PRESS INFORMATION

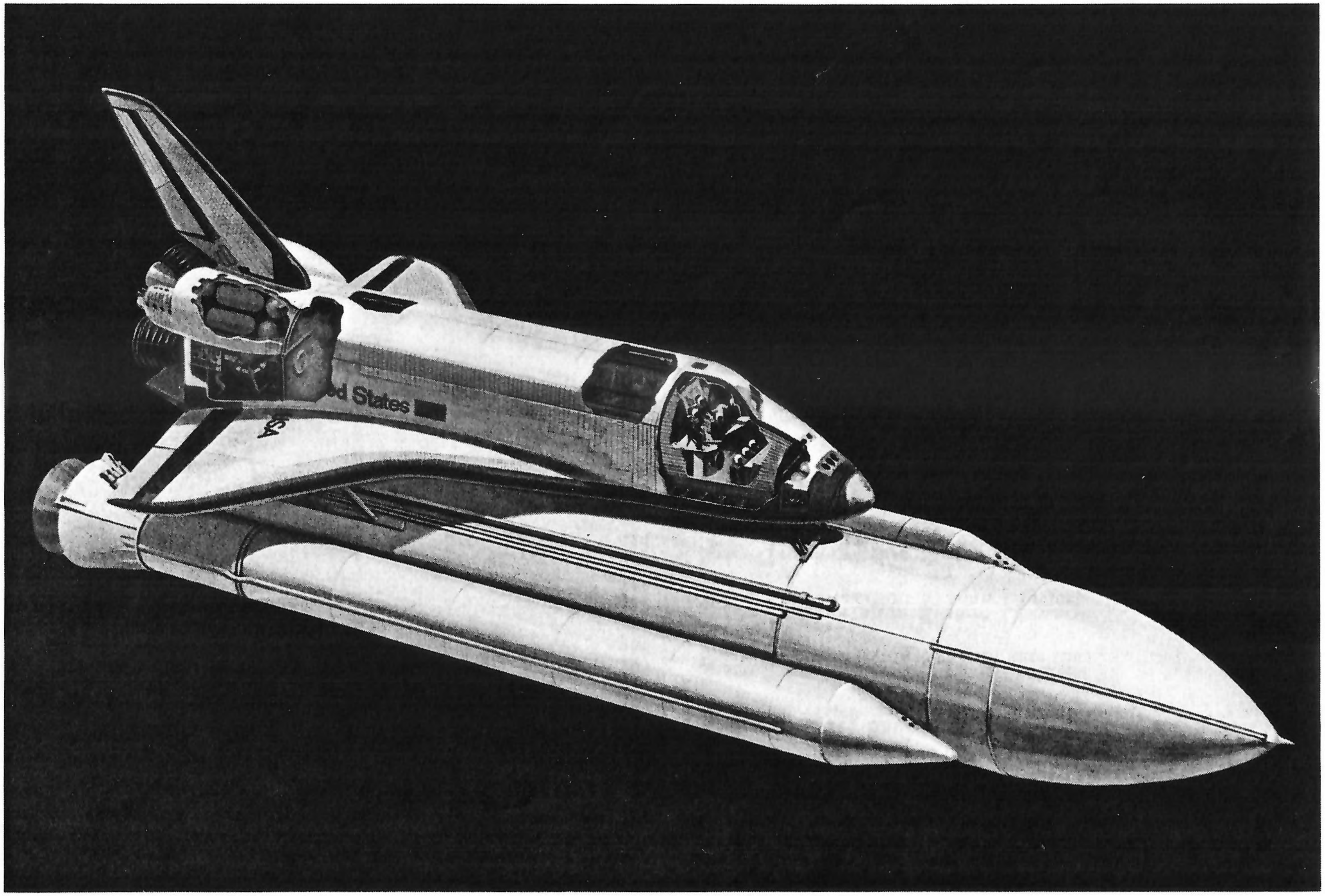
SPACE SHUTTLE TRANSPORTATION SYSTEM

February 1981

Rockwell International

Contents

°C (°F) = Degrees Celsius (Degrees Farenheit)



Shuttle System/Orbiter Cutaway

Space Transportation System

The Space Shuttle is America's newest and most versatile manned spacecraft. Unlike its predecessors–Mercury, Gemini, and Apollo–it is a reusable, aircraft-like ship which is designed for years of service and makes space flight a relatively economical and routine event.

The Shuttle will provide a flexibility never before achieved in space operations; it will allow space to be treated as the resource it is, rather than as a hostile environment to be tested, examined, and explored.

Shuttle will enable man to use hard vacuum and zero gravity for further development of material processing in space. Space flight has opened up this new environment which we have only begun to understand or take advantage of.

Our earth-bound environment limits certain material processes. In almost all human activities and industrial processes, gravity is a dominant force; it constrains our motion, produces friction which must be overcome, and separates fluids. It causes layering in alloys, contamination in glasses, separation in chemicals, and defects in electronic materials.

Experiments planned for Shuttle flights will explore the benefits of zero-gravity and hard vacuum. Actually a tiny gravitational attraction is present: 1/3000 to 1/6000 of what we experience on earth. Neither is the vacuum perfect; there is one-trillionth of an atmosphere. The findings to date from Apollo, Skylab, and Apollo-Soyuz Test Project missions have brought back materials fashioned in space that were more pure and uniform than anything made on earth–almost physically perfect. If we could develop materials to their full physical thermal, chemical, optical, and electric theoretical potential, major breakthroughs in medicine, electronics, energy, instrumentation, and construction could be achieved. The development of the Space Shuttle and Spacelab opens the door to routine work on commercially attractive material processes in space.

The spacecraft's large cargo capacity and relatively mild launch environment will enable it to carry into orbit a variety of satellites, including some which could not be launched before because of size, shape, weight, or sensitivity to launch forces.

Malfunctioning satellites can be repaired in space by Shuttle crewmen or the satellite can be returned in the cargo bay for repair on earth. The Shuttle also could serve as a launch platform for higher-orbit satellites or for interplanetary craft.

Eventually, the Shuttle can transport elements of large space structures to orbit and support the construction crews in space. Such large structures could include satellite power systems, permanent space stations, multi-satellite service platforms, and orbiting space power stations.

The potential of space is unlimited.

A special NASA study team recently reviewed the U.S. space program and identified specific contributions to the welfare of mankind it could make in the next 25 years. These contributions–compiled in the team's report, *Outlook for Space*–range from weather and communications through manufacturing-processing to earth-oriented activities. In the last category, a grouping of contributions aimed at responding to "basic human needs," the use of satellites and space laboratories was seen helping meet the following needs:

- **Food production, forestry management**—Forecasting of global crop production and water availability, land use assessment, timber inventory, assessment of marine resources and rangeland.
- **Environmental protection**—Large-scale weather forecasting, climate prediction, weather modification experiments, stratospheric changes and effects, water quality monitoring, global marine weather forecasting.
- **Protection of life and property**—Storm tracking, tropospheric pollutant monitoring, hazard forecasting, communication-navigation, earthquake prediction, control of harmful insects.
- **Energy and mineral exploration**—Solar power relay to earth from satellite power stations, disposal of hazardous waste in space, world geologic atlas.
- **Information transfer**—Domestic, intercontinental, and personal communications.
- **Use of space environment**—Basic physics and chemistry, material science, commercial inorganic processing, biological material research, effects of gravity on terrestrial life, physiology and disease processes.
- **Earth science**—Earth's magnetic field, crustal dynamics, ocean interior and dynamics, dynamics and energetics of lower atmosphere, ionosphere-magnetosphere coupling, and structure, chemistry, and dynamics of stratosphere/mesosphere.

SPACE SHUTTLE PROGRAM

The Space Shuttle is being developed by the National Aeronautics and Space Administration. NASA will coordinate and manage the Space Transportation System (NASA's name for the overall Shuttle program) through the 1980's, including inter-governmental agency requirements and international and joint projects. NASA also oversees the launch and space flight requirements for civilian and commercial use.

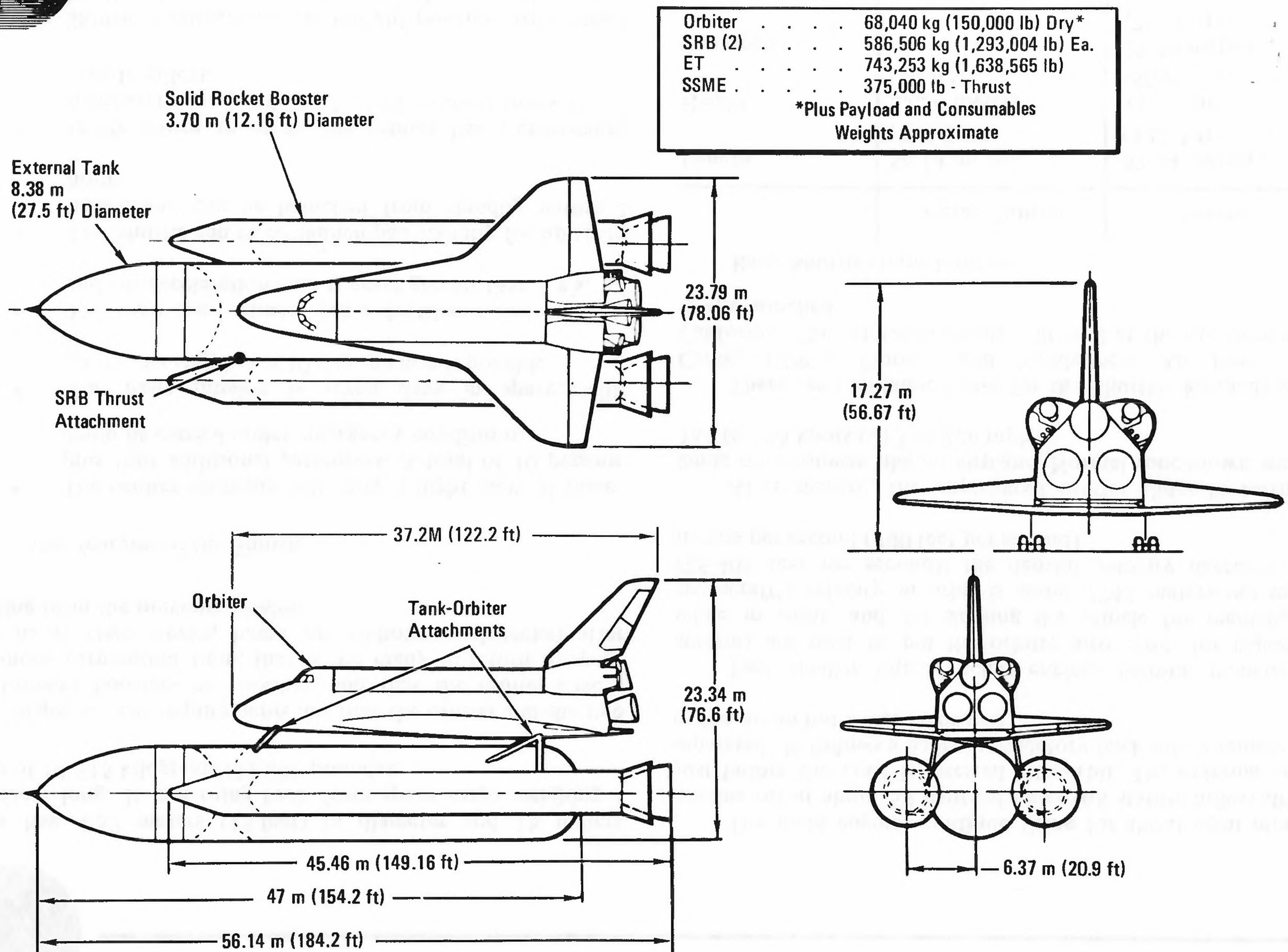
The Space Shuttle system consists of four primary elements: an orbiter spacecraft, two solid-rocket boosters, an external tank to house fuel and oxidizer, and the three main engines.

The orbiter is built by Rockwell International's Downey, Calif., facility. This facility also has contractual responsibility for integration of the overall Space Transportation System. Both orbiter and integration contracts are under the direction of NASA's Johnson Space Center (JSC) in Houston, Tex.

The solid-rocket booster motors are built by the Wasatch Division of Thiokol Corp., the external tank by Martin Marietta Corp., and the Shuttle main engines by Rockwell's Rocketdyne Division. These contracts are under the direction of NASA's George C. Marshall Space Flight Center (MSFC) in Huntsville, Ala.

SHUTTLE REQUIREMENTS

The Shuttle will transport into near earth orbit 100 to 600 nautical miles (115 to 690 statute miles) up to 29,484 kilograms (65,000 pounds) of cargo. This cargo (called payload) is carried



Space Shuttle Statistics

in a bay 4.57 meters (15 feet) in diameter and 18 meters (60 feet) long. It can bring back from space cargo weighing a total of 14,515 kilograms (32,000 pounds).

Major system requirements are that the orbiter and the two solid-rocket boosters be reusable, and that the orbiter have a 160-hour turnaround time; that is, be ready to return to space 160 hours (two weeks, based on 80-hour workweeks) after landing from the previous mission.

Other features of the Shuttle:

- The orbiter normally will carry a flight crew of three, plus four additional passengers. A total of 10 persons could be carried under emergency conditions.
- The basic mission is seven days in space; with additional supplies, a 30-day mission is possible.
- The crew compartment has a shirtsleeve environment, and the acceleration load is never greater than 3 g's.
- The Shuttle can be on launch pad standby for up to 24 hours, and can be launched from standby within 2 hours.
- In its return to earth, the orbiter has a cross-range maneuvering capability of 1100 nautical miles (1265 statute miles).

The Shuttle is launched in an upright position, with thrust provided by the three main engines and the two solid-rocket boosters. After about two minutes, at an altitude of about 24 nautical miles (28 miles), the two boosters are spent and are separated from the orbiter. They fall into the ocean at predetermined points and are recovered for reuse.

The main engines continue firing for about eight minutes, cutting off at about 59 nautical miles (68 statute miles) altitude just before the craft is inserted into orbit. The external tank is separated. It follows a ballistic trajectory back into a remote area of the ocean but is not recovered.

Two smaller liquid rocket engines (orbital maneuvering system) are used to put the orbiter into orbit, for maneuvers while in orbit, and for slowing the vehicle for reentry. The spacecraft's velocity in orbit is about 7743 meters per second (25,405 feet per second); the deorbit velocity decrease is 91 meters per second (300 feet per second).

After reentry, the unpowered orbiter glides to earth and lands on a runway like an airplane. Normal touchdown speed is 184 to 196 knots (213 to 226 mph).

There are two launch sites for the Shuttle: Kennedy Space Center (KSC), Florida, and Vandenberg Air Force Base, California. The orbiter normally will land at the site from which it was launched.

Basic Shuttle characteristics:

	Overall Shuttle	Orbiter
Length	56.14 meters (184.2 ft)	37.24 meters (122.2 ft)
Height	23.34 meters (76.6 ft)	17.27 meters (56.67 ft)
Wingspan	–	23.79 meters (78.06 ft)
Approx. weight		
Gross liftoff	2,041,200 kilograms (4.5 million pounds)	–
Landing with payload	–	96,163 kilograms (212,000 pounds)

Space Transportation System

	Overall Shuttle	Orbiter
Thrust (sea level)		
Solid-rocket boosters	12,889,200 Newtons (2.9 million pounds) of thrust each at sea level	–
Orbiter main engines	–	1,668,000 Newtons (375 thousand pounds) of thrust each at sea level
Cargo bay:		
Length	–	18.28 meters (60 ft)
Diameter	–	4.57 meters (15 ft)

BACKGROUND AND STATUS

On July 26, 1972, NASA selected Rockwell's Space Transportation System Development and Production Division, in Downey, Calif., as the industrial contractor for design, development, test, and evaluation (DDT&E) of the orbiter. The contract called for fabrication and testing of two orbiter spacecraft, a full-scale structural test article, and a main propulsion test article. The award followed years of NASA and Air Force studies on definition and feasibility of a reusable space transportation system.

NASA previously (March 31, 1972) had selected Rockwell's Rocketdyne Division to design and develop the Space Shuttle main engines. Contracts followed to Martin Marietta for the external tank (August 16, 1973) and Thiokol's Wasatch Division for the solid-rocket boosters (June 27, 1974).

In addition to the orbiter DDT&E contract, Rockwell's Space Transportation System Development and Production Division was given contractual responsibility as industrial integrator for the overall Shuttle system.

The first orbiter spacecraft, *Enterprise* (OV–orbiter vehicle–101), was rolled out on Sept. 17, 1976. On Jan. 31, 1977, it was transported overland from Rockwell's assembly facility at Palmdale, Calif., to the Dryden Flight Research Center at Edwards Air Force Base (36 miles) for the approach and landing test (ALT) program.

The ALT program was conducted from February through November, 1977, and demonstrated that the orbiter could fly in the atmosphere and land as an airplane. The program consisted of:

- Five "captive" flights in which the orbiter, unmanned, was mounted atop a specifically modified 747 Shuttle carrier aircraft (SCA).
- Three manned captive flights in which two-man astronaut crews operated the orbiter control systems.
- Five "free" flights, in which the orbiter was released from the SCA and maneuvered to a landing at Edwards. In the first four such flights the landing was on a dry lake bed; in the fifth, the landing was on Edwards' main concrete runway under conditions simulating a return from space. The last two free flights were made without the tail cone, which is the spacecraft's configuration during an actual landing from earth orbit.

On March 13, 1978, the *Enterprise* was ferried atop the SCA to NASA's Marshall Space Flight Center in Huntsville, Ala., to undergo a series of mated vertical ground vibration tests. These were completed in March, 1979. On April 10, 1979, the

Enterprise was ferried to the Kennedy Space Center, mated with the external tank and solid rocket boosters, and transported via the mobile launch platform to launch complex 39A. At launch complex 39A, the *Enterprise* served as a practice and launch complex fit check verification tool representing the flight vehicles. It was ferried back to NASA's Dryden Research Center at Edwards AFB, CA, on August 16, 1979, and then returned overland to Rockwell's Palmdale final assembly facility on October 30, 1979. Certain components were refurbished for use on flight vehicles being assembled at Palmdale. It will eventually be used as a practice and fit check verification tool at Vandenberg AFB. The *Enterprise* was built as a test vehicle and is not equipped for space flight.

The second orbiter (OV-102), *Columbia*, will be the first to fly into space. It was transported overland on March 8, 1979, from Palmdale to Edwards for mating atop the SCA and ferrying to Kennedy Space Center, Fla. It arrived at KSC on March 25, 1979, to begin preparations for the first flight into space.

Columbia will land at Edwards AFB in the initial development flight tests, with the last development flight landing on the concrete runway at Edwards AFB.

The structural test article, after 11 months of extensive testing, is being modified at Rockwell's final assembly facility at Palmdale to become the second orbiter available for operational missions. It is redesignated OV-099, the *Challenger*.

The main propulsion test article (MPTA-098) consists of an orbiter aft fuselage, a truss arrangement which simulates the orbiter mid fuselage, and the Shuttle main propulsion system (three main engines and the external tank). This test structure is at the National Space Technology Laboratory in Mississippi. A series of static firings were conducted in 1978 through 1981 in support of the first flight into space.

NASA named the first four orbiter spacecraft after famous sailing ships. In the order they will become operational, they are:

- **Columbia (OV-102)**, after a sailing frigate launched in 1836, one of the first Navy ships to circumnavigate the globe. Columbia also was the name of the Apollo 11 command module which carried Neil Armstrong, Michael Collins, and Edward (Buzz) Aldrin on the first lunar landing mission, July 20, 1969.

- **Challenger (OV-099)**, also a Navy ship, which from 1872 to 1876 made a prolonged exploration of the Atlantic and Pacific Oceans. It also was used in the Apollo program, for the Apollo 17 lunar module.

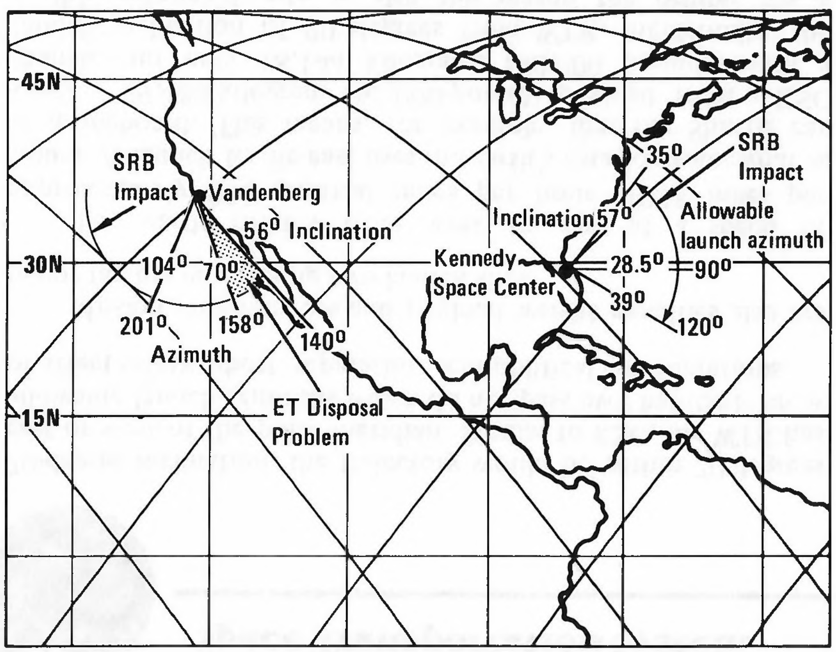
- **Discovery (OV-103)**, after two ships, Henry Hudson's which in 1610-11 attempted to search for a northwest passage between Atlantic and Pacific oceans and instead discovered Hudson Bay, and Captain Cook's which discovered the Hawaiian Islands and explored southern Alaska and western Canada.

- **Atlantis (OV-104)**, after a two-masted ketch operated for the Woods Hole Oceanographic Institute from 1930 to 1966, which traveled more than half a million miles in ocean research.

On Jan. 29, 1979, NASA contracted with Rockwell for the manufacture of two additional orbiters, OV-103 and OV-104 (*Discovery* and *Atlantis*), conversion of the STA to space flight configuration (*Challenger*), and modification of *Columbia* from its developmental configuration to that required for operational flights.

LAUNCH SITES

During normal operations, Shuttle launchings will be from both KSC and the Western Test Range (Vandenberg AFB). Shuttle missions destined for equatorial orbital trajectories will be launched from KSC and those requiring polar orbital planes will be launched from WTR.



Space Shuttle Launch Sites

Orbital mechanics and the complexities of mission requirements, plus safety and the possibility of infringement on foreign air and land space, prohibit polar orbit launches from KSC.

KSC launches have an allowable path no less than 35 degrees northeast and no greater than 120 degrees southeast. These are azimuth degree readings based on due east from KSC as 90 degrees.

A 35-degree azimuth launch places the spacecraft in an orbital inclination of 57 degrees. This means the spacecraft in its orbital trajectories around the earth will never exceed an earth latitude higher or lower than 57 degrees north or south of the equator.

A launch path from KSC at an azimuth of 120 degrees will place spacecraft in an orbital inclination of 39 degrees (it will be above or below 39 degrees north or south of the equator).

These two azimuths 35 and 120 degrees–represent the launch limits from KSC. Any azimuth angles further north or south would launch a spacecraft over a habitable land mass, adversely affect safety provisions for abort or vehicle separation conditions, or would be undesirable because of the possibility that the solid rocket boosters or external tank could land on foreign land or sea space.

Launches from the Western Test Range have an allowable launch path suitable for polar insertions south, southwest, and southeast.

The launch limits at WTR are 201 and 158 degrees. At a 201-degree launch azimuth, the spacecraft would be orbiting at a 104-degree inclination. Zero degree would be due north of the launch site and the orbital trajectory would be within 14 degrees east or west of the north-south pole meridian. At a launch azimuth of 158 degrees, the spacecraft would be orbiting at a

70-degree inclination, the trajectory would be within 20 degrees east or west of the polar meridian. Similar to KSC, the WTR has allowable launch azimuths which do not pass over habitable areas or affect safety, abort, separation, and political considerations.

Mission requirements and payload weight penalties also are major factors in selecting two launch sites.

The earth rotates from west to east at a speed of approximately 900 nautical miles per hour (1035 miles per hour). A launch to the east uses the earth's rotation somewhat as a springboard. This means, for example, that the Shuttle can carry a 29,484-kilogram (65,000-pound) payload from a KSC launch, but only 18,144 kilograms (40,000 pounds) with a launch inclination of 90 degrees from WTR. Incidentally, the earth's rotational rate is also the reason the orbiter has a cross-range capability of 1100 nautical miles (1265 statute miles) to provide the abort once around (AOA) capability in polar orbit launches.

Attempting to launch and place a spacecraft in polar orbit from KSC to avoid habitable land mass would be uneconomical because the Shuttle's payload would be reduced severely—down to 7711 kilograms (17,000 pounds). A northerly launch into polar orbit 8 to 20 degrees azimuth—would necessitate a path over a land mass, and most safety, abort, and political constraints would have to be waived. This prohibits polar orbit launches from KSC.

The following orbital insertion inclinations and payload weights exemplify the Shuttle's capabilities:

1. Equatorial orbit from KSC (for low earth orbit, geosynchronous orbit, or interplanetary escape)—at an orbital inclination of 28.5 degrees from KSC, maximum payload weight is 29,484 kilograms (65,000 pounds); at an inclination of 57 degrees, maximum payload weight is 25,855 kilograms (57,000 pounds).

2. Polar orbit from WTR—at an orbital inclination of 90 degrees, maximum payload weight is 18,144 kilograms (40,000 pounds) or 14,515 kilograms (32,000 pounds) at an inclination of 104 degrees.

MISSION PROFILE

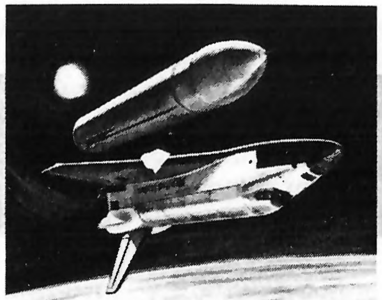
In the launch configuration, the orbiter and two solid-rocket boosters are attached to the external tank and all are in a vertical position (nose up) on the launch pad. Each solid-rocket booster is attached at its aft skirt to the mobile launch platform by four bolts.

Emergency exit for the flight crew while on the launch pad up to 30 seconds prior to liftoff is by slide wire. There are five 365-meter (1200-foot) slide wires, each with one basket. Each basket is designed to carry normally two persons but could handle three. The baskets, 1.5 meters (5 feet) in diameter and 106 centimeters (42 inches) deep, are suspended beneath the slide mechanism by four cables. The slide wires carry the baskets to a bunker at ground level. The bunker is designed to protect personnel even from an explosion on the launch pad.

At launch, the three Shuttle main engines—fed liquid hydrogen fuel and liquid oxygen oxidizer from the external tank—are ignited first. When it has been verified that the engines are operating at the proper thrust level, a signal is sent to ignite the solid-rocket boosters. At the proper thrust-to-weight ratio, initiators (small explosives) at the eight holddown bolts are fired to release the Shuttle for liftoff. All this takes only a few seconds.

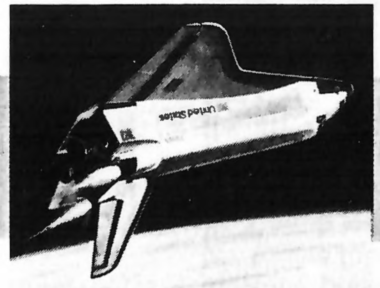
Maximum dynamic pressure (max q) is reached early in the ascent, nominally at 10,241 meters (33,600 feet) about 60 seconds after liftoff.

Approximately a minute later (two minutes into the ascent phase), the two solid-rocket boosters have consumed their



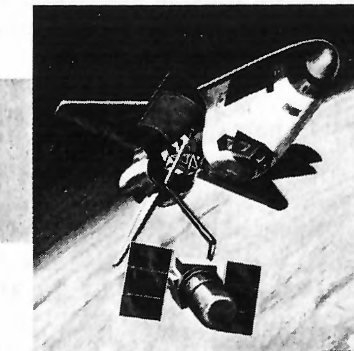
MAIN ENGINE CUTOFF, EXTERNAL TANK SEPARATION

Altitude: 59 nmi (68 miles); velocity: 7796 mps (25,581 fps, 17,440 mph) about 8 minutes after launch (just before orbit insertion)



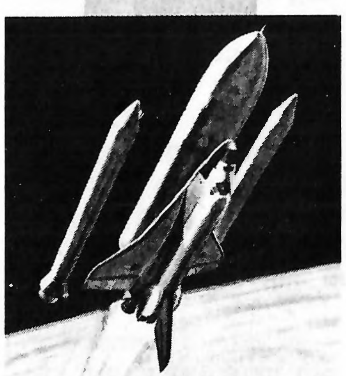
ORBIT INSERTION AND CIRCULARIZATION

Altitude varies according to mission



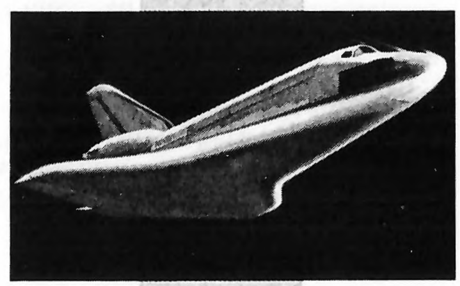
ORBITAL OPERATIONS

Mission from 7 to 30 days; 100 to 600 nmi (115 to 690 miles) orbits; 7743 mps (25,405 fps, 17,321 mph)



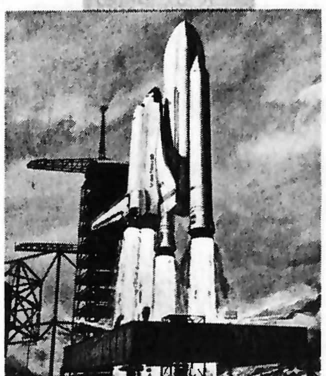
SRB SEPARATION

Altitude: 24 nmi (28 miles); velocity: 1383 mps (4,538 fps, 3,094 mph) 2 minutes after launch



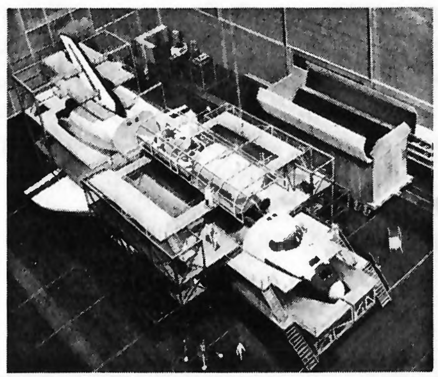
DEORBIT

Velocity decreased nominal 91 mps (300 fps, 204 mph) from earth orbit operations

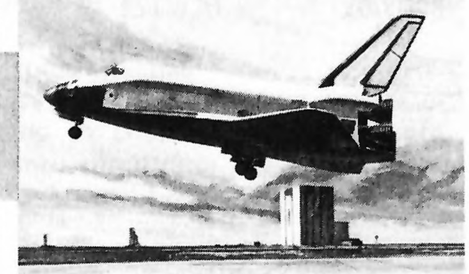


LAUNCH

Maximum dynamic pressure at 10,241 meters (33,600 ft); about 60 seconds after launch



MAINTENANCE Two-week turnaround (14 days—160 hours)



LANDING Touchdown speed 184 to 196 knots (213 to 226 mph)

Shuttle Mission Profile

propellant and are jettisoned from the external tank. This is triggered by a separation signal from the orbiter.

The boosters briefly continue to ascend, while small motors fire to carry them away from the Shuttle. The boosters then turn and descend, and at a predetermined altitude, parachutes are deployed to decelerate them for a safe splashdown in the ocean. Splashdown will occur approximately 141 nautical miles (162 statute miles) from the launch site. The boosters are recovered and reused.

Meanwhile, the orbiter and external tank continue to ascend, using the thrust of the three main engines. Approximately eight minutes after launch and just short of orbital velocity, the three engines are shut down (main engine cutoff—MECO) and the external tank is jettisoned on command from the orbiter.

The external tank continues on a ballistic trajectory and enters the atmosphere, where it disintegrates. Its projected impact is in the Indian Ocean, in the case of equatorial orbits (KSC launch), and in the extreme southern Pacific Ocean, in the case of WTR launch.

Two orbital maneuvering systems (OMS) at the aft end of the orbiter are used in a two-step firing, to complete insertion into earth orbit, and to circularize the spacecraft's orbit. Forward and aft reaction control system (RCS) thrusters provide attitude control (pitch, yaw, and roll) of the orbiter, as well as any minor translation maneuvers along a given axis. The orbiter is designed to operate in earth orbit between 100 to 600 nautical miles (115 to 690 statute miles).

At completion of the orbital operations (from 1 to 30 days), the orbiter is oriented to a tail-first attitude. The two OMS engines are then used to slow the vehicle for deorbit.

The RCS thrusters then turn the orbiter nose forward for entry. These thrusters continue to control the orbiter until atmospheric density is sufficient for the pitch and roll aerodynamic control surfaces to become effective.

Entry is considered to occur at 121,920 meters (400,000 feet) altitude approximately 4400 nautical miles (5063 statute miles) from the landing site and at approximately 7620 meters per second (25,000 feet per second) velocity.

At 121,920 meters the spacecraft is maneuvered to zero degrees roll and yaw (wings level) and a predetermined angle of attack for entry. (In the initial development flights, the angle of attack is 40 degrees; in later flights, it will be between 28 degrees and 38 degrees.) The flight control system issues the commands to roll, pitch, and yaw RCS jets for rate damping.

The forward RCS jets are inhibited at 121,920 meters, and the aft RCS jets maneuver the spacecraft until a dynamic pressure of 517 mmHg per square meter (10 pounds per square foot) is sensed, which is when the orbiter's ailerons become effective. The aft RCS roll jets are then deactivated. At a dynamic pressure of 1035 mmHg per square meter (20 pounds per square foot), the orbiter's elevators become active and the aft RCS pitch jets are deactivated. The orbiter's speed brake is used below Mach 10 to induce a more positive downward elevator trim deflection. At Mach 3.5, the rudder becomes activated and the aft RCS yaw jets are deactivated at 13,716 meters (45,000 feet).

Entry guidance must dissipate the tremendous amount of energy the orbiter posseses when it enters the earth's atmosphere to assure that the orbiter does not either burn up (entry angle too steep) or skip out of the atmosphere (entry angle too shallow) and that the orbiter is properly positioned to reach the desired touchdown point.

During entry, energy is dissipated by the atmospheric drag on the orbiter's surface. Higher atmospheric drag levels enable faster energy dissipation with a steeper trajectory. Normally, the angle of attack and roll angle enable the atmospheric drag of any flight vehicle to be controlled. However, for the orbiter, angle of attack was rejected because it creates surface temperatures above the design specification. The angle of attack schedule used during entry is loaded into the orbiter computer as a function of relative velocity, leaving roll angle for energy control. Increasing the roll angle decreases the vertical component of lift, causing a higher sink rate and energy dissipation rate. Increasing the roll rate does raise the surface temperature of the orbiter, but not nearly as drastically as an equal angle of attack command.

If the orbiter is low on energy (current range-to-go much greater than nominal at current velocity) entry guidance will command lower than nominal drag levels. If the orbiter has too much energy (current range-to-go much less than nominal at the current velocity), entry guidance will command higher than nominal drag levels to dissipate the extra energy.

Roll angle is used to control cross-range. Azimuth error is the angle between the plane containing the orbiter's position vector and the heading alignment cylinder tangency point and the plane containing the orbiter's position vector and velocity vector. When the azimuth error exceeds a computer-loaded number, the orbiter's roll angle is reversed.

Thus, descent rate and down ranging are controlled by bank angle. The steeper the bank angle, the greater the descent rate and the greater the drag; conversely, the minimum drag attitude is wings level. Cross-range is controlled by bank reversals.

The entry thermal control phase is designed to keep the backface temperatures within the design limits. A constant heating rate is established until below 5791 meters per second (19,000 feet per second).

The equilibrium glide phase shifts the orbiter from the rapidly increasing drag levels of the temperature control phase to the constant drag level of the constant drag phase. The equilibrium glide flight is defined as flight in which the flight path angle, the angle between the local horizontal and the local velocity vector, remains constant. Equilibrium glide flight provides the maximum downrange capability. It lasts until the drag acceleration reaches 33 feet per second squared.

The constant drag phase begins at that point. In the development flight the angle of attack is initially 40 degrees but it begins to ramp down in this phase to approximately 36 degrees by the end of this phase.

The transition phase is where the angle of attack continues to ramp down, reaching the approximately 14-degree angle of attack at the entry terminal area energy management (TAEM) interface, approximately 25,298 meters (83,000 feet) altitude, 762 meters per second (2500 feet per second), Mach 2.5, and 52 nautical miles (59 statute miles) from the landing runway. Control is then transferred to TAEM guidance.

During the entry phases described, the orbiter's roll commands keep the orbiter on the drag profile and control crossrange.

TAEM guidance steers the orbiter to the nearest of two heading alignment cylinders (HAC's) whose radii are 5480 meters (18,000 feet), which are located tangent to and on either side of the runway centerline on the approach end. In TAEM guidance, excess energy is dissipated with an S turn and the speed brake can be utilized to modify drag, L/D (lift/drag) ratio, and flight path angle in high energy conditions. This increases the ground track range as the orbiter turns away from the nearest HAC until sufficient energy is dissipated to allow a normal approach and landing guidance phase capture, which begins at 3048 meters (10,000 feet) altitude. The orbiter also can be flown near the velocity for maximum lift over drag or wings level for the range stretch case. The spacecraft slows to subsonic velocity at approximately 14,935 meters (49,000 feet) altitude, about 22 nautical miles (25.3 statute miles) from the landing site.

At TAEM acquisition, the orbiter is turned until it is aimed at a point tangent to the nearest HAC and continues until it reaches the point, WP-1 (way point one). At WP-1, the TAEM heading alignment phase begins. The HAC is followed until landing runway alignment plus or minus 20 degrees has been achieved. In the TAEM prefinal phase, the orbiter leaves the HAC, pitches down to acquire the steep glide slope, increases airspeed, and banks to acquire the runway centerline and continues until on the runway centerline, on the outer glide slope, and on airspeed. The approach and landing guidance phase begins with the completion of the TAEM prefinal phase and ends when the spacecraft comes to a complete stop on the runway.

The approach and landing trajectory capture phase begins at the TAEM interface and continues to guidance lock on to the steep outer glide slope. The approach and landing phase begins at about 3048 meters (10,000 feet) altitude at an equivalent air speed (EAS) of 290 plus or minus 12 knots, 6.9 nautical miles (7.9 statute miles) from touchdown. Autoland guidance is initiated at this point to guide the orbiter to the minus 20° glide slope (which is over seven times that of a commercial airliner's approach) aimed at a target 0.86 nautical mile (one statute mile) in front of the runway. The spacecraft speedbrake is positioned to hold the proper velocity. The descent rate in the later portion of TAEM and approach and landing is greater than 3048 meters (10,000 feet) per minute (approximately 20 times higher rate of descent than a commercial airliner's standard three-degree instrument approach angle).

At 533 meters (1750 feet) above ground level, a pre-flare maneuver is started to position the spacecraft for a 1.5-degree glideslope in preparation for landing with the speed brake positioned as required. The flight crew deploys the landing gear at this point.

The final phase reduces the sink rate of the spacecraft to less than 2.7 meters per second (9 feet per second). Touchdown occurs approximately 762 meters (2500 feet) past the runway threshold at a speed of 184 to 196 knots (213 to 226 mph).

ABORTS

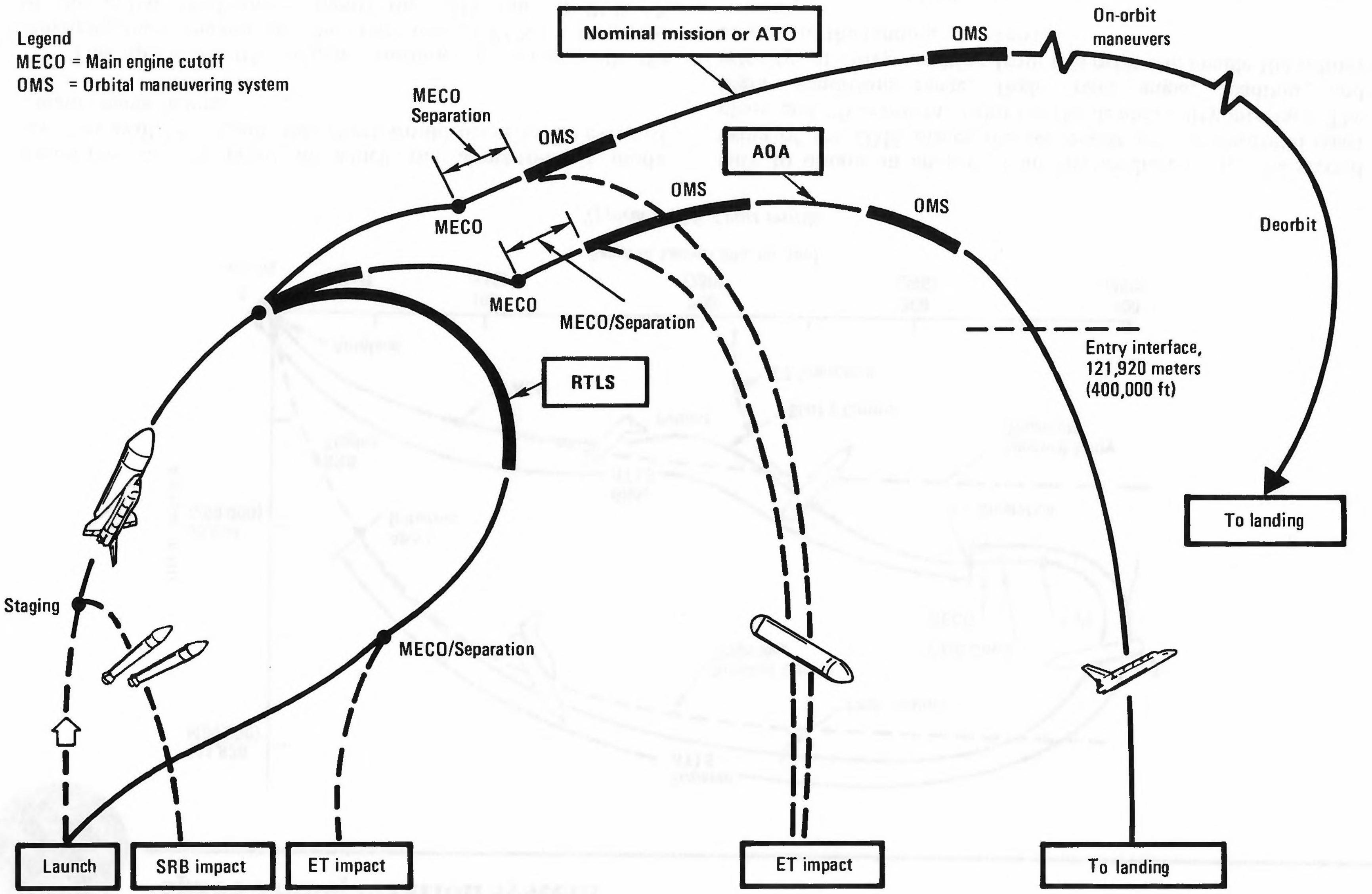
The Shuttle has three abort alternatives, depending on when it becomes necessary. These are to return to launch site (RTLS), abort once around (AOA), and to abort to orbit (ATO).

RETURN TO LAUNCH SITE. This mode will be used in the event of a main engine failure between liftoff and the point at which the next abort mode (AOA) is available. RTLS will not begin until the solid-rocket boosters complete their normal thrusting period and are jettisoned, as in a normal ascent.

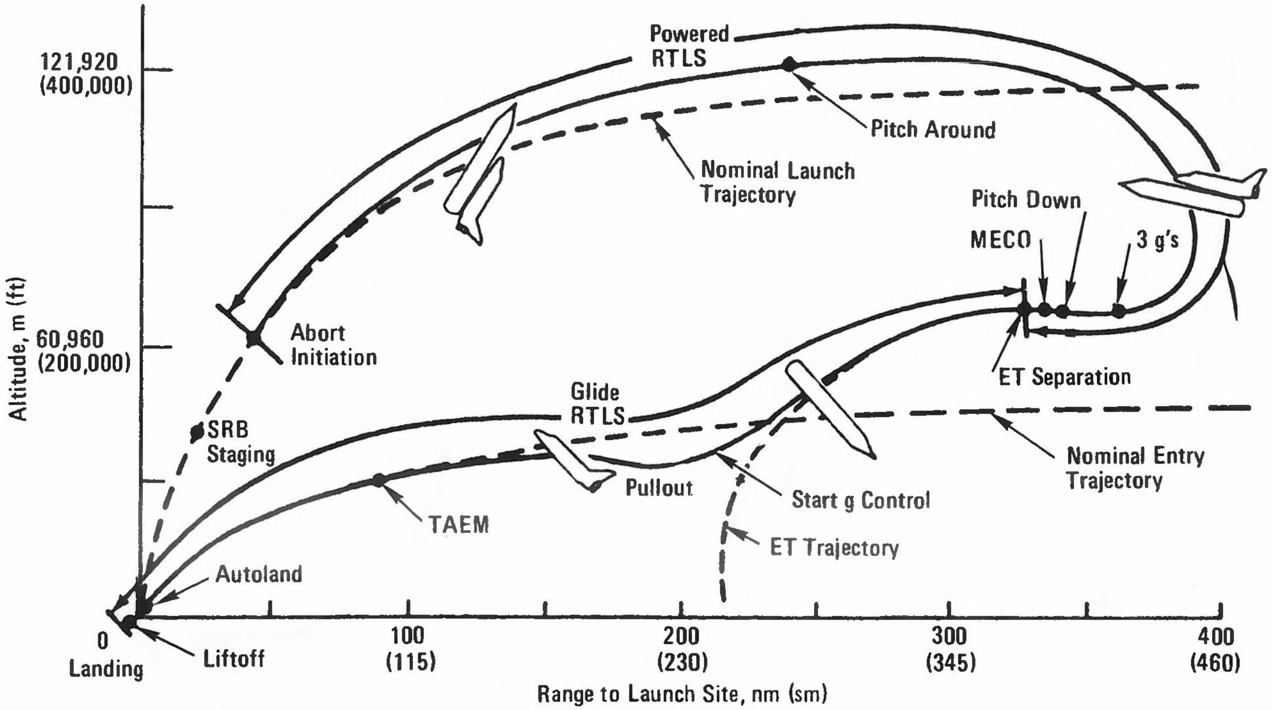
The Space Shuttle (orbiter and external tank) continues to thrust down range, with the two remaining main engines, the two OMS, and the four aft +X RCS thrusters firing, until the remaining propellant for the main engines equals the amount required to reverse the direction of flight.

A pitch-around (plus pitch) maneuver is then performed at approximately 5 degrees per second; this places the orbiter and external tank in a "heads-up" attitude, pointing back toward the launch site. Main engine cutoff is commanded when altitude, attitude, flight path angle, heading, weight, and velocity/range conditions combine for acceptable orbiter-external tank separation [tank impact no closer than 24 nautical miles (28 statute miles) from the U.S. coast] and orbiter glides to the launch site runway.

ABORT ONCE AROUND. This mode will be used from approximately two minutes after normal solid-rocket booster



Abort and Normal Mission Profile



Typical RTLS Abort Profile

separation to the point at which the abort-to-orbit mode becomes available. Again, this abort would occur in the event of a main engine failure.

The Space Shuttle vehicle continues to thrust with the remaining main engines and the OMS and aft RCS +X thrusters. In the initial development flights the OMS and aft RCS +X thrusters will not be used before MECO. The OMS and RCS thrusting periods terminate when the amount of propellant remaining in these two systems will support two OMS thrusting periods after MECO.

Main engine cutoff is followed by jettisoning of the external tank. The OMS thrusters are fired after jettisoning the external tank to obtain an apogee of an intermediate orbit. The second firing of the OMS places the spacecraft into a suborbital coast phase and "free return" orbit for the desired entry interface. The flight conditions–range, flight path angle, headings, and velocity–at entry resulting from this orbit will enable the orbiter to glide to the landing site runway.

ABORT TO ORBIT. This mode begins after the AOA point is passed and also would occur in the event of a main engine failure. The Space Shuttle continues to thrust with the remaining main engines to main engine cutoff and external tank jettisoning. The OMS thrusters fire twice, to insert the orbiter into orbit and then to circularize the orbit. The orbit coast time altitude and the coast time before the deorbit maneuver depend on when the abort was initiated and the mission. Alternate missions may be planned in case of an ATO orbit. The deorbit, entry, and landing would be similar to a normal mission.

ORBITER GROUND TURNAROUND

NASA's Kennedy Space Center, Fla., is responsible for orbiter recovery ground operations at all primary and contingency landing sites. In addition to the prime sites at Edwards Air Force Base, Calif., and the Kennedy Space Center, Fla., KSC recovery operations are responsible for landing activities at contingency sites at Northrup Strip, White Sands, New Mexico; ROTA Naval Air Station, Spain; Kadena Air Force Base, Okinawa, and Hickam Air Force Base, Hawaii.

The first four flights of the *Columbia* will land at Edwards Air Force Base, Calif. The fifth and subsequent flights are scheduled to land at the Kennedy Space Center.

The spacecraft recovery operations at Edwards Air Force Base will be supported by approximately 160 Kennedy Space Center team members. Ground team members wearing "SCAPE" suits that protect them from toxic chemicals will approach the spacecraft as soon as it stops rolling. The ground team members will take sensor measurements to insure the atmosphere in the vicinity of the spacecraft is not explosive. In the event of propellant leaks, a wind machine truck carrying a large fan will be moved into the area to create a turbulent air flow that will break up gas concentrations and reduce the potential for an explosion.

An air conditioning purge unit is attached to the orbiter so cool air can be directed through the orbiter's aft fuselage, payload bay, forward fuselage, wings, vertical stabilizer, and orbital maneuvering system/reaction control system pods to dissipate the heat of entry. This heat, if not dissipated, will "soak" to the orbiter systems within 15 minutes of landing.

A second ground cooling unit is connected to the spacecraft Freon coolant loops to provide cooling for the flight crew and avionics during post landing and system checks. The spacecraft fuel cells remain powered up at this time. The flight crew will then exit the spacecraft and a ground crew will power down the spacecraft.

Within one to two hours the spacecraft and ground support equipment convoy will be ready to move the spacecraft to the service area at NASA's Dryden Flight Research Center at Edwards. After detailed inspection and preparations at DFRC, the *Columbia* is ferried atop the Shuttle Carrier Aircraft to the Kennedy Space Center.

When the spacecraft lands and completes its runout at the Kennedy Space Center, the same procedures as at Edwards Air Force Base are accomplished, with the exception being that it is expected that only one hour will be required before the spacecraft and convoy is ready to move to the Orbiter Processing Facility (OPF).

In later missions, the orbiter must be refurbished and readied for another launch within 160 hours (14 working days). This short turnaround decreases the maintenance cost (part of the cost per flight), decreases the number of orbiters and support elements needed, and increases the utilization rate of each orbiter.

The spacecraft is towed to the Orbiter Processing Facility where it is safed (fuel and oxidizer systems drained, tanks purged, and ordnance removed). The OMS and RCS pods are removed and reinstalled, if required, and other vehicle maintenance performed. The payload is then installed and spacecraft functioning verified. Activity in the OPF will take about 96 hours.

The spacecraft is then towed to the Vehicle Assembly Building (VAB), and mated to the external tank. These elements were stacked and mated on the mobile launch platform while the orbiter was being refurbished. Shuttle connections and the integrated vehicle are checked and ordnance is installed. Activity in the VAB is scheduled for about 39 hours.

The mobile launch platform moves the entire Space Shuttle system on four crawlers to the launch pad, where connections are made and servicing, checkout, and pre-launch activities are conducted. This takes about 24 hours.

The Space Shuttle is then ready for launch within two hours.

In the event of a landing at an alternate site, a crew of about eight team members will move to the landing site to assist the astronaut crew in preparing the orbiter for loading aboard the Shuttle Carrier Aircraft for transport back to the Kennedy Space Center. If the landing is outside the U.S., personnel at the contingency landing sites will be provided minimum training on safe handling of the orbiter with emphasis on crash rescue training, how to tow the orbiter to a safe area, and prevention of propellant contamination.

Space Shuttle flights from the Western Test Range, Calif., will utilize the Vandenberg Launch Facility (SL6) which was built for but never used for the manned orbital laboratory program. This facility will be modified for Space Transportation System use.

The runway at Vandenberg will be strengthened and lengthened from 2,438 meters (8,000 feet) to 3,657 meters (12,000 feet) to accommodate the orbiter returning from space.

Shuttle buildup at Vandenberg will differ from the NASA Kennedy Space Center plan in that the integration-on-pad technique will be employed. Solid rocket boosters will start on-the-pad buildup followed by the external tank. The orbiter will then be mated to the external tank.

The orbiter maintenance and checkout facility at Vandenberg will be used for orbiter processing. It will also provide an area for processing security classified payloads. SL6 includes the launch mount, access tower, mobile service tower, launch control tower, payload preparation room, payload changeout room, solid rocket booster refurbishment facility, solid rocket booster disassembly facility and liquid hydrogen and liquid oxygen storage tanks.

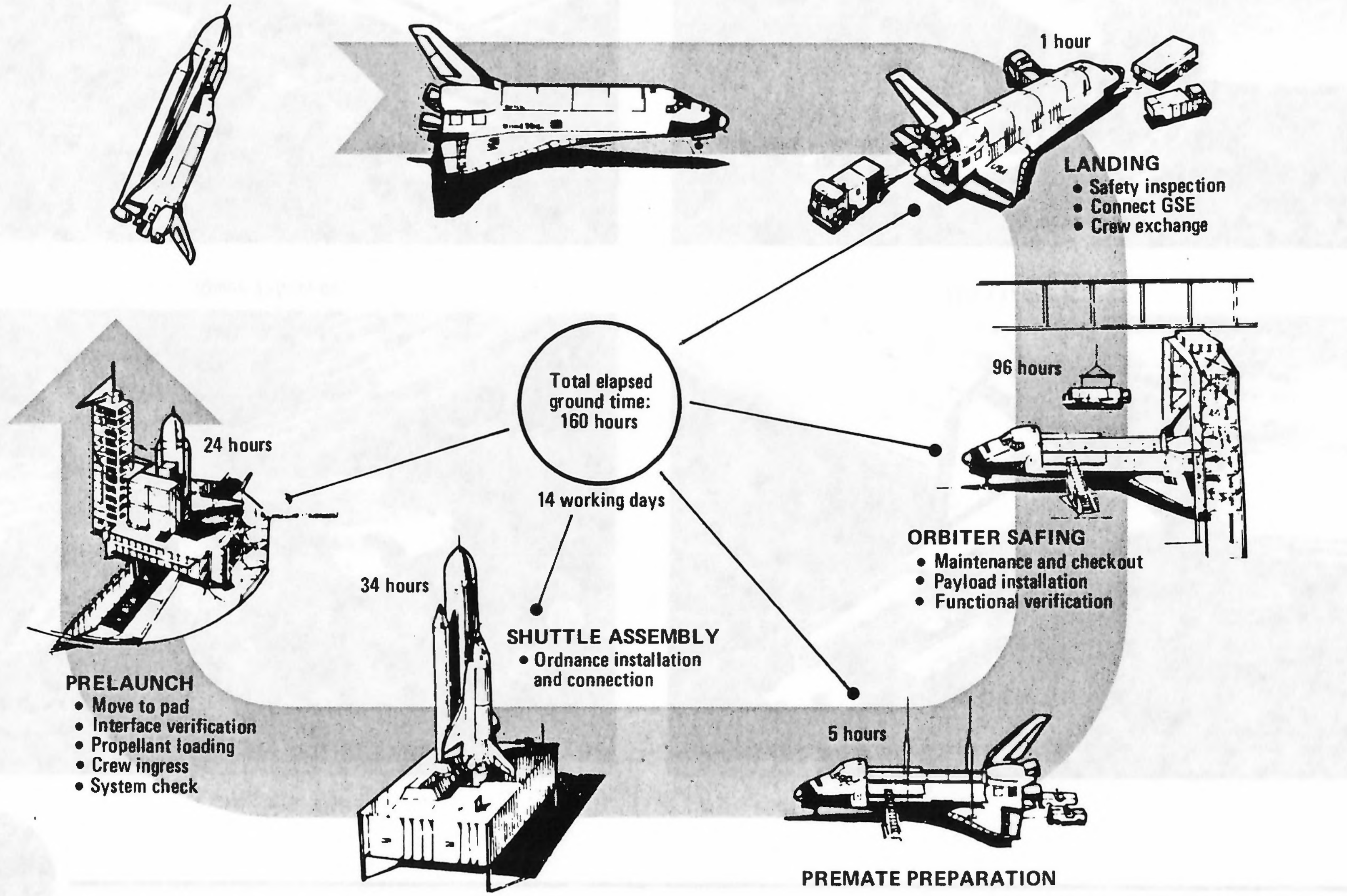
A future liquid propellant thrust augmentation system which will use storable propellants will require some modification to the pad to facilitate loading of propellants. The launch processing system will be similar to the one at the Kennedy Space Center.

PAYLOADS

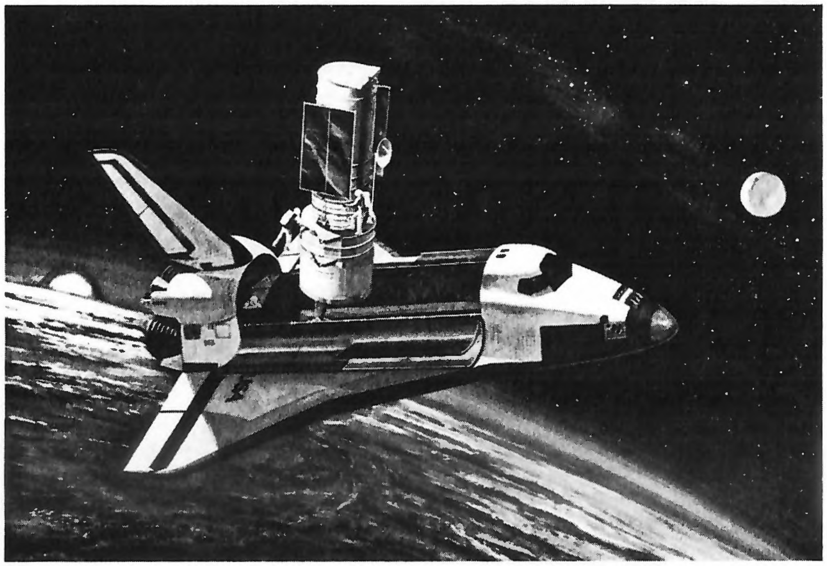
The Space Shuttle is a transportation system. What it carries to earth orbit and back is its reason for existence. The Shuttle orbiter's 18.28-meter-long (60-foot) cargo bay will take hundreds of payloads into space.

All of NASA's centers are vitally concerned with the Shuttle's payload capabilities. In the first 11 years of operation through 1991, more than 400 Shuttle missions will be flown, each carrying one or more payloads. Estimates are that nearly every federal agency—as well as universities, the scientific community, and private industry—will have Shuttle payloads during the rest of this century.

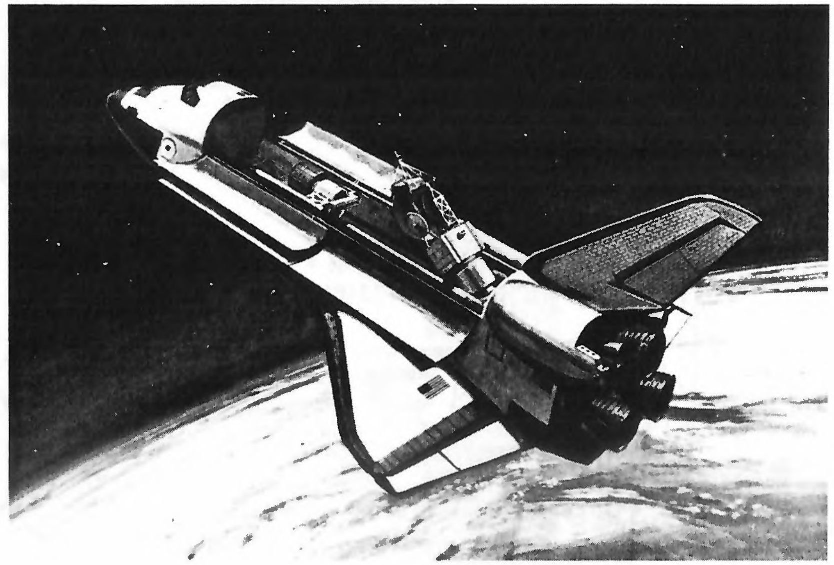
Payloads already scheduled to be carried aboard the orbiter include the Tracking and Data Relay Satellites, communications and weather satellites, Spacelab, Department of Defense non-weapon military payloads, the Space Telescope, multi-mission modular spacecraft, Long Duration Exposure Facility. The Department of Defense will total about 25 percent of the Shuttle missions.



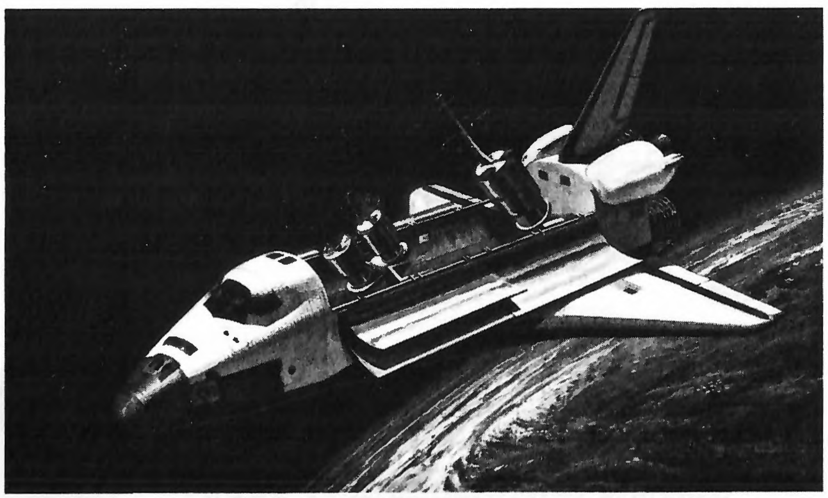
Ground Turnaround Sequence



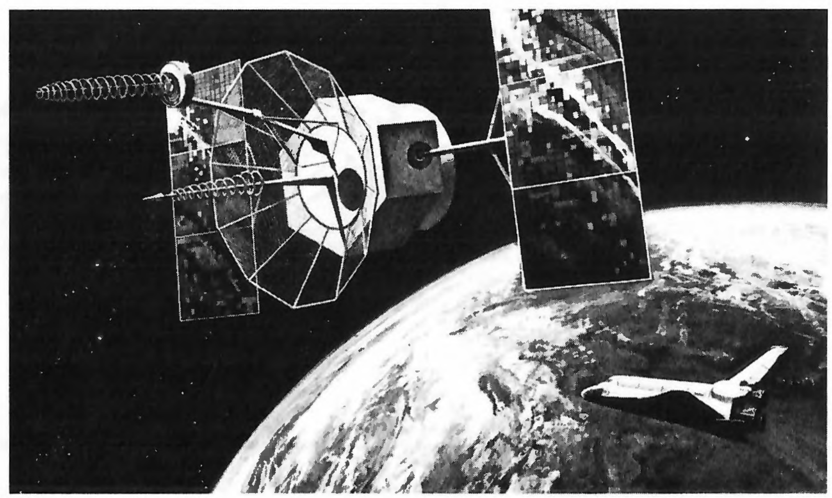
Space Telescope



IUS/COMSAT



WESTAR/MARISAT/INTELSAT IV-A



FLEET SATCOM

Satellites that require high orbital altitudes will be carried aboard the orbiter with a spinning solid upper stage (SSUS) or other type of booster stage. The satellite and upper stage will be deployed together from the payload bay, and the upper stage used to boost the satellite to the required orbit. Weather, communication, and navigation satellites, as well as deep space probes, would require upper stages.

OSTA
(OFFICE OF SPACE TERRESTRIAL APPLICATIONS)–1

OSTA-1 is scheduled to fly on STS-2 and consists of five experiments installed on a U-shaped 3-meter (10 feet) long pallet built by the British Aerospace Corp. under contract to ERNO (Zentral Gesellschaft VFW-Fokker mbh) and the ESA (European Space Agency). Rockwell's Space Operations is responsible for the final assembly of the pallet, installation, integration, and testing of the payload.

NASA's OSTA-1 scientific payload will occupy approximately 0.84 cubic meters (30 cubic feet) of the *Columbia's* payload bay in STS-2 and will weigh 2,449 kilograms (5,400 pounds).

OSTA-1 consists of five experiments, two from NASA's Langley Research Center, two from the Jet Propulsion Laboratory, and one from NASA's Goddard Research Center. The five experiments are a Shuttle Imaging Radar-A (SIR-A); Shuttle Multispectral Infrared Radiometer (SMIRR); a Measurement of Air Pollution from Satellites (MAPS); a Feature Identification and Location Experiment (FILE); and an Ocean Color Experiment (OCE).

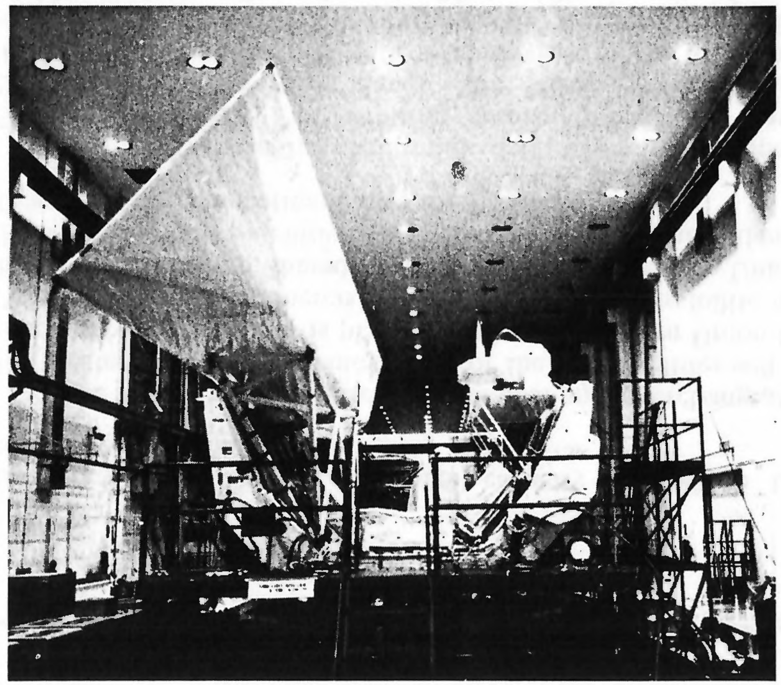
The SIR-A experiment is to evaluate potential use of spaceborne imaging radar for geological exploration and for mineral exploration.

The SMIRR experiment is designed to measure the earth's surface radiance. This will lead toward determining whether instruments can be developed to discriminate between geological formations.

MAPS is an experiment to measure concentration of carbon monoxide in the troposphere over tropical areas.

FILE is an experiment to develop technology which will recognize, acquire and track earth surface features, spectrally sensing and classifying quantatively the surface of the earth into cloud cover, water, bare earth and vegetation.

OCE is an experiment to map distribution of marine algae. This will detect chlorophyll in the algae and will be used to locate marine life and determine effects of pollution and chemical waste.


OSTA-1

The Jet Propulsion Laboratory manages the SIR-A and SMIRR experiments, NASA's Langley Research Center manages the MAPS and FILE experiments, and NASA's Goddard Research Center manages the OCE.

The Pallet provides a platform for mounting the experiments and can also cool equipment, provide electrical power, and furnish connections for commanding and acquiring data from the experiments.

TRACKING AND DATA RELAY SATELLITE SYSTEM. Four Tracking and Data Relay Satellites (TDRS)/Advanced Westar-shared service satellites and ground support equipment are being built by TRW's Defense and Space Systems Group, Redondo Beach, Calif., for Western Union Space Communications Incorporated. Western Union will own and operate the entire TDRS system, leasing tracking and data relay services to NASA in its commercial communications capacity to the Western Union Telegraph Company for its satellite transmission.

The advantage to NASA and commercial users of sharing telecommunications equipment and services is more flexibility in handling different kinds of traffic and greater reliability at lower cost. TDRS is the first such large-scale space undertaking to share government and business telecommunications services. This shared use concept will mean cost savings to the government and will assure Western Union that its Advanced Westar satellite needs will be fulfilled. For Western Union's customers, this will provide continuity of existing services and the availability of the most technologically advanced system.

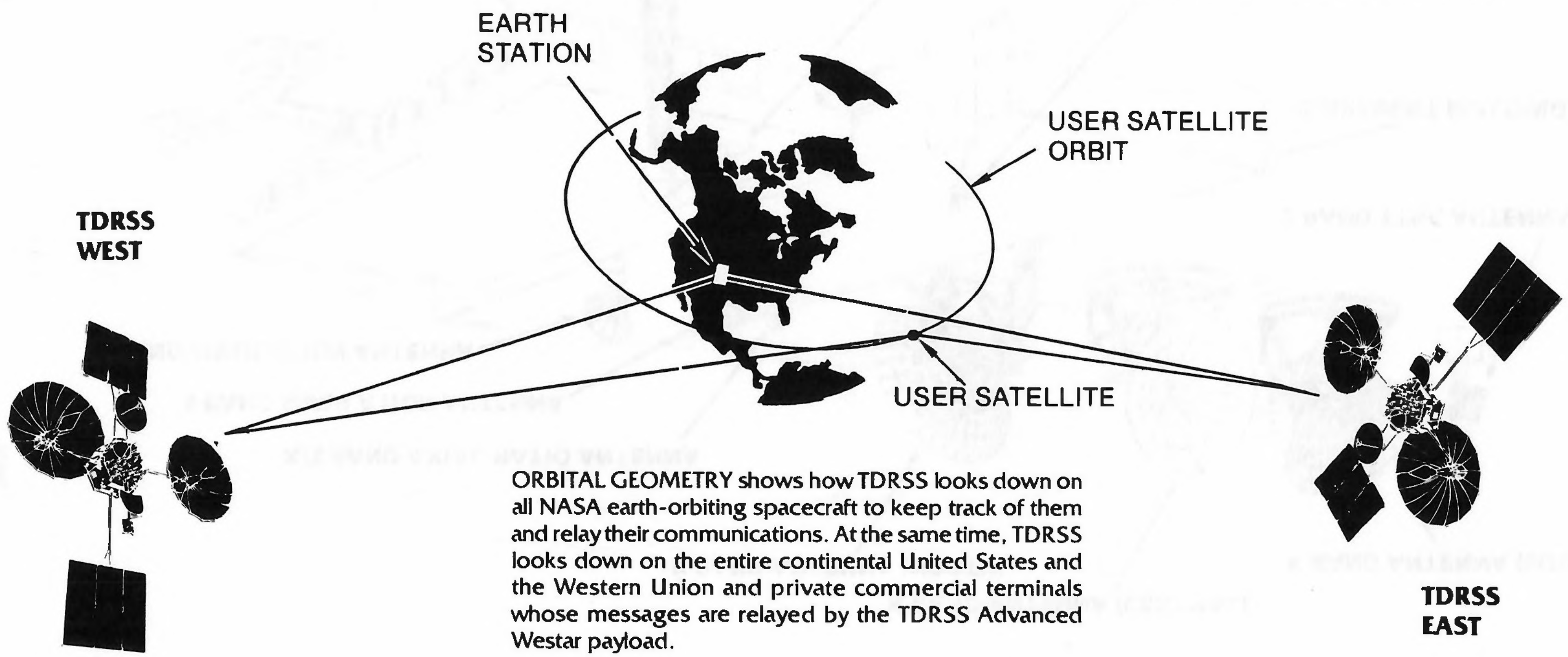
Both the NASA tracking and data relay portion of the telecommunications payload and the Western Union Advanced Westar portion are fully integrated and share common equipment. This payload is separate from but carried by the basic bus which contains satellite operating and housekeeping equipment. In another space first, TDRS incorporates three frequency bands, adding the high capacity K-band to the conventional S- and C-bands.

The four TDRS satellites are identical and interchangeable for greater system redundancy. Two of the four satellites will be used by NASA, another is planned for use by Western Union for Advanced Westar commercial service, and the fourth satellite will be an on-orbit spare, shared by both NASA and Western Union. Two of the four satellites will be stationed over the Atlantic Ocean and two are stationed over the Pacific Ocean.

The TDRS satellites will relay data to and from Space Shuttle, unmanned Earth-orbiting spacecraft and the Earth Station at NASA's White Sands, New Mexico, Test Facility. TDRS offers simultaneous data relay services for up to 32 user spacecraft in orbits up to approximately 2,693 nautical miles (3,100 statute miles) through 85 to 100 percent of each spacecraft's orbit. Today's Earth stations only provide an average of 15 percent orbital tracking for each revolution due to the line of sight acquisition and loss of signal line of sight of that ground station in respect to the spacecraft.

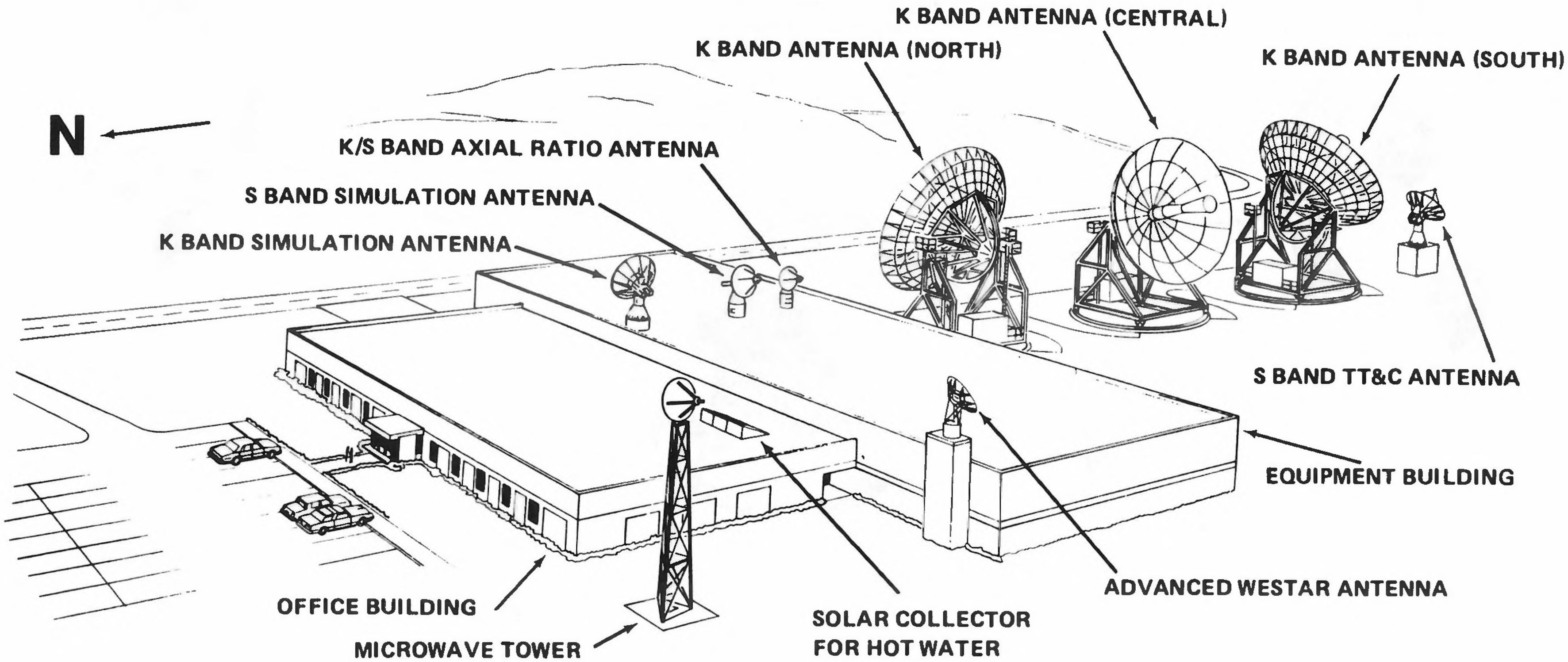
The TDRS is carried into Earth orbit in the payload bay of the Space Shuttle. At an on-orbit altitude of approximately 150 nautical miles (172 statute miles), the Space Shuttle remote manipulator system is utilized to remove the TDRS with its attached upper stage from the payload bay into space. The upper stage is ignited with the attached TDRS to place the TDRS into its 22,300 nautical mile (25,662 statute mile) geosynchronous orbit. Once the TDRS is on-orbit, the upper stage backs off, leaving the TDRS to drift in its assigned geostationary position.

Each TDRS satellite is 17.42 meters (57.2 feet) wide with its solar arrays deployed, 12.98 meters (42.6 feet) wide at the deployed antennas, 4.5 meters (15 feet) deep and weighs 2,932 kilograms (4,700 pounds) on orbit. Each consists of solar panels for electrical power supplemented with batteries during eclipse; a

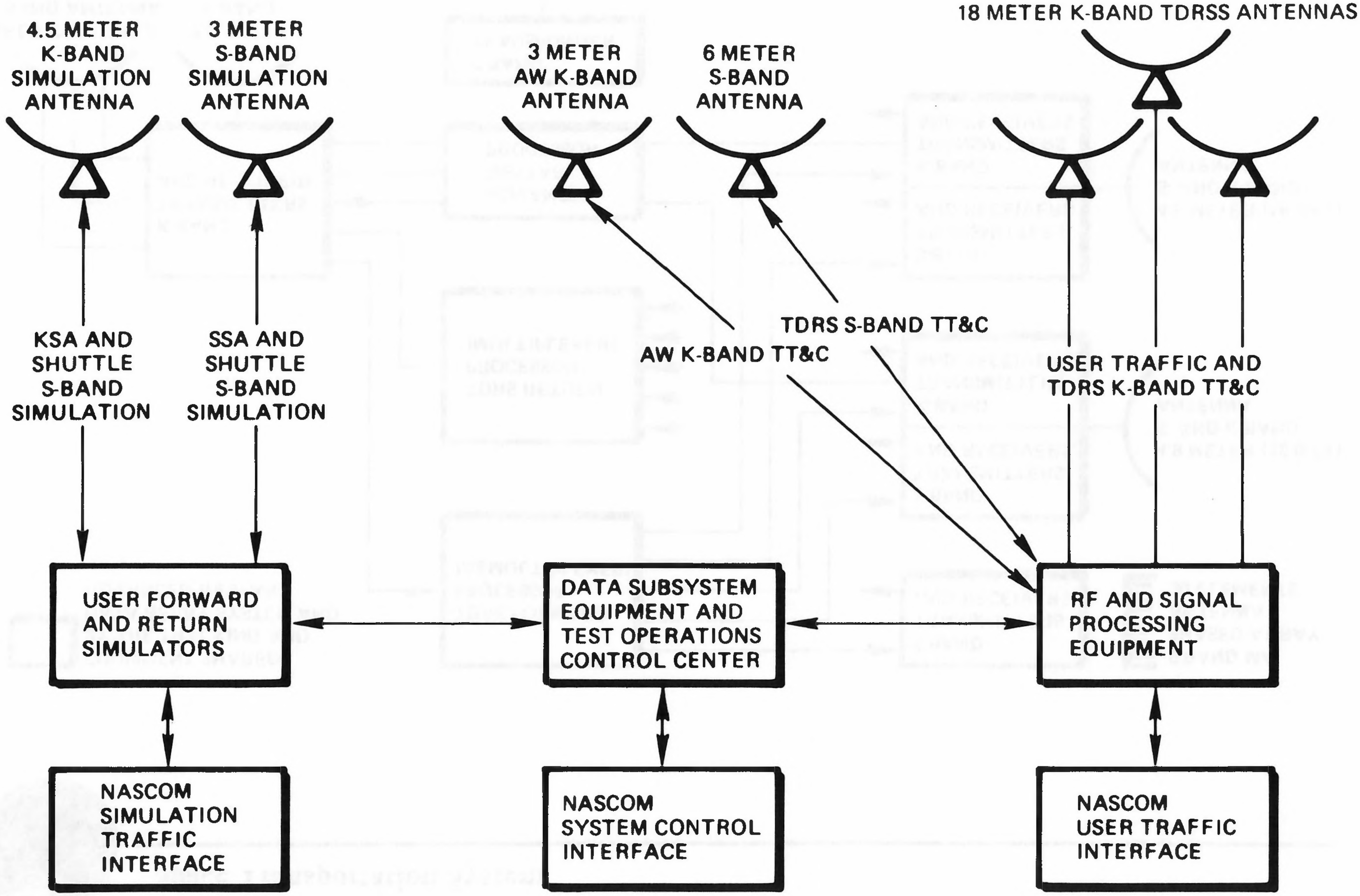


ORBITAL GEOMETRY shows how TDRSS looks down on all NASA earth-orbiting spacecraft to keep track of them and relay their communications. At the same time, TDRSS looks down on the entire continental United States and the Western Union and private commercial terminals whose messages are relayed by the TDRSS Advanced Westar payload.

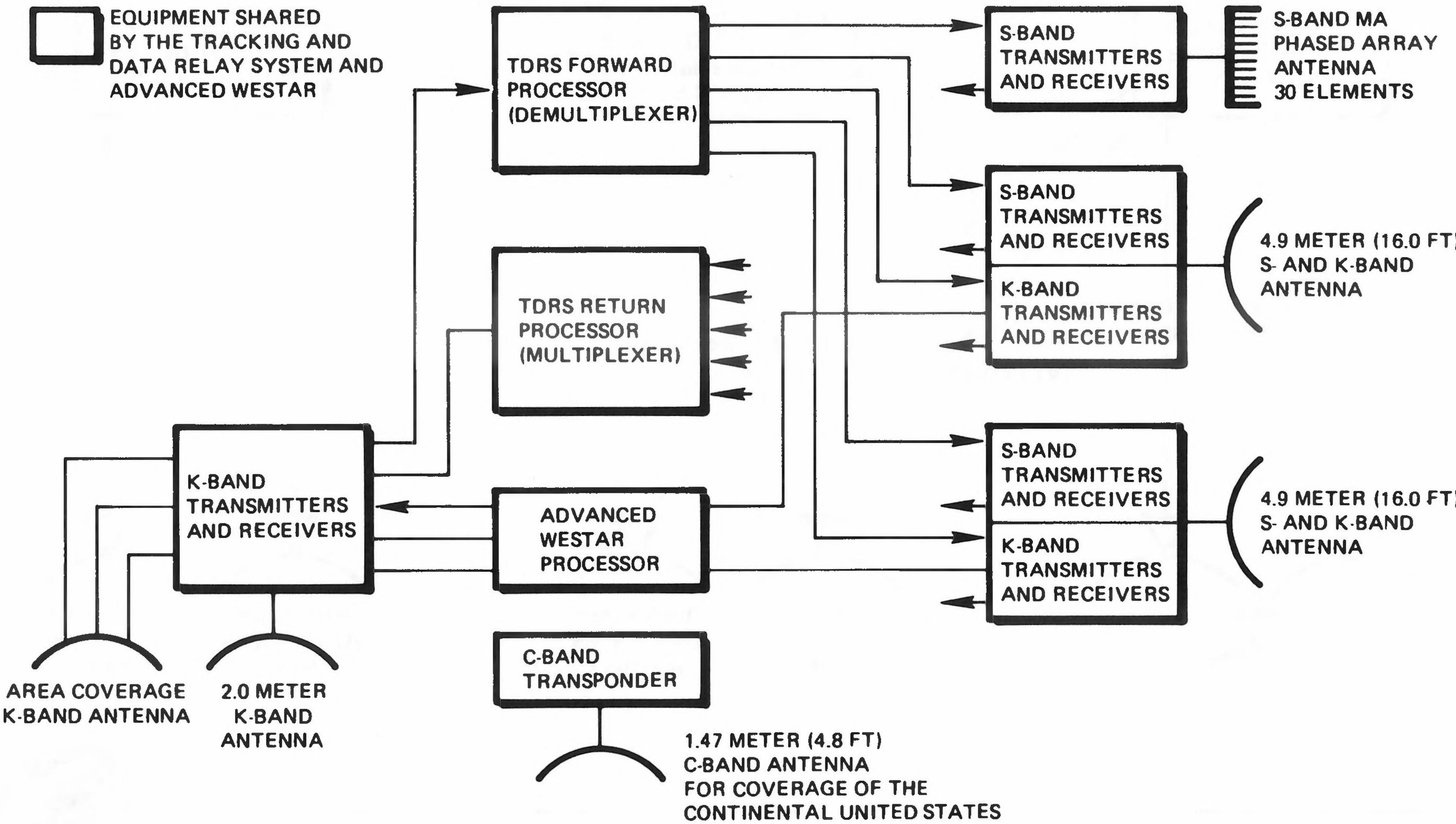
Tracking and Data Relay Satellite System



Tracking and Data Relay Satellite System
Ground Station, White Sands, New Mexico



Tracking and Data Relay Satellite System Antenna



Tracking and Data Relay Satellite System
Transmission and Receive System

three-axis stabilization system maintains attitude control; body fixed momentum wheels and monopropellant hydrazine thrusters interact with earth and sun sensors to stabilize and point the satellite fixed antennas at Earth targets; the three steerable antennas are pointed by command through gimbal drives; and hydrazine thrusters are used for satellite positioning and north-south and east-west station keeping.

The TDRS satellite communications payload operates as a repeater, relaying signals to and from the Earth station and user spacecraft. No signal processing is done on board the TDRS. As many functions as possible have been removed from the TDRS and implemented in the Earth station. This increases system reliability and availability since Earth stations hardware is available for repair and calibration throughout the life of the TDRS, which is 10 years.

The Earth station at NASA's White Sands, New Mexico, Test Facility is one of the largest and most complex ever built. Signals relayed from the TDRS satellites are processed in the Earth station by demodulation equipment which incorporates the latest technology in high-data-rate reception. Automatic data processing equipment aids in making user satellite tracking measurements. This equipment also controls all communications equipment in the Earth station and collects TDRS status data for transmission along with the user spacecraft data.

All data to and from NASA will be handed over to NASA at the station at White Sands. NASA will route the data to the various using NASA centers such as the Johnson Space Center Mission Control Center in Houston, Texas, for Space Shuttle flights. The station also takes command from NASA for transmission to user spacecraft via the TDRS satellites.

Advanced Westar commercial data links will operate independently of this station, except for satellite command and control through the K-band tracking, telemetry, and command link.

Electronic hardware is jointly supplied by Harris Electronics System Division in Melbourne, Fla., and TRW, who also performs integration and test of the total Earth station. TRW is also developing software for the overall system and will integrate this software with the Earth station and the satellites. This software ties together the space segment and the ground segment.

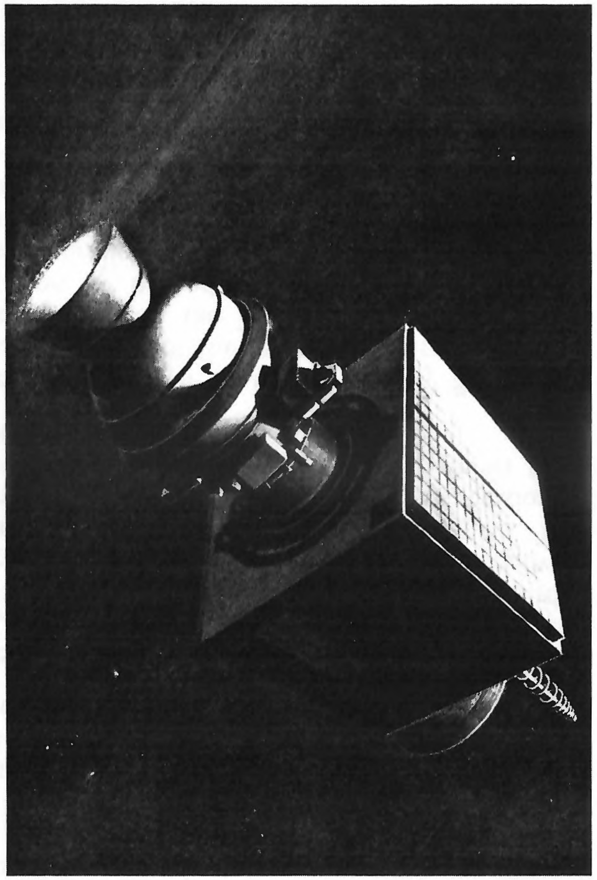
Because TDRS long-term reliability and adaptability are very important, the Earth station performs many functions which are ordinarily found in the space segment of a system. For example, the beam of the multiple-access phased array is formed and controlled by the Earth station, as are the control and tracking functions of the satellite single-access antennas.

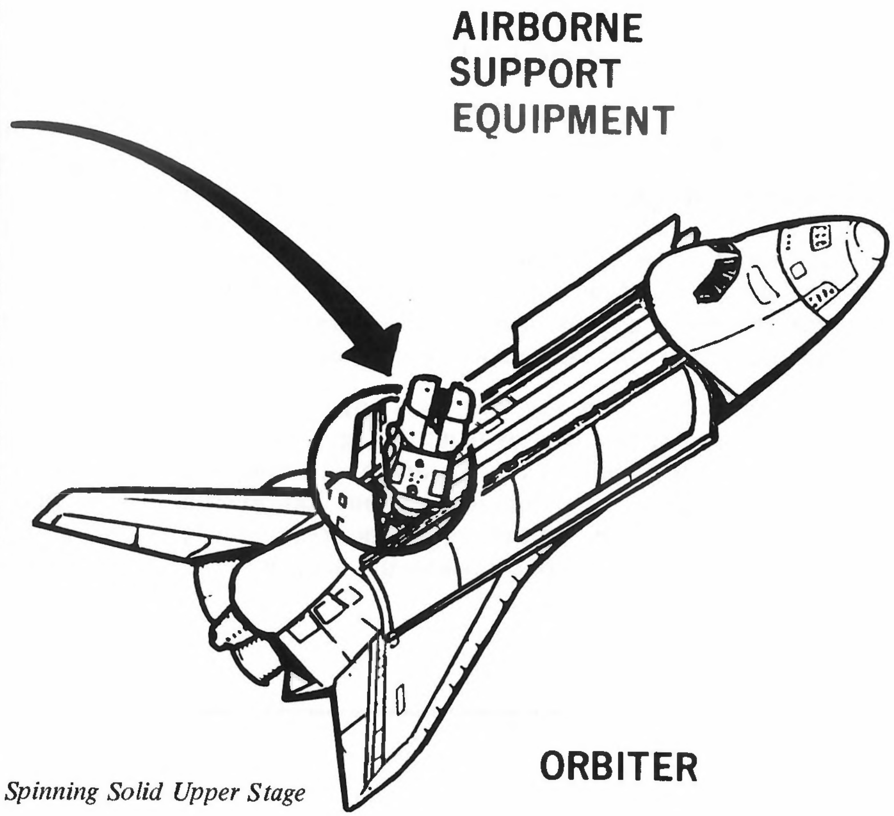
The earth station covers nine acres at NASA's White Sands, New Mexico, Test Facility.

SPINNING SOLID UPPER STAGE. The Spinning Solid Upper Stage (SSUS) is a privately developed rocket vehicle by McDonnell Douglas Astronautics Company in Huntington Beach, Calif., for service in the 1980's. The SSUS is designed to be released from the payload bay of the Space Shuttle orbiter then boost unmanned spacecraft to orbits higher than the orbiter's maximum operating altitude.

Communications satellites aimed toward geosynchronous orbits at 35,887 kilometers (22,300 miles), and others intended for orbits above the orbiter's limits, must be transferred upward from the orbiter with a boost from another rocket. The SSUS provides the boost. Two versions are in development. The SSUS-A is designed for heavy payloads and the SSUS-D is for smaller payloads.

The SSUS-A will be able to lift a satellite weighing as much as 1,996 kilograms (4,400 pounds) when released from the orbiter into a geosynchronous orbit. The SSUS is powered by a solid propellant rocket motor developed by Thiokol Corporation as a commercial derivative of the third stage for the U.S.A.F.





Spinning Solid Upper Stage

Minuteman Missile. SSUS-A is approximately 2.29 meters (7.5 feet) long and 1.5 meters (4.9 feet) in diameter.

SSUS-D will be able to place up to 1,058 kilograms (2,320 pounds) of payload on a geosynchronous transfer track. Options are available to raise this capability to 1,245 kilograms (2,750 pounds). A new Thiokol Corporation solid propellant rocket motor designated STAR 48, is being developed for SSUS-D. The motor is 121.9 centimeters (48 inches) in diameter and approximately 1.82 meters (6 feet) long. Variations in propellant loading and in rocket nozzle length are available to tailor the system for specific mission requirements.

The SSUS and its spacecraft will be mounted on a special cradle developed by McDonnell Douglas for installation in the orbiter payload bay. A payload attach fitting at the top of the rocket motor joins the SSUS to its spacecraft. At the base of the motor are a spin table and a separation system.

The spring-loaded separation system releases the SSUS and its payload from the payload bay of the orbiter, pushing it out of the payload bay before the SSUS rocket motor is fired to send the payload to its final destination. Moments before the SSUS launch from low earth orbit the spin table is activated to provide both SSUS and payload a rotating motion which stabilizes their flight.

McDonnell Douglas is developing SSUS with private funding as a commercial venture. The company has received orders for both models, with three customers to date: NASA, Hughes Aircraft Company's Space and Communications Group, and the Ford Aerospace and Communications Corporation.

SPACELAB. Spacelab is the manned laboratory being built by a group of European nations. This laboratory, which normally will take up the entire cargo bay, will be able to take scientists and researchers into space with their experiments. The Spacelab is not deployed free of the orbiter. The Spacelab is scheduled to make a number of flights aboard the Shuttle.

NASA, with the Marshall Space Flight Center as lead center and the European Space Agency (ESA), formerly known as ESRO (European Space Research Organization), signed a memorandum of understanding on September 24, 1973, in which ESA would design, develop, and test a space laboratory to be flown in the cargo bay of the orbiter. ESA has 11 member nations: Belgium, Denmark, France, Ireland, Italy, Netherlands, Spain, Sweden, Switzerland, United Kingdom, and West Germany. A twelfth, Austria, is an observer rather than a full member. All except Sweden are participating in the Spacelab program.

The NASA-ESA agreement represented a major set in the sharing of space costs. The estimated cost of Spacelab to be borne by ESA member nations is calculated at approximately $400 million. ESA will deliver to NASA one Spacelab engineering model, one flight unit with spares, and two sets of ground support equipment. In addition, ESA will provide engineering post-development support and provide follow-on production.

The Spacelab flight unit will be delivered to NASA one year before the first mission and is designed to have a lifetime of 50 missions or five years. Nominal mission duration of the Spacelab is seven days, but is designed so that missions up to 30 days can be completed by trading payload capability for consumables and a power extension package.

An industrial consortium headed by ERNO-VFW Fokker was named by ESA in June 1974 to build the Spacelab. The six-year contract presently calls for the delivery of the first fully qualified Spacelab flight unit in 1980.

Spacelab is developed on a modular basis and can be varied to meet specific mission requirements. Its two principal components are the pressurized module, which provides a laboratory with a shirtsleeve working environment, and the open pallet that exposes materials and equipment to space. Each module is segmented, permitting additional flexibility.

The pressurized module or laboratory comes in two segments. One, called the core segment, contains supporting systems such as data processing equipment and utilities for both the pressurized modules and the pallets. It also has laboratory fixtures such as floor-mounted racks and work benches.

The second, called the experiment segment, is used to provide more working laboratory space. It contains only floor-mounted racks and benches. When only one segment is needed, the core segment is used.

Each pressurized segment is a cylinder 4.1 meters (13-1/2 feet) in diameter and 2.7 meters (9 feet) long. When both segments are assembled with end cones, their maximum outside length is 7 meters (23 feet).

A tunnel connects the pressurized laboratory with the pressurized cabin of the orbiter. The tunnel also is segmented so that its length can be varied. An airlock module can be attached to the tunnel to provide additional access to space for extravehicular activities.

Spacelab total weight (its payload and payload chargeable-items) will not exceed 14,515 kilograms (32,000 pounds).

Five pallet segments are available. Each U-shaped pallet is 3 meters (10 feet) long, built by the British Aerospace Corp. under contract to ERNO (Zentralgesellschaft VFW-Fokker mbh) and the ESA (European Space Agench). Each pallet is not only a platform for mounting instrumentation but also can cool equipment, provide electrical power, and furnish connections for commanding and acquiring data from the experiments. When pallets only are used, Spacelab portions of the essential systems for supporting experiments (power, experiment control, data handling, and communication, etc.) are protected in a small pressurized and temperature-controlled housing called an igloo. Communications normally is an orbiter function on Spacelab pallet flights. The pallets are designed for large instruments, experiments requiring direct exposure to space, or those needing unobstructed or broad fields of view. Such equipment includes telescopes, antennas, and sensors such as radiometers and radars.

The experimenters aboard Spacelab are called payload specialists. They are nominated for flight by the organization sponsoring the payload. They are accepted, trained, and certified for flight by NASA. Their training includes zero-gravity exercises, simulation of operations and emergencies, briefings of flight plans, and space flight housekeeping.

From one to four payload specialists can be accommodated for each Spacelab flight. These specialists ride into space and return to earth in the orbiter cabin, but they work in Spacelab from seven to 30 days and return to the orbiter's crew cabin when off duty.

More than 2,000 world scientists were represented in the responses to invitations to participate in the first Spacelab missions in the 1980's. Of these, NASA and ESA selected proposals representing 222 investigators from 15 countries. The countries are Austria, Belgium, Canada, Denmark, Federal Republic of Germany, France, Italy, India, Japan, the Netherlands, Norway, Spain, Switzerland, United Kington and United States. The investigations planned by the 222 scientists will be conducted in orbit by the Spacelab payload specialists who will be in contact with their colleagues on the ground.

Some of the experiments that may be flown about Spacelab include:

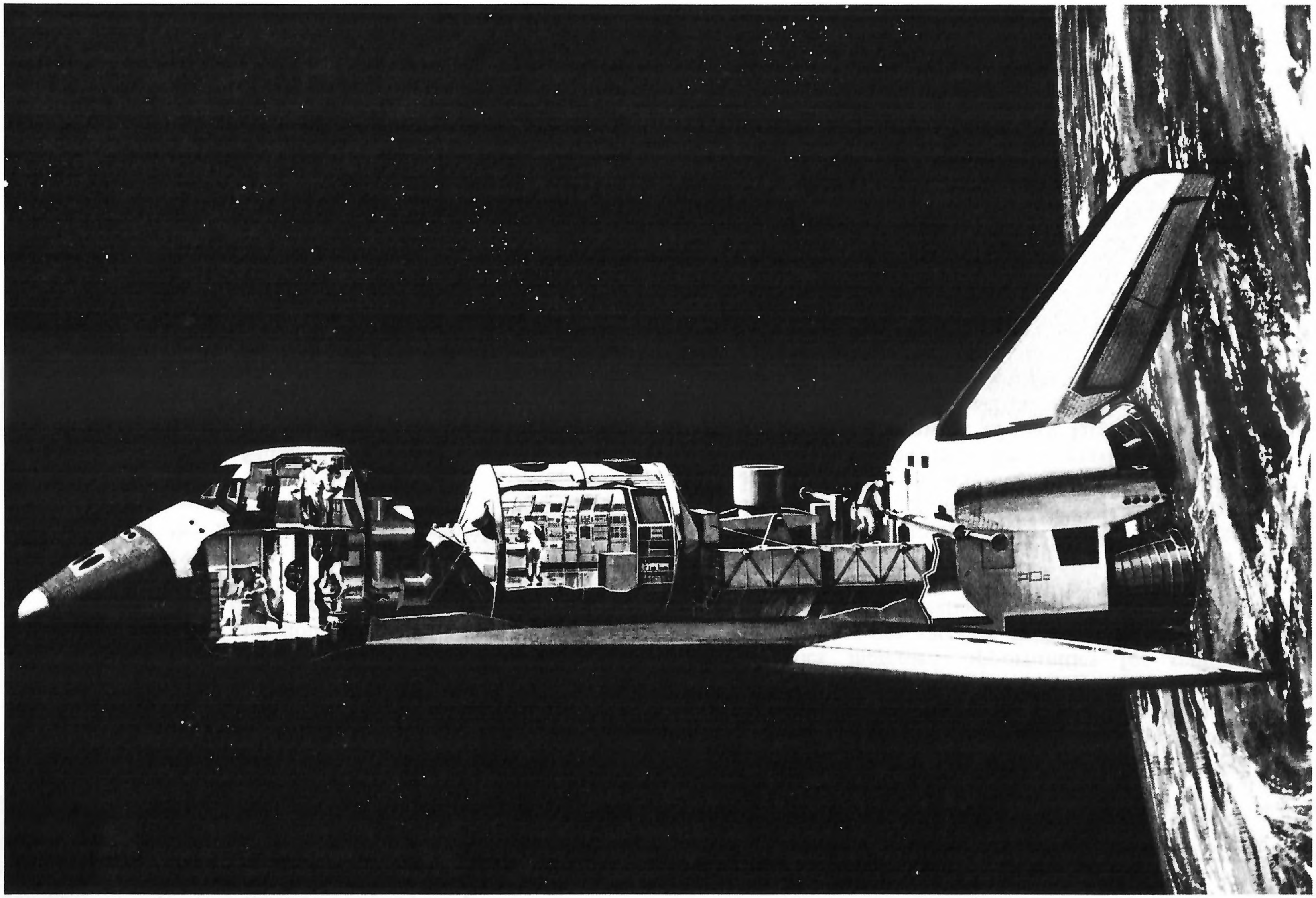
- Earth Surveys—These will gather a variety of information useful in transportation, urban planning, pollution control, farming, fishing, navigation, weather forecasting, and prospecting.

- Astronomy Observations–These are designed to add knowledge about our sun and its interactions with the earth's environment, to view transient events such as comets and novas, and to make observations of high-energy radiation. This radiation–gamma rays, X-rays, ultraviolet light–does not for the most part pass through the atmosphere and cannot be studied on earth. Locked in it are answers to many questions about the nature, origin, and evolution of celestial phenomena.

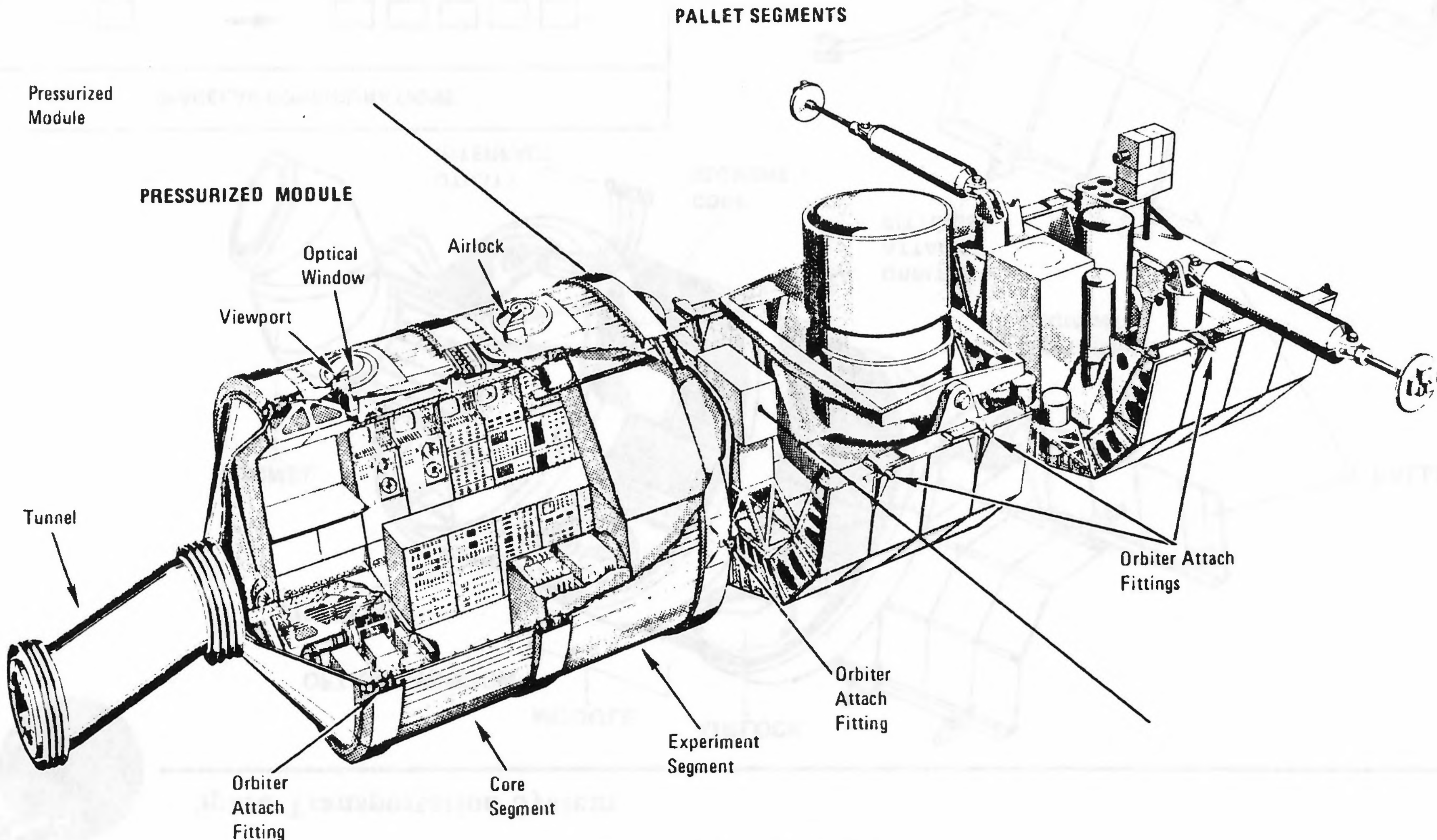
- Life Science–Studies of man and other living things in space have indicated significant metabolic changes resulting from the absence of gravity; continued studies are expected to increase the understanding of these changes and contribute to the advancement of medicine.

- Biomedicine–The gravity-free environment of space has demonstrated significant advantages in separating and purifying biological particles. Space processing provides increased opportunities for removal of impurities from vaccines that cause undesirable side effects and for isolating specific cells or antibodies for treatment of disease.

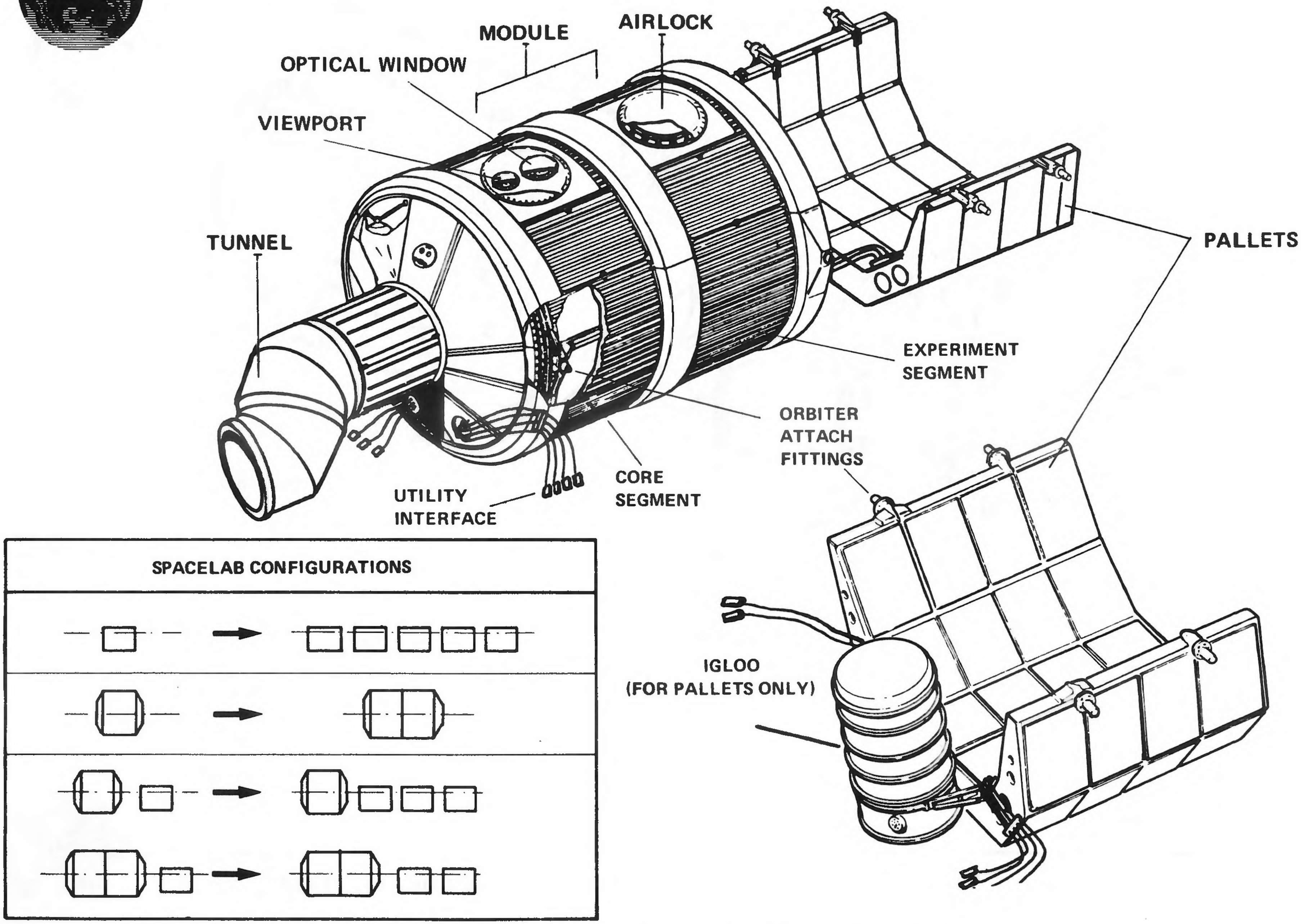
- Industrial Technology–Zero gravity lends itself to manufacturing of new alloys and other composite materials that are uniquely strong, lightweight, and temperature-resistant. Zero gravity has proved conducive to growing of very large crystals of high purity for use in electronics and producing pure glass-free of contamination for optical, electronic and laser uses.



Spacelab



European Space Agency's Spacelab



SPACELAB EXTERNAL DESIGN FEATURES

MATERIALS PROCESSING IN SPACE

Materials processing in space has progressed through NASA's Apollo, Skylab, and the Apollo-Soyuz Test Project with exciting results obtained. Space flight has opened up to science and industry a new environment in which the effects of gravity are absent. We have only begun to understand or take advantage of this new phenomenon, but it holds much promise.

Our Earth-bound environment limits certain materials processes. All our experience on Earth is constrained by the force of gravity—the mutual attraction that the Earth and everybody near it exert on each other. The weight of the objects, their falling if unsupported, and the pressure they exert on their supports are manifestations of gravity. In almost all human activities and industrial processes, gravity is the dominant force we must contend with; it constrains our motion, produces friction which must be overcome, and separates some fluids. It causes layering in alloys, contamination in glasses, separation in chemicals, and defects in electronic materials.

Materials processing is dedicated to exploring the benefits of zero-gravity (actually a tiny gravitational attraction is present; 1/3,000 to 1/6,000 of what we experience on Earth and can be called low, reduced or microgravity) and ultra-high vacuum of space (one-trillionth of an atmosphere). The findings to date from Apollo, Skylab and Apollo-Soyuz Test Project missions have brought back materials fashioned in space that were more pure and uniform than ever seen before—almost physically perfect. If we could develop materials to their full physical, thermal, chemical, optical and electric theoretical potential, major breakthroughs in medicine, electronics, energy, instrumentation, and construction could be achieved. The development of the Space Shuttle/Spacelab opens the door to working routinely in space and commercially attractive materials processing in space ventures.

NASA awarded the first major contract for Materials Processing in space payloads to the TRW Defense and Space Systems Group, Redondo Beach, Calif. TRW designs, develops, tests, integrates and delivers two Space Shuttle/Spacelab payloads to NASA. NASA's Marshall Space Flight Center, Huntsville, Ala., manages the projects. They will be used to conduct applied research in fluids technology and materials solidification relating to electronic materials, metals, ceramics, glass, and chemicals in the low-gravity space environment.

The fluid investigation will take place inside Spacelab. The solidification experiment will be located on a single pallet outside Spacelab in the payload bay of the spacecraft. The experiments will compare results with ground-based research to examine such questions as how gravity or lack of it affects crystal growth, purification, solidification, and processing in fluids. The direct results from these experiments may or may not provide answers to much-needed questions on earth. No one knows if spaceborne products will soon find their way into the homes of earthbound humans, but new forms of glass, metal alloys, and pharmaceuticals are postulated.

ATMOSPHERIC, MAGNETOSPHERIC, AND PLASMAS IN SPACE

The Atmospheric, Magnetospheric, and Plasmas (AMPS) in space program utilizes the versatile Space Shuttle/Spacelab in conducting a variety of active experiments and coordinated passive measurements.

The atmospheric and magnetic field surrounding Earth form a protective barrier against the solar wind and hostile environment of space. This life-sustaining buffer zone determines our planet's ecosystem and has a decisive effect on the Earth's climate.

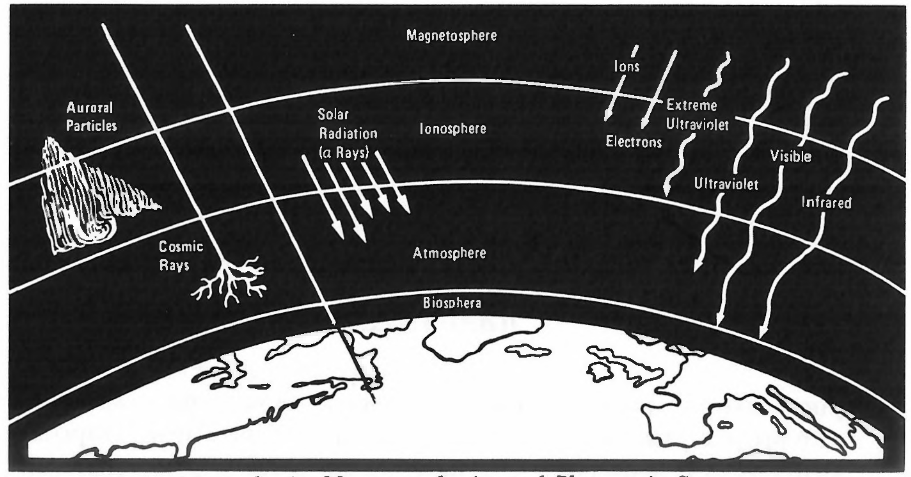
Scientific spacecraft probing of the near-earth environment has enabled the identification and sampling of the nature of the three outermost regions; the magnetosphere, ionosphere, and atmosphere. Important knowledge has been gained of these complex regions and their interactions; however, limitations of automated spacecraft payload weight and size have precluded carrying into space the broad instrument complement necessary for performing studies in a coordinated and comprehensive manner. For the first time, a full manned laboratory of instruments can be carried into a region of interest to conduct remote and in-situ investigations in a structured, evolutionary program.

The experiments involve observing the effects produced by natural processes or by disturbances that are purposely imposed. Understanding the near-earth region involves studying the earth's electric and magnetic field system, energetic particle and electromagnetic wave interactions, physical processes associated with the motion of bodies in rarefied plasmas, and the chemistry and dynamics of the upper atmosphere.

The AMPS program uses various techniques and instruments to explore four specific disciplines: particle interaction, plasma interaction and flow, atmospheric science and wave phenomena. AMPS will allow investigators to perturb the near-earth environment in a controlled and systematic manner.

Acoustic gravity wave and gas cloud expansion dynamics such as plasma density, temperature, and composition and initial investigation of wave-particle and wave-plasma instabilities in the ionosphere will be studied for possible relation between changes in solar wind velocity and magnetic fields.

Determination of atmospheric constituents by measuring infrared and ultraviolet absorption, in addition to detection of atmospheric minor constituents using an infrared interferometer and infrared radiometer will study temperature profiles, wind velocities and constituents such as hydroxyl (oxygen-hydrogen) radical chlorine chemistry, ozone and minor atmospheric constituents such as nitric oxide and chlorine oxide.



Atmospheric, Magnetospheric, and Plasmas in Space

SPACE TELESCOPE

Lockheed Missiles and Space Company, Incorporated, Sunnyvale, Calif., designs and builds the Space Telescope support systems module which is 13 meters (43 feet) long, 4.2 meters (14 feet) in diameter–very similar in size to the 3 meter (120 inch) telescope at Lick Observatory–and houses optics, sensors, support system. They are also responsible for systems engineering, analysis, integration, verification of the complete assembly, in addition to providing support of ground and flight operations under contract from NASA. Perkin-Elmer Corporation's Optical Technology Division, Danbury, Conn., is responsible for the Space Telescope optical telescope assembly, including the optics, focal plane structure, fine guidance sensor, and associated controls. NASA's Goddard Space Flight Center, Greenbelt, Md., is responsible for the scientific instruments, mission operations and data reduction. ESA provides one scientific instrument, solar arrays and participates in flight operations. NASA's Office of Space Science is responsible for overall direction of the Space Telescope System and NASA's Marshall Space Flight Center, Huntsville, Ala., is responsible for overall project management.

The Space Telescope would be placed about 270 nautical miles (310 statute miles) above the Earth where it will be free of atmospheric interference, such as filtering, haze, twinkling and light pollution, that handicaps conventional telescopes such as the famous Hale telescope at Mount Palomar, Calif., which is a 5-meter (200-inch) telescope. The Space Telescope 2.4-meter (95-inch) diameter optics will enable astronomers to detect objects 50 times fainter and view items with a clarity or resolution seven times better than ground-based observations for periods as long as 30 to 40 hours. A major technical challenge is to lock onto viewing targets and insure the telescope maintains a pointing stability of .007 arc seconds, which is roughly equivalent to a marksman in Boston zeroing in on a baseball in Los Angeles, Calif.

In orbit, the telescope consists of three parts: an Optical Telescope Assembly, Scientific Instruments, and a Support Systems Module.

The Optical Telescope Assembly comprises 2.4-meter (95-inch) class reflecting telescope. A meteoroid shield light baffles, and sunshade protects the optics.

The five Scientific Instruments (SI's) provide the means of converting the telescope images to useful scientific data. The SI's for the first flight include the wide field/planetary camera (Jet Propulsion Laboratory), faint object camera (ESA/Dormer), faint object spectrograph (Martin Marietta), high speed photometry (University of Wisconsin), and high resolution spectrograph (Ball Brothers). Astronomy (University of Texas) is performed using one of the five guidance sensors.

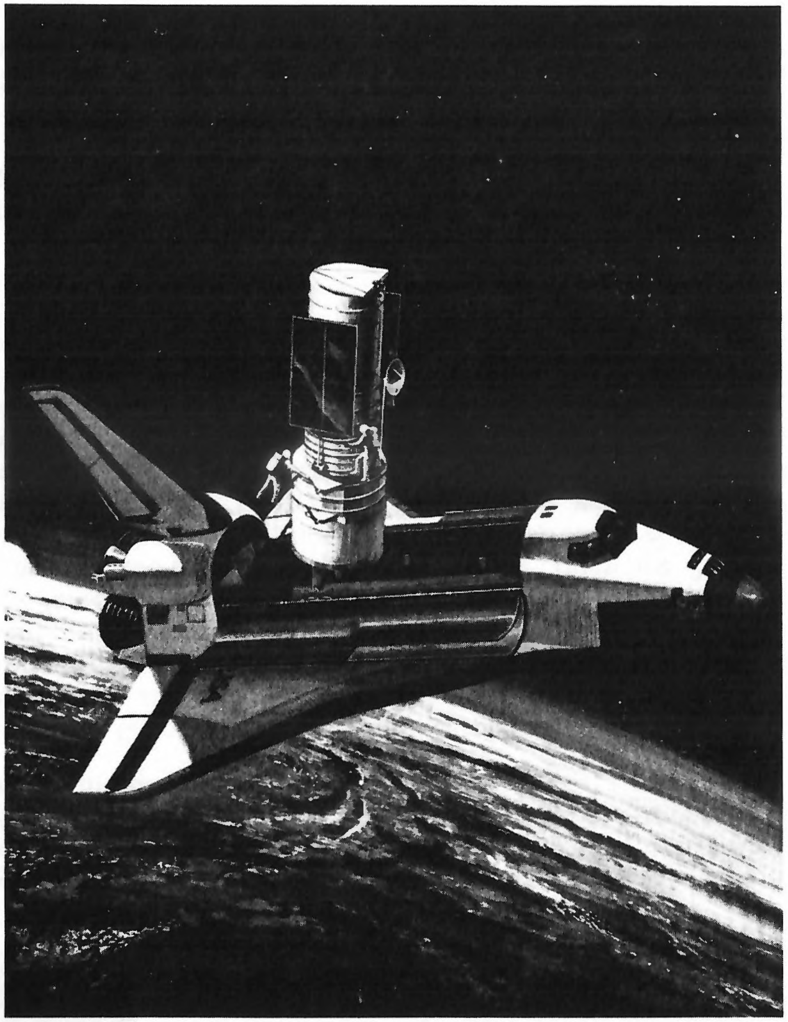
The Support Systems Module contains a very precise pointing and control system, communications system, thermal control system, and the power system. Electrical power is supplied by solar panels and batteries.

In operation, light enters the open end of the telescope, is projected by the primary mirror onto the smaller secondary mirror and from there is deflected to the scientific instruments for analysis. The Space Telescope will also make ultraviolet and infrared measurements not possible on Earth.

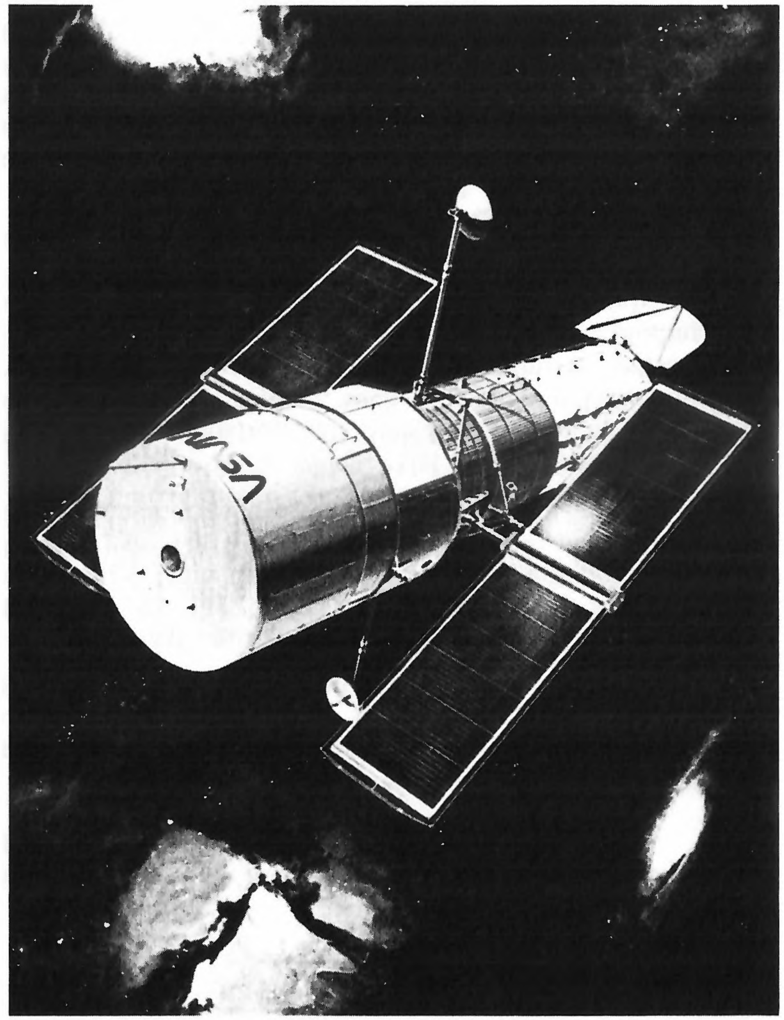
Celestial objects which exist under conditions of gravity, temperature, radiation and time that cannot be duplicated here on Earth will be studied, such as gaseous nebulae, dust clouds, variable stars, binary stars, novae, supernovae, pulsars, neutron stars, black holes, forming galaxies, quasars, exploding galaxies, and quasars. The nearby stars, solar system planets, comets, and the planetary systems of stars within 100 light years will also be subjected to Space Telescope scrutiny.

The detailed study of plasmas in space or of mysterious quasars for instance could unlock secrets of tremendous new energy sources. Closer observation of planets in the Earth's own solar system should also help us better understand the significance of changes now taking place on this planet. Such insights could be vital to preserving the world's environment or adapting to changes beyond our control.

The Space Telescope is planned to operate for 15 years on-orbit. It is capable of being serviced and repaired while it is in orbit and new instruments can be installed. It can also be returned to Earth for major repairs and improvements and then replaced into orbit.



Space Telescope Servicing



Space Telescope

SPACE SHUTTLE FLIGHTS

In preparation for the first flight into space of Orbiter 102 (*Columbia*), tens of thousands of hours of tests and simulations are expended at government and contractor facilities throughout the nation to qualify all the structures, flight equipment, and computer programs software.

Major test programs included the 13 flights of the Approach and Landing Test (ALT) program at NASA's Dryden Flight Research Center, Edwards Air Force Base, Calif. The main propulsion test article–consisting of the orbiter aft fuselage, three Space Shuttle main engines, external tank, and truss arrangement to simulate the mid fuselage–is qualifying the main propulsion system at the National Space Technology Laboratory in Mississippi. The solid rocket booster's solid-propellant rocket motor is being qualified at Thiokol Chemical Corp. near Brigham City, Utah. The orbiter reaction control and orbital maneuvering systems are being qualified at NASA's White Sands Test Facility near Las Cruces, N.M. Flight simulation and avionics testing are qualifying the Space Shuttle avionics at JSC and at Rockwell's Space Systems Group in Downey, Calif. The orbiter full-scale structure was tested at Lockheed's facility in Palmdale, Calif. The orbiter vehicle (OV-101), external tank, and solid rocket boosters have undergone a mated vertical ground vibration test program at MSFC. OV-101, external tank and solid rocket boosters have also completed a fit check and checkout with the mobile launch platform and launch complex 39A at KSC.

The initial four flights of *Columbia* will be launched from KSC and will verify design and operational capability of the Shuttle and all of the ground-based monitoring, communications, and support systems. The first flight is structured to minimize risks and complexity and will be two days in duration. The subsequent three flights will progress from five to seven days and will develop and demonstrate mission and payload capabilities which become progressively more complex.

The first four orbital space flights of *Columbia* are scheduled to land at Dryden FRC, Edwards AFB, Calif.

KSC launch operations has responsibility for all mating, prelaunch testing, and launch control ground activities until the Space Shuttle vehicle has cleared the umbilical tower; then responsibility is turned over to JSC's Mission Control Center in Houston. This responsibility includes earth entry and approach and landing until runout completion, at which time the orbiter is handed over to the post-landing operations at the landing site for turnaround and relaunch. KSC also will process the solid rocket boosters and external tank for launch, as well as for re-cycling the solid rocket boosters for reuse.

The following table gives tentative plans for the first 68 STS flights. It should be remembered that these details are preliminary and may be changed as the program progresses.

STS Flt	Vehicle	No. Crew	Length (days)	Launch/Land Sites	Inc./Altitude [°/mi (nm)]	Payloads*	Remarks
1	Columbia	2	3	KSC/DFRC	40.3/172 (150)	1, 3	Objectives: safe ascent, return; assess ascent, entry loads, SRB/external tank performance, payload bay door performance, communications, attitude control and maneuvering, closed-circuit TV, IMU, passive thermal control Main engine throttling limits: 65-100% RPL Max q: 580/660 psf (ascent) AOA landing site: Northrup Strip, N.M. Entry angle of attack: 40° Crossrange: 690 mi (600 nm) Autoland control mode: control stick steering (Runway 23) DFRC at EAFB dry lake bed Payload weight: 10,190 lb (up and down)
2	Columbia	2	5	KSC/DFRC	40.3/172 (150)	1, 2, 3, 5	Objectives: Hydraulic thermal control tests, payload deployment retrieval system unloaded arm tests, payload bay door latch tests, communications tests, IMU performance, OMS performance, RCS/flight control tests, airlock/EMU checkout OSTA-1 data takes, radiator heat sink tests, flutter boundary test, entry aero/structural test. +Z local vertical 88 hr. Main engine throttling limits: 65-100% RPL Max q: 600/720 psf (ascent) AOA landing site: Northrup Strip, N.M. Entry angle of attack: 40° Crossrange: 690 mi (600 nm) Autoland control mode to flare: control stick steering (Runway 23) DFRC at EAFB dry lake bed Payload weight: 15,860 lb (up and down)

*See Page 27 for Payload listings.

Space Transportation System

STS Flt	Vehicle	No. Crew	Length (days)	Launch/Land Sites	Inc./Altitude [°/mi (nm)]	Payloads*	Remarks
3	Columbia	2	7	KSC/DFRC	38/172 (150)	1, 2, 3, 6	Objectives: Payload deployment retrieval system loaded arm test, IECM plume survey, payload bay door thermal test, OMS performance, star tracker/IMU tests, backup flight system test, radiator heat sink test, propulsion thermal soakback, entry aero/structural tests, flutter boundary test, on-orbit thermal test, autoland demonstration, active thermal control system test. Passive thermal control, 10 hrs; +X solar inertial, 80 hr; -Z solar inertial, 40 hr; +Z solar inertial, 3 revolutions Main engine throttling limits: 65-100% RPL Max q: 650/765 psf (ascent) AOA landing site: DFRC at EAFB 23 Entry angle of attack: 40° Crossrange: 690 mi (600 nm) Autoland control mode: automatic through rollout (Runway 23) DFRC at EAFB dry lake bed Payload weight: 18,760 lb (up and down)
4	Columbia			KSC/DFRC	38.5/184 (160)	1, 2, 3, 4, 7 or 21	Objectives: Payload bay door thermal tests, IECM contamination monitoring, OMS performance, IMU performance, propulsion thermal soakback, radiator heat sink test, UHF entry test, entry aerodynamics test, flutter boundary test, on-orbit thermal test, OSS-1 data takes, autoland demonstration. Passive thermal control, 10 hrs; -Z solar inertial, 80 hr; +Z solar inertial, 26 hr Main engine throttling limits: 65-100% RPL AOA landing site: DFRC at EAFB Max q: 680/790 psf (ascent) Entry angle of attack: 38°/28° Crossrange: between 690 mi (600 nm) and 920 mi (800 nm) Autoland control mode: through rollout at EAFB Runway 04 Payload weight: 19,640 lb (up), 19,640 lb (down)
5	Columbia	2	2	KSC/KSC	28.5/176 (153.5)	3, 8	Main engine throttling: 102/109 RPL lightweight ET
6	Challenger	3	5	KSC/KSC	28.5/184 (160)	10, 17, 18, 20	First flight of Challenger (OV-099) Main engine throttling: 102/109% RPL
7	Columbia	3	3	KSC/KSC	28.5/176 (153.5)	3, 13, 69, 70, 71, 77	Main engine throttling: 102/109% RPL lightweight ET
8	Challenger	3	5	KSC/KSC	28.5/184 (160)	24, 34, 35, 68	Main engine throttling: 102/109% RPL
9	Challenger			KSC/KSC		32	Main engine throttling: 102/109% RPL
10	Columbia	6	7	KSC/KSC	57/155 (135)	3, 15, 69, 70, 71	Ferried to Western Test Range after this flight Main engine throttling: 102/109% RPL

*See Page 27 for Payload Listings

STS Flt	Vehicle	No. Crew	Length (days)	Launch/Land Sites	Inc./Altitude [°/mi (nm)]	Payloads*	Remarks
11	Challenger			KSC/KSC		45	
12	Challenger	3	3	KSC/KSC	28.5/176 (153.5)	3, 16, 27, 69, 70, 71	Lightweight ET
13	Challenger			KSC/KSC		58	
14	Challenger	6	7	KSC/KSC	50/232 (202)	22, 28	Limited to 7 days with four EPS tank sets
15	Challenger	3	3	KSC/KSC	28.5/176 (153.5)	23, 74	Lightweight ET
16	Discovery	3	3	KSC/KSC	28.5/368 (320)	9, 53	First flight of Discovery (OV-103), 500 fps OMS kit
17	Challenger	3	7	KSC/KSC	28.5/184 (160)	12, 30, 33, 36	
18	Discovery	2	1	KSC/KSC	28.5/115 x 253 (100 x 220)	65	
19	Challenger	2	1	KSC/KSC	28.5/115 x 253 (100 x 220)	66	Discovery used as a backup for this mission
20	Discovery	6	7	KSC/KSC	57/230 (200)	31	
21	Challenger	3	5	KSC/KSC	28.5/184 (160)	42, 44, 48, 87	
22	Discovery	6	7	KSC/KSC	46/184 (160)	67	
23	Challenger			KSC/KSC		38	
1VA	Columbia	2	2	WTR/KSC	98.8/184 (160)	3, 69, 70, 71, 88	First Western Test Range launch, lands at KSC
24	Discovery	4	5	KSC/KSC	28.5/184 (160)	3, 11, 38, 51, 55, 57, 69, 70, 71	Lands at Vandenberg AFB
25	Challenger	6	7	KSC/KSC	57/184 (160)	52	
26	Columbia	3	5	KSC/KSC	28.5/184 (160)	3, 54, 61, 69, 70, 71, 72, 75	Main engine throttling: 102/109% RPL
27	Challenger	3	3	KSC/KSC	28.5/316 (275+)	14, 25	500 fps OMS kit
28	Columbia	3	7	KSC/KSC	28.5/184 (160)	3, 14, 25, 69, 70, 71, 73, 90	Main engine throttling: 102/109% RPL
1V	Discovery			WTR/WTR		89	
29	Challenger	3	3	KSC/KSC		91	
30	Challenger	6	7	KSC/KSC	57/230 (200)	79	

*See Page 27 for Payload Listings

Space Transportation System

STS Flt	Vehicle	No. Crew	Length (days)	Launch/Land Sites	Inc./Altitude [°/mi (nm)]	Payloads*	Remarks
51	Challenger			KSC/KSC		105	
52	Columbia	3	5	KSC/KSC	28.5/184 (160)	46, 49, 107	
53	Atlantis			KSC/KSC		109	
7W	Discovery			WTR/WTR		111	
54	Challenger	6	7	KSC/KSC	28.5/184 (160)	110	
55	Columbia			KSC/KSC		112	
56	Atlantis			KSC/KSC		46	
8V	Discovery			WTR/WTR		114	
57	Challenger			KSC/KSC		46	
58	Columbia	3	5	KSC/KSC	28.5/184 (160)	46, 115	
59	Atlantis	2	1	KSC/KSC	28.5/184 (160)	62	
60	Challenger	6	7	KSC/KSC	57/230 (200)	117	
61	Columbia			KSC/KSC		121	
62	Discovery	3	5	KSC/KSC	28.5/184 (160)	46, 63	
63	Challenger			KSC/KSC		116	
9V	Discovery	3	3	WTR/WTR	99/184 (160)	46, 78	
64	Columbia			KSC/KSC		118	
65	Atlantis	3	7	KSC/KSC	28.5/184 (160)	46, 113	
10V	Discovery	6	7	WTR/WTR		122	
66	Challenger	6	7	KSC/KSC	46/184 (160)	123	
67	Columbia			KSC/KSC		46	
68	Atlantis			KSC/KSC		119	

*See Page 27 for Payload Listings

STS Flt	Vehicle	No. Crew	Length (days)	Launch/Land Sites	Inc./Altitude [°/mi (nm)]	Payloads*	Remarks
31	Columbia			KSC/KSC		93	
32	Challenger	4	5	KSC/KSC	28.5/184 (160)	60, 80, 82, 92	
2V	Discovery			WTR/WTR		94	
33	Columbia			KSC/KSC		96	
34	Challenger	6	7	KSC/KSC	57/230 (200)	47	
35	Atlantis	2	1	KSC/KSC	28.5/115 x 253 (100 x 220)	37	First flight of Atlantis (OV-104)
3V	Discovery			WTR/WTR		97	
36	Challenger	2	1	KSC/KSC	28.5/115 x 253 (100 x 220)	95	
37	Columbia			KSC/KSC		100	
38	Atlantis	6	7	KSC/KSC	28.5/184 (160)	86	
39	Challenger	3	3	KSC/KSC	28.5/184 (160)	41, 98	
40	Columbia	3	5	KSC/KSC	28.5/184 (160)	19, 81, 101, 103	
41	Atlantis	6	7	KSC/KSC	57/230 (200)	99	
4V	Discovery			WTR/WTR		29	
42	Challenger			KSC/KSC		40	
43	Columbia			KSC/KSC		38	
44	Atlantis	3	4	KSC/KSC	28.5/184 (160)	26, 84, 85	
45	Challenger	6	7	KSC/KSC	57/230 (200)	83	
5V	Discovery			WTR/WTR		59	
46	Columbia	3	7	KSC/KSC	28.5/184 (160)	39, 43, 76, 108	
47	Atlantis	3	3	KSC/KSC	28.5/TBD	9, 46, 50	OMS kit 500 fps
48	Challenger	6	7	KSC/KSC	46/184 (160)	102	
49	Columbia	3	3	KSC/KSC	28.5/184 (160)	46, 56, 120	
6V	Discovery			WTR/WTR		104	
50	Atlantis	6	7	KSC/KSC	57/230 (200)	106	

*See Page 27 for Payload Listings

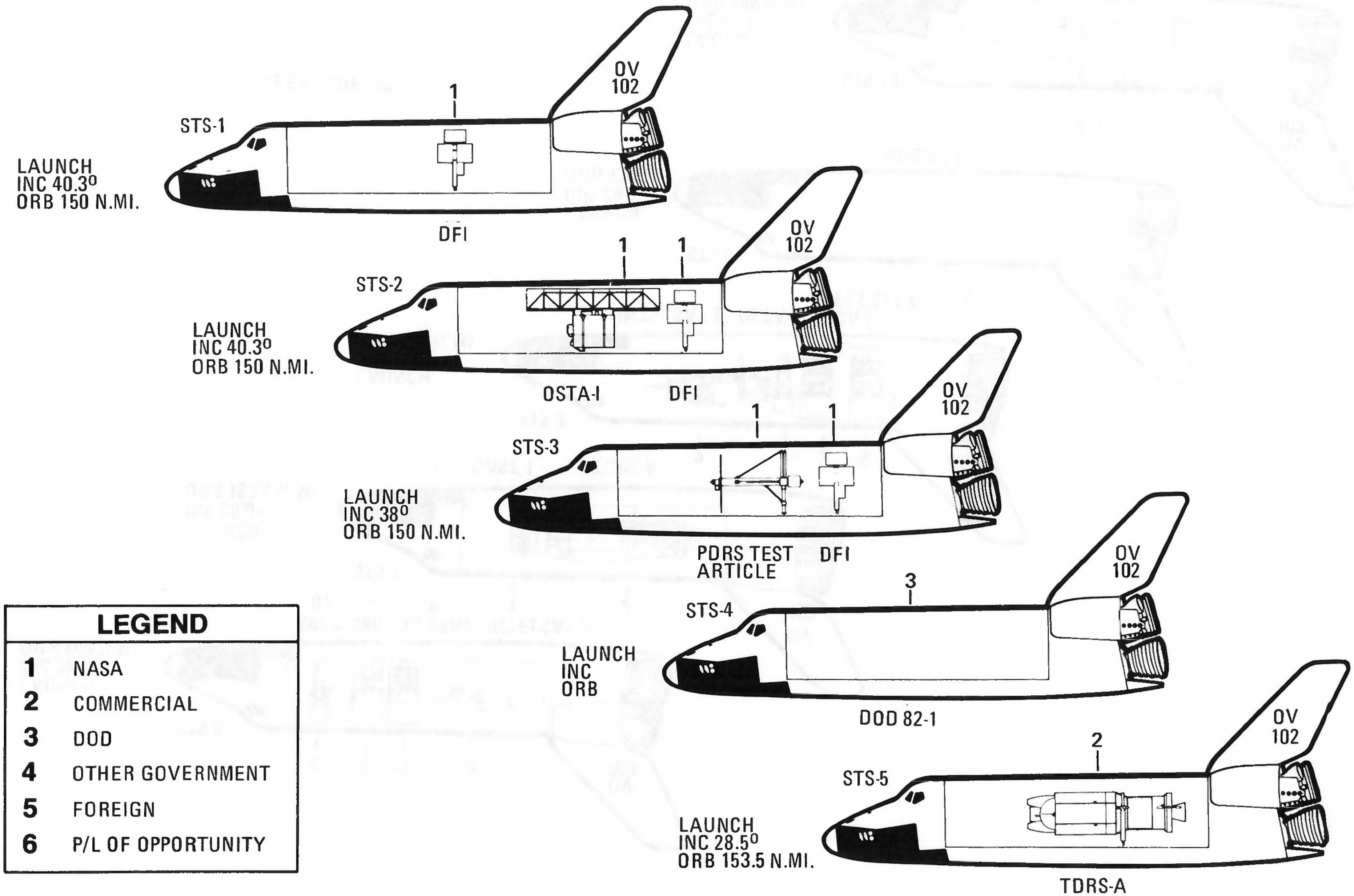
*Payloads

1. DFI (development flight instrumentation)
2. IECM (induced environmental control monitor)
3. ACIP (aerodynamic coefficient identification package)
4. Gas test article
5. OSTA–1 (Office of Space and Terrestrial Applications) OFT (orbital flight test) pallet
6. PDRS (payload deployment/retrieval test article)/unique carrier
7. OSS–1 (office of space science experiment)/OFT pallet
8. TDRS–A (tracking and data relay satellite) IUS–2 (inertial upper stage): two–stage booster used to put payloads into higher orbit
9. Orbital maneuvering system kit/support system
10. Intelsat (international communication satellite) F–5/SSUS–A (Spinning Solid Upper Stage–A – Heavy Payloads Class)
11. SBS–B (satellite business system)/SSU–D (Light Payloads Class)
12. MPS (materials processing science)/OFT pallet
13. TDRS–B/IUS–2
14. SMM (solar maximum mission)
15. Spacelab–1/long pressurized module plus pallet
16. TDRS–C/IUS–2
17. SPAS–01 (Indonesian communication satellite)/special structure
18. SBS–C/SSUS–D
19. AUST DOMSAT–A (Australian domestic satellite)/SSUS–D
20. Telesat–E (Canada telecommunications satellite)/SSUS–D
21. DOD 82–1
22. Spacelab–2/igloo plus three pallets
23. TDRS–D/IUS–2
24. Telesat–F/SSUS–D
25. LDEF (long–duration exposure facility) deploy/LDEF
26. OSS–4/OFT pallet
27. Insat–1A (India Communications Satellite)/SSUS–D
28. Cosmic ray/unique structure
29. Opportunity Flight
30. Telesat–G/SSUS–D
31. Spacelab–3/long pressurized module plus pallet
32. DOD 83–1 (department of defense)
33. PALAPA–B–1 (Indonesian communication satellite)/SSUS–D
34. RCA (Radio Corporation of America)–E/SSUS–D
35. OSTA–2/special structure
36. Synchom IV–1 (Hughes synchronous communication satellite)/ unique stage
37. Solar polar/IUS–3 (NASA)
38. Reflight opportunity
39. Telesat–H/SSUS–D
40. DOD 85–8
41. Synchom IV–5/unique stage
42. Synchom IV–2/unique stage
43. AUST–DOMSAT–B/SSUS–D
44. PALAPA–B–2/SSUS–D
45. DOD 83–2
46. Payload of Opportunity
47. Spacelab–5 astrophysics/igloo plus four pallets
48. RCA–F/SSUS–D
49. AUST–DOMSAT–C/SSUS–D
50. LDEF retrieve/LDEF
51. OSS–2/OFT pallet
52. Spacelab D–1 (West Germany)/long pressurized module plus Spacelab pallet
53. Space telescope/unique structure
54. Synchom IV–3/unique stage
55. Arabsat–A (Saudi Arabian communications satellite)/SSUS–D
56. GOES–G/SSUS–D
57. AT&T–1 (American Telephone and Telegraph satellite)/SSUS–D
58. DOD 84–1
59. DOD 86–1
60. Synchom IV–4/unique stage
61. Arabsat–B/SSUS–D
62. VOIR (Venus orbiting imaging radar)/IUS–3
63. OSTA–5/special structure
64. GSAT (geosynchronous satellite)A–1/SSUS–D
65. Galileo orbiter/IUS–3
66. Galileo probe/IUS–3
67. Spacelab–4/long pressurized module life science/ERBS (earth radiation budget satellite) unique structure
68. Intelsat F–6/SSUS–A
69. SILTS (Shuttle infrared leeside temperature sensing)
70. SEADS (Shuttle entry air data sensor)
71. SUMS (Shuttle upper atmosphere mass spectrometer)
72. OSTA–3/special structure
73. Intelsat–C/SSUS–A
74. Intelsat–1/SSUS–D

***Payloads (Cont)**

75. RCA–G/SSUS–D
76. MPS–3/OFT pallet
77. OAST–1 (Office of Aeronautics and Space Technology)/special structure
78. COBE (cosmic background explorer)/special stage
79. Spacelab–6 solar terrestrial/Spacelab short module plus three pallets
80. OSS–3/OFT pallet
81. Intelsat–D/SSUS–A
82. RCA–H/SSUS–D
83. Spacelab–7 earth observation, igloo plus four pallets
84. Intelsat–E/SSUS–A
85. Payload of opportunity/SSUS–D
86. Spacelab–8 materials processing/long pressurized module plus one pallet
87. TELESAT–I/SSUS–D
88. NOAA–G/unique stage
89. DOD 85–1
90. MPS–2/DFT pallet
91. DOD 85–2
92. AT&T–3/SSUS–D
93. DOD 85–3
94. DOD 85–4
95. Solar polar/IUS–3 (ESA)
96. DOD 85–5
97. DOD 85–6
98. GRO (gamma ray observatory)/MMS (multimission modular spacecraft)
99. Spacelab D–4 (West Germany) igloo plus four pallets
100. DOD 85–7
101. OSTA–4/special structure
102. Spacelab–10 life sciences long pressurized module plus one Spacelab pallet
103. GSAT–A2/SSUS–D
104. NASA free flyer
105. DOD 86–2
106. Spacelab–16 solar physics/igloo plus four Spacelab pallets
107. OSS–5/OFT pallet
108. EUVE (extreme ultra–violet experiment)/special stage
109. DOD 86–3
110. Spacelab–13 materials science long pressurized module/plus one Spacelab pallet
111 DOD 86–4
112. DOD 86–5
113. MPS–4/OFT pallet
114. DOD 86–6
115. OSS–6/DFT pallet
116. DOD 86–7
117. Spacelab–9 Astrophysics/igloo plus four Spacelab pallets
118. DOD 86–8
119. DOD 86–9
120. Intelsat–F/SSUS–A
121. Reflight opportunity
122. Spacelab–11 space plasma short pressurized module/plus three spacelab pallets
123. Spacelab–15 life sciences pressurized long module/plus one Spacelab pallet.

STS FLIGHT ASSIGNMENT BASELINE



LEGEND	
1	NASA
2	COMMERCIAL
3	DOD
4	OTHER GOVERNMENT
5	FOREIGN
6	P/L OF OPPORTUNITY

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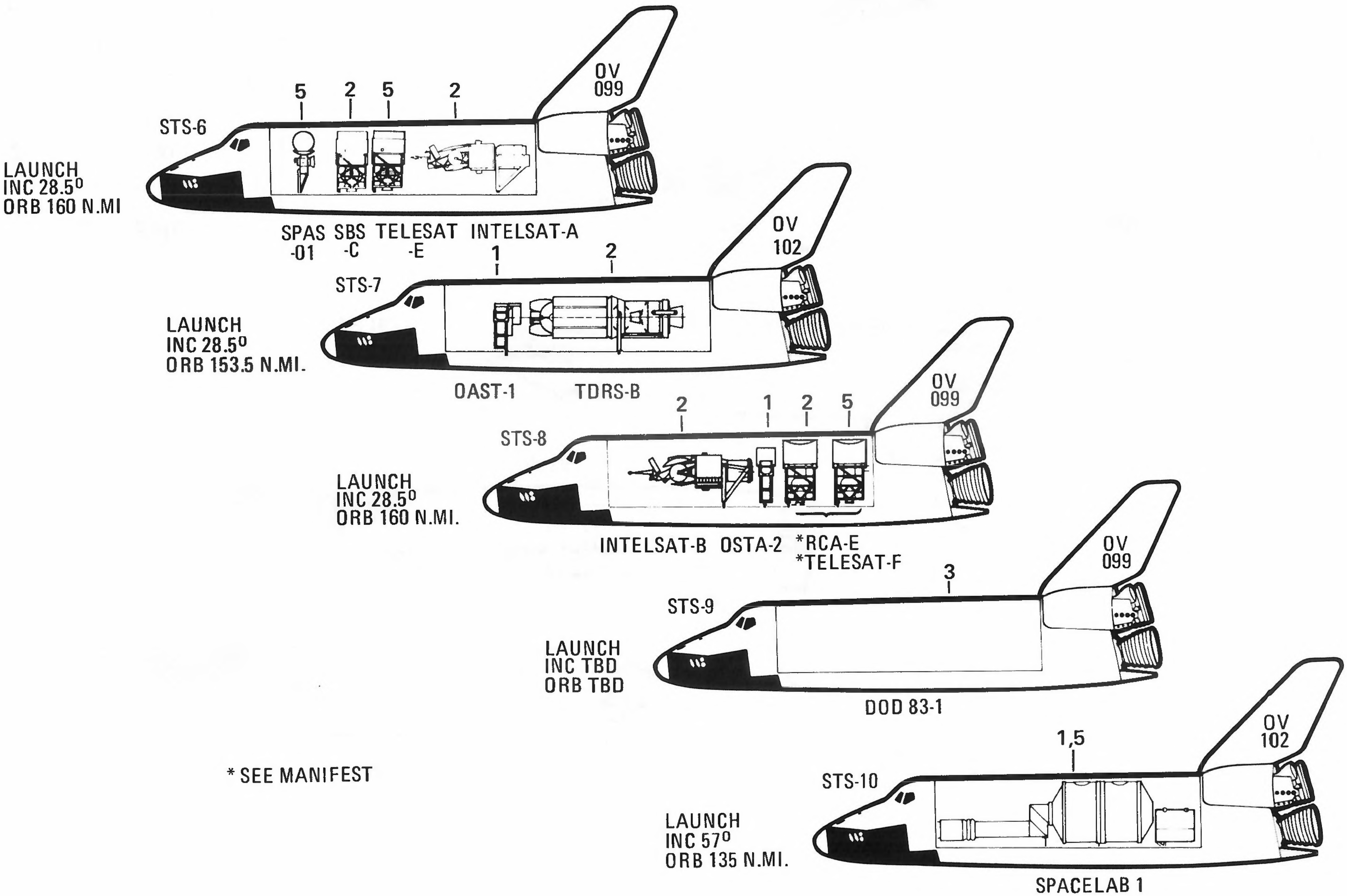

* SEE MANIFEST

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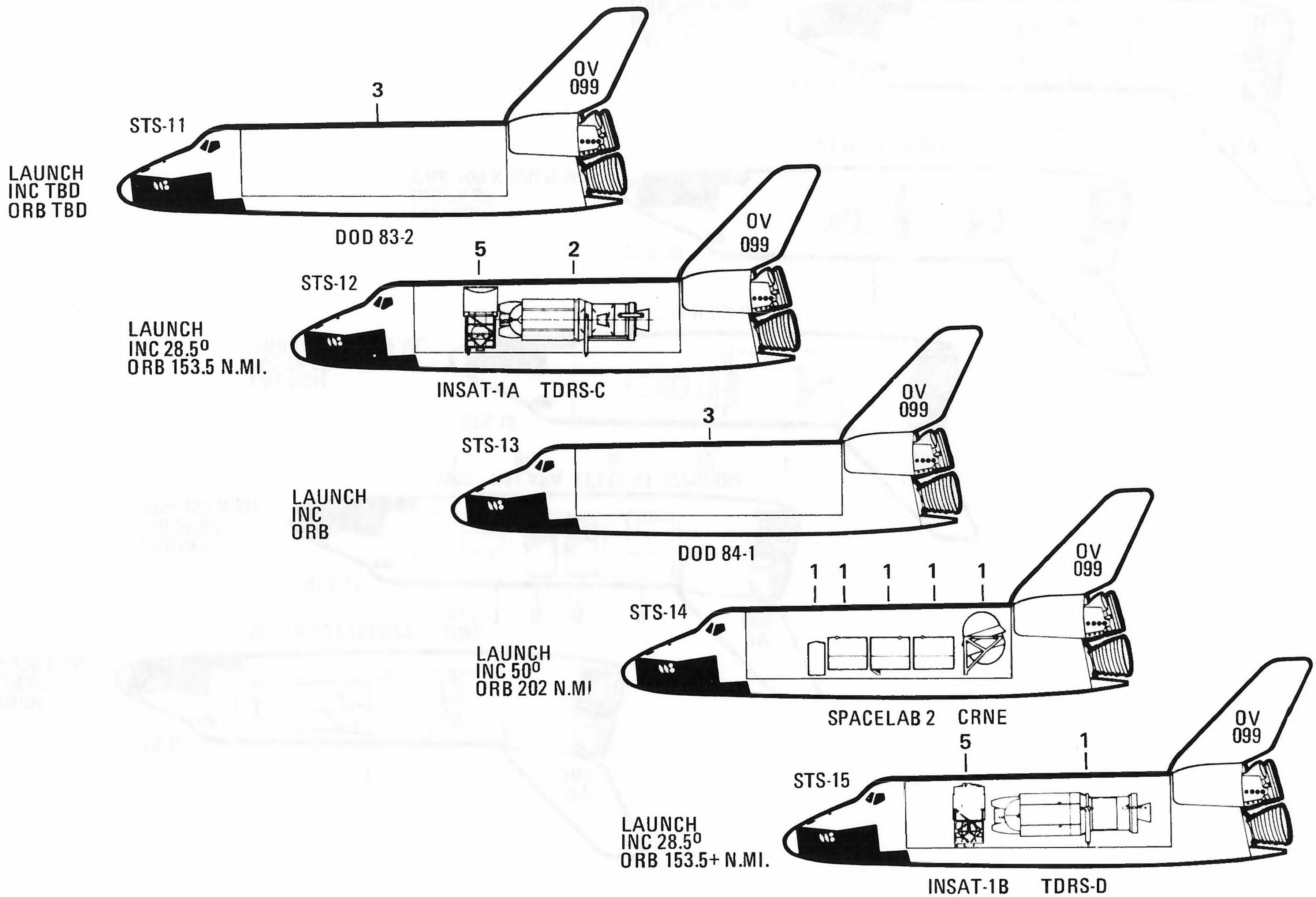

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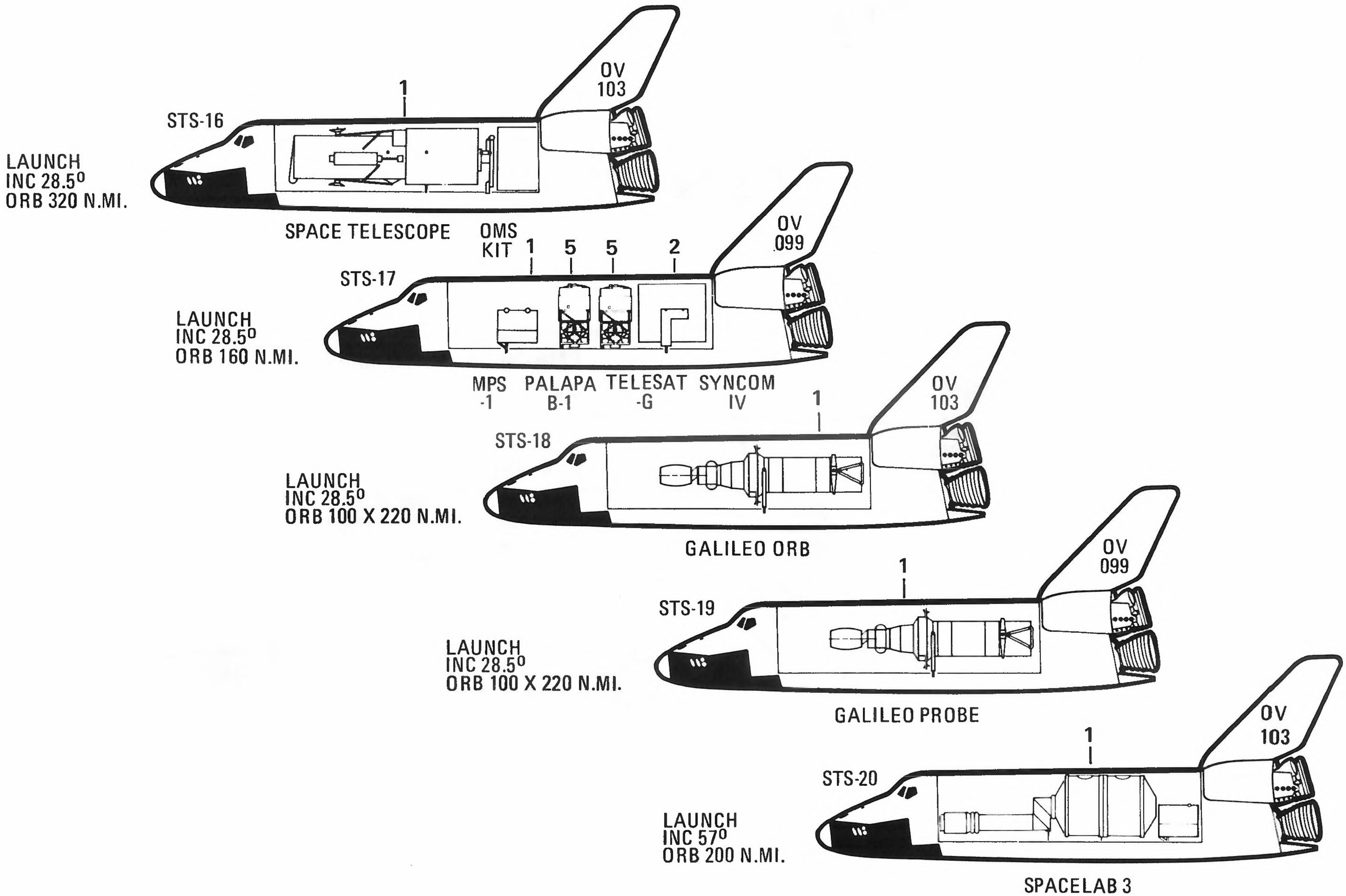

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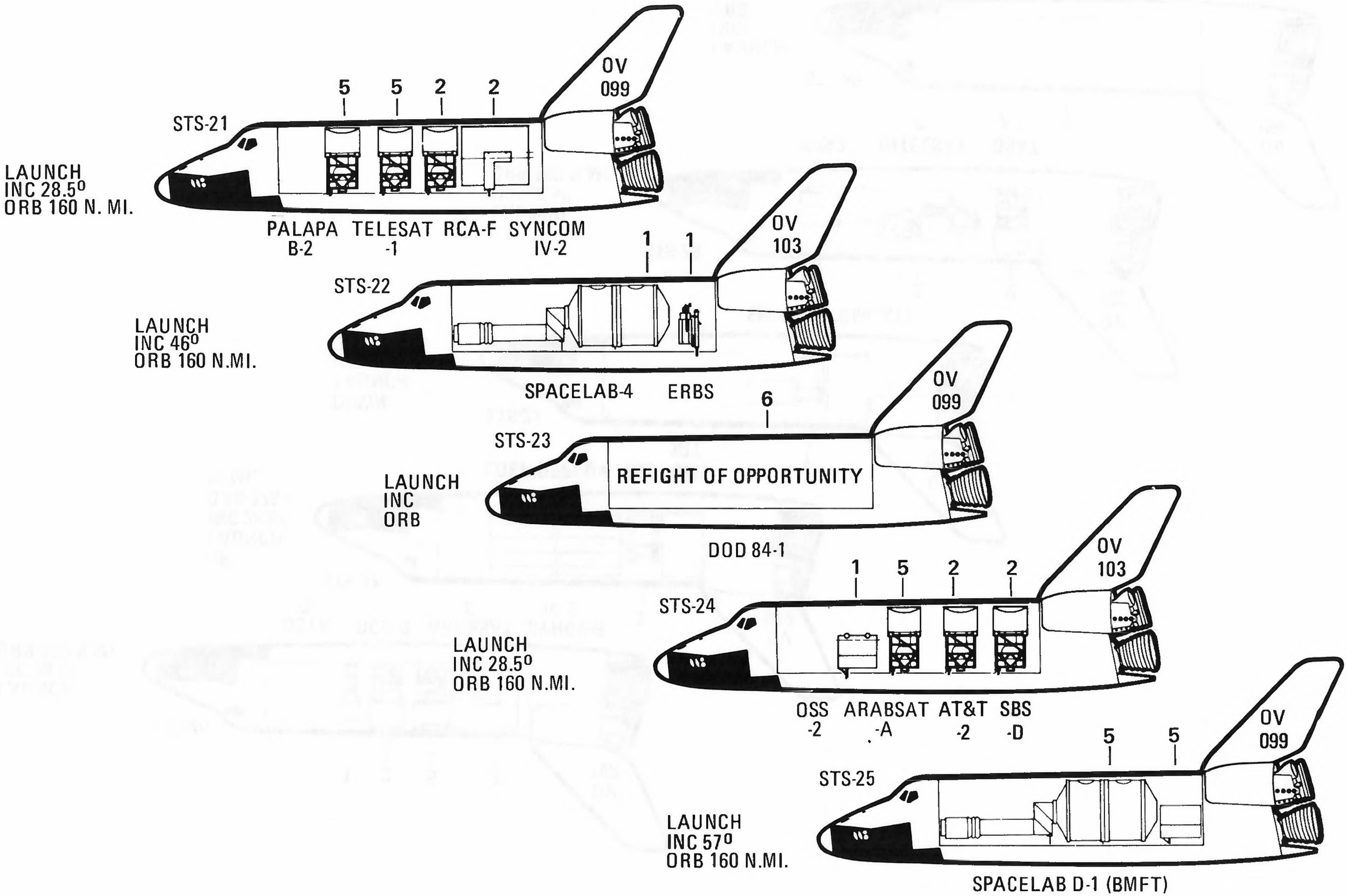

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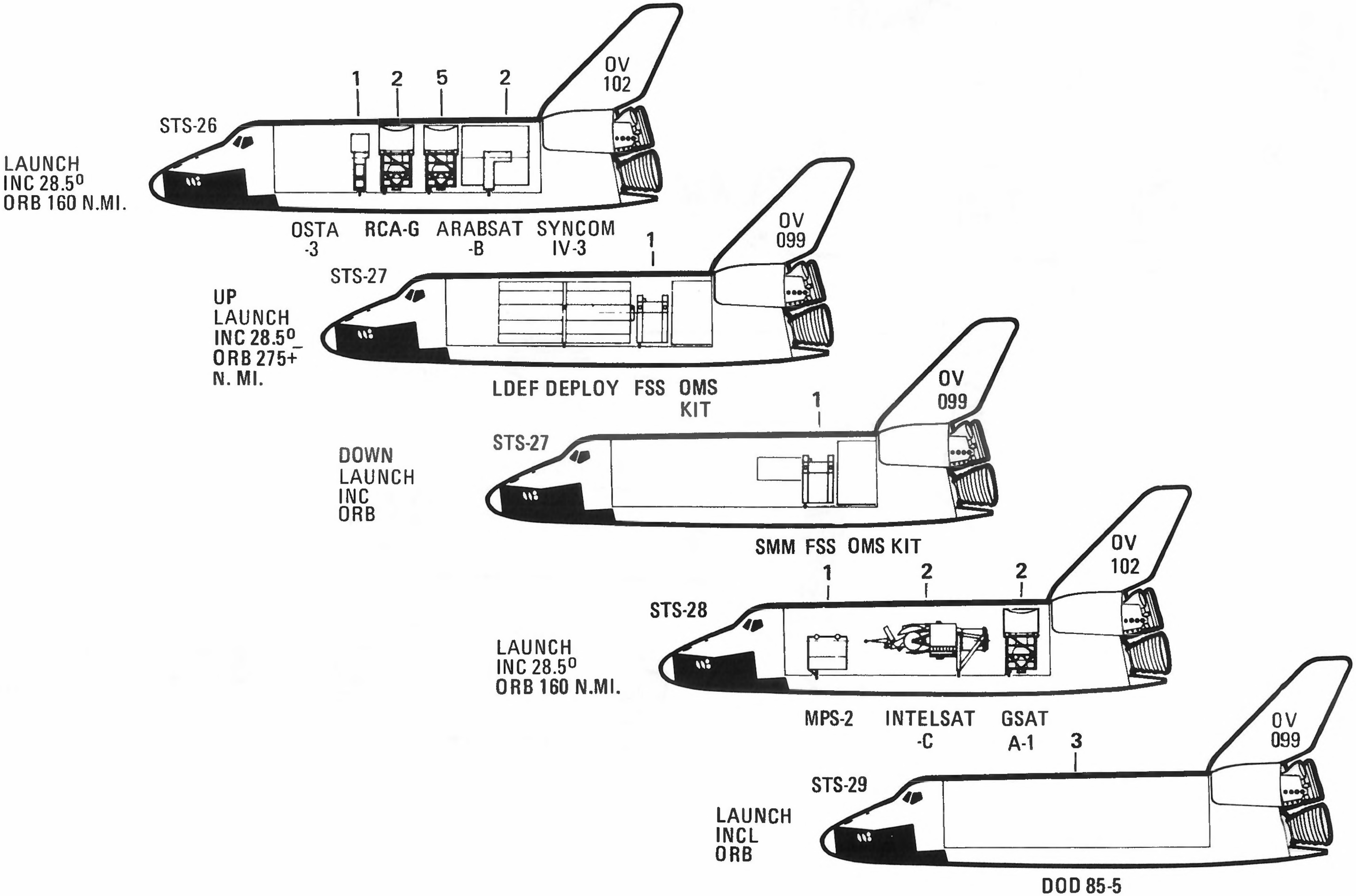

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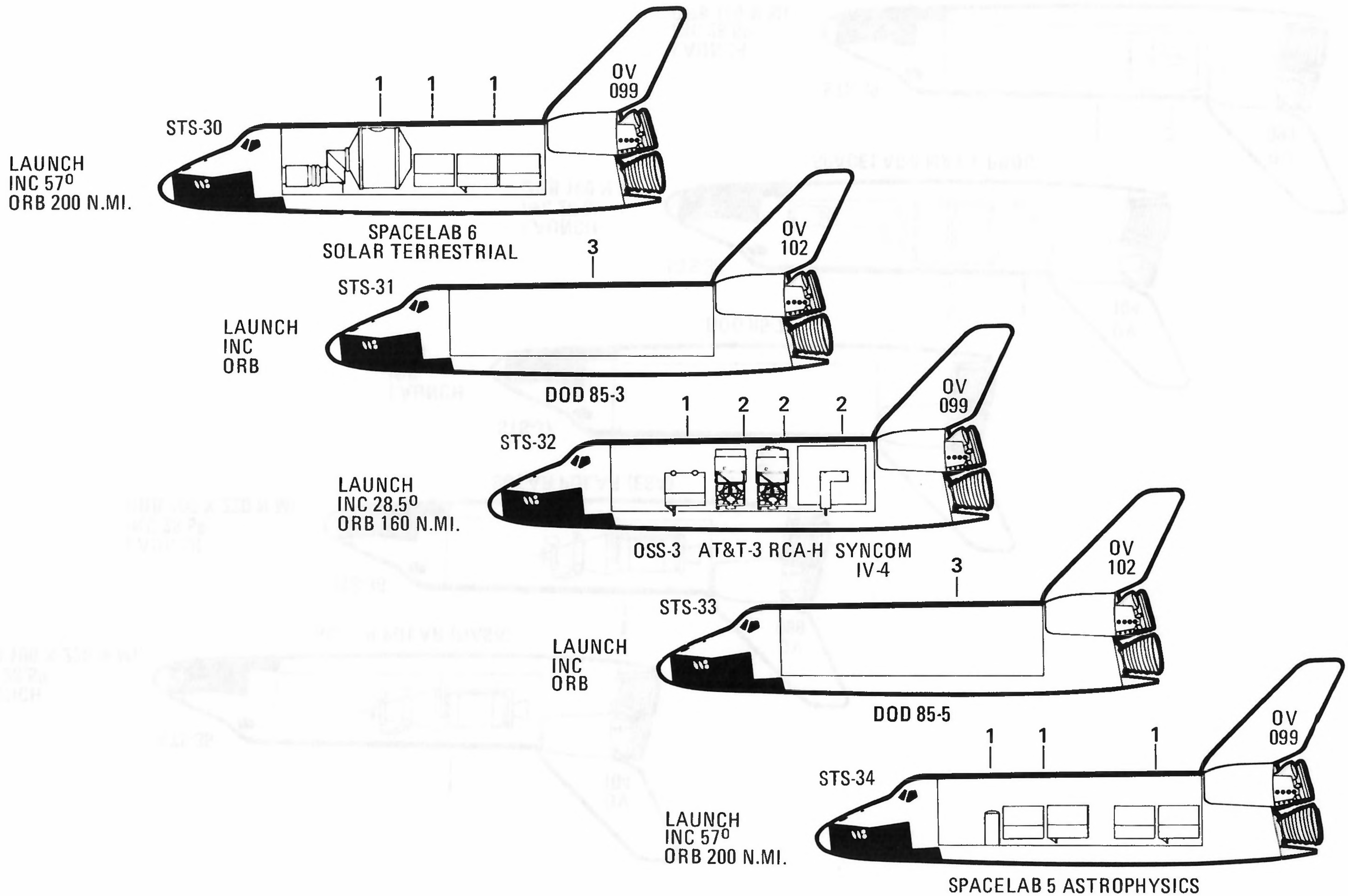

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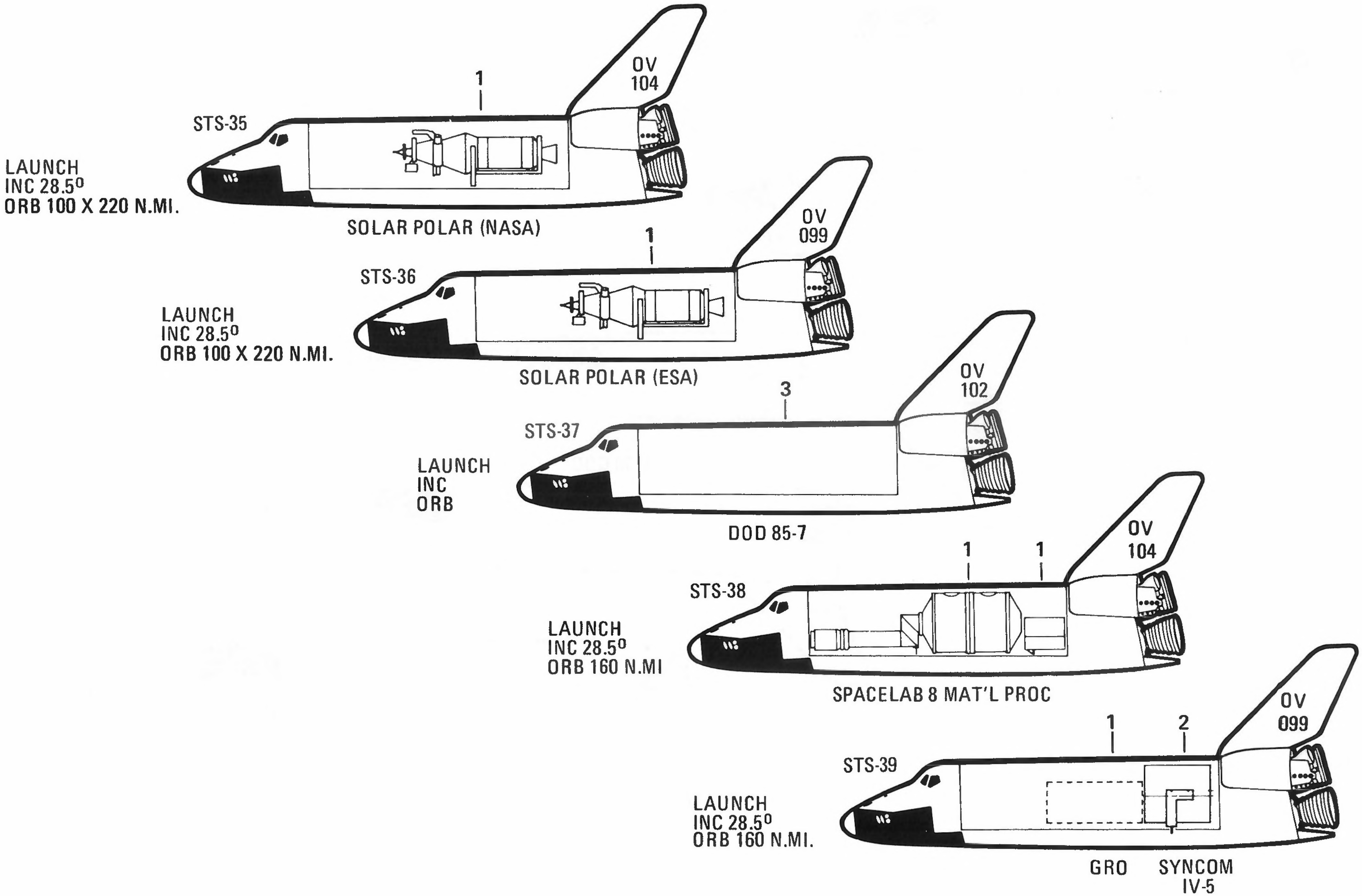

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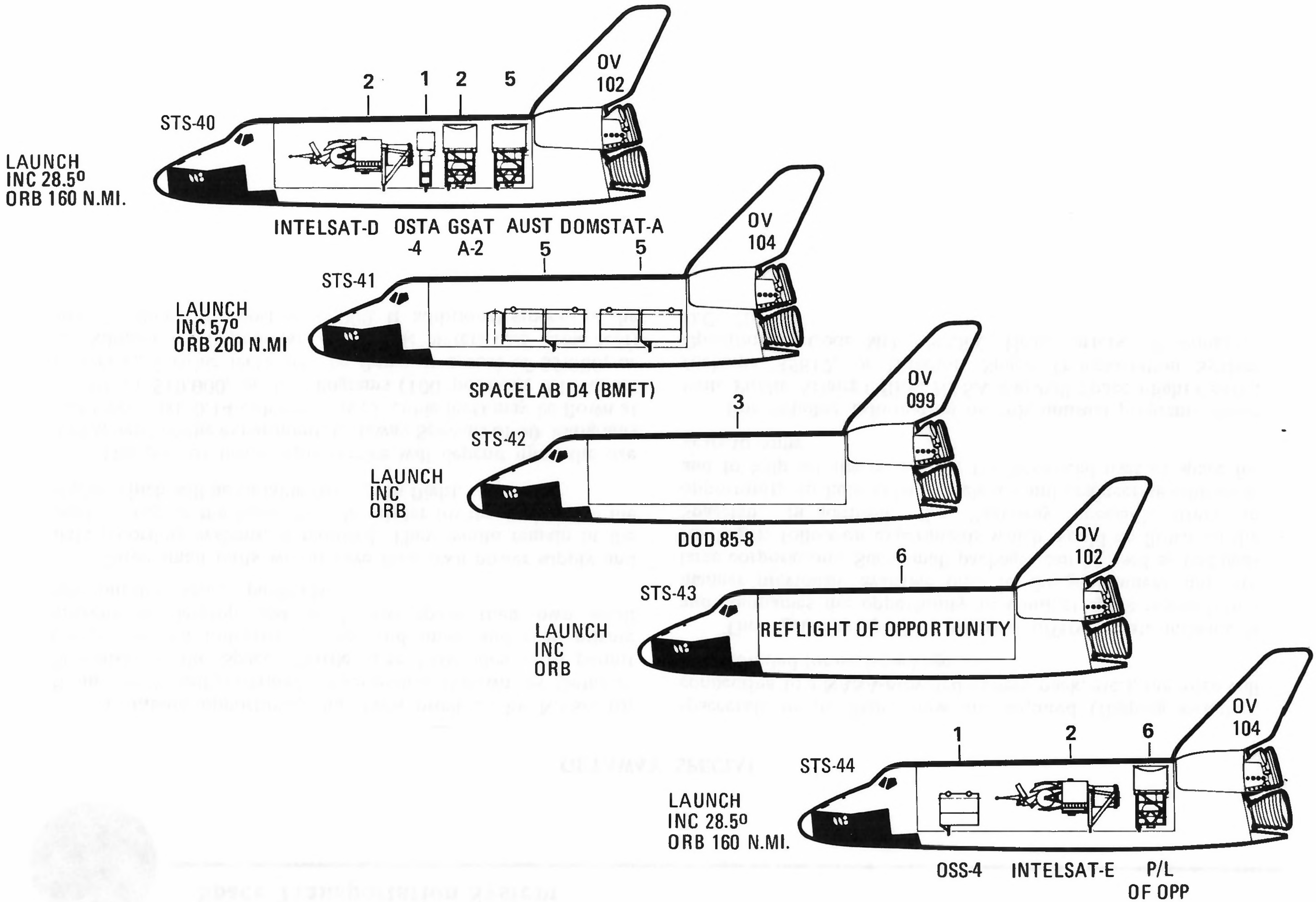


GETAWAY SPECIAL

A unique opportunity has been provided by NASA for flying small self-contained experiments (known as Getaway Specials) on the Space Shuttle. The basic idea is to permit companies and universities, large and small–and even private citizens–to develop and send into space their own small self-contained space "payloads."

Three small units would have their own power supply and data recording systems, if required. They would remain in the payload bay of the Space Shuttle orbiter for the duration of the flight, which will be variable for a given flight.

The cost of this unique service will depend upon the size and weight of the experiment. Getaway Specials of 90 kilograms (200 pounds), 0.14 cubic meters (5 cubic feet) may be flown at a cost of $10,000, or 45 kilograms (100 pounds), 0.14 cubic meters (2.5 cubic feet) may be flown at a cost of $5,000, or 27 kilograms (60 pounds), 0.17 cubic meters (2.5 cubic feet) may be flown at a cost of $3,000. If additional services of the spacecraft or its flight crew are required (flipping switches, connecting to a NASA-provided battery pack, etc.), the price will be negotiated for each package.

The "getaway special" program offers private individuals and companies the opportunity to conduct space research in a manner previously available only to the government and very large corporations. Such small packages can be used as test beds for larger, follow-on experiments which would be flown on the Spacelab. In addition, the "getaway special" offers an opportunity to help enhance science and engineering education and to help set the course for the beneficial uses of space for years to come.

For detailed information on this unusual program, please write Public Affairs Office, NASA Marshall Space Flight Center, Alabama, 35812, or Director, Space Transportation System Operations, Code MO, NASA Headquarters, Washington, D.C., 20546.

SOLID-ROCKET BOOSTERS

The two solid-rocket boosters (SRB's) provide the main thrust to lift the Space Shuttle off the pad and up to an altitude of about 45,720 meters (150,000 feet), 24 nautical miles (28 statute miles). In addition, the two SRB's carry the entire weight of the external tank and orbiter and transmit the weight load through their structure to the mobile launch platform. Each booster has a thrust (sea level) of 12,899,200 Newtons (2.9 million pounds) at launch. They are ignited after the three Shuttle main engine thrust level is verified. The two SRB's provide 71.4 percent of the thrust at lift-off and during first stage ascent. After SRB separation, 75 seconds later, SRB apogee occurs at an altitude of 67,056 meters (220,000 feet), 35 nautical miles (41 statute miles). SRB impact occurs in the ocean around 122 nautical miles (141 statute miles) downrange.

The SRB's are the largest solid-propellant motors ever flown and the first designed for reuse. Each is 45.46 meters (149.16 feet) long and 3.70 meters (12.16 feet) in diameter. At launch, each weighs 586,506 kilograms (1,293,004 pounds), of which 85 percent, 503,627 kilograms (1,110,290 pounds) is propellant. The boosters are designed to be used at least 20 times.

Primary elements of each booster are the motor (including case, propellant, igniter, and nozzle), structure, separation systems, operational flight instrumentation, recovery avionics, pyrotechnics, deceleration system, thrust vector control system and range safety destruct system.

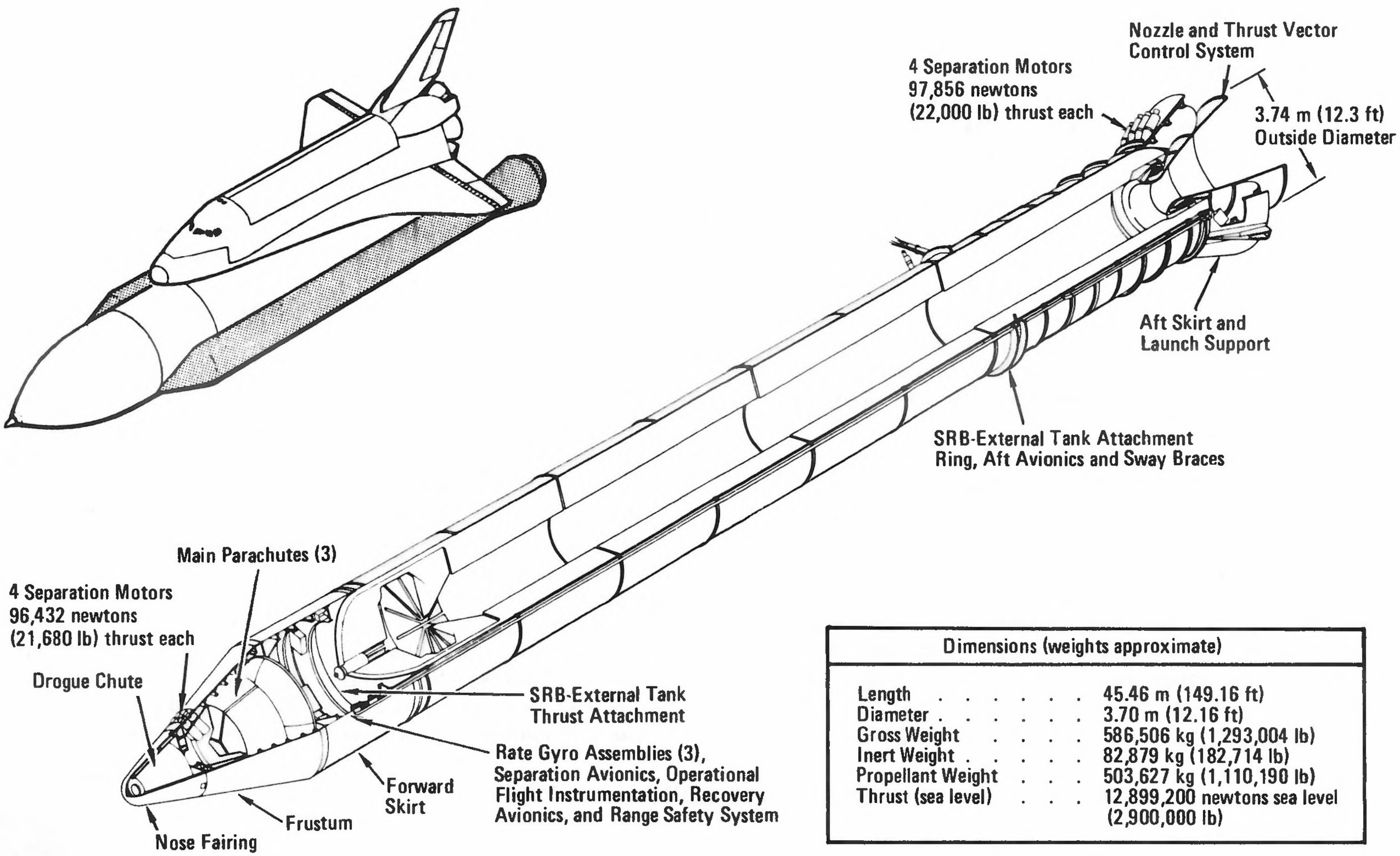
Each booster is attached to the external tank at the SRB's aft frame by two lateral sway braces and a diagonal attachment. The forward end of each SRB is attached to the external tank at the forward end of the SRB forward skirt. On the launch pad, each booster also is attached to the mobile launch platform at the aft skirt by four bolts which are severed by small explosives at liftoff.

The propellant mixture in each SRB motor consists of an ammonium perchlorate (oxidizer, 69.93 percent by weight), aluminum (fuel, 16 percent), iron oxide (a catalyst, 0.07 percent), a polymer (a binder that holds the mixture together 12.04 percent), and an epoxy curing agent (1.96 percent). The propellant is an 11-point star-shaped perforation in the forward motor segment and a double-truncated-cone perforation in each of the aft segments and aft closure. This configuration provides high thrust at ignition, then reduces the thrust by approximately a third 50 seconds after liftoff to prevent overstressing of the vehicle during maximum dynamic pressure (max q).

The SRB's are interchangeable. They are used as matched pairs and each is made up of four solid rocket motor segments. The pairs are matched by loading each of the four motor segments in pairs from the same batches of propellant ingredients to minimize any thrust imbalance. The segmented casing design give maximum flexibility in fabrication and ease of transportation and handling. Each segment is shipped to the launch site on a heavy-duty rail car with a specially built cover.

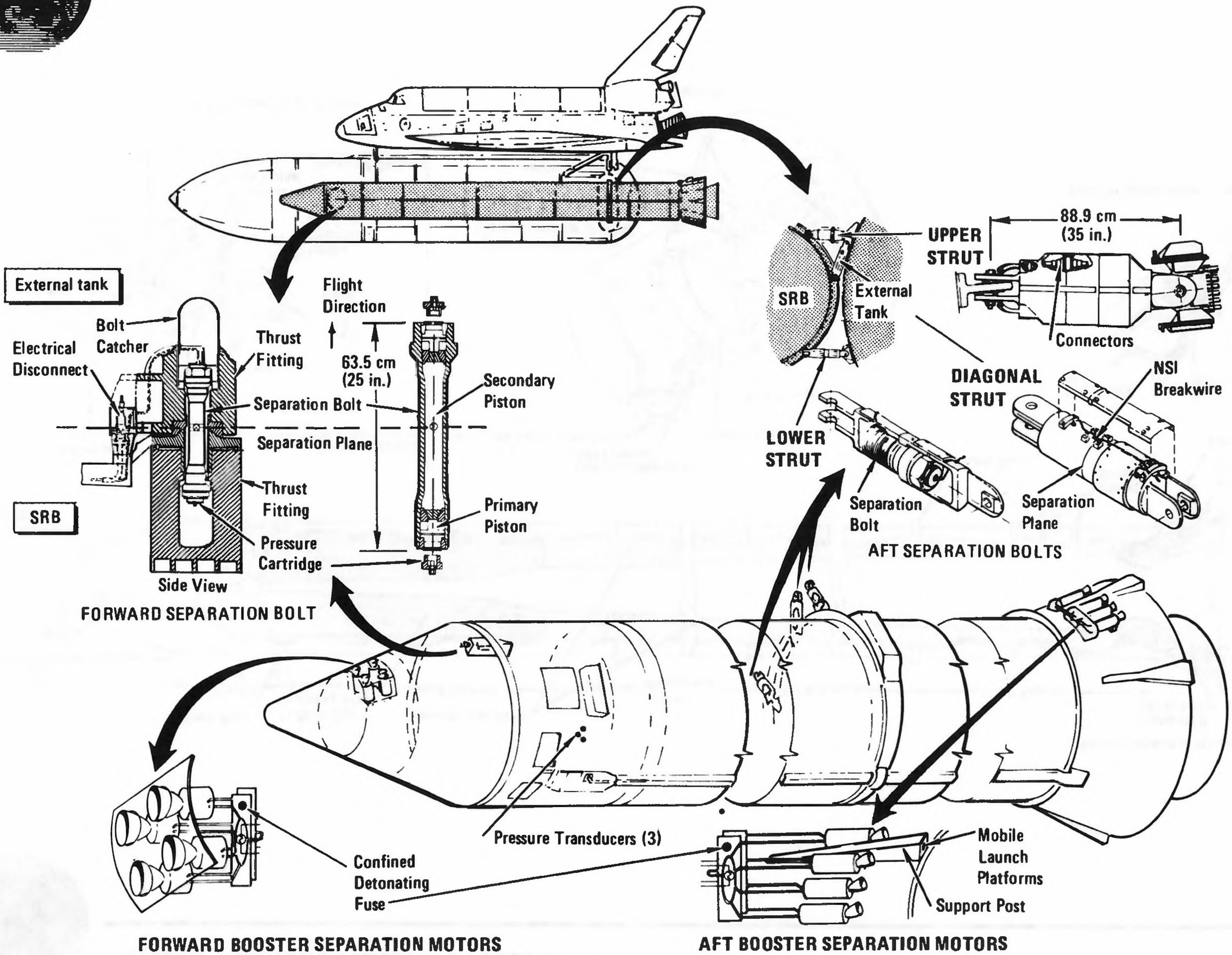
The nozzle expansion ratio of each booster is 7.16:1. The nozzle is gimbaled for thrust vector (direction) control. Each SRB has its own redundant auxiliary power units and hydraulic pumps. The all-axis gimbaling capability is 8 degrees. Each nozzle has a carbon cloth liner which erodes and chars during firing. The nozzle is a convergent-divergent, movable design in which an aft pivot-point flexible bearing is the gimbal mechanism.

The cone-shaped aft skirt reacts the aft loads between the SRB and the mobile launch platform. The four aft separation motors are mounted on the skirt. The aft section contains avionics, thrust vector control system which consists of two auxiliary power units and hydraulic pumps, hydraulic systems, and nozzle extension jettison system.

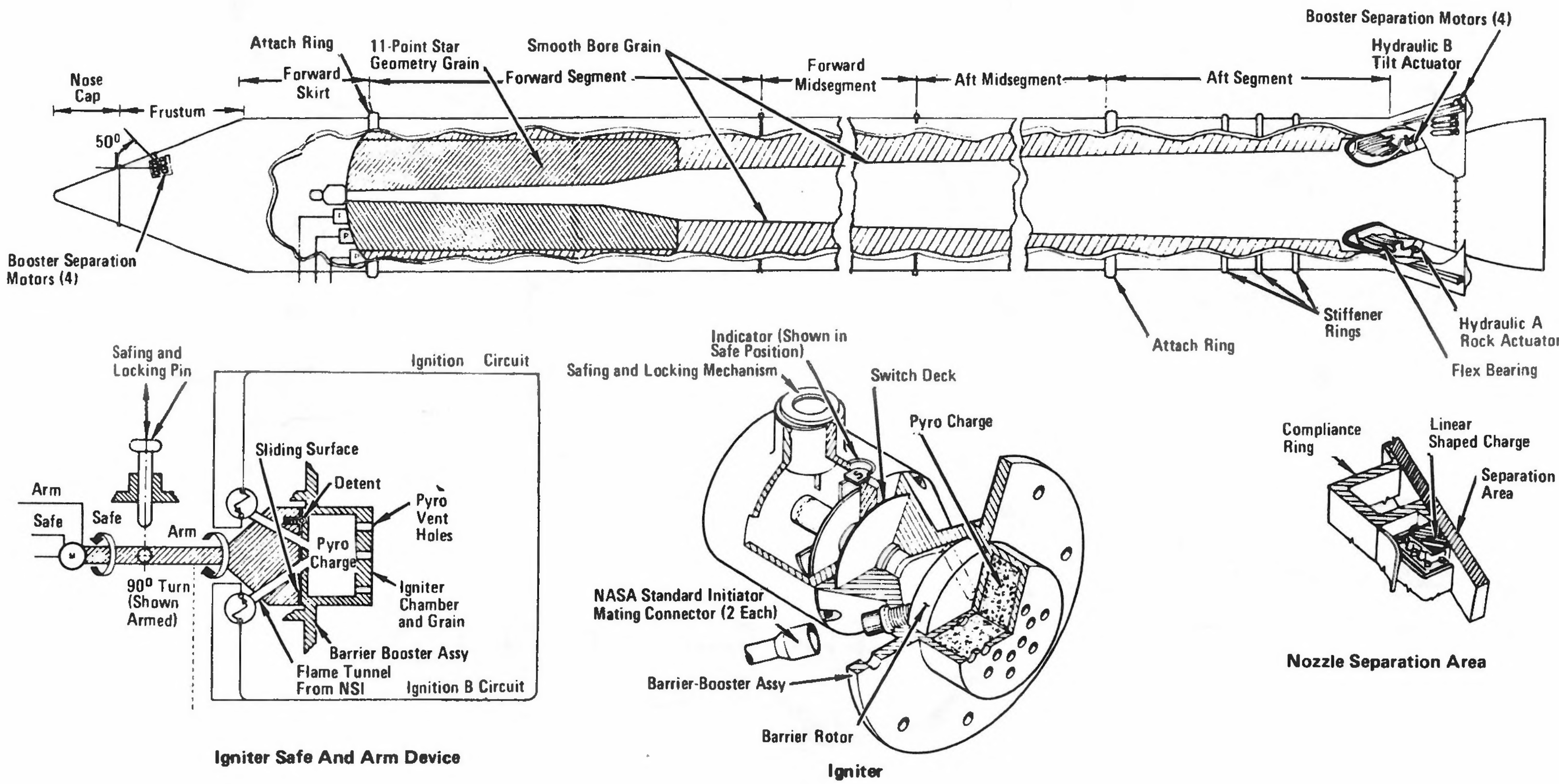


Dimensions (weights approximate)	
Length	45.46 m (149.16 ft)
Diameter	3.70 m (12.16 ft)
Gross Weight	586,506 kg (1,293,004 lb)
Inert Weight	82,879 kg (182,714 lb)
Propellant Weight . . .	503,627 kg (1,110,190 lb)
Thrust (sea level) . . .	12,899,200 newtons sea level (2,900,000 lb)

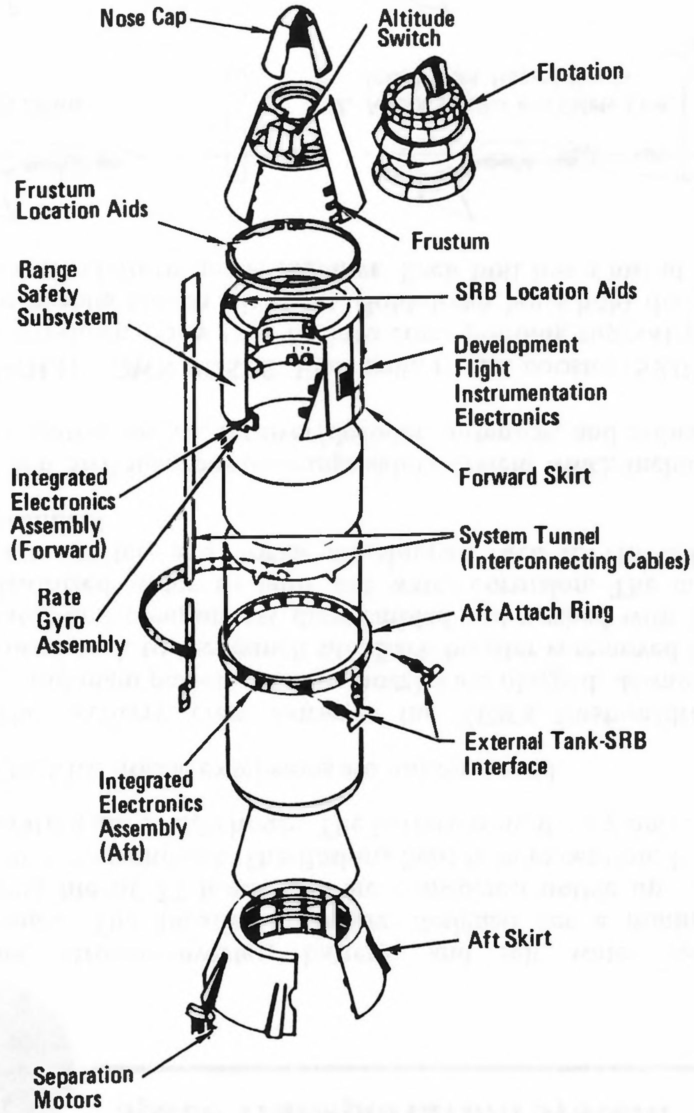
Solid Rocket Booster



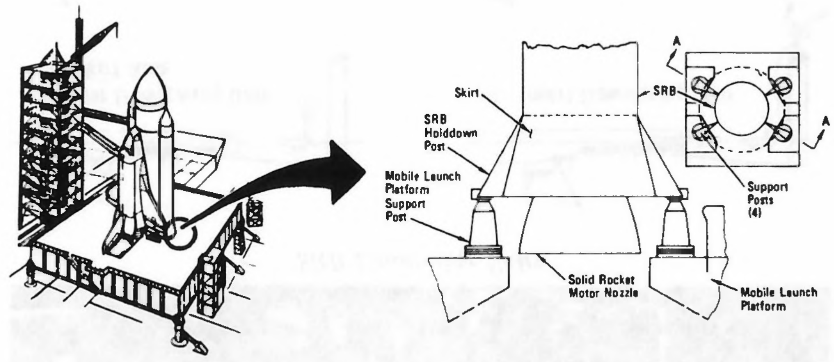
Separation System Elements



Motor Segments, Igniter, and Nozzle Severance



SRB – Exploded View



SRB Holddown Configuration

The forward section of each booster contains avionics, sequencer, forward separation motors, nose cone separation system, drogue and main parachutes, recovery beacon, recovery light, and a range safety system.

Each SRB has two integrated electronic assemblies, one forward and one aft. The forward assembly jettisons the nozzle after burnout, initiates release of the nose cap and frustum, detaches the parachutes, and turns on recovery aids. The aft assembly, mounted in the external tank/SRB attach ring, connects with the forward assembly and the orbiter avionics systems for SRB ignition commands and nozzle thrust vector control. Each integrated electronic assembly has a multiplexer/demultiplexer which sends or receives more than one message, signal, or unit of information on a single communication channel.

Eight booster separation motors (four in the nose frustum and four in the aft skirt) of each SRB thrust for 0.68 second at SRB separation from the external tank. Each solid-rocket separation motor is 78 centimeters (31 inches) long and 31 centimeters (12.5 inches) in diameter.

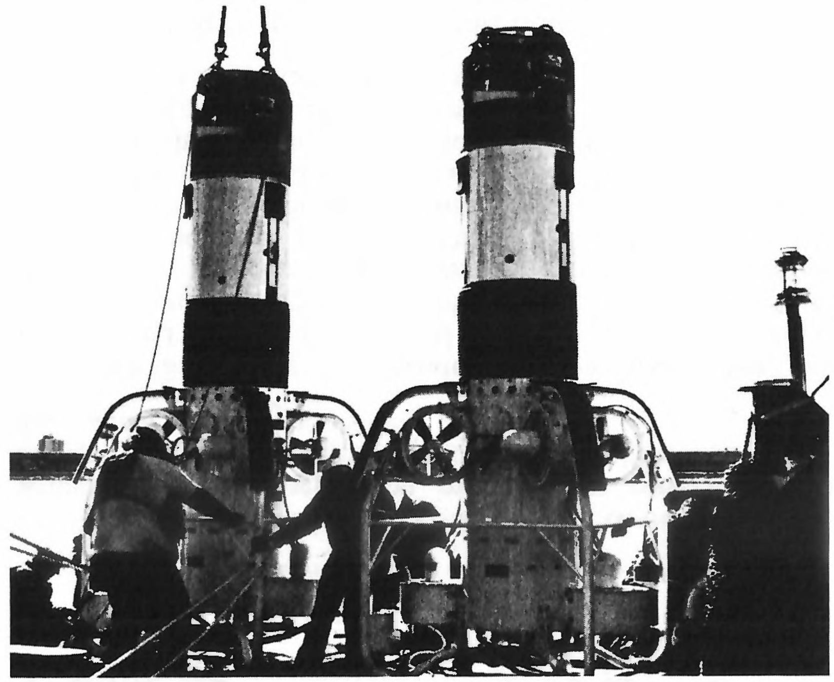
Location aids are provided for each SRB, frustum/drogue chute, and main parachutes. These include a transmitter, antenna, strobe/converter, battery, and salt water switch electronics. The location aids are designed for a minimum operating life of 72 hours and are considered usable up to 20 times by refurbishment. The flashing light is an exception. It has an operating life of 280 hours. The battery is used only once.

The SRB nozzle extensions are not recovered.

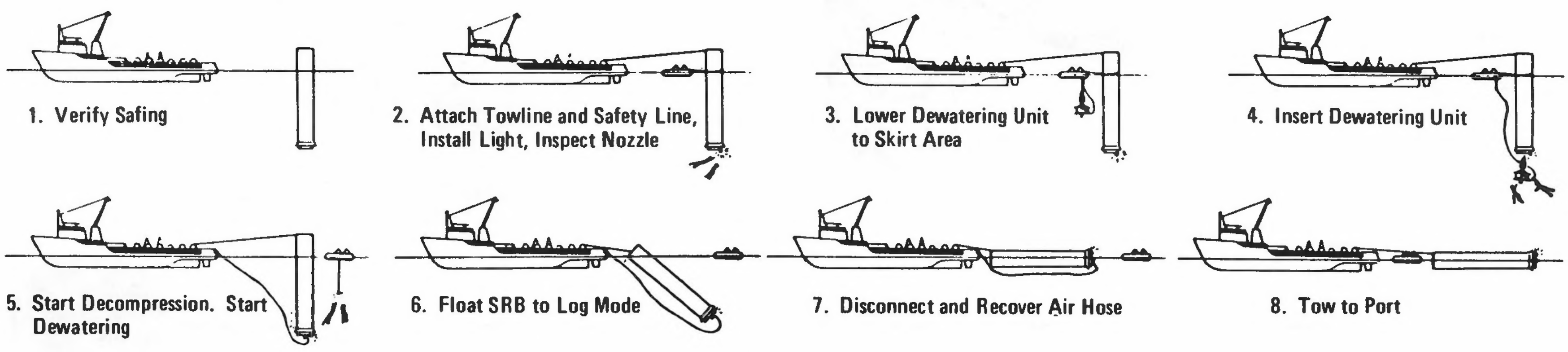
The recovery crew retrieves the SRB's, frustum/drogue chutes, and main parachutes. The nozzles are plugged, dewatered, and towed back to the launch site. Each booster is removed from the water and components disassembled and washed with fresh and deionized water to limit salt water corrosion. The motor segments, igniter, and nozzle are shipped back to Thiokol for refurbishment.

Each SRB incorporates range safety system which includes a battery power source, receiver/decoder, antennas, and ordnance.

HOLD DOWN POSTS. Each solid rocket booster (SRB) has four hold-down posts that fit into corresponding support posts on the mobile launch platform. Hold-down bolts hold the SRB and launch platform posts together. Each bolt has a nut at each

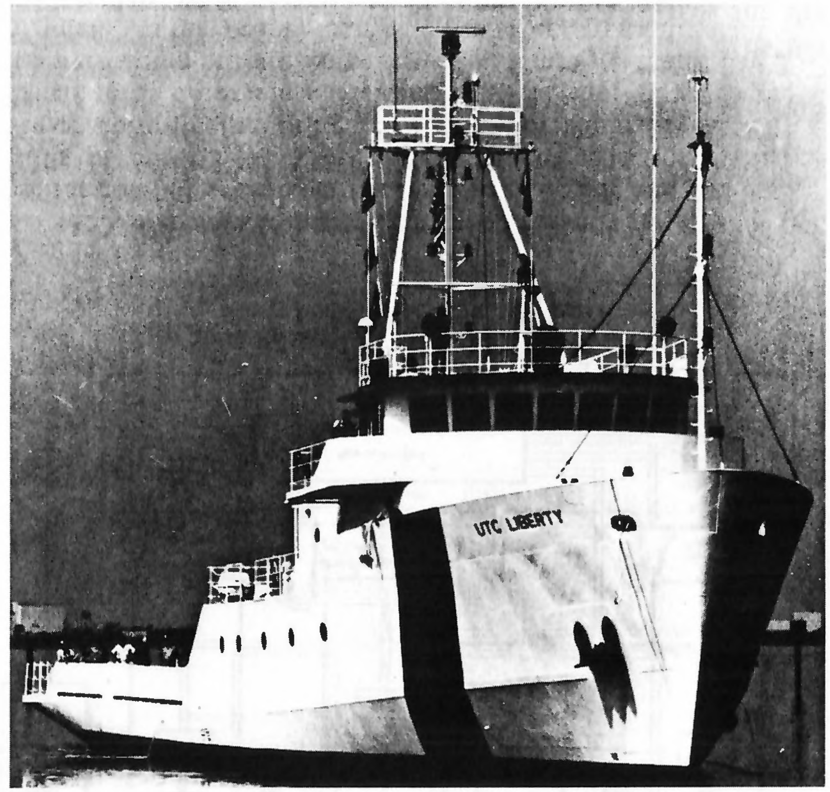


SRB Dewatering Units



SRB Recovery Procedure

UTC Freedom

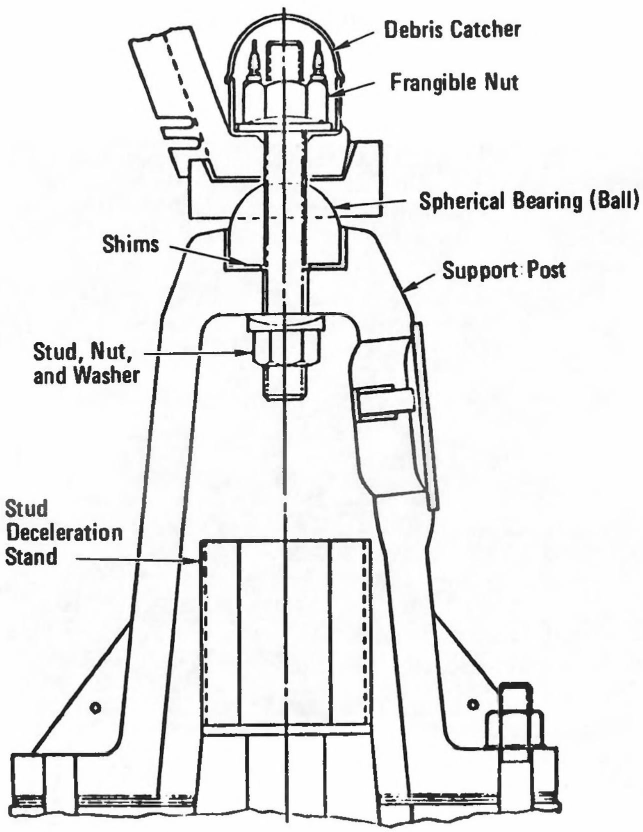


UTC Liberty

end, but only the top nut is frangible. It contains two NASA standard initiators (NSI's), which are ignited at solid rocket motor ignition commands.

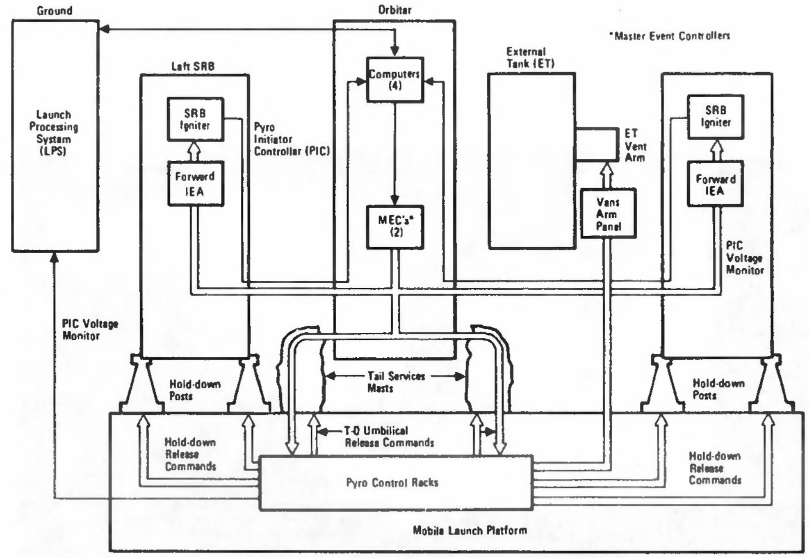
When the two NIS's are ignited at each hold down, the hold-down bolt travels downward because of the release of tension in the bolt (pre-tensioned priot to launch), NSI gas pressure, and gravity. The bolt is stopped by the stud deceleration stand, which contains sand. The SRB bolt is 71 centimeters (28 inches) long and is 8.89 centimeters (3.5 inches) in diameter.

The solid rocket motor ignition commands are issued by the orbiter's computers through the master event controllers (MEC's) to the hold-down pyrotechnic initiator controllers (PIC's) on the mobile launch platform. They provide the ignition to the hold-down NSI's. The launch processing system (LPS) monitors the SRB hold-down PIC's for low voltage during the last 12 seconds before launch. PIC low voltage will initiate a launch hold.



SRB Support/Hold-Down Post

SRB IGNITION. SRB ignition can occur only when a lock pin from each SRB safe and arm device has been removed. The ground crew removes the pin during pre-launch activities. The solid rocket motor ignition commands are issued when the three Space Shuttle main engines (SSME's) are at or above 90 percent rated thrust, no SSME fail and/or SRB ignition PIC low voltage is indicated, and there are no holds from the LPS.



Solid Rocket Ignition, Hold-Down Release, and Umbilical Retract Commands

The solid rocket motor ignition commands are sent by the orbiter computers through the MEC's to the safe and arm device NSI's in each SRB. A PIC single-channel capacitor discharge device controls the firing of each pyrotechnic device. Three signals must be present simultaneously for the PIC to generate the pyro firing output. These signals – Arm, Fire 1, and Fire 2 – originate in the orbiter computers and are transmitted to the MEC's. The MEC's reformats them to 28 Vdc signals for the PIC's. The "arm" signal charges the PIC capacitor to 40 Vdc (minimum of 20 Vdc).

The computer launch sequence also controls certain critical main propulsion system valves and monitors the engine-ready indications from the main engines. The MPS start commands are issued by the onboard computers at T minus 3.8 seconds (staggered start–engine three, engine two, engine one–all approximately within one-fourth of a second) and the sequence monitors the thrust buildup of each engine. All three engines must reach the required 90 percent thrust within this time period or an orderly shutdown is commanded up to 4.6 seconds and safing functions are initiated.

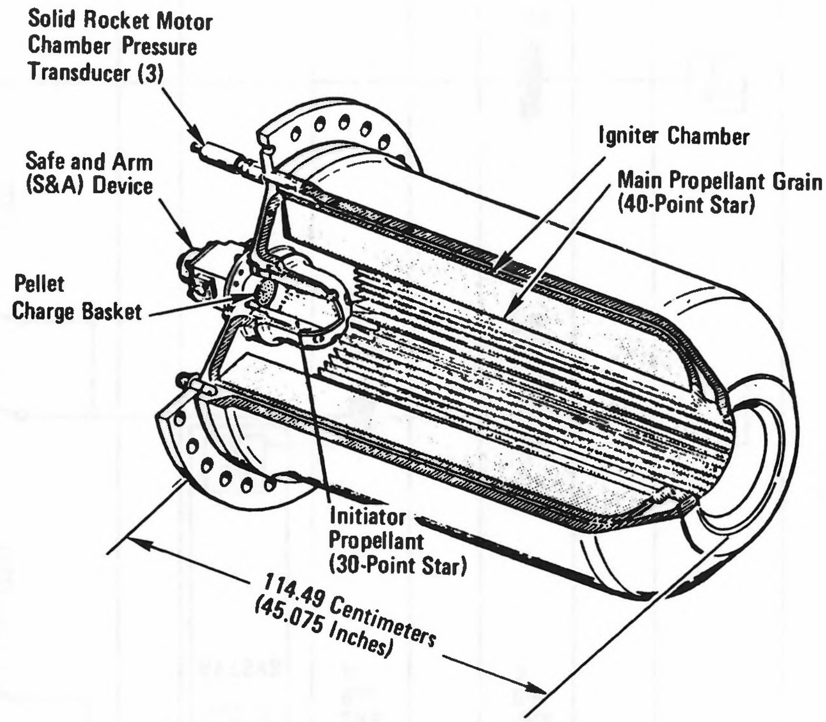
Normal thrust build-up to the required 90-percent thrust level will result in the engines being commanded to the liftoff position at T-0, start of the onboard timer, termination of LPS porting, issue a Fire 1 command to arm the SRB's, and start a 2.64 second timer which allows the vehicle base bending load modes to initialize (movement of approximately 76 centimeters [30 inches] measured at the tip of the external tank–with movement towards the external tank).

When the 2.64 second timer has timed out, the Fire 2 command is issued which causes SRB ignition, T-0 umbilical release, SRB hold-down release commands, reset of the onboard master timing unit to T-0, start of the onboard event timer, reset of the mission event timer, and modes the computers.

The "arm" signal causes a barrier rotor to move into a position from which redundant NSI's fire through a thin barrier seal down a flame tunnel. This ignites a pyro booster charge, which is retained in the safe the arm device behind a perforated plate. The booster charge ignites the propellant in the igniter initiator, and combustion products of this propellant ignite the solid rocket motor initiator, which fires down the length of the solid rocket motor igniting the solid rocket motor propellant.

The solid rocket motor thrust profile is tailored to reduce thrust during the maximum dynamic pressure (max q) region.

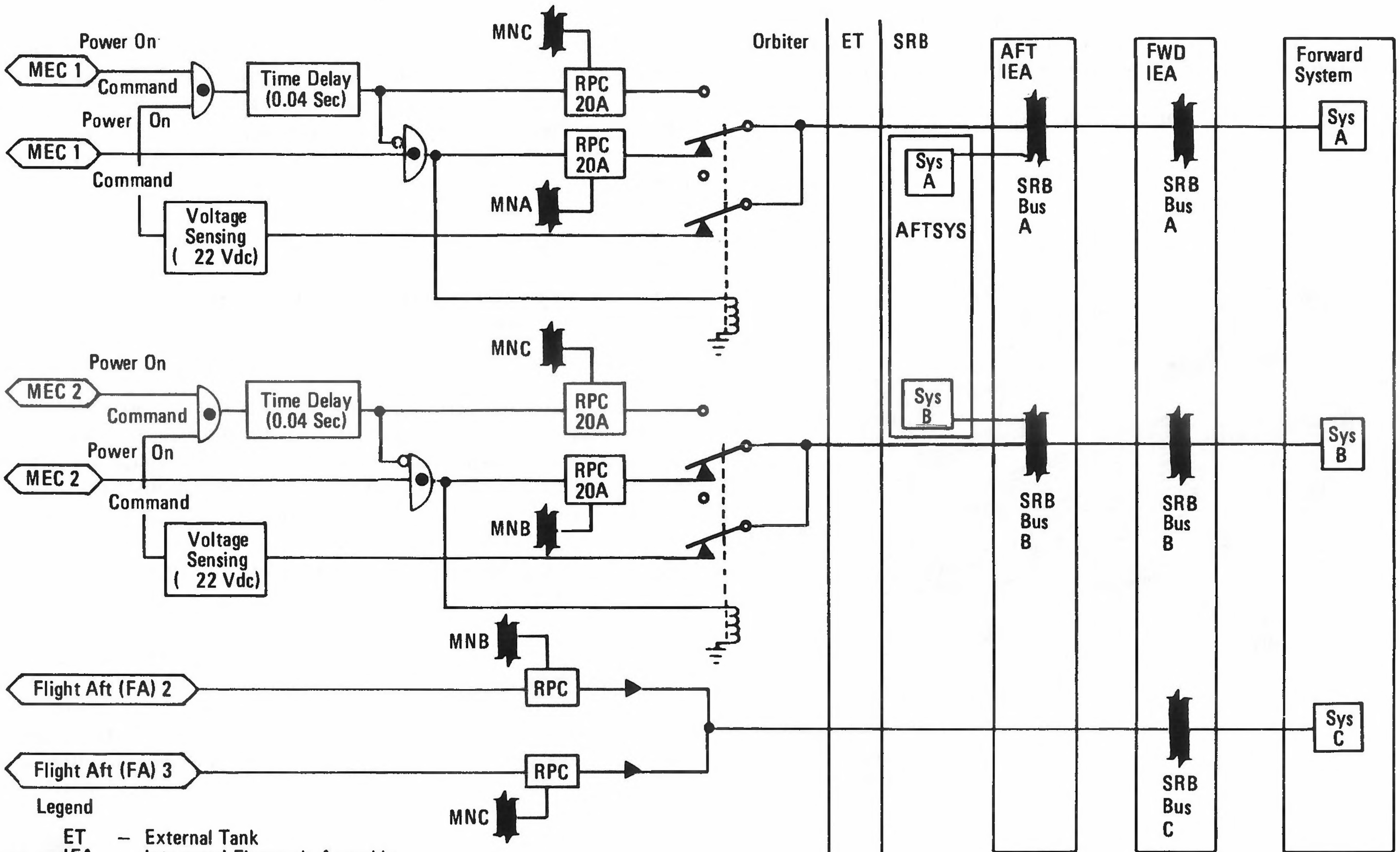
ELECTRICAL POWER DISTRIBUTION. Electrical power distribution in each SRB consists of orbiter-supplied main dc bus



Solid Rocket Motor Igniter

power to each SRB via SRB buses A, B, and C. The orbiter main dc buses A, B, and C supply main dc bus power to the respsective SRB buses A, B, and C. In addition, orbiter main dc bus C supplies backup power to SRB buses A and B, and orbiter bus B supplies backup power to SRB bus C. This electrical power distribution arrangement allows all SRB buses to remain powered in the event one orbiter main bus fails.

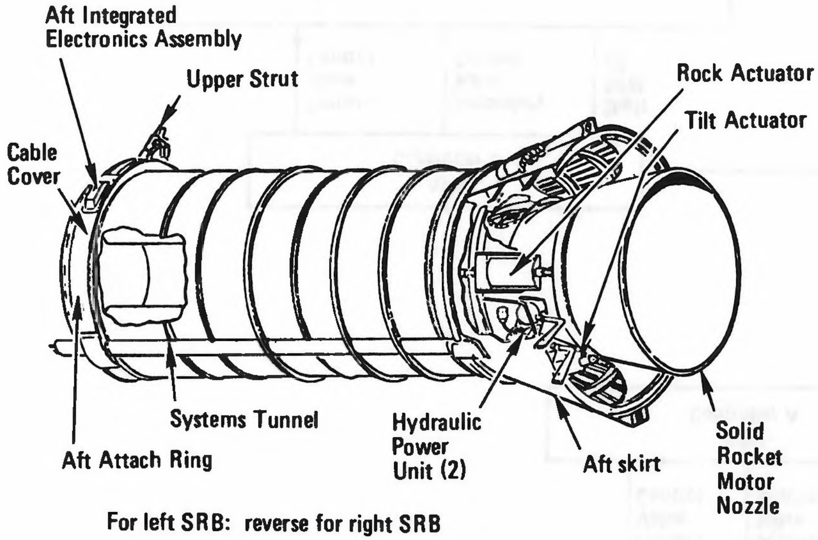
The nominal dc voltage is 28 Vdc with an upper limit of 32 Vdc and lower limit of 24 Vdc.



Legend

ET – External Tank
IEA – Integrated Electronic Assembly
MEC – Master Event Controller
RPC – Remote Power Controller
SRB – Solid Rocket Booster

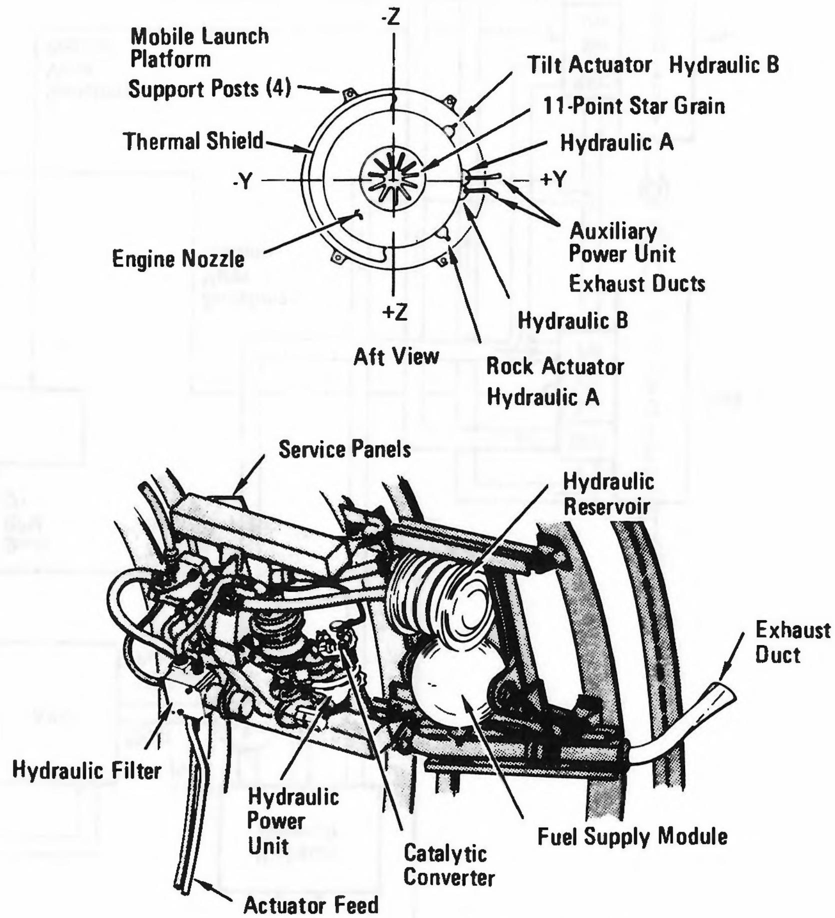
SRB Electrical Power Distribution



SRB Thrust Vector Control Component Location

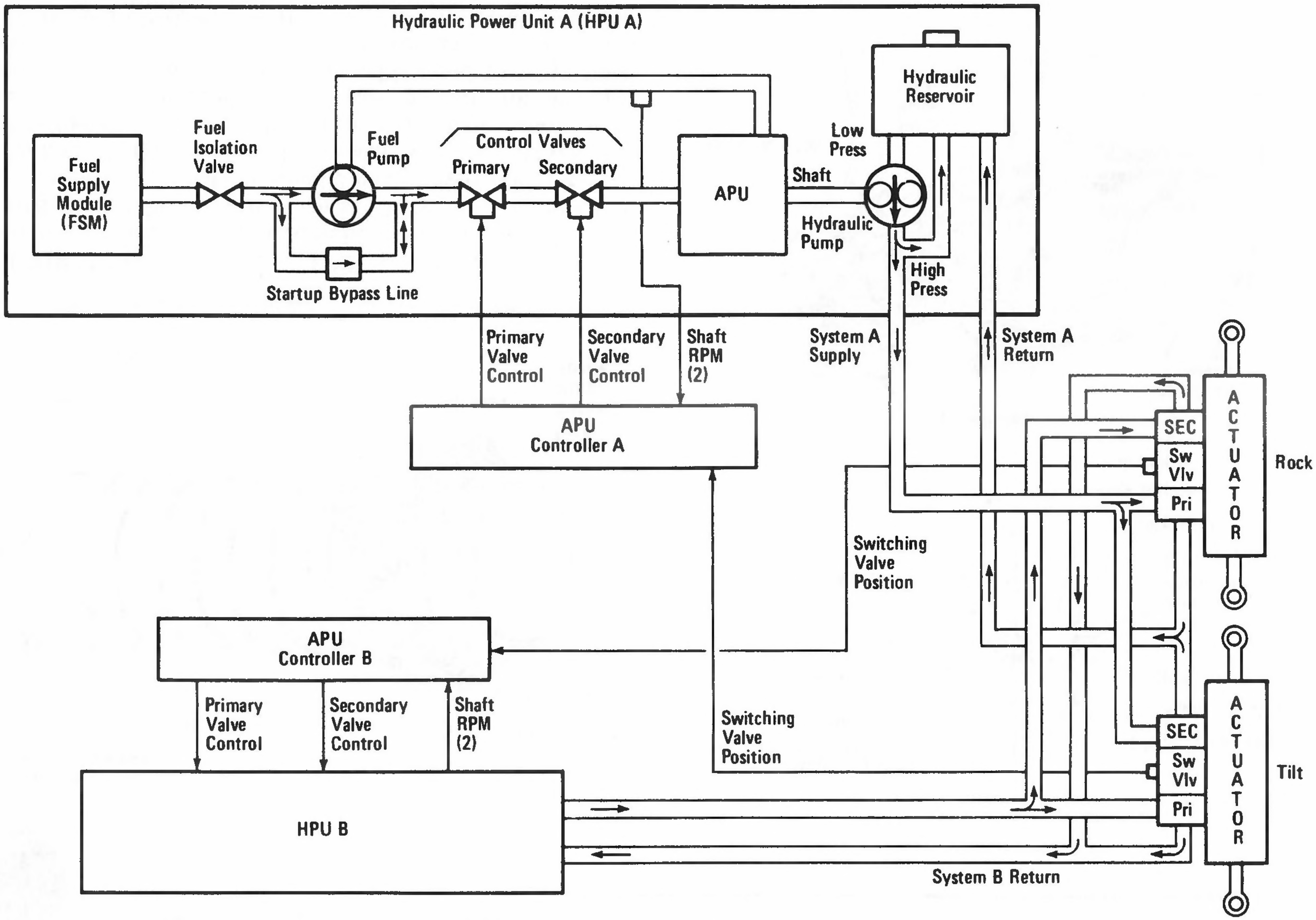
HYDRAULIC POWER UNITS. There are two self-contained, independent, hydraulic power units (HPU) on each SRB. Each HPU consists of an auxiliary power unit (APU), fuel supply module (FSM), hydraulic pump, hydraulic reservoir, and hydraulic fluid manifold assembly. The APU's are hydrazine-fueled and generate mechanical shaft power to a hydraulic pump that produces hydraulic pressure for the SRB hydraulic system. The two separate HPU's and two hydraulic systems are located on the aft end of each SRB, between the SRB nozzle and aft skirt. The HPU components are mounted on the aft skirt between the rock and tilt actuators. The two systems operate from just before launch until SRB jettison. The two independent hydraulic systems are connected to the rock and tilt servoactuators.

The APU controller electronics are located in the SRB aft integrated electronic assemblies (IEA's) on the aft external tank attach rings.



SRB Thrust Vector Control Component Location

The APU's and their fuel systems are isolated from each other. Each fuel supply module (tank) contains 9.9 kilograms (22 pounds) of hydrazine. The fuel tank is pressurized with gaseous nitrogen at 20,700 millimeters of mercury (MMHg) (400 psi) which provides the force to expel (positive expulsion) the fuel from the tank to the fuel distribution line, maintaining a positive fuel supply to the APU throughout its operation.



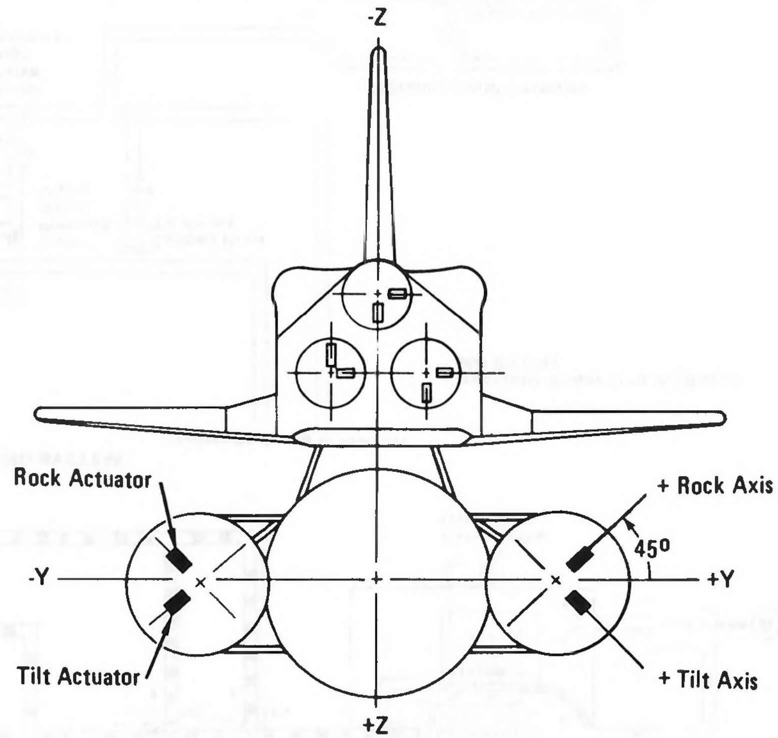
Hydraulic Power Unit System

The fuel isolation valve is opened at APU startup to allow fuel to flow to the APU fuel pump and control valves, then to the gas generator. The gas generator catalytic action decomposes the fuel and creates a hot gas and feeds the hot gas exhaust product to the APU two-stage gas turbine. Fuel flows primarily through the startup bypass line until the APU speed is such that the fuel pump outlet pressure is greater than the bypass line. Then all the fuel is supplied to the fuel pump.

The APU turbine assembly provides mechanical power to the APU gearbox. The gearbox drives the APU fuel pump, hydraulic pump, and lube oil pump. The APU lube oil pump provides oil lubrication of the gearbox. The turbine exhaust of each APU flows over the exterior of the gas generator, cooling it, and is then directed overboard through an exhaust duct.

When the APU speed reaches 100-percent, the APU primary control valve closes, and the APU speed is controlled by the APU controller electronics. If the primary control valve logic fails to the "open" state, the secondary control valve assumes control of the APU at 112-percent speed.

Each HPU is connected to both servoactuators on that SRB. One HPU serves as the primary hydraulic source for the servoactuator and the other HPU serves as the secondary hydraulics for the servoactuator. Each servoactuator has a switching valve that allows the secondary hydraulics to power the actuator if the primary hydraulic pressure drops below 106,087 mmHg (2050 psi). A switch contact on the switching valve will close when the switching valve is in the secondary position. When the valve is closed, a signal is sent to the APU controller that inhibits the 100-percent APU speed control logic and enables the 100-percent APU speed control logic. The 100-percent APU speed enables one APU/HPU to supply sufficient operating hydraulic pressure to both servoactuators of that SRB.

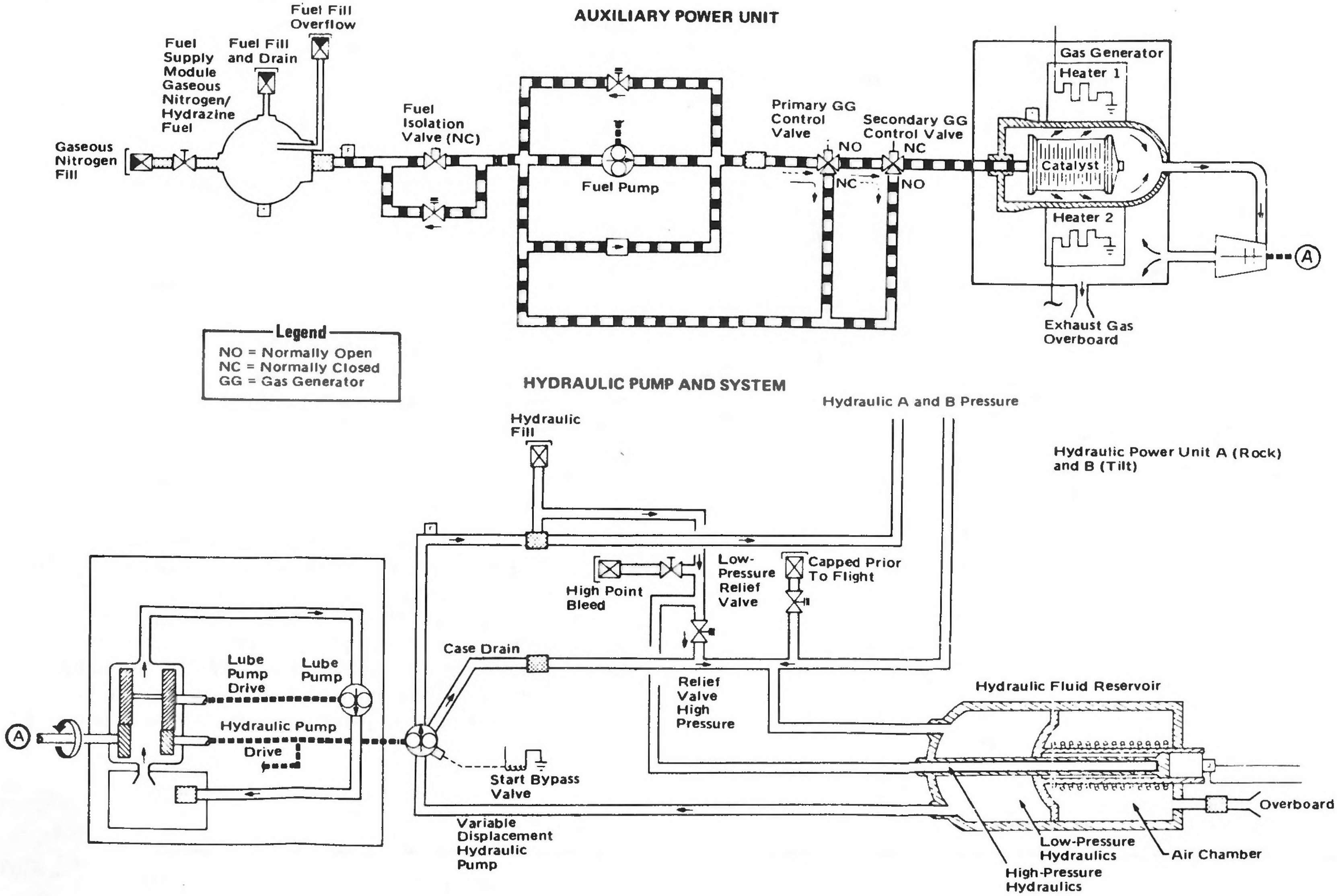


SRB Actuator Orientation

The APU 100-percent speed corresponds to 72,000 rpm of the APU, 110-percent to 72,200 rpm, and 112-percent to 80,640 rpm.

The hydraulic pump speed is 3600 rpm and supplies hydraulic pressure of 157,837 plus or minus 2587 mmHg (3050 plus or minus 50 psi). A high-pressure relief valve provides overpressure protection to the hydraulic system and relieves at 188,887 mmHg (3650 psi).

The APU's/HPU's and hydraulic systems are reusable.



Auxiliary Power Unit and Hydraulic Pump System

THRUST VECTOR CONTROL. Each SRB has two hydraulic gimbal servoactuators: one for rock and one for tilt. The servoactuators provide the force and control to gimbal the nozzle for thrust vector control.

The Space Shuttle ascent thrust vector control (ATVC) portion of the flight control system directs the thrust of the three Shuttle main engines and the two SRB nozzles to control Shuttle attitude and trajectory during liftoff and ascent. Commands from the guidance system are transmitted to the ATVC drivers, which transmit signals proportional to the commands to each servoactuator of the main engines and SRB's. Four independent flight control system channels and four ATVC channels control six main engine and four SRB ATVC drivers, with each driver controlling one hydraulic port on each main and SRB servoactuator.

Each SRB servoactuator consists of four independent, two-stage servovalves that receive signals from the drivers. Each servovalve controls one power spool in each actuator, which positions an actuator ram and the nozzle to control the direction of thrust.

The four servovalves in each actuator provide a force-summed majority voting arrangement to position the power spool. With four identical commands to the four servovalves, the actuator force sum action prevents a single erroneous command from affecting power ram motion. If the erroneous command persists for more than a predetermined time, differential pressure sensing activates a selector valve to isolate and remove the defective servovalve hydraulic pressure, permitting the remaining channels and servovalves to control the actuator ram spool.

Failure monitors are provided for each channel to indicate which channel has been bypassed. An isolation valve on each channel provides the capability of resetting a failed or bypassed channel.

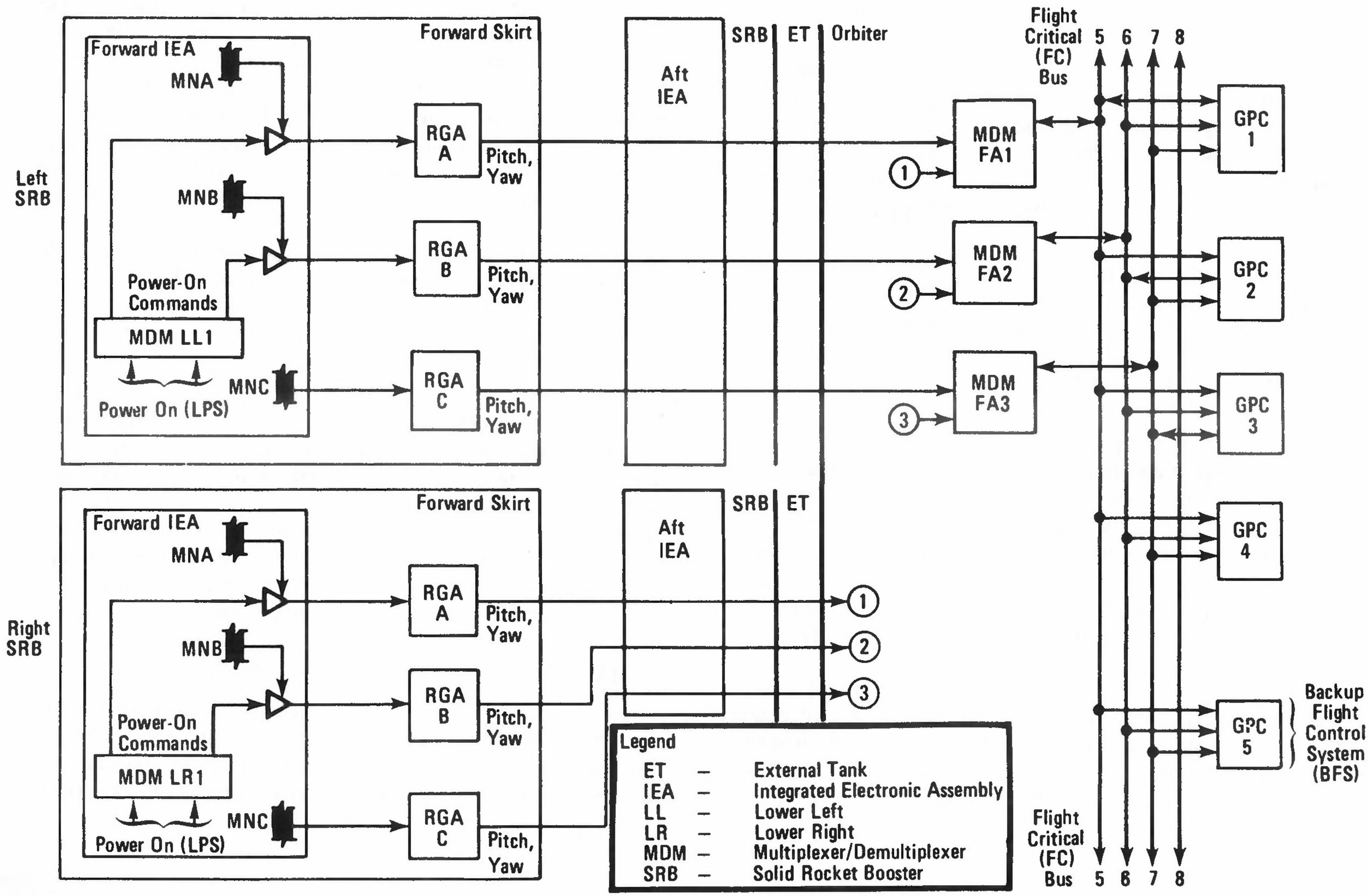
Each actuator ram is equipped with transducers for position feedback to the thrust vector control system. Within each servoactuator ram is a splashdown load relief assembly to cushion the nozzle at water splashdown and prevent damage to the nozzle flexible bearing.

SRB RATE GYRO ASSEMBLIES. Each SRB contains three rate gyro assemblies (RGA's), with each RGA containing one pitch and one yaw gyro. These provide an output proportional to angular rates about the pitch and yaw axis to the orbiter computers and guidance, navigation, and control system during first-stage ascent flight in conjunction with the orbiter roll rate gyros until SRB separation. At SRB separation, a switchover is made from the SRB RGA's to the orbiter RGA's.

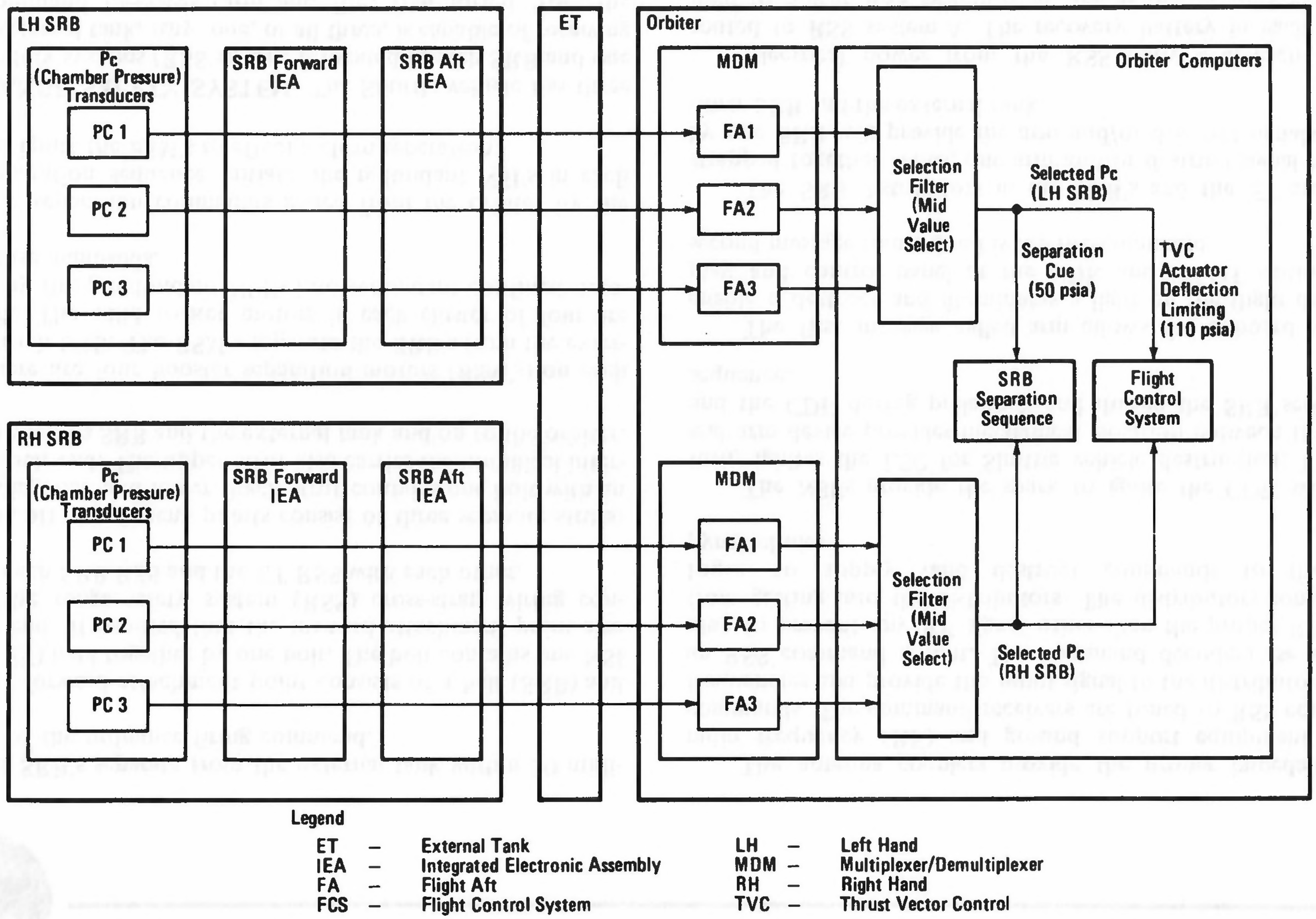
The SRB RGA rates pass through the flight aft multiplexer/demultiplexers (MDM's) to the orbiter computers. The RGA rates are then mid-value selected in redundancy management (RM) to provide one rate from each SRB pitch and yaw rate gyro (left and right SRB) to the user software.

SRB SEPARATION. SRB separation is initiated when the three solid rocket motor chamber pressure transducers are processed in the redundancy management middle value select and the head-end chamber pressure of both SRB's is less than or equal to 2,587 mmHg (50 psi). A backup cue is the time-elapsed from booster ignition.

The separation sequence is initiated, commanding the thrust vector control actuators to the null position and putting the main propulsion system into a second-stage configuration (at 0.8 seconds from sequence initialization) that ensures the thrust of each SRB is less than 444,800 newtons (100,000 pounds). Orbiter yaw attitude is held for four seconds, while SRB thrust drops to less than 266,800 newtons (60,000 pounds).



SRB Rate Gyro Assembly (RGA)



Solid Rocket Motor Chamber Pressure Data Flow

The SRB's separate from the external tank within 30 milliseconds of the ordnance firing command.

The forward attachment point consists of a ball (SRB) and socket (ET) held together by one bolt. The bolt contains one NSI at each end. It is noted that the forward attachment point also carries the range safety system (RSS) cross-strap wiring connecting each SRB RSS and the ET RSS with each other.

The aft attachment points consist of three separate struts: upper, diagonal, and lower. Each strut contains one bolt with an NSI at each end. The upper strut also carries the umbilical interface between its SRB and the external tank and on to the orbiter.

There are four booster separation motors (BSM's) on each end of each SRB. The BSM's separate the SRB's from the external tank. The solid rocket motors in each cluster of four are ignited by firing redundant NSI's into redundant confined detonating fuse manifolds.

The separation commands issued from the orbiter by the SRB separation sequence initiate the redundant NSI's in each bolt and ignite the BSM's to effect a clean separation.

RANGE SAFETY SYSTEM. The Shuttle vehicle has three range safety systems (RSS's). One is located in each SRB and one in the external tank. Any one, or all three, is capable of receiving two command messages (arm and fire) transmitted from the ground station. The RSS is used only when the Shuttle vehicle violates a launch trajectory red line.

An RSS consists of two antenna couplers, command receivers and command decoders, a dual distributor, a safe and arm device with two NSI's, two confined detonator fuse (CDF) manifolds, and two linear shaped charges (LSC's).

The antenna couplers provide the proper impedance for radio frequency (RF) and ground support equipment (GSE) commands. The command receivers are tuned to RSS command frequencies and provide the input signal to the distributors when an RSS command is sent. The command decoders use a coded plug to prevent any RF signal other than the proper RF signal from getting into the distributors. The distributors contain the logic to supply valid destruct commands to the RSS pyrotechnics.

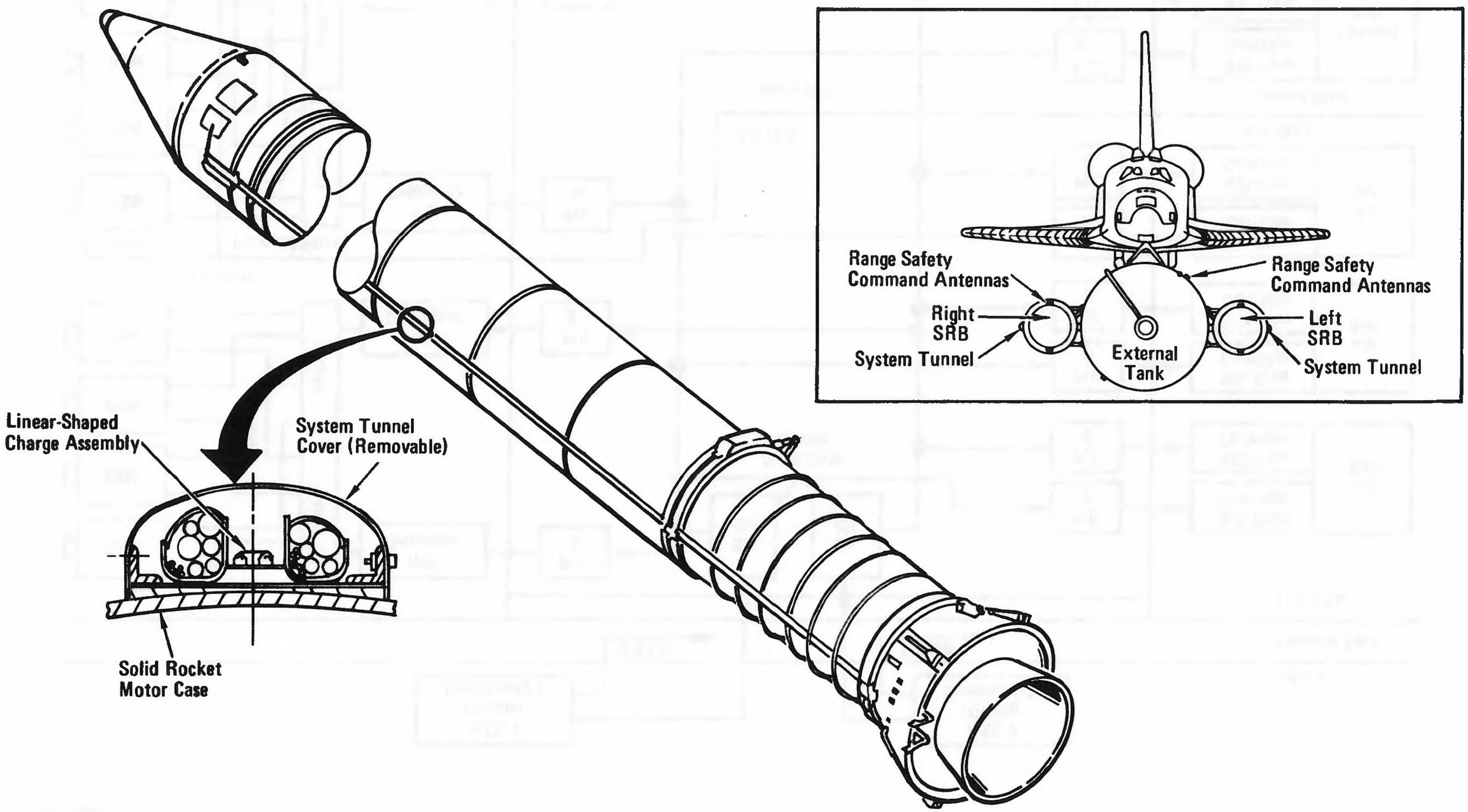
The NSI's provide the spark to ignite the CDF, which, in turn, ignites the LSC for Shuttle vehicle destruction. The safe and arm device provides mechanical isolation between the NSI's and the CDF during prelaunch and during the SRB separation sequence.

The first message called arm allows the onboard logic to enable a destruct and illuminates a light on the flight deck display and control panel at the CDR and PILOT station. The second message transmitted is the fire command.

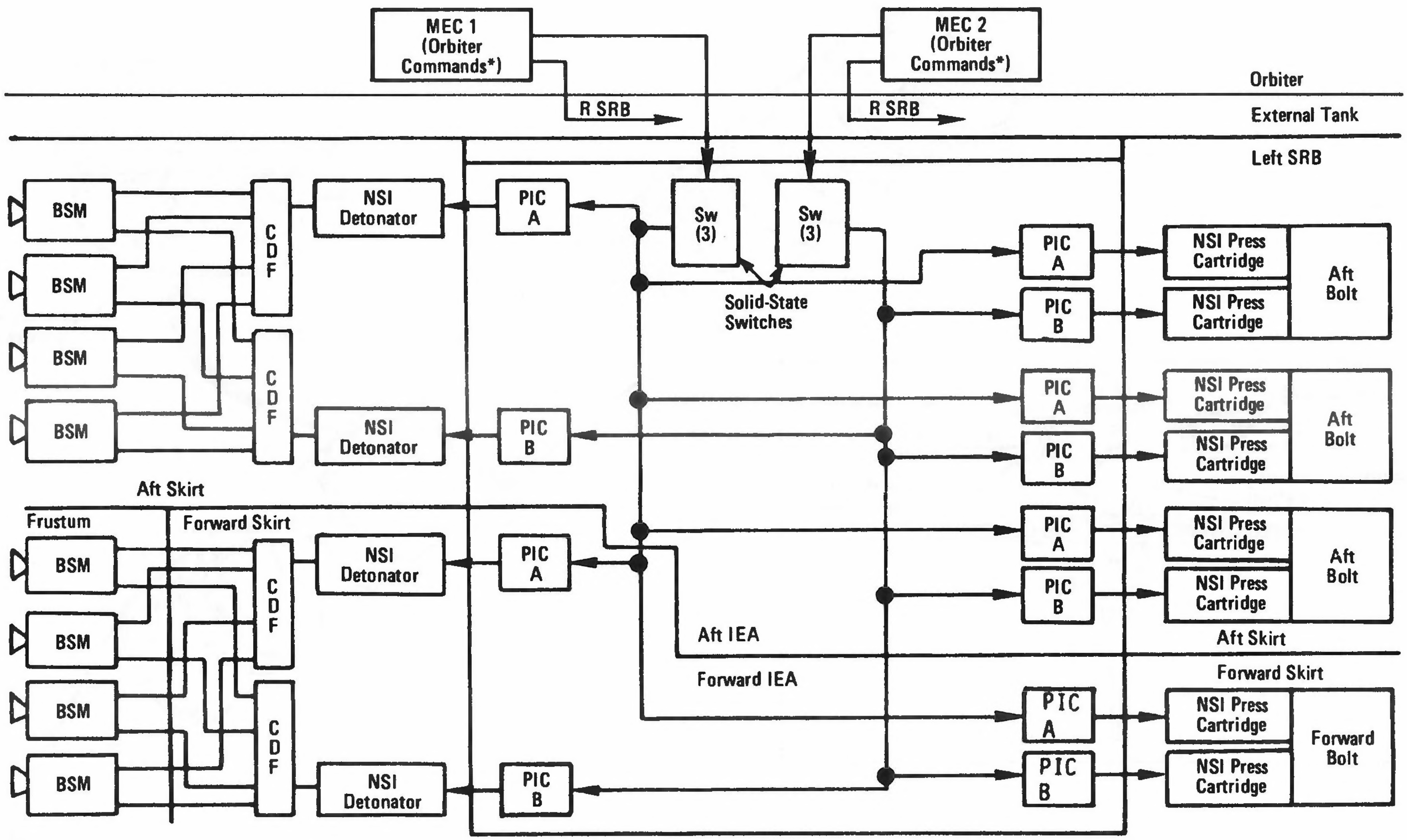
The SRB distributors in the SRB's and the ET are cross-strapped together. Thus, one arm and/or destruct signal received by one SRB will provide the arm and/or destruct signals to the other SRB and the external tank.

Electrical power from the RSS battery in each SRB is routed to RSS system A. The recovery battery in each SRB is used to power RSS system B, as well as the recovery system in that SRB. The SRB RSS is powered down during the separation sequence, and the recovery system for that SRB is powered up.

SRB DESCENT AND RECOVERY. At approximately 70 seconds after separation and SRB apogee, a linear-shaped charge severs the nozzle extension. During the descent portion of the trajectory, the SRB's achieve a nose-up trim condition.



SRB Range Safety System

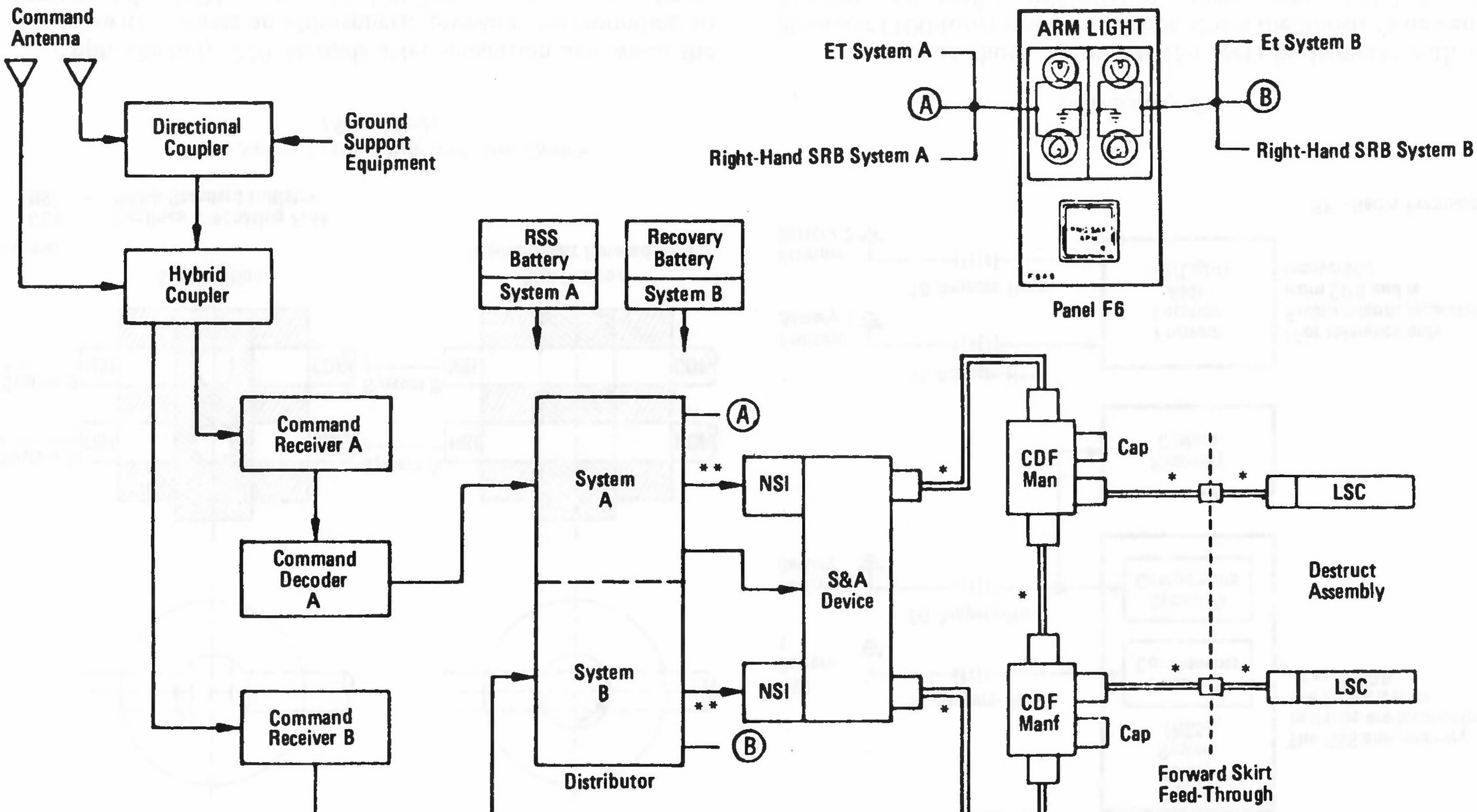


*Separation Commands: Arm, Fire 1, Fire 2

Legend

BSM	–	Booster Separation Motor	
CDF	–	Confined Detonator Fuse	
IEA	–	Integrated Electronic Assembly	
MEC	–	Master Event Controller	
NSI	–	NASA Standard Initiator	
R	–	Right	
SW	–	Switch	

SRB Separation System

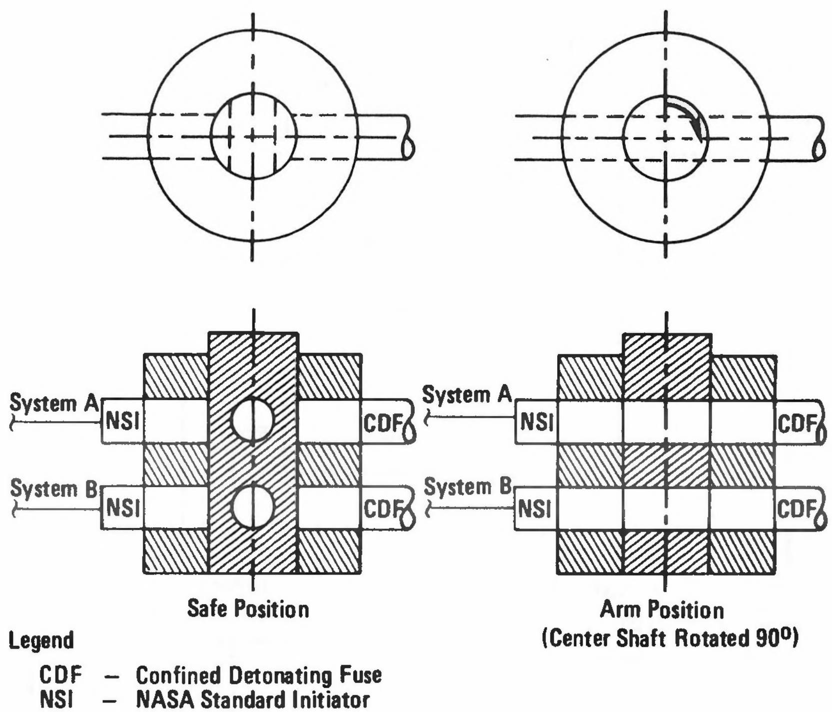


Legend

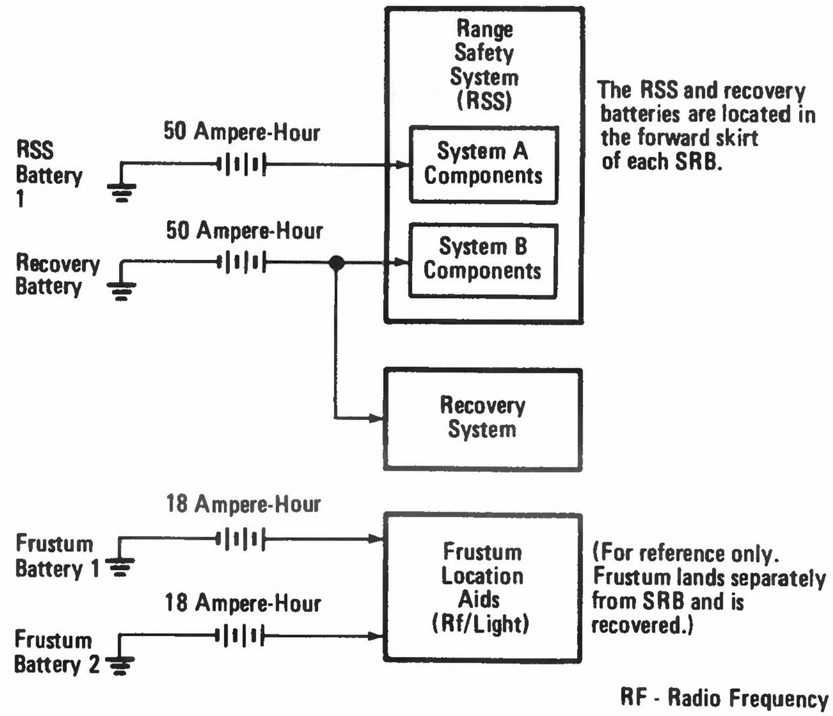
CDF – Continued Detonating Fuse
LSC – Linear Shaded Charge
NSI – NASA Standard Initiator
S&A – Safe and Arm

* Confined Detonating Fuse
** NSI With Adapter

SRB Range Safety System (RSS)



Range Safety System Safe and Arm Device (Simplified)

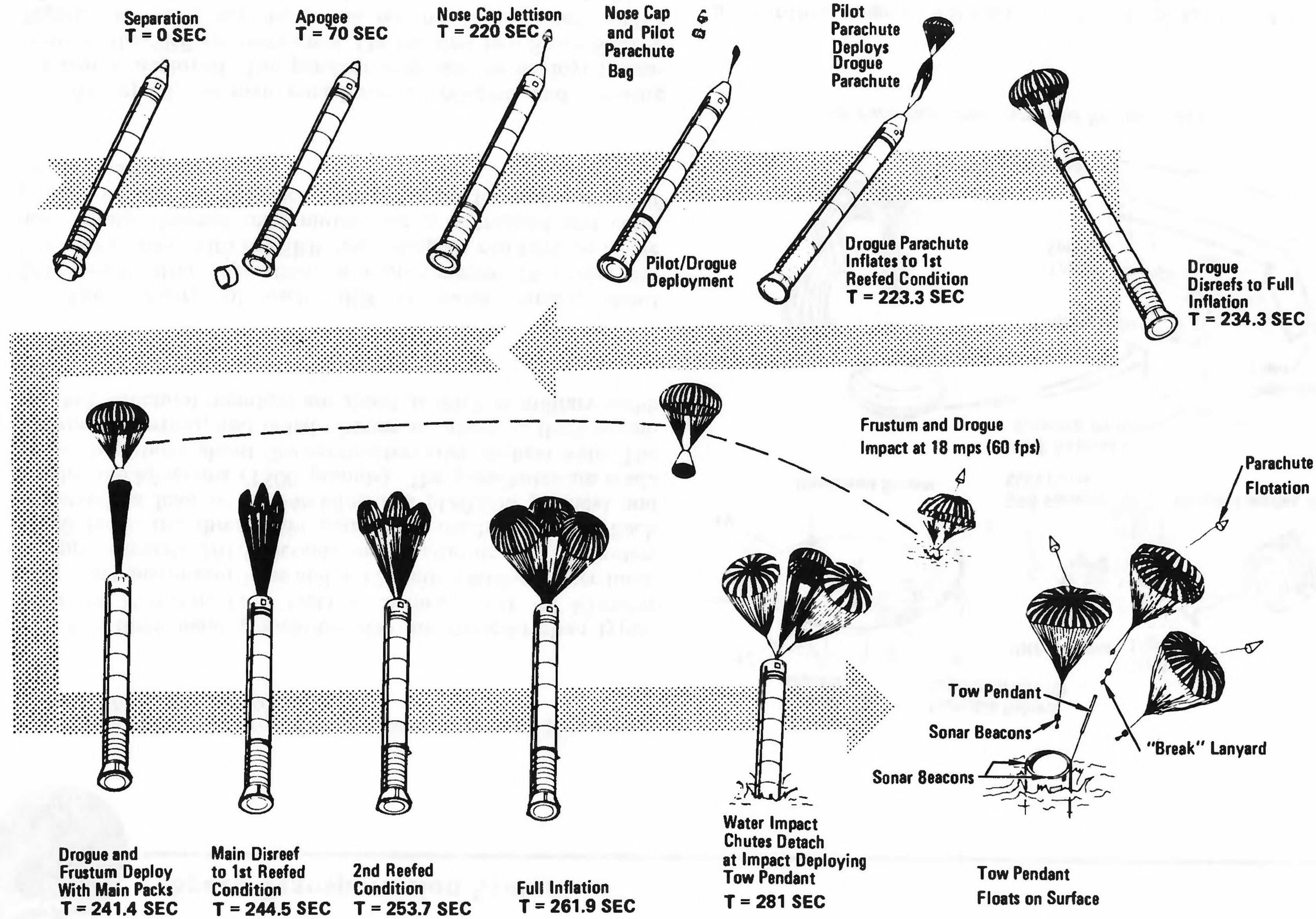


SRB Battery Power

Approximately 220 seconds after separation and when the altitude switch senses an atmospheric pressure corresponding to approximately 4693 meters (15,400 feet), a nose cap initiator fires, and the energy is sent through confined detonating fuse manifold and assemblies to three thrusters on the top ring of the frustum. The nose cap is jettisoned, which fires a pyrotechnic charge to deploy a 3.5-meter-diameter (11.5-foot) conical, ribbon, pilot chute with a 5.4-meter (18-foot) suspension line and 9.7-meter (32-foot) riser line, which then strips the pilot bag and extracts the drogue parachute.

The drogue chute, 16 meters (54 feet) in diameter with 12 30-meter (100-foot) suspension lines, slows the booster's descent. The drogue disreefs to full inflation approximately 234.3 seconds after separation at an altitude of approximately 2834 meters (9300 feet). It withstands a load of 122,472 kilograms (270,000 pounds) and weighs 544 kilograms (1200 pounds).

Approximately 241.4 seconds after separation, another initiator fires, and its energy is sent through a confined detonating fuse to a linear-shaped charge, which separates the frustum



SRB Descent and Splash Down

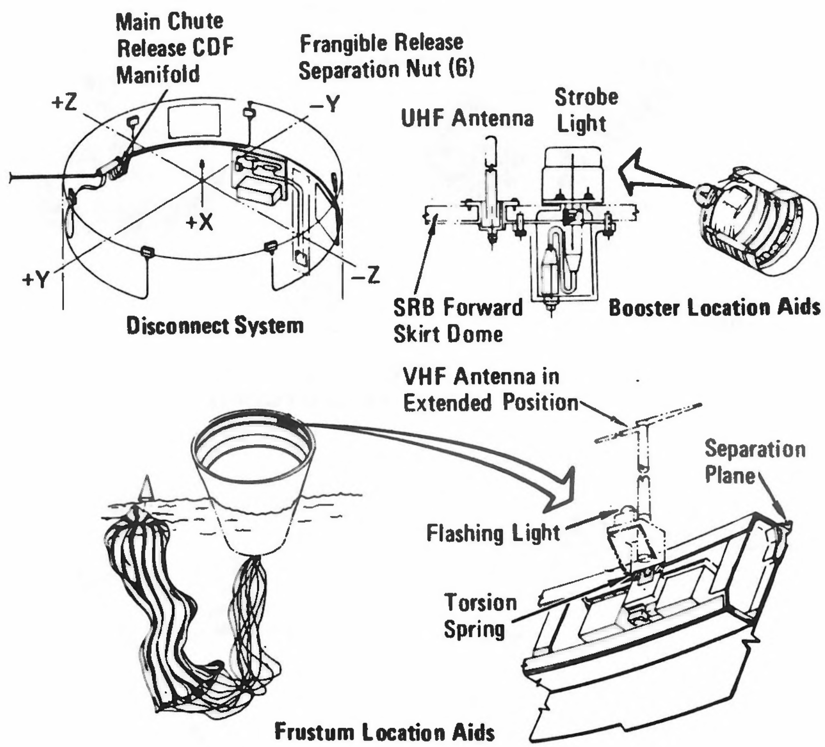
and drogue chute and strips the main parachute bags. This occurs at about 2011 meters (6600 feet).

The three main parachutes also are conical-ribbon types. They are 35 meters (115 feet) in diameter with 48 40-meter (132-foot) suspension lines and 8 12-meter (40-foot) riser lines. At approximately 261.9 seconds and an altitude of 6707 meters (2200 feet), the three main parachutes reach full open. Each withstands a load of 81,648 kilograms (180,000 pounds) and weighs 680 kilograms (1500 pounds). The parachutes are made of nylon ribbons about five centimeters (two inches) wide. The horizontal, vertical, and radial ribbons are about as thick as canvas, but structural members are about as thick as military webb belts.

The velocity of each SRB at water impact, about 281 seconds after separation, is approximately 26 meters (87 feet) per second. With the SRB descending aft end first, air in the now empty (burned out) motor casing is trapped and compressed, causing the booster to float with the forward end out of the water after splashdown.

At impact, the main parachutes are released, and a towing pendant is deployed. The pendant may also be deployed manually by the SRB recovery crew. On the first two Space Shuttle flights, one main parachute will not be disconnected, and a 12-meter (40-foot) riser attached to one of the main parachutes will be used for towing.

The radio transponder in each SRB has a range of 8.9 nautical miles (10.35 statute miles), and the flashing light has



SRB Parachute Disconnect and Recovery Aids

a night-time range of 4.9 nautical miles,(5.75 statute miles) with 10 power optical aids, with wave heights of 2.9 meters (9.5 feet). The flashing light is activated upon water impact.

Each frustum/drogue chute has an RF transmitter with a range of 8.9 nautical miles (10.35 statute miles) and a flashing light with a night-time range of 4.9 nautical miles (5.75 statute miles) with wave heights of 2.9 meters (9.5 feet). The flashing light is activated upon water impact.

The main parachutes have an RF transmitter with a range of 8.9 nautical miles (10.35 statute miles) and a flashing light with a night-time range of 4.9 nautical miles (5.75 statue miles) with wave heights of 2.9 meters (9.5 feet). The flashing light is activated upon water impact. In addition, sonar beacons are used with a range of 2.9 nautical miles (3.45 statute miles).

Various parameters of SRB operation are monitored and displayed on the orbiter flight deck control and display panel and are transmitted to ground telemetry.

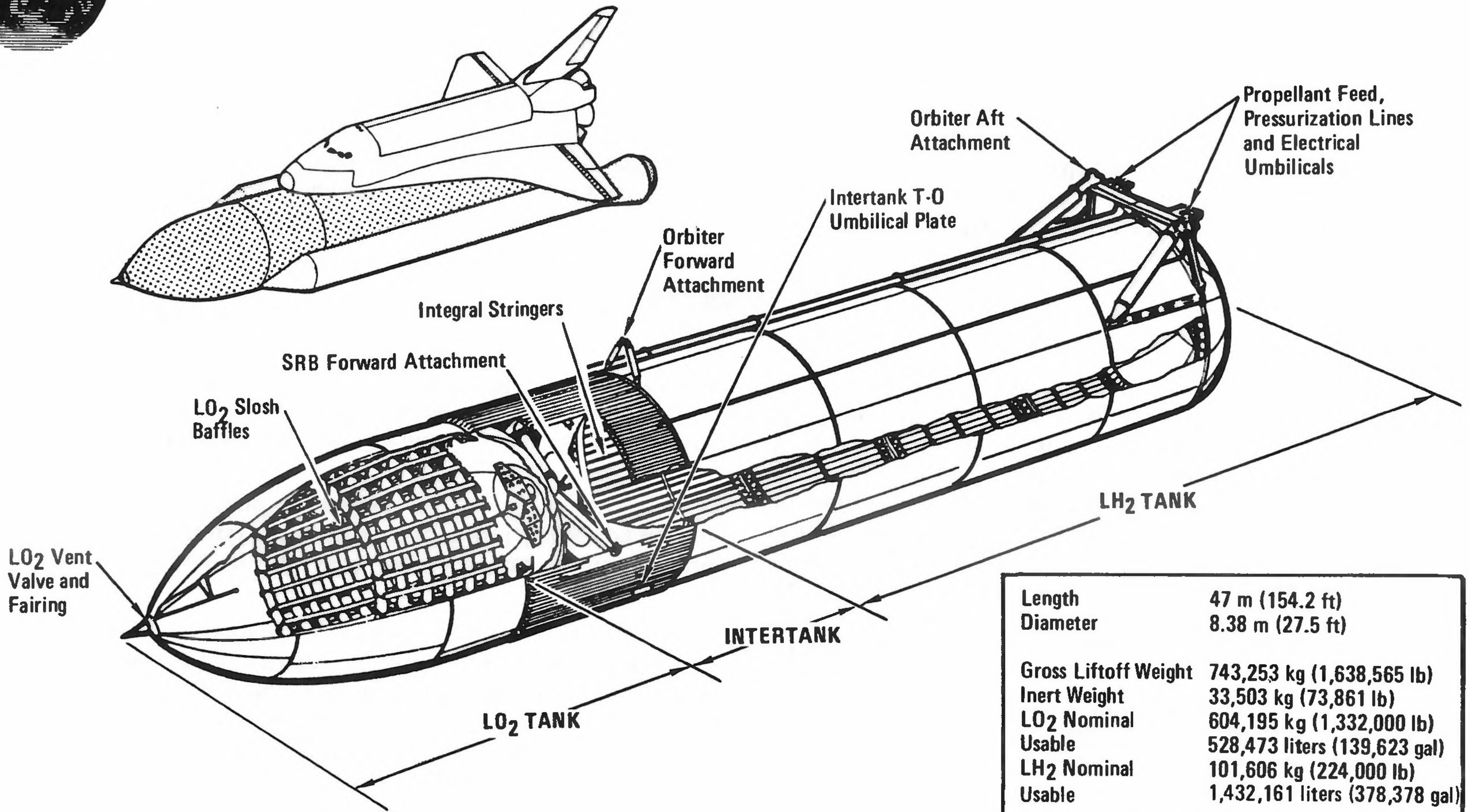
CONTRACTORS. Thiokol Chemical Corp., Wasatch Division, Brigham City, Utah, is prime contractor for the solid-rocket booster motors. Other contractors include McDonnell Douglas Astronautics Co., Huntington Beach, Calif. (SRB structures); United Space Boosters Inc., Sunnyvale, Calif. (SRB checkout, assembly, launch, and refurbishment, except for motors); Pioneer Parachute Co., Manchester, Conn. (parachutes); Abex Corp., Oxnard, Calif. (hydraulic pumps); Arde Inc., Mahwah, N.J. (hydrazine fuel modules); Arkwin Industries Inc., Westbury, N.Y. (hydraulic reservoirs); Aydin Vector Division, Newtown, Pa. (integrated electronic assemblies); Bendix Corp. Teterboro, N.J. (integrated electronic assemblies); Consolidated Controls Corp., El Segundo, Calif. (fuel isolation valves, hydrazine); Eldec Corp. Lynnwood, Wash. (integrated electronic assemblies); Explosive Technology, Fairfield, Calif. (CDF manifolds); Martin Marietta, Denver, Colo. (pyro initiator controllers); Moog Inc., East Aurora, N.Y. (servoactuators); Sperry Rand Flight Systems, Phoenix, Ariz. (multiplexers/demultiplexers); Teledyne, Lewisburg, Tenn. (location aid transmitters); United Technology Corp., Sunnyvale, Calif. (separation motors); Sundstrand, Rockford, Ill. (auxiliary power units); Motorola Inc., Scottsdale, Ariz. (range safety receivers).

EXTERNAL TANK

The external tank contains the liquid hydrogen fuel and liquid oxygen oxidizer and supplies them under pressure to the three main engines in the orbiter during liftoff and ascent. When the main engines are shut down the external tank is jettisoned, enters the earth's atmosphere, breaks up, and impacts in a remote ocean area. It is not recovered.

The largest and heaviest (when loaded) element of the Space Shuttle, the external tank has three major components: the forward liquid oxygen tank, an unpressurized intertank which contains most of the electrical components, and the aft liquid hydrogen tank. The external tank is 47 meters (154.2 feet) long and has a diameter of 8.38 meters (27.5 feet). It weighs approximately 33,503 kilograms (73,861 pounds) empty and 743,253 kilograms (1,638,565 pounds) full [604,195 kilograms (1,332,000 pounds) of LO_2 and 101,606 kilograms (224,000 pounds) of LH_2].

The external tank is attached to the orbiter at one point forward attachment and two points aft. In the aft attachment area, there are also umbilicals which carry fluids, gases, electrical signals, and electrical power between the tank and the orbiter.



Length	47 m (154.2 ft)
Diameter	8.38 m (27.5 ft)
Gross Liftoff Weight	743,253 kg (1,638,565 lb)
Inert Weight	33,503 kg (73,861 lb)
LO_2 Nominal	604,195 kg (1,332,000 lb)
Usable	528,473 liters (139,623 gal)
LH_2 Nominal	101,606 kg (224,000 lb)
Usable	1,432,161 liters (378,378 gal)

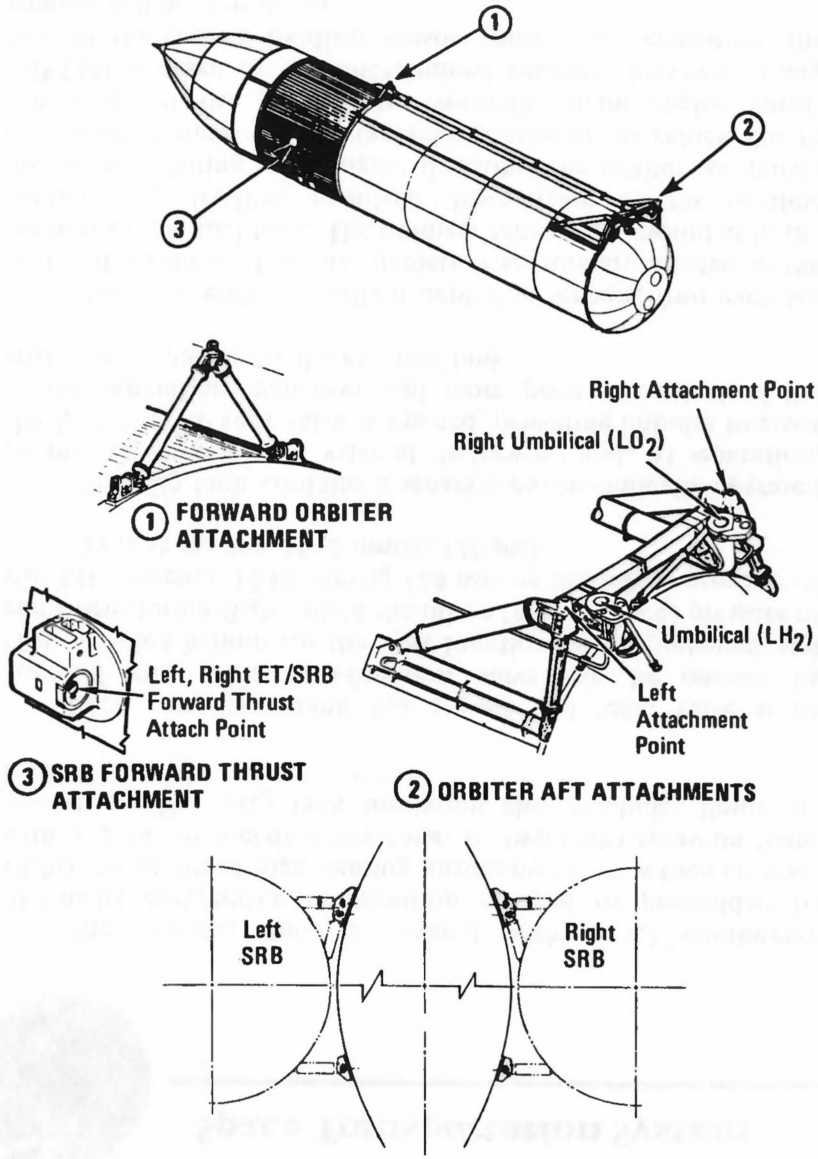
(All Weights Approximate)

External Tank

Electrical signals and controls between the orbiter and the two solid-rocket boosters also are routed through those umbilicals.

The LO_2 tank is an aluminum monocoque structure composed of a fusion-welded assembly of preformed, chem-milled gores, panels, machined fittings, and ring chords. It operates in a pressure range of 1,035 to 1,138 mmHg (20 to 22 psi). The tank contains antislosh and antivortex provisions to minimize liquid residuals and for damping fluid motion.

An antigeyser system line is installed to prevent geysering of the LO_2. The antigeyser system consists of a 10 centimeter (4 inch) diameter line in parallel with the 43 centimeter (17 inch) diameter LO_2 feedline, which provides a convection flow loop to maintain cold LO_2 in the feedline. The tank feeds into a 43 centimeter (17 inch) diameter feedline which conveys the LO_2 through the intertank, then external to the ET to the aft right hand ET/orbiter disconnect umbilical. The 43 centimeter (17 inch) diameter feedline permits LO_2 flow of 71,979 liters



Orbiter – External Tank Attachment Points

per minute (19,017 gallons per minute). The LO_2 tank's double wedge nose cone reduces drag, heating, contains the vehicle's ascent air data system, and serves as a lightning rod. The LO_2 tank volume is 552 cubic meters (19,495 cubic feet). It s diameter is 840 centimeters (331 inches), its length is 1,664 centimeters (655.5 inches) and the empty weight of 5,647 kilograms (12,450 pounds).

The intertank is a steel/aluminum semi-monocoque cylindrical structure with flanges on each end for joining the LO_2 and LH_2 tanks. The intertank houses the ET instrumentation components and provides an umbilical plate that interfaces with the ground facility arm for purge gas supply, hazardous gas detection, and hydrogen gas boiloff during ground operations. It consists of mechanically joined skin, stringers, and machined panels of aluminum alloy. The intertank is vented during flight. The intertank contains the forward SRB-external tank attach thrust beam and fittings which distribute the SRB loads to the LO_2 and LH_2 tanks. The intertank is 658 centimeters (270 inches) long and weighs 6,259 kilograms (13,800 pounds).

The LH_2 tank is an aluminum semimonocoque structure of fusion-welded barrel sections, five ring frames, and forward and aft ellipsodial domes. Its operating pressure range is 1656 to 1759 mmHg (32 to 34 psi). The tank contains an antivortex baffle and siphon outlet to transmit the LH_2 from the tank through a 43-centimeter-diameter (17-inch) line to the left aft umbilical. The LH_2 feedline flow rate is 184,420 liters per minute (48,724 gallons per minute). The LH_2 tank provides at its forward end the ET/orbiter forward attach pod strut and at its aft end the two ET/orbiter aft attach ball fittings, as well as the aft SRB-external tank stabilizing strut attachments. The LH_2 tank is 840 centimeters (331 inches) in diameter, 2948 centimeters (1161 inches) long, has a volume of 1573 cubic meters (55,552 cubic feet) and a dry weight 14,451 kilograms (31,860 pounds).

The external tank is covered with a 1.27-centimeter (0.5-inch) cork/epoxy composition sprayed or premolded to withstand localized high heating during boost. It is then covered with a 2.54- to 5-centimeter (one- to two-inch) spray-on foam insulation. The LH_2 tank insulation also precludes liquid air formation on the external surface.

Each propellant tank has a vent and relief valve at its forward end. This dual-function valve can be opened by GSE-supplied helium for the vent function during prelaunch and can open during flight when the ullage (empty space) pressure of the LH_2 reaches 1242 mmHg (24 psi) or the ullage pressure of the LO_2 tank reaches 1863 mmHg (36 psi).

The LO_2 tank contains a separate pyrotechnically-operated propulsive, tumble vent valve at its forward end. At separation, the LO_2 tumble vent valve is opened, providing impulse to assist in the separation maneuver and more positive control of the entry aerodynamics of the external tank.

There are eight propellant depletion sensors, four each for fuel and oxidizer. The fuel depletion sensors are located in the bottom of the fuel tank. The oxidizer sensors are mounted in the orbiter LO_2 feedline manifold downstream of the feedline disconnect. During main engine thrusting, the orbiter computers constantly compute the instantaneous mass of the vehicle due to the usage of the propellants. Normally, main engine cutoff (MECO) is based on a predetermined velocity; however, if any two of the fuel or oxidizer sensors sense a dry condition, the engines will be shut down.

Location of the LO_2 sensors allow the maximum amount of oxidizer to be consumed in the engines, while allowing sufficient time to shut down the engines before the oxidizer pumps cavitate (run dry). In addition, 498 kilograms (1100 pounds) of LH_2 are loaded over and above that required by 6:1 oxidizer/fuel engine mixture ratio. This assures that main engine cutoff from the depletion sensors is fuel-rich; oxidizer-rich engine shutdowns can cause burning and severe erosion of engine components.

Four pressure tranducers located at the top end of the LO_2 and LH_2 tanks monitor the ullage pressures.

Each of the two aft external tank umbilical plate mates with a corresponding plate on the orbiter. The plates help maintain alignment among the umbilicals. Physical strength at the umbilical plates is provided by bolting corresponding umbilical plates together. When the orbiter computers command external tank separation, the bolts are severed by pyrotechnic devices.

The external tank has five propellant umbilicals that interface with orbiter umbilicals: two for the LO_2 tank and three for the LH_2 tank. One of the LO_2 tank umbilicals is for liquid oxygen, the other for gaseous oxygen. The LH_2 tank umbilical has two for liquid and one for gas. The smaller-diameter LH_2 umbilical is a recirculation umbilical used only during LH_2 chilldown sequence in prelaunch.

The external tank also has two electrical umbilicals, which carry electrical power from the orbiter to the tank and the two solid-rocket boosters and provide information from the SRB's and external tank to the orbiter.

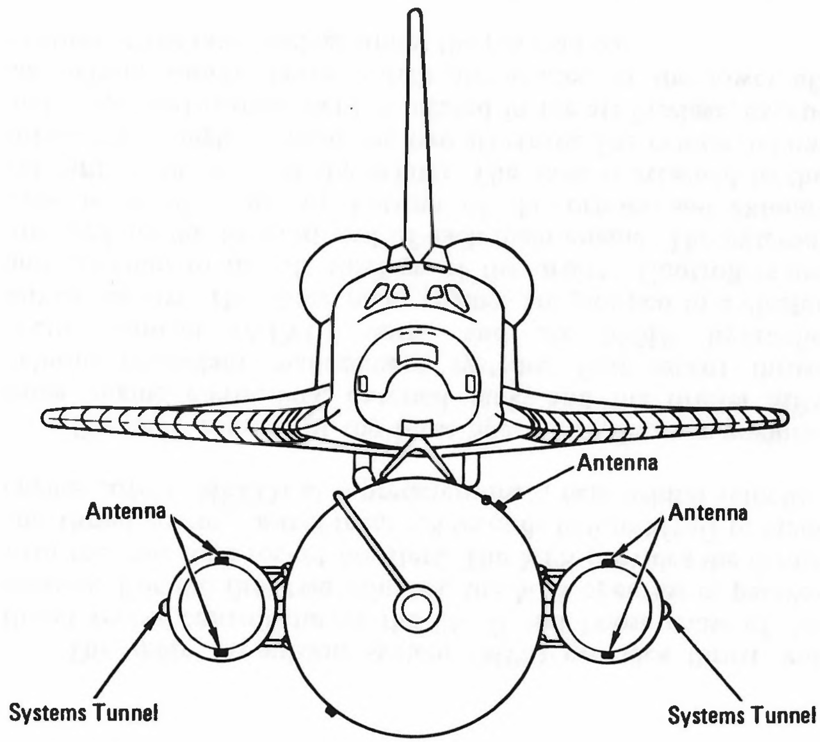
A swing-arm-mounted cap to the fixed service structure covers the oxygen tank vent on top of the external tank during the countdown and is retracted about two minutes prior to liftoff. The cap will siphon off oxygen vapor that threatened to form large ice on the external tank, thus protecting the orbiter's thermal protection system during launch.

A range safety system provides a means for destructively dispersing the propellants. It includes a battery power source, a receiver/decoder, antennas, and ordnance.

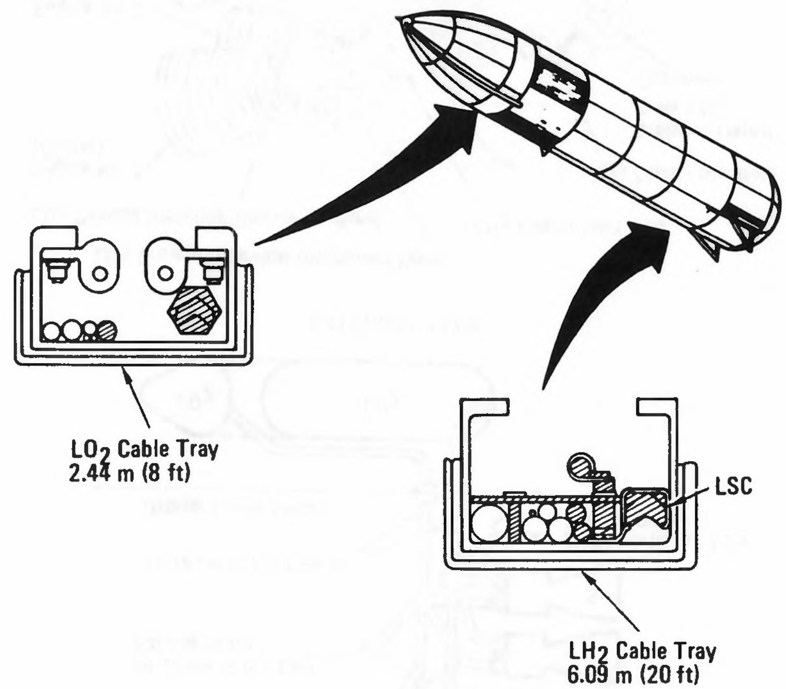
Various parameters are monitored and displayed on the flight deck display and control panel and are transmitted to the ground.

The contractor for the external tank is Martin Marietta Aerospace, Denver, Colo.; the tank is manufactured at Michoud, La. Motorola Inc., Scottsdale, Ariz., is the contractor for range safety receivers.

MAIN PROPULSION SYSTEM



External Tank Range Safety System Antennas



External Tank Range Safety System Linear Shaped Changes

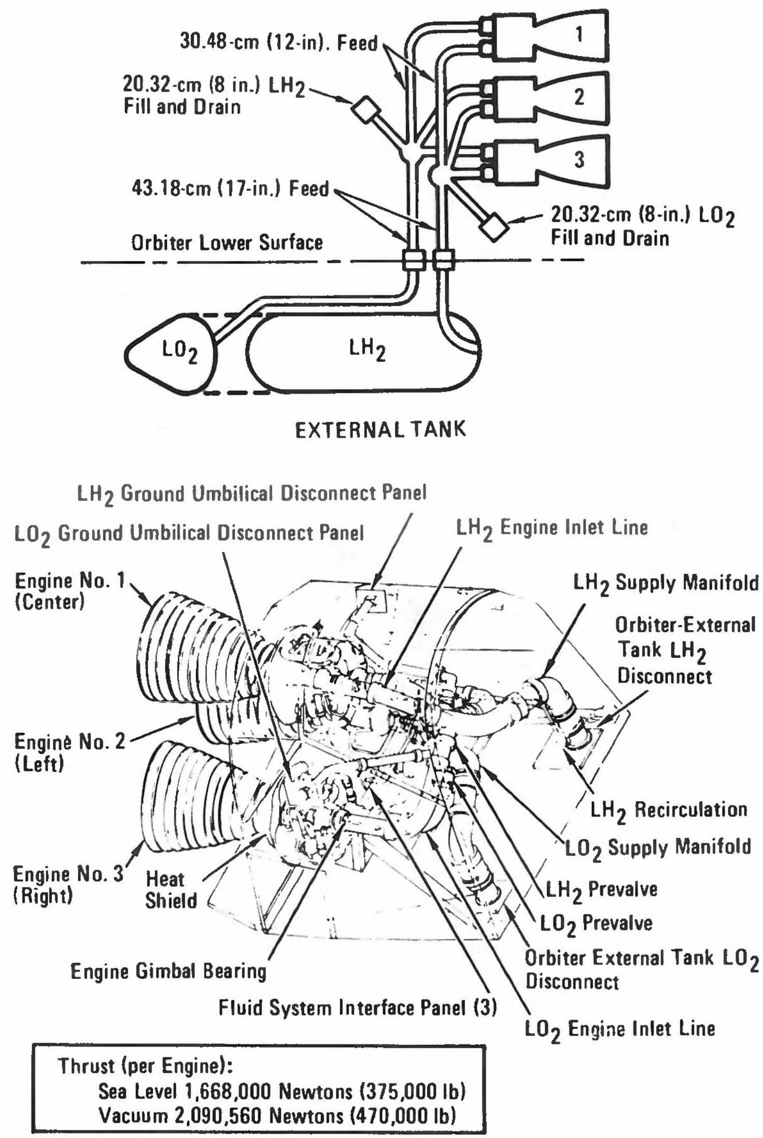
The main propulsion system (MPS), assisted by the SRB's during the initial phases of ascent provides the velocity increment (ΔV) from lift-off to a predetermined velocity increment (ΔV) prior to orbit insertion. After the two SRB's are expended and jettisoned, the MPS continues to thrust until the predetermined velocity is achieved, at which time main engine cutoff (MECO) is initiated. The external tank is jettisoned and the orbital maneuvering system (OMS) is ignited to provide the final ΔV's for orbital insertion. The magnitude of the ΔV's supplied by the OMS is dependent on payload weight, mission trajectory requirements, and systems limitations.

The main propulsion system (MPS) provides thrust and thrust vector control during the liftoff and boost phase of the mission. For the first two minutes, the MPS operates in parallel with the two solid-rocket boosters. The MPS provides the thrust and thrust vector control from 3.8 seconds before liftoff to main engine cutoff (MECO) at a predetermined, near-orbital velocity.

The MPS consists of the three Space Shuttle main engines, three engine controllers, external tank, and the orbiter MPS helium, propellant management systems, four ascent thrust vector control (ATVC) units, and six SSME hydraulic servoactuators. The three main engines are grouped in a cluster and attached to the aft fuselage of the orbiter. Controllers are attached to the forward end of each main engine. The external tank is attached to the bottom of the orbiter and extends forward of the nose of the orbiter. The tank is attached to the orbiter by a single forward and two aft struts. The orbiter helium and propellant management is located in the aft fuselage, except six helium supply tanks, which are located in the lower aft portion of the mid fuselage under the payload bay.

The main engines are reusable, high-performance liquid-propellant rocket engines with variable thrust. The propellant fuel is liquid hydrogen (LH_2) and the oxidizer is liquid oxygen (LO_2). The propellant is carried in separate tanks in the external tank and supplied to the main engine under pressure. Each engine can be gimbaled plus or minus 10.5 degrees in pitch and plus or minus 8.5 degrees in yaw for thrust vector control by hydraulically powered gimbal actuators.

Valves on each engine which control the propellant flow are hydraulically actuated using the orbiter hydraulic system pressure. All propellant valves on a given engine are operated from the same hydraulic system; only one of the hydraulic systems supplies pressure to each engine. In an emergency, all the



Main Propulsion System

propellant valves on an engine can be fully closed by using MPS-supplied helium pressure. Control of the valves is via commands from each engine's controller.

The main engines can be throttled over a range of 65 to 109 percent of their rated power level (RPL) in one-percent increments. In the development flights, the throttling range will be 65 to 100 percent. A value of 100 percent corresponds to a thrust level of 1,668,000 newtons (375,000 pounds) at sea level and 2,090,560 newtons (470,000 pounds) in a vacuum. RPL corresponds to a thrust chamber pressure of 153,180 mmHg (2960 psia).

At sea level, the engine throttling range is reduced due to flow separation in the nozzle, prohibiting operation of the engine at its 65-percent throttle setting, referred to as minimum power level (MPL). All three main engines receive the same throttle command at the same time. Normally these come automatically from the general-purpose computers through the engine controllers. During certain contingency situations, manual control of engine throttling is possible by use of the speedbrake/engine throttle controller handle. The throttling ability reduces vehicle loads during maximum aerodynamic pressure, limits vehicle acceleration to 3 g's maximum during boost, and makes it possible to abort with all main engines thrusting or one engine out.

Normal-main engine shutdown is executed by computer commands, based on the attainment of a specified total velocity. In a normal shutdown, all three engines are shut down simultaneously. If an engine continues to thrust after velocity is reached, or an emergency requires a premature engine shutdown, the engines can be shut down manually. The engines also can be shut down by LO_2 or LH_2 depletion. Four oxidizer depletion sensors are located in the orbiter LO_2 feedline manifold and four fuel depletion sensors are located in the external tank. Any two LO_2 or two LH_2 sensors reporting depletion will cause the computers to issue a MECO command.

Each engine is designed for 7-1/2 hours of operation over a life span of 55 starts. Throughout the throttling range, the LO_2-LH_2 mixture ratio is 6:1. Each nozzle area ratio is 77:5:1. The engines are 4.2 meters (14 feet) long and 2.4 meters (8 feet) in diameter at the nozzle exit.

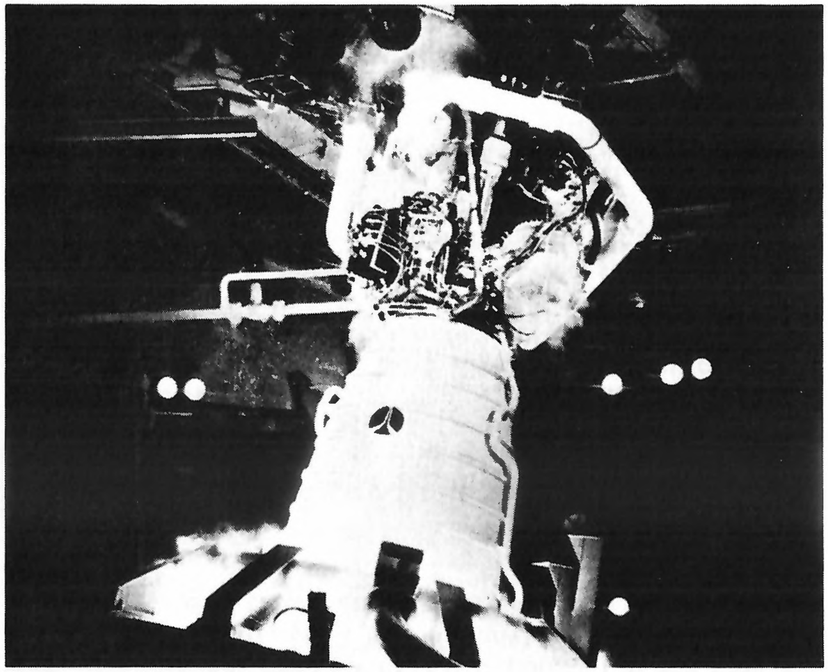
Five fluid lines connect with the external tank through connections at the bottom of the orbiter aft fuselage. The three hydrogen connections are mounted on a carrier plate on the left side of the orbiter (facing forward) and the two oxygen connections are mounted on the right side. Ground servicing is through umbilicals on both sides of the aft fuselage, hydrogen on the left side and oxygen on the right.

The orbiter portion of the MPS consists of the following systems: propellant feed, propellant fill and drain, prestart propellant conditioning, external tank pressurization, helium storage and supply, nitrogen purge (ground only), propellant management, and pogo suppression.

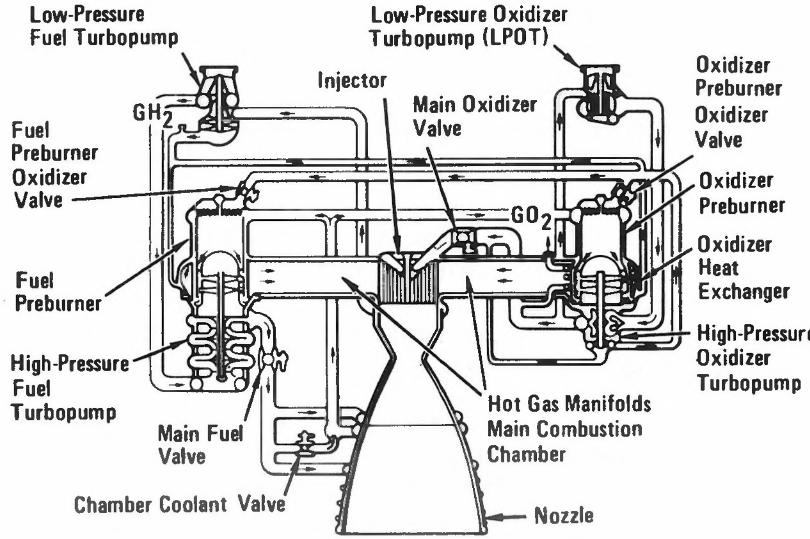
The propellant fill and drain system provides propellants from the ground umbilical to the external tank during propellant loading and draining.

The external tank is pressurized with ground-supplied helium plus hydrostatic head to attain the required prestart pressure at the main engine pump inlets. After engine thrust buildup, LO_2 and LH_2 tank pressure is maintained with vaporized propellants extracted from the engines.

The propellant feed system supplies the LO_2 and LH_2 to the SSME's from the external tank through separate



Space Shuttle Main Engine



Main Engine Flow

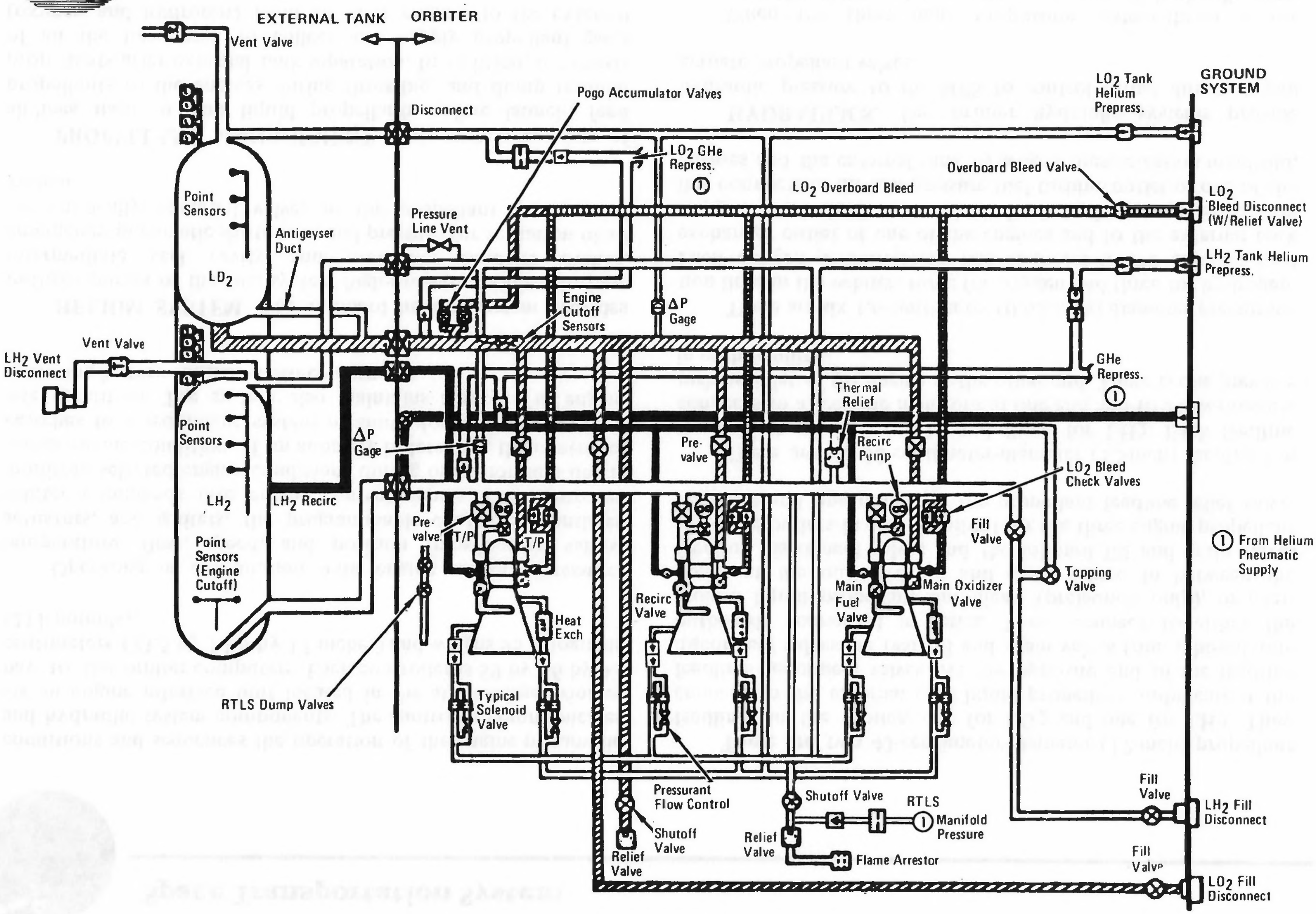
43 centimeter (17-inch) ducts in the bottom aft fuselage of the orbiter, where they each branch out in the aft fuselage to three separate 30-centimeter (12-inch) ducts, one to each engine.

The prestart propellant conditioning system provides conditioned propellants to the main engines. The nitrogen ground purge system is used for inert purging of the engines before start.

The helium storage and supply system in the orbiter is divided into a pneumatic and engine systems. The former supplies helium pressure to actuate all pneumatically-operated valves in the propellant management system, to aid in expelling propellants during MPS propellant dump, and to repressurize MPS propellant lines before entry. The latter is used in the main engines for in-flight purges and provides backup emergency actuation (shutdown) of engine propellant valves.

The propellant management system controls the loading of propellants in the external tank, diverts gases from the three main engines back to the external tank through two gas umbilicals to maintain pressure in the LO_2 and LH_2 tanks, and provides a low-level main engine cutoff as a backup to the normal orbiter velocity cutoff. During engine thrusting, the propellants flow through two umbilicals (one for LO_2 and one for LH_2) through a system of manifolds, distribution lines, and valves to the engines.

Each main engine controller is mounted on and hardwired to the engine it controls. It automatically controls checkout, start, mainstage, and engine shutdown. The solid-state controller is based on a digital computer that monitors engine operating



MPS Fluid System

conditions and sequences the operation of the engine pneumatic and hydraulic system components. The controller communicates via an engine interface unit located in the aft fuselage avionics bay to the orbiter computers. Each controller is 59 by 36 by 43 centimeters (23.5 by 14.5 by 17 inches) and weighs 95 kilograms (211 pounds).

Operating in conjunction with engine sensors (pressure, temperature, flow, speed, and position transducers), valves, actuators, and igniters, the programmable controller translates orbiter commands into engine control signals. The controller monitors selected engine conditions during operation and determines engine condition. If an anomaly is detected, the controller switches to a redundant system or shuts down the engine in a safe condition. The system also maintains a record of engine operating history for maintenance purposes.

HELIUM SYSTEM. The onboard helium system provides in-flight purges of the fuel system high-pressure oxidizer turbine intermediate seal cavity and preburner oxidizer domes, emergency pneumatic shutdown, and pressure for actuation of all pneumatically operated valves in the propellant management system.

PROPELLANT MANAGEMENT. This system consists of all lines used to load liquid propellants before launch, feed propellants to the engines during thrusting, and dump residual propellants after external tank separation. In addition, it consists of all the lines used to collect and supply propellant gases (oxygen and hydrogen) from all three engines to the external tank to maintain tank pressurization.

The liquid propellant supply and distribution network consists of feedline manifolds, engine propellant feedlines, and relief valves.

There are two 43-centimeter-diameter (17-inch) propellant feedlines in the orbiter, one for LO_2 and one for LH_2. They connect to the external tank liquid propellant umbilicals at the feedline disconnect valves. At the opposite end of the feedline disconnect valves are two fill and drain valves (one inboard, one outboard) connected in series. These connect to either the ground liquid propellant umbilicals (prelaunch only), or overboard at the outboard fill and drain valves. In between the feedline disconnect valves and the inboard fill and drain valves are three outlets in each manifold for the three engine propellant feedlines and one outlet for the propellant feedline relief valve.

There are six 30-centimeter-diameter (12-inch) feedlines in the orbiter, three for LO_2 and three for LH_2. Each feedline connects to a feedline manifold at one end and to a low-pressure turbine inlet of the engine at the other end. There is one prevalve in each feedline.

There are six 1.6-centimeter (0.63-inch) diameter pressurization lines in the orbiter, three for oxygen and three for hydrogen. Each oxygen pressurization line connects to the oxidizer heat exchanger outlet of one of the engines and to the external tank oxygen pressurization manifold. Each hydrogen pressurization line connects to the low-pressure fuel turbine outlet of one of the engines and the external tank hydrogen pressurization manifold.

HYDRAULICS. The orbiter hydraulic systems provide hydraulic pressure to the MPS to control thrust direction and actuate propellant valves.

When the three main propulsion system/thrust vector control (MPS/TVC) isolation valves are opened, hydraulic pressure is used to actuate the engine main fuel valve, main oxidizer valve, fuel preburner oxidizer valve, oxidizer preburner oxidizer valve, and chamber coolant control valve. All hydraulic actuators on an engine receive hydraulic pressure from the same system.

The left engine operates from orbiter hydraulic system 2, the center engine from system 1, and the right engine from system 3. Each actuator on an engine is controlled by dual-redundant signals (channel A/servo valve 1 and channel B/servo valve 2) from that engine controller. All actuators on an engine are supplied with helium pressure from the same engine helium tank supply system. In the event of hydraulic lock-up, helium pressure is used to actuate the propellant valves to their fully closed position upon shutdown.

MAIN ENGINE HYDRAULIC GIMBAL ACTUATORS. Each engine has two hydraulic gimbal servoactuators, one for pitch and one for yaw. These provide the force to gimbal (turn) each engine for thrust vector control during liftoff and ascent.

Hydraulic pressure from the orbiter hydraulic systems to each servoactuator is controlled by the MPS/TVC isolation valves in the three hydraulic systems. When these three valves are opened, hydraulic pressure is supplied to the servoactuators, except that only two of three hydraulic systems are connected to each servoactuator (one as primary and one as standby). The two hydraulic systems are connected to a switching valve at each servoactuator which senses loss of the primary hydraulic system and automatically provides pressure from the standby system. This prevents any loss of thrust vector control.

The Space Shuttle ascent thrust vector control (ATVC) portion of the flight control system directs the thrust of the three main engines and two SRB nozzles to control attitude and trajectory during liftoff and ascent. The TVC system, responding to commands from the guidance system, transmits commands to the ATVC drivers, which send signals proportional to the commands to each servoactuator of the main engines and SRB's. Four independent flight control system channels and four ATVC channels control six main engine and four SRB ATVC drivers, with each driver controlling one hydraulic port on each main engine and SRB servoactuator.

SPACE SHUTTLE MAIN ENGINES. Oxidizer from the external tank enters the orbiter at the LO_2 feedline disconnect valve, then into the orbiter LO_2 feedline manifold, where it branches out into three parallel paths, one to each engine. In each of these branches an LO_2 prevalve must be opened to permit flow to the low-pressure oxidizer turbopump (LPOT).

The LPOT is an axial flow pump driven by a six-stage turbine powered by LO_2. It boosts the pressure of the LO_2 from 5,175 mmHg (100 psi) to 23,649 mmHg (457 psi). The flow from the LPOT is supplied to the high-pressure oxidizer turbopump (HPOT). During engine operation, the pressure boost provided by the LPOT permits the HPOT to operate at high speeds without cavitation. The LPOT operates at approximately 5,075 rpm. The LPOT is approximately 45 by 45 centimeters (18 x 18 inches) and is connected to the vehicle propellant ducting and supported in a fixed position by the orbiter structure.

The HPOT consists of two single stage centrifugal pumps (main pump and preburner pump) mounted on a common shaft driven by a two-stage turbine. The main pump boosts the LO_2 pressure from 23,649 mmHg (457 psi) to 239,757 mmHg (4,633 psi) while operating at approximately 29,120 rpm. The HPOT discharge flow splits into several paths, one of which is routed to drive the LPOT turbine. Another path is routed to and through the main oxidizer valve and enters into the engine main combustion chamber. Another small flow path is tapped off and sent to the oxidizer heat exchanger. Before the LO_2 enters the oxidizer heat exchanger, it flows through an anti-flood valve that prevents LO_2 from entering the heat exchanger until sufficient heat is present to convert the LO_2 to gas. The heat exchanger utilizes the heat contained in the discharge gases from the HPOT turbine to convert the LO_2 to gas which is then sent to a manifold where it joins gas from the other engines and is sent to the external tank to pressurize the LO_2 tank. Another path enters the HPOT second-stage (preburner pump) to provide additional boost of the LO_2 pressure from 239,757 mmHg (4,633 psi) to 396,146 mmHg (7,655 psi) and passes through the oxidizer preburner oxidizer valve (OPOV) into the oxidizer preburner (OPB), and through the fuel preburner oxidizer valve (FPOV) and into the fuel preburner (FPB). The HPOT is approximately 60 by 91 centimeters (24 x 36 inches). The HPOT is flange attached to the hot gas manifold.

Fuel enters the orbiter at the LH_2 feedline disconnect valve, then into the orbiter GH_2 feedline manifold, and branches out into three parallel paths to each engine. In each of these LH_2 branches is an LH_2 prevalve, which when open permits LH_2 flow to the low-pressure fuel turbopump (LPFT).

The LPFT is an axial flow pump driven by a two-stage turbine powered by GH_2. It boosts LH_2 pressure from 1,552 mmHg (30 psi) to 12,161 mmHg (235 psi) and supplies it to the high-pressure fuel turbopump (HPFT). During engine operation, the pressure boost provided by the LPFT permits the HPFT to operate at high speeds without cavitation. The LPFT operates at approximately 14,760 rpm. The LPFT is approximately 45 by 60 centimeters (18 x 24 inches). The LPFT is connected to the vehicle propellant ducting and is supported in a fixed position by the orbiter structure 180° from the LPOT.

The HPFT is a three-stage centrifugal pump driven by a two-stage hot gas turbine. It boosts LH_2 pressure from 12,161 mmHg (235 psi) to 319,504 mmHg (6,174 psi). The HPFT operates at approximately 30,015 rpm. The discharge flow from the turbopump is routed to and through the main valve and then splits into three flow paths. One flow path is through the jacket of the main combustion chamber where the hydrogen is used for cooling the chamber walls. It is then routed from the main combustion chamber to the LPFT where it is used to drive the LPFT turbine. A small portion of the flow from the LPFT is then directed to a common manifold from all three engines to form a single path to the external tank to maintain LH_2 tank pressurization. The remaining hydrogen passes between the inner and outer walls to cool the hot gas manifold and is discharged into the main combustion chamber. The second hydrogen flow path from the main fuel valve is through the engine nozzle (for cooling of the nozzle), which then joins the third flow path from the chamber coolant valve. The combined flow is then directed to the fuel and oxidizer preburner. The HPFT is approximately 55 by 111 centimeters (22 x 44 inches). The HPFT is flange attached to the hot gas manifold.

The oxidizer (OPB) and fuel (FPB) preburners are welded to the hot gas manifold. The fuel and oxidizer enter each preburner where they are mixed so that efficient combustion can occur. The augmented spark igniter is a small combustion chamber located in the center of the injector of each preburner. The two dual spark igniters which are activated by the controller are used during the engine-start sequence to initiate combustion in each preburner. They are turned off after approximately three seconds as the combustion process is self-sustaining. The preburners produce the fuel-rich hot gas which passes through the turbines to generate the power for operation of the high-pressure turbopumps. The oxidizer preburner outflow drives the turbine which is connected to the HPOT and oxidizer preburner pump. The fuel preburner outflow drives the turbine which is connected to the HPFT.

The HPOT turbine and HPOT pumps are mounted on a common shaft. Mixing of the fuel-rich hot gas in the turbine section and the LO_2 in the main pump could create a hazard. To prevent this, the two sections are separated by a cavity which is continuously purged with the MPS engine helium supply during engine operation. Leakage into the cavity is minimized by two seals, one between the turbine section and the cavity and other between the pump section and cavity. Loss of helium pressure to this cavity results in automatic shutdown of that engine.

The speed of the HPOT and HPFT turbines depends on the position of the corresponding oxidizer and fuel preburner oxidizer valves. These valves are positioned by the engine controller, which uses them to throttle the flow of LO_2 to the preburners and thus control engine thrust. The oxidizer and fuel preburner valves increase or decrease the LO_2 and LH_2 flow, thus increasing or decreasing preburner chamber pressure, HPOT and HPFT turbine speed, and LO_2 and GH_2 flow into the main combustion chamber, increasing or decreasing the thrust of that engine, thus throttling the engine. The oxidizer and fuel preburner valves operate together to throttle the engine and maintain the engine mixture ratio constant at 6:1.

The main oxidizer valve (MOV) and main fuel valve (MFV) control LO_2 and GH_2 flow into the engine and are controlled by each engine controller. When an engine is operating the main valves are fully open.

A coolant control valve (CCV) is mounted on the combustion chamber coolant bypass duct of each engine. The engine controller regulates the amount of GH_2 allowed to bypass the nozzle coolant loop, thus controlling its temperature. The chamber coolant valve is 100 percent open before engine start and after shutdown. During engine operation, it will be 100-percent open for throttle settings of 100 to 109 percent for minimum cooling. For throttle settings between 65 to 100 percent, its position will range from 66.4- to 100-percent open for maximum cooling.

Each engine main combustion chamber (MCC) receives fuel-rich hot gas from a hot gas manifold (HGM) and GH_2 from the hot gas manifold cooling circuit. The GH_2 and LO_2 enter the chamber at the injector which mixes the propellants. A small augmented spark igniter (ASI) chamber is located in the center of the injector. The dual-redundant igniter is used during the engine start sequence to initiate combustion. After approximately three seconds, the igniters are turned off and the combustion process is self-sustaining. The main injector and dome assembly is welded to the hot gas manifold. The main combustion chamber is bolted to the hot gas manifold.

The inner surface of each combustion chamber, as well as the inner surface of each nozzle, are cooled by GH_2 flowing through coolant passages. The nozzle assembly is a bell-shaped extension which is bolted to the main combustion chamber. The nozzle is 287 centimeters (113 inches) long and has a 238 centimeter (94 inch) exit outside diameter. An engine to orbiter heat shield support ring is welded to the forward end of the nozzle and is the engine attach point for the orbiter supplied heat shield. Thermal protection for the nozzles is provided due to the thermal environment the exposed portions of the nozzles experience during the launch, ascent, on-orbit and entry phases of a mission. The insulation consists of four layers of metallic batting covered with a metallic foil and screening.

The five propellant valves on each engine (oxidizer preburner oxidizer, fuel preburner oxidizer, main oxidizer, main fuel, and chamber coolant) are hydraulically actuated and controlled by electrical signals from the controller. They can be fully closed using the MPS engine helium supply system as a backup actuation system.

The low-pressure oxygen (LPOT) and fuel (LPFT) turbopump are mounted 180 degrees apart on the orbiter aft fuselage thrust structure. The lines from the low-pressure turbopumps to the high-pressure turbopumps contain flexible bellows that enable the low-pressure turbopumps to remain stationary while the rest of the engine is gimbaled for thrust vector control purposes. The LH_2 line from the LPFT to the HPFT is insulated to prevent the formation of liquid air.

The main oxidizer and fuel valves also are used after shutdown. During a propellant dump, the main oxidizer valve will be opened to allow residual LO_2 to be dumped overboard through the engine and the residual LH_2 is dumped overboard through the LH_2 fill and drain valves. After completion of the dump, the valves close and remain closed for the remainder of the mission.

The gimbal bearing is bolted to the main injector and dome assembly and is the thrust interface between the engine and orbiter. The gimbal bearing assembly is approximately 28 by 35 centimeters (11.3 by 14 inches).

The overall weight of an SSME is approximately 3,001 kilograms (6,618 pounds).

MPS SHUTDOWN, DUMP, PURGE AND INERTING. Main engine cutoff (MECO) normally is based on attainment of a specified velocity. The engines also can be shut down by LO_2 or LH_2 depletion. Four oxidizer depletion sensors are located in the orbiter LO_2 feedline manifold and four fuel depletion sensors are located in the external tank. Any two LO_2 or two LH_2 sensors reporting depletion will cause the computers to issue a MECO command.

The MPS pneumatic control assembly on each engine provides an emergency backup mode of closing the engine propellant valve pneumatically by using helium pressure. The normal engine shutdown of the engine propellant valves is accomplished by hydraulic actuation.

After MECO and before external tank separation, the computers isolate the orbiter LO_2 and LH_2 feedlines from the external tank by simultaneously closing the LO_2 and LH_2 feedline disconnect valves, one on each side of the separation interface. The GO_2 and GH_2 lines are isolated at separation by self-sealing disconnects. The three LO_2 and LH_2 prevalves also are closed before separation.

At MECO, the computers open LO_2 and LH_2 feedline relief isolation valves, allowing any pressure buildup generated by the propellant trapped in the feedline manifolds to be vented overboard. At the same time, the pneumatic control assemblies perform a six-second purge of preburner oxidizer domes.

Approximately 11 seconds after MECO, the external tank separates from the orbiter. About 109 seconds later, the orbital maneuvering system (OMS) burst thrusting period begins. Concurrent with this, the computers automatically initiate the LO_2 dump. Approximately 771 kilograms (1,700 pounds) of propellants will be trapped in the engines and 1,678 kilograms (3,700 pounds) in the orbiter propellant lines at separation. This represents an overall center-of-gravity shift for the orbiter of approximately 17 centimeters (7 inches) and could cause guidance problems during entry or a return-to-launch-site abort maneuver. The residual LO_2, the heavier of the two propellants, creates the greatest impact on center-of-gravity travel. The greatest hazard from the trapped LH_2 occurs during entry, when any LH_2/GH_2 remaining in the propellant lines may combine with atmospheric oxygen to form an explosive mixture. In addition, unless dumped overboard, trapped propellants will sporadically outgas through the relief valves, causing spacecraft accelerations which are of such a low level that they cannot be sensed by onboard guidance yet represent a significant source of navigation error over an entire mission. Outgassing propellants also are a source of contamination for scientific experiments contained in the payload bay.

The computers initiate LO_2 dump and also command the engine controllers to initiate their portion of the sequence.

Trapped LO_2 in the feedline manifolds is expelled under pressure from the MPS helium system through each engine and out its nozzle. The flight crew can perform the LO_2 dump manually although all valve opening and closing sequences are still automatic.

Simultaneously with LO_2 dump, the remaining liquid hydrogen is dumped overboard through the hydrogen fill and drain valves for six seconds, when the hydrogen inboard fill and drain valve is closed and the hydrogen recirculation valve is opened and flows through the engine bleed valves to the orbiter hydrogen MPS line between the inboard and outboard hydrogen fill and drain valves, dumping the remaining hydrogen out through the outboard fill and drain valve for approximately 100 seconds.

After the LH_2 dump, the crew inerts the orbiter LO_2 and LH_2 feedline manifolds and the LH_2 tank pressurization line. The pneumatic helium isolation valves are opened by the flight crew, which permits helium to activate the two LO_2 and two LH_2 fill and drain valves. This allows any traces of LO_2 or LH_2 left over after the propellant dumps to be vented to space (vacuum inerting). For LO_2, the crew opens the three LO_2 prevalves, and LO_2 inboard and outboard fill and drain valves. At the same time, the LH_2 inboard and outboard fill and drain valves are opened, as well as the GH_2 pressurization line vent valve to vacuum inert the external tank GH_2 pressurization manifold. Approximately 30 minutes is allowed for vacuum inerting.

During the early part of orbiter entry, the MPS propellant feed system and external tank pressurization lines are repressurized with helium from the MPS helium supply. This prevents atmospheric contaminants from being drawn into the manifolds and lines. The flight crew repressurizes the propellant feed system by opening the LO_2 and LH_2 pressurization valves and the MPS pneumatic helium system. These valves allow helium to flow through the pressurization valves into the feedline manifolds and pressurization.

If engine cutoff is preceded by a return-to-launch site (RTLS) abort, the LH_2 and LO_2 dump will start simultaneously 10 seconds after the external tank separation command. The dumps are initiated and terminated automatically. The LO_2 dump sequence is normal except that the LO_2 feedline manifold is not pressurized. As a result, very little LO_2 is dumped overboard through the engine nozzles, because of the LO_2 feedline location and its orientation to the orbiter's acceleration vector during RTLS aborts. Thus, RTLS LO_2 dump is more of a venting process than dump. The computers terminate the LO_2 dump approximately 280 seconds after initiation. The LH_2 dump sequence opens the inboard and outboard RTLS dump valves and two RTLS manifold repressurization valves and trapped LH_2 in the feedline manifold is expelled under helium system pressure through a special opening on the port (left) side of the orbiter between the wing and the OMS/RCS pod. LH_2 dump is terminated automatically and inboard and outboard RTLS dump valves and the two RTLS manifold repressurization valves close. No vacuum inerting or system repressurization is performed in an RTLS abort.

POGO. A major problem in large pump-fed liquid rocket engines has been a longitudinal instability phenomenon termed "pogo." This instability involves the fluid-feed system, the engines, and the vehicle structure. A pogo condition begins with small vehicle accelerations which produce perturbations through the external tank propellant tanks, feedline, and turbopump supports into the propellant feed system pressure and flow rate, which in turn cause thrust oscillations resulting in increased vehicle oscillations. Accumulator-type devices installed in the lines between the engine low pressure oxidizer turbopump and high pressure oxidizer turbopump are used to suppress the perturbations.

Various functions and conditions of the main propulsion system are monitored on the flight crew display and control panel and are telemetered to the ground.

CONTRACTORS. The Rocketdyne Division of Rockwell International, Canoga Park, Calif., is prime contractor for the Space Shuttle main engines. Other contractors include Aeroflex Laboratories, Plainview, N.Y. (MPS vibration mounts); Airite Division, Sargent Industries, El Segundo, Calif. (MPS surge pressure receiver); Ametek Calmec, Pico Rivera, Calif. (1-1/2 inch and 2-inch LO_2 and LH_2 shutoff valve, 4-inch LH_2 disconnect, 2-inch GH_2/GO_2 disconnect); Ametek Straza, El Cajon, Calif. (8-inch LH_2/LO_2 fill/drain, 2-inch and 4-inch LH_2 recirculation lines, high point bleed line manifold, gimbal joint); Arrowhead Products, Division of Federal Mogul, Los Alamitos, Calif. (12 to 17-inch diameter LO_2 and LH_2 feedlines, flexible purge gas connector); Astech, Santa Ana, Calif. (MPS heat shield); Brunswick, Lincoln, Neb. (17.3-cubic-foot and 4.7-cubic-foot capacity helium tanks); Brunswick-Circle Seal, Anaheim, Calif. (helium check valves, GO_2 and GH_2 1-inch helium pressurization line, 3/8-inch LH_2 relief valve, engine isolation check valves); Brunswick-Wintec, El Segundo, Calif. (helium filter); Coast Metal Craft, Compton, Calif. (metal flex hose), Conrac Corp., West Caldwell, N.J. (engine interface unit); Consolidated Controls, El Segundo, Calif. (oxygen primary flow control hydraulic valve, hydrogen/oxygen pressurant flow control valves, 20-psi helium regulator, 850-psi helium relief valve, 750-psi helium regulator); Fairchild Stratos, Manhattan Beach, Calif. (12-inch prevalves, 1-1/2 inch LO_2 disconnect, 8-inch LO_2 and LH_2 fill and drain valves and GN_2 and GH_2 disconnects); Gulton Industries, Costa Mesa, Calif. (pogo pressure transducer); K-West, Westminster, Calif. (LO_2 and LH_2 external tank ullage pressure signal conditioner, MPS differential pressure transducer and electronics propellant head pressure); Megatek, Van Nuys, Calif. (MPS line flange cryo seals); Moog, Inc., East Aurora, N.Y. (main engine gimbal actuators); Parker Hannifin Corp., Irvine, Calif. (1-inch relief isolation valves, pogo check valves, 17-inch LH_2 and LO_2 disconnects, 8-inch LO_2 and LH_2 disconnects, LO_2 and LH_2 relief valves); Simmonds Precision Instruments, Vergennes, Ver. (LO_2 and LH_2 point sensors and electronics); Sterer Engineering, Los Angeles (main engine hydraulic solenoid shutoff valve); Whittaker Corp., North Hollywood, Calif. (750/20-psi helium regulator); Wright Components, Inc. Clifton Springs, N.J. (two-way pneumatic solenoid valve, three-way helium solenoid valve, hydraulic latching solenoid valve).

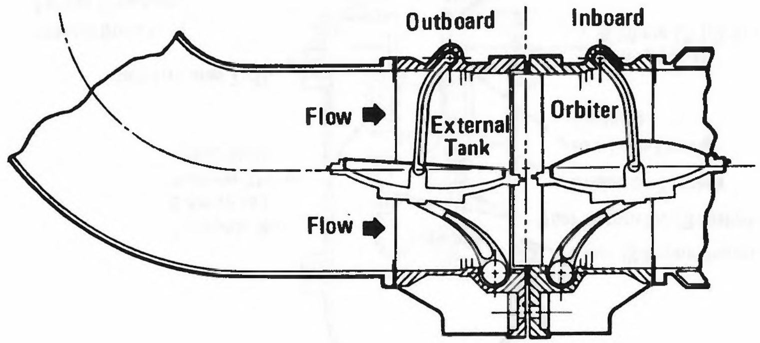
ORBITER-EXTERNAL TANK SEPARATION SYSTEM

This system separates the orbiter from the external tank at three structural points and two umbilicals. Separation of the orbiter and external tank occurs just before orbit insertion, and is triggered automatically by the orbiter computers. Orbiter crewmen also can initiate separation.

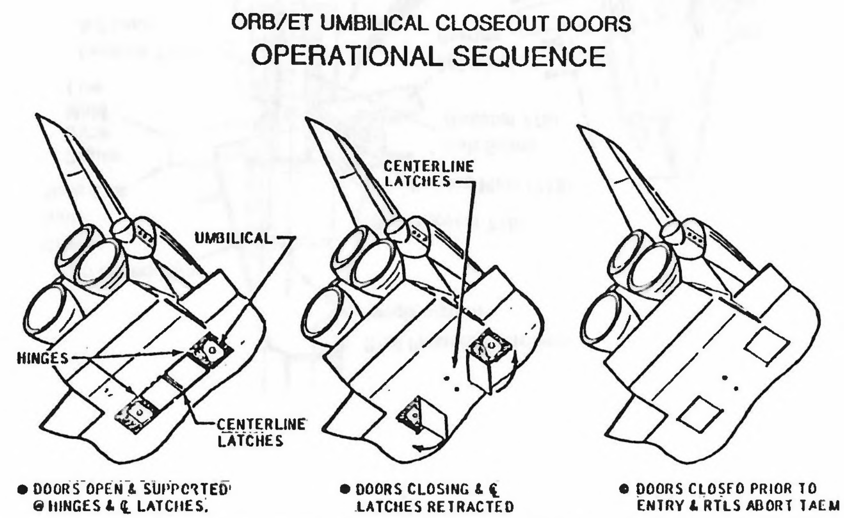
The forward structural attachment point consists of a spherical bearing and a single-piston, dual-pressure cartridge-actuated frangible nut coupled with a standard bolt. After separation, two centering plungers and springs and a stud align the bolt-bearing separation plane with the orbiter moldline by rotating the portion of the bolt contained in the ball portion of the spherical bearing assembly. No closeout door is required because the bolt stub and spherical bearing are essentially flush with the orbiter thermal protection system moldline.

The aft structural attachments consist of left- and right-side dual-detonator frangible nuts coupled with two bolts. Each bolt has a retraction spring which, after nut fragmentation, retracts the bolt into the external tank strut hemisphere to avoid interference in the separation sequence. On the orbiter side, the frangible nuts with dual detonators are enclosed in a cover assembly that contains nut fragments and hot gas generated by operation of the detonators, either of which will fracture the nut.

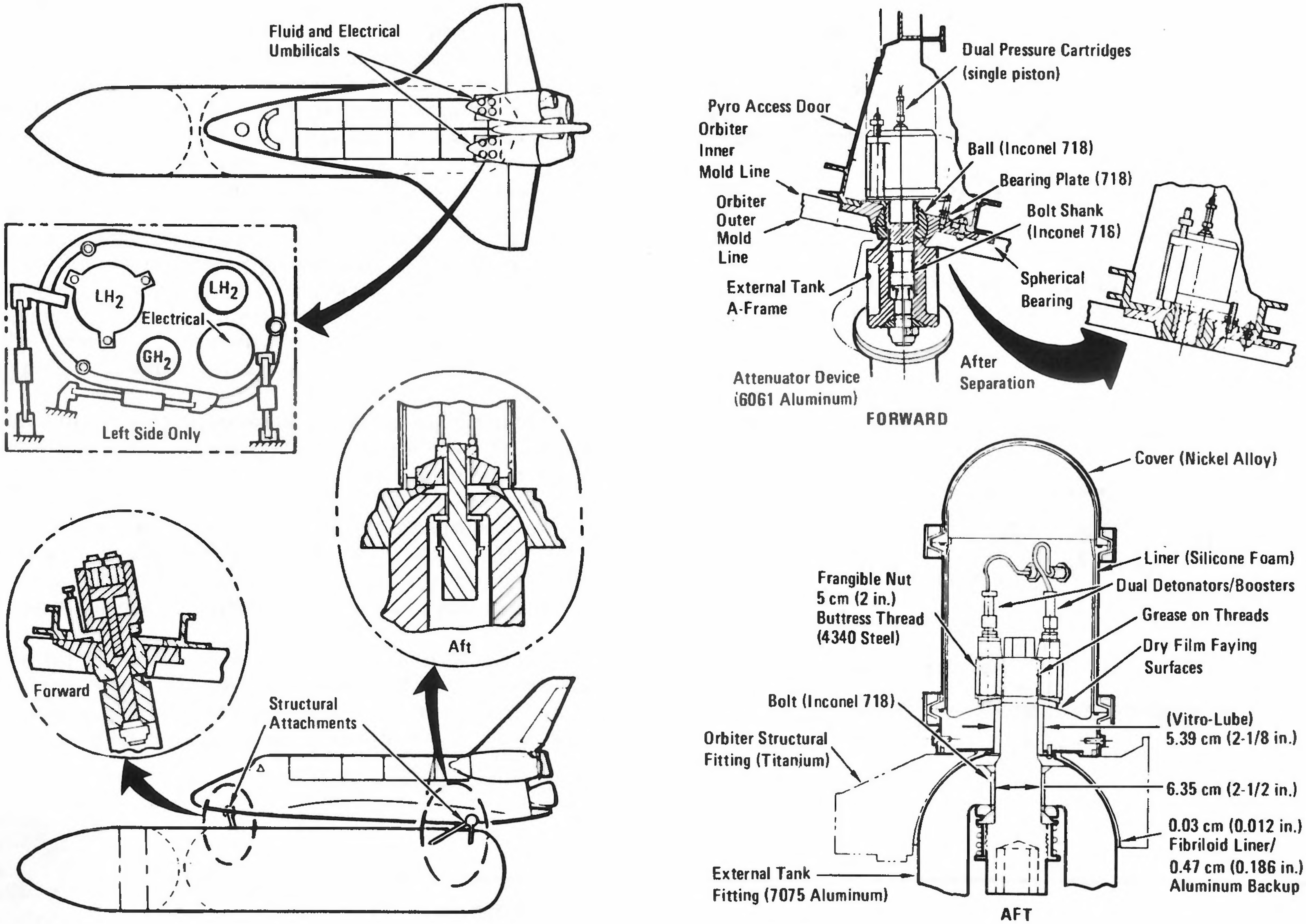
The orbiter-external tank umbilical plate separation consists of right and left assemblies. Each assembly contains three dual-detonator frangible nut/bolt combinations that hold the orbiter and external tank umbilical plates together during mated flight. Each bolt has a retraction spring which, after release of the nut, retracts the bolt to the external tank side of the interface. On the orbiter side, each frangible nut with its detonators is enclosed in a debris container that contains nut fragments and hot gases generated by operation of the detonators, either of which will fracture the nut. Each orbiter umbilical plate has three hydraulic retractors which, after release of the three frangible nut/bolt combinations, retract the plate approximately 6 centimeters (2-1/2 inches). The retraction disconnects the orbiter-external tank electrical umbilical in the first 1 centimeter (0.5 inch) of travel, releases the fluids between the disconnected LO_2 and LH_2 shutoff valves, and (as a secondary function of the retract motion through linkage) closes the LO_2 and LH_2 main feedline disconnect valves by MPS pneumatic helium supply. Each orbiter umbilical plate has three bungees that hold the plate in position after separation from the external tank umbilical plate.



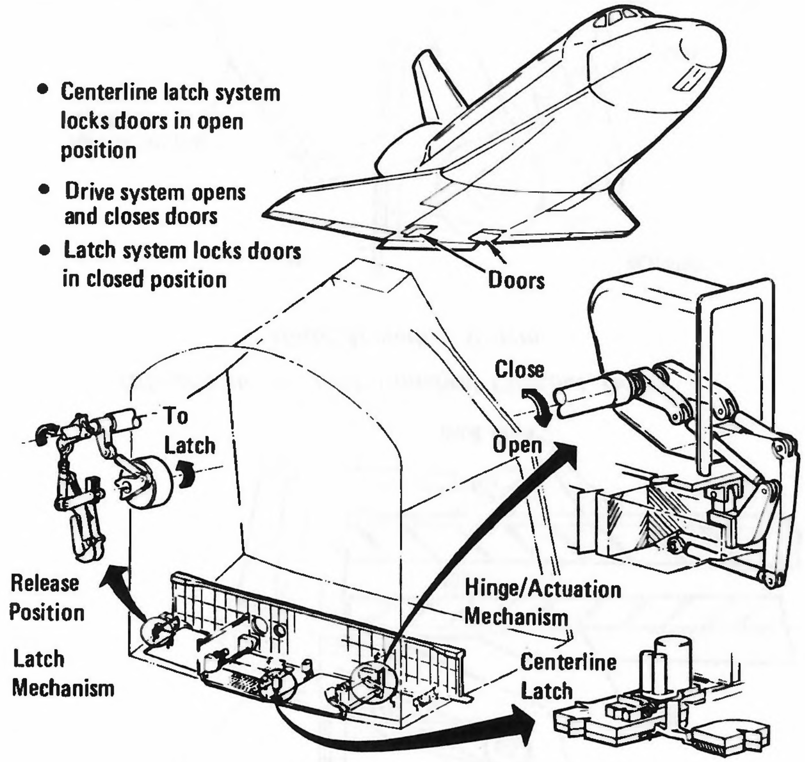
External Tank-Orbiter Disconnect Valves



Orbiter/External Tank Umbilical Door Operational Sequence



Orbiter – External Tank Separation System

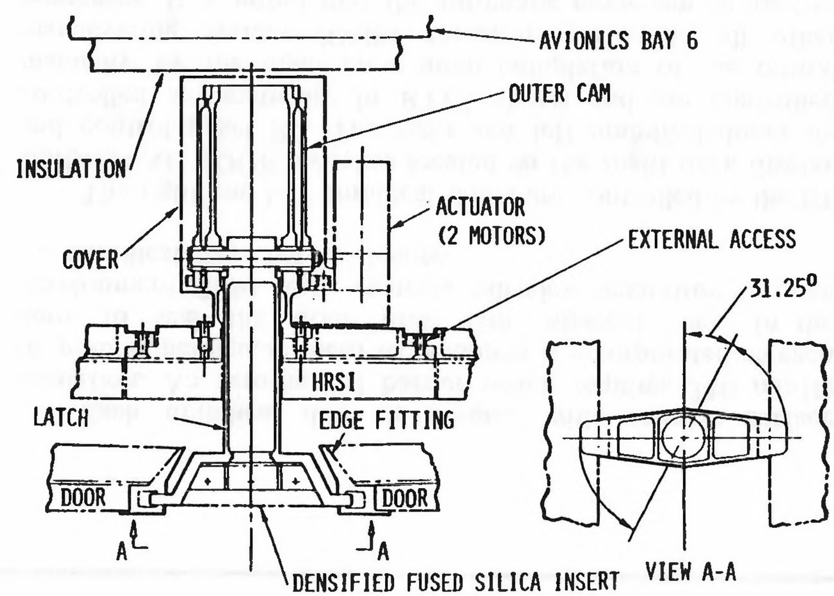


Orbiter Umbilical Doors

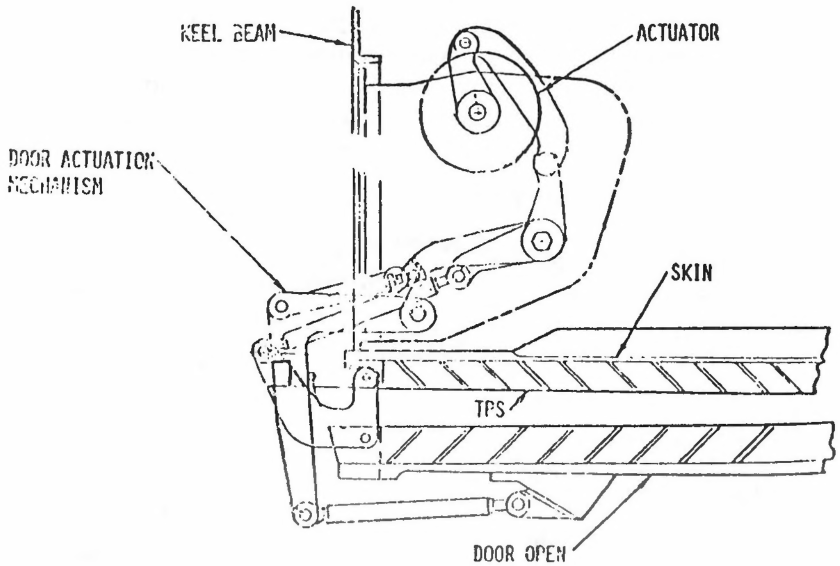
An electromechanical actuation system provides power to latch the orbiter-external tank umbilical doors in the open position, drive the doors closed, and latch them in the closed position. The door system consists of two doors (right and left) which close the left and right cavities in the orbiter structure after the external tank is jettisoned and the left and right umbilical plates are retracted. The umbilical doors are controlled manually by the flight crew. Each door is 127 centimeters (50 inches) square.

The doors are held fully open during the liftoff and ascent by two centerline latches, one forward and one aft, each latch engaging both doors. After separation, the two redundant ac motors for each door operate an electromechanical actuator to rotate the latches 31.25 degrees to free the doors for closure and retract the latches flush with the orbiter thermal protection system moldline. Hinges are on the inboard edge of each door. Two redundant ac motors for each door operate an electromechanical actuator using push rods and bell cranks to provide the force to close the door.

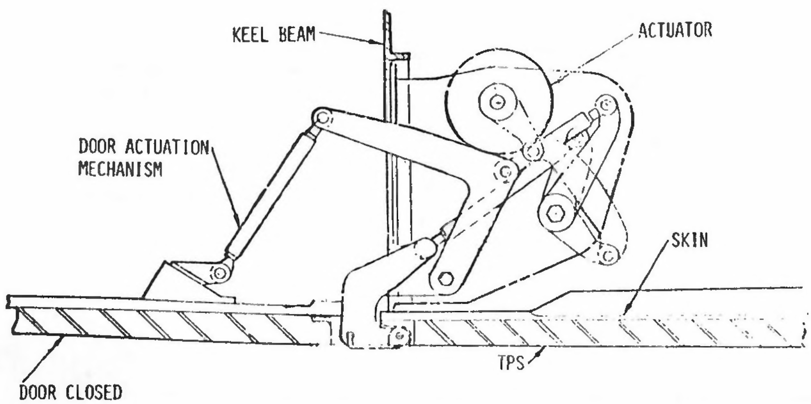
When the door is approximately 5 centimeters (2 inches) away from full closure, the three latch mechanisms are activated and engage rollers on the outboard edge of each door to pull it to the closed position. After the doors are closed, the latches and the drive mechanisms hold the doors closed for entry.



Orbiter/External Tank Umbilical Closeout Doors – Centerline Latch



Orbiter/External Tank Umbilical Closeout Doors – Hinge/Actuation System



Orbiter/External Tank Umbilical Closeout Doors – Hinge/Actuation System

Each umbilical door is covered with reusable surface insulation. An aerothermal barrier which requires 310 mmHg (6 pounds per square inch) to compress is incorporated on each door to seal the door tiles with adjacent tiles. In the development flight tests, cameras will view separation through the umbilical doors before closure.

The right and left umbilical doors are controlled by the ET UMBILICAL DOOR switches located on the flight deck display and control panel R2. The right and left umbilical doors are controlled automatically in RTLS aborts and are controlled manually by the flight crew upon completion of the orbital maneuvering system (OMS) thrusting period in all other sequences. It is noted that the automatic mode can be used to provide a backup to the manual sequence.

The manual sequence is enabled by positioning the ET UMBILICAL DOOR MODE switch on panel R2 to MAN (manual). The STOW position of the ET UMBILICAL DOOR CENTERLINE LATCH switch on panel R2 releases the two umbilical door centerline latches and stows both centerline latches. The ET UMBILICAL DOOR CENTERLINE LATCH talkback indicator indicates STO (stow) when the latches complete their motion in six seconds. The GRD (ground) position of the ET UMBILICAL DOOR CENTERLINE switch is used during ground operations. Prior to release of the centerline latches the ET UMBILICAL DOOR CENTERLINE LATCH talkback indicates BARBERPOLE.

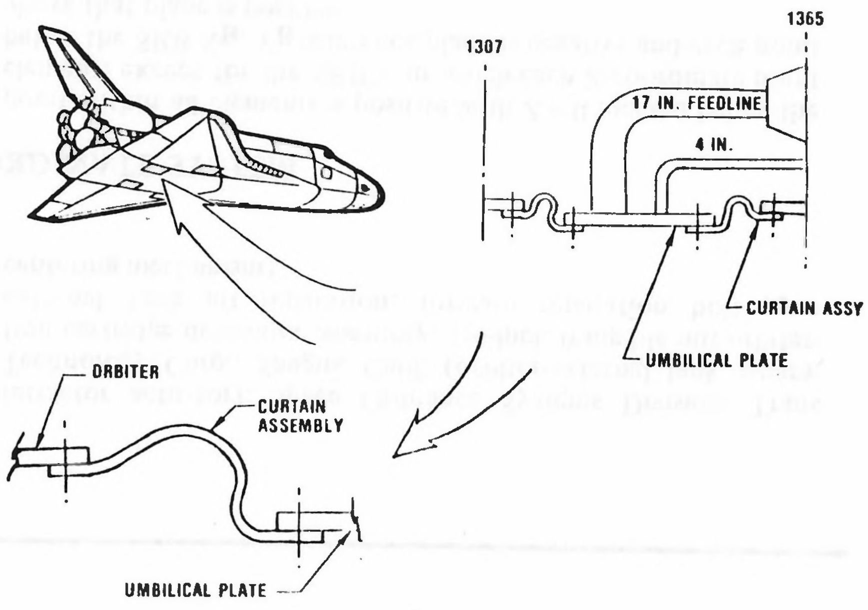
The manual control of opening or closing the right and left umbilical door is controlled by the respective LEFT and RIGHT ET UMBILICAL DOOR OPEN, CLOSE, OFF switch on panel R2. The CLOSE position commands the respective door closed in approximately 24 seconds and when the door reaches its limit of travel, the electrical motors are automatically turned off. The LEFT and RIGHT ET UMBILICAL DOOR talkback indicator above the respective switch on panel R2 indicates OP (open) prior to door closure, BARBERPOLE when the door is in transit, and CL (close) when two of the three ready-to-latch switches for that door has sensed door closure. The OFF position of the respective switch, removes electrical power from the manual door closure sequence. The OPEN position of the respective switch is used for ground operations.

The manual control of latching or releasing the right and left umbilical door is controlled by the respective LEFT and RIGHT ET UMBILICAL DOOR LATCH, RELEASE, OFF, LATCH switch on panel R2. The LATCH position commands the respective door to its latched position in approximately six seconds, providing two of the three ready-to-latch switches have been sensed and the latches hold the door closed. The LEFT and RIGHT ET UMBILICAL DOOR, talkback indicator above the respective switch on panel R2, indicates REL (release) during ascent, BARBERPOLE when that door is in transit, and LAT (latch) when that door is in its latched position. The OFF position of the respective switch, removes electrical power from the door latch closure sequence. The REL (release) position of the respective switch is used for ground operations.

The automatic sequence of the left and right umbilical doors occurs during return-to-launch-site (RTLS) aborts. The automatic sequence is enabled by positioning the ET UMBILICAL DOOR MODE switch on panel R2 to GPC. Two seconds after external tank (ET) separation the centerline latches release the doors and the latches are stowed. The ET UMBILICAL DOOR CENTERLINE LATCH talkback indicator will indicate STO when the centerline latches complete their motion at ET separation plus six seconds. The left and right umbilical doors are closed and the ET UMBILICAL DOOR LEFT and RIGHT DOOR talkback indicates CL at ET separation plus 32 seconds. The left and right umbilical door latches, latch the doors closed and the ET UMBILICAL DOOR LEFT and RIGHT talkback indicates LAT at ET separation plus 62 seconds.

A curtain is installed at each of the orbiter/external tank umbilicals. After ET separation, the residual LO_2 is dumped through the three Space Shuttle Main Engines and the residual LH_2 is dumped overboard. The umbilical curtain prevents hazardous gases (GO_2, GH_2) from entering into the orbiter aft fuselage through the orbiter umbilical openings prior to umbilical door closure. The curtain also acts as a seal during the ascent phase of the mission to permit the aft fuselage to vent through the orbiter purge and vent system, thereby providing protection to the orbiter aft bulkhead at station 1307.

Various parameters are monitored and displayed on the flight deck control panel and CRT (cathode ray tube) and transmitted by telemetry.



Orbiter/External Tank Umbilical Closeout Curtain

Contractors for the separation system include Hoover Electric, Los Angeles (external tank umbilical door centerline latch and actuator, umbilical door actuator, and umbilical door latch actuator); U.S. Bearing, Chatsworth, Calif. (external tank-orbiter spherical bearing); Bertea Corp., Irvine, Calif. (umbilical retractor actuator); Space Ordnance Systems Division, Trans Technology Corp., Saugus, Calif. (orbiter-external tank separation cartridge detonator assembly, 3/4-inch frangible nut orbiter-external tank aft separation, forward separation bolt pyro centering mechanism).

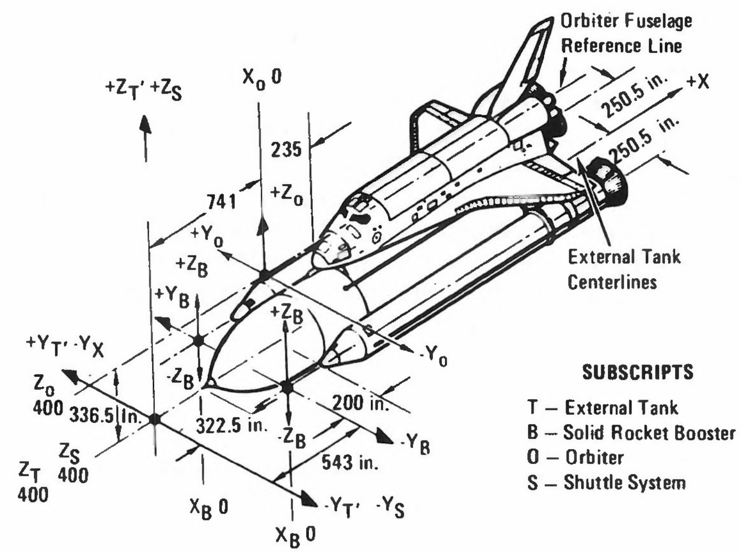
SPACE SHUTTLE COORDINATE SYSTEM

The Space Shuttle coordinate reference system is a means of locating specific points on the Shuttle. The system is measured in inches and decimal places: X_O designates the longitudinal (forward and aft) axis, Y_O the lateral (inboard and outboard) axis, and Z_O the vertical (up and down) axis. The subscript 0 indicates orbiter; similar reference systems are used for the external tank (T), solid rocket booster (B), and overall Space Shuttle System (S).

In each coordinate system, the X-axis zero point is located forward of the nose tip; that is, the orbiter nose tip location is 236 inches aft of the zero point (at X_O 236); the external tank nose cap tip location is at X_T 322.5, and the solid rocket booster (SRB) nose tip location is at X_B 200. In the orbiter, the horizontal X_O, Y_O reference plane is located at Z_O 400, which is 336.5 inches above the external tank horizontal X_T, Y_T reference plane located at Z_T 400. The solid rocket booster horizontal X_B, Y_B reference plane is located at Z_B 0 and coincident with the external tank horizontal plane at Z_T 400. The solid rocket booster central vertical X_B, Z_B planes are located at $+Y_S$ 250.5 and $-Y_S$ 250.5. Also, note that the orbiter, external tank, and Shuttle system center X, Z planes coincide.

From the X = 0 point, aft is positive, and forward is negative for all coordinate systems. Looking forward, each Shuttle element Y-axis point right of the center plane (starboard) is positive and left of center (port) is negative. The Z axis of each point within all elements is positive with Z = 0 located below the element, except for the SRB's, in which each Z-coordinate point below the SRB X_B, Y_B reference plane is negative and each point above that plane is positive.

The Shuttle system and Shuttle element coordinate systems are related as follows: The external tank X_T 0 point coincides with X_S 0, the SRB X_B 0 point is located 543 inches aft, and the orbiter Y_O, Z_O reference plane is 741 inches aft of X_S 0.



Coordinate Systems

ORBITER STRUCTURE

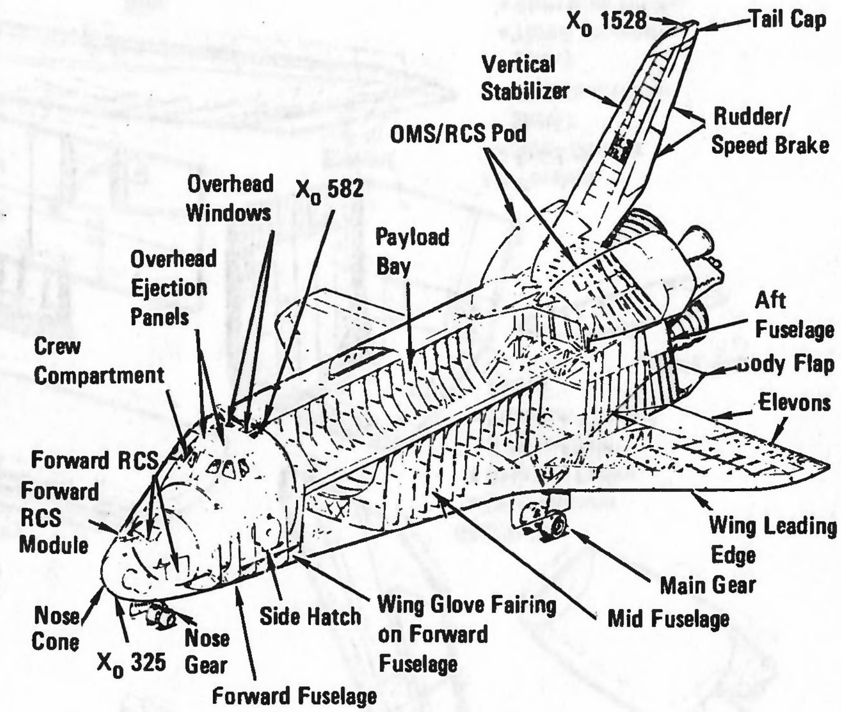
The orbiter structure is divided into major sections: the forward fuselage, which consists of upper and lower sections that fit clam-like around a pressurized crew compartment; wings; mid fuselage; payload bay doors; aft fuselage; and the vertical tail. The majority of the structures are of conventional aluminum construction protected by reusable surface insulation.

The forward fuselage structure is composed of 2024 aluminum alloy skin-stringer panels, frames, and bulkheads.

The crew compartment, which is supported within the forward fuselage at four attachment points, is welded to create a pressure-tight vessel. The three-level compartment has a side hatch for normal passage and a hatch from the airlock into the payload bay for extra-vehicular activity (EVA) and intra-vehicular activity (IVA).

The mid fuselage is an 18.28-meter (60-foot) section of primary load-carrying structure between the forward and aft fuselages. It includes the wing carry-through structure and the payload bay doors. The skins are integral-machined aluminum panels and aluminum honeycomb sandwich panels. The frames are constructed as a combination of aluminum panels with riveted or machined integral stiffeners and a truss structure center section. The upper half of the mid fuselage consists of structural payload bay doors hinged along the side and split at the top centerline. The doors are graphite epoxy frames and honeycomb panel construction.

The aft fuselage includes a truss-type internal structure of diffusion-bonded elements that transfers the main engine thrust loads to the mid fuselage and external tank. The aft fuselage external surface is of standard construction except for the removable orbital maneuvering system/reaction control system (OMS/RCS) pods. They are constructed of graphite epoxy skins

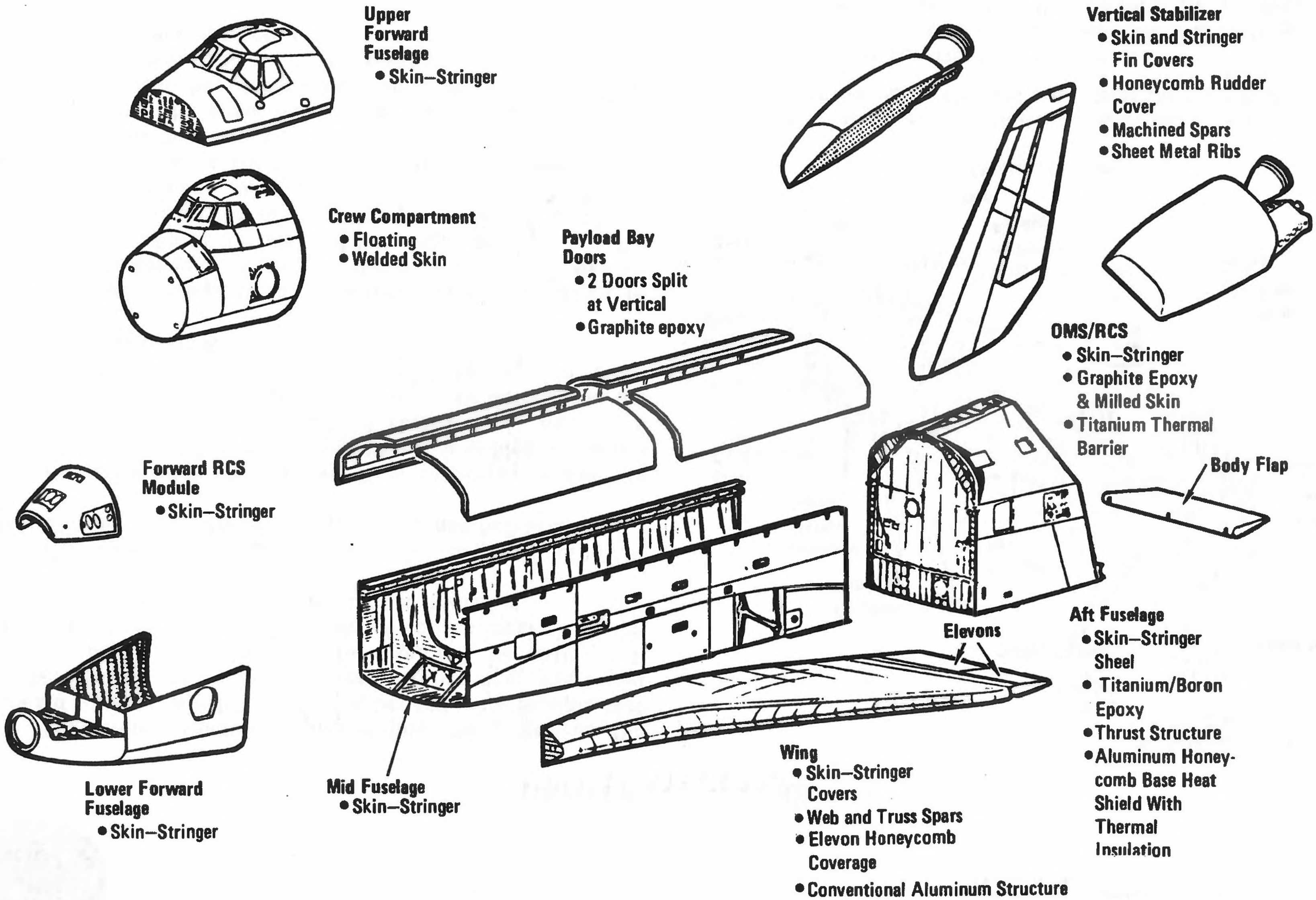


Orbiter Structure

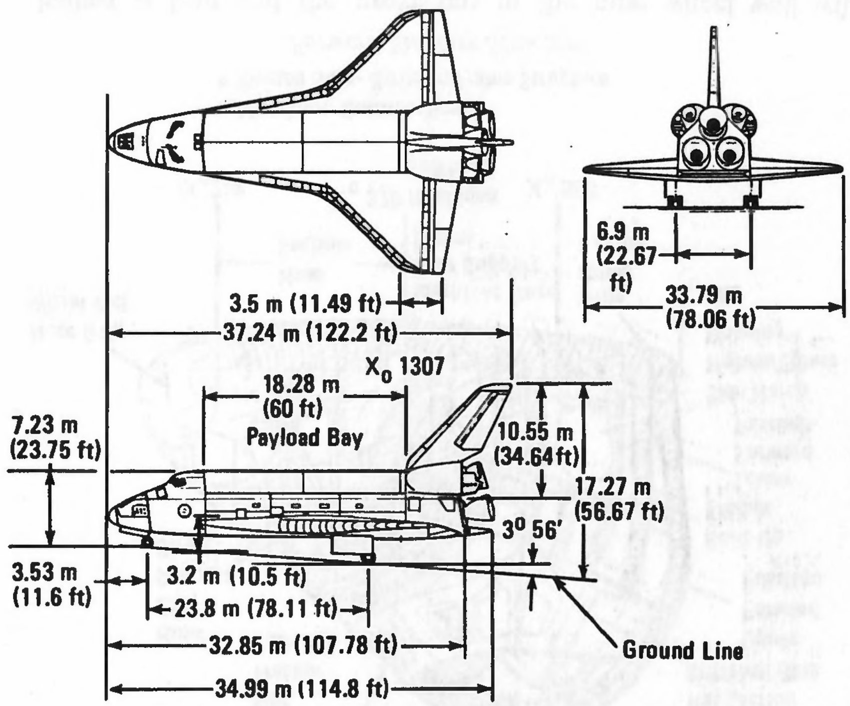
and frames. An aluminum bulkhead heat shield with reusable insulation at the rear of the orbiter protects the main engines.

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

The vertical tail, a conventional aluminum alloy structure, is a two-spar, multi-rib, integrally machined skin assembly. The tail is attached to the aft fuselage by bolted fittings at the two main



Orbiter Structural Elements



Note: All Dimension In Meters (Feet)

Minimum Ground Clearances	Meters (Ft)
Body Flap (Aft End)	3.67 (12.07)
Main Gear Door	0.86 (2.85)
Nose Gear Door	0.89 (2.95)
Wing Tip	3.63 (11.92)

Orbiter Dimensions

spars. The rudder/speed brake assembly is divided into upper and lower sections. Each is also split longitudinally and individually actuated to serve as both rudder and speed brake.

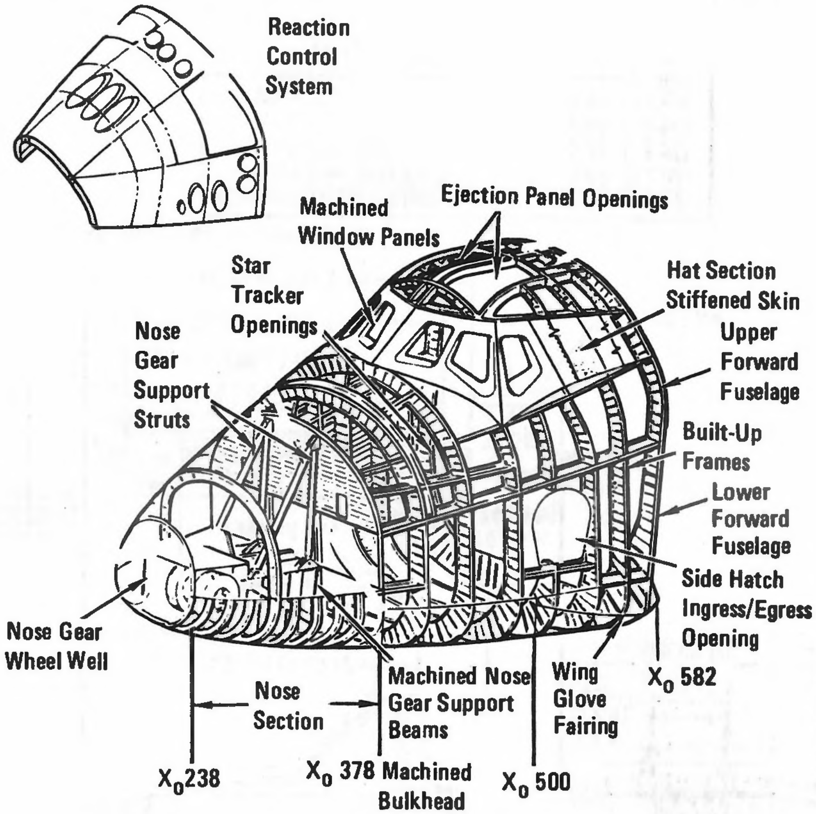
These major structural assemblies are mated and held together with rivets and bolts. The mid fuselage is joined to the forward and aft fuselage primarily with shear ties, with the mid fuselage overlapping the bulkhead caps at stations X_o582 and X_o1307. The wing is attached to the mid fuselage and aft fuselage primarily with shear ties, except in the area of the wing carry-through. There the upper panels are attached with tension bolts. The vertical tail is attached to the aft fuselage with bolts that work in both shear and tension. The body flap, which has aluminum honeycomb covers, is attached to the aft lower fuselage by four rotary actuators.

FORWARD FUSELAGE

The forward fuselage consists of the upper and lower fuselages. The forward fuselage houses the crew compartment and supports the forward RCS module, nose cap, nose gear wheel well, nose gear, and nose gear doors.

The forward fuselage is constructed of conventional 2024 aluminum alloy skin-stringer panels, frames, and bulkheads. The panels are single curvature and stretch-formed skins with riveted stringers spaced 7.62 to 12.7 centimeters (3 to 5 inches) apart. The frames are riveted to skin-stringer panels. The major frame spacing is 76.2 to 91.44 centimeters (30 to 36 inches). The Y_o378 forward bulkhead is constructed of flat aluminum and formed sections riveted and bolted together (upper) and a machined section (lower). The bulkhead provides the interface fitting for the nose section.

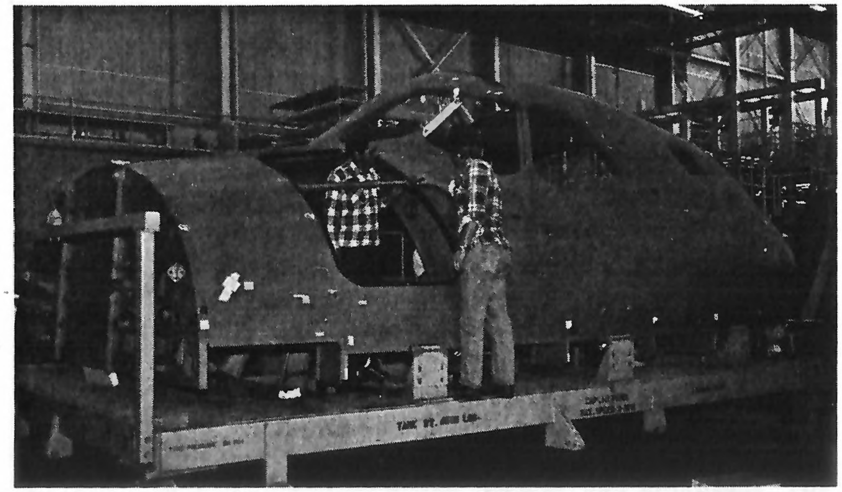
The nose section contains large machined beams and struts. The structure for the nose landing gear wheel well consists of two support beams, two upper closeout webs, drag link support struts, nose landing gear strut and actuator attachment fittings, and the nose landing gear door fittings. The two (left and right) landing gear doors are attached by hinge fittings in the nose section. The doors are constructed of aluminum alloy honeycomb. The left door is wider than the right, although both are the same length. Each door has an up-latch fitting at the forward and aft ends to lock the doors closed when the gear is retracted. Each door has a pressure seal in addition to a thermal barrier. Ballast provisions are provided in the nose wheel well and on the X_o378 bulkhead for weight and center of gravity control. The

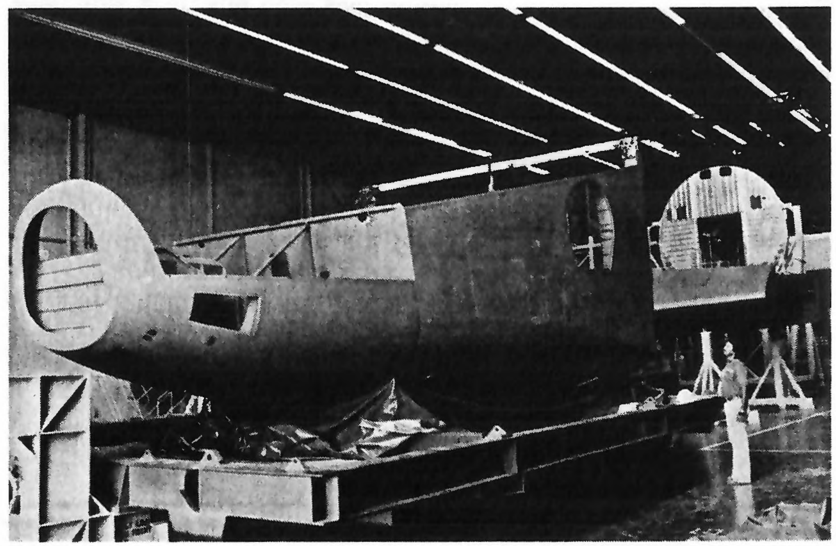


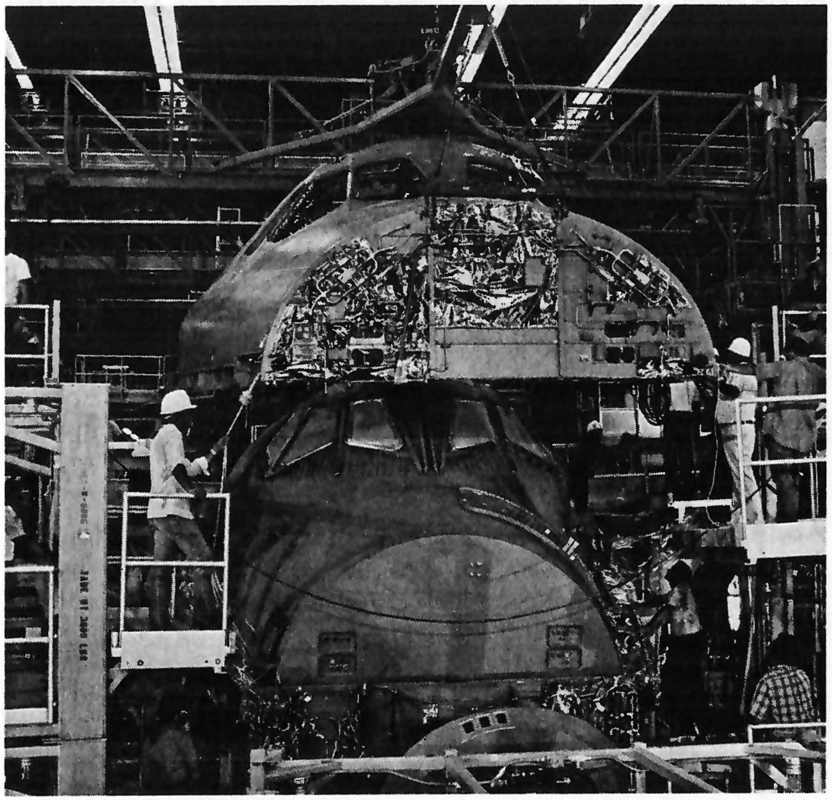
Forward Fuselage Structure

ballast is lead and the provisions in the nose wheel well will accomodate 612.36 kilograms (1,350 pounds) and the $X_0$378 bulkhead will accomodate a maximum of 1,206.57 kilograms (2,660 pounds).

The forward fuselage carries the basic body bending loads (a tendency to change the radius of a curvature of the body) and reacts nose landing gear loads.


Upper Forward Fuselage


Lower Forward Fuselage



Upper Forward Fuselage Being Mated to Lower Forward Fuselage

The forward fuselage will be covered with the reusable thermal protection system, except for the six windshields, two overhead windows, side hatch window, and some of the area around the forward RCS engines. The nose cap is also a reusable thermal protection system. It is constructed of reinforced carbon-carbon and will have thermal barriers at the nose cap-structure interface.

In the forward fuselage skin are structural provisions for installation of antennas, the deployable air data probes, and the door eyelet openings for the two star trackers. An opening in the upper forward fuselage is required for star tracker viewing. The opening has a door for environmental control.

The forward orbiter/external tank attach fitting is provided by the X_o378 bulkhead and skin panel structure aft of the nose gear wheel well. Purge and vent control is provided with installation of flexible boots between the forward fuselage and crew compartment around the windshield windows, overhead observation window, crew hatch window, and star tracker opening. Isolation between the forward fuselage and payload bay is provided by a flexible membrane between the forward fuselage and crew compartment at X_o582.

The six forward outer-pane windshields are installed on the forward fuselage and are described in the section on windows. The window structural frames in the forward fuselage are five-axis machined parts.

The overhead ejection panels in the initial development flights for the commander and pilot on the forward fuselage operate in concert with the crew compartment ejection panels. They are described under the crew compartment.

The forward RCS module is constructed of conventional 2024 aluminum alloy skin-stringer panels and frames. The panels are composed of single-curvature and stretch-formed skins with riveted stringers. The frames are riveted to the skin-stringer panels. The forward reaction control system module is secured to the forward fuselage nose section and forward bulkhead of the forward fuselage with 16 fasteners, which permit installation and removal of the forward RCS module. The components of the forward RCS are mounted and attached to the forward RCS module. The forward RCS module will be covered with reusable thermal protection in addition to thermal barriers installed around the forward RCS module and engine interface and forward RCS module interface-attachment area to the forward fuselage.

The forward fuselage and forward RCS module are built by Rockwell's Space Transportation System Development and Production Division, Downey, CA.

CREW COMPARTMENT

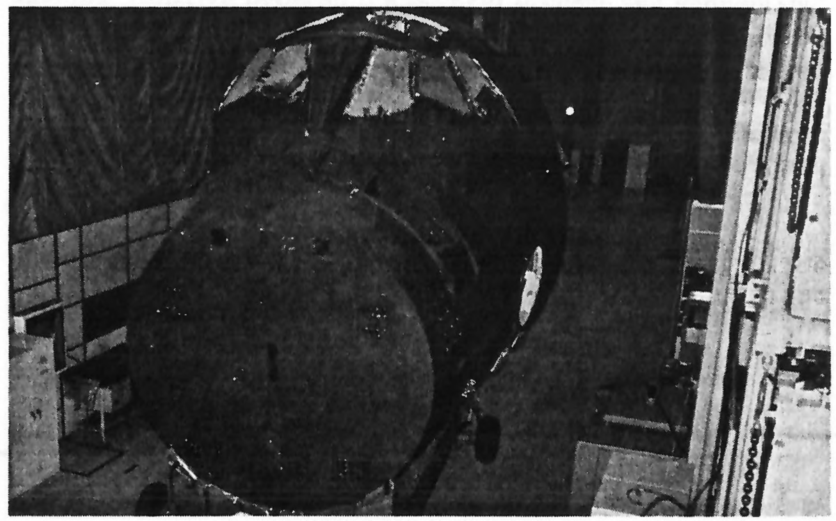
The three-level crew compartment is constructed of 2219 aluminum alloy plate with integral stiffening stringers and internal framing welded together to create a pressure-tight vessel. The compartment has a side hatch for normal ingress and egress, a hatch into the airlock from the mid deck, and a hatch through the aft bulkhead into the payload bay for extra-vehicular activity, intra-vehicular activity, and payload bay access.

Redundant pressure window panes are provided in the windshield and in the overhead, aft viewing, and side hatch windows; they are described in the window section. The approximately 300 penetrations in the pressure shell are sealed with plates and fittings. A large removable panel in the aft bulkhead provides access to the crew compartment interior during initial fabrication and assembly and provides for airlock installation and removal. Equipment supported in the compartment includes the environmental control life and support system (ECLSS), avionics, guidance and navigation (G&N), displays and controls (D&C), navigation star tracker base, and crew accommodations for sleeping, waste management, seats, and galley.

The crew compartment is supported within the forward fuselage at only four attach points to minimize the thermal conductivity between them. The two major attach points are at the aft end of the crew compartment at the flight deck section floor level. The vertical load reaction link is on the centerline of the forward bulkhead. The lateral load reaction links are on the lower segment of the aft bulkhead.

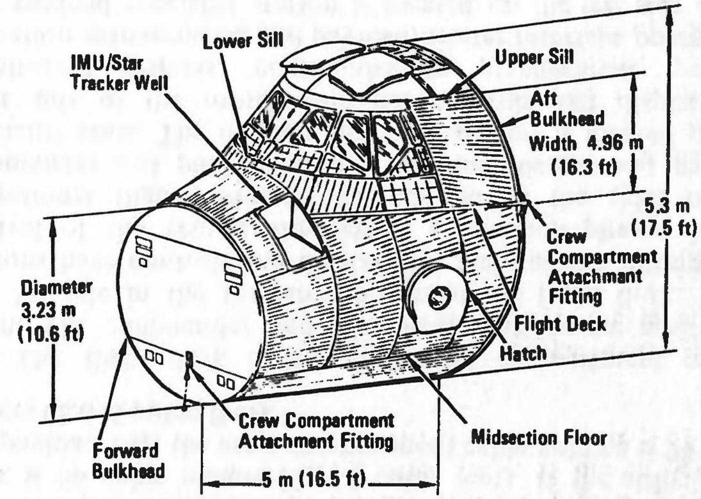
The compartment is configured to accommodate a crew of three on the flight deck with an additional seat for one more on

Forward RCS Module

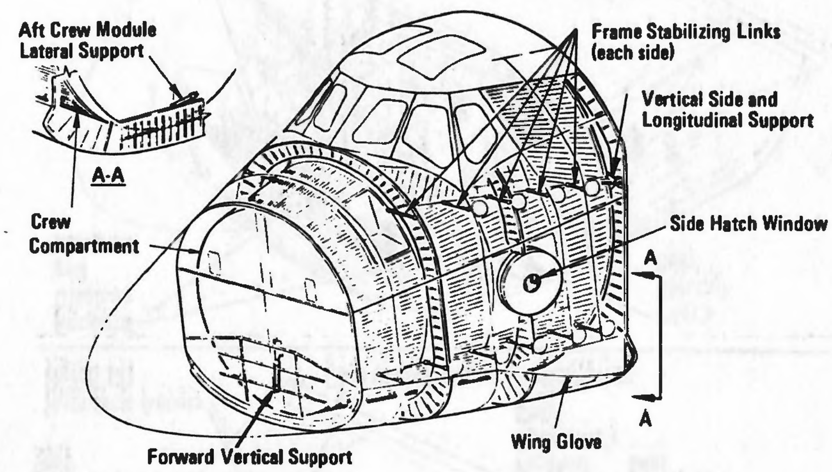


Crew Compartment

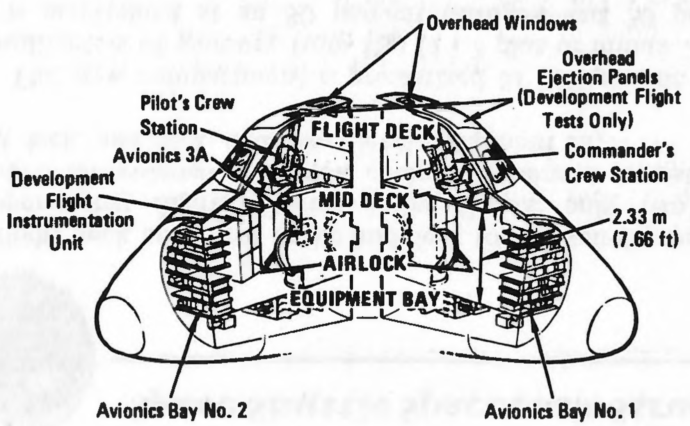
Space Shuttle Spacecraft Structures



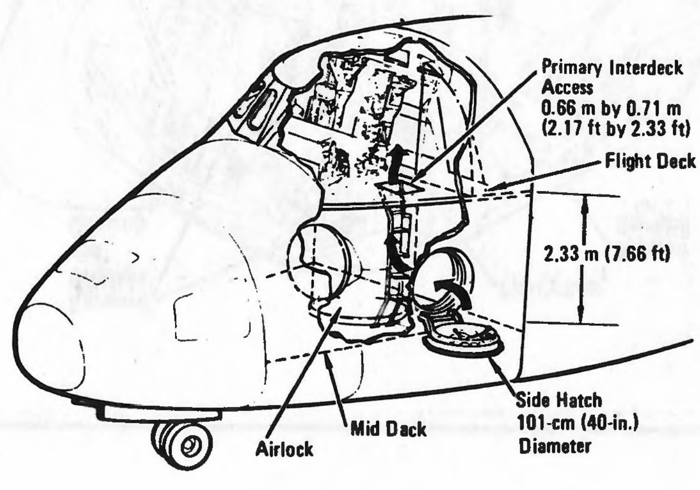
Crew Compartment



Crew Compartment-Forward Fuselage Interface



Crew Compartment and Arrangement



Normal Crew Entrance and Exit

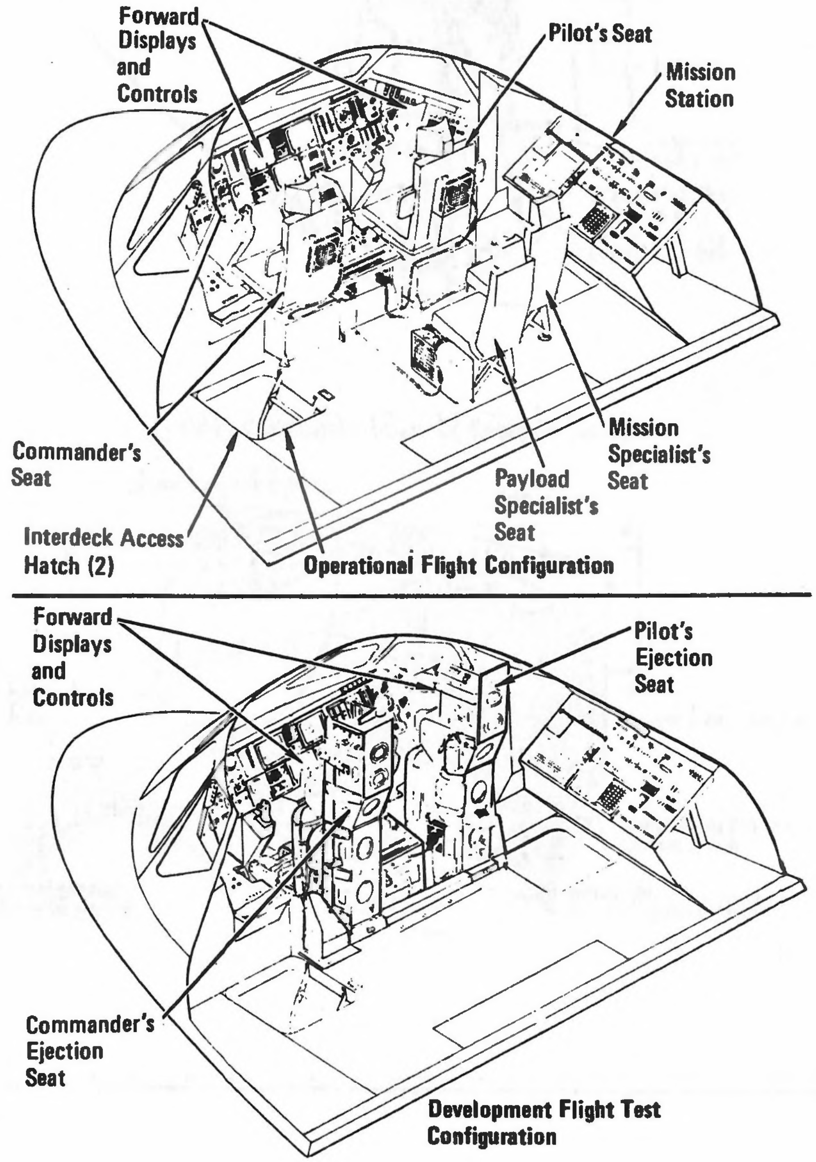
the flight deck and three in the mid deck for a seven-day mission in operational flights. During initial flights, only two crew members are assigned. The crew cabin arrangement consists of a flight deck, mid deck, and lower level equipment bay.

The crew compartment is pressurized to 760 plus or minus 10 millimeters of mercury (mm Hg) (14.7 plus or minus .2 psia) and is maintained at an 80 percent nitrogen and 20 percent oxygen composition by the ECLSS, which provides a shirtsleeve environment for the flight crew. The crew compartment is designed for 828 mmHg (16 psia).

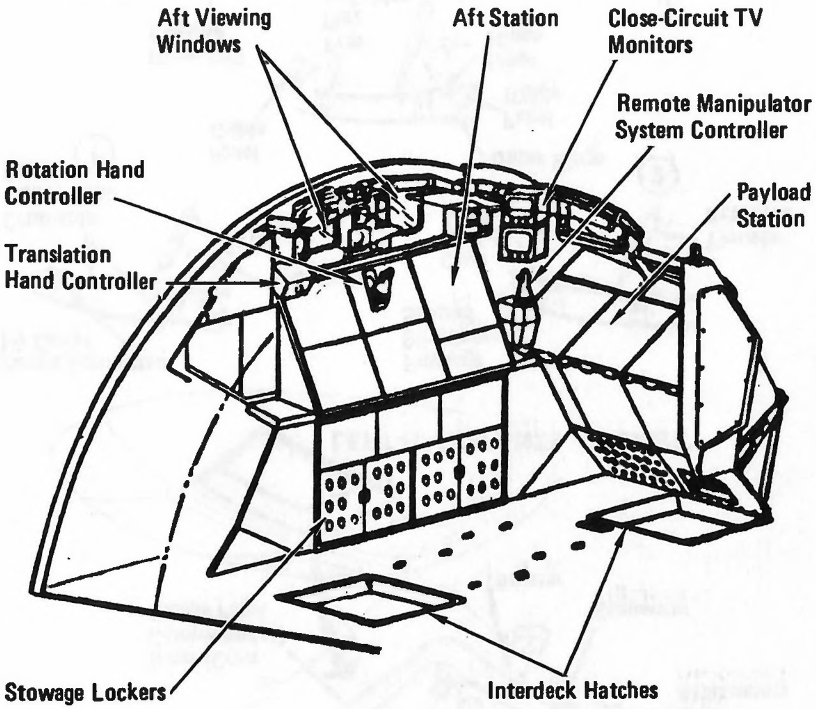
The crew compartment volume with the airlock in the mid deck is 66 cubic meters (2,325 cubic feet). If the airlock is in the payload bay, the crew compartment cabin volume is 74 cubic meters (2,625 cubic feet).

The flight deck is the uppermost compartment of the cabin. The commander and pilot work stations are positioned side by side in the forward portion of the flight deck. These stations have controls and displays for maintaining autonomous control of the vehicle throughout all mission phases. In the operational flights, directly behind and to the sides of the commander and pilot centerline are the mission and payload specialist seats. The mission specialist station is located on the right side of the orbiter and has controls and displays for monitoring systems, communication management, payload operation management, and payload/orbiter interface operations. The payload specialist station is located on the left side of the orbiter and contains controls and displays.

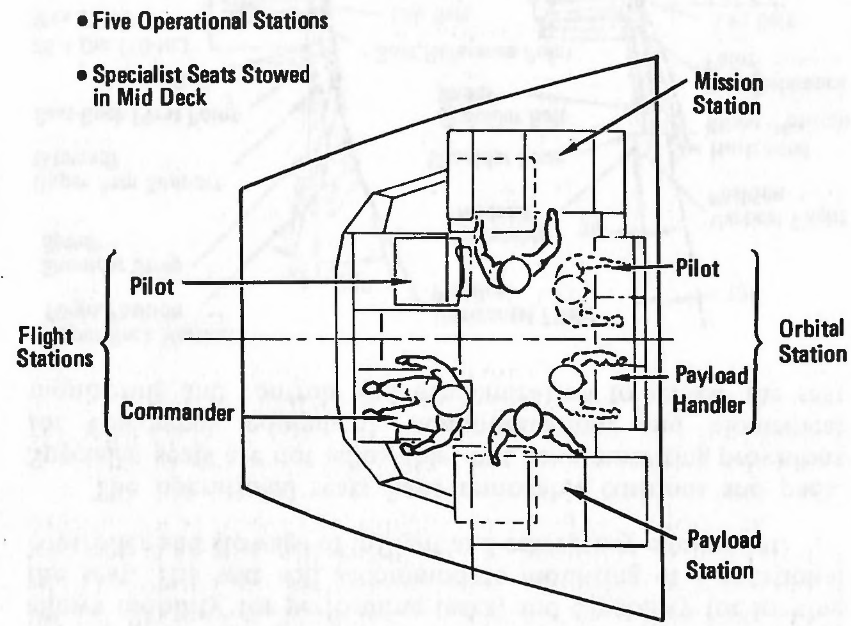
Between these two aft stations are the on-orbit pilot and payload handling stations. These rearward facing stations are unoccupied during launch. Orbital operation visibility is provided by overhead and aft viewing windows. The orbital stations contain displays and controls for executing attitude or translational maneuvers for terminal-phase rendezvous, station-keeping and docking, and payload deployment and retrieval. The



Crew Compartment Flight Deck



Aft Flight Deck Station



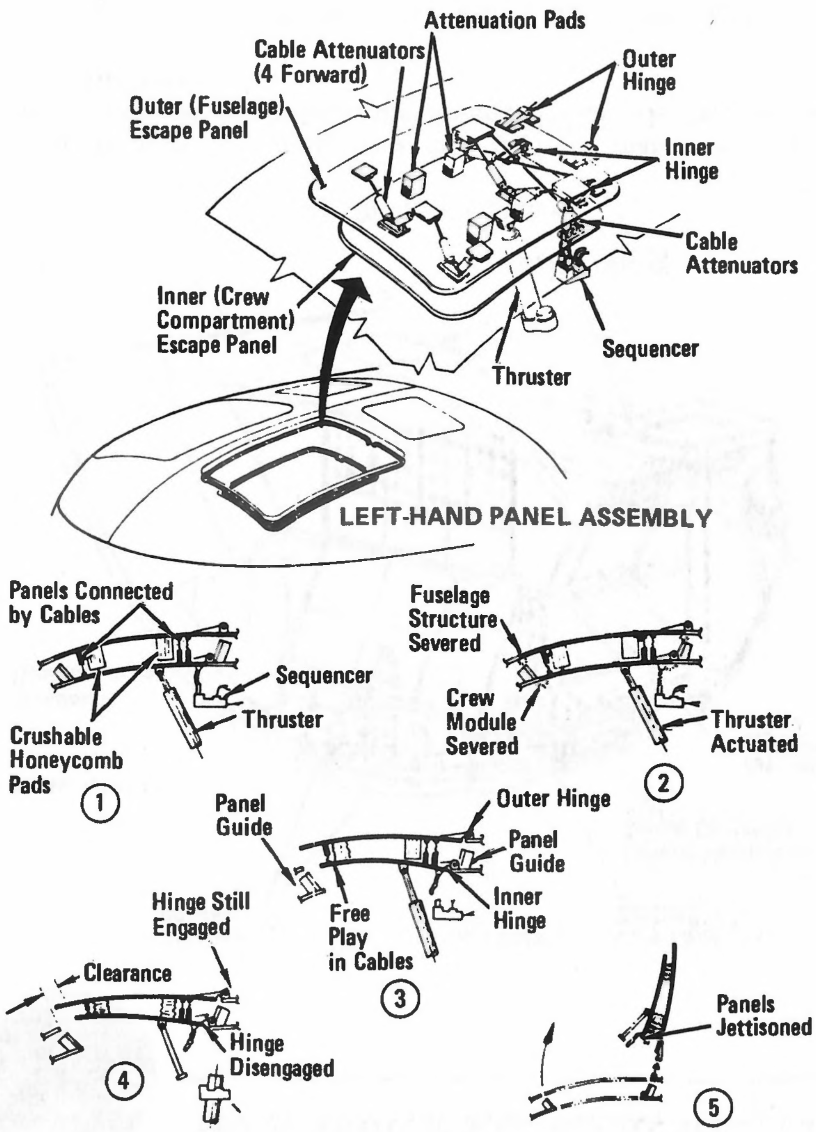
Flight Deck Station – Orbital Operations

pilot is normally designated the payload manipulator and docking operator; however, the commander can also accomplish these functions.

In the early development flights an escape system is provided for the commander and pilot. The system consists of the seat, ejection/escape panel, energy transmission and system sequencing, and panel jettison actuation for non-ejection ground entrance and exit.

Above the commander and pilot ejection seats for the development flight tests are overhead ejection panels. The panels provide the commander and pilot with an emergency exit or the capability of ejecting in the seat through the overhead panel area after ejection of the overhead panel. The ejection panel assembly consists of crushable honeycomb pads for shock attenuation, cable retention between the crew module and forward fuselage panel with breakaway hinges, and an inner-outer severance system and thruster to propel the severed panel clear of the orbiter.

For the development flight tests, the commander and pilot's seats have electrical adjustments of fore, aft, up, and down for vertical launch and horizontal flight. Both seats have provisions for stowage of inflight and emergency equipment. The seats' ejection components consist of seat assembly, seat structure, crew restraint, escape controls, seat guidance rails, and attachments to cabin structure and the escape propulsion thruster. The

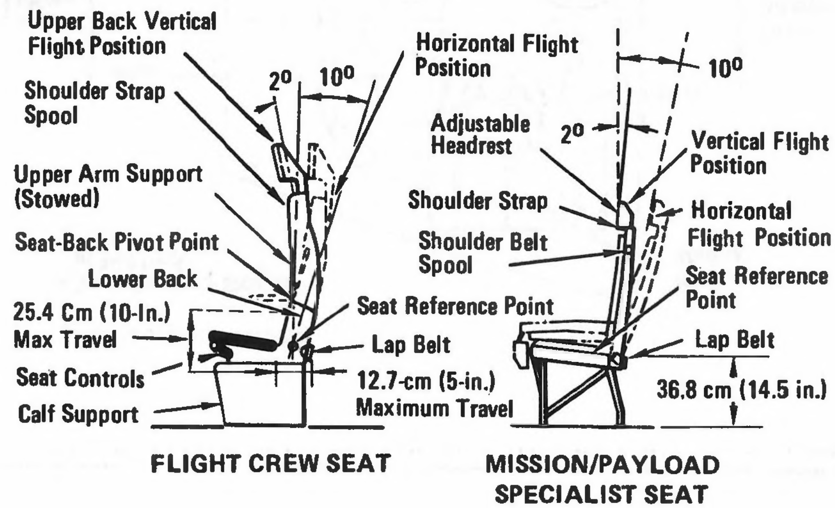


Overhead Ejection Panels - Development Flight Tests

escape propulsion thruster provides the energy to propel the seat from its normal flight position to its point of separation from the orbiter and sustain and stabilize the motion of the escapee for orbiter clearance, stabilization/deceleration, seat-crew member separation, recovery, and survival systems.

In the later flights, the ejection seats and components are replaced with operational seats. The seat system will have controls for electrically adjusting the seat fore, aft, up, and down and for support during vertical launch and horizontal flight and tilt for back angle positioning. It also has an inertia reel, which allows mobility for performing tasks, and capability for locking the seat. The seat will accommodate mounting of a rotational controller and stowage of inflight and emergency equipment.

The operational seats have removable cushions and pads. Specialist seats are not adjustable, but have mounting provisions for emergency equipment, communication, and biomedical monitoring and controls and mechanization to release the seat



Operational Flight Seats

from the flight deck for stowage during zero-g orbital flight. The specialists' seats will have restraint devices and controls to lock and unlock the seat back for tilt change and removable cushions and pads.

In the operational configuration, the left-hand overhead window will provide an emergency exit route.

Directly beneath the flight deck is the mid deck. Access to the mid deck-flight deck is by two interdeck openings 66 by 71 centimeters (26 by 28 inches). A ladder attached to the left interdeck access allows easy passage in one-g conditions. The mid deck provides crew accommodations. Three avionics equipment bays are also located here. The two forward avionics bays utilize the complete width of the cabin and extend into the mid deck 99 centimeters (39 inches) from the forward bulkhead. The aft bay extends into the mid deck 99 centimeters (39 inches) from the aft bulkhead on the right side of the airlock. Just forward of the waste management system is the side hatch.

The side hatch in the mid deck is used for normal crew entrance/exit and may be operated from within the crew cabin mid deck or externally. The pressure side hatch is assembled to the crew cabin tunnel through hinges, torque tube, and support fittings. The side hatch opens outwardly 90 degrees down with the orbiter horizontal or 90 degrees sideways with the orbiter vertical. The side hatch is 101 centimeters (40 inches) in diameter. It has a 25.4-centimeter-diameter (10-inch) clear-view window in the center of the hatch. The window consists of three panes of glass. The side hatch seal provides a pressure seal that is compressed by the side hatch latch mechanisms when the hatch is locked closed. A thermal barrier of Inconel wire mesh spring with a ceramic fiber braided sleeve is installed between the reusable surface insulation tiles on the forward fuselage and the side hatch.

In the initial development flights, an instrumentation unit is installed on the mid deck in addition to two potable water tanks and two sleep stations.

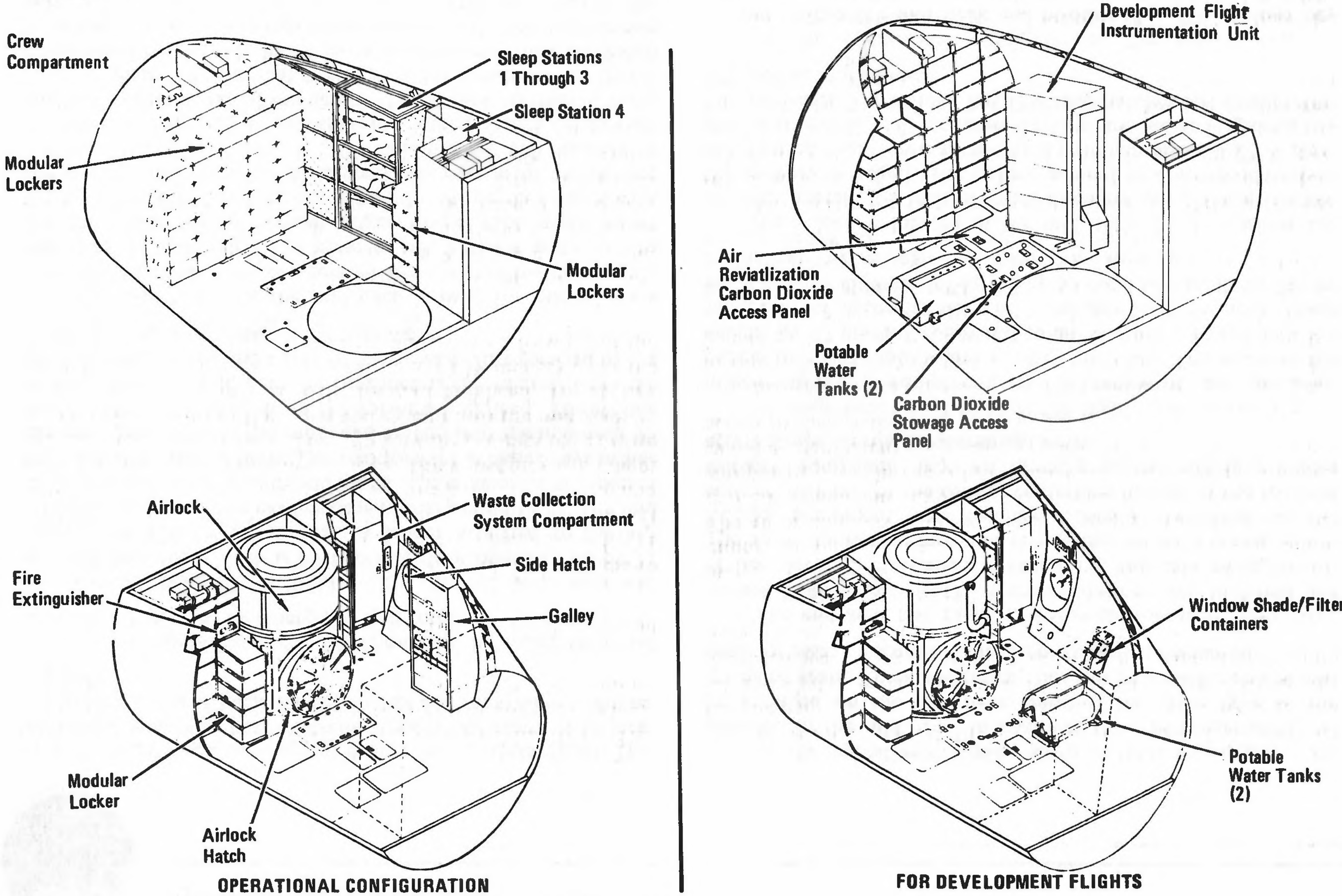
In the operational flights, sleep stations and a galley are installed in the mid deck; in addition, three passenger seats can be installed on the mid deck floor of the same type as the specialist seats. Three additional seats can be installed on the mid deck for rescue missions if the sleep stations are removed.

The mid deck also provides a stowage volume of 3.96 cubic meters (140 cubic feet). Accommodations are included for dining, sleeping, maintenance, exercising, and data management. The floor contains removable panels that provide access to the ECLSS equipment and stowage beneath the floor in the equipment bay. On the orbiter centerline, just aft of the forward avionics equipment bay, an opening in the ceiling provides access to the inertial measurement units.

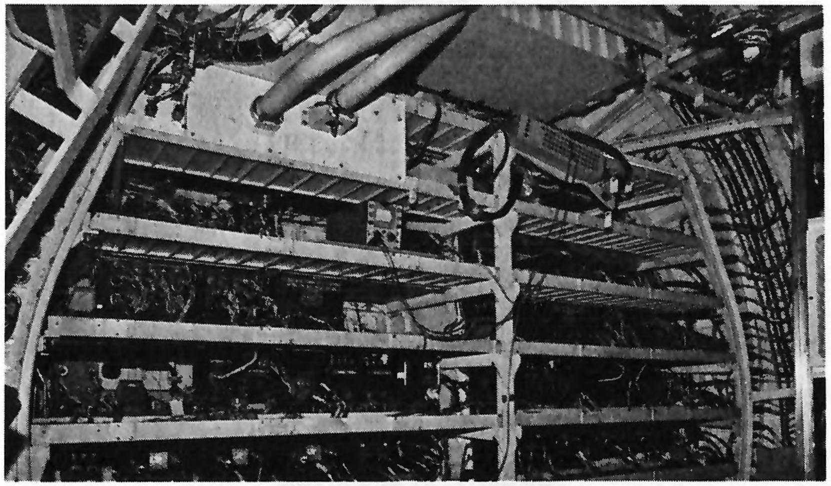
Stowage boxes used to store the flight crew's personal gear, mission-necessary equipment, and experiments use sandwich panels of Kevlar/epoxy and nonmetallic core. This reduced the weight by 83 percent compared to all aluminum boxes. This is a reduction of approximately 68 kilograms (150 pounds). There are 42 identical boxes 27 by 45 by 53 centimeters (11 by 18 by 21 inches) which can be used for stowage.

An airlock is located in the mid deck. Its dimensions are 160 centimeters (63 inches) inside diameter and 210 centimeters (83 inches) in height. The airlock is sized to accommodate two fully suited flight crew members simultaneously. All EVA gear, checkout panel, and recharge stations are located against the internal walls. The airlock has D-shaped hatches 101 centimeters (40 inches) in diameter.

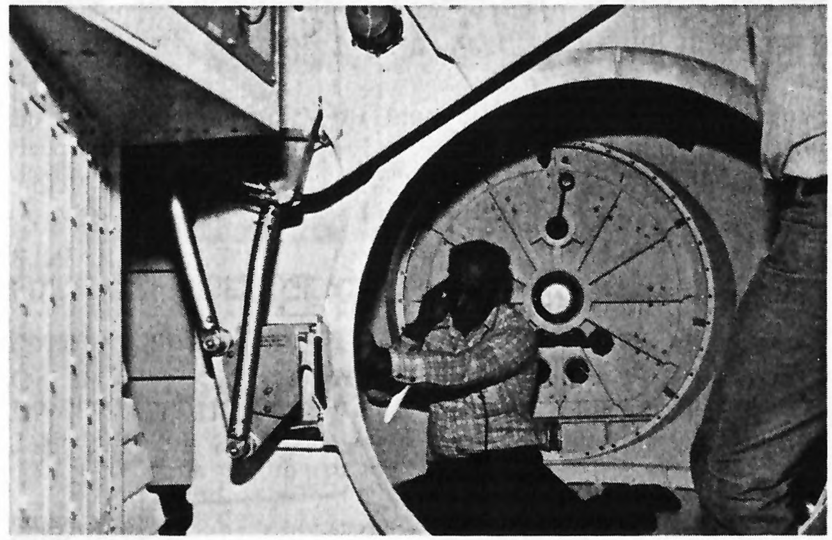
In the operational configuration, the airlock may be removed from the mid deck and positioned in the payload bay area to provide full-time shirtsleeve access to a Spacelab in the payload bay and EVA capability without depressurizing either the Spacelab or the crew module. In this configuration, a tunnel adapter with the airlock mounted atop the adapter will provide EVA capabilities. When the airlock is in the payload bay, insulation is installed on the airlock exterior for protection from



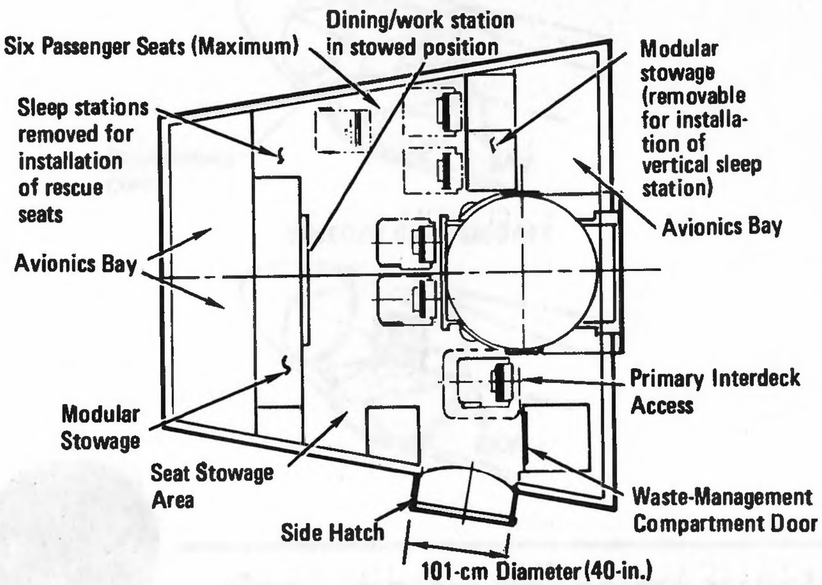
Crew Compartment Mid Deck



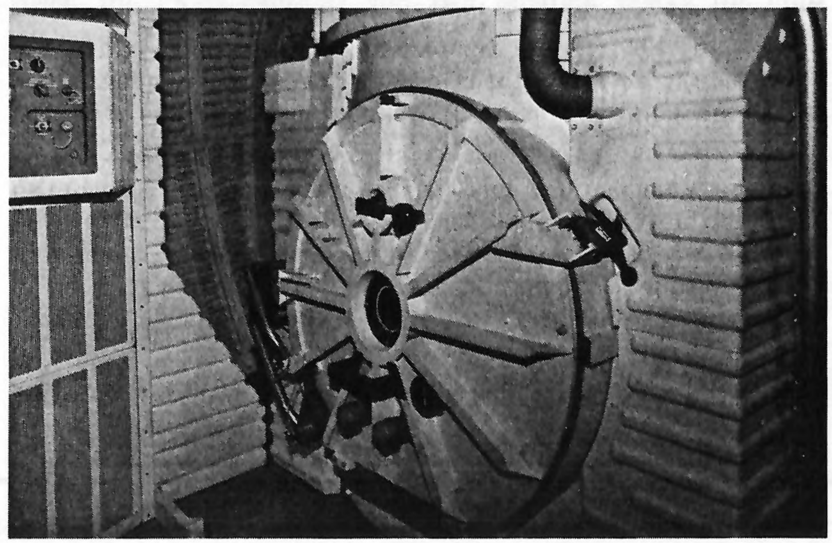
Avionics Bays 1 and 2



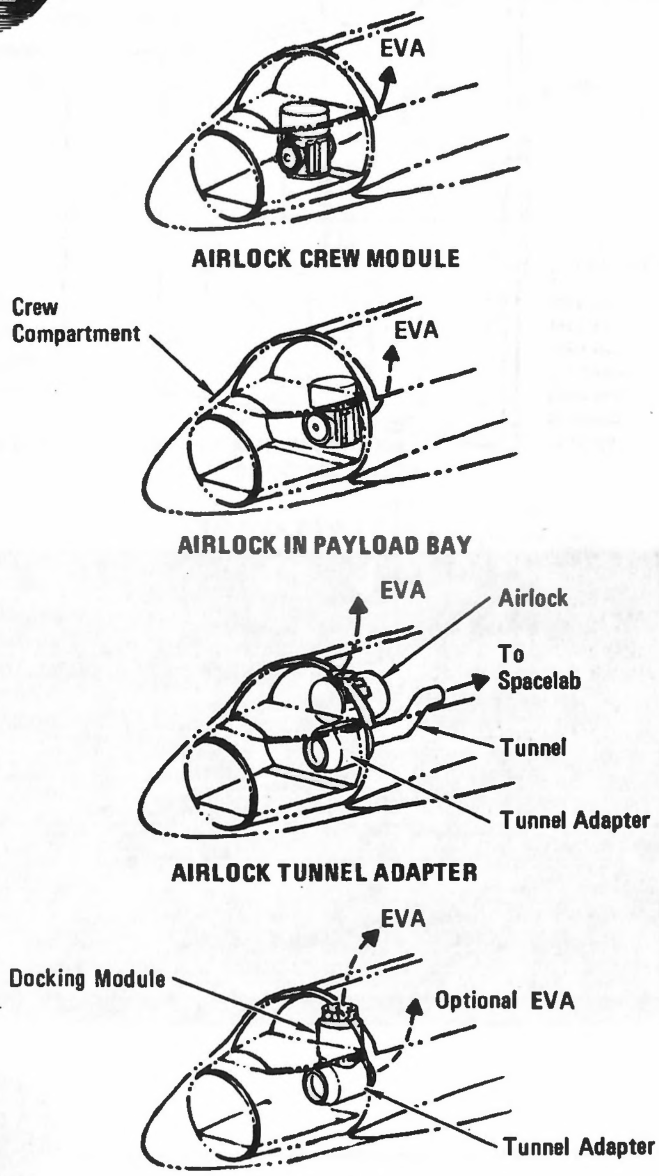
Airlock



Mid-Deck-Rescue Configuration



Airlock



Airlock Configuration

the extreme temperatures of space. For missions requiring docking of two vehicles, a docking module can be substituted for the airlock and installed on the tunnel adapter. The docking module is extendible and provides an airlock for one crew member for EVA when retracted and for two crew members when extended.

The equipment bay houses the major components of the waste management and atmospheric revitalization systems such as pumps, fans, lithium hydroxide (LiOH) absorbers, heat exchangers, and ducting. This compartment provides stowage space for LiOH canisters and five separate spaces for crew equipment stowage with a volume of 0.8 cubic meters (29.93 cubic feet).

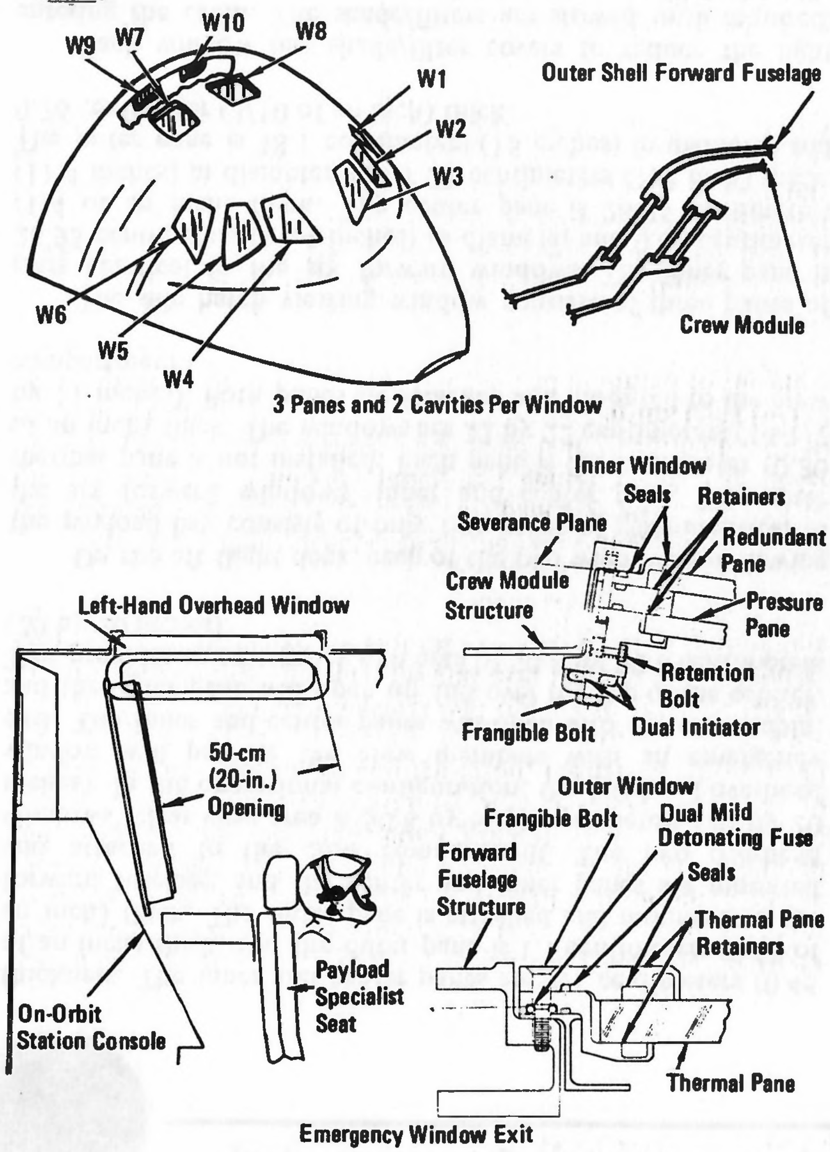
The crew compartment is built by Rockwell's Space Transportation System Development and Production Division, Downey, CA. The ejection set contractor is Lockheed-California Co., Burbank, CA. The operational crew seat contractor is AMI, Colorado Springs,CO.

FORWARD FUSELAGE AND CREW COMPARTMENT WINDOWS

The orbiter windows provide visibility for entry, landing, and on-orbit operations. For atmospheric flight, the flight crew needs forward, left, and right viewing areas. On-orbit mission phases require visibility for rendezvous, docking, and payload handling.

The six windows located at the forward flight deck commander and pilot stations provide the forward, left, and right viewing. The two overhead windows and two payload viewing windows at the aft station location on the flight deck provide rendezvous, docking, and payload viewing. A window in the mid deck side hatch is for viewing by a crew person from that position.

The six planform-shaped forward windows are the largest pieces of glass ever produced in the optical quality for see-



Emergency Window Exit – Operational Flights

through viewing. Each consists of three individual panes. The innermost pane is constructed of tempered aluminosilicate glass to withstand the crew compartment pressure. It is 1.58 centimeters (5/8 of an inch) thick. Aluminosilicate glass is a low expansion glass that can be tempered to provide maximum mechanical strength. The exterior of this pane, called a pressure pane, is coated with a red reflector coating to reflect the infrared (heat portion) rays while transmitting the visible spectrum.

The center pane is constructed of low expansion fused silica glass due to its high optical quality and excellent thermal shock resistance. This pane is 3.3 centimeters (1.3 inches) thick. The exterior and interior are coated with a high efficiency, anti-reflection coating to improve visible light transmission. This pane is redundant to the pressure pane. These windows withstand a proof pressure of 445,050 mm Hg (8600 psi) at 115°C (240°F) and 0.017 relative humidity. This pane is also a redundant pane to the outer thermal pane.

The outer pane's exterior is of the same material as the center pane and is 1.58 centimeters (5/8 of an inch) thick. The exterior is uncoated, but the interior is coated with the high efficiency anti-reflective coating. The outer surface withstands approximately 482°C (900°F) and the interior surface withstands approximately 426°C (800°F).

Each of the forward six windows' outer panes measures 106.68 centimeters (42 inches) diagonally, and the center and inner panes each measure 88.9 centimeters (35 inches) diagonally. The outer panes of the forward six windows are mounted and attached to the forward fuselage. The center and inner panes are mounted and attached to the crew compartment. Redundant seals are employed for each window. No sealing/bonding compounds are used.

The two overhead windows at the flight deck aft station are identical in construction to the six forward windows except for thickness. The inner and center panes are 1.1 centimeters (0.45 of an inch) thick, and the outer pane is 1.7 centimeters (0.68 of an inch) thick. The outer pane is attached and mounted to the forward fuselage, and the center and inner panes are mounted and attached to the crew compartment. The two overhead windows' clear view area is 50.8 by 50.8 centimeters (20 by 20 inches). In the operational configuration, the left-hand overhead window will provide the crew members with an emergency exit. The inner and center panes will open into the crew cabin, and the outer pane will open up and over the top of the orbiter. This provides an emergency exit area of 50.8 by 50.8 centimeters (20 by 20 inches).

On the aft flight deck, each of the two windows for viewing the payload bay consists of only two panes of glass identical to the six forward windows' inner and center pane. The outer thermal pane is not installed. Each pane is 0.7 centimeter (0.30 of an inch) thick. The windows are 37 by 27 centimeters (14-1/2 by 11 inches). Both panes are attached and mounted to the crew compartment.

The side hatch viewing window consists of three panes of glass identical to the six forward windows. The inner pane is 28.95 centimeters (11.4 inches) in diameter and 0.63 centimeter (1/4 of an inch) thick. The center pane is 28.95 centimeters (11.4 inches) in diameter and 1.27 centimeters (1/2 inch) thick. The outer pane is 38.1 centimeters (15 inches) in diameter and 0.76 centimeter (3/10 of an inch) thick.

Each window has shade/filter covers to reduce the light entering the cabin. The shade/filters are stowed until required. Attachment mechanisms and devices are provided for the installation at each window on the flight deck and the mid deck side hatch window.

Contractor for the windows is Corning Glass Co., Corning, NY.

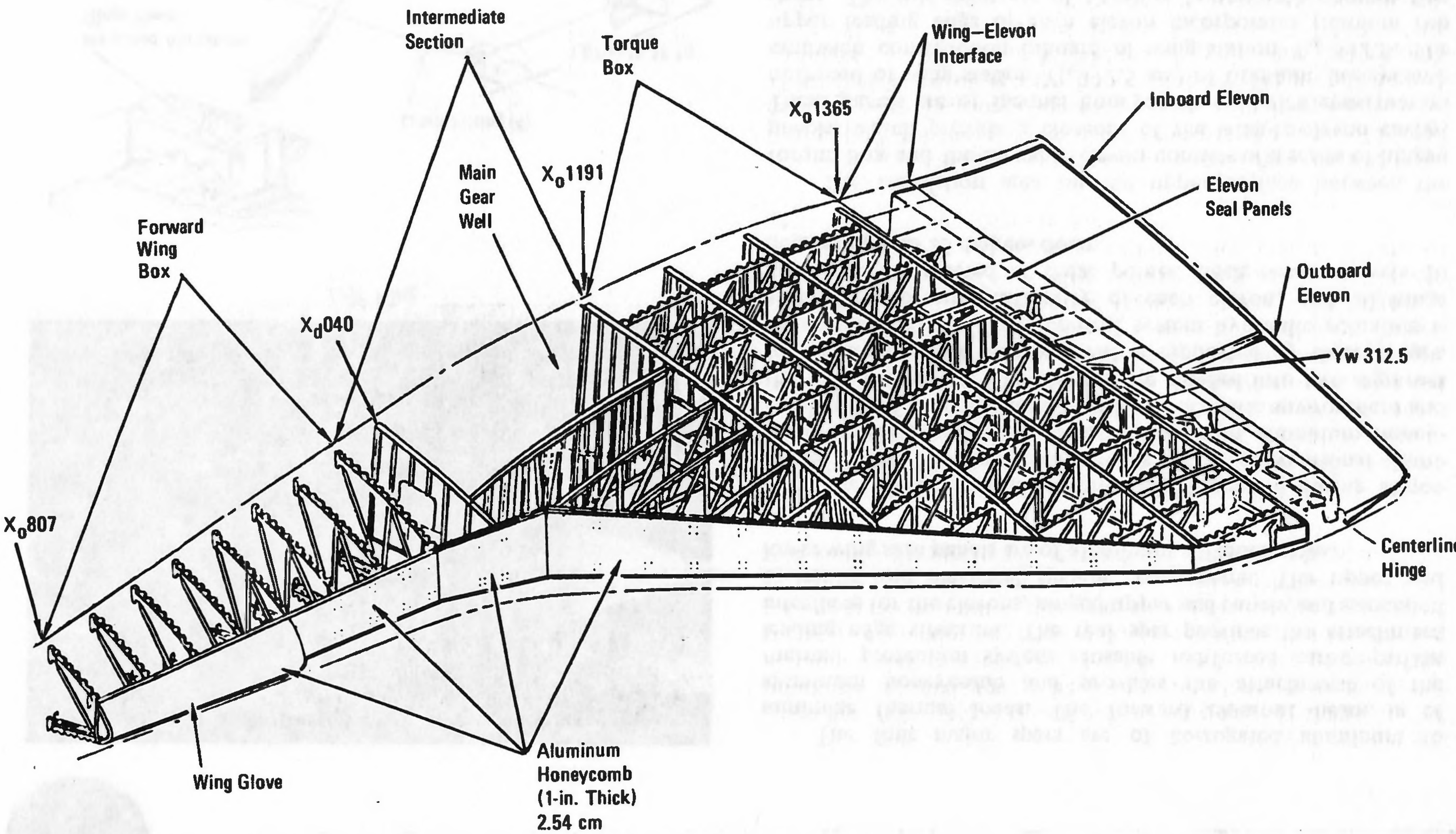
WING

The wing is an aerodynamic lifting surface that provides conventional lift and control for the orbiter. The left and right wings consist of the wing glove, the intermediate section which includes the main landing gear well, the torque box, the forward spar for the mounting of the reusable reinforced carbon-carbon leading edge structure thermal protection system, the wing/elevon interface, the elevon seal panels, and the elevons.

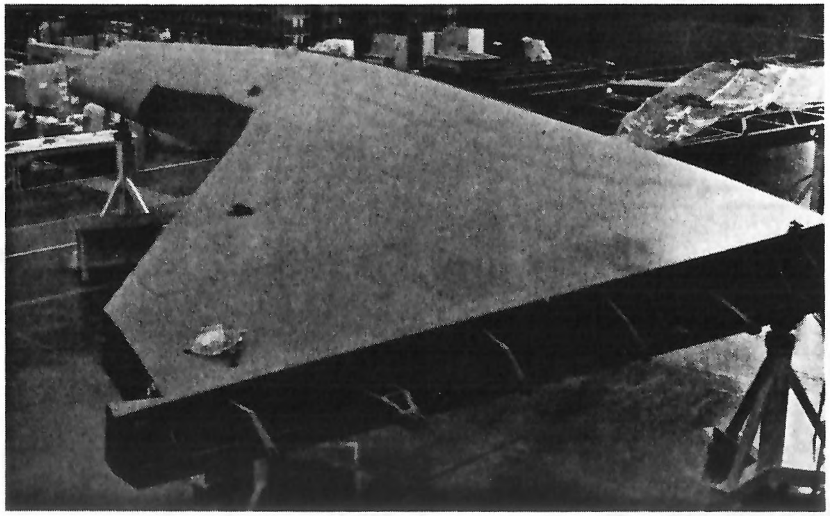
The wing is constructed of conventional aluminum alloy with a multi-rib and spar arrangement with skin-stringer-stiffened covers or honeycomb skin covers. Each wing is approximately 18.28 meters (60 feet) long at the fuselage intersection, with a maximum thickness of 1.52 meters (5 feet).

The forward wing box is an extension of the basic wing that aerodynamically blends the wing leading edge into the mid fuselage wing glove. The forward wing box is of a conventional design of aluminum multi-ribs, aluminum tubes, and tubular struts. The upper and lower wing skin panels are stiffened aluminum. The leading edge spar is of aluminum honeycomb panel construction.

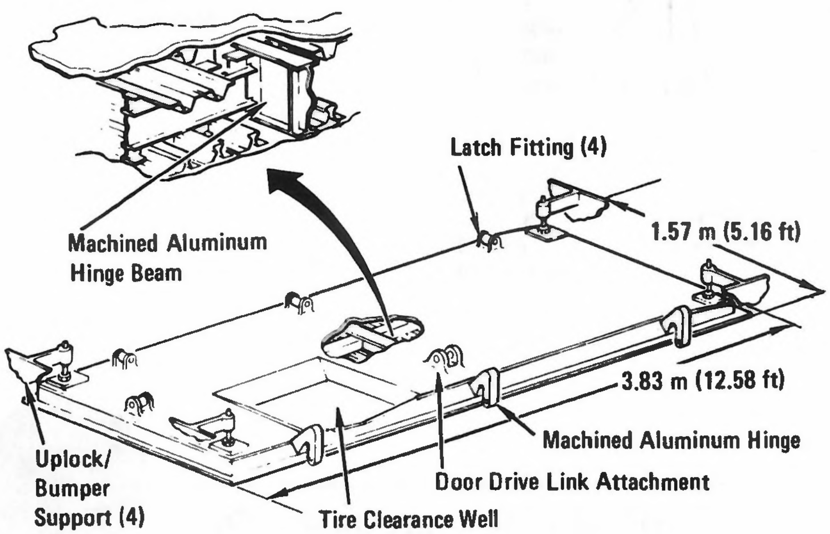
The intermediate wing section consists of the conventional aluminum multi-ribs and aluminum tubes. The upper and lower skin covers are of aluminum honeycomb. A portion of the lower wing surface skin panel is made up of the main landing gear door. The intermediate section houses the main landing gear compartment and reacts a portion of the main landing gear loads. A structural rib supports the outboard main landing gear door hinges and the main landing gear trunnion and drag link. The support for the inboard main landing gear trunnion and drag link attachment is provided by the mid fuselage. The main landing gear door is a conventional aluminum multi-rib and spar configuration.



Wing Configuration



Left Wing



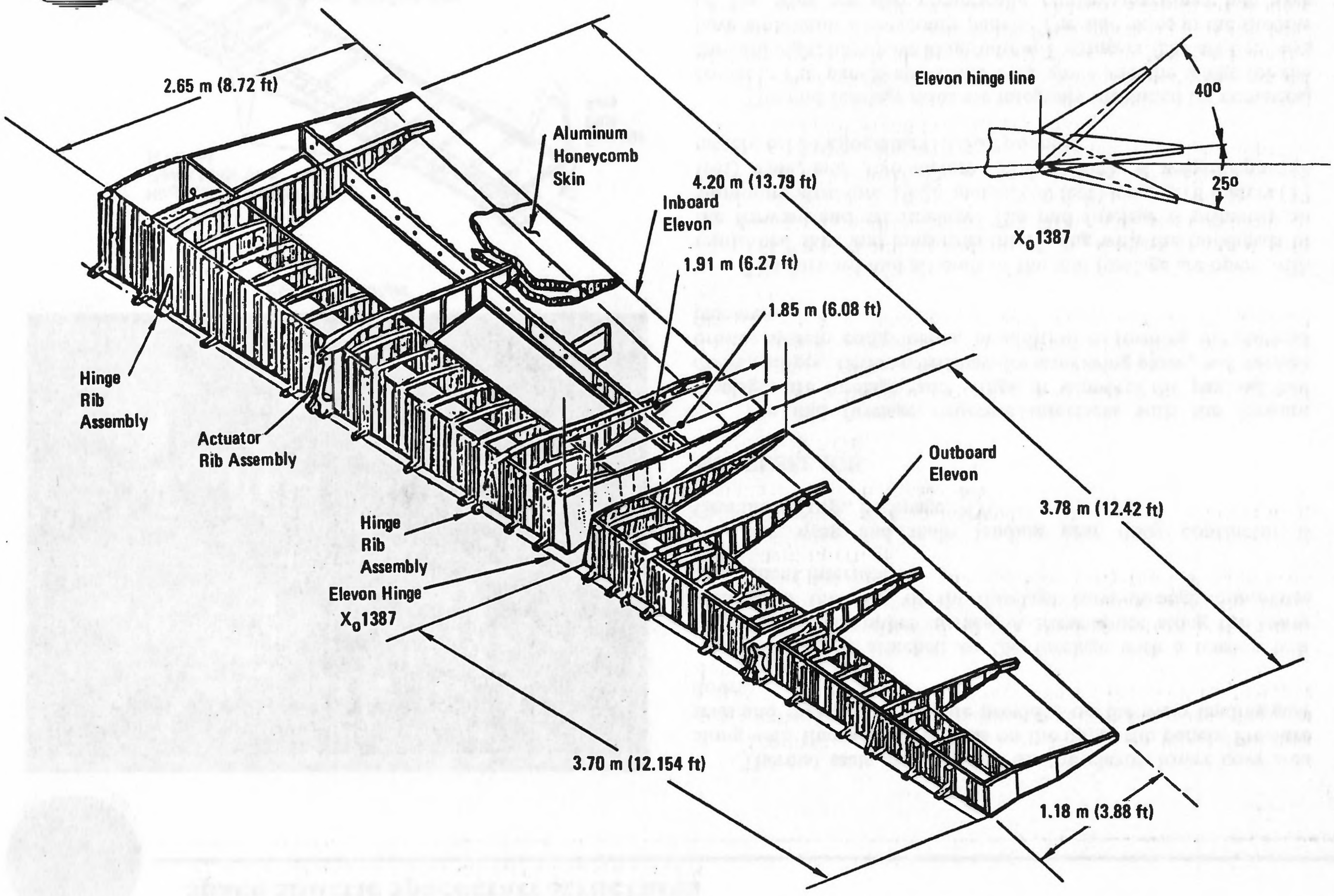
Main Landing Gear Door Construction

The four major spars are of corrugated aluminum to minimize thermal loads. The forward closeout beam is of aluminum honeycomb and provides the attachment of the thermal protection system reusable reinforced carbon-carbon leading edge structure. The rear spar provides the attachment interfaces for the elevons, hinged upper seal panels, and associated hydraulic and electrical system components. The upper and lower wing skin panels are of aluminum stiffened skins.

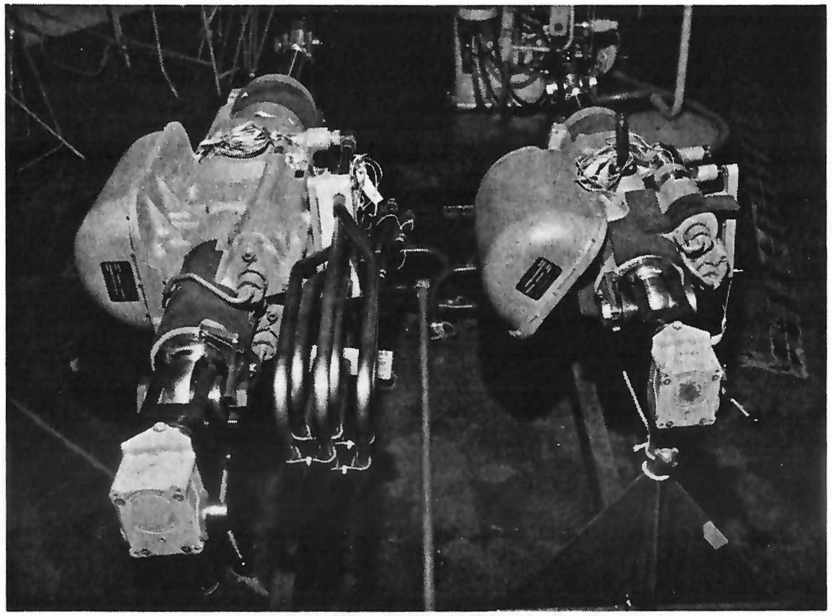
The elevons provide orbiter flight control during atmospheric flight. The two-piece elevons are of conventional aluminum multi-rib and beam construction with aluminum honeycomb skins for compatibility with the acoustic environment and thermal interaction. The elevons are divided into two segments for each wing and each segment is supported by three hinges. Attachment to the flight control system hydraulic actuators is along the forward extremity of each elevon, and all hinge moments are reacted at these points. Each elevon travels 40 degrees up and 25 degrees down.

The transistion area on the upper surface between the torque box and the movable elevon consists of a series of hinged panels, which provide a closeout of the wing-to-elevon cavity. These panels are of Inconel honeycomb sandwich construction outboard of wing station Y_W312.5 and of titanium honeycomb sandwich construction inboard of wing station Y_W 312.5. The upper leading edge of each elevon incorporates titanium rub strips. The rub strips are of titanium honeycomb construction and are not covered with the thermal protection system reusable surface insulation. The rub strips provide the sealing surface area for the elevon seal panels.

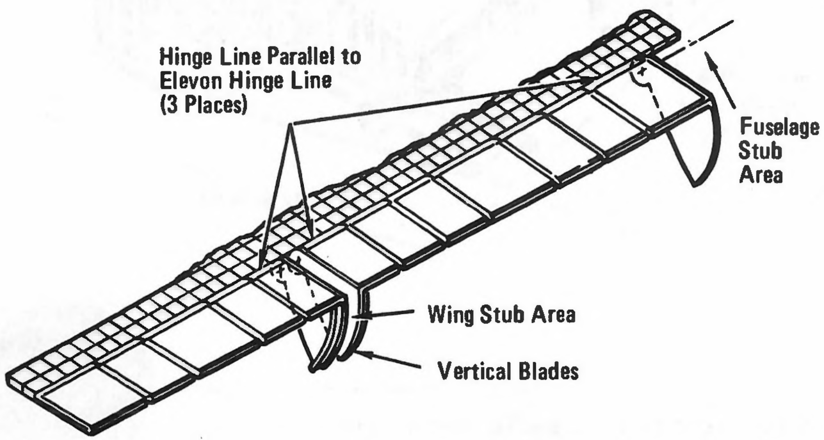
The exposed areas of the wings, main landing gear doors, and elevons are covered with the thermal protection system reusable surface insulation materials except for the elevon seal panels.



Elevon Construction



Elevon Actuators



Wing Seal Panel Arrangement

Thermal seals are provided on the elevon lower cove area along with thermal spring seals on the upper rub panels. Pressure seals and thermal barriers are provided on the main landing gear doors.

The wing is attached to the fuselage with a tension bolt splice along the upper surface. A shear splice along the lower surface in the area of the fuselage carry-through completes attachment interface.

The wing and main landing gear door contractor is Grumman Corp., Bethpage, NY.

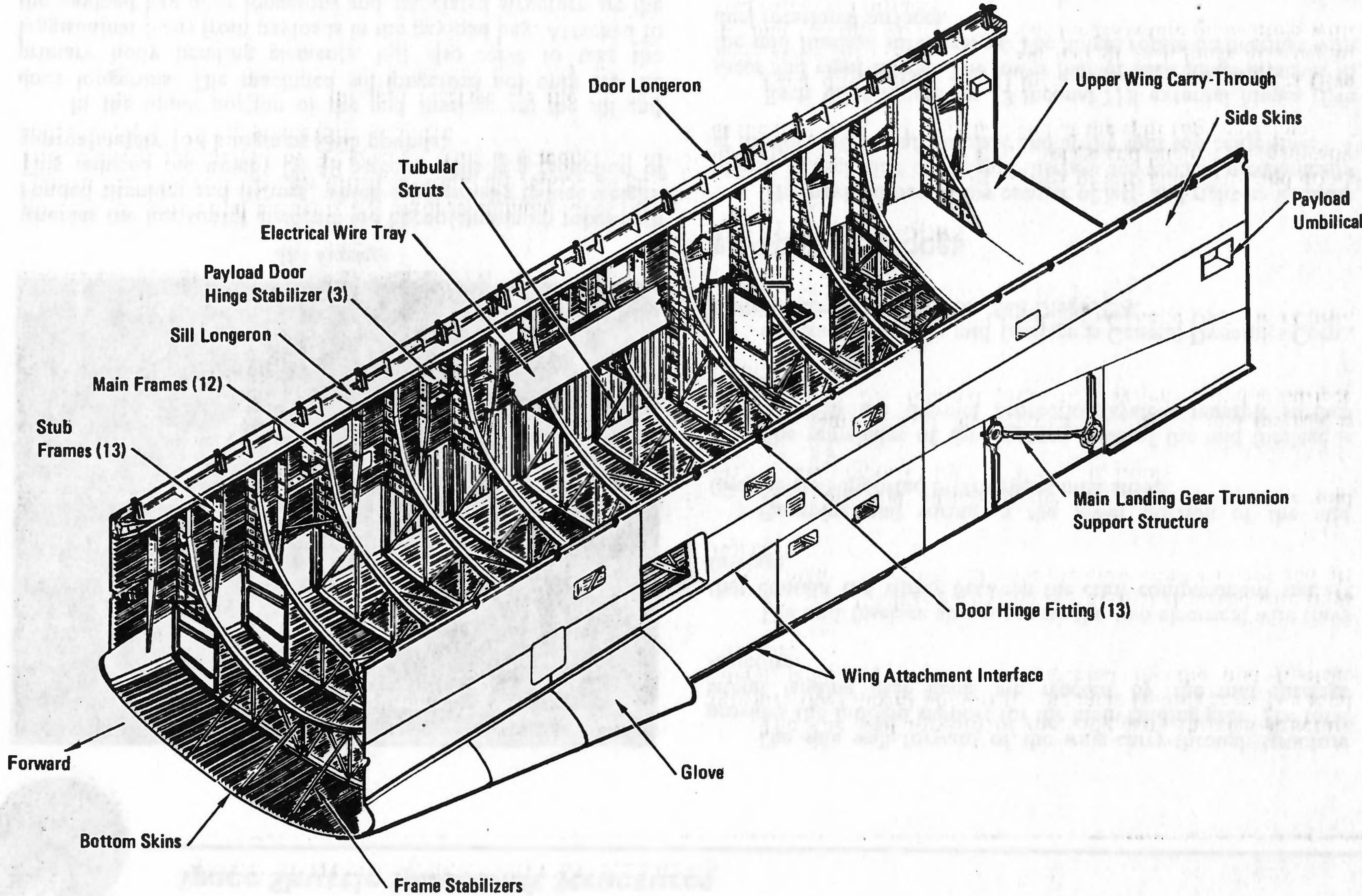
MID FUSELAGE

The mid fuselage structure interfaces with the forward fuselage, aft fuselage, and wings. It supports the payload bay doors, hinges, tiedown fittings, forward wing glove, and various orbiter system components, in addition to forming the payload bay area.

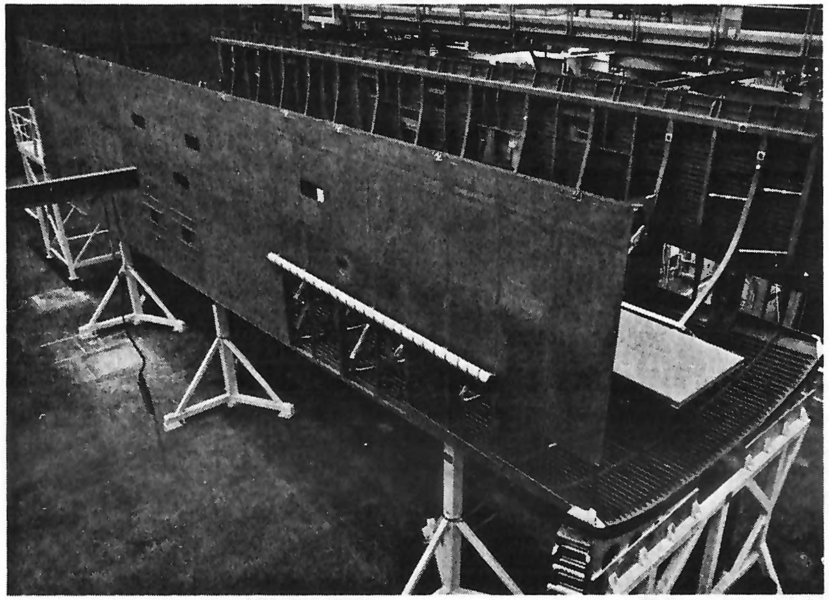
The forward and aft ends of the mid fuselage are open, with reinforced skin and longerons interfacing with the bulkheads of the forward and aft fuselage. The mid fuselage is primarily an aluminum structure 18.28 meters (60 feet) long, 5.18 meters (17 feet) wide, and 3.96 meters (13 feet) high. It weighs approximately 6,124 kilograms (13,502 pounds).

The mid fuselage skins are integrally machined by numerical control. The panels above the wing glove and the wings for the forward eight bays have longitudinal T stringers. The aft five bays have aluminum honeycomb panels. The side skins in the shadow of the wing are also numerically control machined but have vertical stiffeners.

There are 12 main frame assemblies that stabilize the mid fuselage structure. The assemblies consist of vertical side elements and horizontal elements. The side elements are machined,



Mid Fuselage Structure



Mid Fuselage

whereas the horizontal elements are boron/aluminum tubes with bonded titanium end fittings, which substantially reduce weight. This reduced the weight by 49 percent. This is a reduction of approximately 138 kilograms (305 pounds).

In the upper portion of the mid fuselage are the sill and door longerons. The machined sill longerons not only are the primary body bending elements, but also serve to take the longitudinal loads from payloads in the payload bay. Attached to the payload bay door longerons and associated structure are the 13 payload bay doors hinges. These hinges provide the vertical reaction from the payload bay doors and five of the hinges react the payload bay door shears. The sill longeron also provides in the operational configuration the base support for the payload bay manipulator arm or arms and its storage provisions, the Ku-band rendezvous antenna and antenna base support and its sto...ge provisions, and the payload bay door actuation system.

The side wall forward of the wing carry-through structure provides the inboard support for the main landing gear. The total lateral landing gear loads are reacted by the mid fuselage structure.

The mid fuselage also supports the two electrical wire trays that contain the wiring between the crew compartment and aft fuselage.

Plumbing and wiring in the lower portion of the mid fuselage are supported by fiberglass milk stools.

The remainder of the exposed areas of the mid fuselage is covered with the thermal protection system reusable surface insulation.

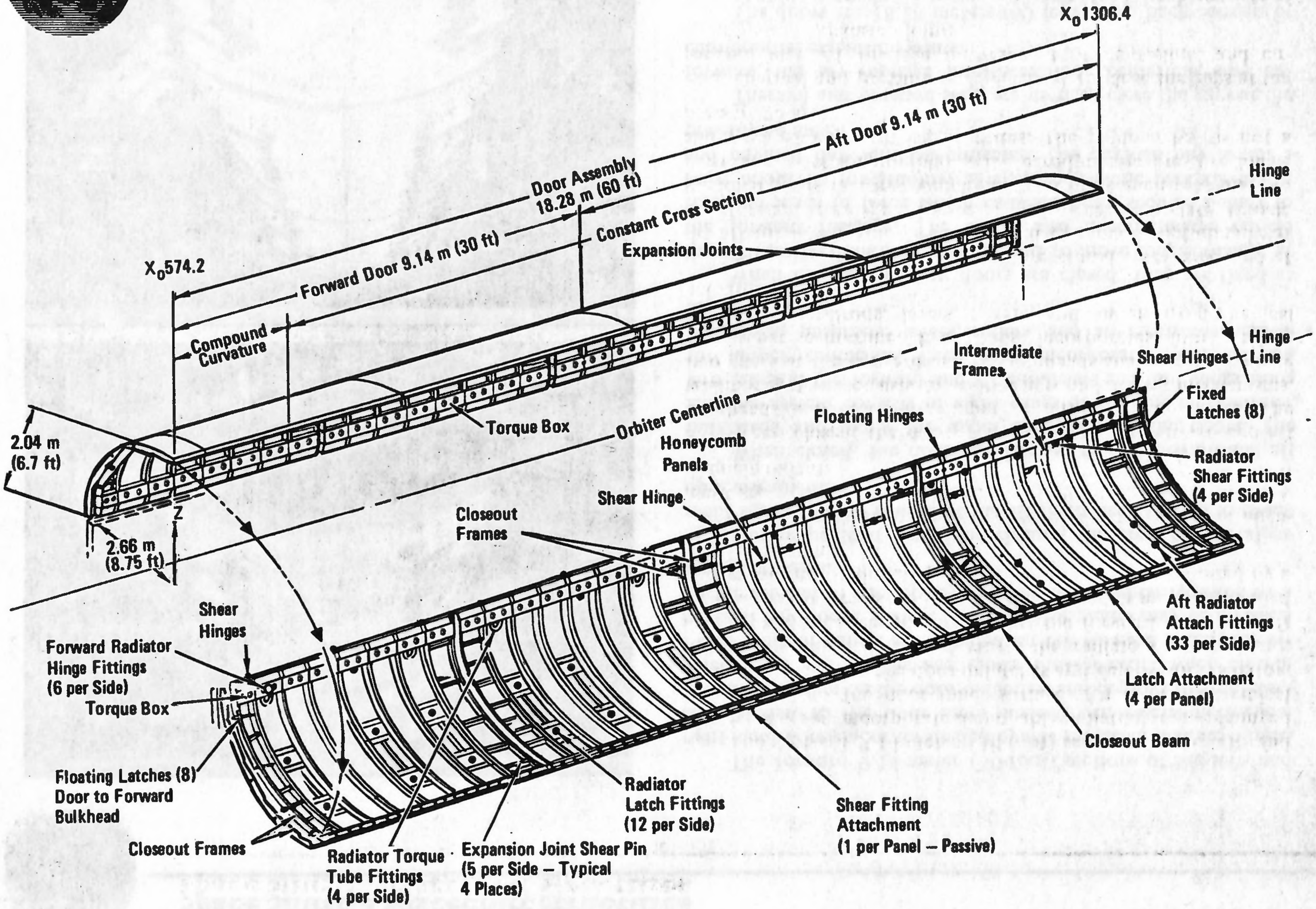
Contractor for the mid fuselage is General Dynamics Corp., Convair Aerospace Division, San Diego, CA.

PAYLOAD BAY DOORS

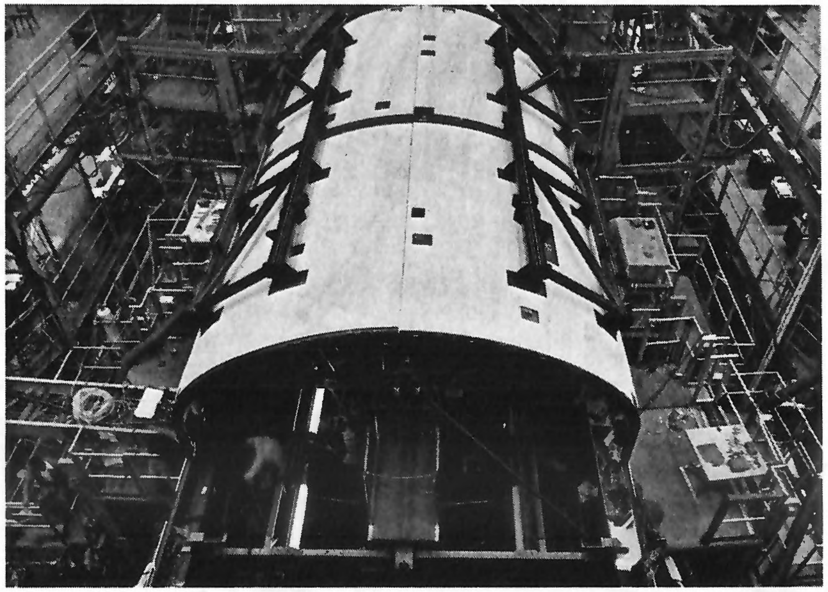
The payload bay doors consist of left- and right-hand doors hinged at each side of the mid fuselage and latched mechanically at the forward and aft fuselage and at the split top centerline.

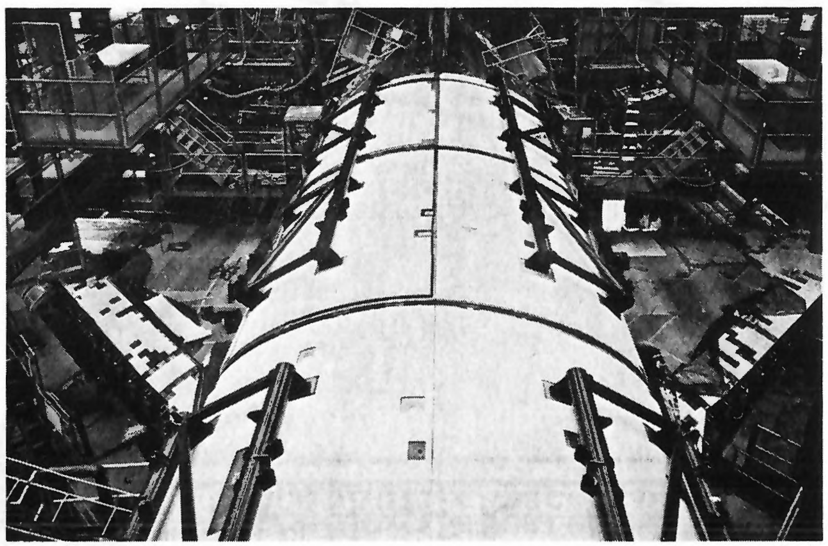
Each door hinges on 13 Inconel 718 external hinges (five shear and eight idlers). The lower half of each hinge attaches to the mid fuselage sill longeron. The hinges rotate on bearings with dual rotational surfaces.

Each door actuation system provides the mechanism to drive each door side to the open or closed position. Each mechanism consists of an electromechanical power drive unit and six rotaty gear actuators. The actuators are connected by torque tubes to each other, to the power drive unit, and to the door linkages.



Payload Bay Doors





Payload Bay Doors

The forward 9.14-meter (30-foot) sections of the left- and right-hand doors incorporate deployable radiators that are hinged and latched to the door inner surface. An electromechanical actuation system on the door unlatches and deploys the radiators when open and latches and stows the radiators when closed. Fixed radiator panels are installed on the forward end of the aft payload bay doors. Kitted, fixed radiator panels may be installed on the aft end of the aft payload bay doors when required by a specific mission.

During payload bay door closure, the crew optical alignment sight (COAS) is used at the aft flight deck station to check door alignment.

When closed, the doors are latched to the forward and aft bulkheads and along the upper centerline of the doors. The latching system consists of eight gangs of latching mechanisms, two gangs at the forward and aft bulkheads and four gangs along the upper centerline. Each gang incorporates four latches, bellcranks, pushrods, levers, rollers, and an electromechanical actuator.

When the payload bay doors are closed, they are fixed at the aft fuselage bulkhead and allowed to move longitudinally at the forward fuselage. The doors also accommodate vehicle torsional loads (a force which causes a body such as a shaft to twist about its longitudinal axis), aerodynamic pressure loads, and payload bay vent lag pressures. The payload bay is not a pressurized area.

Thermal and pressure seals are used to close the gaps at the forward and aft fuselage interface, door centerline, and circumferential expansion joints.

The doors are 18.28 meters (60 feet) long. Each consists of five segments interconnected by expansion joints. The chord of each half of these curved doors is approximately 3.04 meters (10 feet) and 4.57 meters (15 feet) in diameter. The surface area is approximately 148.64 square meters (1600 square feet).

The doors are constructed of graphite/epoxy composite material, which reduces the weight by 23 percent over that aluminum honeycomb sandwich. This is a reduction of approximately 408 kilograms (900 pounds), which brings the weight of the doors down to approximately 1480 kilograms (3264 pounds). The payload bay doors are the largest aerospace structure to be constructed from composite material.

The composite doors will withstand 163 dB acoustic noise and a temperature range of minus 112° to plus 57°C (minus 170° to plus 135°F).

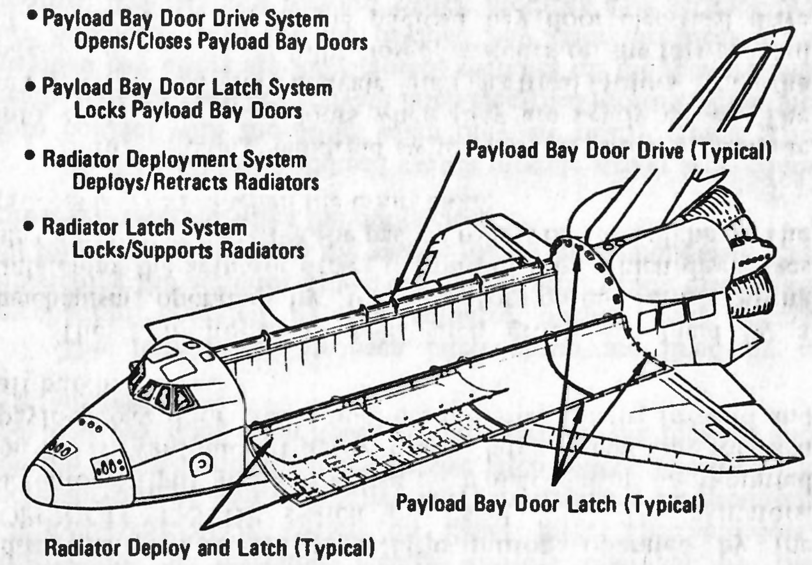
The doors are made up of subassemblies consisting of graphite/epoxy honeycomb sandwich panel, solid graphite/epoxy laminate frames, expansion joint frames, torque box, seal depressor, centerline beam intercostals, gussetts, end fittings, and clips. There are also aluminum 2024 shear pins, titanium fittings and Inconel 718 floating and shear hinges. The assembly is joined by mechanical fasteners. Lightning strike protection is provided by bonding aluminum mesh wire to the outer skin.

Extra-vehicular activity handholds are attached in the torque box areas.

The payload bay doors are covered with the reusable surface insulation.

The port (left-hand) door weighs approximately 1,077 kilograms (2,375 pounds) and the starboard weighs about 1,149 kilograms (2,535 pounds). The starboard door contains the centerline latch active mechanisms, which accounts for the weight difference. These weights do not include the radiator panel system, which adds 377 kilograms (833 pounds) per door.

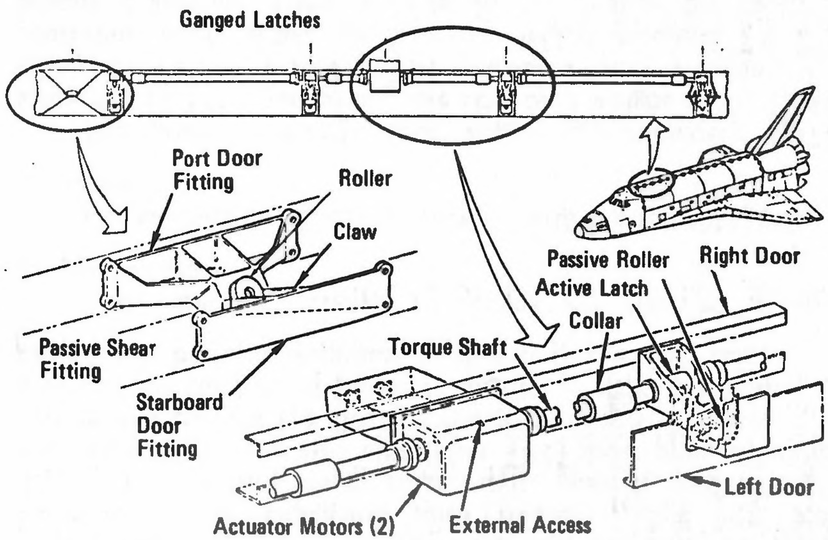
The payload bay doors can be operated automatically. The flight crew also can operate the system manually, selecting the door to be opened or closed on the computer keyboard.



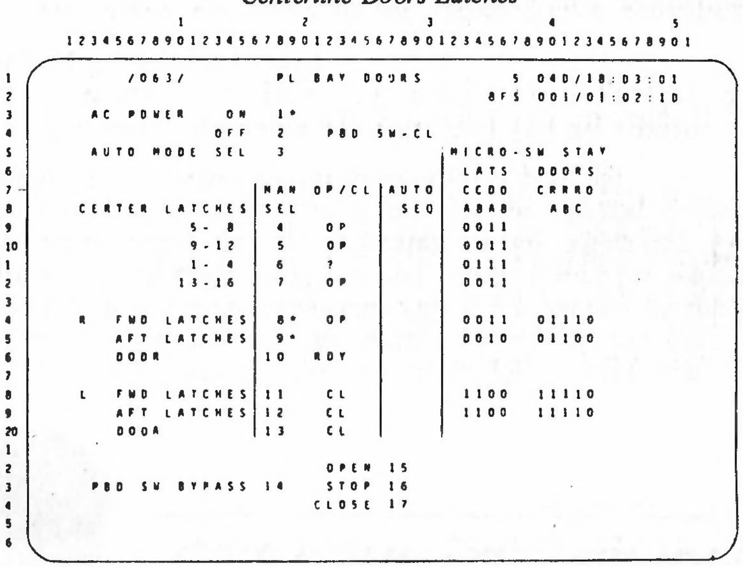
Payload Bay Doors and Radiators

The payload bay door power and control system consists of the data processing system (DPS), midmotor control assemblies (MMCA) and electro-mechanical actuator assembly electrical motors. Auto or manual commands from the DPS are sent to the MMCA's, which activate the proper actuator motors to open or close latches and doors. Microswitch feedback signals are sent to the DPS to indicate the status of the payload bay door (PBD) latch and drive system on the CRT display and to operate the PBD status indicator on the aft flight deck crew display and control panel. The status of the payload bay door limit switches is also displayed on the CRT.

Sixteen payload bay door centerline latches (eight aft and eight forward) and 16 bulkhead latches (eight aft and eight forward) secure the payload bay doors in the closed position and permit the opening of the doors.



Centerline Doors Latches



Payload Bay Doors Deployment and Closure System

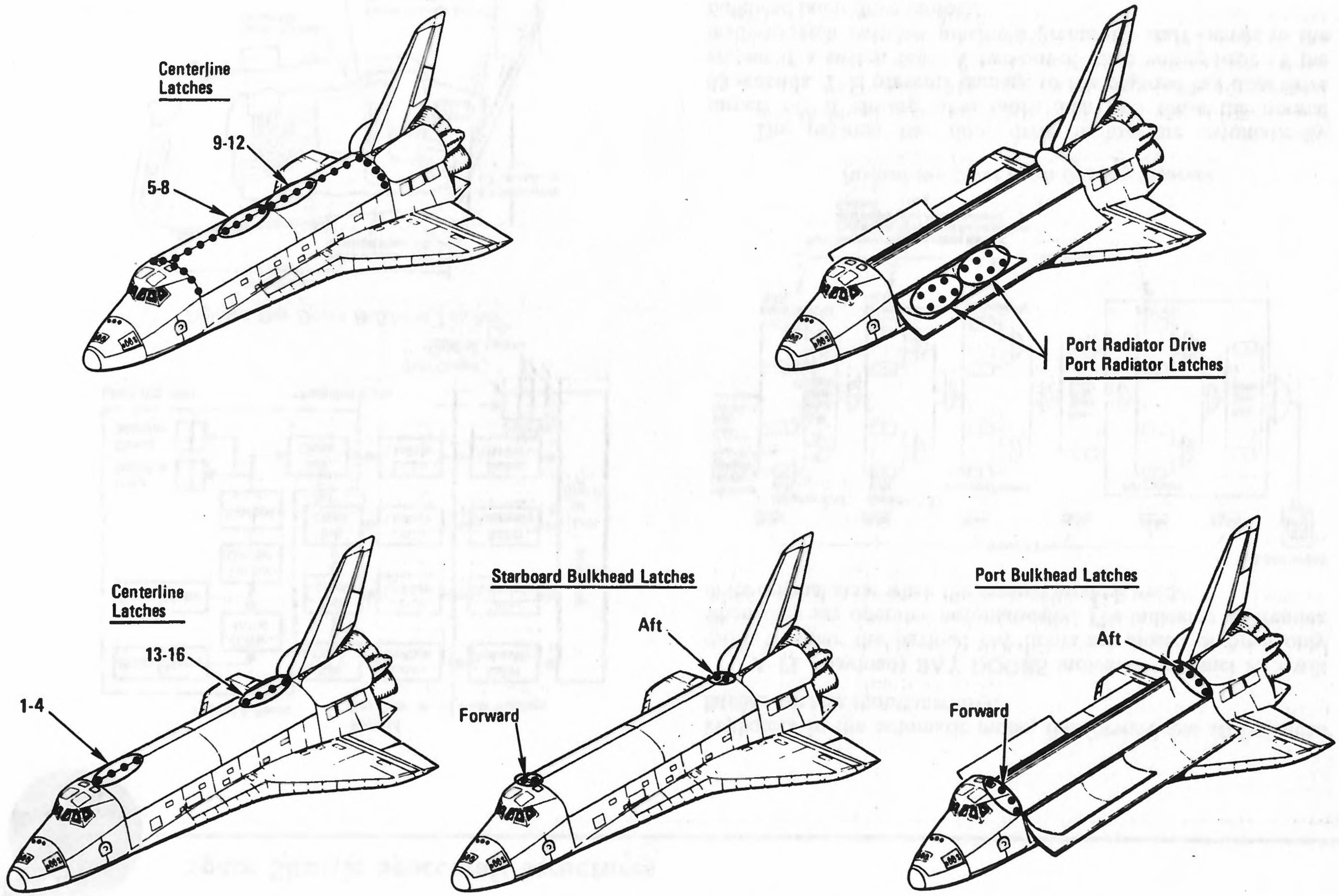
The two forward payload bay doors on each side have deployable radiator panels which reject the excess heat of the Freon-21 coolant loops from both sides of the radiator panels when the doors are open. Fixed radiator panels are installed on the forward end of the aft doors and radiate from one side only. Fixed radiator panels will be installed on the aft end of the aft doors when required by a specific mission and radiate only from one side.

The forward and aft bulkhead latches are in groups of four ganged latch hooks. Each group of hooks is opened or closed by an electro-mechanical actuator consisting of two redundant three-phase ac reversible electric motors operated by the OPEN/CLOSE/STOP switch on Panel R13. The actuators, actuator output arm and active latch mechanisms are mounted on the forward and aft doors. Passive latch rollers, one for each payload bay door latch hook, are mounted on the forward and aft bulkheads.

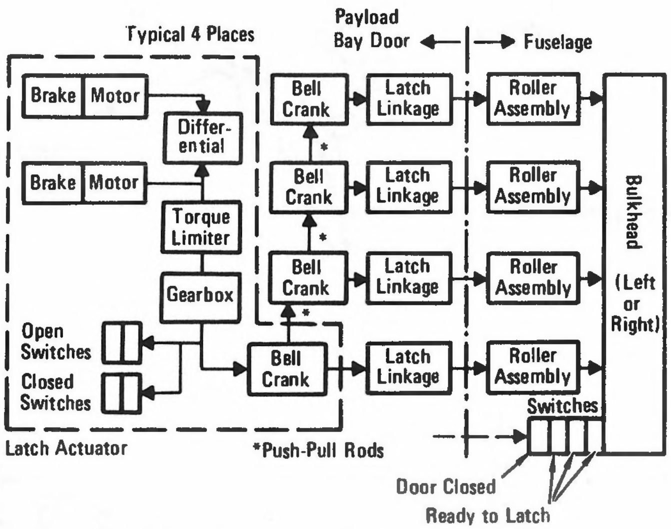
The four hooks in each latch group are lined by a mechanism operated by the actuator output arm. During unlatching the actuator drives the output arm, which disengages all four latch hooks from the passive rollers on the bulkhead. The process is reversed when the doors close.

During latching, bulkhead switch module striker arms come into contact with the doors when they are nearly closed. The ready-to-latch switches activate latch electrical motors. When the payload bay doors are fully closed, switches on the forward and aft bulkheads turn off the payload bay door electrical drive motors. The aft and forward switch modules also activate the bulkhead latch electrical drive motors, which are turned off by limit switches.

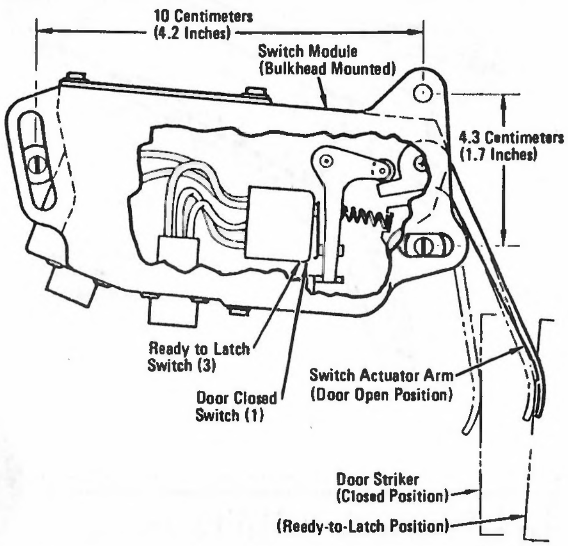
The payload bay door bulkhead latch groups can be opened and closed automatically in a predetermined sequence or manually by individual latch groups through the computer



Payload Bay Doors and Radiator Latches



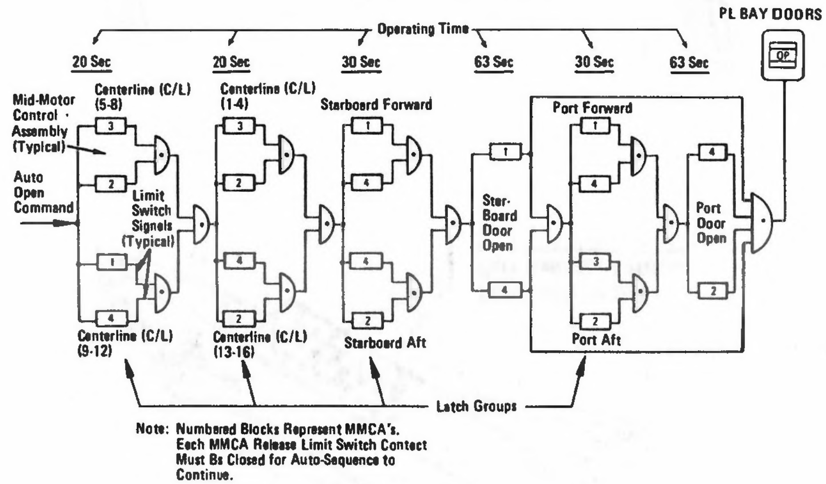
Payload Bay Doors Bulkhead Latches



Payload Bay Doors Bulkhead Switch Module

keyboard. In the automatic mode, the forward and aft bulkhead latches operate simultaneously.

A PL (payload) BAY DOORS indicator on Panel R13 will show whether the payload bay doors are closed or open only when they are operated automatically. The indicator will remain in its original state when the manual mode is used.



Payload Bay Doors Latch Opening Sequence

The payload bay door drive motors are automatically turned off if closing takes more than two times the normal 63 seconds. This prevents damage to the payload bay door drive system if a switch fails. A two-out-of-three voting logic of the ready-to-latch switches precludes premature start signals to the bulkhead latch drive motors.

The bulkhead latch drive motors are turned off automatically only if the operating time is more than twice the normal 30 seconds. If only one bulkhead electrical drive motor operates, 60 seconds are required to open or close the bulkhead latches. Each MMCA receives commands from the DPS and is turned on by limit switch closure. Each has its own timer set to twice the normal operating time. This allows enough time for single-motor operation of a bulkhead latch group without causing a sequence fail signal PLB (payload bay) DOORS CRT message and SM ALERT.

All limit switch contact closures are sent to the DPS and are shown on the PBD (payload bay door) CRT display under MICRO-SW (switch) STAT (status). The flight crew can observe the change in the status of these switches. Microswitch status is also transmitted to telemetry.

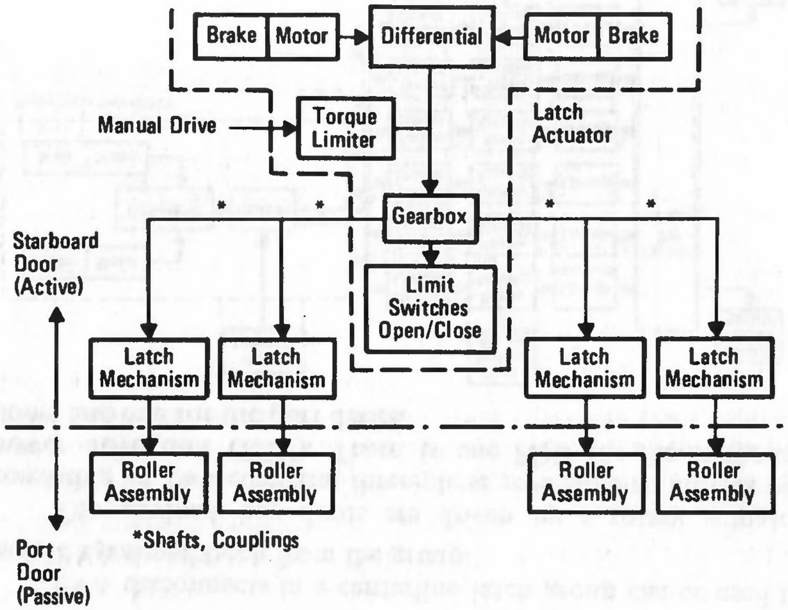
Torque limiters in the bulkhead latch groups allow slippage if a limit switch fails to turn off an electrical drive motor or a mechanism jams during latch operations, thus preventing damage to the motors or mechanisms.

Extra-vehicular activity (EVA) disconnect points in the bulkhead latch group mechanisms are provided in case the mechanism jams when the doors close. This permits crew members to close the doors manually from outside the orbiter.

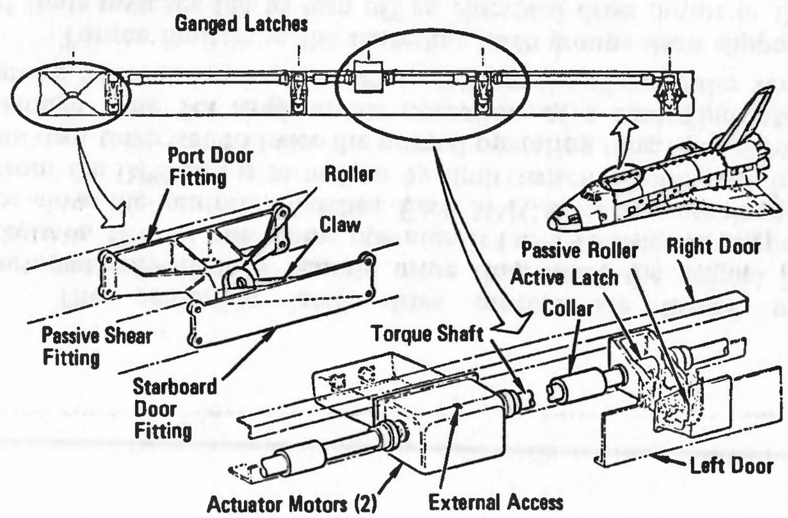
The centerline latch hooks secure the doors closed or permit them to open in conjunction with the forward and aft bulkhead latches. The centerline latches are in groups of four ganged latches that are opened or closed by an electro-mechanical actuator consisting of two redundant three-phase ac reversible electric motors.

The starboard payload bay doors must be opened first and closed last because of the arrangement of the centerline latch mechanism and the structural and seal overlap.

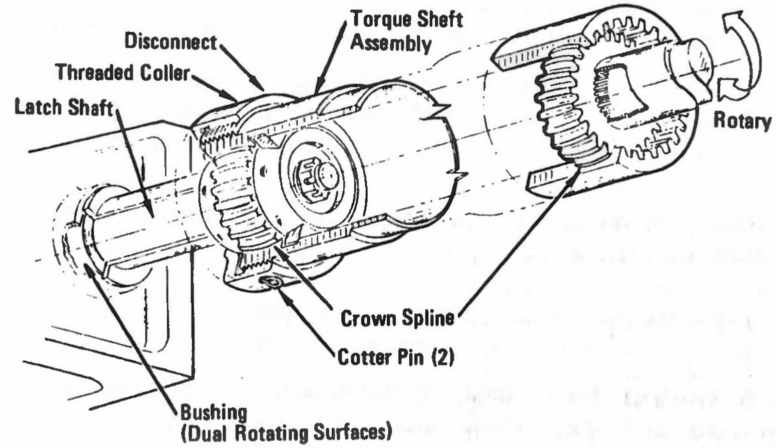
During latching, the electrical drive motors turn the rotary shaft, bellcrank, and link, causing the hook to engage the passive roller. Alignment rollers on the starboard doors eliminate overlapping of the closed doors caused by thermal distortion. All 16 centerline hook assemblies contain the alignment rollers. The passive shear fittings in each centerline latch group align the closing doors and cause the fore and aft shear loads to react once the doors are closed.



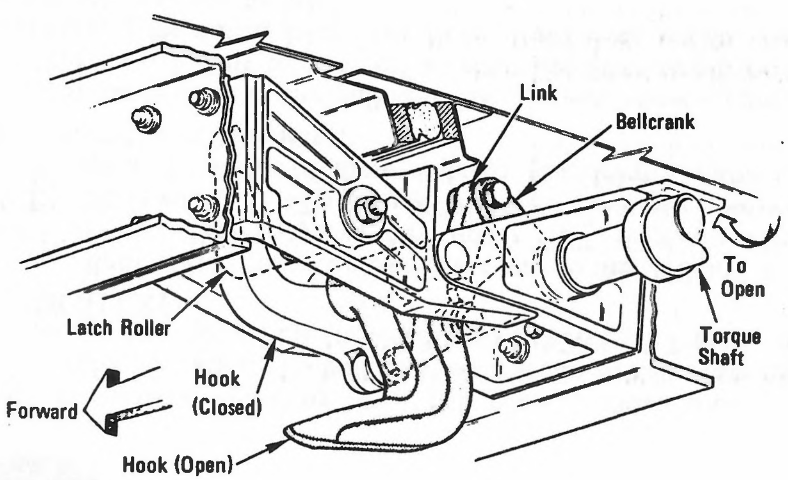
Payload Bay Doors Centerline Latches



Payload Bay Doors Centerline Latches



Centerline Door Latch Drive



Centerline Door Latch Drive

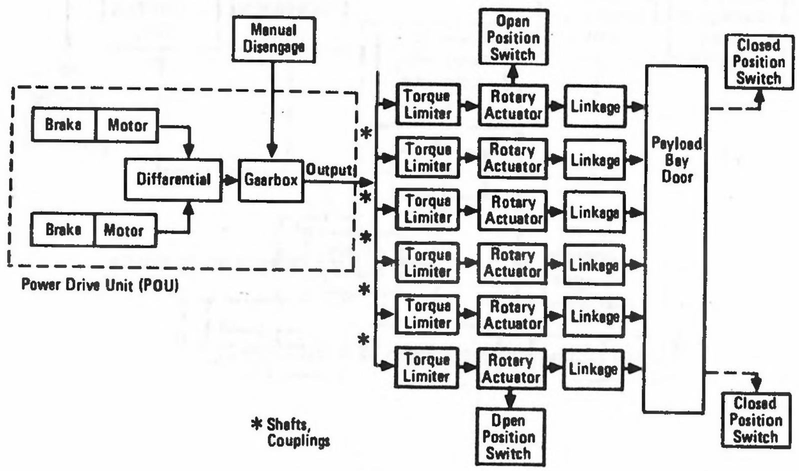
The centerline latch groups can be opened and closed automatically in a predetermined sequence manually or by single individual latch groups through the computer keyboard.

The centerline latch drive motors are turned off automatically if they operate more than twice the normal 20 seconds. If only one motor operates, it takes 40 seconds to open or close the centerline latches. Each MMCA receives commands from the DPS and is turned on by limit switch closure; each has its own timer set to twice the normal operating time. This allows enough time for single-motor operation of a centerline latch group without causing a sequence fall signal and computer alert.

Torque limiters in the centerline latch groups allow slippage if limit switches fail to turn off an electrical drive motor or the mechanisms jam.

EVA disconnects in a centerline latch group can be used to isolate a jammed latch from the group.

The payload bay doors are driven by a rotary actuator consisting of two electrical three-phase reversible ac motors per power drive unit (PDU). There is one PDU for the starboard doors and one for the port doors.



Payload Bay Door Control

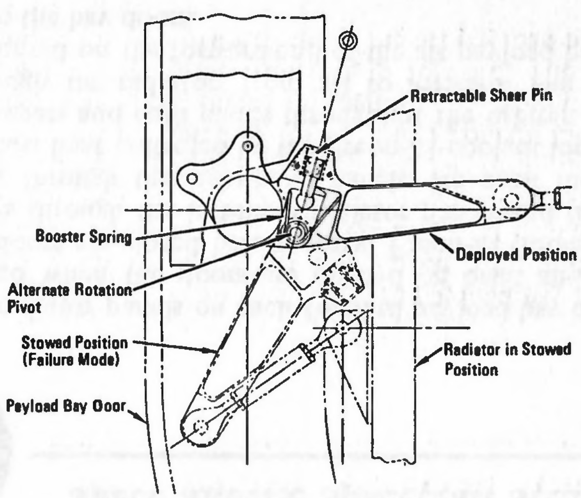
The PDU drives a 16-meter (55-foot) long torque shaft. The shaft turns the rotary actuators, which causes the push rod, bellcrank, and link to push the door open. The same arrangement pulls the doors closed. Limit switches on each drive system turn off drive motors when the doors are open.

The payload bay door drive motors are turned off automatically if both motors run more than two times the normal 63 seconds. It takes 126 seconds for just one motor to open or close the doors. Each MMCA times is set to twice the normal operating time, which allows enough time for single-motor operation of the payload bay door.

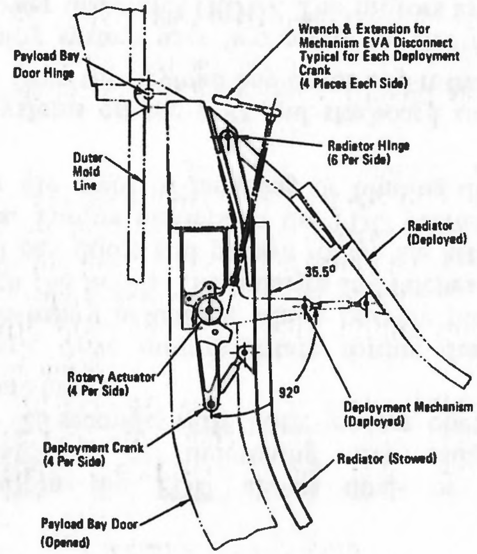
Torque limiters are incorporated in the rotary actuators to avoid damaging the drive motors or mechanisms if limit switches fail to turn off an electrical drive motor or the mechanisms jam.

Two bolts on the bellcrank and the bolt connecting the link to the rotary actuator can be EVA disconnect points if the linkage fails when the doors close. The PDU's can be disengaged manually on the ground.

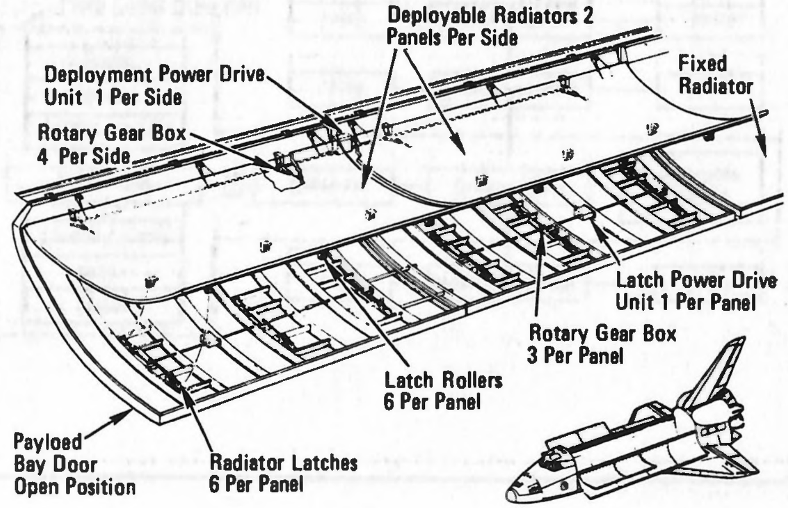
The payload bay doors open through an angle of 175.5 degrees.



Payload Bay Door, Deployable Radiator Stowed



Payload Bay Door, Deployable Radiator Deployed



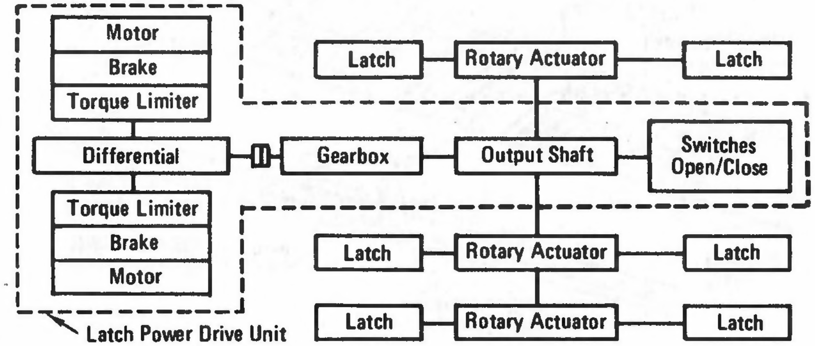
Payload Bay Door, Deployable Radiator Mechanism

Two radiator panels on each forward payload bay door will be deployed when the doors are opened on orbit and stowed when the doors are closed before entry. Freon-21 Coolant Loop No. 1 flows through the left-hand radiator panels and the No. 2 loop flows through the right-hand panels. On orbit the panels radiate excess heat collected by the Freon-21 coolant loops from heat exchangers and cold plates throughout the orbiter. Coolant flows through the radiators from aft to forward. The radiator panels mounted on the forward end of the aft payload bay doors are fixed to the bay doors.

The radiator deploy and stow operation is controlled manually from the aft flight deck crew display and control panel. The PL BAY MECH (payload bay mechanisms) PWR switches and the RADIATOR LATCH and RADIATOR CONTROL SYS switches control the panels. Four indicators show the radiator latch and deploy status.

When the payload bay doors are fully open, the SYS1 and SYS2 PL BAY MECH PWR switches are turned to ON and the RADIATOR LATCH CONTROL switches to RELEASE. In approximately 26 seconds, the status indicators will show REL (release). The RADIATOR CONTROL SYS A and SYS B switches are turned to DEPLOY, and in approximately 43 seconds, the indicators will show DEP (deploy). The stow sequence is reversed. The electrical power and control system for the deployable radiators is designed to permit the loss of one latch or radiator control switch or one PL BAY MECH PWR SYS1 or SYS2 switch and not affect the operation of the radiators.

Each deployable radiator panel is secured to the payload bay door in the stowed position by six ganged latches. Two electrical drive motors latch or unlatch the six latches in each panel simultaneously. The motors can be reversed by the LATCH and UNLATCH switches on Panel R13. Limit switches turn off the motors when the latches are opened or closed. Each electrical drive motor is controlled by a separate set of switches. A



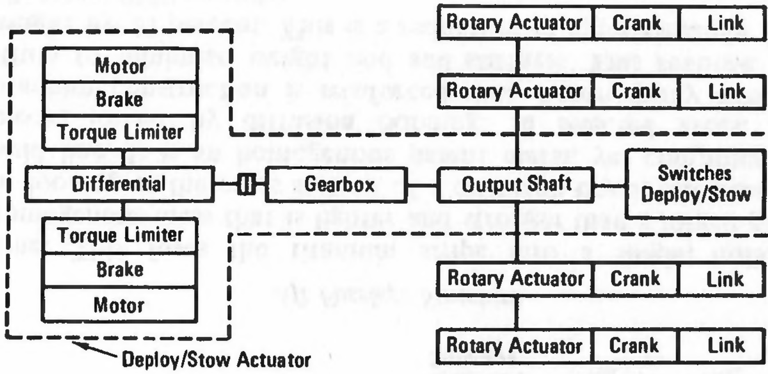
Radiator Latch Control

differential within the PDU allows dual- or single-motor operation. Latching or unlatching the radiators takes approximately 26 seconds with both motors operating or 52 seconds with one motor.

The electric drive motors rotate torque shafts, and the shafts turn the rotary actuators, which operate push rods that latch or unlatch the hooks. The linkages and latches are attached to the payload bay doors and passive rollers are attached to the radiator panels. Torque limiters in the PDU prevent damage to the system in the event of jamming or binding during system operation.

Deploy systems on the port and starboard sides drive the radiators away from the payload bay doors and retract them.

Each deploy system uses two reversible electric motors to operate the power drive unit (PDU). The motors are not turned on until the MMCA's have received two signals from the radiator panel latch drives. This prevents inadvertent deployment of the radiators while still latched. As the rotary actuator shaft turns, the deployment crank and link straighten out and push the radiator panels away from the payload bay door to the deployed position. Stowing the radiators is the reverse operation. Limit



Radiator Deploy/Stow Control

switches turn off the electric drive motors when the radiators are deployed or stowed. Both port and starboard radiators are deployed or stowed simultaneously. The radiators deploy 35.5 degrees from the payload bay doors in 43 seconds with both electrical motors operating or in 86 seconds with one electrical motor operating. The DEPLOY/STOW switches on Panel R13 allow single- or dual-motor operation.

Torque limiters in the PDU prevent damage in the event of jamming or binding during operation.

Each rotary crank can be disengaged from the rotary actuator (via EVA operations) by retracting shear pin. Retraction allows the crank to rotate and lets the crew stow the panels if the system fails. If the PDU fails, all four shear pins must be removed to allow manual stowing of the radiators. The pins are accessible when the radiators are fully deployed. No disengagement is planned if the radiators fail to deploy.

Contractors are Rockwell's Tulsa Division, Tulsa, OK (payload bay doors); Curtiss Wright, Caldwell, NJ (payload bay door power drive unit rotary actuators, drive shafts, torque tubes and couplings, radiator deploy actuator and latch mechanism); Hoover Electric, Los Angeles, CA (payload bay door electro-mechanical rotary actuators); Vought Corp., Dallas, TX (radiators).

AFT FUSELAGE

The aft fuselage consists of an outer shell, thrust structure, and internal secondary structure. It is approximately 5.48 meters (18 feet) long, 6.70 meters (22 feet) wide, and 6.09 meters (20 feet) high.

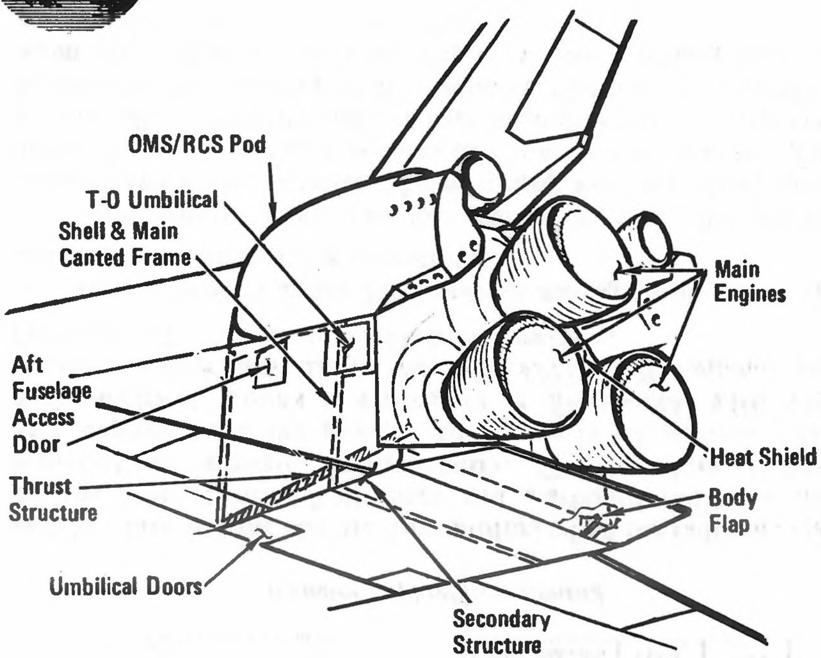
The aft fuselage supports and interfaces with the left-hand and right-hand aft OMS/RCS pods, the wing aft spar, mid fuselage, orbiter/external tank rear attachments, Space Shuttle main engines, heat shield, body flap, vertical tail, and two T-0 launch umbilical panels.

The aft fuselage provides the load path to the mid fuselage main longerons, main wing spar continuity across the forward bulkhead of the aft fuselage, structural support for the body flap, structural housing around all internal systems for protection from operational environments (pressure, thermal, and acoustic), and for controlled internal pressures during flight.

The forward bulkhead closes off the aft fuselage from the mid fuselage and is composed of machined and beaded sheet metal aluminum segments. The upper portion of the bulkhead attaches to the front spar of the vertical tail.

The internal thrust structure supports the three main engines. The upper section of the thrust structure supports the upper main engine, and the lower section of the thrust structure supports the two lower main engines. The internal thrust structure includes the main engines, load reaction truss structures, engine interface fittings, and the actuator support structure and supports the main engines, low pressure turbopumps, and propellant lines. The orbiter/external tank two aft attach points interface at the longeron fittings.

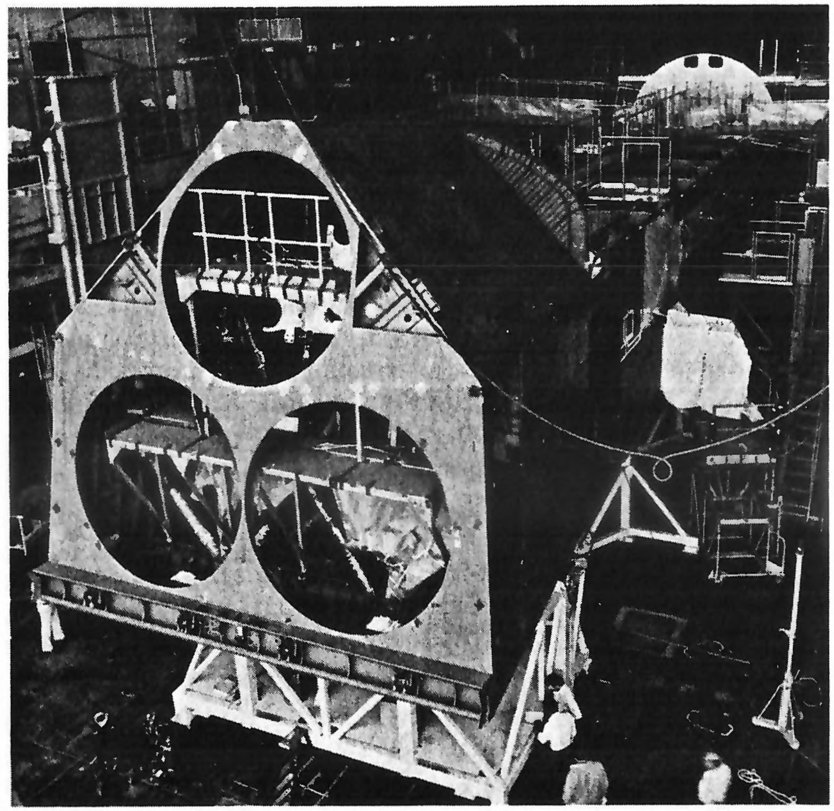
The internal thrust structure is composed mainly of 28 machined, diffusion-bonded truss members. In diffusion bonding, titanium strips are bonded together under heat, pressure, and



Aft Fuselage Structure

time. This fuses the titanium strips into a single, hollow homogenous mass that is lighter and stronger than a forged part. In looking at the cross section of a diffusion bond, one sees no weld line. It is an homogenous parent metal, yet composed of pieces joined by diffusion bonding. In selected areas, the titanium construction is reinforced with boron/epoxy tubular struts to minimize weight and add stiffness. This reduced the weight by 21 percent. This is a reduction of approximately 408 kilograms (900 pounds).

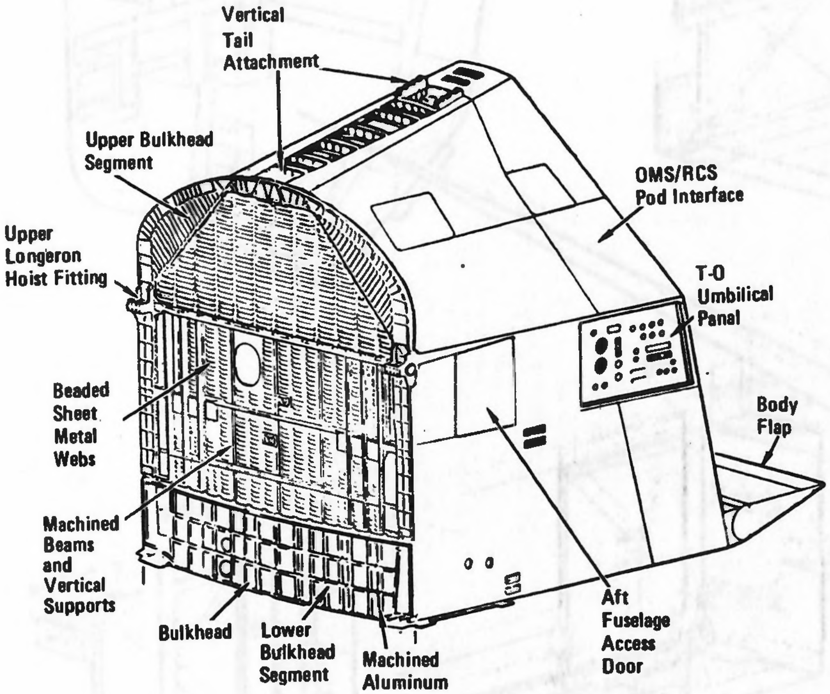
The upper thrust structure of the aft fuselage is of integral machined aluminum construction with aluminum frames, except for the vertical fin support frame, which is titanium. The skin panels are integrally machined aluminum and attach to each side of the vertical fin to react drag and torsion loading.



Aft and Mid Fuselage

The outer shell of the aft fuselage is constructed of integral machined aluminum. Various penetrations are provided in the shell for access to installed systems.

The secondary structure of the aft fuselage is of conventional aluminum construction, except that some titanium and fiberglass is used for thermal isolation of equipment. The aft fuselage secondary structures consists of brackets, build-up webs, truss members, and machined fittings as required by system

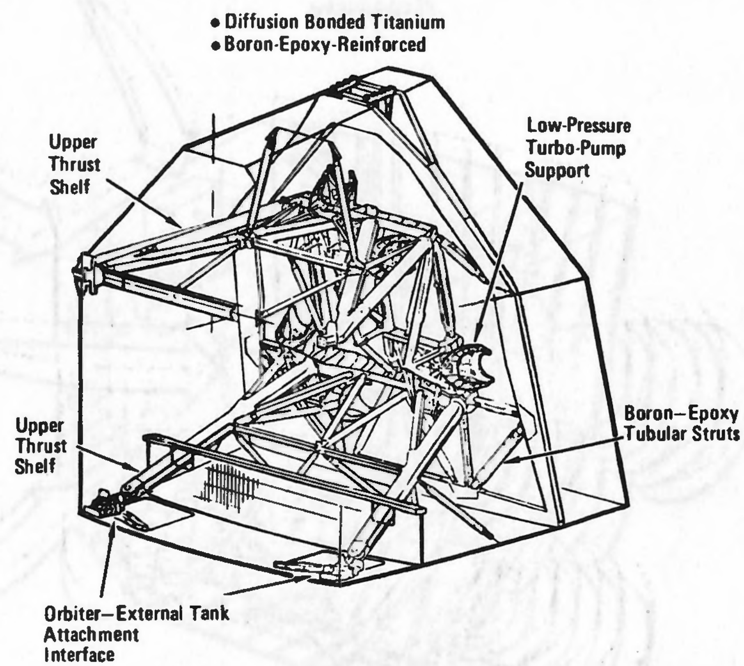


Aft Fuselage

loading and support constraints. Certain system components, such as the avionics shelves, are shock-mounted to the secondary structure. The secondary structure includes support provisions for the auxiliary power units, hydraulics, ammonia boiler, flash evaporator, and electrical wire runs.

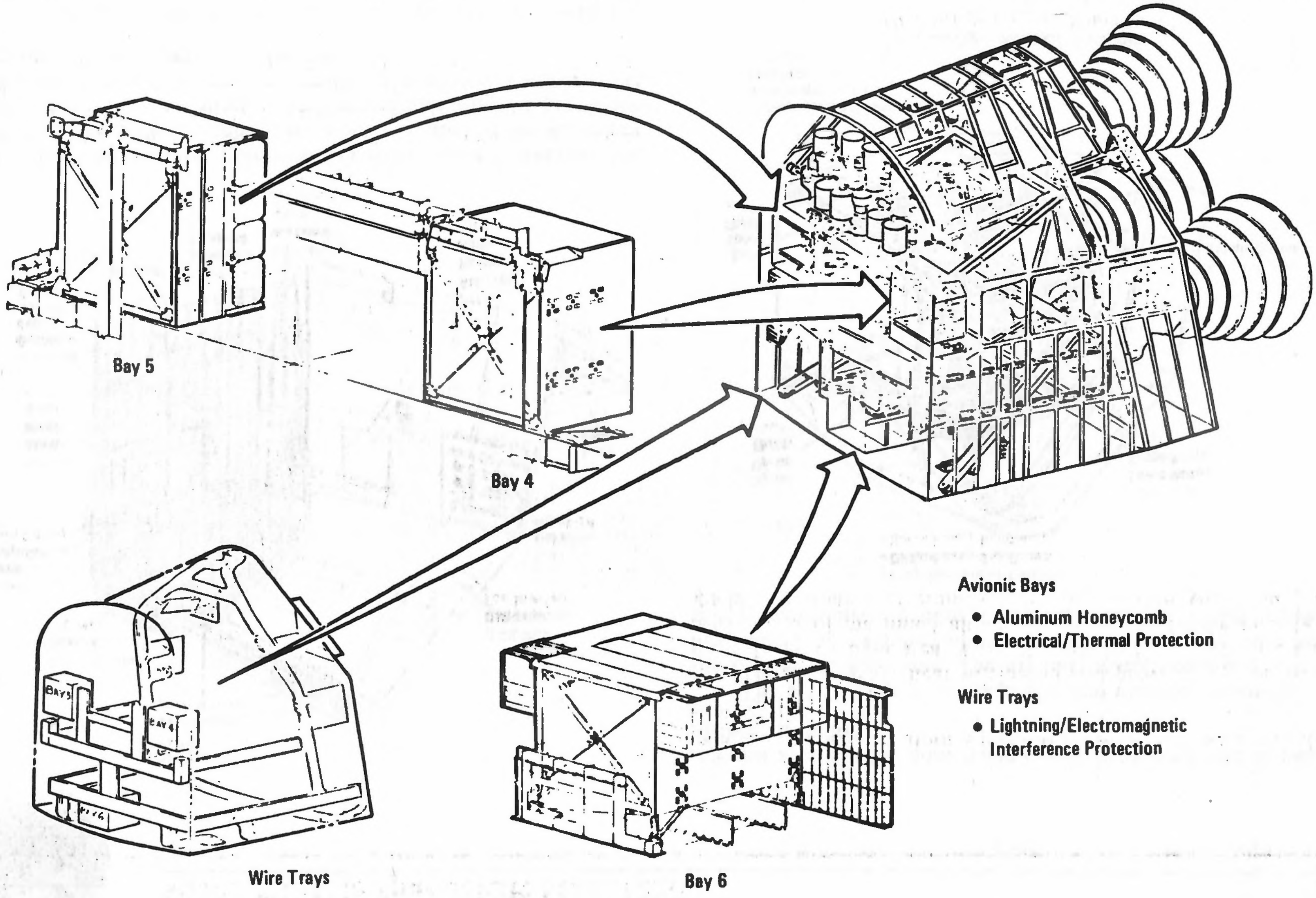
The two external tank umbilical areas interface with the external tank two aft attach points of the orbiter and the external tank liquid oxygen, liquid hydrogen feed lines, as well as electrical wire runs. The umbilicals are retracted, and the umbilical areas are closed off after external tank separation by an electromechanically operated beryllium door at each umbilical. Thermal barriers are employed at each umbilical door. The exposed area of each door when closed is covered with reusable surface insulation.

The aft fuselage heat shield and seal provides a closeout of the orbiter aft base area. The aft heat shield consists of a base heat shield of machined aluminum. Attached to the base heat shield are domes of honeycomb construction which support



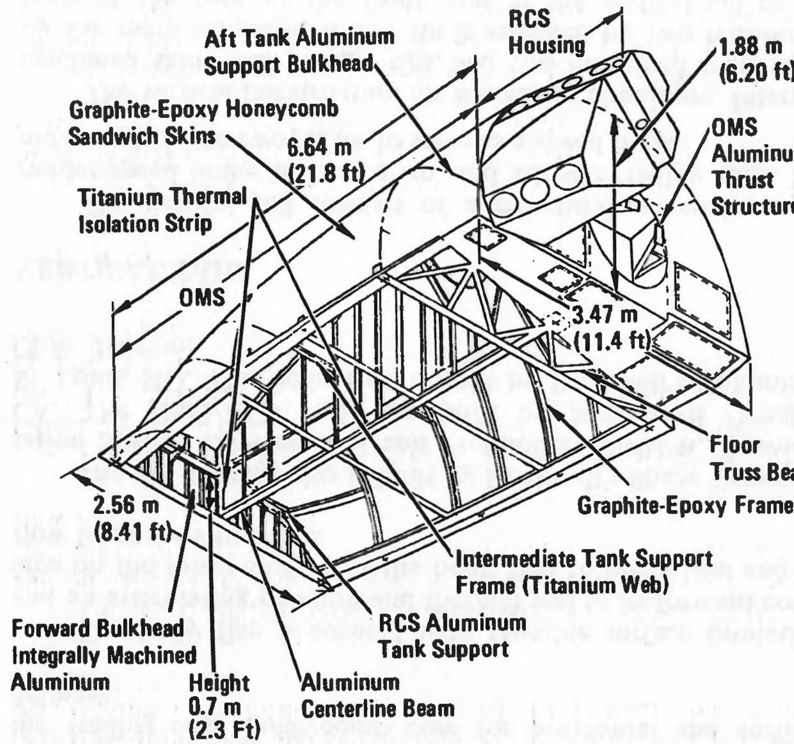
Aft Fuselage Internal Structure

flexible and sliding seal assemblies. The engine-mounted heat shield is of Inconel honeycomb construction and is removable for access to the main engine power heads. The heat shield is covered with reusable surface insulation except for the Inconel segments. The exposed areas of the aft fuselage also are covered with reusable surface insulation.



Aft Fuselage Avionics Provisions

Avionics Bays



Aft OMS/RCS Pod

The OMS/RCS left- and right-hand pods are attached to the upper aft fuselage left and right sides. Each pod is fabricated primarily of graphite/epoxy composite and aluminum. Each pod is 6.45 meters (21.8 feet) long and 3.46 meters (11.37 feet) wide at its aft end and 2.56 meters (8.41 feet) wide at its forward end, with a surface area of approximately 40.41 square meters (435 square feet). Each pod is divided into two compartments: the OMS and the RCS housing. Each pod houses all the OMS propulsion components and RCS propulsion components. Each pod is attached to the aft fuselage with 11 bolts. The pod skin panels are graphite/epoxy honeycomb sandwich. The forward and aft bulkhead, aft tank support bulkhead, and floor truss beam are machined aluminum 2124. The centerline beam is 2024 aluminum sheet with titanium stiffness and graphite/epoxy frames. The frames are graphite/epoxy. The OMS thrust structure is conventional 2124 aluminum construction. The cross braces are aluminum tubing, and the attach fittings at the forward and aft fittings are 2124 aluminum. The intermediate fittings are corrosion-resistant steel. The RCS housing, which attaches to the OMS pod structure, contains the RCS thrusters and associated propellant feed lines. The RCS housing is of aluminum sheet metal construction, including flat outer skins, and the curved outer skin panels are graphite/epoxy honeycomb sandwich. Access to the OMS and RCS and attach points are provided by 24 doors in the skins.

The two graphite/epoxy pods per spacecraft reduces the weight by 10 percent. This is a reduction of approximately 204 kilograms (450 pounds). The pods will withstand 162 dB acoustic noise and a temperature range from minus 112°C to plus 57°C (minus 170° to plus 135°F).

The exposed areas of the OMS/RCS pods are covered with reusable surface insulation and a pressure and thermal seal is installed at the OMS/RCS pod aft fusleage interface. Thermal barriers also are installed and they interface with the RCS thrusters and reusable surface insulation.

The body flap thermally shields the three main engines during entry and provides the orbiter with pitch control trim during its atmospheric flight after entry.

The body flap is an aluminum structure consisting of ribs, spars, skin panels, and a trailing edge assembly. The main upper and lower forward honeycomb skin panels are joined to the ribs, spars, and honeycomb trailing edge with structural fasteners. The removable upper forward honeycomb skin panels complete the body flap structure.

The upper skin panels aft of the forward spar and the entire lower skin panels are mechanically attached to the ribs. The forward upper skin consists of five removable access panels attached to the ribs with quick-release fasteners. The four integral machined aluminum actuator ribs provide the aft fuselage interface through self-aligning bearings. Two bearings are located in each rib for attachment to the four rotary actuators located in the aft fuselage, which are controlled by the flight control system and the hydraulically actuated rotary actuators. The remaining ribs consist of eight stability ribs and two closeout ribs constructed of chemically milled aluminum webs bonded to aluminum honeycomb core. The forward spar web is of chemically milled sheets with flanged holes and stiffened beads. The spar web is riveted to the ribs. The trailing edge includes the rear spar, which is composed of piano-hinge half-cap angles, chemically milled skins, honeycomb aluminum core, closeouts, and plates. The trailing edge attaches to the upper and lower forward panels by the piano-hinge halves and hinge pins. Two moisture drain lines and one hydraulic fluid drain line penetrate the trailing edge honeycomb core for horizontal and vertical drainage.

The body flap is covered with reusable surface insulation and an articulating pressure and thermal seal to its forward cover area on the lower surface of the body flap to block heat and air flow from the structures.

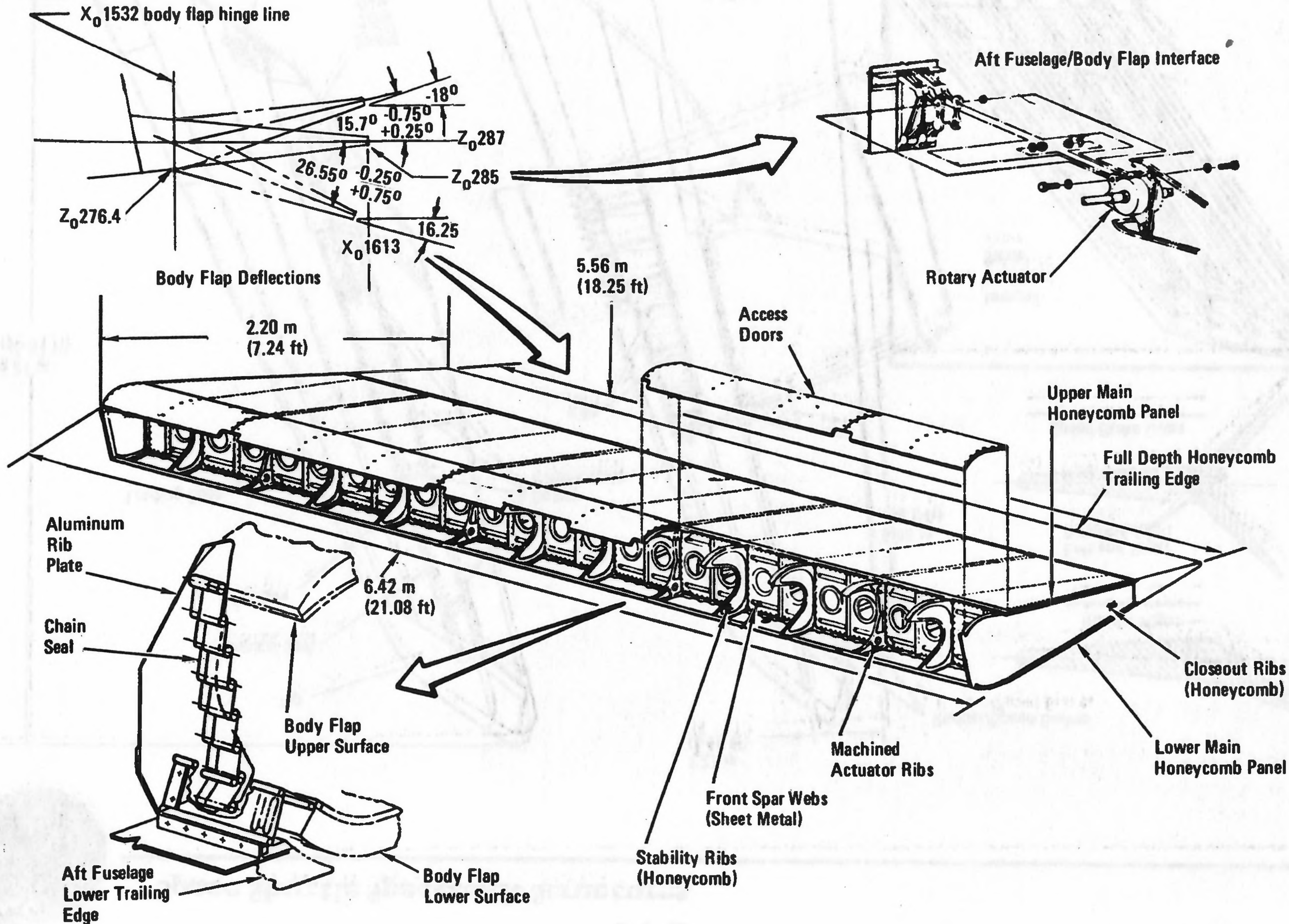
The aft fuselage also is built by Rockwell's Space Transportation System Development and Production Division, Downey, CA. The OMS/RCS pods are built by McDonnell Douglas, St. Louis, MO. The body flap is built by Rockwell's Columbus, Ohio, division.

VERTICAL TAIL

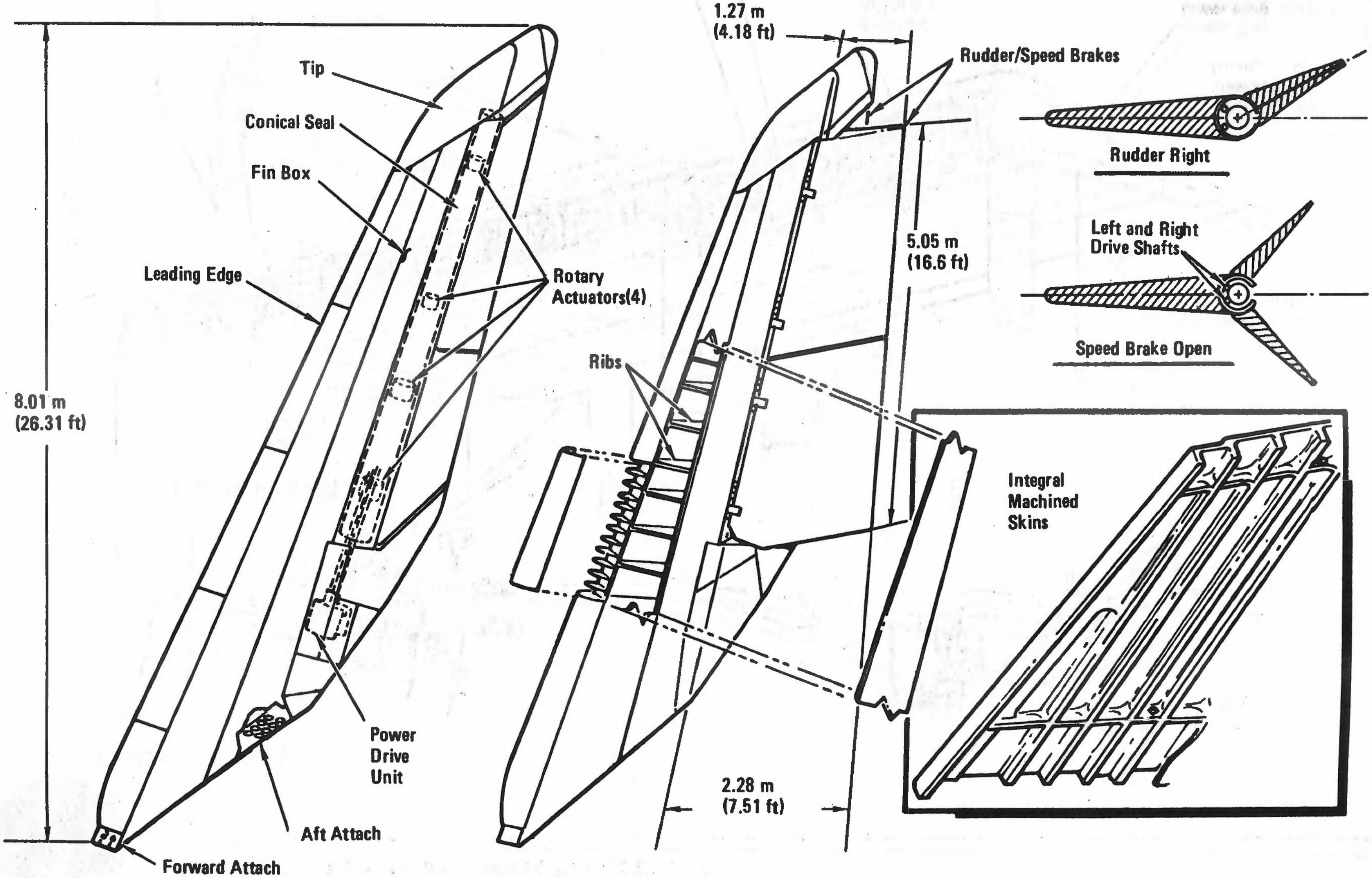
The vertical tail consists of a structural fin surface, the rudder/speed brake surface, a tip, and a lower trading edge. The rudder splits into two halves to serve as a speed brake.

The vertical tail structure fin is made of aluminum. Intergral machined skins and strings, ribs, and two machined spars make up the main torque box. The fin is attached by two tension tie bolts at the root of the front spar of the vertical tail to the forward bulkhead of the aft fuselage and by eight shear bolts at the root of the vertical tail rear spar to the upper structural surface of the aft fuselage.

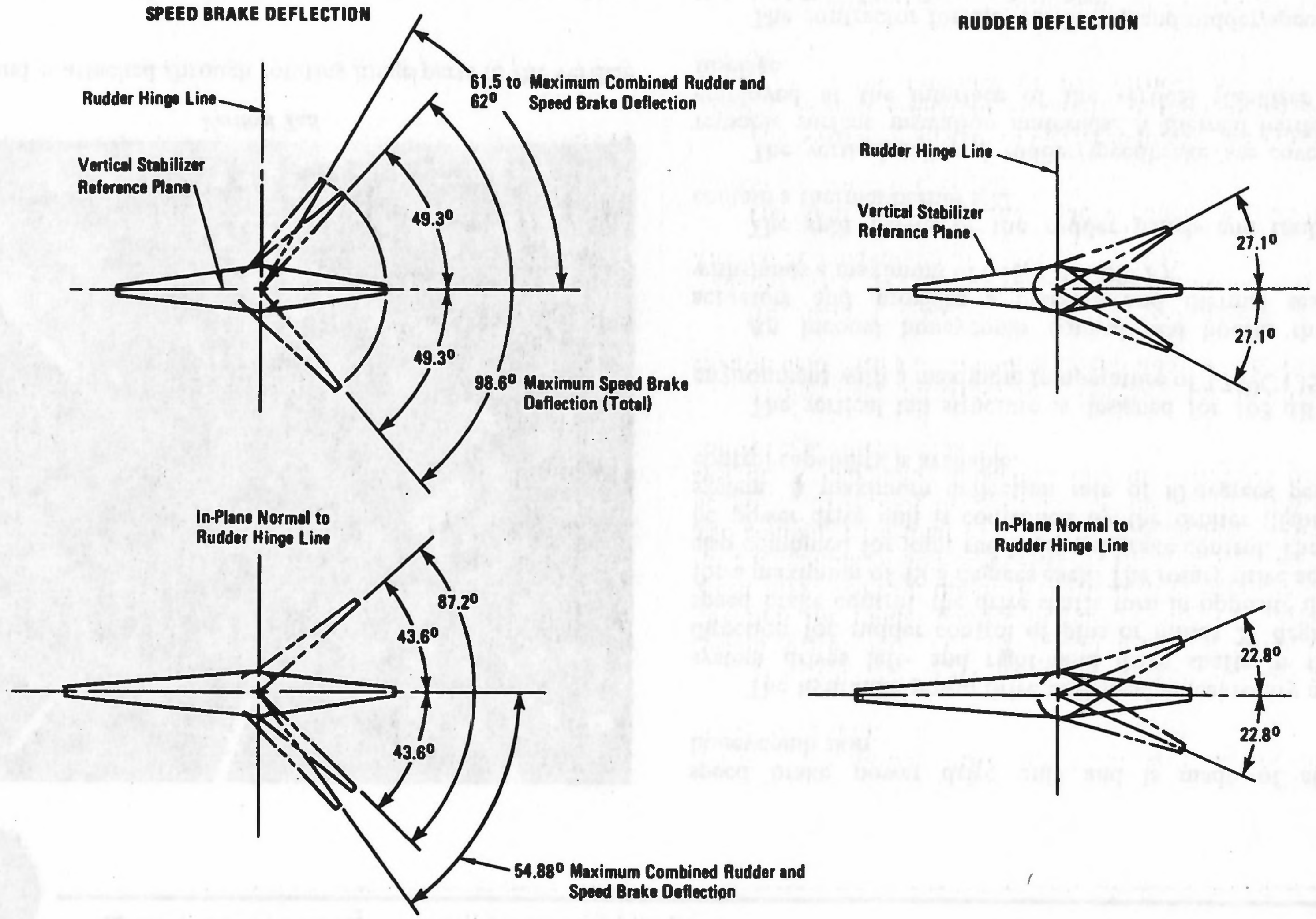
The rudder/speed brake control surface is made of conventional aluminum ribs and spars with aluminum honeycomb skin


$X_0$1532 body flap hinge line
-18°
-0.75°
15.7°
+0.25°
$Z_0$287
26.55°
-0.25°
+0.75°
$Z_0$285
$Z_0$276.4
16.25
$X_0$1613
Body Flap Deflections
Aft Fuselage/Body Flap Interface
Rotary Actuator
5.56 m
(18.25 ft)
2.20 m
(7.24 ft)
Access
Doors
Upper Main
Honeycomb Panel
Full Depth Honeycomb
Trailing Edge
Aluminum
Rib
Plate
6.42 m
(21.08 ft)
Chain
Seal
Closeout Ribs
(Honeycomb)
Body Flap
Upper Surface
Machined
Actuator Ribs
Lower Main
Honeycomb Panel
Front Spar Webs
(Sheet Metal)
Stability Ribs
(Honeycomb)
Aft Fuselage
Lower Trailing
Edge
Body Flap
Lower Surface

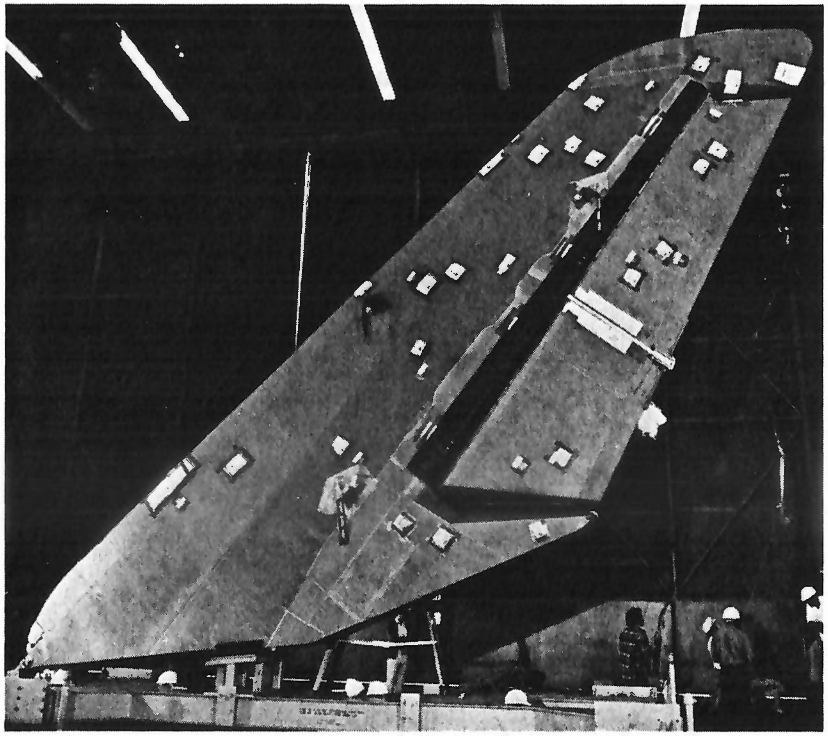
Body Flap



Vertical Tail



Aerosurface Deflections



Vertical Tail

panels and is attached through rotating hinge parts to the vertical tail fin.

The lower trailing edge area of the fin houses the rudder/speed brake power drive unit and is made of aluminum honeycomb skin.

The hydraulic power drive unit/mechanical rotary actuation system drives left- and right-hand drive shafts in the same direction for rudder control of plus or minus 27 degrees. For speed brake control, the drive shafts turn in opposite directions for a maximum of 49.3 degrees each. The rotary drive actions are also combined for joint rudder/speed brake control. The hydraulic power drive unit is controlled by the orbiter flight control system. A maximum deflection rate of 10 degrees per second control capability is available.

The vertical tail structure is designed for 163 dB acoustic environment with a maximum temperature of 176°C (350°F).

An Inconel honeycomb conical seal houses the rotary actuators and provides a pressure and thermal seal which withstands a maximum of 648°C (1200°F).

The split halves of the rudder panels and trailing edge contain a thermal barrier seal.

The vertical tail and rudder/speedbrake are covered with reusable surface insulation materials. A thermal barrier is also employed at the interface of the vertical stabilizer and aft fuselage.

The contractor for the vertical tail and rudder/speedbrake is Fairchild Republic, Farmingdale, NY.

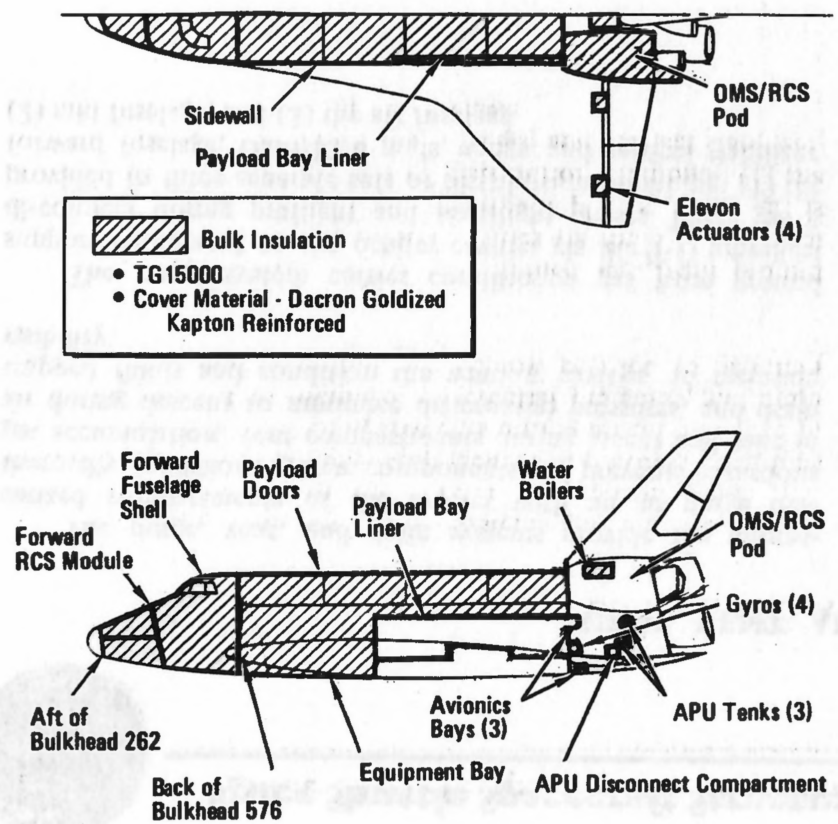
PASSIVE THERMAL CONTROL SYSTEM

A passive thermal control system helps maintain the orbiter systems and components within their temperature limits. This system uses orbiter heat sources and heat sinks and is supplemented by insulation blankets, thermal coatings, and thermal isolation methods. Heaters are provided on components and systems where passive thermal control techniques are not adequate. (The heaters are described in the various systems.)

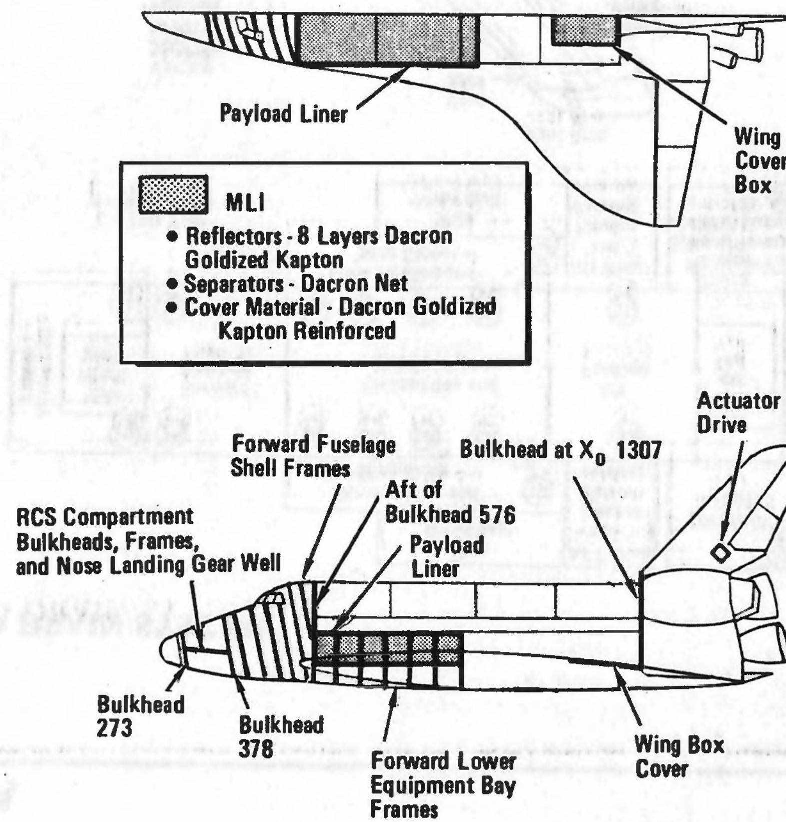
The insulation blankets are of two basic types: fibrous bulk and multilayer. The bulk blankets are fibrous material with a density of 0.9 kilogram (2 pounds per cubic foot) with a sewn cover of reinforced double-goldized Kapton. The cover material has 145,317 holes per square meter (13,500 holes per square foot) for venting. Goldized tape is used for cutouts, patching, and reinforcements. Tufts are used throughout the blankets to minimize billowing during venting.

The multilayer blankets are constructed of alternate layers of perforated double-goldized Kapton reflectors and Dacron net separators for a total of 16 reflector layers, with the two cover halves counting as two layers. Covers, tufting, and goldized tape are similar to the bulk blankets.

The contractors are Hi-Temp Insulation, Inc., Camarillo, CA (fibrous insulation); Scheldahl, Northfield, MN (cover materials and inner layers); Apex Mills, Los Angeles, CA (separators).



Bulk Insulation

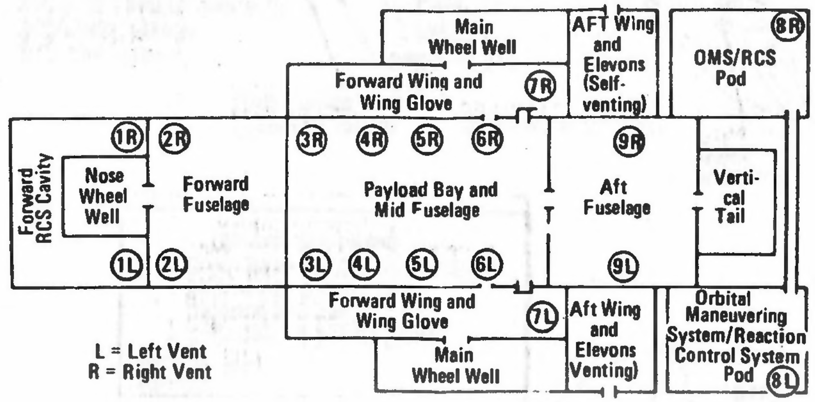


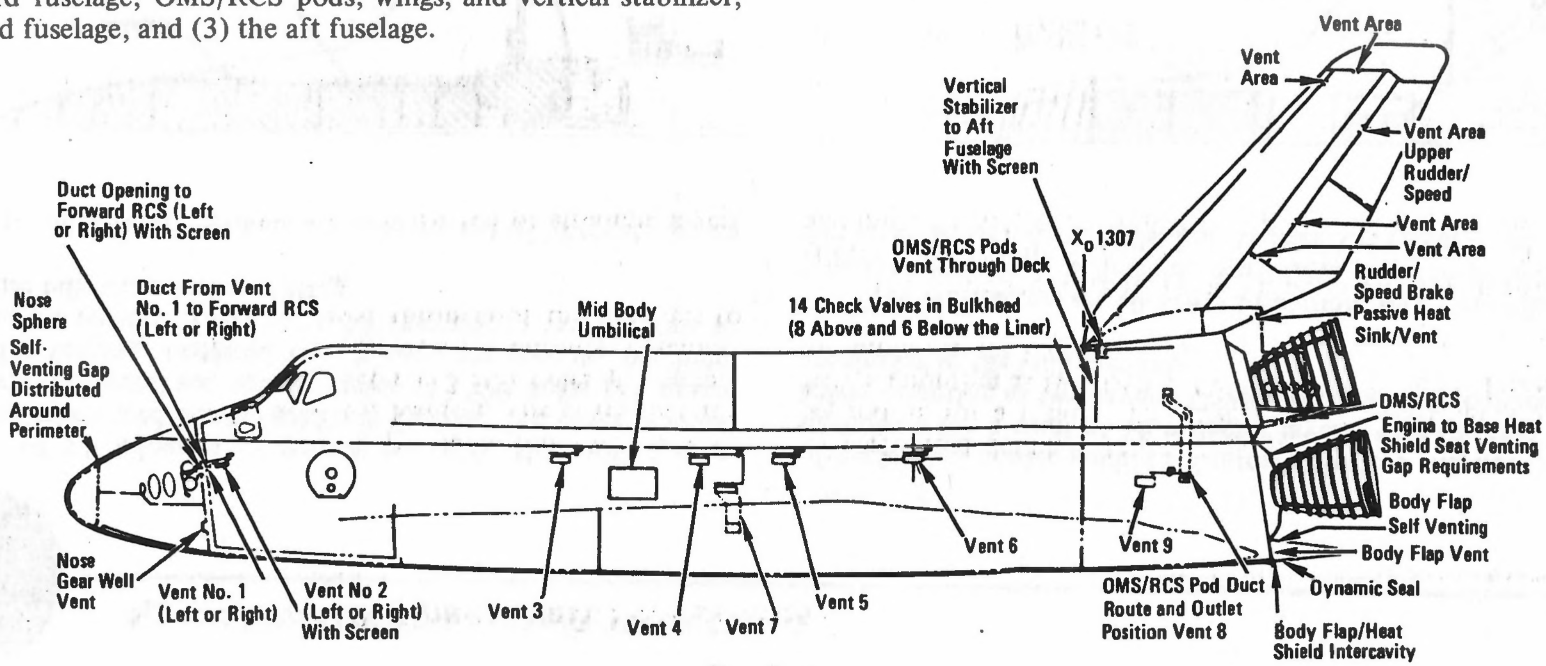
Multilayer Insulation

PURGE, VENT, AND DRAIN SYSTEM

The purge, vent, and drain systems provide the unpressurized compartments of the orbiter with an air purge that thermally conditions system components and prevents hazardous gas accumulation, vent compartments during ascent and take in air during descent to minimize differential pressures, and drain trapped fluids and condition the window cavities to maintain visibility.

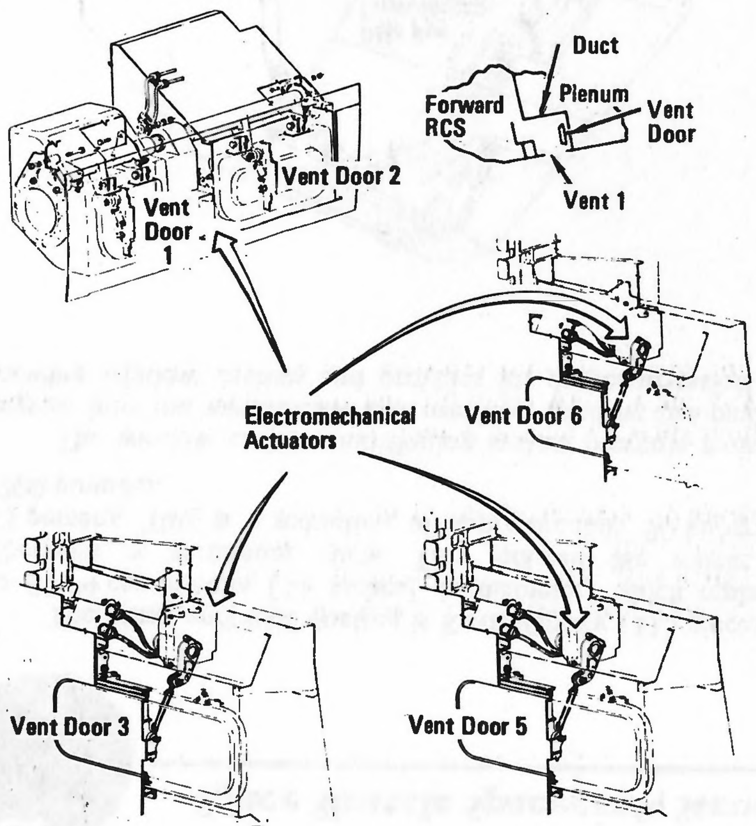
The purge system carries conditioned gas from ground support equipment to the orbiter cavities via the T-O umbilical disconnect during preflight and postflight phases. Purge gas is provided to three separate sets of distribution plumbing: (1) the forward fuselage, OMS/RCS pods, wings, and vertical stabilizer, (2) mid fuselage, and (3) the aft fuselage.





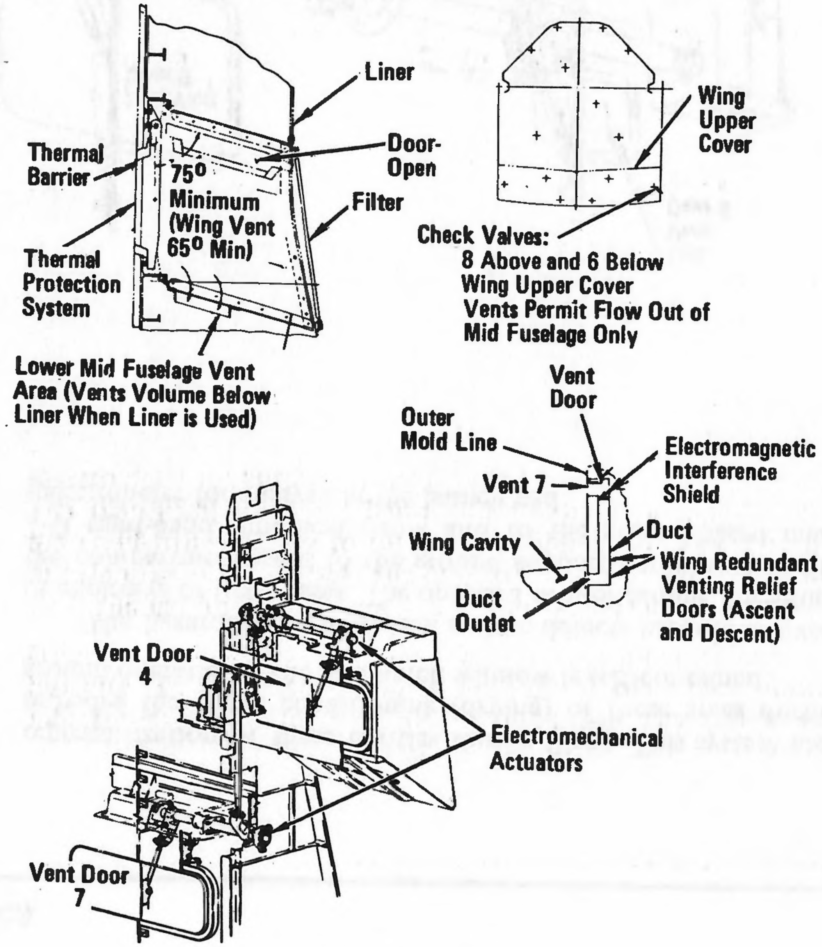
Vent System

The vent system provides the flow area for control of pressure during purge, depressurization during ascent, molecular venting in orbit, and repressurization during descent. There are 18 vent ports in the fuselage skin that are dedicated to specific orbiter cavities. The purge and vent outlets are located and sized to vent the cavities within structural, hazard, and purge limitations. Vent doors are operated by electromechanical actuators and are sequenced during the mission to protect against gas ingestion, high acoustic levels, and entry heating.



Vent Doors (1, 2, 3, 5 & 6)

The drain system provides flow paths and systems to drain or remove accumulated water. The paths are provided through a series of limber holes that allow drainage to the lowest point for removal. Locations that cannot be served by limber holes are evacuated through a series of tubes and disconnects by ground support equipment.

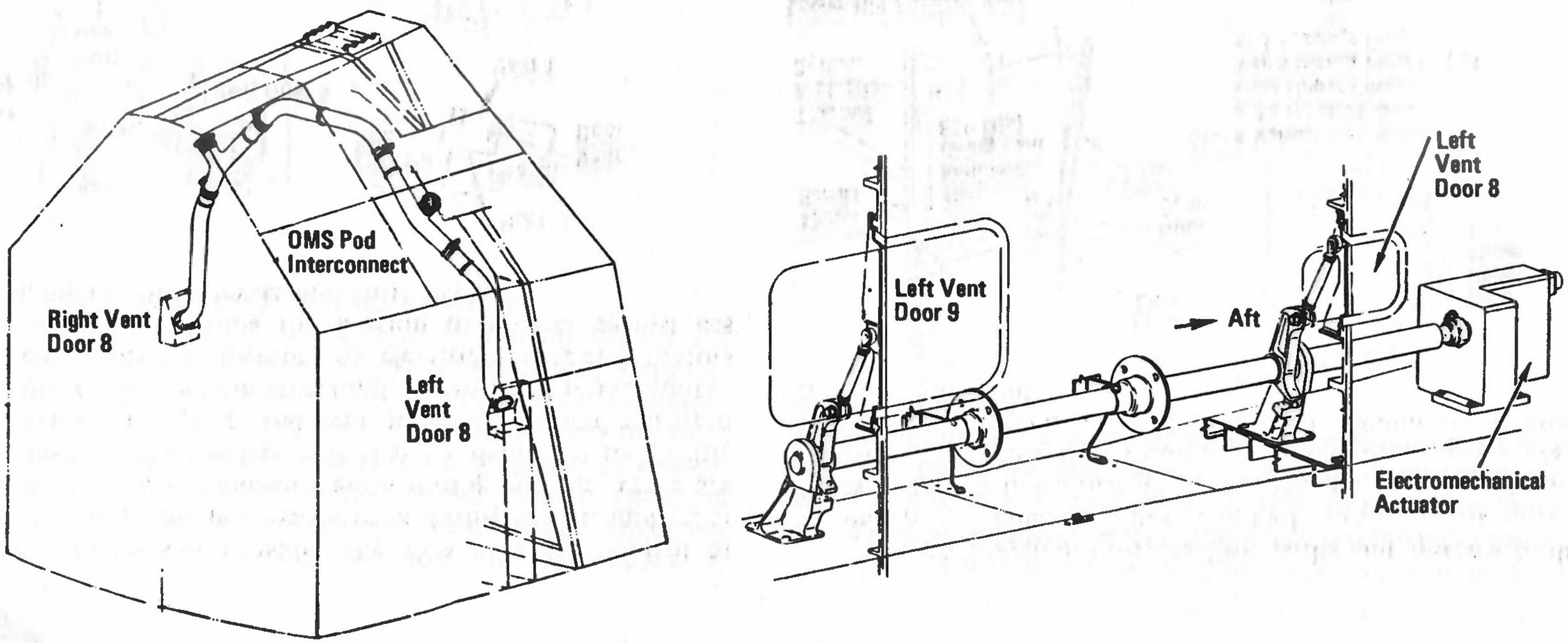


Vent Doors (4 & 7)

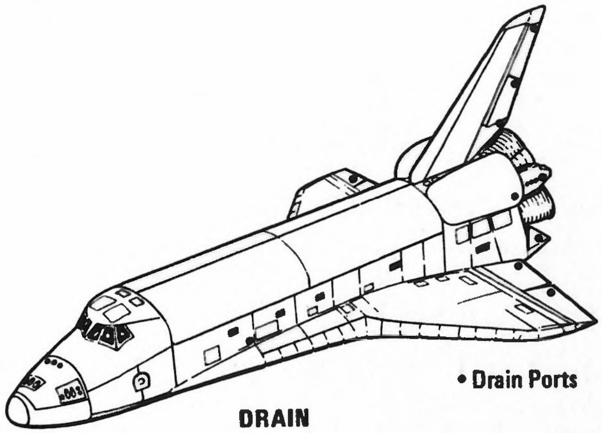
The purge and vent ducting is Kevlar/epoxy (115 pieces up to 27.94 centimeters [11 inches] in diameter) which replaced fiberglass or aluminum ducts. This reduced the weight by 33 percent. This is a reduction of approximately 90 kilograms (200 pounds).

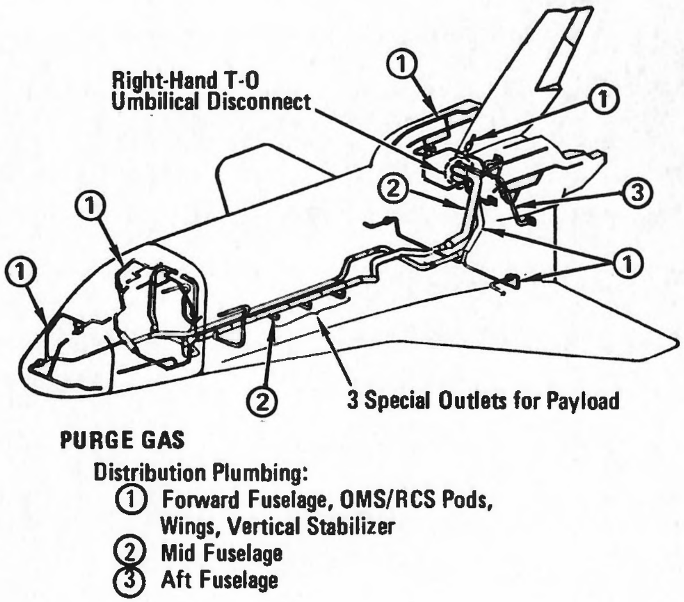
The window cavity conditioning system prevents moisture ingress into the windshields and overhead window and payload viewing window cavities and provides for depressurization and repressurization of these cavities during flight. This system also provides the purge conditioning (drying) of these areas during ground operations. The side hatch window is self-contained.

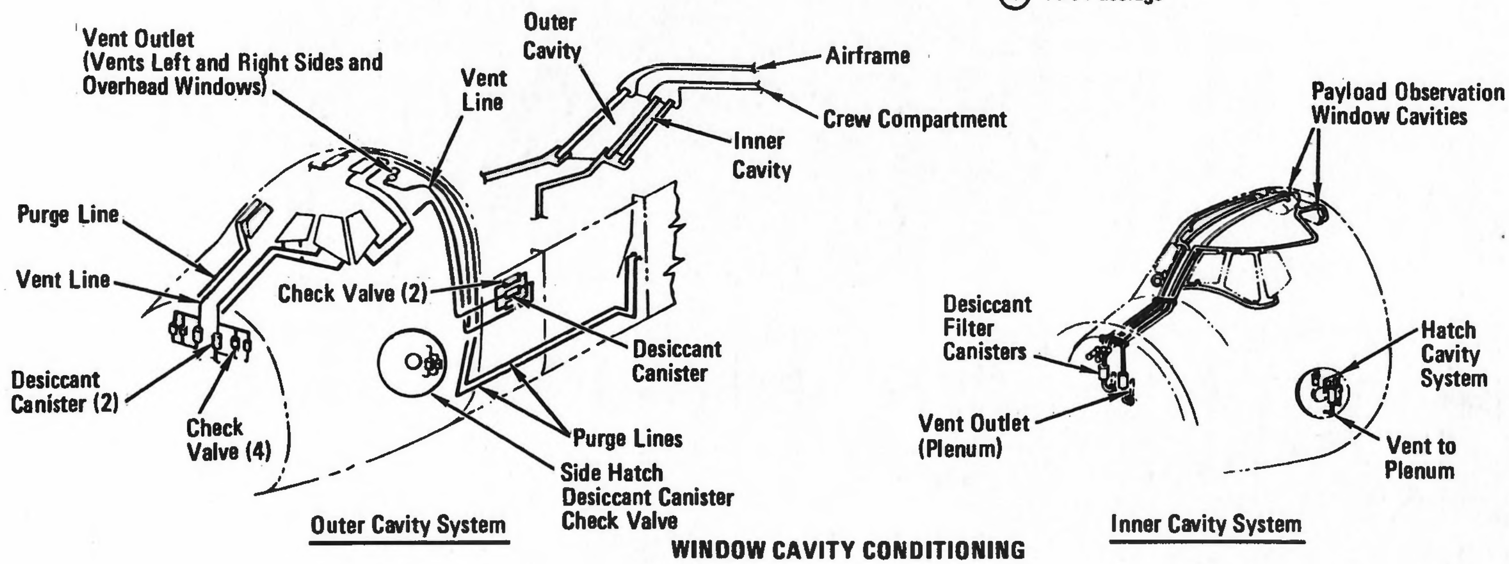
The hazardous gas detection system detects hazardous levels of explosive or toxic gases. The onboard orbiter sample lines duct the compartment gases to the ground support equipment at the T-O right-hand umbilical panel and to the ground-based mass spectrometer for analysis at the launch pad.



Vent Doors (8 & 9)







Window Cavity Conditioning System and Ground Drain and Purge System

THERMAL PROTECTION SYSTEM

The thermal protection system (TPS) consists of materials applied externally to the primary structural shell of the orbiter to maintain the orbiter airframe outer skin within acceptable temperature limits. The skins are constructed primarily of aluminum and/or graphite epoxy. During entry they must be protected from temperatures above 176°C (350°F). The TPS materials must be capable of performing a minimum of 100 missions in which temperatures will range from a minus 156°C (minus 250°F) in the cold soak of space to reentry temperatures that will reach nearly 1,648°C (3,000°F) on the wing leading edge and the nose cap. Interior compartment temperatures are controlled by internal insulation and heaters and through purging techniques.

The TPS is a passive system selected for stability at high temperatures and weight efficiency. Its applications are as follows:

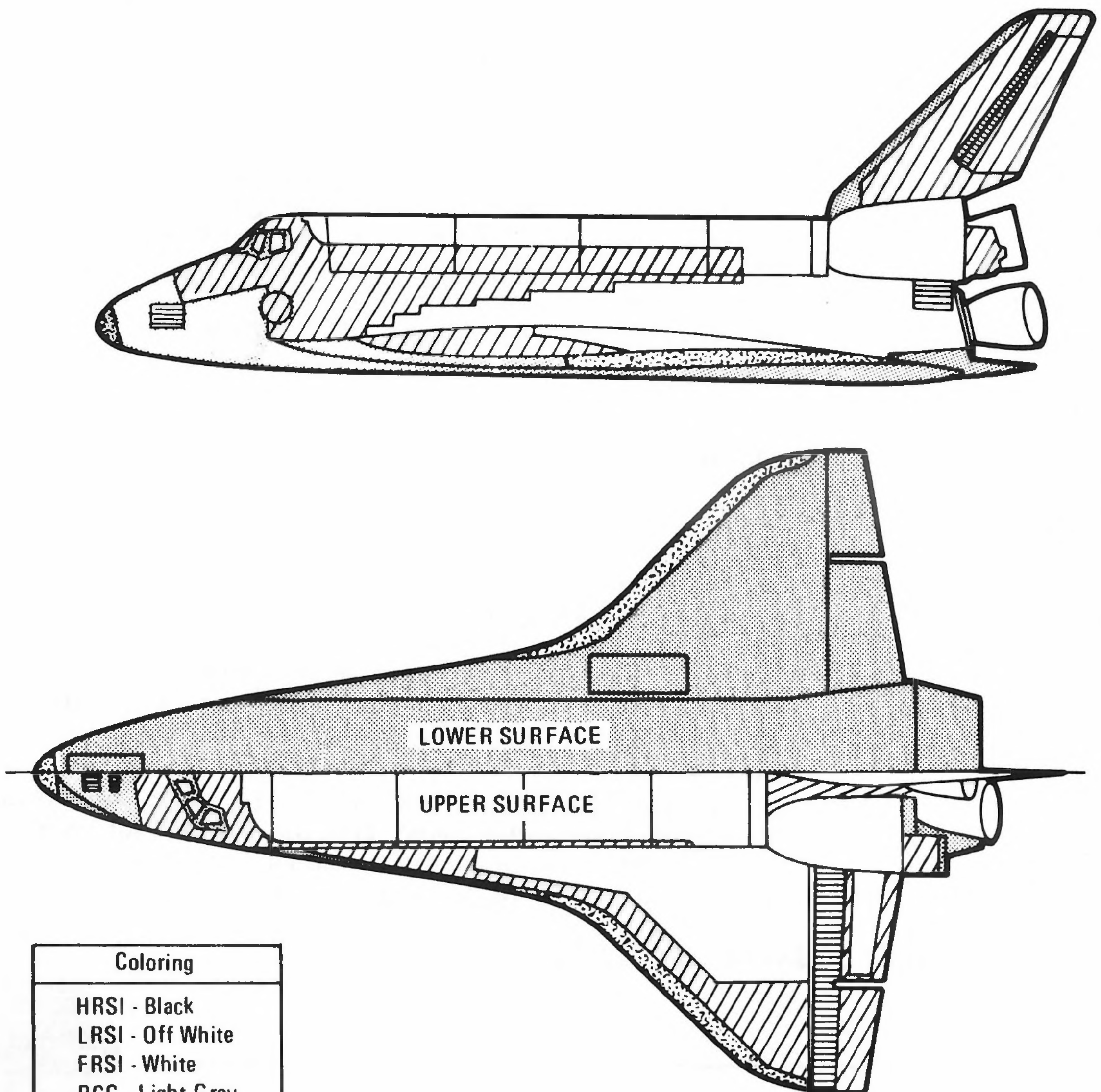
1. Coated Nomex felt reusable surface insulation (FRSI) is used where temperatures are less than 371°C (700°F) during entry and 398°C (750°F) during ascent. FRSI is used on the upper payload bay doors, mid and aft fuselage sides, upper wing, and orbital maneuvering system/reaction control system (OMS/RCS) pods.

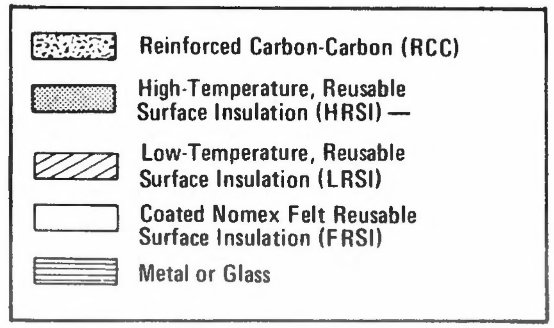
2. On Orbiter 102, the Columbia, low-temperature reusable surface insulation (LRSI) tiles are used where temperatures go below 648°C (1,200°F) and above 371°C (700°F) nominal. These areas are the lower portion of payload bay doors; forward, mid, and aft fuselage; upper wing; and vertical tail.

   On Orbiter 099, the Challenger, the LRSI tiles may be replaced with an advanced, flexible, reusable surface insulation (AFRSI), a quilted fabric blanket that improves producibility and durability, reduces fabrication and installation cost, reduces installation schedule time, and results in a weight reduction. Subsequent orbiters would also use AFRSI.

3. On Orbiter 102, the Columbia, high-temperature reusable surface insulation (HRSI) tiles are used where temperatures are below 1,260°C (2,300°F) and above 648°C (1,200°F). The areas are the forward fuselage, lower mid fuselage, lower wing, selected areas of the vertical tail, and around the forward fuselage windows. The HRSI has two different densities: one weighs 4 kilograms per cubic meter (9 pounds per cubic foot) and is used in all areas except around the nose and main landing gear doors, nose cap interface, wing leading edge, reinforced carbon-carbon/HRSI interface, external tank umbilical doors, vent doors, and vertical stabilizer leading edge. Those areas use HRSI tiles with a density of 9.9 kilograms per cubic meter (22 pounds per cubic foot).

   On Orbiter 099, the Challenger, some of the HRSI 22-pound-per-cubic-foot tiles may be replaced with fibrous refractory composite insulation (FRCI) HRSI tiles. The FRCI-12 tiles have a density of 5.4 kilograms per cubic meter (12 pounds per cubic foot). The FRCI HRSI tiles have improved strength, durability, and resistance to coating cracking. They also provide a weight reduction from that of the 99.7 percent-pure silica HRSI tiles. On Orbiters 103 and 104, FRCI-10 tiles may be used in place of the 9-pound-per-cubic-foot HRSI tiles. The FRCI-10 tiles have a density of 4.5 kilograms per cubic meter (10 pounds per cubic foot).



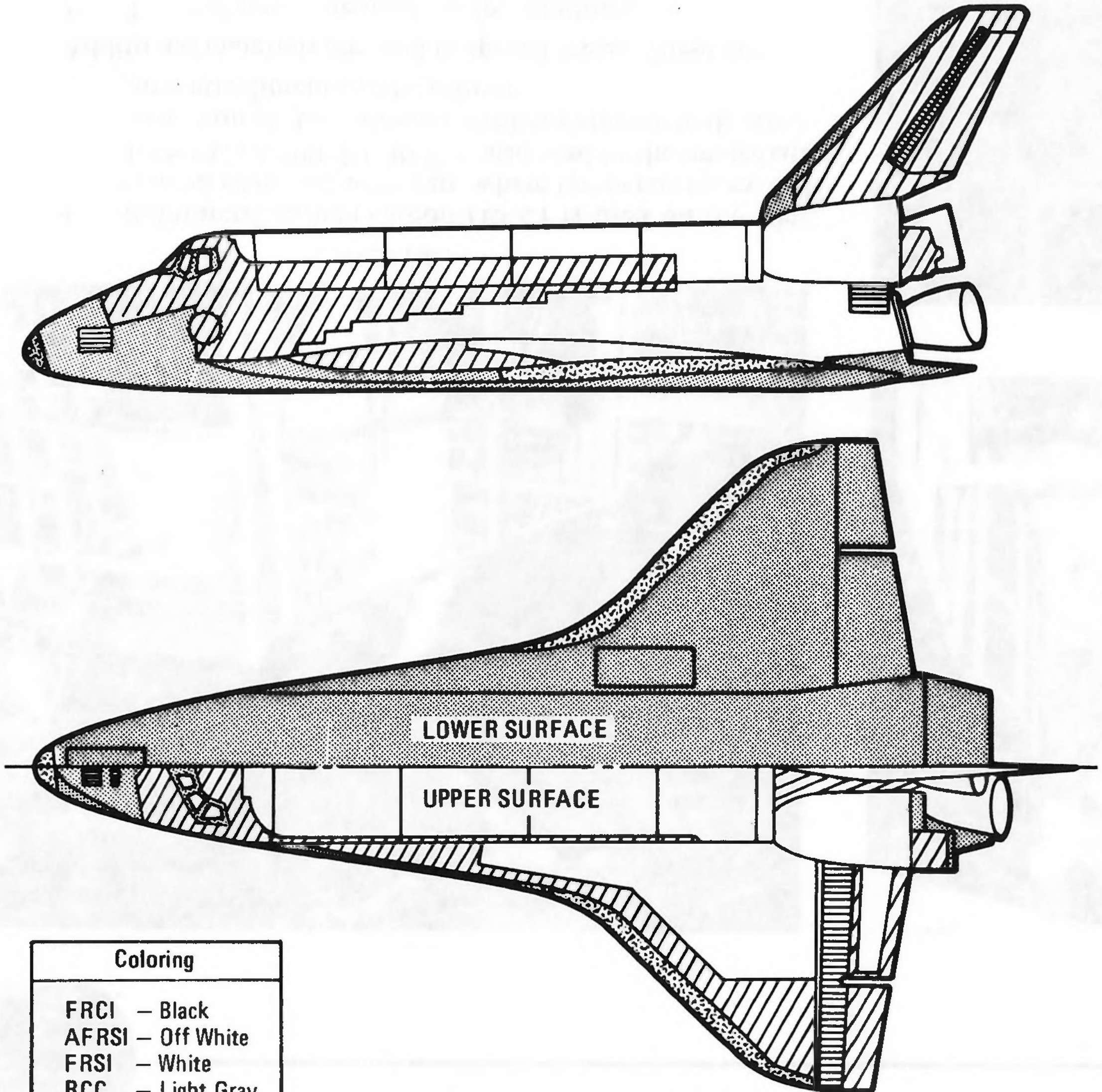


Element*	Area, sq m (sq ft)	Weight, kg (lb)
FRSI	332.7 (3581)	532.1 (1173)
LRSI	254.6 (2741)	1014.2 (2236)
HRSI	479.7 (5164)	4412.6 (9728)
RCC	38.0 (409)	1697.3 (3742)
Miscellaneous		918.5 (2025)
Total	1105.0 (11895)	8574.7 (18,904)

*Includes bulk insulation, thermal barriers, and closeouts

Coloring
HRSI - Black
LRSI - Off White
FRSI - White
RCC - Light Gray

Thermal Protection System, Orbiter 102



- Reinforced Carbon-Carbon (RCC)
- High-Temperature, Reusable Surface Insulation (HRSI)
  Fibrous Refractory Composite Insulation (FRCI)
- Low Temperature, Reusable Surface Insulation (LRSI)
  Advanced Flexible Reusable Surface Insulation (AFRSI)
- Coated Nomex Felt Reusable Surface Insulation (FRSI)
- Metal or Glass

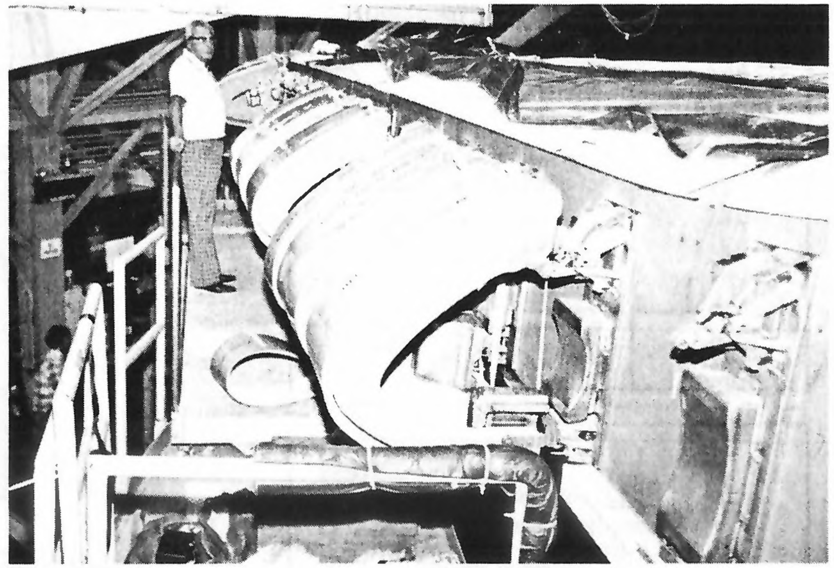
Element*	Area, sq m (sq ft)	Weight Kg (lb)
FRSI	332.7 (3581)	532.1 (1173)
LRSI**	TBD	TBD
AFRSI**-***	TBD	TBD
HRSI**	TBD	TBD
FRCI**-***	TBD	TBD
RCC	38.0 (409)	1697.3 (3742)
Miscellaneous		918.5 (2025)
Total	TBD	TBD

*Includes bulk insulation, thermal barriers, and closeouts
**Possibly some of Orbiter -099
***Orbiter 103 and subsequent
TBD - To Be Determined

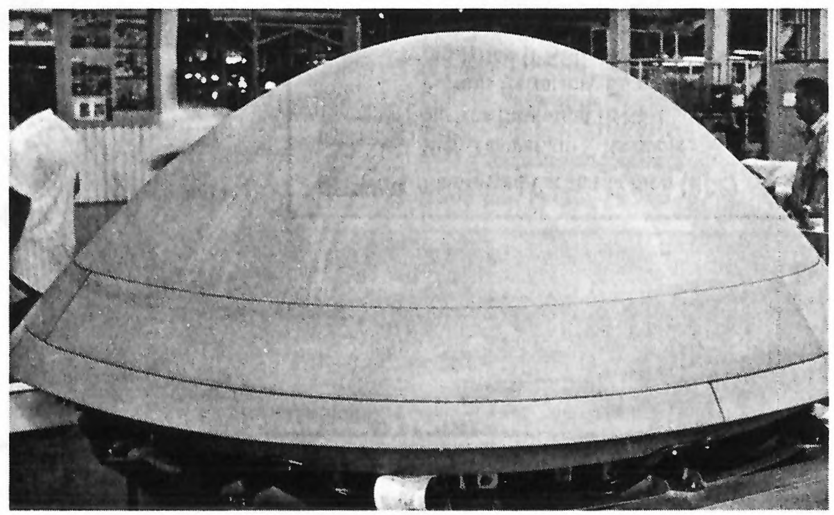
Coloring	
FRCI	– Black
AFRSI	– Off White
FRSI	– White
RCC	– Light Gray

Thermal Protection System, Orbiter 099 and Subsequent Orbiters

HRSI Tiles



RCC Wing



RCC Nose Cap

4. Reinforced carbon-carbon (RCC) is used on the wing leading edge and nose cap, where temperatures exceed 1,260°C (2,300°F). RCC is also used in the immediate area around the forward orbiter-external tank structural attachment on the orbiter.

Additional materials are used in special areas. These are:

1. Thermal panes are used for the windows.
2. Metal is used for the forward reaction control system fairings and elevon seal panels on the upper wing elevon interface.
3. A combination of white and black-pigmented silica cloth for thermal barriers and gap fillers is installed around operable penetrations such as main and nose landing gear doors egress/ingress side hatch, umbilical

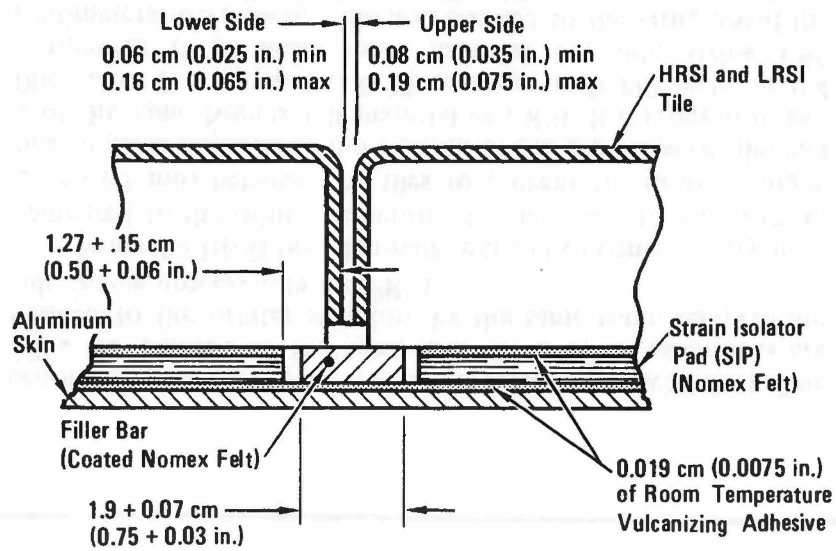
doors, elevons, forward RCS module and RCS thrusters, vent doors, payload bay doors, rudder/speed brake, and OMS/RCS pod and RCS thrusters, and gaps between tiles in high differential pressure areas.

The TPS has been designed for ease of maintenance and for flexibility of ground and flight operations while satisfying its primary function of maintaining acceptable airframe outer skin temperatures. The FRSI Nomex felt varies in thickness from 0.40 centimeter (0.160 inch) to one centimeter (0.40 inch) thick and consists of sheets 0.9 by 1.2 meters (3 to 4 feet) (except for closeout areas) bonded directly to the orbiter exterior. The felt is coated with a white pigmented silicone elastomer to waterproof the felt and to provide the required thermal and optical properties. The FRSI provides an emittance of 0.8 and solar absorptance of 0.32. FRSI covers nearly 50 percent of the orbiter's upper surface.

The felt is a basic Nomex (aramid) fiber. The fibers are two denier 7.62 centimeters (3 inches) long and crimped. The fibers are loaded into a carding machine, which untangles the clumps of fibers and "combs" them to make a tenous mass of lengthwise-oriented, relatively parallel fibers called a web. The cross-lapped web is fed into a loom, where it is lightly needled into a batt. Generally, two such batts are placed face to face, where they are needled togehter to form felt. The felt is then subjected to a multineedle-pass process until the desired strength is reached. The needled felt is then calendered to stabilize thickness 0.40 centimeters (0.16 inch) to one centimeter (0.40 inch) by passing it through heated rollers at selected pressures. The calendered material is then heat-set at approximately 260°C (500°F) to thermally stabilize the felt.

The FRSI is bonded to the orbiter surface by a room temperature vulcanizing silicon adhesive. The silicon adhesive glue is applied at 0.02 centimeters (0.008 inches) thick. The very thin glue line reduces weight and minimizes the thermal expansion during temperatures of 260°C (500°F) at the glue line (entry) and temperatures below minus 112°C (170°F) on orbit. The orbiter structure could be as low as minus 121°C (minus 250°F). The FRSI bond is cured at room temperature, with vacuum bags used to apply pressure.

On Orbiter 102, and possibly some portions of Orbiter 099, the HRSI tiles are nominally 15.24 by 15.27 centimeters (6 by 6 inches). The tiles are made of a low-density (lightweight), high-purity silica (glass) 99.7 percent amorphous fiber (fibers derived from common sand, one to two mils thick) insulation that is made rigid by ceramic (clay) bonding. Ninety percent of the tile is void, and ten percent is material that results in the tiles weighing 4 kilograms per cubic meter (9 pounds per cubic foot). However, the areas around the nose and main landing gear doors, external tank umbilical doors, vent doors, and vertical stabilizer leading edge utilize tiles that have a density of 9.9 kilograms per cubic meter (22 pounds per cubic foot). A slurry containing fibers mixed with water is frame-cast to form soft, porous blocks to which collodial silica binder solution is added. When sintered,



HRSI and LRSI Tile Interface, Orbiter 102 and Possibly Some of Orbiter 099 a rigid block is produced, which is then cut into quarters and then machined to the precise dimensions required for individual tiles.

The HRSI tiles will vary in thickness: 2.54 centimeters (1 inch) to 12 centimeters (5 inches) to minimize the orbiter weight and not permit the orbiter structure to see more than 176°C (350°F). The tiles will vary slightly in sizes and shapes at the closeout areas. The tile thickness also provides adequate on-orbit space cold soak protection, and the tiles must withstand repeated heating and cooling, plus extreme acoustic environments (165 decibels at launch) in some local areas. Resistance to thermal shock is very good. The material can be taken from a 1260°C (2300°F) oven and immersed in cold water without damage. Surface heat dissipates so quickly that an uncoated tile can be held by its edges with an ungloved hand seconds after removal from the oven and while the tile interior still glows red hot. The HRSI tiles are coated on the top and sides with a glass mixture formed by mixing tetra-silicide with boro-silicate glass in a powder with a liquid carrier and sprayed on the tile to a coating thickness of 16 to 18 mills. The coated HRSI tiles are then placed in an oven and heated to a temperature of 1260°C (2300°F). This results in a black waterproof glossy coating covering the tile which has a surface emittance of 0.85 and a solar absorptance of about 0.85. In addition, the silica fibers are treated with a silicone resin after the ceramic coating heating process to provide bulk waterproofing.

Over 20,000 HRSI tiles are used on the bottom portion of the orbiter and nose areas of the orbiter in addition to other selected areas of the orbiter. The HRSI tiles cannot withstand airframe load deformation; therefore, stress isolation is necessary between the tiles and the orbiter structure. This isolation is provided by strain isolation pads (SIP's). The SIP's isolate the HRSI tiles from the orbiter's structural deflections, expansions, and acoustic excitation. They thereby prevent stress failure in the tiles. The SIP is a thermal isolator. The SIP's are made of the same Nomex felt material as the FRSI and are either 0.23 centimeter (0.09 inch) or 0.40 centimeter (0.16 inch) thick. The SIP's are bonded to the tiles, and the SIP/tile assemblies are bonded to the orbiter structure by the same room temperature vulcanizing process as in the FRSI.

Since the HRSI tiles thermally expand or contract very little compared to the orbiter structure, it is necessary to leave gaps of 25 to 65 mils between the tiles to prevent tile-to-tile contact. Insulation is required in the bottom of the gap between tiles and is of the same Nomex felt material as FRSI. It is referred to as a filler bar. The material is 0.23 centimeter (0.090 inch) or 0.4 centimeter (0.16 inch) thick material cut into strips 1.90 centimeters (0.75 inch) wide and bonded to the structure at the same time as the HRSI SIP pads. The filler bar is waterproof and temperature resistant up to approximately 426°C (800°F), top side exposure.

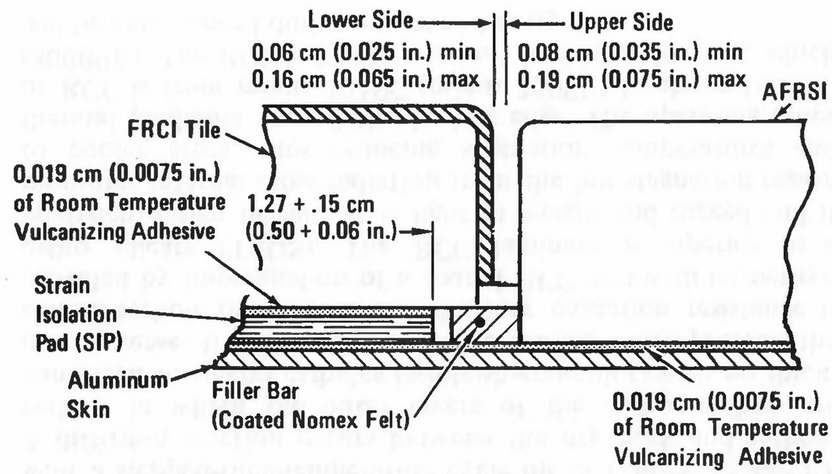
The LRSI and HRSI tiles that were removed from Orbiter 102 at the Kennedy Space Center were replaced with HRSI tiles that are densified. The densification process was required due to fiber polarization in the SIP manufacture, which resulted in stress concentrations on the SIP/tile bond interface. The densification process utilizes a Ludox AS, which is an ammonia-stabilized binder. When mixed with silica slip particles, it becomes a cement. When mixed with water, it drys to a finished hard surface. A silica-tetraboride coloring agent is mixed with the compound for penetration identification. The pigmented Ludox slip slurry is brush-painted with several coats on the tile and allowed to air-dry for 24 hours. A heat treatment and other processing follow prior to installation. The densification coating only penetrates the 4-kilogram-per-cubic-meter HRSI tiles about 0.27 centimeters (0.11 inches) and 0.17 centimeters (0.07 inches) for the 9.9-kilogram-per-cubic-meter tiles, yet the strength and stiffness of the tile/SIP system are increased by a factor of two. Densified tiles will be used on Orbiter 099.

On Orbiter 099, the Challenger, some of the FRCI-12 tiles may replace the HRSI 9.9-kilogram-per-cubic-meter (22-pound-per-cubic-foot) tiles. On subsequent orbiters, the FRCI-12 tiles replace all the 9.9-kilogram-per-cubic-meter HRSI tiles. The FRCI-12 tiles have a density of 5.4 kilograms per cubic meter (12 pounds per cubic foot). On Orbiters 103 and 104, FRCI-10 HRSI tiles may replace the HRSI 4-kilogram-per-cubic-meter tiles. The FRCI-10 HRSI tiles have a density of 4.5 kilograms per cubic meter (10 pounds per cubic foot).

The FRCI-12 and FRCI-10 HRSI tiles were developed by NASA's Ames Research Center, Mountain View, California, and are being manufactured by Lockheed Missiles and Space Division, Sunnyvale, California, the same manufacturer of the original 99.7 percent-pure silica HRSI tiles.

The FRCI-12 and FRCI-10 HRSI tiles are a higher strength tile derived by adding AB 312 (alumino boro silicate fiber) called Nextel developed by the 3M Company in St. Paul, Minnesota to the 80 percent pure silica tile slurry. The Nextel activates boron fusion and figuratively welds the micron-size fibers of pure silica into a rigid structure during sintering in a high-temperature furnace. The resulting composite fiber refractory material composed of 20 percent Nextel and 80 percent silica fiber has entirely different physical properties than the original HRSI 99.7-percent pure silica tiles. The Nextel, with an expansion coefficient 10 times that of the 99.7-percent pure silica, acts like a preshrunk concrete reinforcing bar in the fiber matrix.

One characteristic of the FRCI-12 and FRCI-10 HRSI tiles places the reaction cured glass (black) coating into a compression as it is cured. This sharply reduces the sensitivity to cracking of coating during handling and operations. In addition to the improved coating compatibility, the FRCI-12 and FRCI-10 HRSI tiles are about 10 percent lighter than the HRSI 99.7-percent pure silica tiles. This could reduce weight about 10 percent. The FRCI-12 and FRCI-10 HRSI tiles have also demonstrated a tensile (pull) strength at least three times greater than that of the HRSI 99.7-percent pure silica tiles and a use temperature approximately 37°C (100°F) higher.



FRCI Tile and AFRSI Interface, Possibly Some of Orbiter 099, Orbiter 103 and Subsequent

The FRCI-12 and FRCI-10 HRSI tile manufacturing process is essentially the same as the 99.7-percent pure silica HRSI tiles, the only change being in the "wet end" prebinding of the slurry before it is cast. It also requires a higher sintering temperature, which could mean new furnaces are required. When dried, a rigid block is produced as in the 99.7-percent pure silica HRSI tile. These blocks will be cut into quarters and then machined to the precise dimensions required for each tile. The FRCI-12 and FRCI-10 tiles are the same 15.24- by 15.24-centimeters (6 by 6 inch) tiles and vary in thickness from 2.54 centimeters (1 inch) to 12 centimeters (5 inches). They will also vary in size and shape at the closeout areas as in the 99.6-percent pure silica HRSI tiles. The FRCI-12 and FRCI-10 HRSI tiles are bonded to the orbiter essentially the same as the 99.7-percent silica HRSI tiles.

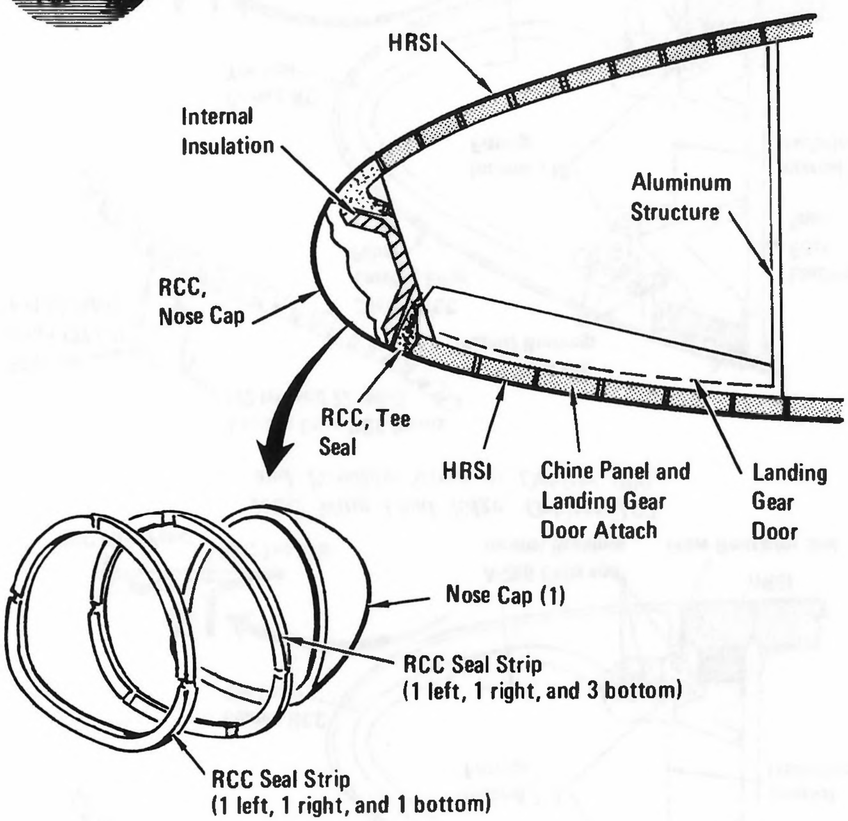
On Orbiter 102, the Columbia, the 99.7-percent pure silica LRSI tiles are the same construction and have the same basic functions as the 99.7-percent pure silica HRSI tiles. They are thinner [0.5 to 3.5 centimeters (0.2 to 1.4 inches)] than the HRSI tiles to minimize orbiter weight. The 99.7-percent pure silica LRSI tiles are manufactured in the same manner as the 99.7-percent pure silica HRSI tiles, except that the tiles are 20 by 20 centimeters (8 by 8 inches) and a white optical and moisture-resistant coating is applied ten mills thick to the top and side of the LRSI tiles. The white coating provides additional on-orbit thermal control for the orbiter. The coating comprises primarily silica compounds with shiny aluminum oxide to obtain optical properties. The coated 99.7-percent pure silica LRSI tiles are treated with bulk waterproofing similar to the 99.7-percent pure silica HRSI tiles. The 99.7-percent pure silica LRSI tiles are installed on the orbiter in the same manner as the 99.7-percent pure silica HRSI tiles (room temperature vulcanizing, SIP, filler bar, and vacuum bond-cure). The 99.7-percent LRSI has a surface emittance of 0.8 and a solar absorptance of 0.32. More than 7,000 99.7-percent pure silica LRSI tiles are used.

On Orbiter 099, the Challenger, some of the 99.7 percent-pure silica LRSI tiles may be replaced with an advanced flexible reusable surface insulation (AFRSI) quilted fabric blanket; Orbiter 103 and subsequent orbiters would use AFRSI. The AFRSI consists of a silica glass cloth sheet with the low-density, high-purity silica (glass), 99.7 percent amorphous fibers (fibers derived from common sand and 1 to 2 mils thick) packed between the two silica glass cloth sheets and sewn with quartz thread, which provides a quilt-like look. The AFRSI is treated with a moisture-resistant coating to provide waterproofing. The AFRSI density is 2.7 kilograms per cubic meter (6 pounds per cubic foot) and varies in thickness from 0.31 to 1.27 centimeters (0.125 to 0.5 inches). The AFRSI quilted fabric blanket sheets are bonded directly to the orbiter structure by a room temperature vulcanizing silicone adhesive. The silicone adhesive glue is applied at a thickness of 0.02 centimeters (0.010 inches). The very thin glue line reduces weight and minimizes the thermal expansion during temperature changes.

The nose cap and the leading edges of the orbiter wing utilize reinforced carbon-carbon (RCC) panels to maintain airfoil shape above 1260°C (2300°F). The wing leading edge is made up of 44 RCC panels (22 each wing), whereas the nose cap is one piece. RCC fabrication begins with a nylon cloth graphitized and impregnated with a phenolic resin. This impregnated cloth is layed up as a laminate and cured in an autoclave. After cure, the laminate is pyrolized (taking the resin out) at high temperature to convert the resin to carbon. The part is then impregnated with furfural alcohol in a vacuum chamber, then cured and pyrolized again to convert furfural alcohol to carbon. This process is repeated three times until the required reinforced carbon-carbon is achieved.

For it to become an oxidation-resistant coating, the material is packed in a retort with a dry pack material made up of a mixture of alumina, silicon, and silicon carbide. The retort is placed in a furnance, and the coating process takes place in argon with a stepped-time-temperature cycle up to 1760°C (3200°F). A diffusion reaction occurs between the dry pack and carbon-carbon in which the outer layers of the carbon-carbon are converted to silicon carbides (whiteish-grey color) with no thickness increase. It is this silicon-carbide coating which protects the carbon-carbon from oxidation. Further oxidation resistance is provided by impregnation of a coated RCC part with tetraethyl-ortho silicate (TEOS). The RCC laminate is superior to a sandwich design because it is light in weight and rugged and it promotes internal cross radiation from the hot stagnation region to cooler areas, thus reducing stagnation temperatures and thermal gradients around the leading edge. The operating range of RCC is from minus 121°C (minus 250°F) to about 1648°C (3000°F). The RCC is highly resistant to fatigue loading, which will be experienced during ascent and entry.

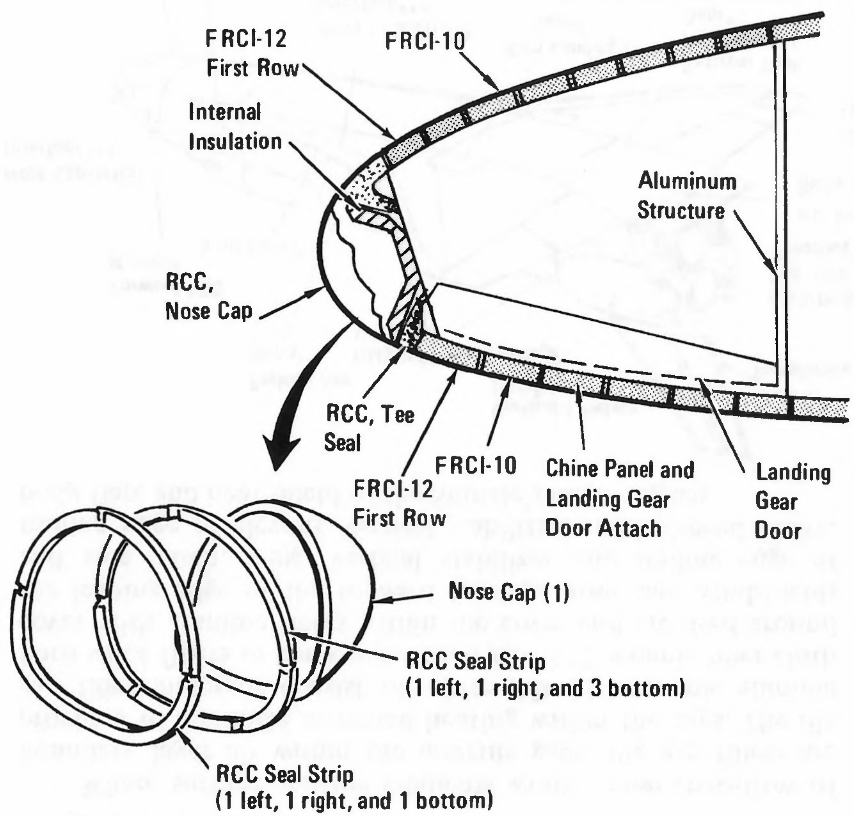
The RCC panels are mechanically attached to the wing with a series of floating joints to reduce loading on the panels due to wing deflections. The seal between each wing leading edge panel is referred to as a "tee" seal. The "tee" seals allow lateral motion and for thermal expansion differences between the RCC and the orbiter wing cooler structure, behind the leading edge. In



RCC Nose Cap, Orbiter 102 and Possibly Some Orbiter 099

addition, it prevents the direct flow of hot boundary layer gases into the wing leading edge cavity during entry. The "tee" seals are constructed of RCC.

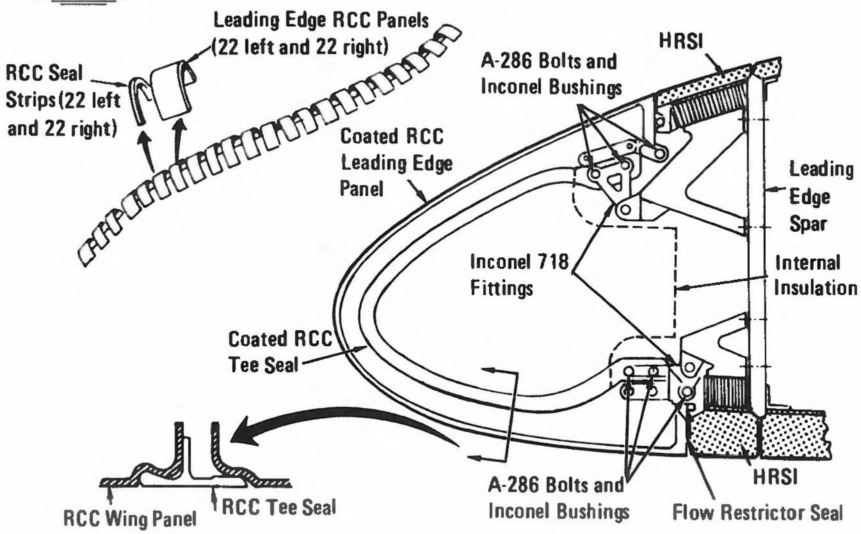
Since carbon is not a good insulator, the adjacent aluminum and the metallic attachments are protected from exceeding temperature limits by internal insulation. Inconel 718 and A-286 fittings are bolted to flanges on the RCC components and attached to the aluminum wing spars and nose bulkhead. Inconel-covered dynaflex insulation protects the metallic attach fittings and spar from the heat emitted from the inside surface of the RCC wing panels. The nose cap thermal insulation utilizes a



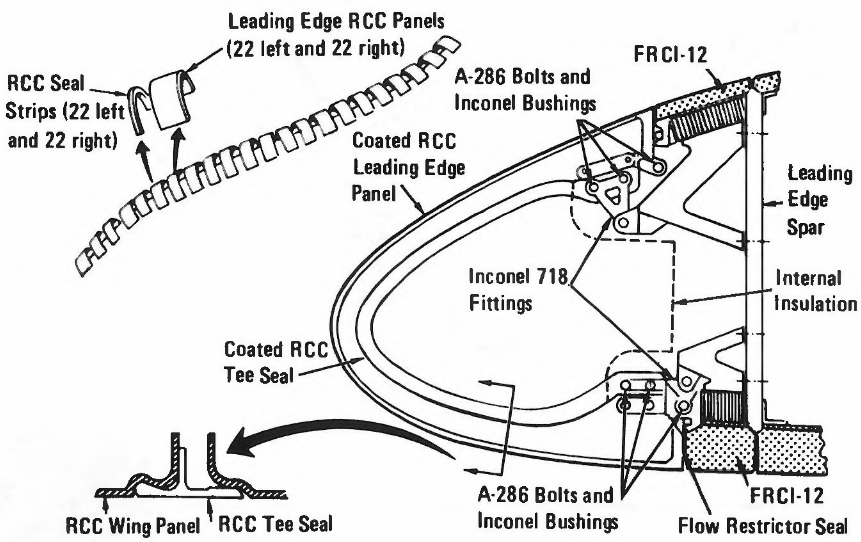
RCC Nose Cap, Possibly Some of Orbiter 099, Orbiter 103 and Subsequent

blanket made from ceramic fibers and filled with silica fibers and some 99.7-percent pure silica HRSI tiles on Orbiter 102 or possibly some FRCI tiles on Orbiter 099 and FRCI tiles on all subsequent orbiters to protect the forward fuselage from the heat emitted from the hot inside surface of the RCC.

Thermal barriers are utilized in the closeout between various components of the orbiter and the TPS such as the forward RCS and aft RCS, rudder/speedbrake, nose and main landing gear doors, crew hatch, vent doors, external tank umbilical doors, vertical stabilizer/aft fuselage interface, payload bay doors, wing



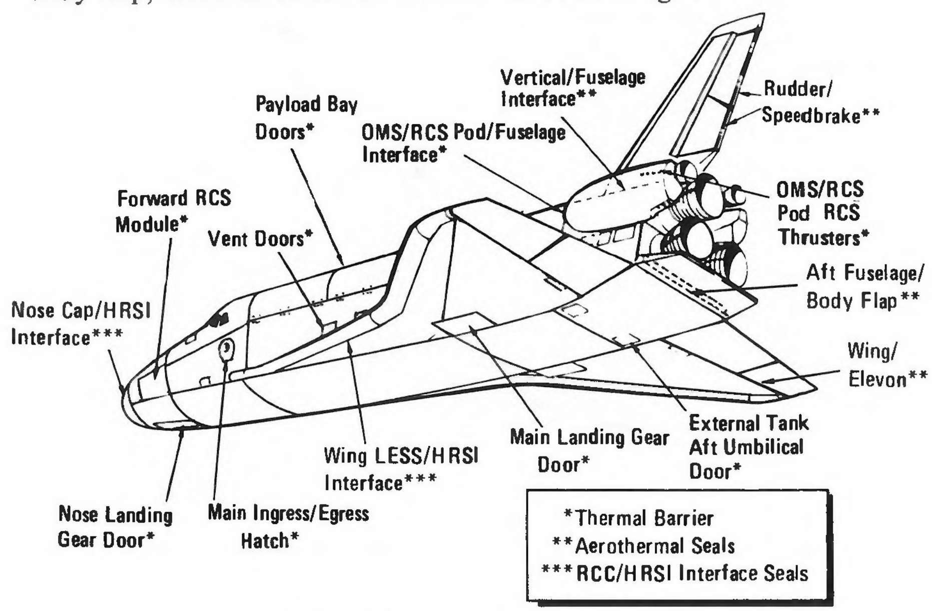
RCC Wing Lead Edge, Orbiter 102 and Possibly Some of Orbiter 099



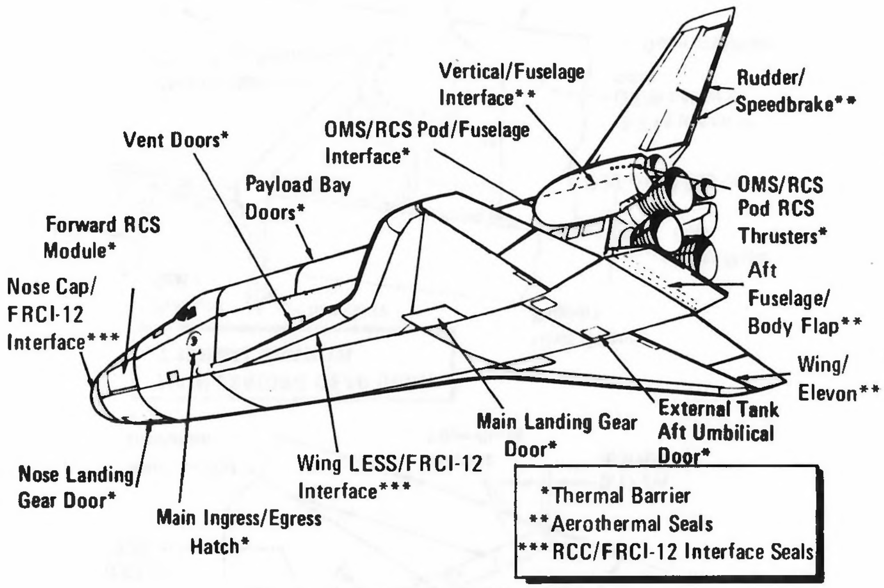
RCC Wing Leading Edge, Possibly Some of Orbiter 099, Orbiter 103 and Subsequent

leading edge RCC/HRSI interface, and nose cap/HRSI interface. The various materials utilized are white AB 312 ceramic alumina boro silica fibers or black pigmented AB 312 ceramic fiber cloth braided around an inner tubular spring made from Inconel X750 wire with silica fibers within the tube, alumina mat, quartz thread, and Macor-machinable ceramic.

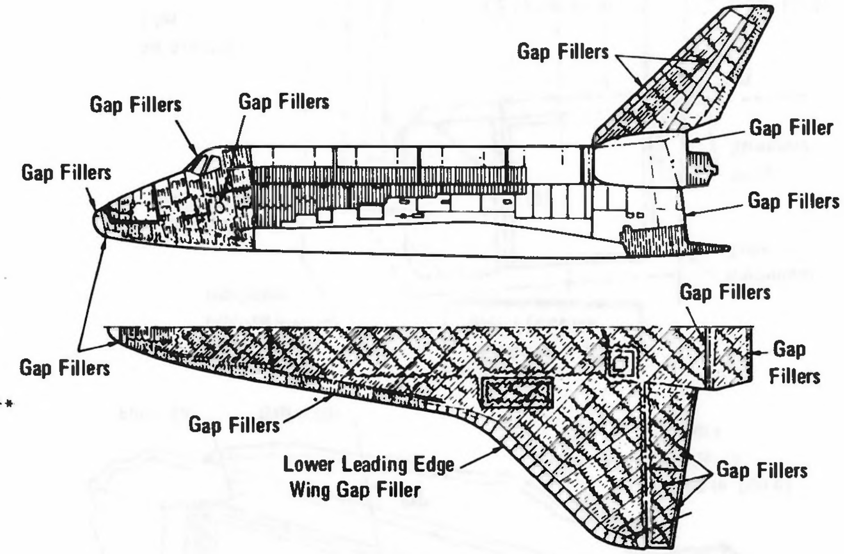
Where surface pressure gradients would cause cross-flow of boundary layer air within the intertile gaps, tile gap fillers are provided to minimize increased heating within the gaps. The tile gap filler materials consist of white AB 312 ceramic alumina boro silica fibers or black pigmented AB 312 ceramic fiber cloth cover with alumina fibers within the cover and are used around the leading edge of the forward fuselage nose cap, windshields and side hatch, wing, vertical stabilizer and trailing edge of trailing edge of elevons, vertical stabilizer, rudder/speed brake, body flap, and heat shield of the Shuttle's main engines.



Thermal Barriers, Orbiter 102



Thermal Barriers, Orbiter 099 and Subsequent Orbiters

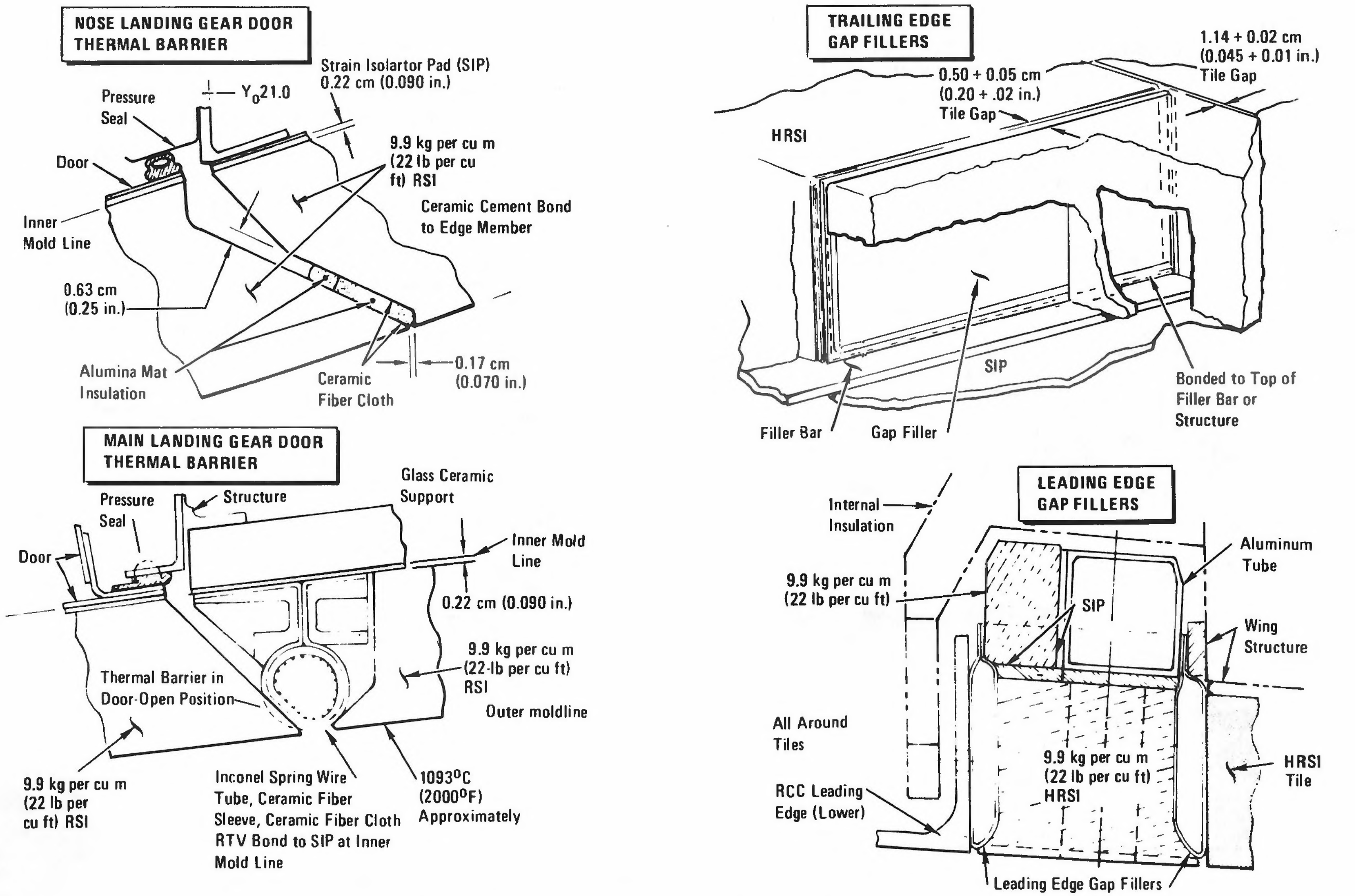


TPS Tile Gap Filler Locations

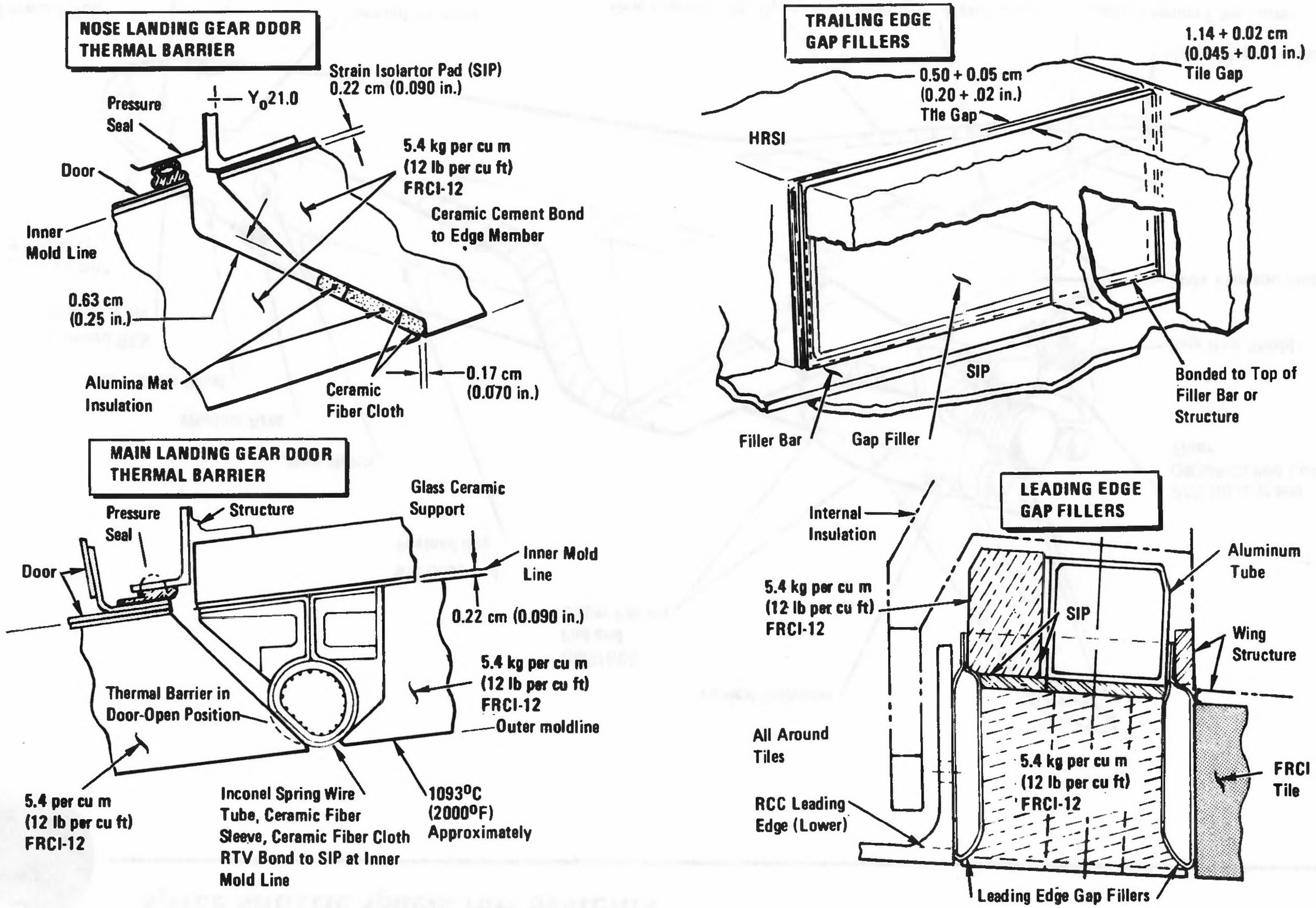
Fused silica threaded inserts and plugs are used in the tiles to provide access through the tiles to remove door or access panel attachments.

The split segments of the elevons (outboard end of each inboard elevon and inboard end of each outboard elevon) utilizes an ablative material that is replaced after each flight. These areas of the elevons could reach approximately 1815°C (3300°F) during entry. The ablative material is similar to that used on the Apollo Command Module heat shields. It consists of an phenolic honeycomb core arrangement with a phenolic resin (a type of reinforced plastic). The heat shield is covered with a black polyurethane paint. This material chars and evaporates during the entry heat load and prevents the heat from reaching the elevon aluminum skin. The ablator segments are attached to the elevon with fasteners and threaded ablative plugs are used to provide access to the fasteners for installation and removal. The weight of each ablator is approximately 25.85 kilograms (57 pounds) each for a total weight of approximately 103.4 kilograms (228 pounds). Alternate materials are being studied for future flights such as; RCC, and a lightweight ablator. The elevon ablator contractor is AVCO Specialty Materials Division, Lowell, MA.

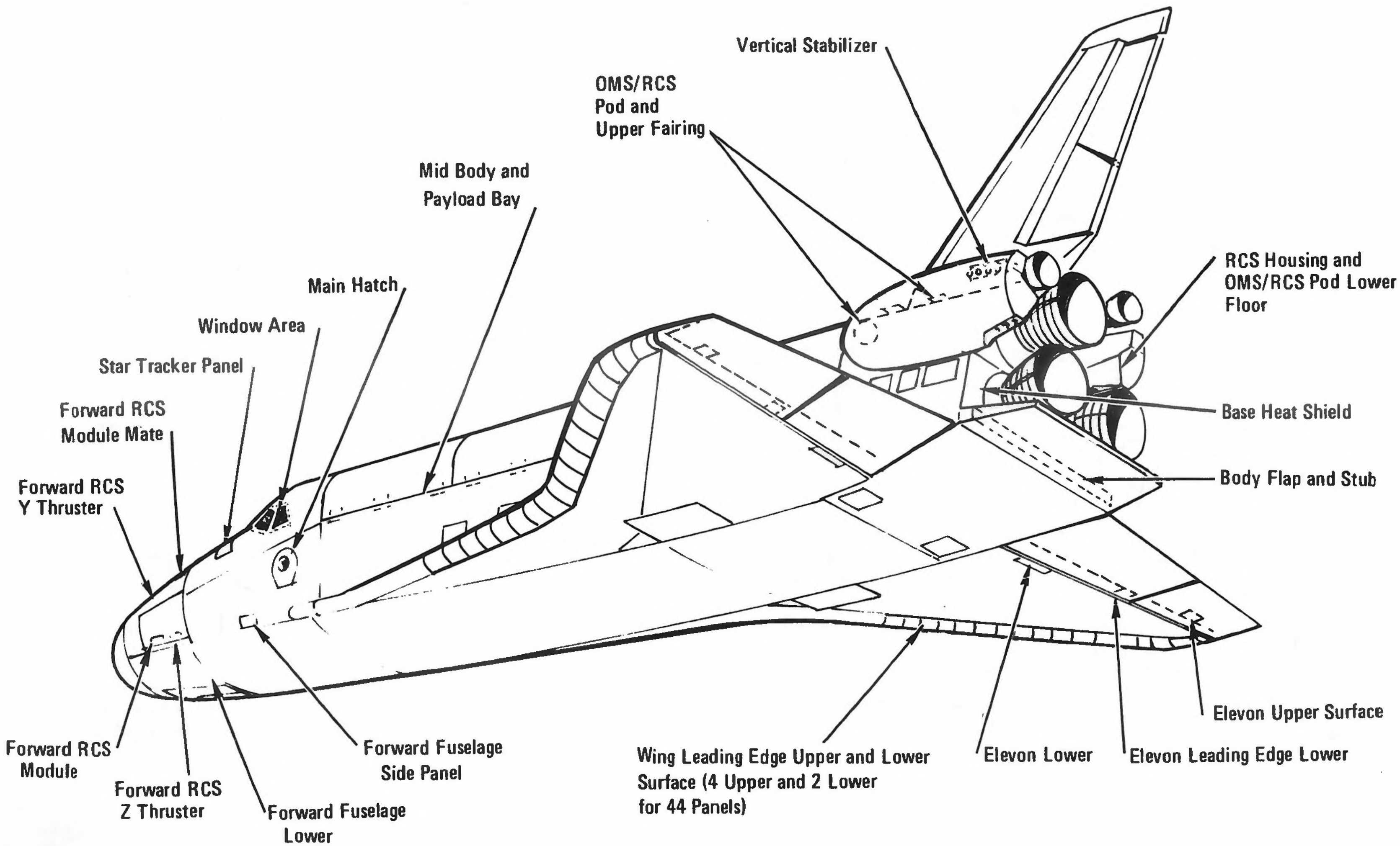
The contractors are Vought Corporation, Dallas, TX (RCC); Lockheed Missiles and Space Co., Inc., Sunnyvale, CA (HRSI and LRSI tiles and HRSI-FRCI-10 and -12 tiles); Albany International Research Co., Dedham, MA (Nomex felt); General Electric, Waterford, NY (room temperature vulcanizing adhesive); 3M Company, St. Paul, MN (AB 312 fibers); Santa Fe Textiles, Santa Ana, CA (Inconel 750 wire spring and fabric sleeving); ICI United States, Inc., Wilmington, DE (alumina mat); J.P. Stevens Co., Los Angeles (quartz thread); Corning Glass Works, Corning, NY (Macor-machinable glass ceramic); Velcro Corp., NY, NY (Velcro hooks and loops); and Prodesco, Perkasie, PA (fibrous pile-S glass).



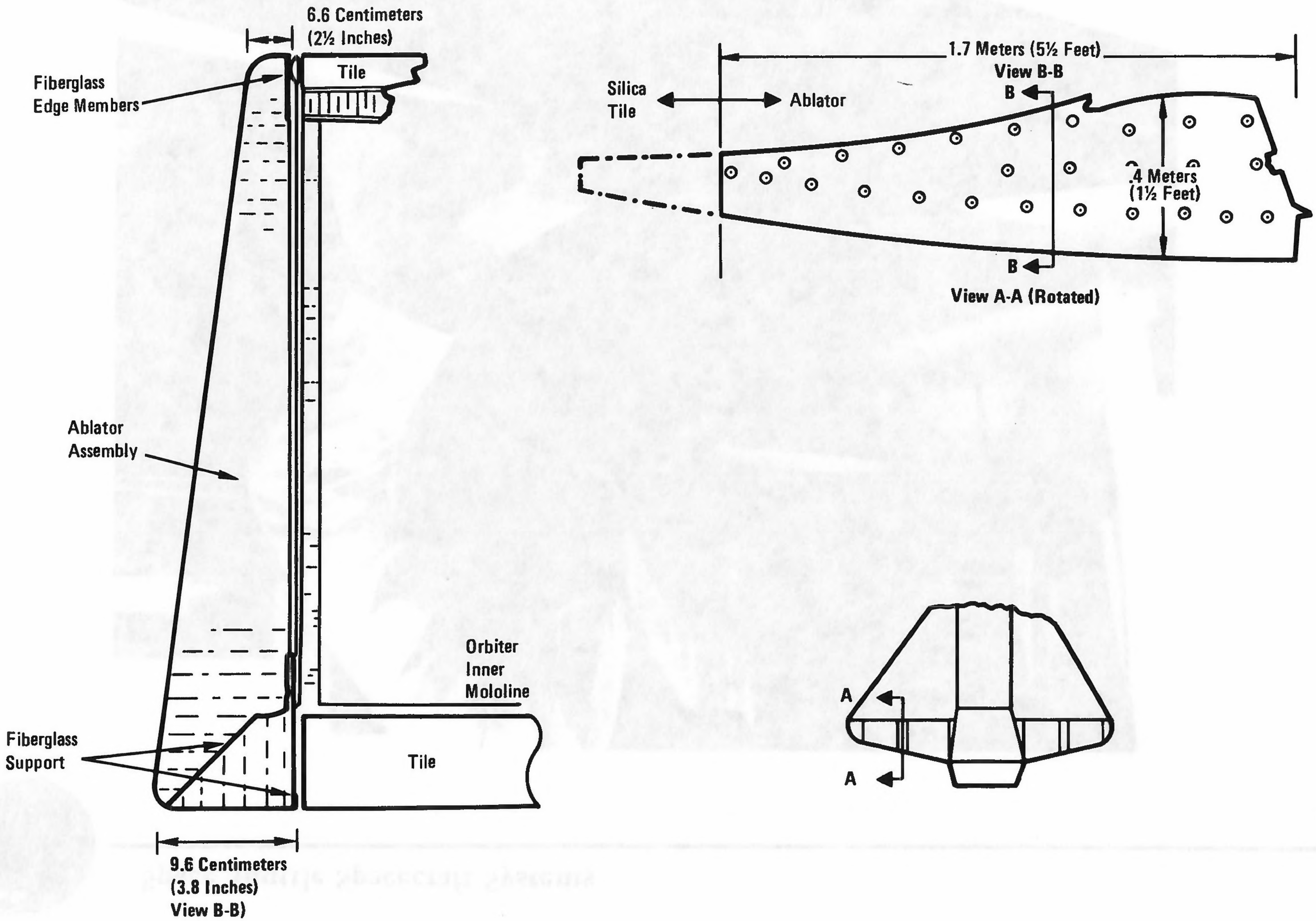
Typical TPS Gap Fillers and Thermal Barriers, Orbiter 102



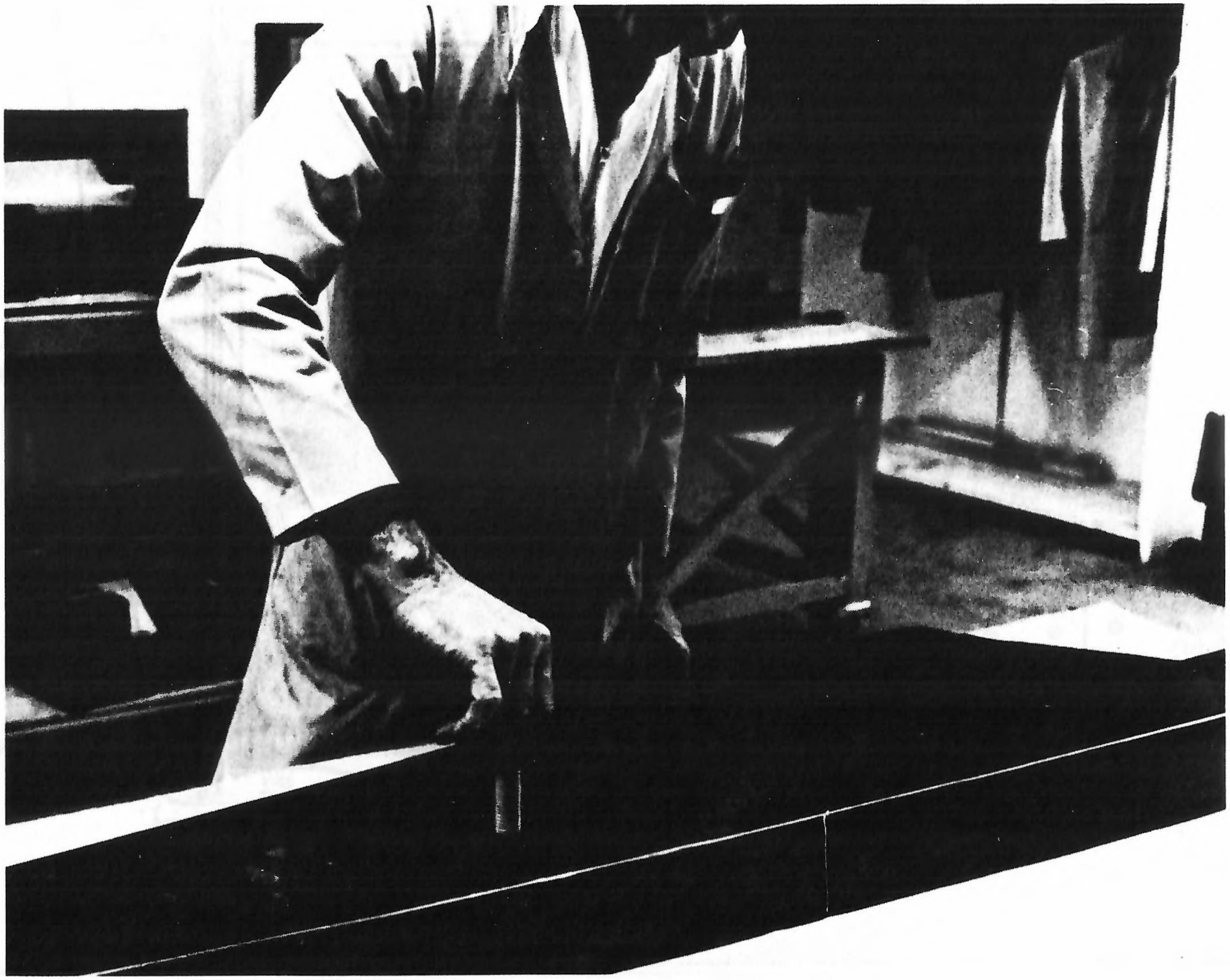
Typical TPS Gap Fillers and Thermal Barriers, Orbiter 099 and Subsequent Orbiters

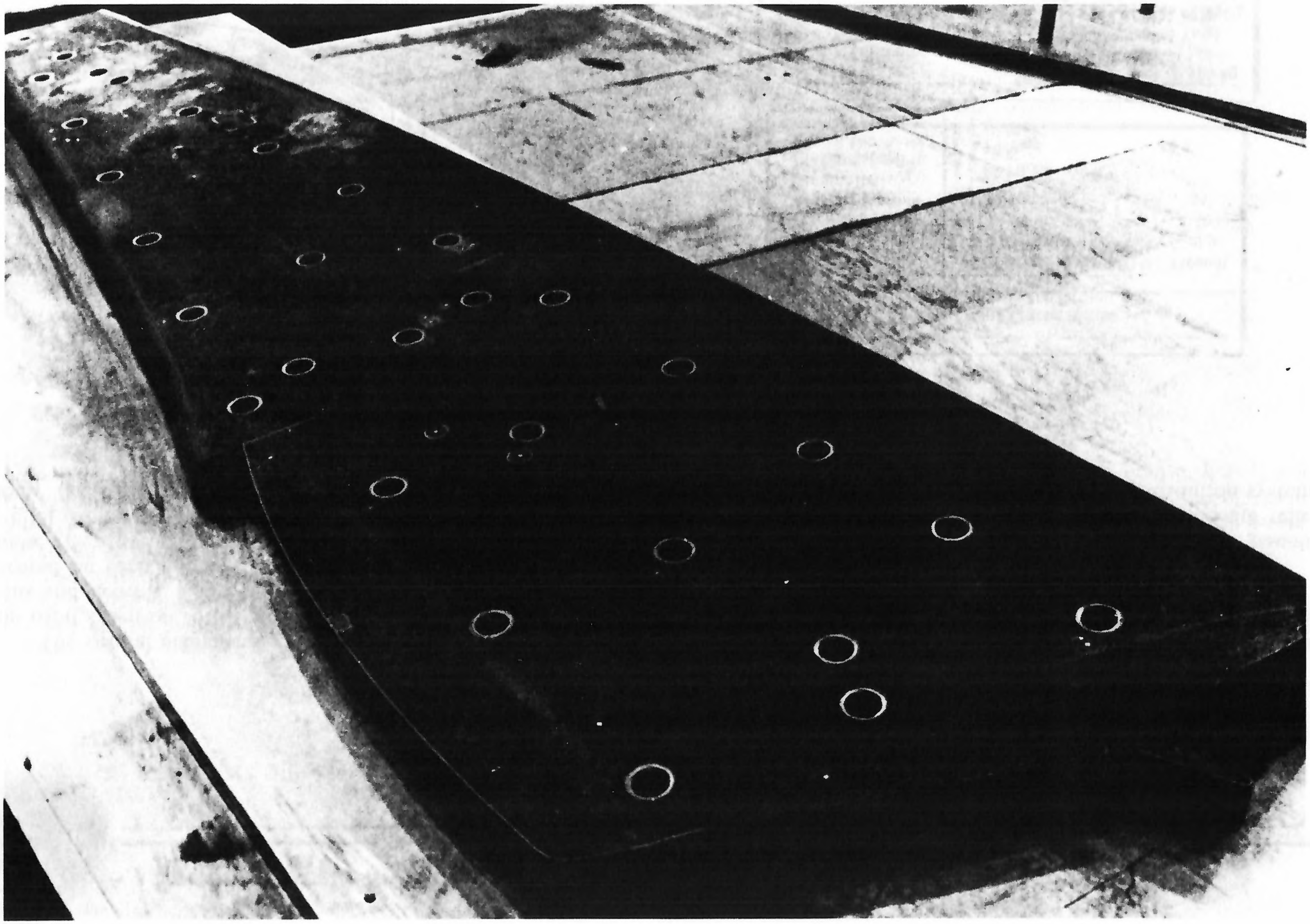


Fused Silica Insert and Plug Locations



Elevon Ablator

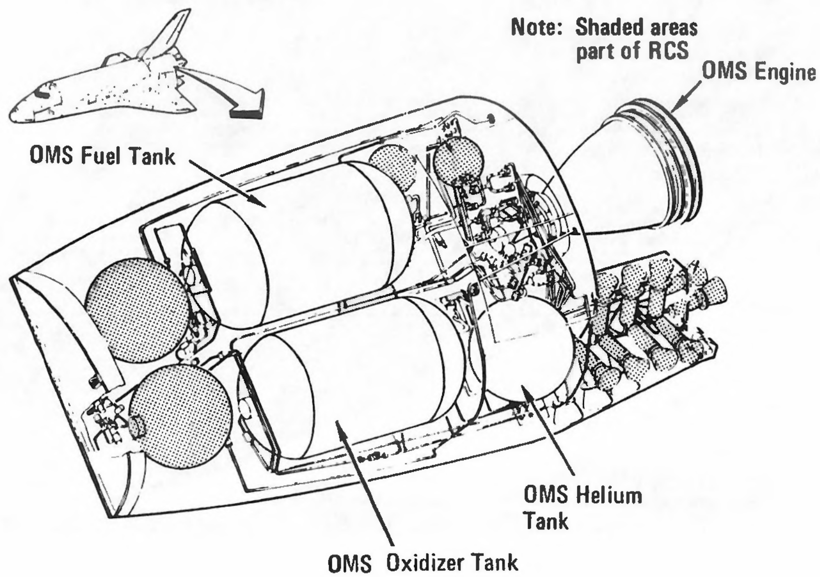




ORBITAL MANEUVERING SYSTEM

The orbital maneuvering system (OMS) provides the thrust for orbit insertion, orbit circularization, orbit transfer, rendezvous and deorbit. The OMS is housed in two independent pods located on each side of the orbiter's aft fuselage. The pods also house the aft reaction control system and are referred to as the orbital maneuvering system/reaction control system (OMS/RCS) pods. Each pod contains one OMS engine and the hardware needed to pressurize, store, and distribute the propellants to perform the velocity maneuvers. The two pods provide the redundancy for the OMS.

The system in each pod consists of a high-pressure gaseous helium storage tank, pressure regulation system, pressure relief valves, fuel tank, oxidizer tank, propellant distribution system,



OMS Engine Characteristics	
Thrust	26,688 Newtons (6000-lb) Vacuum
Specific Impulse	313 Seconds
Chamber Pressure	125 psia
Mixture Ratio	1.65
Gimbal Capability	$\pm 6^{\circ}$ Pitch $\pm 7^{\circ}$ Yaw

OMS Δ V Capability	304 m/s (1000 ft/s) 29,484-kg (65,000-lb) Payload
Usable OMS Propellant:	10,830 kg (23,876 lb) Total 6,743 kg (14,866 lb) of N_2O_4 4, 086 kg (9,010 lb) of MMH

Orbital Maneuvering System

OMS engine, plus 12 RCS primary and two RCS vernier engines. (The RCS is described in the RCS section.)

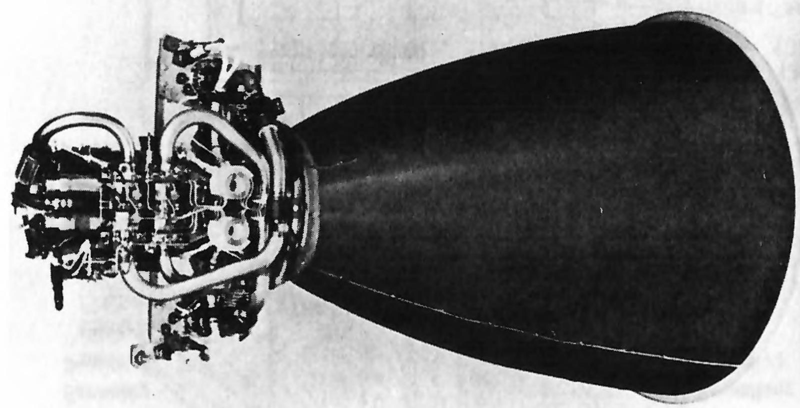
In each of the two OMS pods, gaseous helium pressure is supplied to the oxidizer and fuel tanks, which, in turn, supply the oxidizer and fuel under pressure to that OMS engine. The propellants are nitrogen tetroxide as the oxidizer and monomethyl hydrazine as the fuel. The propellants are earth-storable and hypergolic (ignite upon contact with each other). The propellants are directed to the OMS engine, where they ignite (hypergolic), producing a hot gas, thus thrust.

When the fuel reaches the engine, it is directed first through the combustion chamber walls (this regeneratively cools the chamber walls) and then into the engine injector. The oxidizer goes directly into engine injector. The nozzle extension is radiant-cooled and constructed of aluminum alloy.

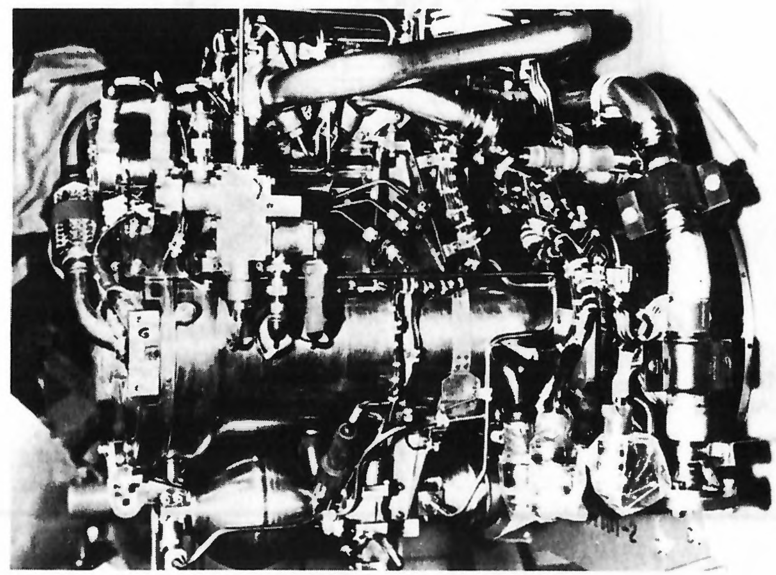
Each OMS engine provides 26,688 Newtons (6000 pounds) of thrust. The oxidizer-to-fuel mixture ratio is 1:65, the expansion ratio of the nozzle exit to the throat is 55:1, the dry weight of the engine is 117 kilograms (260 pounds), and the chamber pressure of the engine is 6,468 millimeters of mercury (mmHg) (125 psia).

Each OMS engine is reusable for 100 missions and capable of 1000 starts and 15 hours of cumulative firing. The minimum firing duration of an OMS engine is two seconds. The OMS may be utilized to provide thrust above 21,336 meters (70,000 feet). For velocity changes of between 0.9 and 1.8 meters per second (3 and 6 feet per second), the normal operating mode is one OMS engine.

Each engine has two electromechanical gimbal actuators, which control the OMS engine thrust direction in pitch and yaw (thrust vector control). The OMS engines can be used singly by



OMS Engine



OMS Engine (– Nozzle)

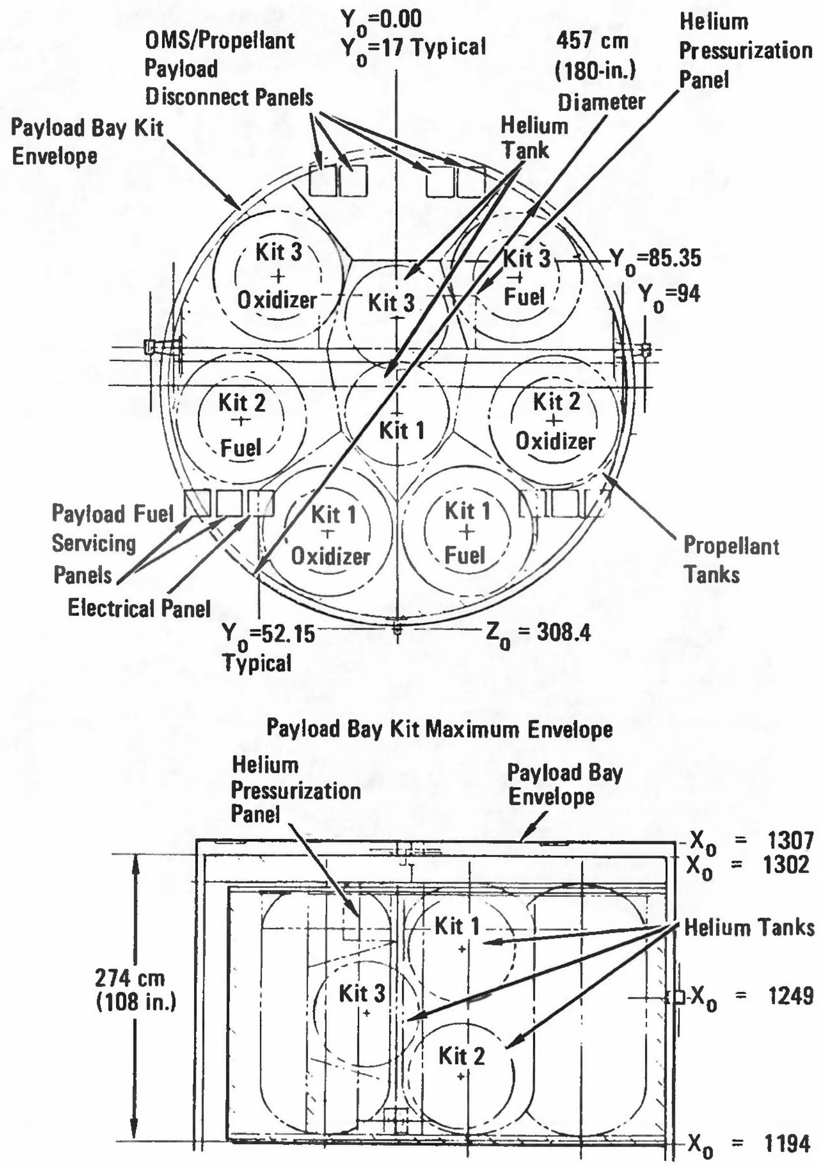
directing the thrust vector through the orbiter center of gravity or together by directing the thrust vector of each engine parallel to the other. It is noted that during a two-OMS-engine thrusting period, the RCS will come into operation only in the event the OMS gimbal exceeds gimbal rate or gimbal limits and should not normally come into operation during the OMS thrust period. However, during a one-OMS engine thrusting period, roll RCS is required. The pitch and yaw actuators are identical except for the stroke length and contain redundant channels (active and standby), which couple to a common drive assembly.

The OMS integral propellant tankage of both pods provides the capability of a 304 meter-per-second (1000-foot-per-second) velocity change with a 29,484-kilogram (65,000-pound) payload in the orbiter's payload bay. An OMS pod crossfeed line allows use of the propellants in both pods for operation of either OMS engine.

Extended missions, which would increase weight in the orbiter but provide additional altitude capabilities, could require up to three self-contained OMS propellant kits (helium, fuel, and oxidizer tanks–same tankage as in the pods). This would be added in the orbiter payload bay. Each OMS kit provides the orbiter with an additional 152 meter-per-second (500 foot-per-second) velocity capability. With the basic OMS and three additional OMS kits, the orbiter would have a total velocity change capability of 762 meters per second (2500 feet per second).

The OMS is thermally controlled by insulating the walls of the OMS pod that enclose the OMS hardware components and by heaters on the lines and OMS structure to maintain the OMS propellant temperature between 4 and 37°C (40 and 100°F).

The OMS/RCS pods are designed to be reused for up to 100 missions with only minor repair, refurbishment, and maintenance. The pods are removable to facilitate orbiter turnaround from a mission.



OMS Payload Bay Kit

The various parameters of the OMS are monitored on the flight deck crew display and control panel, CRT (cathode ray tube), and telemetry.

The contractors are McDonnell Douglas Astronautics Co. St. Louis, MO (OMS pod assembly and integration); Aerojet Liquid Rocket Co., Sacramento, CA (OMS engine); Aerojet Manufacturing Co., Fullerton, CA (OMS propellant tanks); Aircraft Contours, Los Angeles, CA (OMS pod edge member); Brunswick-Wintec, El Segundo, CA (OMS propellant tank acquisition screen assembly); Consolidated Controls, El Segundo, CA (high- and low-pressure solenoid valves); Fairchild Stratos, Manhattan Beach, CA (OMS regulators and hypergolic servicing couplings); Metal Bellows Co., Chatsworth, CA (alignment bellows); Simmonds Precision Instruments, Vergennes, VT (OMS propellant gauging system); SSP Products, Burbank, CA (gimbal bellows assembly); Tayco Engineering, Long Beach, CA (electrical heaters); AiResearch Manufacturing Co., Torrance, CA (gimbal actuators); Futurecraft Corp., City of Industry, CA (OMS engine valve components); L.A. Gauge, Sun Valley, CA (ball valves); PSM Division of Fansteel, Los Angeles, CA (OMS nozzle extension fabrication); Rexnord Inc., Downers Grove, IL (OMS engine bearings); Sterer Engineering and Manufacturing, Pasadena, CA (OMS engine pressure regulators); Parker-Hannifin, Irvine, CA (OMS propellant tank isolation valves, manifold interconnect valves, and additional delta velocity kit valves); Rockwell International, Rocketdyne Division, Canoga Park, CA (OMS check valves); Brunswick, Lincoln, NE (OMS helium tanks).

REACTION CONTROL SYSTEM

The forward and aft reaction control system (RCS) on the orbiter provides the thrust for attitude (rotational) maneuvers (pitch, yaw and roll) as well as the thrust for small velocity changes along the orbiter axis (translation maneuvers) when the orbiter is above 21,336 meters (70,000 feet).

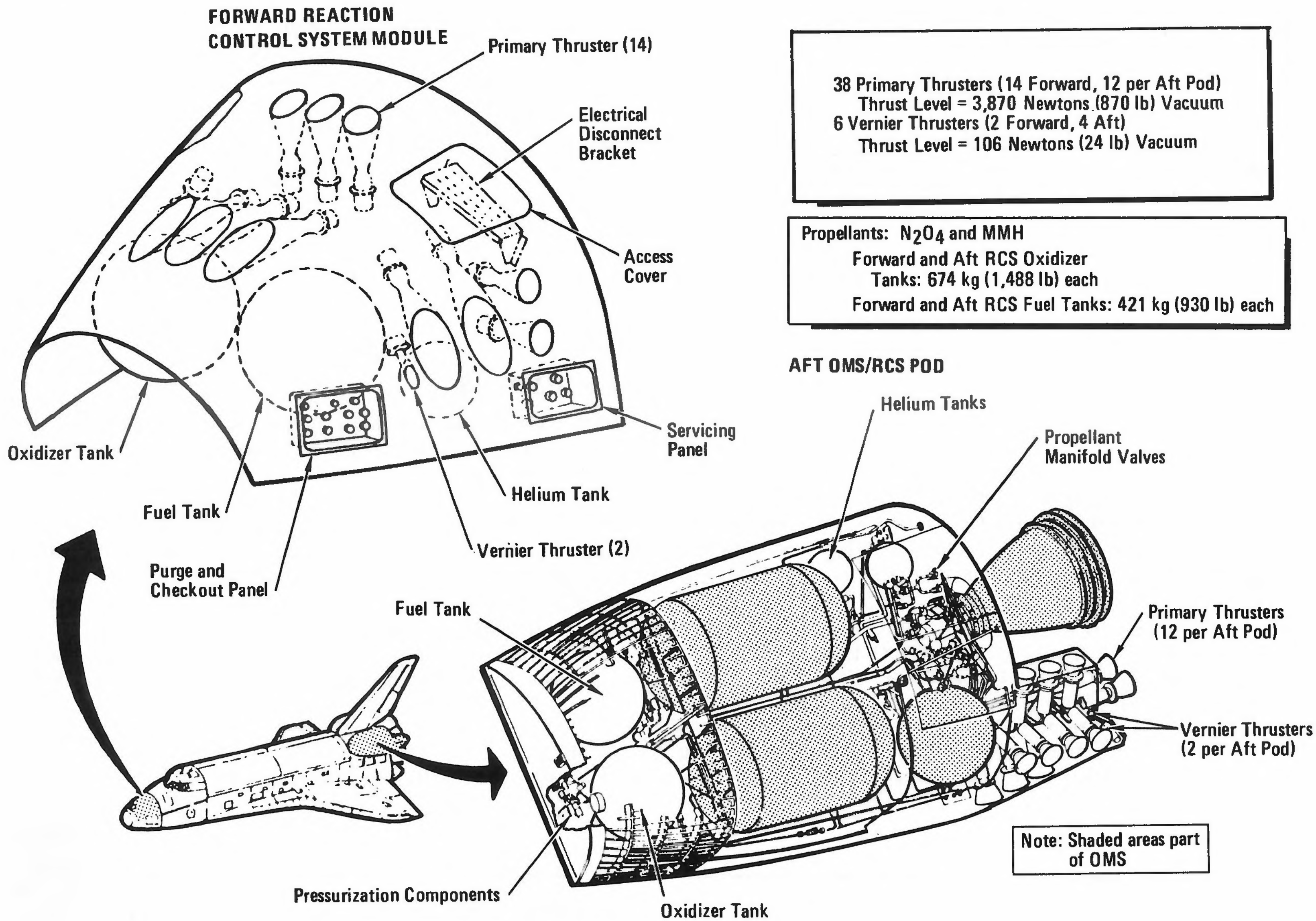
The RCS is located in three different areas of the orbiter. The forward RCS module is located in the forward fuselage nose area. The aft (right and left) RCS is located in the left and right OMS/RCS pods, which are attached to the aft fuselage.

Each RCS consists of high-pressure gaseous helium storage tanks, pressure regulation systems, pressure relief valves, a fuel and an oxidizer tank, and a propellant distribution system to its RCS engines.

The two helium tanks of each RCS supply gaseous helium pressure to the oxidizer and fuel tanks, which supply the oxidizer and fuel under pressure to the RCS engines. The propellants are nitrogen tetroxide as the oxidizer and monomethyl hydrazine as the fuel. The propellants are earth storable and hypergolic (ignite upon contact with each other). The propellants of each RCS are supplied to its RCS engines, where they ignite (hypergolic), producing a hot gas, thus thrust.

The forward RCS has 14 primary and two vernier RCS engines. The aft RCS has 12 primary and two vernier engines in each pod. The primary RCS engines provide 3870 Newtons (870 pounds) of thrust each, and the vernier RCS engines provide 106 Newtons (24 pounds) of thrust each. The oxidizer-to-fuel ratio for each engine is 1:6. The chamber pressure of the primary engines is 7866 millimeters of mercury (152 psia). For each vernier engine it is 5692 mmHg (110 psia).

The primary engines are reusable for a minimum of 100 missions and capable of sustaining 50,000 starts and 20,000 seconds of cumulative firing. The primary engines are used in a steady-state thrusting mode of one to 150 seconds with a contingency of 500 seconds for the aft RCS engines, as well as in a pulse mode with a minimum impulse thrusting time of 80 milliseconds (0.080 second). The expansion ratio (exit area to throat area) of the primary engines ranges from 22:1 to 30:1. The primary engines provide the normal control of the orbiter.

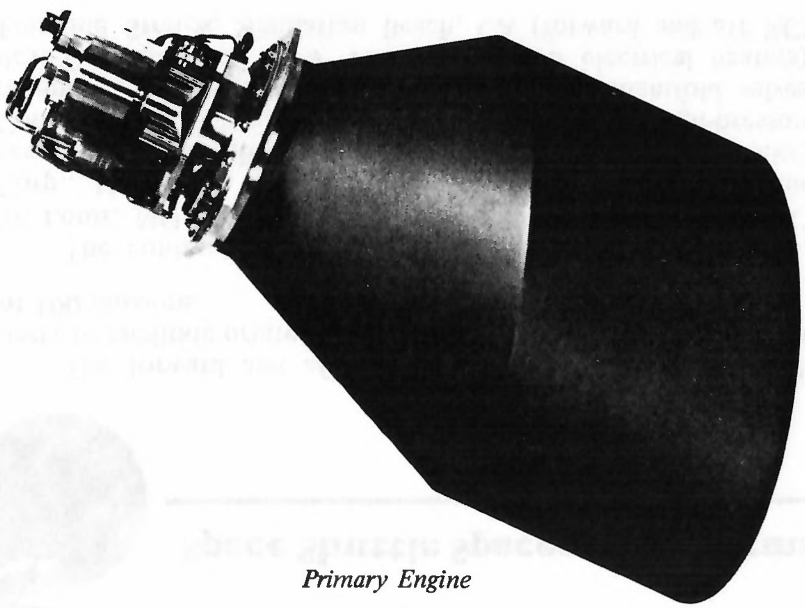


Reaction Control System

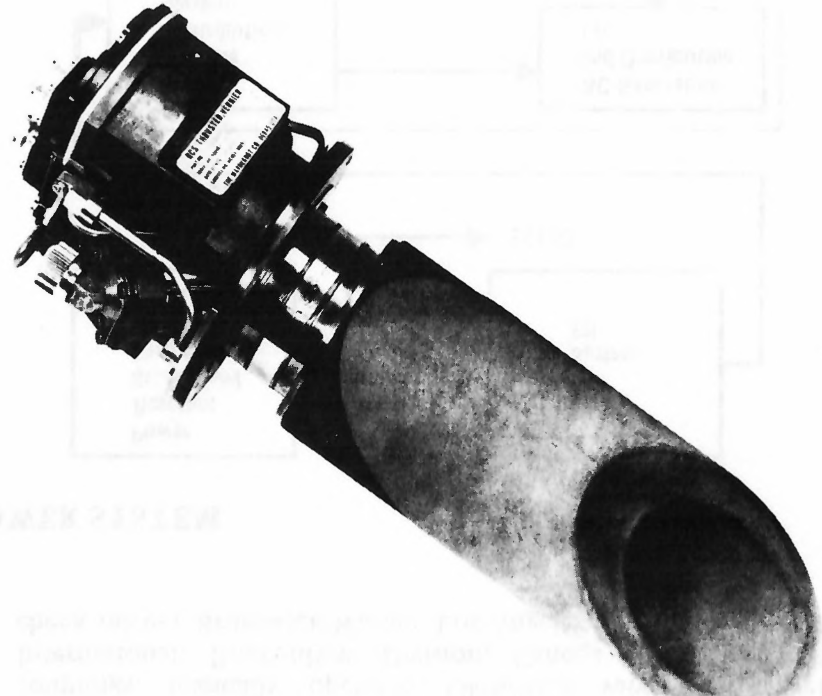
The vernier engines are also reusable for a minimum of 100 missions and capable of sustaining 500,000 starts and 125,000 seconds of cumulative firings. The vernier engines are used in a steady-state thrusting mode of one to 125 seconds as well as in a pulse mode with a minimum impulse time of 80 milliseconds. The vernier engines are used for finite maneuvers and station-keeping (or long-time attitude hold). The expansion ratio of the vernier engine ranges from 20:7 to 50:1.

The combustion chamber of each RCS engine is constructed of columbium with its internal wall fuel film cooled. The nozzle of each RCS engine is radiation cooled. Insulation around the combustion chamber and nozzle prevents the excessive heat–1093 to 1315°C (2000 to 2400°F)–from radiating into the orbiter structure.

The left and right aft RCS are interconnected, which permits the cross feeding of propellants through ac motor-



Primary Engine



Vernier Engine

operated RCS crossfeed valves. In addition, an interconnect is provided between the OMS propellant systems and the aft RCS propellant systems through ac motor-operated interconnect valves, which permits OMS propellant use by the left and right aft RCS.

Thermostatic controlled heaters maintain the RCS propellant tanks and lines between 15 and 37°C (60 and 100°F).

The various parameters of each RCS are monitored and utilized for display on the flight deck crew display and control panel and transmitted to telemetry and fault messages.

The forward and aft (left and right) RCS are removable units to facilitate orbiter turnaround and reusable for a minimum of 100 missions.

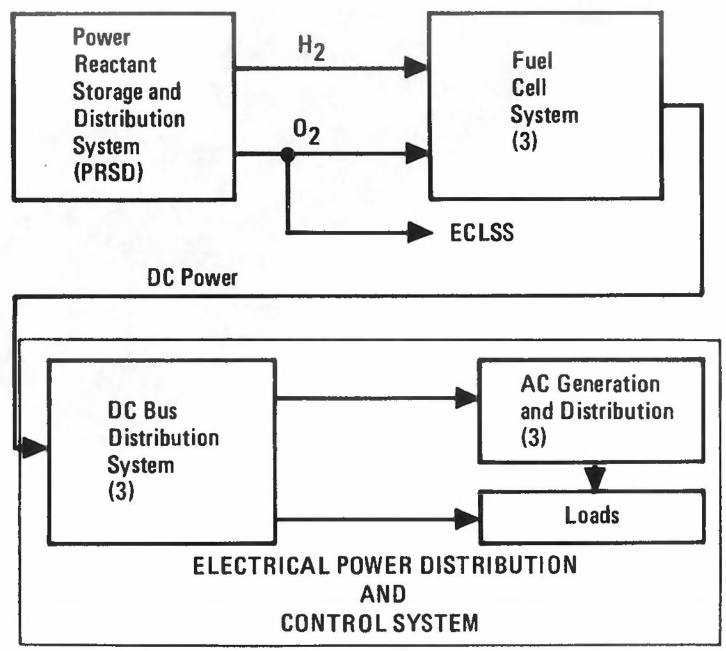
The contractors are McDonnell Douglas Astronautics Co., St. Louis, MO (OMS/RCS pod assembly and integration); CCI Corp., Marquardt Co. Division, Van Nuys, CA (primary and vernier thrusters); Brunswick, Lincoln, NE (RCS helium tanks); Consolidated Controls, El Segundo, CA (RCS high-pressure helium valves and low-pressure vernier engine manifold valves, dc); Cox and Co., New York, NY (RCS electrical heaters); Fairchild Stratos, Manhattan Beach, CA (forward and aft RCS helium pressure regulators, couplings, nitrogen tetroxide/monomethyl hydrazine, helium fill disconnects); Honeywell, Inc., St. Petersburg, FL (forward and aft RCS reaction jet drivers); Martin Marietta, Denver, CO (forward and aft RCS propellant tanks); Metal Bellows Co., Chatsworth, CA (RCS flexible line assembly); Parker-Hannifin, Irvine, CA (ac motor-operated manifold valves, RCS crossfeed valves, interconnect valves payload, delta velocity kit valve and relief valve hypergolic couplings, manually operated OMS/RCS valves); Rockwell International, Rocketdyne Division, Canoga Park, CA (RCS check valves); Brunswick-Wintec, Los Angeles, CA (filters).

ELECTRICAL POWER SYSTEM

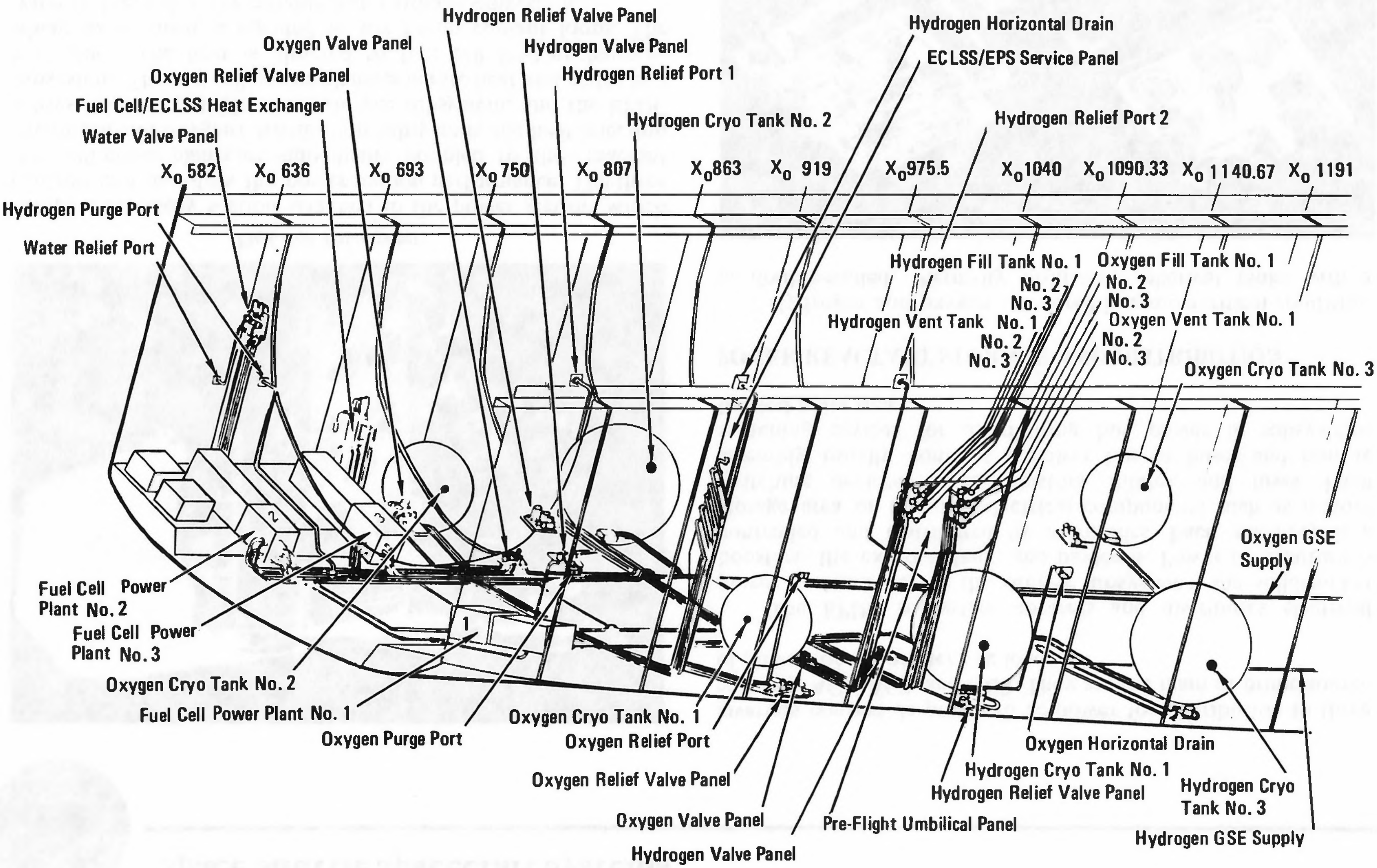
The electrical power system (EPS) consists of three subsystems: power reactant storage and distribution (PRSD), fuel cell power plants (FCP) (the electrical power generation), and electrical power distribution and control (EPDC).

The PRSD subsystem stores and supplies the reactants (hydrogen and oxygen) to the three fuel cell power plants, which generate the electrical power during all mission phases; in addition, oxygen is supplied to the environmental control and life support system (ECLSS) for crew cabin pressurization. The hydrogen and oxygen are stored in their respective storage tanks at cryogenic temperatures and supercritical pressures. The EPDC subsystem distributes the fuel cell power plant direct current and converts direct current to alternating current and distributes alternating current to all Space Shuttle electrical equipment throughout all mission phases.

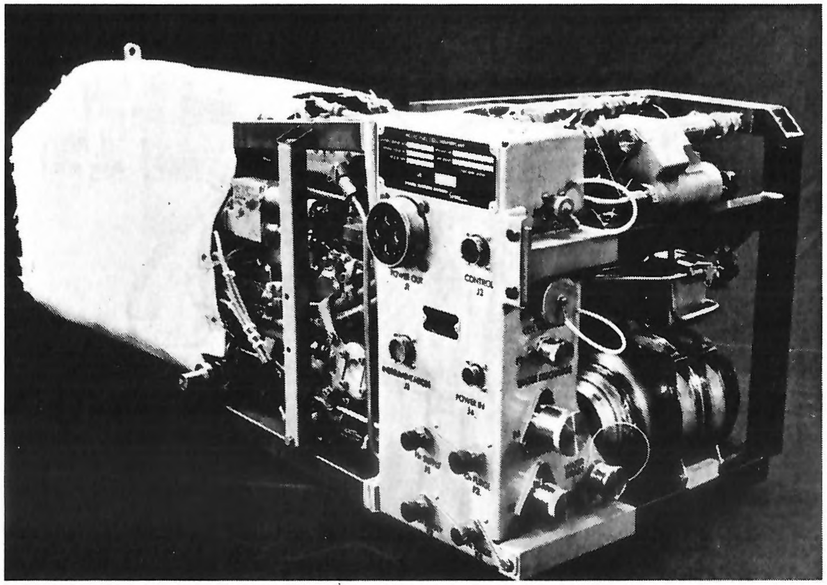
The three fuel cell power plants, through a chemical reaction, generate all of the 28-volt dc electrical power for the Space Shuttle after launch. Electrical power before launch is supplied in conjunction with ground support equipment at the launch pad. Each of the three fuel cell power plants consists of a power section, where the chemical reaction occurs, and a



PRSD, Fuel Cell Systems, and EPDC



Power Reactant Storage and Distribution System



Fuel Cell Powerplant

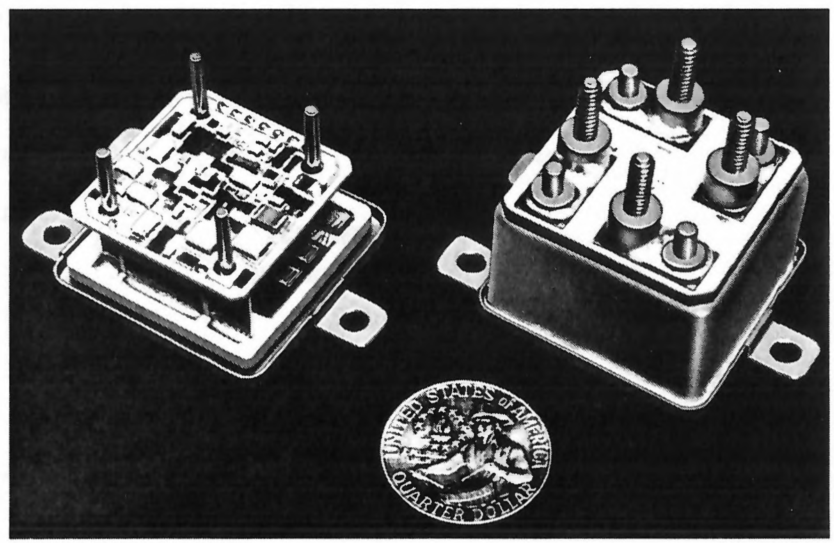
compact accessory section attached to the power section, which controls and monitors the power section performance. The three fuel cell power plants are individually coupled to the reactant (hydrogen and oxygen) distribution subsystem, the heat rejection subsystem, the potable water storage subsystem, and the EPDC subsystem. The fuel cell power plants generate heat and water as a byproduct. The heat is directed to fuel cell heat exchangers, where excess heat is rejected to the Freon coolant loops. The water is directed to the potable water storage subsystem.

The dc power generated by the fuel cells is routed to a three-bus system that distributes dc power to the forward, mid, and aft orbiter sections for equipment in these areas. The three main dc buses–MN (main) A, MNB, and MNC–are the prime source of power for the orbiter's dc loads. Each of the three dc main buses supplies power to three solid (static), single-phase inverters, which constitute one inverter system; thus, the nine inverters convert dc power to ac power for distribution to three ac buses, AC1, AC2, and AC3. They are the main or prime source of power for the orbiter's ac loads.

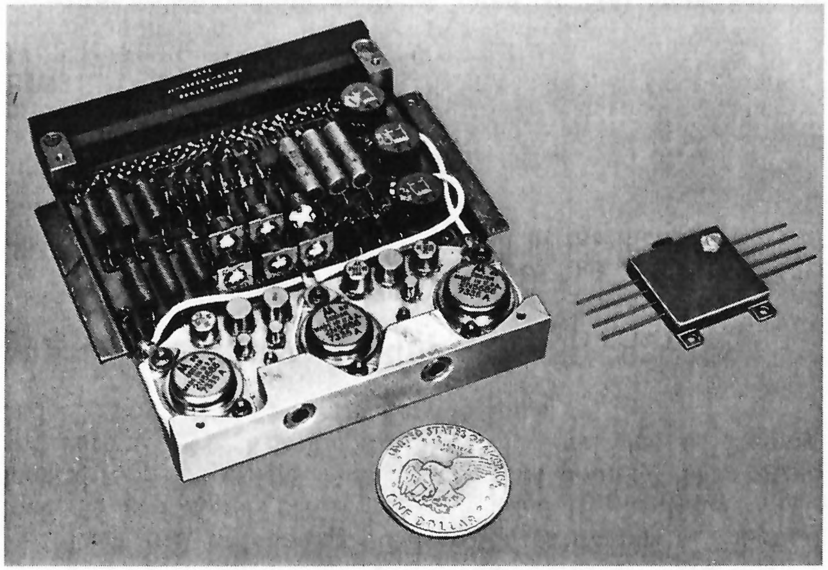
The EPDC subsystem controls and distributes electrical power (ac and dc) to the orbiter subsystems, the solid-rocket boosters, the external tank, and payloads. Power distribution is controlled and distributed by assemblies. Each assembly is a storage area or box for electrical components such as remote switching devices, buses, resistors, diodes, and fuses. Each assembly usually contains a power bus or buses and remote switching devices for distributing bus power to subsystems located in its area.

POWER REACTANT STORAGE AND DISTRIBUTION

Hydrogen and oxygen are stored in a supercritical condition in double-walled, thermally insulated, spherical tanks with a



Remote Power Controller



Hybrid Load Controller

vacuum annulus between the inner pressure vessel and outer shell of the tank. Each tank has multi-layer thermal insulation and heaters to add energy to the reactants during depletion for pressure control. Each tank has capacitance quantity sensing capability for measuring quantity remaining.

The hydrogen and oxygen tanks are grouped in sets consisting of one hydrogen and one oxygen tank. Initial development test flight will consist of two-tank sets (hydrogen and oxygen tanks one and two); the second development flight test will have three-tank sets (hydrogen and oxygen tanks one, two, and three); and the third and subsequent flights will have four-tank sets (hydrogen and oxygen tanks one, two, three and four). The four-tank sets provide seven-day mission capability. Up to five tank sets can be installed. The five tank sets are all located in the mid fuselage under the payload bay area. In the operational flights, additional tank sets can be installed in the payload bay area.

FUEL CELL POWER PLANTS

Each of the three fuel cell power plants is reusable and restartable. The cells are located under the payload bay area in the forward portion of the orbiter mid fuselage.

Each cell's power section or cell stack contains two parallel substacks, which have 32 cells in series. The cells contain electrolyte consisting of potassium hydroxide and water, an oxygen electrode (cathode), and a hydrogen electrode (anode).

The hydrogen and oxygen reactants enter each fuel cell power plant and pass through preheaters. The reactants are warmed from cryogenic temperature to $4.4^{\circ}C$ ($40^{\circ}F$) or greater. Then they pass through filters and then into a two-stage integrated dual gas regulator module.

Hydrogen is routed to the cell's hydrogen electrode, where it reacts with hydroxyl ions in the electrolyte. This electrochemical reaction produces electrons (electrical power), water, and heat with the electrons being routed through the orbiter's EPDC subsystem to perform electrical work. Oxygen is routed to the cell's oxygen electrode, where it reacts with the water and returning electrons to product hydroxyl ions. The hydroxyl ions then migrate to the hydrogen electrode, where they enter into the hydrogen reaction at that electrode. The oxygen and hydrogen are reacted (consumed) in proportion to the orbiter's electrical power demand.

Excess water vapor is removed by an internal hydrogen system. Hydrogen and water vapor from the reactants exits the cell stack, is mixed with replenishing hydrogen from the storage and distribution system, and enters a condensor, where waste heat from the hydrogen and water vapor is transferred to the fuel cell coolant system. The resultant temperature decrease condenses some of the water vapor to water droplets. A centrifugal water separator extracts the liquid water and pressure-feeds it to potable tanks in the lower deck of the pressurized crew cabin. Water from the potable water storage tanks can be used for crew consumption and cooling of the Freon-21 coolant loops. The remaining circulating hydrogen is directed back to the fuel cell stack.

The fuel cell coolant system circulates a liquid fluorinated hydrocarbon (FC-40) and transfers the waste heat from the cell stack through the fuel cell heat exchanger of that fuel cell power plant to the Freon-21 coolant loop system in the mid fuselage. Internal control of the circulating fluid maintains the cell stack at a normal operating temperature of approximately 93°C (200°F).

Each fuel cell power plant is 43 centimeters (17 inches) high, 35 centimeters (14 inches) wide, and 101 centimeters (40 inches) long and weighs 91 kilograms (201 pounds). The voltage and current of each is 2 kilowatts at 32.5 volts dc, 61.5 amps, and 12 kilowatts at 27.5 volts dc, 436 amps.

ELECTRICAL POWER DISTRIBUTION AND CONTROL

The dc power generated by the three fuel cells is routed to main buses and essential buses located in the distribution and controller assemblies.

The three main dc buses (MNA, MNB, and MNC) are the primary source of dc power for all orbiter electrical equipment. Each fuel cell is dedicated to a specific main bus: FC-1 to MNA, FC-2 to MNB, and FC-3 to MNC. Each cell is connected to its corresponding main bus by power contactors, which are dc-motor-driven remote switches rated at 500 amps and are located in the distribution assembly (DA). The DA, through fuses, powers the coreesponding buses in the mid power controller (MPC) assembly; the forward power controller (FPC) assembly; the forward load controller (FLC) assembly; the aft power controller (APC) assembly and the aft load controller (ALC) assembly.

The power controller assemblies (PCA's) and load controller assemblies (LCA's) use remote switching devices to distribute the loads. The PCA's contain remote power controllers (RPC's), which are solid-state, remote switching devices used for applying loads up to 20 amps and relays for remote switching. The LCA's contain hybrid circuit devices, which are solid-state, remote switching devices used as logic switches and as remotely controlled switches for electrical loads of 5 amps or less. The mid power controllers contain RPC's, relays, and hybrid circuit drivers. The remote switching devices permit location of major electrical distribution buses close to the major loads, which eliminates the need for heavy electrical feeders to and from the crew cabin flight deck display and control panels. This greatly reduces spacecraft wiring, and thus weight, and permits more flexible electrical load management.

Three essential buses, ESS 1BC, ESS 2CA, ESS 3AB, supply control power to the flight deck crew display and control switches which are necessary to restore power to a failed main dc or ac bus and to essential non-EPS electrical loads and switches.

The nominal fuel cell voltage is 27.5 to 32.5 volts dc, and the nominal main bus voltage range is 27 to 32 volts dc, which correspond to 12 and 2 kilowatt loads, respectively.

Alternating-current power is generated and made available to using systems through three ac buses: AC1, AC2, and AC3. The inverters convert dc to ac and the ac bus distribution assemblies contain the ac buses and the ac bus sensors. The ac power is distributed to the flight and mid-deck display and control panels and in the payload bay deployment mechanisms, in payloads, and in the aft main engine controllers and payload bay doors. The inverters are located in the forward avionics bays of the crew compartment.

The three phase inverters for AC1 receive power from MNA, AC2 from MNB, and AC3 from MNC.

There are 10 motor controller assemblies (MCA's) used on the orbiter; three are in the forward area, four are in the mid body area, and three are in the aft area. Their only function is to supply ac power to the ac motors used for vent doors, air data doors, star tracker doors, payload bay doors, payload bay latches, and reaction control system/orbital maneuvering system (RCS/OMS) motor-actuated valves. The MCA's contain main buses, ac buses, and hybrid relays, which are the remote switching devices for switching the ac power to electrical loads. The main buses are used only to supply control or logic power to the hybrid relays so that ac power can be switched on/off. If a main bus is lost, the hybrid relays using that main bus will not operate.

The various parameters of the PRSD subsystem, the fuel cell power plans, and the EPDC subsystem are monitored and utilized for display on the flight deck display and control panel, and for transmission to telemetry and fault messages.

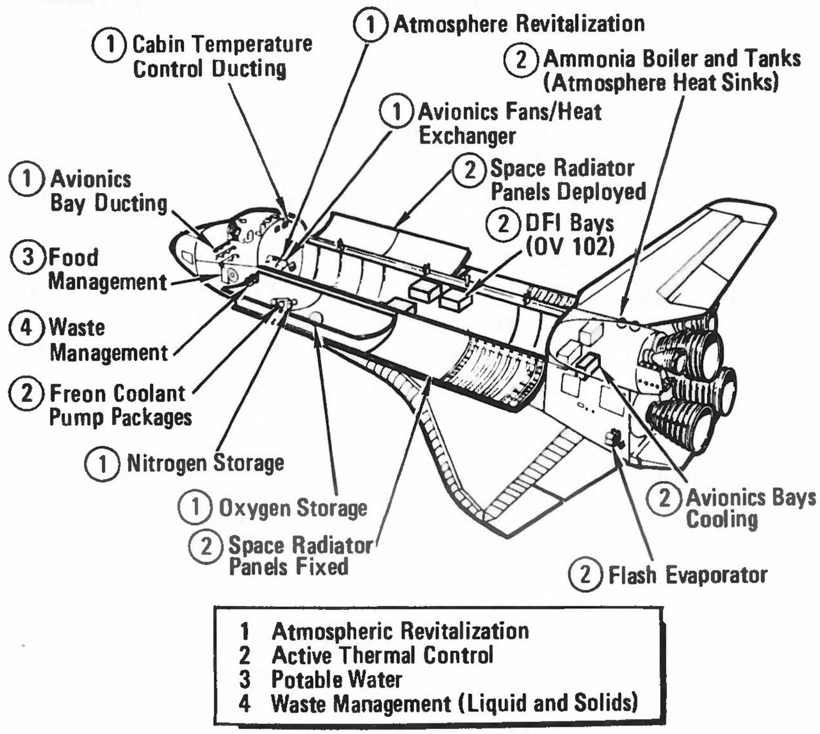
The contractors are: Aerodyne Controls Corp., Farmingdale, NY (oxygen, hydrogen check valve, water pressure relief valve); Aiken Industries, Jackson, MI (thermal circuit breakers; three-phase circuit breakers); AIL Cutler Hammer, Milwaukee, WI (remote control circuit breaker); American Aerospace Farmingdale, NY (ac and dc current sensors, current level detector); Applied Resources, Fairfield, NJ (rotary switch); Rockwell International Autonetics Group, Anaheim CA (ac bus sensor, load controller assemblies); Beech Aircraft Corp., Boulder, CO (power reactant storage hydrogen and oxygen tanks, gaseous oxygen and hydrogen ground support equipment); Bell Industries, Gardena, CA (modular terminal boards); Bendix Corp., Sidney, NY, and Franklin, IN (high density connectors); Bussman Division of McGraw Edison, St. Louis, MO (fuses, fuse holders, fuse dc limiter high current); Brunswick-Circle Seal, Anaheim, CA (water check valve); Consolidated Controls, El Segundo, CA (hydrogen, oxygen solenoid valve, undirectional/bidirectional shutoff valve); Cox and Co., New York, NY (heaters); Deutsch, Banning, CA (general-purpose connector); Fairchild Stratos, Manhattan Beach, CA (cryogenic fluid and gas supply disconnects); G/H Technology Co., Santa Monica, CA (connector cryo); Hamilton Standard, Windsor Locks, CT (fuel cell heat exchanger); Haveg Industries, Inc., Winooski, VT (general-purpose wire); ITT Cannon, Santa Ana, CA (connectors, bulkhead feedthrough); Labarge, Santa Ana, CA (general-purpose wire); Leach Relay, Los Angeles, CA (relay); Malco Microdot Corp., Pasadena, CA (connector); Pratt and Whitney Division of United Technologies, East Hartford, CT (fuel cell powerplants); R.V. Weatherford, Glendale, CA (shunt); Statham Instruments, Oxnard, CA (cryo pressure transducer); Tayco Engineering, Long Beach, CA (fuel cell water dump nozzle); Teledyne Kinetics, Solana Beach, CA (dc power contractor); Teledyne Thermatics, Elm City, NC (general-purpose wire); Westinghouse Electric Corp., Lima, OH (remote power controller electrical system inverters); Weston Instruments, Newark, NJ (electrical indicator meter) Brunswick-Wintec El Segundo, CA (reactant and coolant filters).

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM

The environmental control and life support system (ECLSS) consists of an atmospheric revitalization subsystem (ARS), water coolant loop subsystem (WCLS), atmosphere revitalization pressure control subsystem (ARPCS), active thermal control subsystem (ATCS), food, water, and waste management subsystem (FWWS), and airlock support subsystem. These subsystems interact to provide a habitable environment in the crew compartment for the crew and passengers.

The ARS provides humidity, carbon dioxide, and carbon monoxide control for the crew compartment, controls cabin temperature and ventilation, and provides cooling to the flight deck avionics, cabin, and avionics bays.

The ARPCS controls cabin pressure and oxygen partial pressure, nitrogen pressurization of the potable and waste water tanks, and storage of nitrogen and emergency oxygen consumables.



Environmental Control and Life Support System

The WCLS collects heat from the cabin atmosphere and electronics and transfers it to the ATCS, which rejects it to space via water and Freon coolant loops. The ATCS also removes and rejects waste heat from the fuel cells, payload, and mid-body and aft-located electronic units, while providing heating of the hydraulic systems when needed.

Potable water is produced by the three fuel cells aboard the spacecraft and stored in tanks for crew consumption and personal hygiene; during certain phases of the mission it is also used for cooling of the Freon coolant loops. Waste water collected from the cabin heat exchanger is stored in tanks along with crew members' waste water. Solid waste remains in the waste management system until the orbiter is serviced on the ground.

The orbiter crew compartment provides a life-sustaining environment for a crew of four and has accommodations for four passengers. The crew cabin volume with the airlock inside the mid deck of the crew compartment is 66 cubic meters (2325 cubic feet). If the airlock and the tunnel adapter are outside in the payload bay area, the volume is 74 cubic meters (2625 cubic feet).

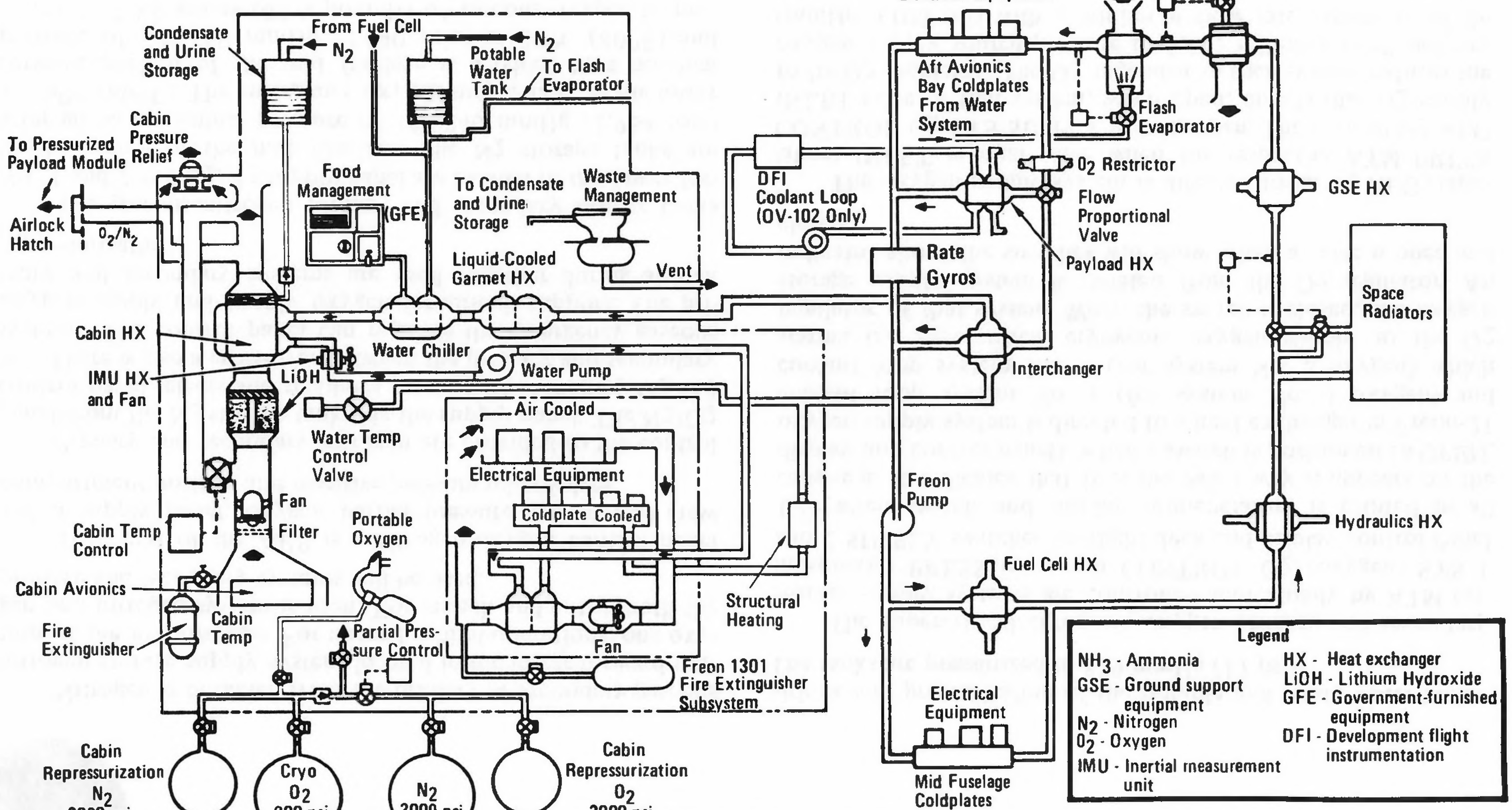
ATMOSPHERIC REVITALIZATION CONTROL SUBSYSTEM

The ARS maintains a habitable environment for the crew and passengers and a conditioned environment for the electronic avionics equipment located inside the crew cabin. The ARS consists of the water coolant loops (WCL), and the cabin air loops, and pressure control.

CABIN PRESSURE. The cabin is pressurized to 760 millimeters of mercury (mmHg) ±103 mmHg (14.7 ±0.2 psia) and maintained at an average 80-percent nitrogen and 20-percent oxygen mixture by the ARS.

Oxygen partial pressure is maintained between 152 and 178 mmHg (2.95 and 3.45 psi), with sufficient nitrogen added to achieve the cabin total pressure of 760 plus or minus 10 mmHg (14.7 ±0.2 psia) for operational orbiters and 750 +10 mmHg (14.5 ±0.2 psia) for the development test flights.

Oxygen is obtained from three sources: the primary and secondary power reactant super-critical cryogenic oxygen storage supply system and an emergency gaseous oxygen supply system. The supercritical cryogenic oxygen supply system is located in the lower portion of the mid fuselage and is also utilized by the three power reactant fuel cells. The emergency gaseous oxygen supply is located in the lower forward portion of the mid fuselage.



ECLSS Schematic

Nitrogen is obtained from the primary or secondary gaseous nitrogen storage supply system located in the lower forward portion of the mid fuselage. For normal orbital operations, one oxygen and nitrogen system is used. For launch and entry, both the primary and secondary systems will be used.

The heart of the ARS is a nitrogen/oxygen control panel and a supply panel, oxygen partial pressure sensor, and crew compartment positive and negative pressure relief valves.

Primary and secondary nitrogen are provided to the control panel from the N_2 storage tanks via the supply panel. The N_2/O_2 control panel selects and regulates primary or secondary O_2 and N_2. There is also a cross-over between the primary and secondary systems. The control panel can regulate the emergency gaseous oxygen supply and supply oxygen for airlock support. The primary and secondary systems are used together during airlock repressurization.

The gaseous nitrogen primary and secondary storage tanks No. 1 and 2 and O_2/N_2 supply panel are located in the lower forward portion of the mid fuselage. The N_2 storage tanks are serviced to a nominal pressure of 153,240 mmHg (2,964 psia) at 26°C (80°F). The emergency oxygen supply tank in the lower forward portion of the mid fuselage is serviced to a nominal pressure of 126,150 mmHg (2,440 psia) at 26°C (80°F) and stores 29.7 kilograms (67.6 pounds) of gaseous oxygen to provide high flow along with gaseous nitrogen for emergency entry in the event of a crew compartment puncture. This emergency supply maintains the cabin at 414 mmHg (8 psi) and oxygen partial pressure at 103.5 mmHg (2 psia).

The O_2 and N_2 systems provide makeup gas for oxygen consumption by the crew and passengers. Four crew members use an average of 0.79 kilogram (1.76 pounds) of O_2 per person per day. Up to 3.49 kilograms (7.7 pounds) of nitrogen and 4 kilograms (9 pounds) of oxygen are expected to be used per day for normal loss of crew cabin air to space and metabolic usage. The O_2 and N_2 also provides for repressurization of the airlock and pressurization of the potable and waste water tanks. The tanks are pressurized to 879 mmHg (17 psi).

The supercritical cryogenic oxygen primary and secondary storage supply systems are controlled individually by ATM (atmospheric) PRESS (pressure) CONTROL O_2 (oxygen) SYS 1 and 2 SUPPLY switches on flight deck and display control Panel L2 (when switch and display nomenclature is printed in all caps—e.g. it indicates that it is the exact way it appears on the display and control panel). When a switch is positioned to OPEN, oxygen supply system is directed to a heat exchanger in Freon-21 coolant loop system No. 1 (for system No. 1 oxygen) and coolant loop system No. 2 (for system No. 2 oxygen) which warms the supercritical cryogenic oxygen supply to the O_2 regulator of that system. When the switch is closed, the oxygen storage supply system is isolated from the O_2 regulator. An indicator above the switches will show when a valve is open and closed.

The oxygen supply system is directed to an O_2 REG (regulabor) INLET manual valve when the respective ATM PRESS CONTROL O_2 SYS SUPPLY valve is open. The manual O_2 REG INLET valve of that system, when open, directs that O_2 supply to its O_2 regulator. The O_2 regulator in each system reduces the oxygen supply source pressure to 5,175 millimeters of mercury (mmHg) (100 psi) with a minimum flow rate capability of 34 kilograms (75 lbs) per hour. Each regulator is a two-stage regulator with the second stage functioning as a relief valve when the differential pressure across the second stage is 12,678 mmHg (215 psi). The relief pressure is vented into the crew module cabin. The regulated pressure from system No. 1 and system No. 2 is routed to its respective 14.7 psi CABIN REG (regulator) INLET manual valve, 8 psi EMERG (emergency) REG (regulator), the PAYLOAD O_2 manual valve and the XOVER (crossover) O_2 valve. The check valve in each O_2 supply line between the O_2 regulator and the 14.7 CABIN REG INLET prevents O_2 reverse flow to the 5,175 mmHg (100 psi) regulator.

The two valves in the crossover manifold are controlled by ATM PRESS CONTROL O_2 PRESS SYS 1 and SYS 2 XOVER switches on Panel 12. When a switch is opened, that oxygen supply system is directed to the AIRLOCK SUPPLY OXYGEN 1 and 2 manual valves, AIRLOCK O_2 1 and 2 EMU (extravehicular mobility unit), POS (portable oxygen system), the CDR (commander) and PLT (pilot) ejection seat oxygen system, and the three face mask outlets. If both ATM PRESS CONTROL O_2 PRESS SYS 1 and SYS 2 XOVER valves are open, oxygen supply systems No. 1 and 2 are interconnected. When the respective ATM PRESS CONTROL O_2 PRESS switch is closed, that oxygen supply system is isolated from the crossover manifold.

The emergency O_2 tank is made of a filament wound Kevlar fiber with an Inconel liner. The O_2 tank is serviced to a nominal 126,150 mmHg (2,440 psia) at 26°C (80°F) and stores 29.7 kilograms (67.6 lbs) of gaseous oxygen. The emergency oxygen supply system is controlled by the ATM PRESS CONTROL O_2 EMER switch on Panel L2. The emergency O_2 supply system is directed to its regulator when the ATM PRESS CONTROL O_2 EMER switch is open and isolated from the regulator when the switch is closed. An indicator above the switch indicates OP (open) when the valve is open, CL (close) when the valve is closed, and BARBERPOLE when the motor operated valve is in transit.

The emergency O_2 regulator reduces emergency O_2 supply pressure to 15,525 mmHg (300 psi) when the O_2 emergency valve is open. The two-stage regulator has a relief valve. When the differential pressure across the relief valve is 64,687 mmHg (1,250 psi), the valve will operate. The relief pressure is vented overboard.

The emergency O_2 regulated pressure is directed to an O_2 EMER manual valve. When the manual valve is open, the emergency O_2 supply is directed into the O_2 crossover manifold. The emergency O_2 supply is normally isolated except for ascent and entry.

The check valve between the Freon-21 coolant loop and crossover valve in the primary and secondary supercritical oxygen supply system prevents O_2 flow from one supply source to the other when the crossover valves are open.

The primary and secondary gaseous nitrogen supply tanks are identical to the emergency gaseous oxygen supply tank except that a titanium liner is used. Each gaseous nitrogen (N_2) tank is serviced to a nominal pressure of 153,240 mmHg (2,964 psia) at 26°C (80°F) with a volume of 134,086 cubic centimeters (8,181 cubic inches). The two N_2 tanks in each system are manifolded together. The primary and secondary nitrogen supply systems are controlled by ATM PRESS CONTROL N_2 SYS 1 and 2 SUPPLY switches on Panel L2. When a switch is opened, that nitrogen supply system is directed to an ATM PRESS CONTROL N_2 SYS REG (regulator) INLET valve. When the switch is closed, the nitrogen supply system is isolated from the N_2 SYS REG INLET valve. An indicator above each switch regulator. An indicator above each switch indicates CL (close) when the valve is closed, OP (open) when the valve is open and BARBERPOLE when the motor operated valve is in transit.

Each nitrogen inlet valve is controlled by its respective ATM PRESS CONTROL N_2 SYS REG INLET 1 and 2 switch on Panel L2. When the individual switch is open, that valve permits that N_2 source pressure to that system N_2 regulator, providing the respective N_2 supply valve is open. When the individual switch is closed, the N_2 supply pressure is isolated from that system N_2 regulator. An indicator above each switch shows the open or indicates CL (close) when the valve is closed, OP (open) when the valve is open and BARBERPOLE when the motor operated valve is in transit.

The N_2 regulators in the primary and secondary supply systems reduce pressure to 10,350 mmHg (200 psi). Each N_2 regulator is a two-stage regulator with the second stage functioning as a relief valve. The second stage relieves pressure overboard at 14,237 mmHg (245 psi).

The regulated N_2 pressure of each N_2 system is supplied to the N_2 crossover valve, the H_2O (water) tank regulator inlet valve, and the O_2/N_2 controller valve in each N_2 system.

The N_2 XOVER (crossover) manual valve connects both regulated N_2 systems when the valve is open. When it is closed, the N_2 regulated supply systems are isolated from each other. A check valve between the N_2 regulator and N_2 crossover valve in each N_2 regulated supply line prevents reverse flow of N_2 when the crossover valve is open.

The H_2O TK (tank) N_2 REG INLET valve in each N_2 regulated supply system permits N_2 to flow into the H_2O regulator when that valve is manually opened. The closed position isolates the N_2 regulated supply from the H_2O regulator.

The H_2O tank regulator of each system reduces the 10,351 mmHg (201 psi) supply pressure to 879 mmHg (17 psi). Each H_2O tank regulator is a two-stage regulator. The second stage relieves pressure into the crew module cabin at a differential pressure of 1,034 mmHg (20 psi).

Two partial pressure oxygen (PPO_2) sensors are located in the crew module mid deck cabin air supply duct and another PPO_2 sensor is located in the flight deck return air duct. The PPO_2 sensors in the mid deck cabin air supply duct are sensor A and B and provide inputs to the PPO_2 CONTR (controller) SYS 1 controller and switch. Sensor C provides inputs to a flight deck controller.

When a PPO_2 CNTR switch is positioned to NORM (normal), the corresponding PPO_2 controller and PPO_2 sensor, in conjunction with the ATM PRESS CONTROL PPO_2 SNSR (sensor)/VLV switch on panel L2 in the NORM position, provide electrical power to the corresponding ATM PRESS CONTROL O_2/N_2 CNTLR VLV switch on Panel L2 for SYS 1 and SYS 2. When the O_2/N_2 CNTRL VLV switch is set on AUTO, electrical power automatically controls the nitrogen valve in the corresponding regulated nitrogen supply. If O_2 is required in the crew module cabin, the nitrogen valve is automatically closed, the 14.7 psi cabin regulator opens (providing the 14.7 psi CABIN REG INLET manual valve on that system is open), the 10,350 mmHg (200 psi) nitrogen supply in the manifold drops below 5,175 mmHg (100 psi), and oxygen flows through the check valve and cabin regulator into the crew module cabin. When sufficient oxygen is present in the cabin as determined by the PPO_2 sensor, the nitrogen valve is opened and 10,350 mmHg (200 psi) nitrogen enters the manifold. The nitrogen closes the oxygen check valve and flows through the 14.7 psi cabin regulator into the crew module cabin. Oxygen partial pressure is maintained at 178 mmHg (3.45 psi). The OPEN and CLOSE positions of the N_2/O_2 CNTLR VLVE SYS 1 and SYS 2 switch on Panel L2 permit the crew to control the nitrogen valve in each system manually. The REVERSE position of the PPO_2 SNSR/VLV switch on Panel L2 allows Controller B to SYS 1 and Controller A to SYS 2.

If the 14.7 psi CABIN REG INLET manual valves of SYS 1 and SYS 2 are closed, the crew module cabin pressure will decrease to 8 psi. The PPO_2 CONTR SYS 1 and SYS 2 switches on Panel L2 are positioned to EMER (emergency) for the corresponding nitrogen system which selects the 113.8 mmHg (2.2 psi) oxygen partial pressure. The corresponding PPO_2 sensor and controller, through the corresponding PPO_2 CONTR switch and the PPO_2 SNSR/VLV switch on NORM, provide electrical inputs to the corresponding O_2/N_2 CNTRL VLV switch. The electrical output from the applicable O_2/N_2 CNTLR VLV switch controls the nitrogen valve in that supply system in the same manner as in the 14.7 psi mode except that the crew module cabin partial pressure is maintained at 113.8 mmHg (2.2 psi). In this mode, the crew members would use supplemental oxygen from the portable oxygen system and masks.

The O_2 system Nos. 1 and 2 and N_2 system Nos. 1 and 2 flows are monitored and sent to the O_2/N_2 FLOW rotary switch on Panel 01. The rotary switch permits SYS 1 O_2 or N_2 or SYS 2 O_2 or N_2 flow to be monitored on the O_2/N_2 FLOW meter on Panel 01 in PPH (pounds per hour).

PPO_2 Sensors A and B monitor the oxygen partial pressure and transmit the signal to the PPO_2 SENSOR select switch on Panel 01. When the PPO_2 SENSOR switch is positioned to SENSOR A, oxygen partial pressure from Sensor A is monitored on the PPO_2 meter on Panel 01 in psia. If the switch is set on SENSOR B, oxygen partial pressure from Sensor B is monitored.

The cabin pressure sensor transmits directly to the CABIN PRESS meter on Panel 01 and is monitored in psia.

A separate PPO_2 Sensor C monitors the crew module cabin oxygen partial pressure and transmits it to a flight deck CRT.

The RED CABIN ATM caution and warning light on Panel F7 would illuminate from any of the following monitored parameters:

Cabin pressure below 14.0 psia or above 15.4 psia

PPO_2 below 2.8 psia or above 3.6 psia

O_2 flow rate above 5 lbs/hr

N_2 flow rate above 5 lbs/hr

A klaxon will sound in the crew module cabin if the dP/dT (which stands for change in pressure versus change in time) is greater than 0.05 psi per minute.

The temperature and pressure of the primary and secondary nitrogen and emergency oxygen tanks are monitored and transmitted to the SM (systems management) computer. This information is used to compute O_2 and N_2 quantities.

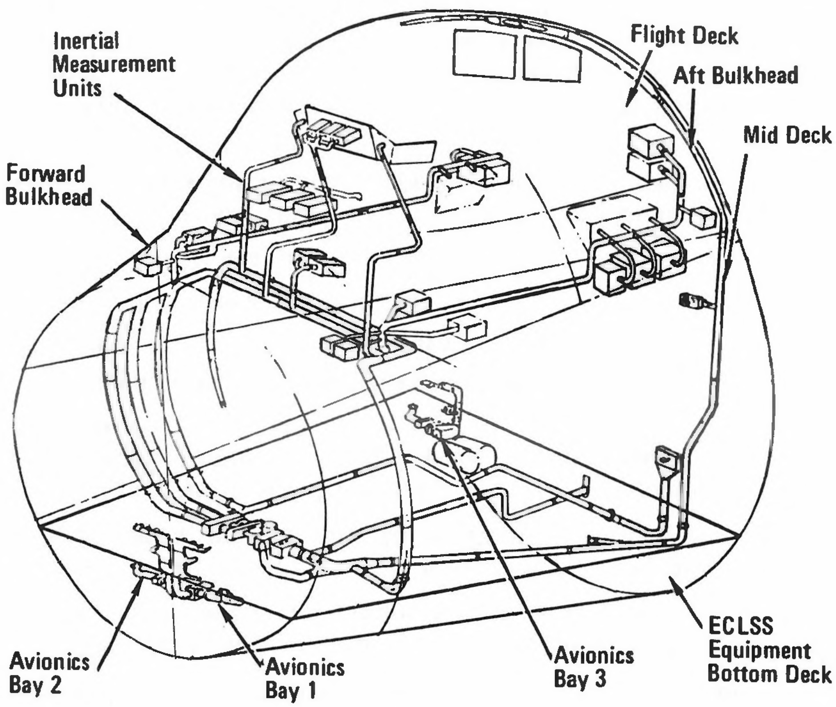
If the crew module cabin pressure is lower than the pressure outside the cabin, the negative pressure relief valves will open in a 10 to 36 mmHg (0.2 to 0.7 psi) range differential permitting flow into the crew module cabin. The maximum flow rate at 25 mmHg (0.5 psi) differential is 0 to 296 kilograms (654 lbs) per hour.

The crew module cabin vent and vent isolation valves provide the capability of venting the cabin to ambient following the prelaunch cabin pressure integrity test. These two valves are in series, thus both valves must be open to vent the cabin. The CABIN VENT VALVE ISOL (isolation) switch on Panel L2 opens and closes the cabin vent isolation valve. An indicator above the switch indicates OP (open) when the valve is open, CL (close) when the valve is closed, and BARBERPOLE when the valve is in transit. The CABIN VENT VALVE switch on Panel L2 opens and closes the cabin vent valve. An indicator above the switch indicates in the same manner as in the cabin vent isolation valve. When both these valves are open, the maximum flow 10 mmHg (0.2 psi) differential is 408 kilograms (900 lbs) per hour.

The two parallel cabin relief valves provide protection against overpressurization of the crew module cabin above 828 mmHg (16.0 psi) differential. Each cabin relief valve has a back-up, motor-operated isolation valve. CABIN RELIEF switch A controls cabin relief valve A and CABIN RELIEF switch B controls cabin relief valve B. When a switch is positioned ENABLE, the corresponding cabin relief valve is enabled and when the switch is positioned to CLOSE, the valve is disabled. An indicator above each switch shows whether the valve is open or closed. Each cabin relief valve will flow a maximum of 68 kilograms (150 lbs) per hour.

Approximately one hour and 26 minutes before launch, the crew module cabin is pressurized by ground support equipment to approximately 864 mmHg (16.7 psi) for a leak check. The CABIN VENT ISOL and VENT valves are then opened and the crew module is vented down to 786 mmHg (15.2 psi) or lower. The CABIN VENT ISOL and VENT valves are then closed.

CABIN AIR. Cabin air cools the cabin avionics electronic units, the crew, and passengers. Some avionics are cooled by air loops within the avionics bays. These loops are not pressure isolated from the crew cabin, although each avionics bay contains a closeout cover to minimize air interchange and thermal losses to the cabin environment: therefore, equipment contained in these air loops meets outgassing and flammability requirements to minimize toxicity levels resulting from outgassing materials. Low-toxicity materials also are used in the crew cabin habitable areas.



Cabin Air

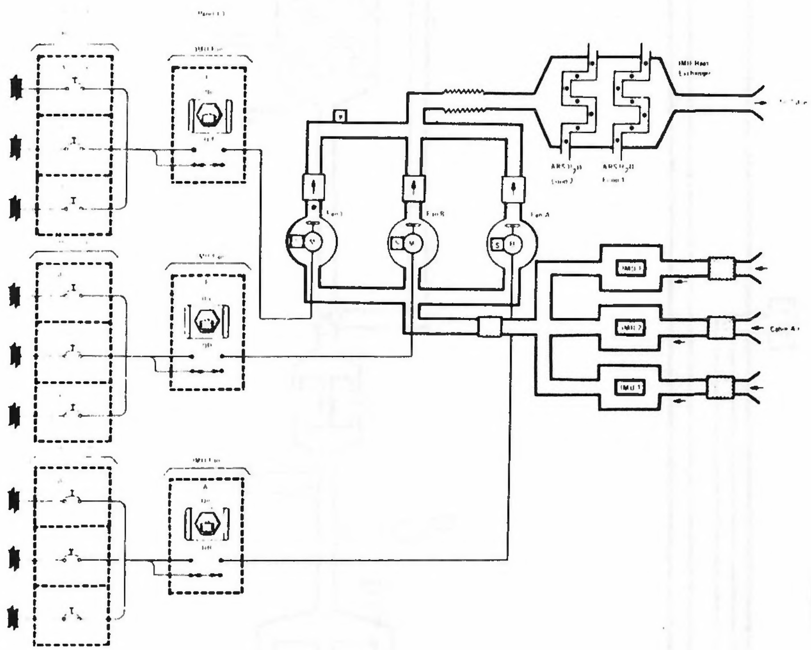
The crew module cabin contains five air loops: the cabin, three avionics bays and the inertial measurement unit cooling loop. The crew module cabin atmosphere is drawn through the cabin through a 300-micron filter by one of two cabin fans located downstream of the filter.

Each of the cabin fans is controlled individually by the CABIN/FAN switch on Panel L1. CABIN FAN A switch turns cabin fan A on and off. CABIN FAN B switch controls cabin fan B. Normally only one fan is used at a time.

The cabin air is then ducted to the two lithium hydroxide (LiOH) canisters where carbon dioxide (CO_2) is removed and activated charcoal removes odors and trace contaminants. CO_2 is maintained at 7.6 mmHg (0.147 psia) maximum. The two LiOH-activated charcoal canisters are replaced alternately every 12 hours (for four crew members) via an access door in the mid deck floor.

The cabin atmosphere is then ducted to the crew module cabin heat exchanger where the cabin air is cooled by the water coolant loops. Humidity condensation is removed from the cabin heat exchanger by a fan separator which draws air and water from the cabin heat exchanger, separates the air and water, routes the water into waste water tanks, and ducts the air through its exhaust into the cabin. The two separator fans are individually controlled by the HUMIDITY SEP (separator) switch on Panel L1. HUMIDITY SEP switch A controls separator fan A and HUMIDITY SEP switch B operates separator fan B. The fan separators separate the air and water by centrifugal force and remove up to 118 kilograms (4 lbs) of water per hour. Only one fan separator is used at a time.

A small portion of the revitalized/conditioned air from the cabin heat exchanger is ducted to the carbon monoxide removal unit (installed after the second flight of Orbiter 102 and subsequent orbiters), which converts carbon monoxide into CO_2. A bypass duct carries cabin air around the heat exchanger and it mixes with the revitalized/conditioned air to control the crew module cabin return air at the selected temperature of between 18 to 26 plus or minus 1°C (65 to 80 plus or minus 2°F). The CABIN TEMP (temperature) CNTLR (controller) switch on Panel L1 selects the cabin temperature controller and the CABIN TEMP SELECTOR rotary switch on Panel L1 selects the desired cabin temperature which controls the mixing of the bypass air and revitalized/conditioned air before its return to the crew module cabin ducting.



IMU Air Loop

The three inertial measurement units (IMU's) are cooled by cabin air drawn through the 300-micron filter and across the three IMU's by one of three parallel fans. The air is cooled by the water coolant loops which flow through the IMU heat exchanger and the cooled air is returned to the cabin. Each of the IMU fans is controlled by individual IMU FAN A, B, and C switches on Panel L1. When the applicable switch is positioned to ON, that fan is on and when positioned to OFF, that fan is off. A check valve installed at the outlet of each fan prevents reverse air flow through the standby fan chambers.

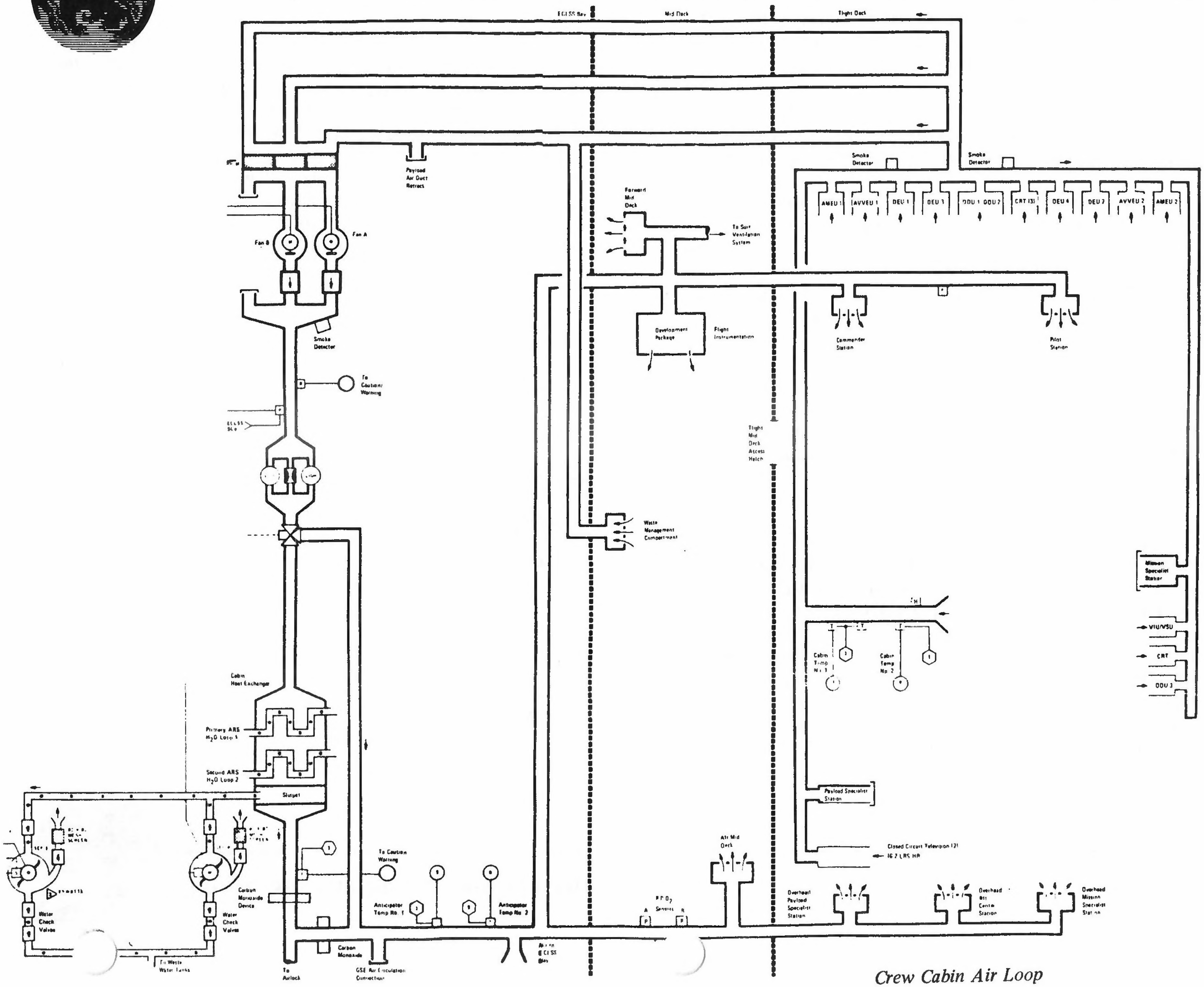
Each of the three electronic avionics equipment bays has identical air-cooled systems. Air is directed into the avionics bays at floor level and is drawn through avionics units by connectors at the back of each unit. The air then returns to the fan package inlet and 300-micron filter upstream of two fans. The air is cooled by the heat exchanger for each avionics bay. The water coolant loops cool the air and the cooled air is returned to the avionics bays. The two fans in each avionics bay are controlled by an AV (avionics) BAY FAN switch on Panel L1. When the applicable AV BAY FAN switch is positioned to ON, that fan is on and when positioned to OFF, that fan is off. A check valve in the outlet of each fan prevents a reverse flow in the standby fan.

The air outlet temperature in each avionics bay and the cabin heat exchanger is monitored and sent to the AIR TEMP (temperature) rotary switch on Panel 01. The rotary switch positioned to AV BAY 1, 2 or 3 position or CAB (Cabin) Hx (heat exchanger) out position permits that temperature to be displayed on the AIR TEMP meter on Panel 01.

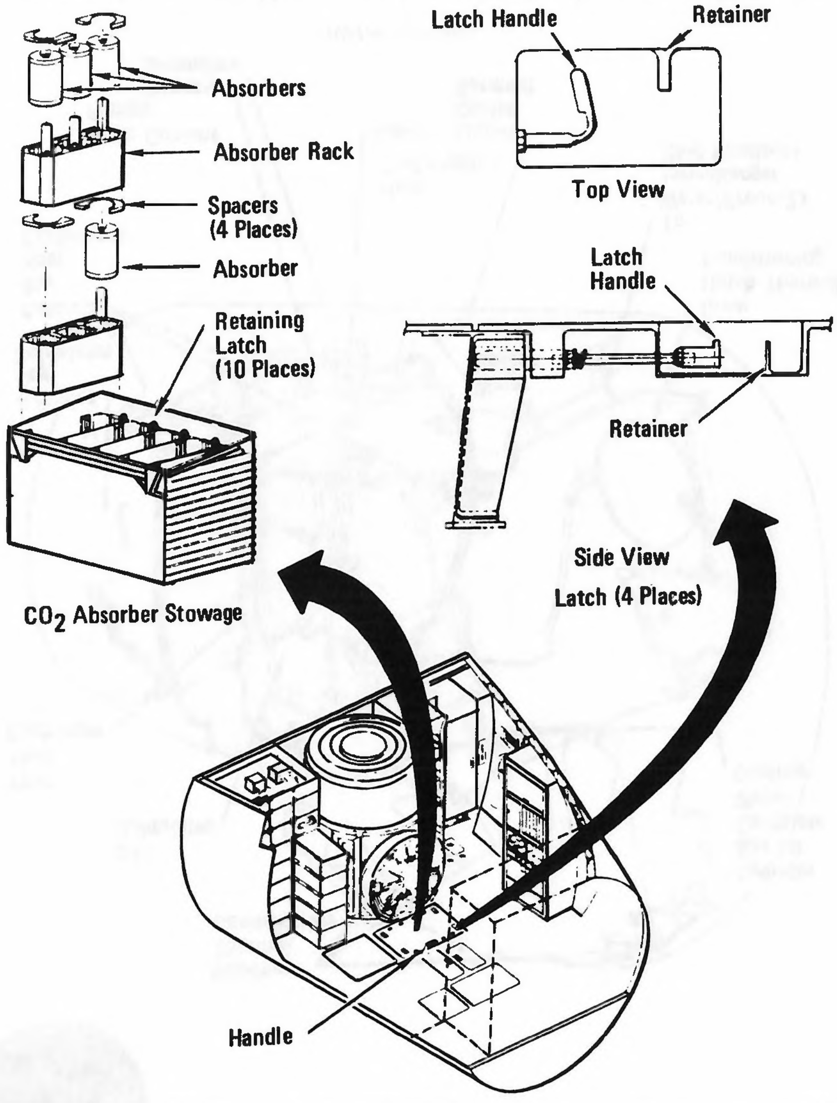
The air outlet temperature in each avionics bay and the cabin heat exchanger provides inputs to the YELLOW AV BAY CABIN AIR caution and warning light on Panel F7. The light would illuminate if any of the avionics bay outlet temperatures are above 57°C (135°F), if the cabin heat exchanger outlet temperature is above 18°C (65°F), or if the cabin fan delta pressure is below 5 mmHg (0.1 psi) or above 15 mmHg (0.3 psi).

If the payload bay contains the Spacelab pressurized module, a kit is installed to provide ducting from the crew cabin into the tunnel from the crew compartment mid deck to theSpacelab.

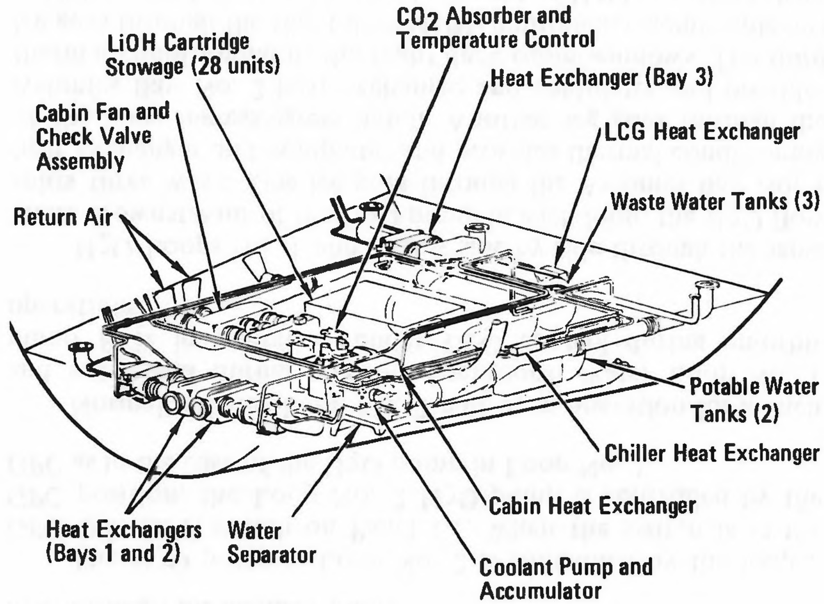
The fan separators, cabin heat exchanger, avionics heat exchangers and inertial measurement unit heat exchanger, waste water tanks, LiOH filters, carbon monoxide unit, and waste and potable water tanks are located beneath the mid deck crew compartment floor.



Crew Cabin Air Loop



Carbon Dioxide Absorbers



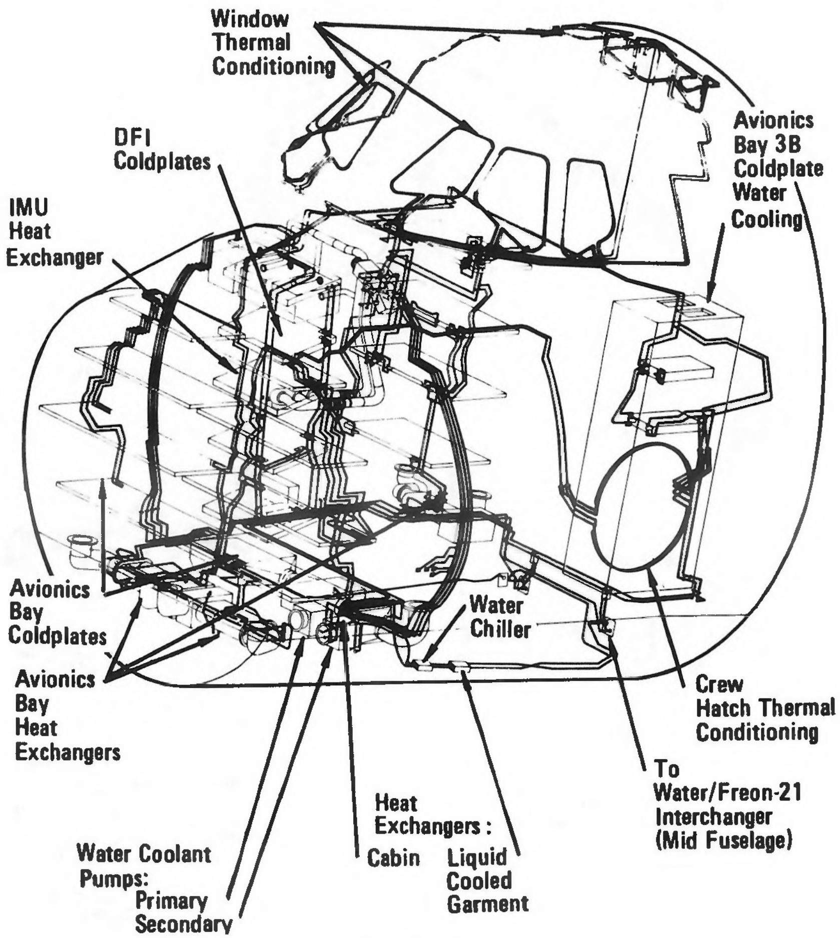
Crew Cabin Bottom Deck ECLSS

WATER COOLANT LOOP SUBSYSTEM

The WCLS provides thermal conditioning of the crew cabin by collecting heat through air-to-water heat exchangers and transferring the heat from the water coolant loops to the Freon coolant loops.

There are two complete and separate water (H_2O) coolant loops that flow side by side and have the capability of operating at the same time. The only difference between H_2O Loop No. 1 and 2, is that Loop No. 1 has two H_2O pumps and Loop No. 2 has one pump.

Some of the electronic units in the avionics bays are mounted on coldplates with H_2O flowing through the coldplates. The heat generated by that electronic unit is transferred to the cold-



Water Coolant

plate and into the H_2O, which carries the heat away from the electronic unit. Coldplates mounted on the shelves in the avionics bays are connected in a series – parallel arrangement with respect to the H_2O flow.

The H_2O pumps in Loop No. 1 are controlled by the H_2O pump Loop 1 A and B switch on Panel L1 in conjunction with the Loop 1 GPC (general purpose computer) OFF and ON switch on Panel L1. The H_2O PUMP LOOP 1 switch positioned to GPC permits the GPC to energize relays, which cycles H_2O pump A or B for six minutes over a period determined by the GPC and the position of the H_2O PUMP LOOP 1 A or B position. The ON position of the Loop No. 1 switch energizes the relays and allows H_2O pump A or B to operate as determined by the position of the H_2O PUMP LOOP 1 A or B switch. The OFF position of the Loop No. 1 switch de-energizes the relays, which prohibits operation of the Loop No. 1 H_2O pump A and B. A ball-type check valve downstream of the H_2O pumps prevents reverse flow through the standby pump.

The H_2O pump in Loop No. 2 is controlled by the loop 2 GPC ON, OFF switch on Panel L1. When the switch is in the GPC position, the Loop No. 2 H_2O pump is controlled by the GPC as in the case of the H_2O pump in Loop No. 1.

Normally Water Loop No. 2 will be in operation for launch and entry and during on-orbit operations. Water Loop No. 1 pump B is in operation under GPC control during on-orbit operations.

H_2O Loops No. 1 and 2 flow side by side through the same areas. Downstream of the H_2O pump in each loop, the H_2O flow splits three ways. One leg goes through the Avionics Bay No. 1 heat exchanger and coldplates and provides thermal conditioning of the crew ingress/egress hatch. Another leg goes through the Avionics Bay No. 2 heat exchanger and coldplates and provides thermal conditioning of the flight deck cabin windows. The third leg goes through the flight deck MDM (multiplexer/demultiplexer) coldplates with a predetermined amount of H_2O bypassing these coldplates and splits again into two parallel paths for the Avionics Bay No. 3 and 3B coldplates. All of these paths come together again and flow through the forward development flight instrumentation (DFI) coldplates to the Freon-21/H_2O heat exchanger where excess heat from the WCL is transferred to the Freon-21 coolant loops.

The WCL loop flows from the Freon-21/H_2O interchanger through the liquid-cooled garment heat exchanger, H_2O water chiller, cabin heat exchanger, and IMU heat exchanger to the H_2O pump inlet.

The controller for each H_2O coolant loop is enabled by its respective loop BYPASS MODE and MAN switch on Panel L1. The AUTO position of the LOOP 1 BYPASS and LOOP 2 BYPASS switch allows the corresponding controller to position the bypass valve of that H_2O loop automatically. The MAN (manual) position of the LOOP 1 BYPASS and LOOP 2 BYPASS switch disables the automatic control of the bypass value of that H_2O loop and enables the corresponding LOOP 1 BYPASS and LOOP 2 BYPASS MAN INCR DECR switch. The crew would position the LOOP 1 BYPASS and LOOP 2 BYPASS MAN switch to INCR (increase) or to DECR (decrease) to control that bypass valve in that H_2O coolant loop manually. The bypass valve is adjusted manually prior to launch to provide 408 to 453 kilograms (900 to 1,000 pounds per hour) through the interchanger. The control system remains in the manual mode for the entire flight.

The accumulator in each H_2O coolant loop provides a positive pressure on the H_2O pump of the corresponding H_2O loop in addition to providing for thermal expansion capability in that H_2O loop. Each accumulator is pressurized with gaseous nitrogen at 983 to 1,811 mmHg (19 to 35 psi).

The pressure at the outlet of the H_2O pump in each coolant loop is monitored and sent to the H_2O PUMP OUT PRESS switch on Panel 01. When the switch is on LOOP 1 or LOOP 2, that H_2O coolant loop pressure can be monitored on the H_2O PUMP OUT PRESS meter on Panel 01 in psia.

The YELLOW H_2O LOOP caution and warning light on Panel F7 would illuminate if H_2O Coolant Loop No. 1 pump outlet pressure is below 2,328 mmHg (45 psi) or above 4,114 mmHg (79.5 psi) or if H_2O Coolant Loop No. 2 pump outlet pressure is below 2,328 mmHg (45 psi) or above 4,191 mmHg (81 psi).

The pump inlet and outlet pressure of each H_2O coolant loop is monitored and transmitted to the SM GPC for CRT capabilities.

In summary, with use of the crew module cabin structural thermal capacity, the crew cabin will not exceed 32°C (90°F) during entry or until after flight crew egress, assumed to be 15 minutes after touchdown.

ACTIVE THERMAL CONTROL SUBSYSTEM

The ATCS provides orbiter heat rejection during all mission phases. The ATCS is composed of two Freon coolant loops (FCL's), coldplate networks for avionics cooling, liquid/liquid heat exchangers for orbiter systems cooling, and three heat sink subsystems (radiators, flash evaporator, and ammonia boiler).

During ground operations (checkout, prelaunch, and postlanding), orbiter heat rejection is provided by the ground support equipment (GSE) heat exchanger in the Freon coolant loops through ground system cooling.

From liftoff to an altitude greater than 42,672 meters (140,000 feet) – approximately 125 seconds – thermal lag is utilized. Approximately 125 seconds after liftoff, the flash evaporator subsystem is activated and provides orbiter heat rejection of the Freon coolant loops via water boiling. Flash evaporator operation continues until the payload bay doors are opened on orbit.

When the payload bay doors are opened, radiator panels attached to the forward payload bay doors are deployed. The forward two panels on each side of the orbiter are deployed away from the payload bay doors and radiate from both sides. The aft radiator panels on the forward portion of the aft payload bay doors remain affixed to the doors and radiate only from the upper surface. On-orbit heat rejection is provided by the radiator panels; however, during orbital operations where combinations of heat load and spacecraft attitude exceed the capacity of the radiator panels, the flash evaporator subsystem is automatically activated to meet total system heat rejection requirements.

At the conclusion of orbital operations the flash evaporator subsystem is activated, and the payload bay doors closed with the radiator panels retracted in preparation for entry.

The flash evaporator subsystem operates during entry to an altitude of 36,576 meters (120,000 feet) where boiling water can no longer provide adequate Freon coolant temperatures. Through the remainder of the entry phase and postlanding until ground cooling is connected, heat rejection of the Freon coolant loops is provided by the evaporation of ammonia through the use of the ammonia boiler. When ground cooling is initiated during postlanding, the ammonia boilers are shut down and heat rejection of the Freon coolant loops is provided by the GSE heat exchanger.

There are two complete and identical Freon coolant loops; Loop No. 1 and Loop No. 2. Both Freon coolant loops operate at the same time. There are two Freon coolant pumps in each loop with only one pump active per loop. The FCL's flow side by side except for the radiator panels. Freon-21 is utilized in the FCL's.

The Freon pumps in each Freon coolant loop are controlled by individual FREON PUMP switches on Panel L1. When the FREON PUMP LOOP 1 or LOOP 2 switch is positioned to A, the Freon pump A in that Freon coolant loop is in operation. If positioned to B, the Freon pump B in that loop is in operation. The OFF position prohibits either Freon pump operation. A ball-check valve downstream of the pumps in each Freon coolant loop prevents a reverse flow through the standby pump.

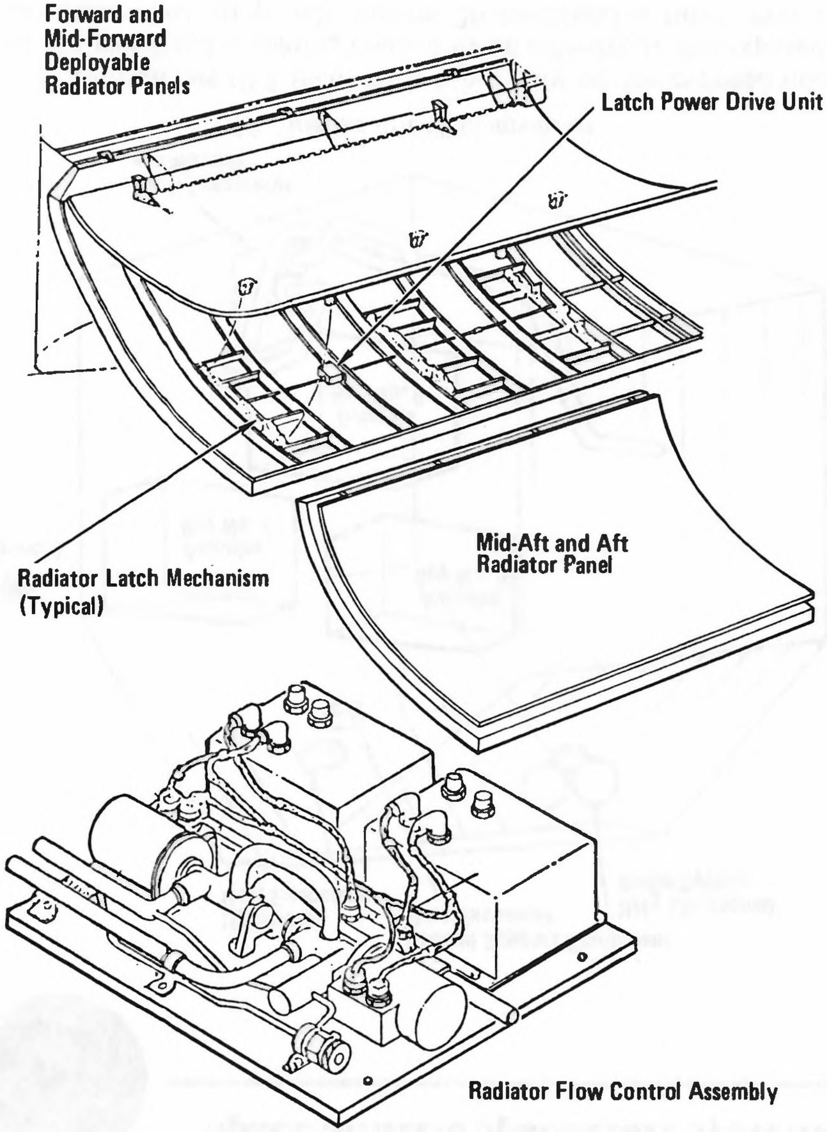
When a Freon coolant pump is operating, Freon is routed in parallel through the fuel cell heat exchangers and the midbody coldplate network to cool electronics avionics units. The Freon coolant from the midbody coldplate network and fuel cell heat exchanger reunites in a series flow path before entering the hydraulics heat exchanger, which extracts energy from theFreon-21 coolant loop to heat the hydraulic systems fluid loops during on-orbit hydraulic circulation thermal conditioning operations. During the prelaunch and boost phase of the mission and during the atmospheric flight portion of entry through touchdown, the hydraulic system heat exchanger transfers excess heat from the hydraulic systems to the Freon-21 loops.

The FCL's flow to the radiator system, which consists of three radiator panels (baseline configuration – two deployable and one fixed) attached to the inside of the forward payload bay doors (deployable) and the forward section of the aft payload bay doors (fixed) and a flow control assembly for each loop. The radiator panels are normally bypassed during ascent and descent. On-orbit, the flow control assembly controls the temperature of the loop (mixed radiator outlet) through use of a variable flow control valve which mixes hot bypassed flow with cold flow from the radiator. The temperature is controlled to either 3°C (38°F) or 13°C (57°F) setpoint temperature.

The radiator panels on each side of the orbiter are configured to flow in series while flow within each panel is parallel through a bank of tubes connected by an inlet and outlet collector manifold.

To increase heat rejection capability for large payloads requiring 29,000 Btu (British thermal units) per hour of heat rejection an additional radiator panel can be kitted into the network by attaching a fixed radiator panel to the inside of the aft section of the aft payload bay doors. The baseline configuration is designed for payloads rejecting 21,500 Btu per hour.

A bypass valve in each FCL system permits Freon-21 to bypass the radiators except on-orbit. When Freon-21 temperaures at the radiator outlet exceed 5°C (41°F), the radiator system heat rejection capability has been exceeded and the flash evapor-



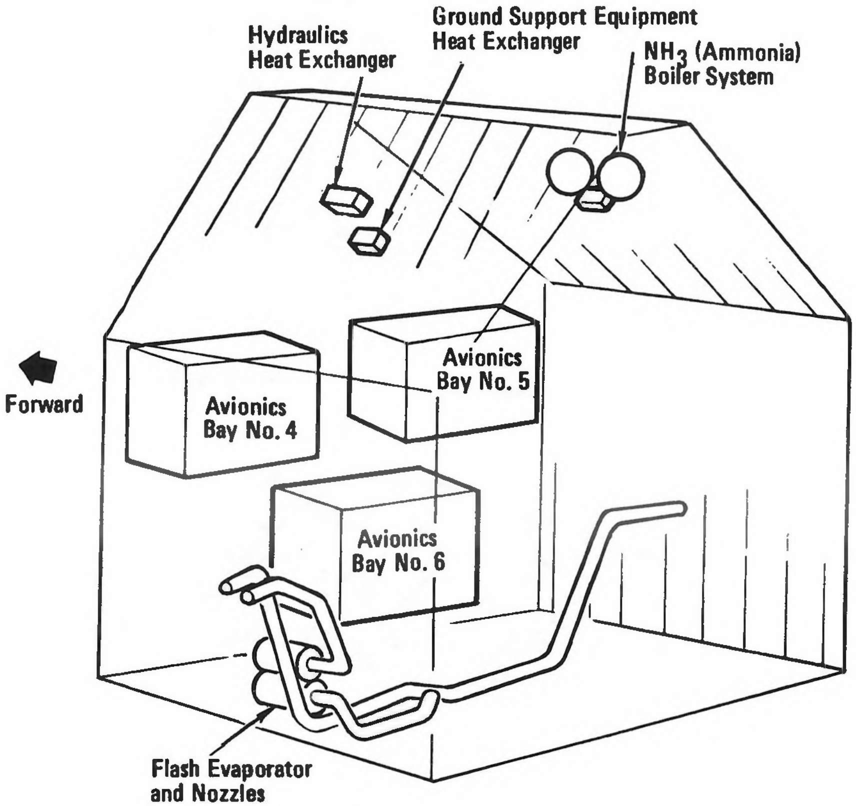
Radiators and Radiator Flow Control Assembly

ators are activated automatically to produce the required Freon-21 temperature.

During boost and entry, each deployable radiator panel is secured to the payload bay door by six motor-driven latches. Deployment for on-orbit operations is by a motor-driven, torque-tube-lever arrangement. The aft four fixed radiator panels are attached to the payload bay doors by a ball joint arrangement at a maximum of 12 places. The ball joints compensate for movement of the payload bay door and radiator panel caused by thermal expansion and contraction of each member. The forward four radiator panels, when deployed, expose both sides of the radiator panels to increase the heat rejection capability of the Freon-21 loops. The four forward radiator panels are deployed 35.5° from the payload bay doors.

The ATCS pumps, interchanger, fuel cell heat exchanger, payload heat exchanger, flow proportioning valve modules, and mid body coldplates are located in the lower forward portion of the mid fuselage. The radiators are attached to the payload bay doors. The hydraulic systems' heat exchanger, ground support equipment heat exchanger, ammonia boiler, flash evaporator, and aft avionic bay coldplates are located in the orbiter aft fuselage. The radiator flow control assemblies are located in the lower aft portion of the mid fuselage.

The radiator panels are constructed of an aluminum honeycomb face sheet 320 centimeters (126 inches) wide and 459 centimeters (320 inches) long. The forward deployable radiator panels are two sided with a core thickness of 2.2 centimeters (0.9 inch). They have longitudinal tubes bonded to the internal side of both facesheets. The forward deployable panels contain 68 tubes each, with a tube spacing of 4.82 centimeters (1.90 inches). Each tube has an inside diameter of 0.332 centimeter (0.131 inch). Each side of the forward deployable radiator panels has a coating bonded by an adhesive to the facesheet consisting of silver backed Teflon tape for proper emissivity properties. The aft fixed panels are one sided with a core thickness of



Aft Fuselage ECLSS Components

1.27 centimeters (0.5 inch) with tubes only on the exposed side of the panel and a coating bonded by an adhesive to the exposed facesheet. The aft panels contain 26 longitudinal tubes with a tube spacing of 12.5 centimeters (4.96 inches), and each tube has an inside diameter of 0.45 centimeter (0.18 inch). The additional thickness of the forward radiator panels is required to meet deflection requirements when the orbiter is exposed to ascent acceleration.

The radiator panels on the left-hand side (port) facing forward are connected in series with Freon-21 coolant loop 1. The panels on the right-hand-side (starboard) facing forward are connected in series with Freon-21 coolant loop 2.

The RAD (radiator) CONTROLLER LOOP 1 and LOOP 2 switch on Panel L1 enables Loops No. 1 and 2 controllers A and B. When the switch for a loop is positioned to AUTO A, radiator controller A automatically controls the radiator flow control valve in that loop, which maintains the desired radiator mixed outlet temperature as determined by the RAD CONTROLLER OUT TEMP switch on Panel 1A. When the switch is turned to AUTO B, controller B is enabled and automatically controls the radiator control valve in the corresponding loop as it did in the case of AUTO A.

The RAD CONTROLLER OUT (outlet) TEMP switch on Panel L1 enables the selected controller A or B in Loop No. 1 or 2 to control the radiator outlet temperature of that loop. The radiator outlet temperatures in Loops No. 1 and 2 are automatically controlled at 3°C (38°F) when the switch is on NORM (normal) and at 13°C (57°F) when it is on HI. The flash evaporator is activated automatically when the radiator outlet temperature exceeds 5°C (41°F).

The RAD CONTROLLER MODE 1 and MODE 2 switch on Panel L1 permits automatic control of radiator flow control valve or manual control of the radiator bypass valve. When in the AUTO position for Loop No. 1 or 2, the respective radiator flow control valve automatically controls the radiator outlet temperature. When in the MAN position, the automatic control of the radiator flow control valve in the corresponding loop is inhibited and the bypass valve is controlled by the RAD CONTROLLER MAN (manual) SEL (select) 1 or 2 switch on Panel L1. If the MAN SEL switch is positioned to the RAD FLOW, the bypass valve permits the Freon-21 to flow through the radiators. If positioned to BYPASS position for that loop, the bypass valve permits the Freon-21 in that loop to bypass the radiators.

The indicator located above the RAD CONTROLLER MAN SEL 1 and 2 switch on Panel L1 indicates the position of the bypass valve in that loop. The indicator indicates BYPASS when the bypass valve is in the bypass position, BARBERPOLE when the motor operated valve in that loop is in transit from the bypass or radiator flow position, and indicates RAD when the bypass valve in that loop is in the radiator flow position.

Freon-21 from the radiator flow control valve assembly is routed to the ground support equipment (GSE) heat exchanger, which is used during ground operations (checkout, prelaunch and postlanding) for orbiter heat rejection of the Freon-21 coolant loops. Freon-21 from the GSE heat exchanger is then directed through the ammonia boiler, then the flash evaporator.

The flash evaporator is used to reject orbiter heat loads from the Freon-21 coolant loops during ascent above 42,672 meters (140,000 feet) and entry above 36,576 meters (120,000 feet) altitude and to supplement the radiators in orbit.

There are two flash evaporators (high-load and topping) contained in one envelope. The evaporators are cylindrical and have a finned inner core. The hot Freon-21 from the coolant loops flows around the finned core and water is sprayed onto the core from the nozzles in each evaporator. The water vaporizes, cooling the Freon-21. In the low-pressure areas above 36,576 meters (120,000 feet), water vaporizes quickly. The water changing from liquid to vapor removes approximately 1,000 Btu per hour per 2.2 kilograms (1 pound) of water. The water supply is obtained from the potable water storage tanks and supply systems A and B.

The flash evaporators are controlled by the FLASH EVAP CONTROLLER switches on Panel L1. The evaporators have three controllers: PRI (primary) A, PRIB and SEC (secondary). These controllers are controlled by the PRIA, PRIB and SEC switches on Panel L1. Normally, only one of these switches is used at a time. When one of the PRIA, PRIB or SEC switches is positioned to GPC, that controller is turned on by the BFS (backup flight system) computer as the orbiter ascends above 42,672 meters (140,000 feet) and is turned off by the BPS computer during entry at 36,576 meters (120,000 feet). The ON position of the PRIA, PRIB, or SEC switch provides power to the flash evaporator controller directly. OFF removes power from the flash evaporator controller. The PRIA controller utilizes water system A; the PRIB controller utilizes water system B. The SEC controller uses water system A if the SEC switch on Panel L1 is in SPLY A or B if the SEC switch on Panel L1 is in SPLY B and the HI-LOAD EVAP switch is in the ENABLE position.

The PRIA and B controllers control evaporator outlet Freon-21 loop temperatures at 3°C (39°F) and the SEC controller controls evaporator outlet Freon-21 loop temperatures at 16°C (62°F).

The applicable flash evaporator controller pulses water into the evaporators, cooling the Freon-21. The steam generated in the topping evaporator is ejected through two sonic nozzles at opposing sides on each side of the aft end of the orbiter, reducing payload water vapor pollutants on orbit and minimizing vent thrust effects on the orbiter guidance navigation and control system. The hi-load evaporator is used in conjunction with the topping evaporator during ascent and entry when higher Freon-21 coolant loop temperatures impose a greater heat load which requires a higher heat rejection. The HI-LOAD EVAP ENABLE switch on Panel L1 must be in the ENABLE position for hi-load evaporator operation. After leaving the high-load evaporator, the Freon-21 would also flow through the topping evaporator for additional cooling. The steam generated by the hi-load evaporator is ejected through a single sonic nozzle on the left-hand (port) side aft end of the orbiter facing forward. The hi-load evaporator would not normally be used on orbit because it has a propulsive vent and might pollute the payload.

The topping evaporator can be used to dump excess potable water from the storage tanks. In this mode, the radiator flow control valve assembly has an alternate control temperature of 13.8°C (57°F), which is used during forced water dumping.

Heaters are employed on the topping and hi-load steam ducts of the flash evaporator to prevent freezing. The HI-LOAD DUCT HTR (heater) switch on Panel L1 positioned to A provides electrical power to the thermostatically controlled A heaters on the hi-load evaporator steam duct and steam duct exhaust. The B position provides electrical power to the thermostatically controlled B heaters on the hi-load evaporator steam duct and steam duct exhaust. The A/B position provides electrical power to both the A and B heaters. The C position provides electrical power to the thermostatically controlled C heaters on the hi-load evaporator steam duct and steam duct exhaust. The OFF position removes electrical power from the heaters.

The TOPPING EVAPORATOR HEATER DUCT switch on Panel L1 positioned to A provides electrical power to the thermostatically controlled A heaters on the topping evaporator. The B position provides electrical power to the thermostatically controlled B heaters. The A/B position provides electrical power to both the A and B heaters. The C position provides electrical power to the thermostatically controlled C heaters. The OFF position removes electrical power from the heaters.

The TOPPING EVAPORATOR HEATER L (left) and R (right) NOZZLE switches on Panel L1 provide electrical power to the topping evaporator left and right nozzles. The L and R AUTO A position provides electrical power to the left and right A nozzle heaters to maintain nozzle temperatures between 4° and 21°C (40° and 70°F). The L and R AUTO B position provides electrical power to the left and right B nozzle heaters.

The ammonia boilers are used below 36,576 meters (120,000 ft) during entry. There are two individual storage and control systems that feed a boiler containing common ammonia passages and individual Freon-21 coolant loop passages. This provides a safe return from orbit for any combination of failures in both the Freon-21 coolant loops and ammonia boilers. The ammonia boilers are enabled by the NH_3 CONTROLLER A and B switches on Panel L1. The NH_3 CONTROLLER A and B switches are positioned to PRI (primary)/GPC before entry. As the orbiter descends through 36,576 meters (120,000 ft), the BFS commands controller A and B on. The ammonia (NH_3) boiler is a shell-and-tube type with a single pass on the ammonia side and two passes for each Freon-21 coolant loop. The NH_3 flows in the ammonia tubes and the Freon-21 coolant loops flow over the tubes, cooling the Freon-21. Freon temperature is maintained at 1.1°C (34°F) by regulating the flow of NH_3 through the boiler. At each Freon-21 coolant loop outlet of the ammonia boiler, Freon-21 temperature is monitored by three temperature sensors. One sensor is associated with the primary NH_3 flow control valve of that loop and another with the secondary NH_3 flow control valve. The third sensor automatically switches control from the primary to secondary system in the event of low Freon-21 coolant loop outlet temperature. The NH_3 boiler exhaust is vented overboard in the aft section of the orbiter adjacent to the bottom right side of the vertical tail. The boiler continues to operate and provide heat rejection until a ground cooling cart is connected to the GSE heat exchanger after touchdown.

The NH_3 CONTROLLER A and B switch positioned to SEC (secondary) ON provides electrical power directly to the NH_3 controllers and boiler, which operates the NH_3 boiler. The OFF position removes electrical power from the NH_3 controllers and boiler. The two complete ammonia boiler systems each have an ammonia storage tank. The capacity of each tank is 29 kilograms (64.7 lbs). There is about 20 kilograms (46 lbs) usable for each system. Each ammonia tank is pressurized with helium at an operating pressure between 4,295 and 28,462 mmHg (83 to 550 psi). There are three valves in the plumbing leading to each ammonia boiler. The isolation valve is opened by the primary or secondary controller. The next two control valves in line are modulating valves whose position is dependent on the amount of current to the control motor. The full closed position of these two valves inhibits about 75 percent of the flow. The fail mode of the control valves is open. If the fault circuitry detects the Freon-21 outlet temperature below minus 0.4°C (31.25°F) for greater than 10 seconds, an automatic switchover occurs in the controller for that NH_3 system and the secondary controller takes over. A relief valve provides over pressurization protection for each ammonia tank.

Freon Coolant Loops No. 1 and 2 are routed from the flash evaporator in series, then into parallel paths. A portion of the Freon-21 loop is directed to the aft avionics bays in the aft fuselage of the orbiter, where some of the electronic avionics units in Avionics Bays 6, 5 and 4 are mounted on coldplates which transfer the heat generated from the electronic avionics units to the Freon-21, which carries the generated heat away. The Freon-21 coolant loops also flow through the coldplates of Rate Gyro Assemblies 1, 2, 3 and 4.

The remaining parallel path downstream of the flash evaporator heats oxygen by means of a heat exchanger to 4.4°C (40°F) prior to entering the cabin. The source of the oxygen is the cryogenic storage and distribution system. This path branches in parallel at the flow proportioning valve in each Freon-21 loop.

The FLOW PROP (proportioning) VLV (valve) switch on Panel L1 for each coolant loop controls the flow of Freon-21 to the payload heat exchanger or the water/Freon-21 interchanger. The INTCHGR (interchanger) position of the LOOP 1 and LOOP 2 switches controls the respective flow proportioning valve to allow maximum Freon-21 flow through the water/Freon-21 interchanger. The PAYLOAD HX (heat exchanger) position of the LOOP 1 and LOOP 2 switch controls the respective flow proportioning valve to allow maximum Freon-21 flow through the payload heat exchanger. The indicator above the respective LOOP 1 and LOOP 2 switch on panel 01 indicates ICH when the water/Freon interchanger position, BARBERPOLE when that valve is in transit from ICH or PL, and indicates PL (payload) when that valve is in the payload heat exchanger position.

The parallel paths from the water/Freon-21 interchanger and payload heat exchanger are reunited with the parallel path from the aft avionics bay and rate gyro assemblies. The coolant loop then returns to its Freon-21 coolant pump.

The accumulator in each Freon-21 coolant loop is a metal, bellows-type pressurized with gaseous nitrogen. The accumulator provides for thermal expansion and keeps a positive suction on the coolant pumps.

The Freon-21 coolant loop temperature and flow rate are monitored on Panel O1. When the FREON FLOW EVAP OUT TEMP switch is positioned to LOOP 1 or LOOP 2, the respective Freon-21 coolant loop evaporator outlet temperature is monitored on the FREON EVAP OUT TEMP in degrees Fahrenheit and the Freon flow is monitored on the FREON FLOW in PPH (pounds per hour). This information is also transmitted to the RED FREON LOOP caution and warning light on Panel F7. The RED FREON LOOP light would illuminate if the Freon-21 Coolant Loop No. 1 or 2 evaporator outlet temperature fell below 0°C (32°F) or rose above 15.6°C (60°F) or if the Freon-21 flow rate is below 544 kilograms (1,200 lbs) per hour.

During the development flights of Orbiter 102, a separate development flight instrumentation (DFI) Freon-21 coolant loop system is installed in the payload bay for cooling the electronic units installed on the DFI pallet in the payload bay. The DFI has its own Freon-21 pumps, accumulator and plumbing. The DFI Freon-21 coolant loop is cooled by the payload bay heat exchanger. The DFI FREON PUMPS SELECT switch is located on Panel R11. Position 1 controls pump No. 1 and position 2 controls pump No. 2.

FOOD, WATER AND WASTE MANAGEMENT

The food, water, and waste management (FWW) subsystem provides the basic life support functions for the flight crew.

Each potable water tank has a usable capacity of 74 kilograms (165 pounds), is 90 centimeters (35.5 inches) in length and 39 centimeters (15.5 inches) in diameter, and weighs 17.9 kilograms (39.5 pounds) dry.

Potable water is generated by the three fuel cells at a maximum of 11.34 kilograms (25 lbs) per hour. The hydrogen-enriched water from the fuel cell passes through a hydrogen (H_2) separator, where 95 percent of the excess hydrogen is removed. The H_2 separator consists of a matrix of silver palladium tubes which have an affinity for H_2. The H_2 is dumped overboard through a vacuum vent.

The water from the H_2 separator is directed to the water storage system, which can consist of a total of six tanks for the development flight test missions. The tanks are identified as A, B, C, D, E and F. Tanks E and F are located on the mid deck of the crew module cabin for the development flights of Orbiter 102. Each tank has a solenoid inlet and outlet valve except Tanks E and F, which have manual valves. Water for Tank A passes through a microbial check valve. The microbial check valve adds approximately 3-5 parts per million iodine to the water. The water from the microbial check valve is directed to Tank A and the galley. The crew can select cooled or ambient water. Cooling is accomplished by passing through the water chiller, where heat is rejected to the water coolant loop.

When the Tank A inlet valve is closed or Tank A is full, the water is directed through a 77 mmHg (1.5 psi) relief valve which routes the water to Tank B.

When the Tank B inlet valve is closed or Tank B is full, the water is directed through another 77 mmHg (1.5 psi) relief valve to Tanks C, D, E and F. The inlet and outlet valves for each tank can be opened or closed selectively to use water; however, the Tank A outlet valve fill always remains closed since the water has been treated by passage through the microbial filter for crew consumption.

The controls for the water supply system are located on Panels R12 and ML31C. Tanks A, B, and C are controlled from Panel R12, Tank D from Panel ML31C, and Tanks E and F from manual valves on the tanks.

Tanks A, B and C have their own SUPPLY H_2O INLET and OUTLET switch on Panel R12. When the SUPPLY H_2O INLET TK A, B, or C switch is positioned to OPEN, the inlet valve for that tank allows water into the tank. If positioned to CLOSE, the inlet valve isolates the water inlet from that tank. An indicator located above the respective switch on Panel R12 indicates OP (open) when the corresponding valve is open, BARBERPOLE when that valve is in transit and CL (close) when that valve is closed. Tank D has its own SUPPLY H_2O TK INLET switch and indicator on Panel ML31C and operates in the same manner as the ones on Panel R12.

A SUPPLY H_2O GALLEY SPLY (supply) VLV (valve) switch on panel R12 permits or isolates water from Tank A to the galley. The switch has OPEN and CLOSE positions and an indicator above the switch shows whether the valve is open or closed or BARBERPOLE when that valve is in transit.

When the valve is open, water is supplied to an Apollo water dispenser and water gun at ambient and chilled temperatures for drinking and food reconstitution. The ambient water temperature range is 18 to 35°C (65 to 95°F) and the chilled water temperature range is 6 to 13°C (43 to 55°F) for the development flight tests. In the operational flights, the water supply to the galley is directed to a hot water heater, which provides hot water at a temperature range between 68 and 73°C (155 to 165°F). Chilled water is supplied at the galley at a temperature range of 7 and 12°C (45 to 55°F).

Tank A is used for crew consumption. To prevent contamination the Tank A OUTLET valve will remain closed. Tank B is used for flash evaporator cooling on-orbit. Tanks A and B may also be dumped overboard as necessary to provide space for water storage. Tanks C, D, E, and F are saved full of water for contingency purposes.

Each of the water tanks is pressurized from the nitrogen pressure supply system at a pressure of 905 mmHg (17.5 psi) to force the water from the water storage tanks for flash evaporator use. The H_2O ALTERNATE PRESS switch on Panel L1 provides the capability of referencing the water tank pressurization system to ambient cabin pressure if the N_2 system fails. If the switch is positioned to OPEN, cabin atmosphere pressure is supplied to the water tanks for pressurization. The CLOSE position isolates the cabin atmosphere pressure from the water tank pressurization supply system.

From the water supply tanks, two evaporator feed lines referred to as SYSTEM A and SYSTEM B are routed to the flash evaporators in the aft fuselage. A crossover valve between the two supply systems is controlled by a SUPPLY H_2O CROSSOVER VLV switch on Panel R12. If the switch is positioned to OPEN, the crossover valve allows all six water tanks for the flash evaporator or overboard dumping. When the switch is positioned to CLOSE, the crossover valve is closed, and Tanks C, D, E and F cannot be used for the flash evaporator A water supply. But by opening the B supply isolation valve, these tanks can flow to B water supply and, thus, the flash evaporator. An indicator above the switch indicates OP (open) when the valve is open, CL (close) when the valve is closed, and BARBERPOLE when the valve is in transit.

The water supply system B to the flash evaporator has an additional supply isolation valve. This valve is controlled by the SUPPLY H_2O B SPLY (supply) ISOL VLV switch on Panel R12. An indicator above the switch on Panel R12 indicates the same as in the previous paragraph.

Water from Tank A, when full, and Tank B can also be dumped overboard. The overboard dump consists of a dump isolation valve in the crew module cabin and a dump valve in the mid fuselage. Both valves are closed unless performing a dump. The SUPPLY H_2O DUMP ISOL VALVE switch on Panel R12 opens and closes the dump isolation valve in the crew module cabin. The SUPPLY H_2O DUMP VLV switch on panel R12 controls the dump valve in the mid fuselage. Indicators above the switches indicate OP (open) when the valves are open, CL (close) when the valves are closed, and BARBERPOLE when the valves are in transit.

The water dump nozzle has a heater to prevent freezing. The heater is controlled by the DUMP VALVE ENABLE/ NOZZLE HEATER switch on Panel R12. The nozzle heater is powered when the switch is positioned to ON.

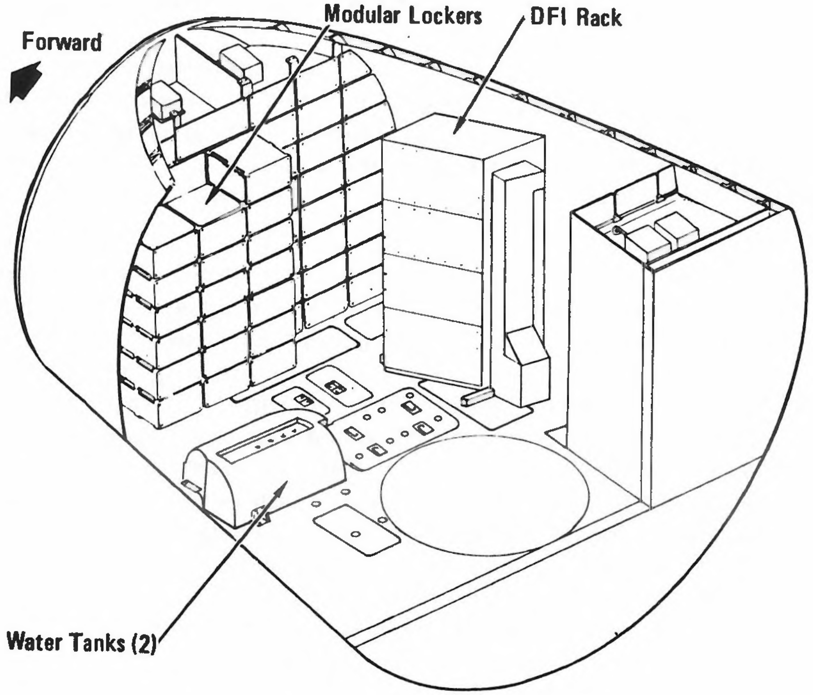
There are thermostatically controlled line heaters upstream of the water dump nozzle on the line. The heaters are powered by circuit breakers H_2O LINE HTR A and B on Panel ML86B.

The system A and B water feedlines to the flash evaporator are approximately 30 meters (100 ft) long. Redundant heaters installed along the length of the water lines are controlled by the FLASH EVAP FEEDLINE HTR switches on Panel L2. The switch enables the thermostatically controlled heaters on H_2O system A and B. Heater circuit one or two on PR1 then A or B may be selected. The OFF position of either switch removes electrical power from the heaters.

The orbiter is equipped with food and facilities for food stowage and preparation and dining to provide each crew member with three meals plus snacks per normal day in orbit; two meals on launch day; one meal on reentry day; and an additional 96 hours of contingency food. The food supply and food preparation facilities are furnished by the government and are designed to accommodate variations in the number of crew and duration of flight, ranging from a crew of two for one day to a crew of seven for 30 days.

In the development test flights, the food preparation system is limited. It consists of the water dispenser, food warmer, food trays, food (meal menu and pantry), and food system accessories.

The food warmer is a small, portable, thermostatically controlled unit that can warm meals for two crew members simultaneously. The food trays serve as a dining surface with restraints for food items and provide each crew member with associated

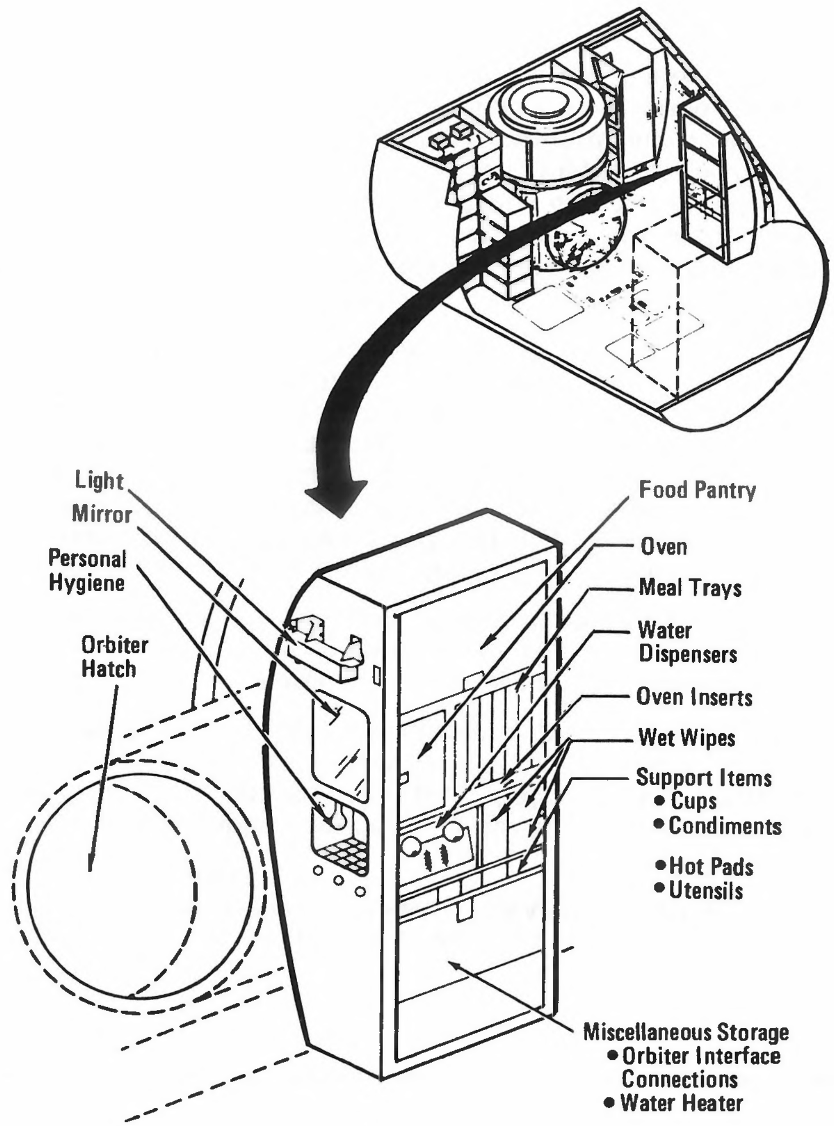


Crew Cabin Mid Deck Potable Water Tanks
(Development Flight Tests)

dining accessories. The food consists of individually packaged items of dehydrated, thermo-stabilized, irradiated, intermediate moisture, natural form and beverage form. Meal accessories include salt, pepper, sauces, etc., as well as candy, gum, vitamin tablets, wipes, utensils, thermal pads, drinking containers, and germicidal tablets.

For the operational flights, the food preparation system consists of the galley, food trays, work/dining table, food, and food system accessories.

The galley is a multi-purpose facility that provides food preparation facilities, stowage of meal accessories, food trays and



Galley and Hygiene (Operational Flights) oven inserts, and food, and volume for seven crew member days of food-related trash. The food consists of individually packaged items of rehydratable, thermo-stabilized, and ready-to-eat food and beverages.

The food warmer will heat a meal for two crew members in approximately two hours in the development flight tests.

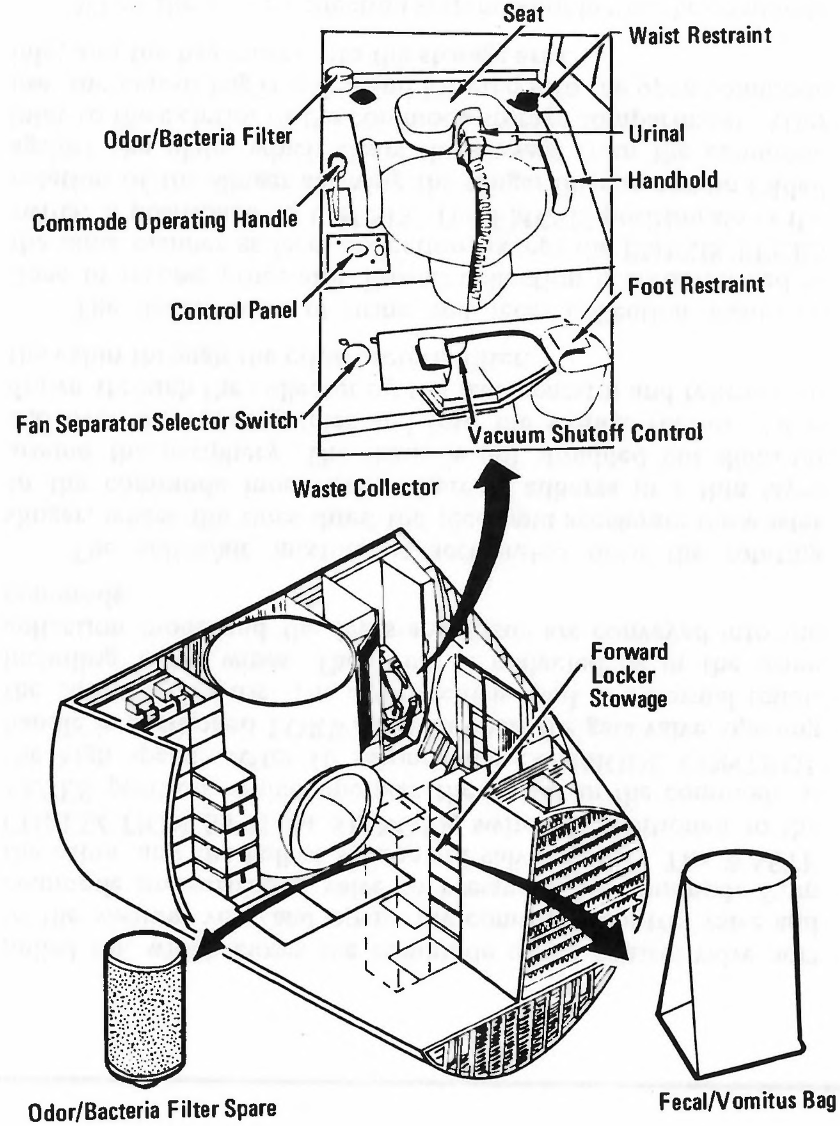
The oven will heat a meal for up to seven in approximately 90 minutes and has a heating range of 62 to 85°C (145 to 185°F) in the operational flights.

The waste management system collects, processes, and stores solid and liquid wastes. Liquid waste consists of urine, perspiration, lung vapor, and liquid waste from the galley and extravehicular mobility unit (EMU). Solid waste consists of feces, emesis, tissues, etc. The waste collection system collects and processes these wastes in zero gravity.

The waste collection system is located in the mid deck of the orbiter crew compartment in a 73-centimeter (29-inch) wide area immediately aft of the crew side hatch. The unit is 68.58 x 68.58 x 73.66 centimeters (27 by 27 by 29 inches) and has two major independent and interconnected assemblies: one handles fluids and the other (commode) the solid waste.

The fluid processing assembly collects liquid waste from the urinal, personal hygiene station or EMU (extra-vehicular mobility unit). The urinal assembly is a flexible hose with a cup that can be used in a standing position or be attached to the commode by a pivoting mounting bracket for use in a seated position. The urinal is a contoured cup that can accommodate both males and females.

The urinal can be used sitting or standing by either male or female crew members. The user positions the waste collection system (WCS) control panel MODE switch to the WCS/EMU, which opens the fan separator valve 1 or 2and turns fan separator 1 or 2 on, dependent upon the WASTE COLLECTION SYSTEM FAN SEP switch in position 1 or 2, opens the fan separator



Location of Waste Collection in Mid Deck control valve, and opens the urine collection valve. The COMMODE CONTROL handle is positioned to OFF, which allows the commode outlet control valve port to vent to the WSC VACUUM VALVE in the OPEN position to expose the wastes in the commode to vacuum for drying, and the ballast air control valve is open. The airflow in the urinal and the ballast airflow enter the system via the debris screen inlet, orifice and ballast air control valve. The ballast air mixes with the urine transport air flow in the fan-separator. The air/liquid mixture is drawn into the fan separator by the fans. The air/fluid separation takes place in the fan separator. The air/fluid mixture in the fan separator is conveyed axially in the rotating chamber and centrifugal force draws the fluid along the outer walls of the chamber. A stationary pitot tube picks up the fluid and pumps it to the waste water tank. The air is drawn axially out of the rotating chamber by blower action and passes through a filter which removes all bacteria and odors before returning the air into the cabin. An EMU water dump is accomplished in the same manner.

Wash water removal from the personal hygiene station (PHS) is accomplished in the same manner, except the WASTE COLLECTION SYSTEM MODE switch is positioned to PHS. This portion of the system is inactive for the development flight tests.

The check valves at the waste water outlet from the fan separators provide back pressure for proper separator operation and prevent back-flow through the inactive separator.

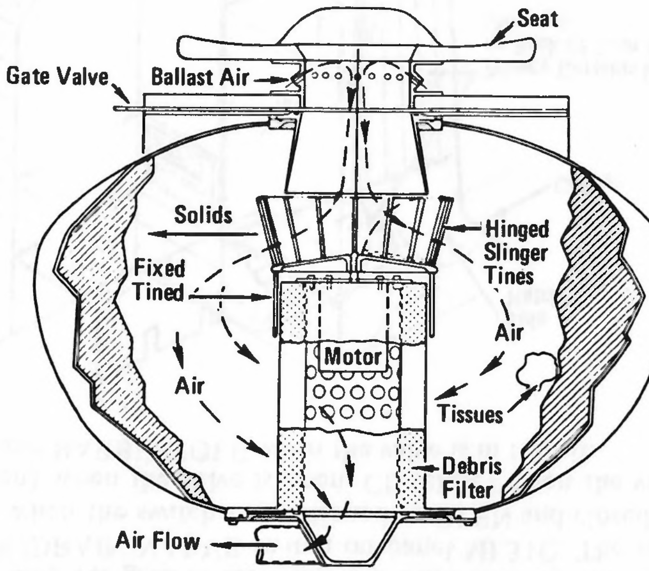
The urine and feces collection mode is accomplished by using the foot restraints, sitting on the commode, locking the restraint belt, and positioning the urinal.

The WCS VACUUM VALVE is positioned to open, opening the vacuum vent valve. This operation is performed prior to the first usage and reversed subsequent to the last usage. The WCS/EMU MODE switch is positioned to WCS/EMU as in the urine collection mode. The COMMODE CONTROL handle is pulled up, which closes the commode outlet control valve port to the vacuum vent and opens the commode control valve and commode pressurization valve to pressurize the commode from the cabin, and the ballast air control valve is open. The WASTE COLLECTION SYSTEM SLINGER switch is positioned to the FECES position, which operates the slinger in the commode at the high speed. After 10 seconds the COMMODE CONTROL handle is positioned FORWARD to open the gate valve, opening the collector for use. The equipment is used as a normal toilet, including toilet wipes. The urine is collected as in the urine collection mode and the feces and tissue are conveyed into the commode.

The solids/air mixture is accelerated onto the rotating slinger, where the tines shred the feces and accelerate the wastes to the commode inner wall, where it adheres in a thin layer around the periphery. The tissue is not shredded but slides up and over the rotating tines and into the storage volume. Air is drawn through the collector by the fan/separator and returned to the cabin through the odor/bacteria filter.

The deactivation of urine and feces collection would be done in reverse procedure. Emesis collection is accomplished in the same manner as feces collection, except the EMESIS/FECES switch is positioned to EMESIS. The EMESIS position slows the rotation of the slinger allowing the slinger tines to remain folded against the plate, which clears the passage from the commode inlet to the exterior of the commode storage compartment. After use, the emesis bag is sealed and lowered into the open commode inlet and the bag moves into the storage area.

When the waste collection system is not in use, the commode is exposed to space vacuum. The COMMODE CONTROL handle on OFF opens the commode through the commode outlet control valve port to the WCS VACUUM VALVE, which is positioned to OPEN to dry and disinfect the wastes. The WASTE COLLECTION SYSTEM MODE switch positioned to OFF closes the urine collection valve and fan separator outlet control valve.



COMMODE OPERATION

Pulling up gate valve control activates slinger motor. As slinger reaches operational speed (1500 rpm), hinged slinger tines unfold outward. Pushing gate valve control forward opens gate valve for commode use. Feces enters commode through seat opening, drawn in by ballast air flow. Slinger tines shred feces and deposit it in thin layer on commode walls. Tissues move up over slinger tines and settle at bottom of collector. Ballast air passes through debris filter and hydrophobic filter to fan separators.

For emesis disposal, fecal/emesis selector switch on WCS control panel is moved to emesis position, rotational speed of slinger tines is slowed and tines do not unfold. This allows unobstructed passage of fecal/vomitus bag into commode.

Waste Collector

The commode has a storage capacity equivalent to 210 crew-member days of vacuum-dried feces and toilet tissue. Each crew member-day usage results in 0.12 kilogram (0.27 pound) of fecal and paper waste, including 0.09 kilogram (0.2 pound) of moisture. The commode can accommodate up to four usages per hour.

Heaters are installed on the vacuum vent line and are thermostatically controlled. Heaters are also installed on the vacuum vent nozzle and are enabled by the WASTE H_2O VAC VENT NOZZLE HEATER switch on panel ML31C when positioned to ON. The WCS waste gases are vented overboard through the vacuum vent line and nozzle.

Personal hygiene accommodations for the crew include a personal hygiene station, personal hygiene kits, pressure-packaged personal hygiene agents, towel dispenser, and tissue dispenser.

In the development flight tests, the personal hygiene station is located in the mid deck of the crew cabin and provides ambient temperature water with no drain. The operational flight personal hygiene station is on the aft side of the galley in the mid deck of the crew cabin and provides ambient and hot water plus a drain to the urinal assembly.

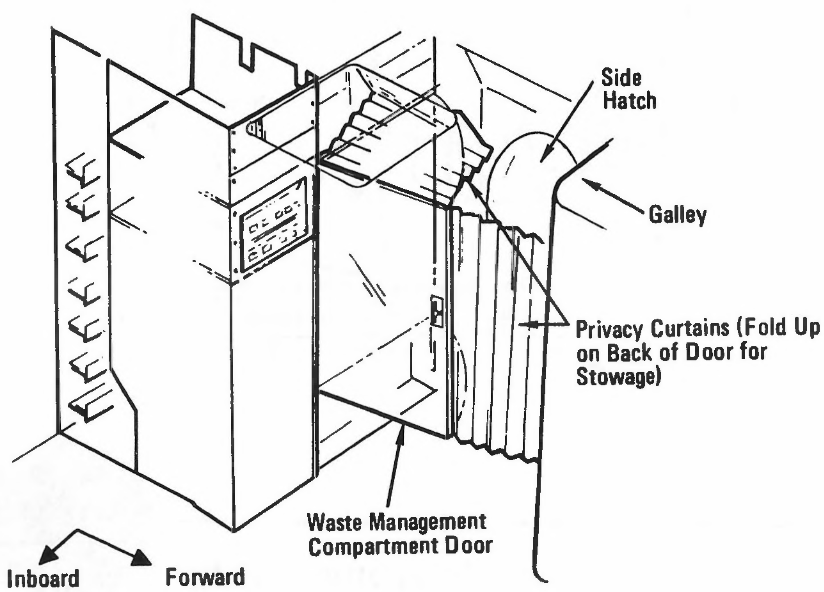
Personal hygiene kits provide for brushing teeth, hair care, shaving, nail care, etc. Pressure-packaged personal hygiene agents are for cleaning hands and for cleaning hands, face, and body. A seven-day supply of towels is provided for each crew member; additional towel dispensers are provided for each crew member for each additional seven days. Tissues are provided to support each crew member for seven days and dispensers are added for each seven days added to a mission.

In the operational flights, two privacy curtains are attached to the waste collection compartment door. One is attached to the top of the door and interfaces with the edge of the inter-deck access. The other is attached to the door and interfaces with the galley. The deployed curtain isolates the waste collection compartment and galley personal hygiene station from the rest of the orbiter mid deck.

The waste water tank receives waste water from the ARS crew compartment humidity separator and the waste collection system urine separator.

The waste water tank usable capacity is 74 kilograms (165 pounds). It is 90 centimeters (35.5 inches) in length and 39 centimeters (15 inches) in diameter and weighs 17.9 kilograms (39.5 pounds) dry.

The waste water tank is pressurized with the same gaseous nitrogen source supply as the potable water tanks. The waste water tank has an inlet and outlet valve. The outlet valve is opened only for ground servicing and controlled by the WASTE H_2O TK/DRAIN VALVE switch on panel ML31C. The valve is opened when the switch is positioned to OPEN and closed when OP (open) when the valve is open, CL (close) when the valve is closed, and BARBERPOLE when the valve is in transit.



Privacy Curtains

The inlet valve permits waste water from the cabin heat exchanger humidity separator and waste fan/separators to enter the waste tank. The inlet valve is controlled by the WASTE H_2O TANK 1 VLVE switch on Panel ML31C. The switch opens and closes the valve. An indicator above the switch indicates the position of the valve in the same manner as the previous paragraph.

In order for waste water to be dumped overboard, the dump isolation valve must be opened. The dump isolation valve is controlled by the WASTE H_2O DUMP ISOL VLV switch on Panel ML31C. An indicator above the switch indicates the position of the valve in the same manner as the previous paragraph.

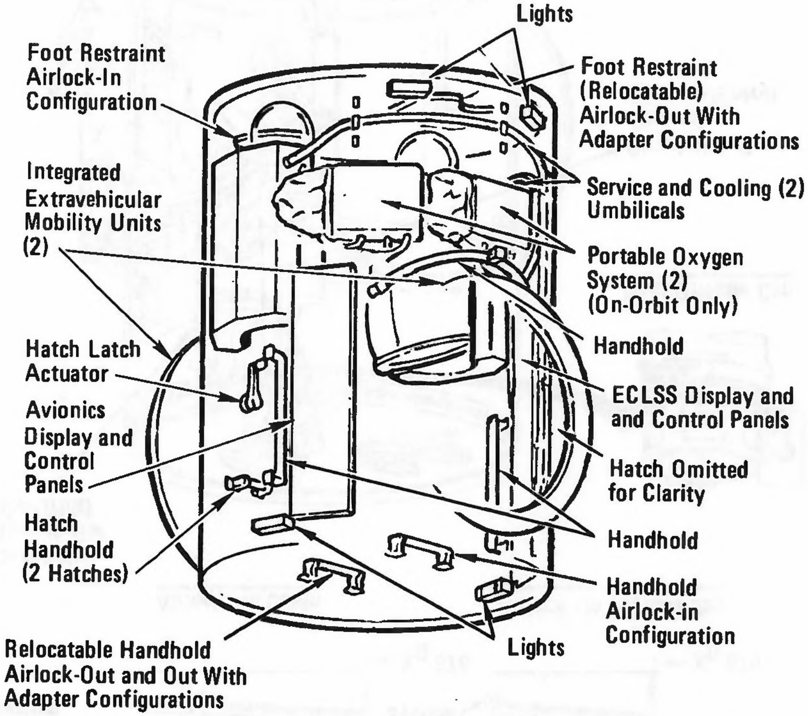
A redundant waste water dump valve must also be opened to dump waste water overboard. This valve is controlled by the WASTE H_2O DUMP VALVE switch on panel ML31C. The WASTE H_2O DUMP ENABLE/NOZ HTR switch must be on before the dump valve can be activated. An indicator above the switch indicates the position of the valve in the same manner as the previous paragraph.

A heater installed on the waste water dump nozzle is turned on and off by the WASTE H_2O DUMP VLV ENABLE/NOZ HTR switch on Panel ML31C. Heaters are installed on the waste water dump line and are thermostatically controlled.

Waste water tanks may be added in parallel if a mission requires additional stowage of waste water. The potable water tanks have a similar overboard dump capability.

AIRLOCK SUPPORT SUBSYSTEM

The airlock eliminates the necessity for crew compartment depressurization for extra-vehicular activity (EVA) and Spacelab operations that require crew/equipment transfer from the crew cabin to the payload bay or to the Spacelab.



Airlock

The airlock is normally located in the mid deck of the crew cabin. It will contain two pressure sealing hatches. The airlock may also be located in one of two additional locations. The airlock may be moved from inside the mid deck of the crew cabin and positioned outside the aft bulkhead of the crew cabin in the payload bay area to provide additional volume in the mid deck. When the airlock is in the payload bay, insulation is installed on the exterior of the airlock for protection against temperature extremes.

If Spacelab is in the payload bay, a transfer tunnel and adapter enables the crew members and equipment to be transferred between Spacelab and the crew cabin. The tunnel mates with a tunnel adapter at the forward end of the payload bay. For EVA, the airlock is positioned on top of the tunnel adapter and stabilized through a structural connection to the crew compartment aft bulkhead. The tunnel adapter will have two access hatches: one on top of the adapter for access to the airlock and the other on the aft end for access to the Spacelab.

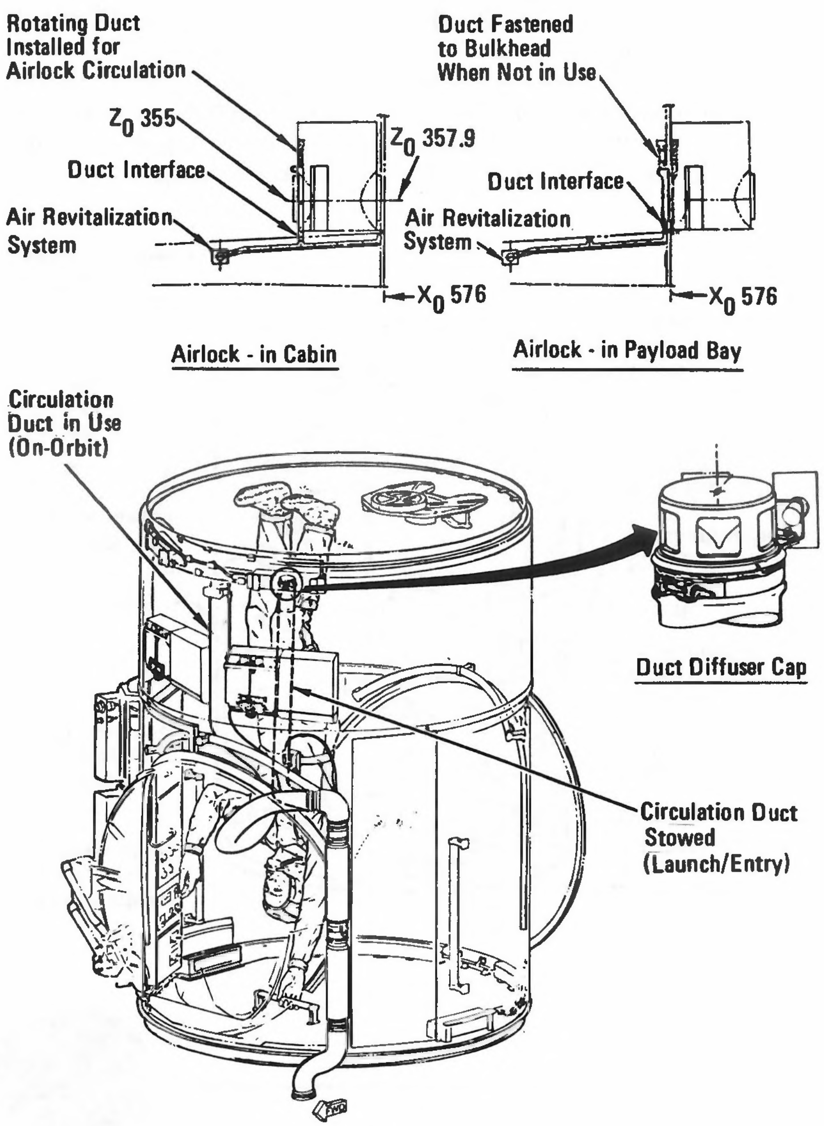
For missions requiring direct docking of two vehicles, a docking module can be substituted for the airlock and installed on the tunnel adapter. The docking module is extendable and provides an airlock function for EVA for two crew members when extended or for one crew member when retracted.

The airlock has an inside diameter of 160 centimeters (63 inches), in 210 centimeters (83 inches) long, and has two 101-centimeter (40-inch) diameter, D-shaped openings, 91 centimeters (36 inches) across, plus two pressure sealing hatches and a complement of airlock support systems.

The airlock volume is 4.24 cubic meters (150 cubic feet). Airlock repressurization is controllable from inside the crew cabin mid deck and from inside the airlock. It is performed by equalizing the airlock and cabin pressure with hatch-mounted equalization valves. Depressurization is controlled from inside the airlock. The airlock is depressurized by venting the airlock overboard.

The airlock provides EVA capabilities up to seven hours and suited intra-vehicular activity (IVA) activities. The airlock support provides airlock pressurization and depressurization, EVA equipment recharge, liquid-cooled garment water cooling, prebreathing support, EVA equipment checkout, donning, and communications.

The EVA equipment recharge station provides oxygen, water, waste water processing, and battery recharge of the extra-vehicular mobility units (EMU's). Oxygen is supplied to the EVA equipment recharge panel from the ARS oxygen supply. Potable water is supplied to the recharge panel from the potable water system. Waste water processing from the EMU is expelled to the waste water system.



Airlock Circulation Duct

A service and cooling umbilical (SCU) is used to transport all supplies (oxygen, H_2O, electrical, communications, etc.) from the airlock control panels to the EMU before and after EVA and during EMU recharge. Cooling for the crew is provided by the liquid-cooled garment circulation system via the SCU and liquid-cooled-garment heat exchanger, which transfers collected heat to the Freon-21 coolant loops.

The EMU's are worn over a liquid-cooled garment similar to a "long john" underwear into which have been woven many feet of flexible tubing that circulates the cooling water.

An EMU power supply/battery charger supplies electrical power to the EMU for pre- and post-EVA operations and recharge of the EMU batteries between EVA's.

PORTABLE OXYGEN SUBSYSTEM

A portable oxygen system is provided for each crew member. The system is used in the event of crew cabin atmosphere contamination and for life support during EVA rescue operations, emergency oxygen after landing if the atmosphere around the orbiter is contaminated, and prebreathing for EVA's for denitrogenizing the crew member's circulatory system.

The portable oxygen system consists of a full face mask with visor, rebreather loop, heat exchanger, oxygen bottle, CO_2 absorber, and condensate collector. It operates independently or connected to the ARS oxygen system. The portable oxygen system rebreather concept limits spillage of oxygen in the crew cabin. A communications cable provides the mask with communications capabilities. A recharge kit consists of the CO_2 absorber cartridge and condensate collector.

The portable oxygen system provides for three hours of operation using the ARS oxygen supply before recharge kit replacement. The portable oxygen provides a walk-around capability for normal and emergency operation. Its internal oxygen supply will provide up to one hour of independent rebreather operation when in EVA personal rescue system. After the internal oxygen supply is depleted, the portable oxygen system can be recharged to provide a 10-minute independent walk-around capability in the rebreather mode.

Oxygen is supplied from the ARS to the portable oxygen system through quick-disconnect flexible hoses.

The development test flights have nine available locations for the portable oxygen system: three at the aft center console, four on the mid deck, and two in the airlock. The operational flights will have 10 portable oxygen system locations available: two in the airlock, four on the mid deck ceiling, and four on the aft center console.

Before the EVA's, the crew member must be denitrogenized to prevent the bends when EVA's are begun in the 212 mmHg (4.1-psi) EMU suit.

If an extra-vehicular activity (EVA) is required to close the payload bay doors, crew cabin pressure will be reduced from 750 mmHg plus or minus 10 mmHg (14.5 plus or minus 0.2 psia) to 465 mmHg plus or minus 23 mmHg (9 plus or minus 0.45 psia) in STS-1 through STS-4 with an 80 percent nitrogen and 20 percent oxygen (mixed gas) composition by the spacecraft's environmental control life support system (ECLSS) atmospheric revitalization pressure control subsystem (ARPCS).

The reduction of the cabin pressure eliminates the 3 1/2 to 4 hours prebreathing of pure oxygen by the two-man flight crew to force nitrogen from their blood in preparation for the 100 percent 258 mmHg (5 psia) of the EVA space suits in addition to providing a prebreathing atmosphere for both crew members simultaneously, thereby facilitating an emergency EVA by the second crewman. The reduction of cabin pressure provides positive denitrogenization and avoids the encumbrance of the portable oxygen system and the servicing and cooling umbilical for prebreathing, permitting the flight crew to accomplish other pre-EVA tasks more effectively.

If an EVA is required, depressurization of the cabin from 750 mmHg plus or minus 10 mmHg (14.5 plus or minus 0.2 psia) to 465 mmHg plus or minus 23 mmHg (9 plus or minus 0.45 psia) will be implemented at an interval of 12 to 67 hours prior to the EVA.

During depressurization, the total cabin pressure and oxygen partial pressure are controlled manually by the flight crew. The cabin pressure reduction results in an oxygen concentration of up to 30 percent. Appropriate analyses and tests are being conducted on equipment and materials to verify their performances in the reduced pressure environment.

During the period of cabin pressure reduction, a procedural powerdown is necessary to maintain operational cabin and avionics temperature levels because of the reduced air cooling capability.

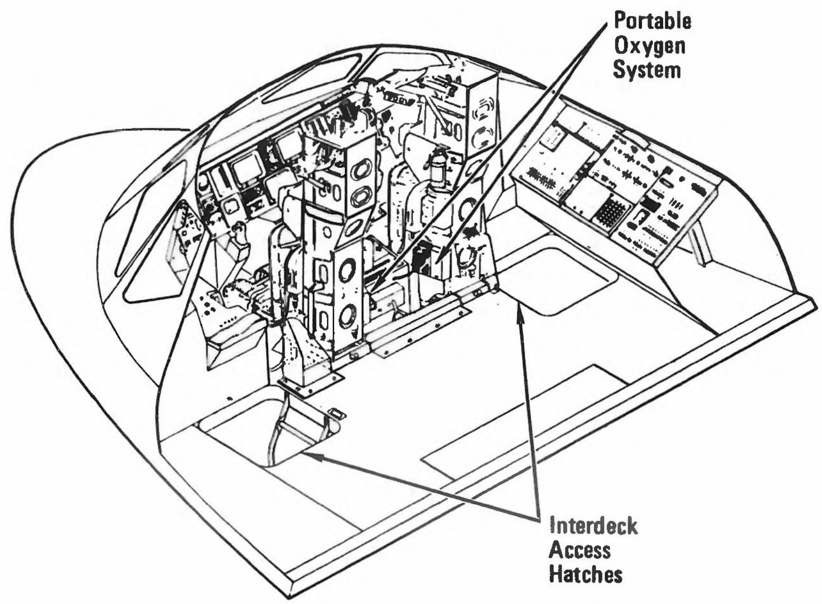
EJECTION ESCAPE SUIT

The ejection escape suit (EES) provides the crew member with a safe environment in the event of crew ejection from the orbiter during high speed – maximum of Mach 2.7 – and high altitude – maximum of 24,384 meters (80,000 feet) – in the development flight tests.

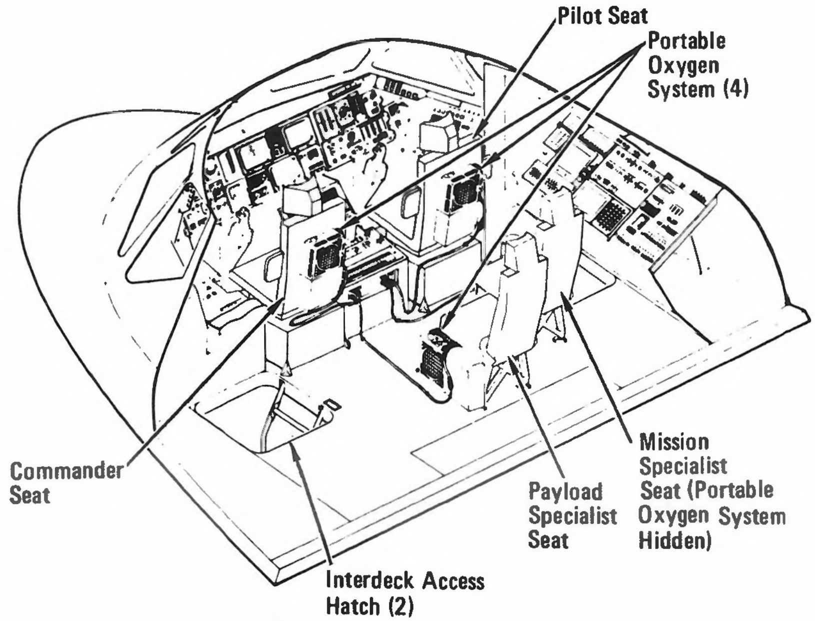
The pressure portion of the EES is a five-layer integral unit covering the torso to the neck, the arms above the wrists, and the legs, including the feet. The EES includes a helmet, gloves, boots, and outer garment.

The EES is donned in preflight phase, removed in orbit, and donned again before descent and worn until after landing.

Regulated oxygen is supplied to the EES helmet by means of a regulator/valve/manifold assembly. The oxygen supply is derived from the power reactant storage and distribution cryogenic system, from the ECLSS ARS emergency breathing oxygen supply, or from the crew member ejection seat oxygen emergency supply during crew ejection. A portable ventilator system is used with the suit before crew members enter the orbiter.



Portable Oxygen Systems (Development Flight)



Portable Oxygen Systems (Operational Flight)

The suit weighs approximately 10 kilograms (23.5 pounds).

The escape suit ventilation system (ESVS) provides ventilation for the EES. Air is drawn from a manifold supplying conditioned cabin air from the ECLSS air circulation duct located in the crew cabin mid deck sleep area. The air is drawn by two compressors and discharged through a check valve to a common manifold for the commander and pilot. The check valve prevents a reverse flow of air in event of a compressor failure. The compressors are located on the crew cabin flight deck immediately behind the ejection seat. The discharge port at the common manifold is connected to the commander's and pilot's suits by flexible hose at the suit ventilation fitting.

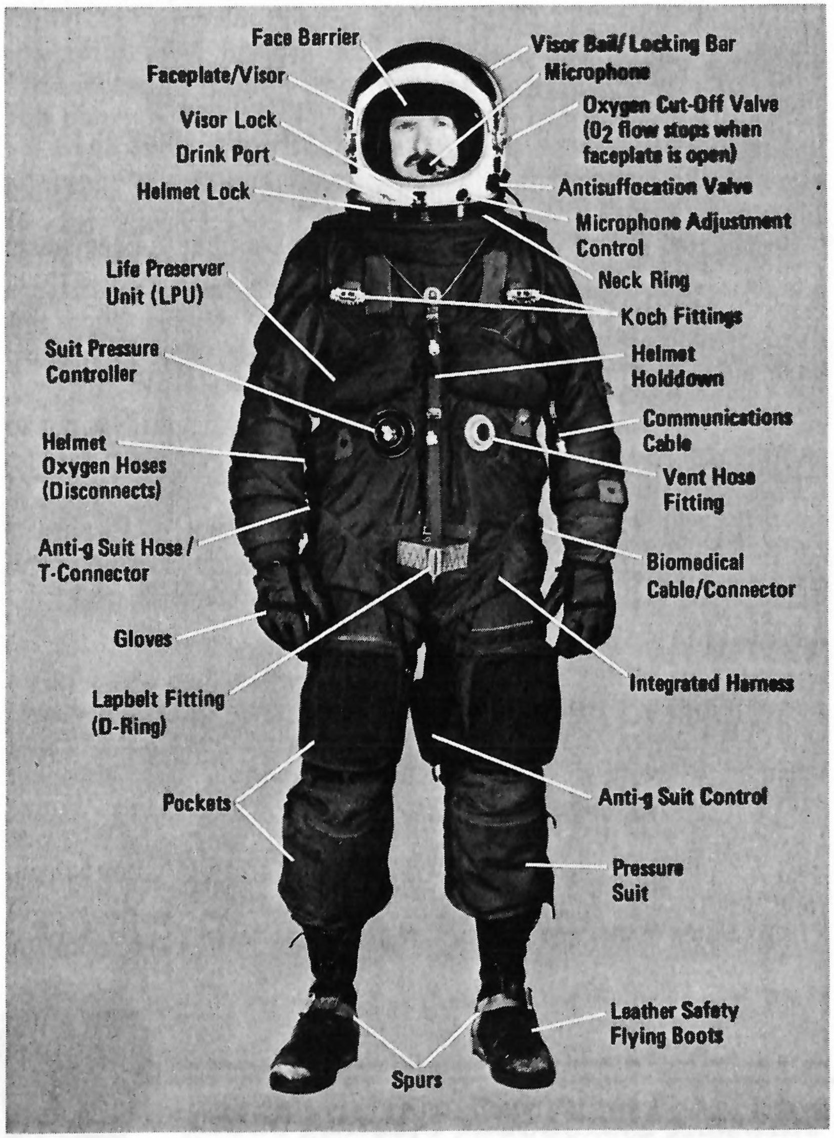
The EES face area is separated from the rest of the suit by a barrier that seals around the face. The barrier exhalation valve exhales gas into the torso area. From the torso area, the air and exhaled gas exit through a chest-mounted pressure controller. When the suit ambient pressure lowers to 165 to 175 mmHg (3.2 to 3.4 psi), the controller closes to maintain suit pressure at this level.

The EES also has a communication system.

ANTIGRAVITY SUIT

The antigravity suit (AGS) is a separate unit but it cannot be operated without the EES in the development flight tests.

The AGS prevents pooling of body fluids and aids in maintaining circulating blood volume. Pooling of body fluids can occur when high "g" loads are imposed on the body. It is particularly noticeable after crew members have had more than three days of zero "g" activity.



Ejection Escape Suit

For the development flight tests, bladders in the AGS receive oxygen from the power reactant storage distribution cryogenic system, ECLSS ARS emergency oxygen system, or the ejection seat emergency oxygen supply. When the bladders of the AGS are activated, pressure is applied to the crew member's lower extremities and to the abdomen to prevent the pooling of body fluids.

The AGS weighs 2.2 kilograms (5 pounds).

The AGS will be worn for entry in the operational flights, and the configuration will be supplied at a later date.

The various parameters of the ECLSS are monitored and are utilized for fault messages, display on the flight deck crew display and control panel, and telemetry.

Contractors involved with the ECLSS are Hamilton Standard Division of United Technologies Corp., Windsor Locks, CT (atmospheric revitalization, Freon-21 coolant loops, heat exchangers, cabin fan assembly, debris trap, CO_2 absorber, humidity control heat exchanger, avionics fan, accumulators, flash evaporators, water management panel EVA life support system and EMU's); Carlton Controls, East Aurora, NY (atmospheric revitalization pressure control subsystem and airlock support components); Aerodyne Controls Corp., Farmingdale, NY (water pressure relief valve, oxygen check valve); Aeroquip Corp., Marman, Los Angeles, CA (couplings, clamps, retaining straps, and flexible air duct); AiResearch Manufacturing Co. Garrett Corp., Torrance, CA (ground coolant unit); Anemostat Products, Scranton, PA (cabin air diffuser); Arrowhead Products Division of Federal Mogul, Los Alamitos, CA (couplings, flex air duct, flexible connector, connector drain system convoluted bellows); Brunswick, Lincoln, NE (atmospheric revitalization oxygen, nitrogen tanks); Brunswick, Circle Seal, Anaheim CA (water relief valve, water check valve); Brunswick Wintec, El

Segundo, CA (water relief valve, water check valve, water filter); Consolidated Controls, El Segundo, CA (unidirectional/bidirectional shutoff valve, water solenoid latching valve); Cox and Co., New York, NY (water relief valve, vent nozzle and port heater, water boiler steam vent line heater); Dynamic Corp. Scranton, PA (cabin diffuser); Fairchild Stratos, Manhattan Beach, CA (ammonia boiler); General Electric, Valley Forge, PA (waste collector); Metal Bellows Co., Chatsworth, CA (potable and waste water tanks, flex metal tubes); RDF Corp., Hudson, NH (temperature sensor/transducer); Symetrics, Canoga Park, CA (Freon fluid disconnects, water boiler quick disconnects); Seaton Wilson, Inc., subsidiary of Systron-Donner, Burbank, CA (water and coolant system quick disconnects); Tavis Corp., Mariposa, CA (Freon flow meter); Tayco Engineering, Long Beach, CA (urine, waste water, O_2, N_2 waste dump); Titeflex Division, Springfield, MA (water coolant flex line); Vacco Industries, El Monte, CA (potable water inline pressure relief valve); Vought Corp., Dallas, TX (radiators and flow control assembly).

AUXILIARY POWER UNIT

The auxiliary power unit (APU) is a hydrazine-fueled turbine-driven power unit and generates the mechanical shaft power to a pump that produces pressure for the orbiter's hydraulic system. There are three separate APU's, three hydraulic pumps, and three hydraulic systems, all located in the aft section of the spacecraft.

The APU's and their fuel systems are isolated from each other. The three independent hydraulic systems are connected to the main engine thrust vector control (TVC) switching valve actuators and the aerosurface actuator switching valves. Thus a single system failure will not affect full operational performance. If two systems fail, the third system will provide sufficient hydraulic power to operate all actuators at a reduced rate.

The three APU's/hydraulic pumps operate during launch and boost. They provide the three main engines with propellant valve control, thrust vector control by hydraulically gimbaling the three main engines, and control of the orbiter hydraulic actuators of the aerosurface elevons for aerodynamic elevon load relief during boost.

The APU's and pumps are restarted prior to deorbit and operate continuously through landing and rollout for hydraulic positioning of orbiter aerosurfaces (elevons, rudder/speedbrake, body flap) during the atmospheric flight portion of entry and to provide hydraulics for nose and main gear deployment, nose gear wheel steering, and main landing gear braking.

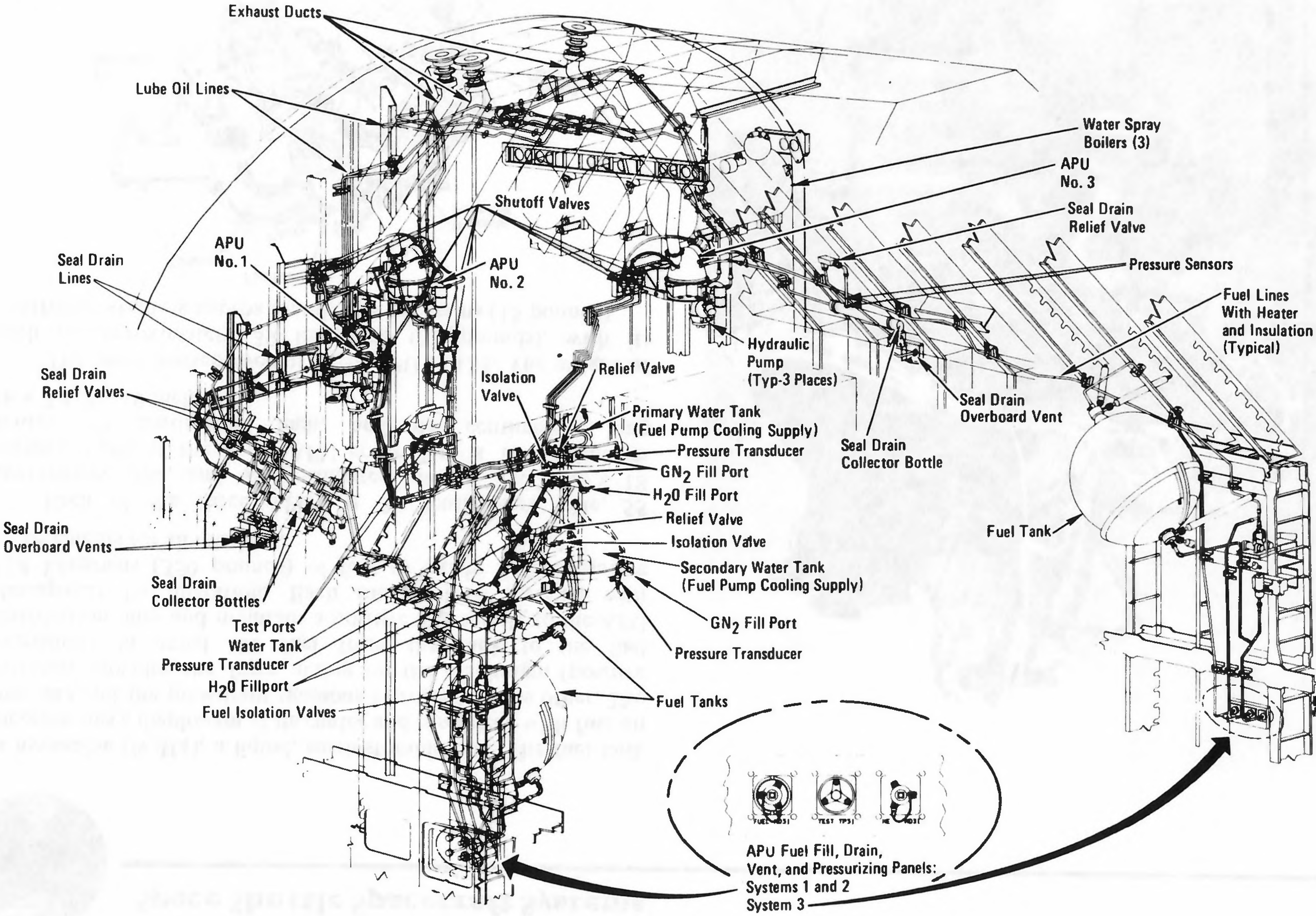
Except for on-orbit checkout, the APU/hydraulic pumps are dormant during the orbital flight phase.

The three APU's are located in the orbiter aft fuselage. Each system consists of a fuel tank, a fuel feed system, an APU and controller, an exhaust duct, lube oil cooling system, and fuel/lube oil vents and drains. Redundant electrical heater systems and insulation thermally control the system above 7°C (45°F) to prevent fuel freezing and provide required oil viscosity. The insulation is used to minimize the electrical heater size and to control high surface temperatures to safe limits on the turbine and exhaust ducts.

Each APU fuel system provides fuel to its respective fuel pump and control valves, then to the gas generator. Gas generator catalytic action decomposes the fuel and the resultant hot gas drives the APU two-stage turbine. The APU turbine assembly provides mechanical power to the APU gearbox which drives the APU fuel pump, hydraulic pump, and oil lube pump. The APU lube oil system is circulated through a lube oil heat exchanger in the APU/hydraulic water spray boiler to cool the lube oil system.

The turbine exhaust of each APU flows over the exterior of the gas generator exterior, cooling it, and then is directed overboard through an exhaust duct at the upper portion of the aft fuselage near the vertical stabilizer.

The APU fuel tanks are mounted on supports cantilevered from the sides of the internal portion of the aft fuselage. The fuel

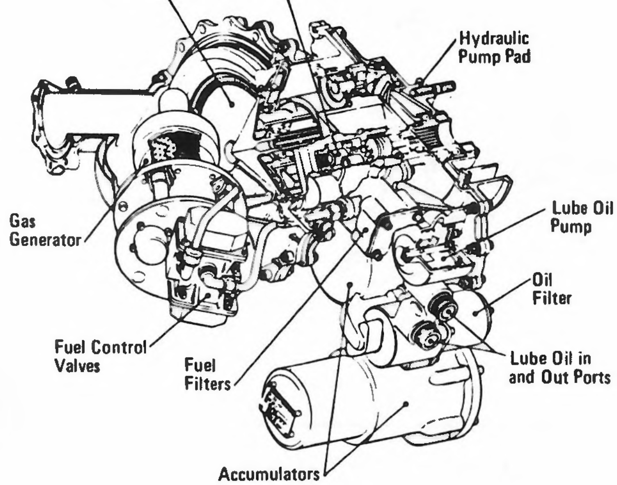


Auxiliary Power Unit/Water Boilers

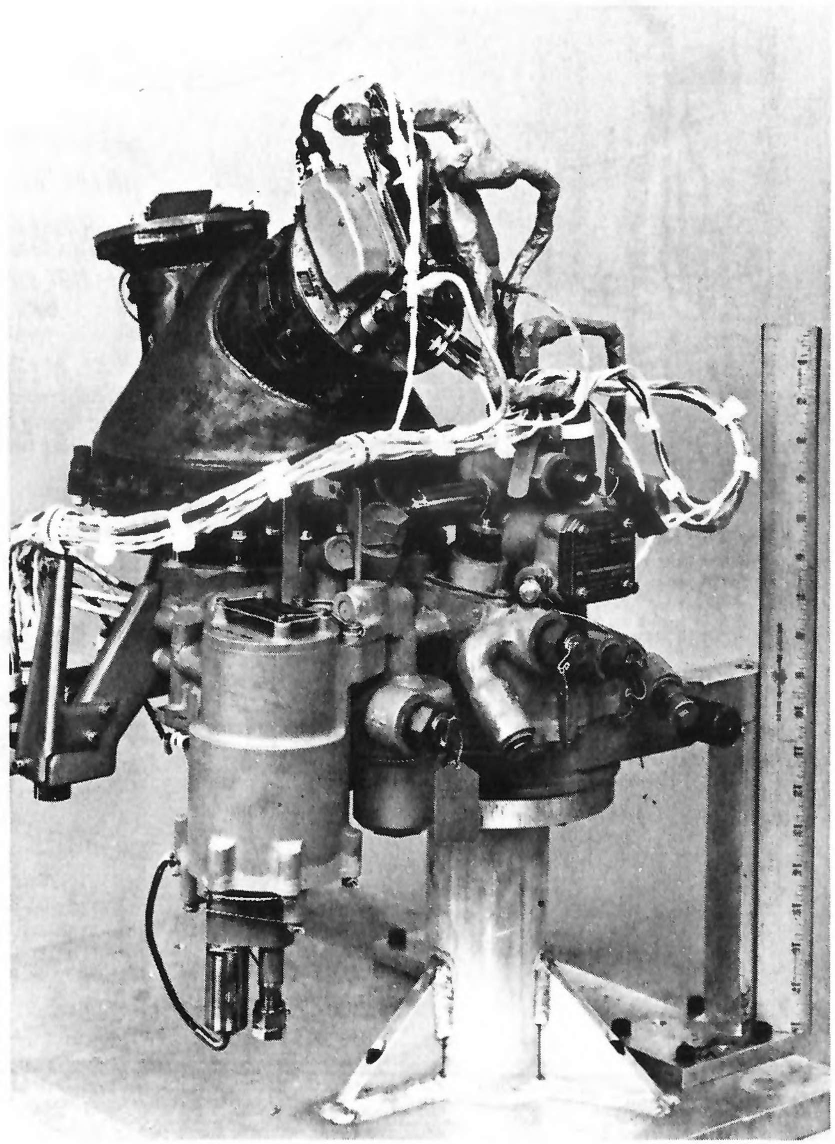
is hydrazine (N_2H_4), a liquid, earth-storable fuel. The fuel tank incorporates a diaphragm at its center and is serviced with fuel on one side and the pressurant (gaseous nitrogen) on the other. The nitrogen provides the force acting on the diaphragm (positive expulsion) to expel the fuel from the tank to the fuel distribution lines and maintain a positive fuel supply to the APU throughout its operation. Each fuel tank is serviced with 158 kilograms (350 pounds) of fuel to satisfy APU operating requirements for all missions.

Each of the three APU's is 50 centimeters wide, 55 centimeters high, and 45 centimeters deep (20 x 21.80 x 18 inches). Each of the three APU controllers is 15 centimeters wide, 19 centimeters high, and 48 centimeters long (6 x 7-1/2 x 19 inches).

The rated horsepower of each APU is 135. The weight of each is approximately 39 kilograms (88 pounds), with its controller weighing approximately 6.8 kilograms (15 pounds).



Auxiliary Power Unit



Auxiliary Power Unit

The fuel tanks are 71-centimeter (28-inch) diameter spheres. Fuel tanks No. 1 and No. 2 are located on the left (minus Y) side of the orbiter's aft fuselage, and tank No. 3 is located on the right (plus Y) side. The fuel tanks are serviced through fill and drain couplings bolted to the corresponding APU servicing panel which is located on the side of the fuselage. The nitrogen service connections are located on the same panel. The initial gaseous nitrogen pressure is 17,077 mmHg (millimeters of mercury) (330 psi).

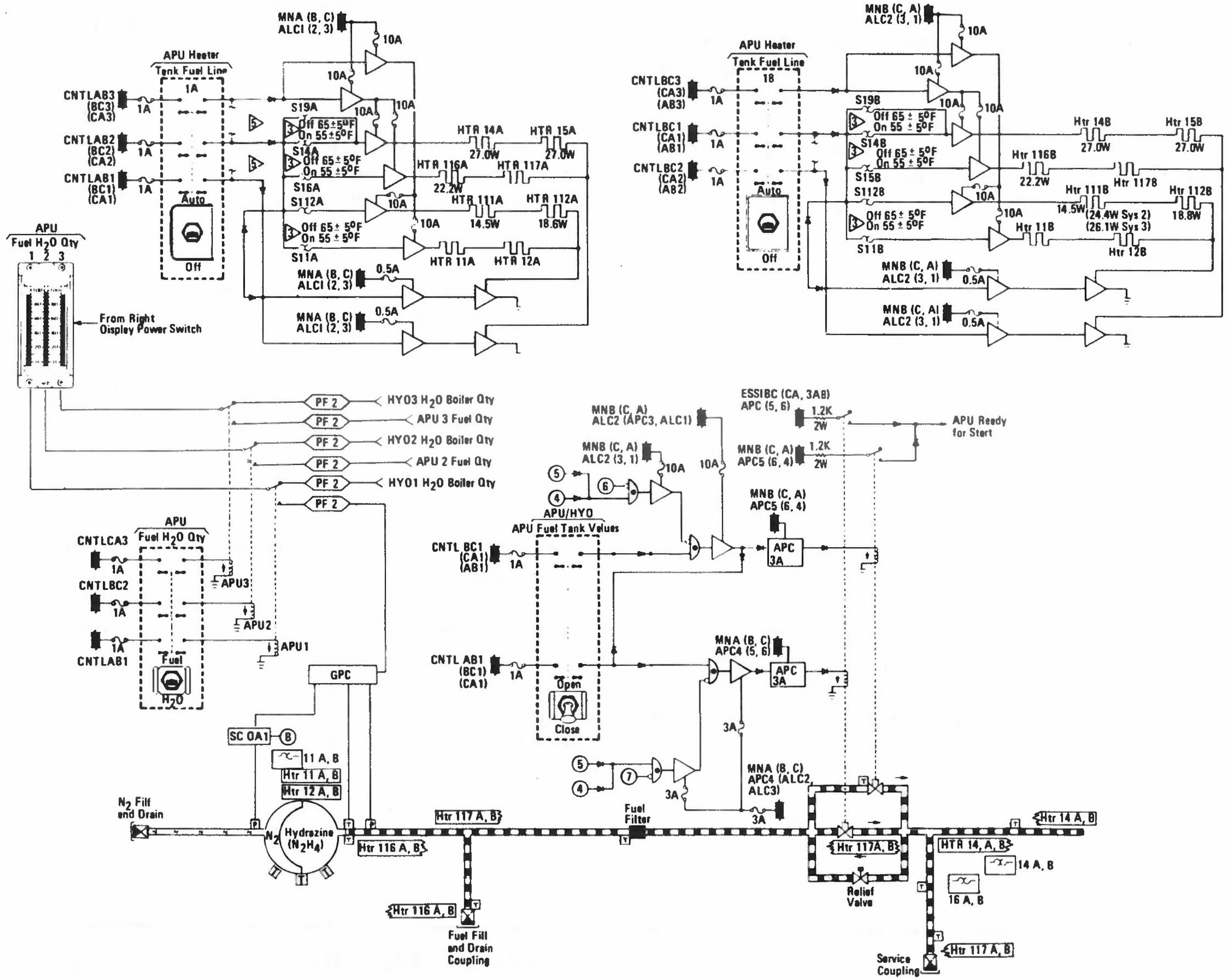
All APU system controls and displays are located on flight deck panels. The temperature and nitrogen pressure in each fuel tank is monitored and processed through the orbiter's system management general-purpose computer (GPC) and transmitted to the APU FUEL/H_2O QTY display on Panel F8. If the display select switch is in the "Fuel" position, the quantities in APU fuel tanks 1, 2, and 3 are displayed simultaneously on the display meter in percent (%). (When switch and display nomenclature is printed in all caps–e.g., APU FUEL/H_2O QTY–it indicates that it is the exact way it appears on the display and control panel.)

The fuel distribution system supplies the fuel from the fuel tank to the APU. Filters are incorporated in the distribution line to remove particles before the fuel arrives at the APU. The fuel distribution line branches into two parallel paths downstream of the fuel tank and filter. An isolation valve is installed in each of these paths, which converge into a single path immediately downstream of the valves. These valves are installed to isolate the fuel tank supply from the APU. The valves are controlled by the applicable system APU FUEL TK VLV switch located on Panel R2. The fuel tank isolation valves are open for APU operation and closed when the APU is not in operation. The valve internal design contains a reverse relief feature to allow relief of pressure trapped in the line when both APU fuel tank valves are closed. The relief feature will function if the downstream pressure increases to a pressure range of 20,700 to 25,875 mmHg (400 to 500 psi).

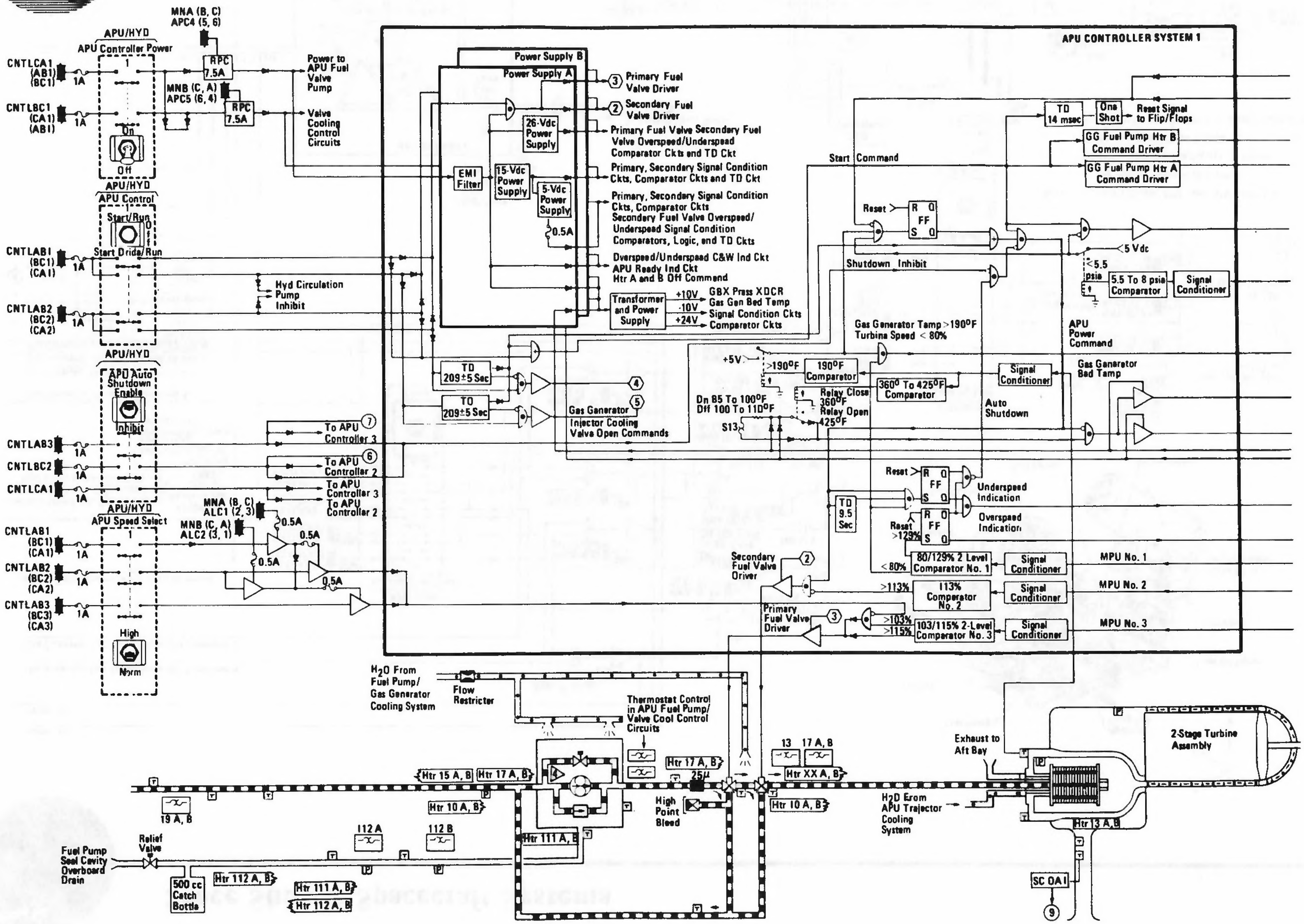
The fuel/nitrogen fill, drain/vent, and test point couplings permit servicing and ground checkout of the fuel distribution system. The lube oil system couplings permit servicing and checkout of the lube oil system.

The APU CONTROL POWER switch for each APU also is located on Panel R2. The switch applies or removes 28 Vdc (volts direct current) power to individual APU controllers. Each APU controller provides the checkout logic prior to starting the APU, detects malfunctions, and controls the APU turbine speed, gearbox pressurization, and fuel pump/gas generator heaters when the APU is not operating. In the OFF position, power is removed from the controller.

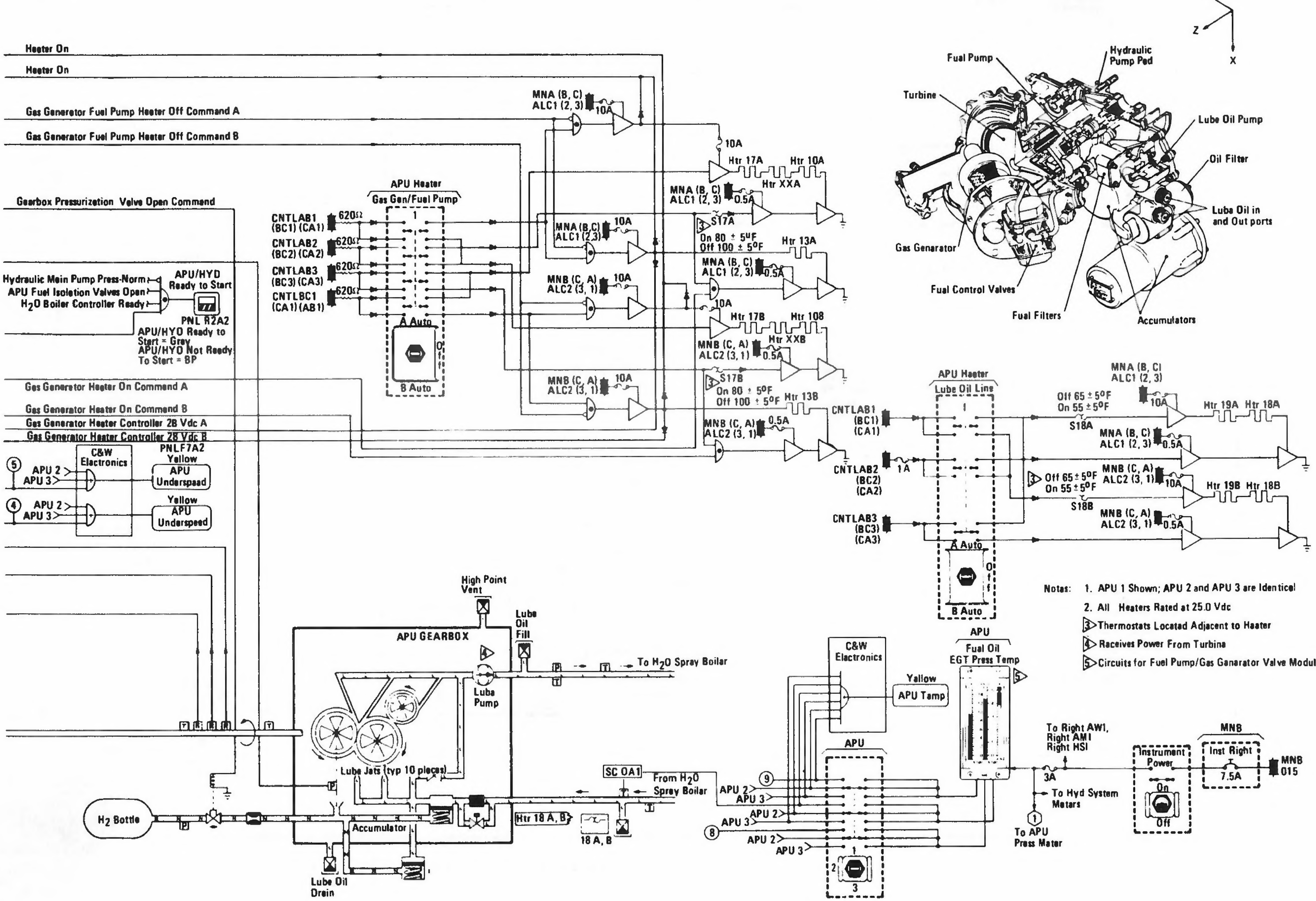
The APU HEATER GAS GEN/FUEL PUMP switch for each APU is located on Panel A12. When the switch is positioned to AUTO A or AUTO B, electrical power is provided to the fuel



APU Heater Gas Gen/Fuel Pump



APU Controller System 1



APU Heater Gas Generator/Fuel Pump

pump and gas generator heaters for each APU. The fuel pump temperature is controlled automatically by thermostats which prevent the fuel from freezing. The thermostats are set to cycle on/off between 26 and 38°C (80-100°F). The gas generator heaters are on when the switch is in AUTO A or AUTO B prior to the start of the APU. When the APU is initiated, the APU controller automatically turns off both gas generator and fuel pump heaters. The OFF position removes power from heater circuits.

The APU/HYD READY TO START indicator for each APU is located on Panel R2. The indicator shows gray when that APU is ready to start; that is, when the APU gas generator temperature is above 87°C (190°F), turbine speed less than 80%, APU gearbox pressure above 284 mmHg (5.5 psi), H_2O boiler controller ready, APU fuel isolation valves open, and hydraulic main pump depressurized. When the APU is initiated, and the turbine speed is greater than 80 percent of normal speed, the indicator will show a barberpole pattern.

The APU CONTROL START/RUN switch for each APU is located on Panel R2. When the switch is in the START/RUN position, the APU controller is activated, and the fuel pump and gas generator heaters are turned off. The APU is then started by the APU controller. The OFF position removes the start signal from the APU controller.

To start, the APU controller opens the normally closed secondary valve shutoff of the gas generator valve module and fuel flows through the fuel pump bootstrap start bypass circuit (the primary fuel control valve of the gas generator valve module was already in the open position) to the APU gas generator. Gas generator catalytic action decomposes the fuel, creates a hot gas, and feeds the hot gas exhaust product to the APU turbine. The APU two-stage turbine assembly provides mechanical power to the APU gearbox and drives the APU fuel pump. The APU turbine must come up to speed in 9.5 seconds or the APU will automatically shut down. Because of the gearbox driving the APU fuel pump, the pump increased the fuel pressure at its outlet and sustains pressurized fuel to the gas generator valve module and gas generator.

The APU SPEED SELECT switch for each APU (Panel R2) selects the speed at which each APU controller operates its APU. The NORM position controls APU speed at 74,160 rpm, 103 plus or minus 8%. The HIGH position controls the APU speed at 81,360 rpm, 113 plus or minus 8%, with a second backup at 82,800 rpm, 115 plus or minus 8%.

When the upper APU turbine speed is reached, the normally open port of the primary fuel control valve of the gas generator valve module assembly closes, and the normally closed port of the control valve opens, allowing the fuel to bypass the fuel pump inlet. A relief valve provides pressure relief for the APU fuel pump outlet pressure. The APU fuel pump operates at a nominal speed of 3,918 rpm and provides a nominal pressure of 73,485 mmHg (1,420 psi).

The APU speed is controlled in the following manner. When the lower speed limit is reached, the primary fuel control valve opens, allowing fuel to the gas generator; it closes when the upper speed limit is reached, shutting off fuel to the gas generator.

The frequency and duration of primary fuel control valve cycling is a function of the hydraulic load on the APU as well as hydraulic system dynamics.

The secondary shutoff valve of the gas generator valve module assembly provides a backup in the event of a malfunction of the primary control valve.

The APU OVERSPEED yellow caution and warning light, on Panel F7, will illuminate if APU 1, 2, or 3 turbine speed is above 92,880 rpm, 129 plus or minus 1%. The APU UNDER-SPEED yellow light will illuminate if the turbine speed is less than 57,600 rpm, 80 plus or minus 3%, normal speed.

In the event of an APU overspeed or underspeed shutdown, the HYD PRESS yellow light on Panel F7 also would illuminate for the corresponding hydraulic system, since the APU driving the hydraulic pump for the system would be off; the yellow HYD PRESS light illuminates when any hydraulic system drops below 144,900 mmHg (2,800 psi).

The INHIBIT position of the APU AUTO SHUTDOWN switch on Panel R2 bypasses the automatic shutdown sequence from the APU controllers and the 9.5-second speed time delay if the APU CONTROL switch is in the START/RUN position.

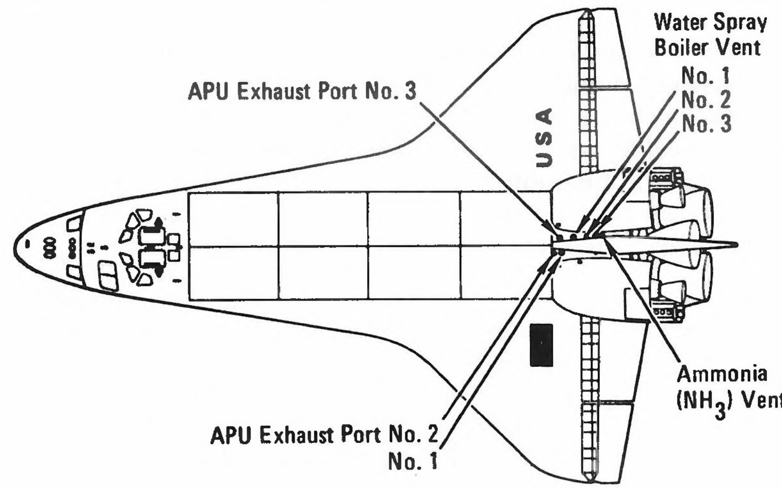
The START ORIDE/RUN position of each APU CONTROL switch on Panel R2 will override the APU pre-start conditions (gas generator temperature above 87°C, turbine speed less than 80%, and gearbox pressure above 284 mmHg [5.5 psi]) to permit a start of the respective APU, if one or more of the prestart conditions are not met. This switch also activates the APU gas generator active cooling system which provides the capability to restart an APU. The restart is inhibited for 209 seconds after the switch is positioned, during which time the gas generator is cooled by flowing water through its cooling passages.

The APU turbine provides the mechanical drive to the APU two-stage gearbox to drive the hydraulic pump at a nominal speed of 3,918 rpm. In addition to driving the hydraulic pump, the output shaft drives the APU fuel pump and the lube oil pump.

Each APU oil lube system uses the power train gears as scavenge pumps to supply lube oil to the inlet of the lube pump. The lube oil pump increases the lube pressure to a nominal 3,984 mmHg (77 psi), directs the lube oil through the APU/hydraulic water spray boiler to cool it and returns the lube oil to the APU gearbox accumulators.

The accumulators allow thermal expansion of the lube oil, accommodate gas initially trapped in the external lube circuit, regulate lube pressure, and act as a zero-gravity, all-attitude lube reservoir. The nitrogen pressurization system for each gearbox is activated when the gearbox case pressure is below 232 mmHg (4.5 psi) plus or minus 77 mmHg (1.5 psi) by the corresponding APU controller, assuring that gearcase pressure is sufficiently above the requirements for proper scavenging and lube pump operation.

Each APU turbine exhaust flows over the exterior of the gas generator, cooling it, and then is vented overboard through its own independent exhaust duct at the upper portion of the aft fuselage near the vertical stabilizer. Overtemperature detection of each APU exhaust is monitored on display and control Panel F8.



APU Exhaust and Water Boiler Vent

Each APU turbine exhaust gas temperature, lube oil temperature, and fuel pressure is routed to Panel F8. The APU SELECT switch position 1, 2, or 3 permits the respective APU exhaust gas temperature, fuel pressure, and lube oil temperature to be displayed on the APU EGT (exhaust gas temperature), FUEL PRESS, and OIL TEMP meter.

The APU TEMP yellow caution and warning light on Panel F7 will illuminate if APU 1, 2, or 3 exhaust gas temperature is above 682°C (1,260°F) or if APU 1, 2, or 3 lube oil temperature is above 143°C (290°F).

The APU HEATER TANK FUEL LINE switches on Panel A12 for each APU provide the operation of the thermostatic controlled heaters located on the respective APU fuel tank fuel lines. The thermostats maintain the temperature between a nominal 12 and 18°C (55 to 65°F).

The heaters for the gas generator and the fuel pump and gas generator valve module water system are also controlled by the APU HEATER TANK FUEL LINE switches on Panel A12. Thermostats for the water systems maintain the temperature between 26 and 35°C (80 to 90°F).

The APU HEATER TANK FUEL LINE switches for each APU are divided into an A and B system switch for each APU. The A switch controls the A heaters and the thermostats provide automatic control. The B switch controls the B heaters and the thermostats provide automatic control. The OFF position of each switch removes power from the respective heater circuits.

The APU lube oil lines on each APU have a heater system. These heaters are controlled by a heater switch for each APU on Panel A12. When the HEATERS LUBE OIL LINE switch on APU is in the A AUTO position, the A lube oil heaters are powered and controlled automatically by thermostats between 12 and 18°C (55-65°F). The B AUTO heater switch position provides the same capability to the B heater system. The OFF position of each switch removes power from the heater circuits.

The APU fuel pump and gas generator valve module is cooled by water spray following APU shutdown on completion of the ascent boost phase and after orbital checkout. The water spray cooling prevents hydrazine decomposition in the APU fuel pump and valve due to heat soakback in each APU. The cooling system consists of a primary (A) and secondary (B) independent water supply to the three APU's. Each water system consists of a 41-centimeter (16.5-inch) diameter tank, a 0.6-centimeter (0.25-inch) diameter line to each APU, heaters, and control valves. Each water tank is loaded with 9.5 plus or minus 0.2 kilograms (21 plus or minus one pound) of water. Each tank is pressurized with nitrogen between 2,587 and 3,053 mmHg (50 to 59 psi). The pressure acts on a diaphragm in each tank to expel the water into the lines to the control valves.

When the APU's are shut down, the APU FUEL PUMP/VLV COOL switch A or B on Panel R2 is positioned to AUTO. With the A switch on AUTO, the 71-76°C (160 to 170°F) thermostats on each APU, through the timer in the water controller for each APU, open control Valve A on each APU to permit water to spray onto the valve module and fuel pump for 1.25 seconds, then close Valve A for 4 seconds, etc. The cooling system is activated for two hours and 45 minutes after APU shutdown. The B switch controls Valve B in the same manner as in the A case. Nitrogen pressurization in each water tank is referred to as a blowdown system (pressure decay continues until the water is expelled from each tank). The water is exhausted into the aft fuselage compartment.

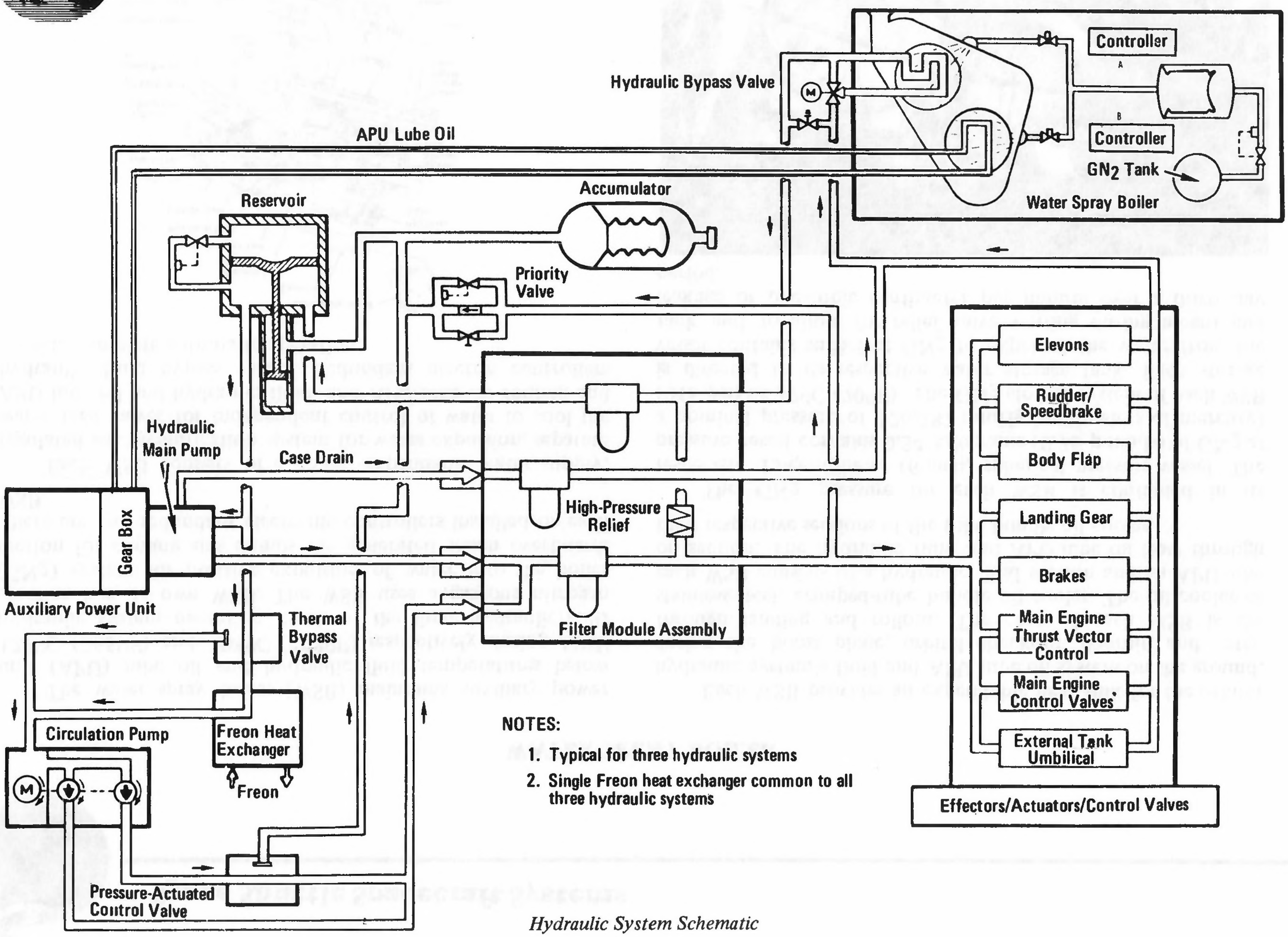
The gas generator active cooling is used only where the normal cooldown time of 180 minutes is not available. This provides cooling of the gas generator injector in each APU. The injector is cooled by circulation of water through the injector. One water tank services all three APU's. The system consists of the one water tank and 0.14-centimeter (1.18-inch) diameter line to each APU, heaters, and a control valve. The water tank is a 23-centimeter (9.4-inch) diameter tank loaded with 2.72 plus or minus 0.2 kilograms (6 plus or minus 0.5 pounds) of water. The tank is pressurized with nitrogen at a nominal pressure of 4,398 mmHg (85 psi) to expel the water into the lines to the control valve.

When the APU CONTROL switch on Panel R2 for each APU is positioned to START ORIDE and the temperature sensor for each APU injector is above 212°C (415°F), each APU controller opens its water control valve for 209 plus or minus 5 seconds and directs water into the gas generator injector to cool it. When the timer in the APU controller has timed out, the control valve is closed. It should be noted that the APU CONTROL switch on panel R2 for each APU will not activate the water cooling system in the START position. The water from the gas generator is exhausted into the aft fuselage.

The APU's are designed for 50 hours of maintenance-free operation. The current life of the gas generator in each APU is 20 hours, however, because of degradation of the catalyst; thus, the gas generator life is 60 hours with two refurbishments of the catalyst bed.

The various parameters of the APU's are monitored and utilized for display on the flight deck crew display and control panel and transmitted to telemetry and fault messages.

The contractors involved with the APU system are Sundstrand Corp., Rockford, IL (APU and APU controller); Consolidated Controls, El Segundo, CA (APU fuel isolation valve); Pressure Systems Inc., Los Angeles, CA (APU fuel tank); SSP Products, Inc., Burbank, CA (APU exhaust duct assembly); Sundstrand Data Control, Redmond, WA (APU heater thermostat); Cox and Co., New York, NY (APU fuel tank, fuel and lube line heaters); Brunswick Wintec, El Segundo, CA (APU fuel line filter); J.C. Carter Co., Costa Mesa, CA (APU servicing coupling); Wright Components Inc., Clifton Springs, NJ (fuel pump seal cavity drain catch, relief valve); Rocket Research Corp., Redmond, WA (APU gas generator).

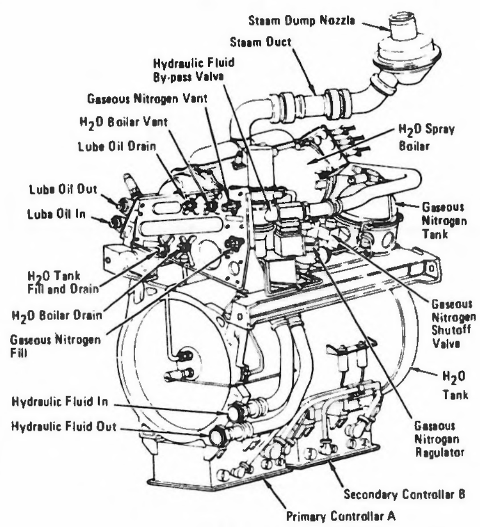


Hydraulic System Schematic

WATER SPRAY BOILER

The water spray boiler (WSB) maintains auxiliary power unit (APU) lube oil and hydraulic fluid temperatures below 129°C (265°F) and 108°C (228°F) respectively during APU/hydraulic system operation. Each of the three hydraulic/APU systems has its own WSB. The WSB uses a gaseous nitrogen (GN_2) system for positive expulsion of water into the boiler section for cooling and dumps the generated steam overboard. There are two redundant electronic controllers installed on each WSB.

Each WSB consists of a boiler, expendable water supply, regulated gas pressurization system for water expulsion, separate water feed valves for independent control of water to cool the APU lube oil and hydraulic fluid, and APU lube oil cooling and hydraulic fluid bypass valves. Redundant electric controllers provide complete automatic operation.

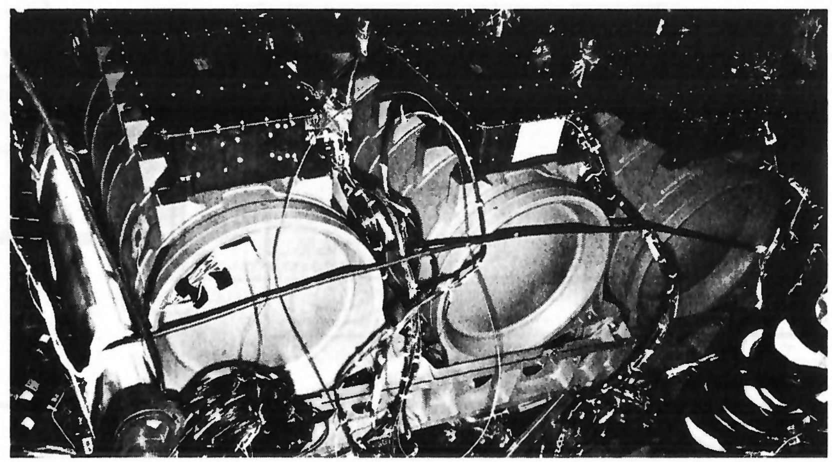


Water Spray Boiler (One of Three)

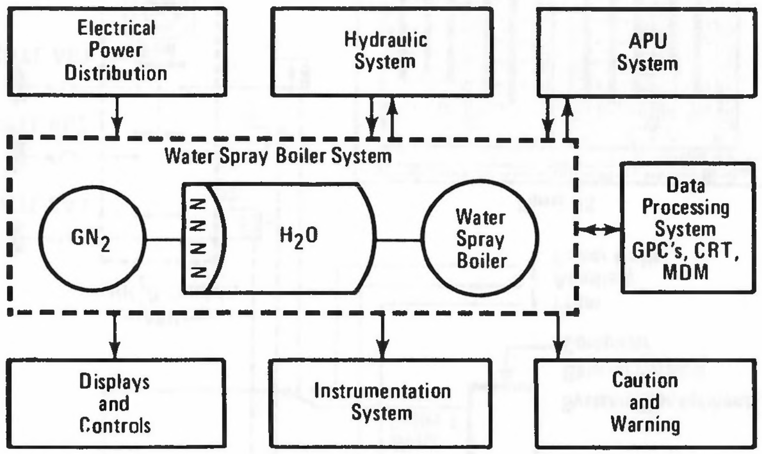
Water Spray Unit

Each WSB provides an expendable heat sink for the orbiter hydraulic system's fluid and APU lube oil system on the ground, during the boost phase, orbital checkout, deorbit, and entry through landing and rollout. The core of each WSB is the stainless-steel, crimped-tube bundle oil cooler. The oil cooler of each WSB consists of a hydraulic fluid section and an APU lube oil section. The hydraulic fluid and APU lube oil flow through their respective sections of the tube bundle oil cooler.

The GN_2 pressure for each WSB is contained in its respective 15-centimeter (6-inch) spherical pressure vessel. The pressure vessel contains 0.34 kilograms (0.76 pounds) of GN_2 at a nominal pressure of 126,787 mmHg (millimeters of mercury) (245 psi) at 37°C (70°F). The GN_2 storage system of each WSB is directed to its respective water storage tank. Each storage vessel contains sufficient GN_2 to expel all the water from the tank and to allow for relief valve venting during ascent and leakage of one cubic centimeter per minute over a thirty-day period.



Water Spray Boilers in Aft Fuselage



Water Spray Boiler System

The GN_2 shutoff valve between the GN_2 pressure vessel and H_2O storage tank of each WSB permits isolation of high pressure GN_2 supply. Each GN_2 valve is controlled by its respective boiler N_2 supply 1, 2, or 3 switch on the flight deck crew display and control panel R2. The GN_2 shutoff valve is latched in the open or close position and consists of two independent solenoid coils which allow valve control from either the primary or secondary controller. The respective switch is positioned to control each valve.

A single-stage regulator is installed between the GN_2 pressure shutoff valve and the H_2O storage tank. The GN_2 regulator for each WSB regulates the high pressure GN_2 between 1,267 and 1,345 mmHg (24.5 to 26 psi) to the H_2O storage tank.

A relief valve is incorporated internally to each GN_2 regulator to prevent the H_2O storage tank pressure from exceeding 1,707 mmHg (33 psi) due to heat soak-back when operating or in the event of a failed open GN_2 regulator. The GN_2 relief valve opens between 1,552 and 1,707 mmHg (30 to 33 psi).

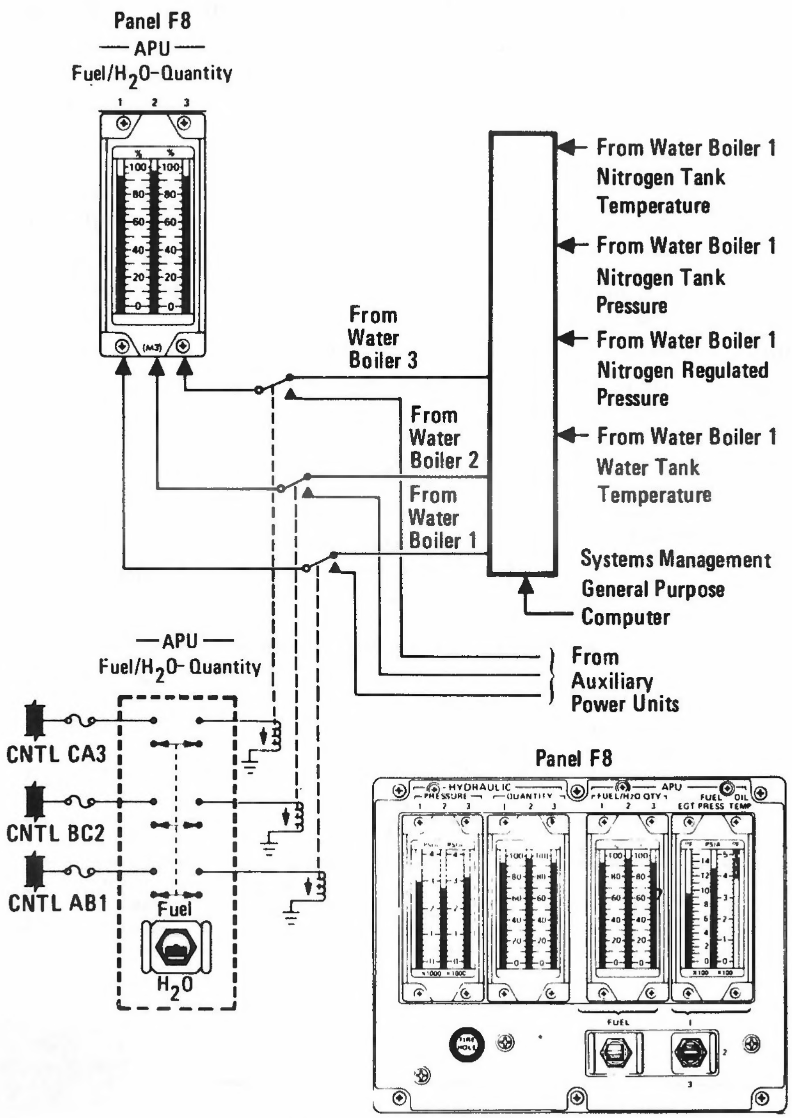
The H_2O supply for each WSB is stored in a positive displacement aluminum water tank containing a welded metal bellows separating the stored water inside the bellows from the GN_2 expulsion gas. The 63 kilograms (140 pounds) of H_2O in each WSB provide total heat rejection.

Non-redundant pressure and temperature sensors located downstream of the GN_2 pressure vessel and on the H_2O tank for each WSB transmit the pressures and temperatures via the A controller to the Systems Management General Purpose Computer (SM-GPC). The SM-GPC computes the pressure/volume/temperature and transmits the H_2O tank quantity to the flight deck crew display and control panel F8 for each WSB. The APU FUEL/H_2O QTY switch on panel F8 is positioned to H_2O which allows the H_2O quantity of each WSB to be displayed on the APU FUEL/H_2O QTY 1, 2, or 3 meter. Thus H_2O quantity is only available when the A controller is powered.

Downstream of the H_2O storage tank, the feedwater lines to each water boiler split into two parallel lines: one line goes to the hydraulic fluid flow section, one to the APU lube oil section. A hydraulic fluid water feed valve is installed in the water line to the hydraulic fluid section and an APU lube oil water feed valve is installed in the water line to the APU lube oil section. Each valve is controlled independently by the WSB controller.

The two WSB controllers are regulated by the respective BOILER CNTLR PWR/HTR switches 1, 2, and 3 on the flight deck crew display and control panel R2. When the applicable switch is positioned to A, the A controller for that WSB is powered; or, if positioned to B, the B controller is powered. The OFF position of the applicable switch removes electrical power from both controllers.

The BOILER CNTLR switches 1, 2, and 3 enable (provide the automatic control functions) the specific controller A or B which was selected for that WSB by the BOILER CNTLR PWR/HTR switch on panel R2. When the applicable controller A or B is enabled for that WSB, a ready signal is transmitted to the

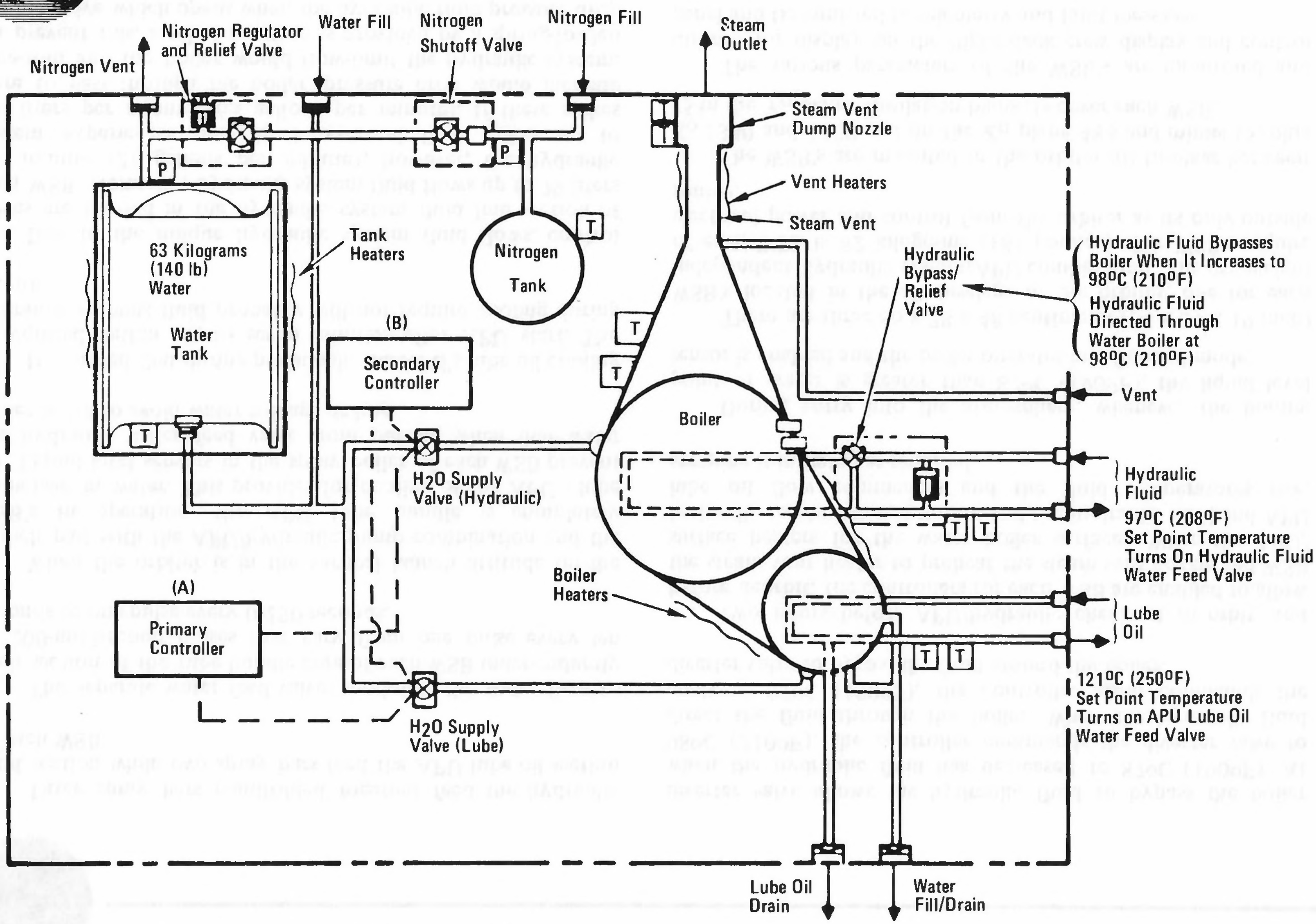


APU and Hydraulic Indicator

corresponding APU/HYD READY TO START talkback indicator (along with other prerequisites from the APU and hydraulic system) on the flight deck crew display and control panel R2, if the following additional conditions are met: GN_2 shutoff valve is open, steam vent nozzle temperature is greater than 54°C (130°F), and the hydraulic fluid bypass valve is in the correct position.

The enabled controller of that WSB monitors the hydraulic fluid and APU lube oil outlet temperature. The hydraulic fluid outlet temperature controls the hydraulic fluid water feed valve and the APU lube oil outlet temperature controls the APU lube oil water feed valve. Signals are generated based on a comparison of the hydraulic system fluid temperature to its 97°C (208°F) set point and of the APU lube oil to its 121°C (250°oF) set point. As the respective feed water valve opens, instantaneous flows of 6.8 kilograms per minute (15 pounds per minute) maximum through the hydraulic section and 4.5 kilograms (10 pounds) per minute maximum through the APU lube oil section enter the water boiler through these respective spray bars to effect evaporative cooling of the hydraulic fluid and APU lube oil with the steam vented out through the overboard steam vent.

The core of each WSB is the stainless steel crimped tube bundle. The hydraulic fluid section is divided into three 43-centimeter (17-inch) long passes of crimped tubes (first pass–234 tubes, second pass–224 tubes, and third pass–214 tubes). The tubes are 3.18 millimeters (0.125 inches) in diameter, with a wall thickness of 0.25 millimeters (0.010 inch). Crimps located every six millimeters (0.24 inches) break up the internal boundary layer and promote enhanced turbulent heat transfer. The APU lube oil section is comprised of two passes with 103 crimped tubes in its first pass and 81 smooth tubes in the second pass. Although the second pass is primarily a low pressure drop return section, approximately 15 percent of the APU lube oil heat transfer occurs there.



Water Spray Boiler (One of Three)

Three spray bars manifolded together feed the hydraulic fluid section while two spray bars feed the APU lube oil section of each WSB.

The separate water feed valves modulate the water flow to each section of the tube bundle core of each WSB independently in 200-millisecond pulses that vary from one pulse every ten seconds to one pulse every 0.250 seconds.

When the orbiter is in the vertical launch attitude on the launch pad with the APU/hydraulic pump combination and the WSB's in operation, the APU tube bundle is completely immersed in water. This provides for cooling of the APU's lube oil. Liquid level sensors in the spray boiler of each WSB prevent the hydraulic water feed valve from pulsing when that water boiler is full to avoid water spillage or loss.

It is noted that during prelaunch, the APU's lube oil cooling is required within six to seven minutes after APU start. The hydraulic systems fluid probably will not require cooling during ascent.

Due to the unique hydraulic system fluid flows, control valves are located in the hydraulic system fluid line section of each WSB. Normally, hydraulic system fluid flows up to 79 liters per minute (21 gallons per minute); however, the hydraulic system experiences one- to two-second flow spikes up to 238 liters per minute (63 gallons per minute). If these spikes were to pass through the boiler, pressure drop would increase nine-fold and the boiler would flow-limit the hydraulic system. To prevent this, a relief function is provided by a spring-loaded poppet valve which opens when the hydraulic fluid pressure drop exceeds 2,484 mmHg (48 psi) and is capable of flowing 162 liters per minute (43 gallons per minute) at a differential pressure of 2,587 mmHg (50 psi) across the boiler. A temperature-controlled diverter valve allows the hydraulic fluid to bypass the boiler when the hydraulic fluid has decreased to 87°C (190°F). At 98°C (210°F), the controller commands the diverter valve to direct the fluid through the boiler. When the hydraulic fluid cools to 87°C (190°F), the controller again commands the diverter valve to bypass the fluid around the boiler.

Two hours before APU/hydraulic checkout in orbit, and before deorbit, the controllers for each WSB are enabled to allow the steam vent heater to preheat the steam vent nozzle and WSB surface heaters for the water boiler surfaces. When the APU/hydraulic combination is started and the hydraulic fluid and APU lube oil flow commences and the fluid temperatures rise, spraying is initiated as required.

During entry into the atmosphere, whenever the boiling point of water is greater than 87°C (190°F), the liquid level sensor is enabled and the boiler operates in a flooded mode.

There are three 86 x 78 x 48 centimeter (34 x 31 x 19 inch) WSB's located in the aft fuselage of the orbiter, one for each independent hydraulic system/APU combination. The dry weight of each WSB is 82 kilograms (181 pounds). The WSB's require electrical power and control from the orbiter as its only outside source.

The WSB's are mounted in the orbiter aft fuselage between X_O 1340 and 1400 and on the Z_O plane 488 and minus 15, plus 15 in the Y_O plane. Insulation blankets cover each WSB.

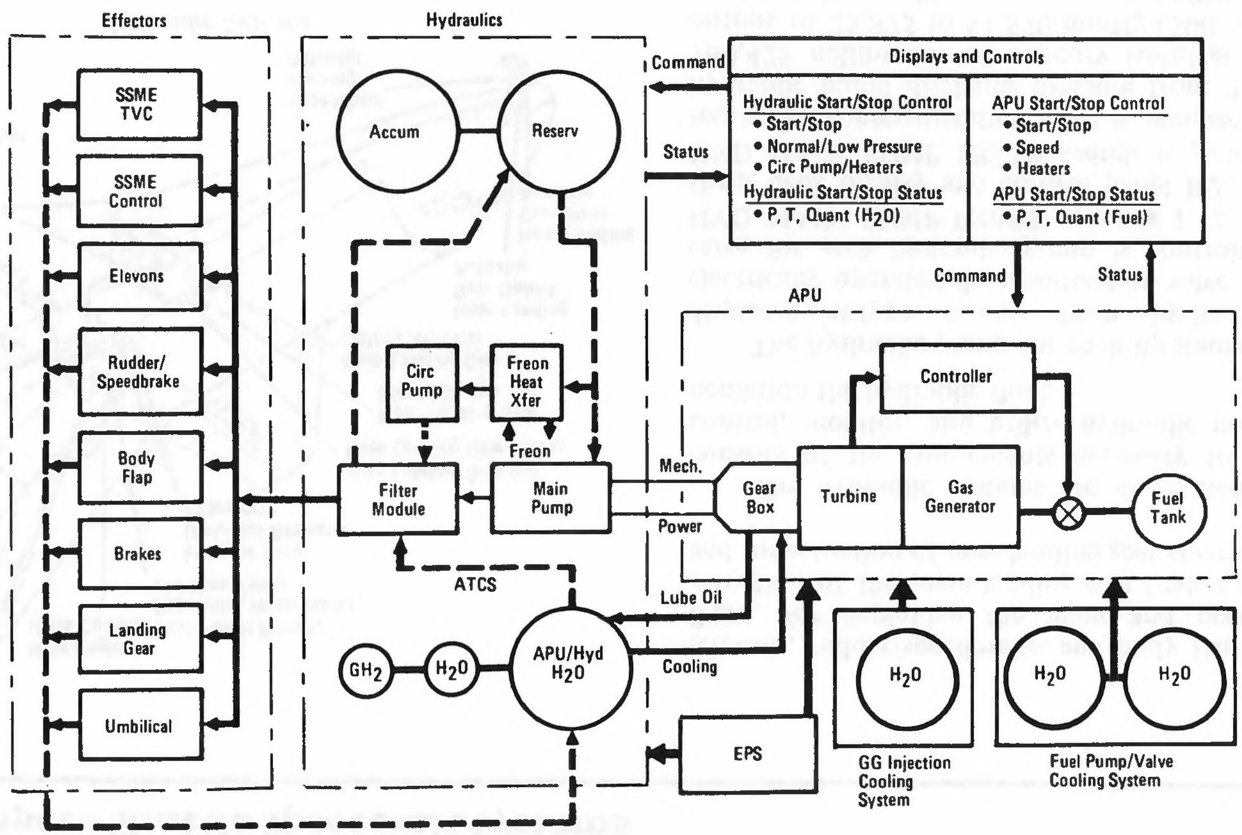
The various parameters of the WSB's are monitored and utilized for display on the flight deck crew display and control panel and transmitted to telemetry and fault messages.

The contractor for the WSB's is Hamilton Standard Division, United Technologies Inc., Windsor Locks, CT.

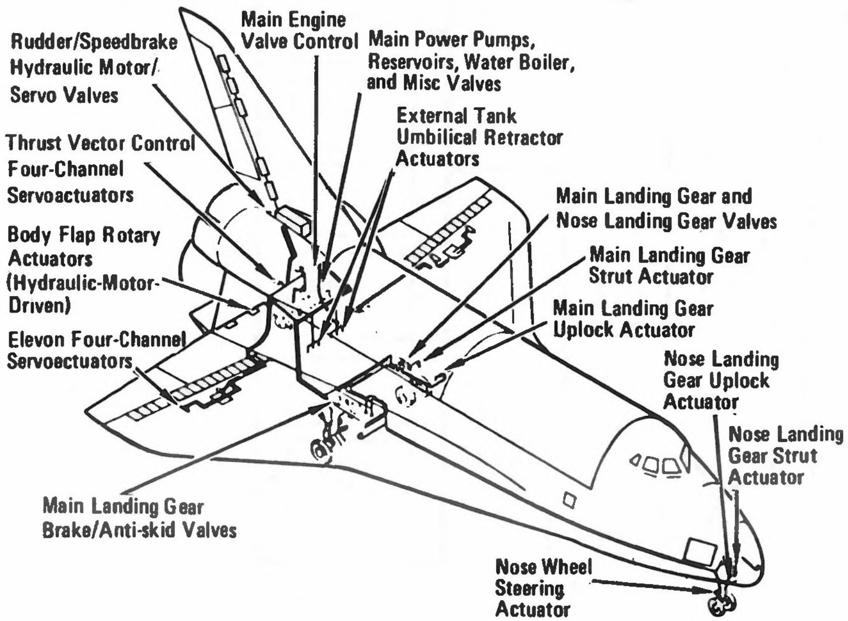
HYDRAULIC SYSTEM

The hydraulic system consists of three independent hydraulic systems. Each of the three auxiliary power units provides mechanical shaft power to drive a hydraulic pump, and each of the three hydraulic pumps provides the hydraulic pressure for the respective hydraulic system.

Each hydraulic system provides hydraulic pressure for operation of actuators to control orbiter aerosurfaces (elevons, rudder/speedbrake, and body flap), the three main engine gimbals (thrust vector control), main engine valves, external tank umbilical retraction, landing gear deployment, main landing gear brakes and anti-skid control, and nose gear steering.



APU/Hydraulic System



Hydraulic Systems

The hydraulic systems are capable of operation when exposed to forces or conditions caused by acceleration, deceleration, normal "g", zero "g", hard vacuum, and extreme low temperatures encountered in orbit. The systems are active during liftoff, ascent, and orbital insertion for main engine thrust vector control and propellant valve control, in addition to elevon load relief during ascent.

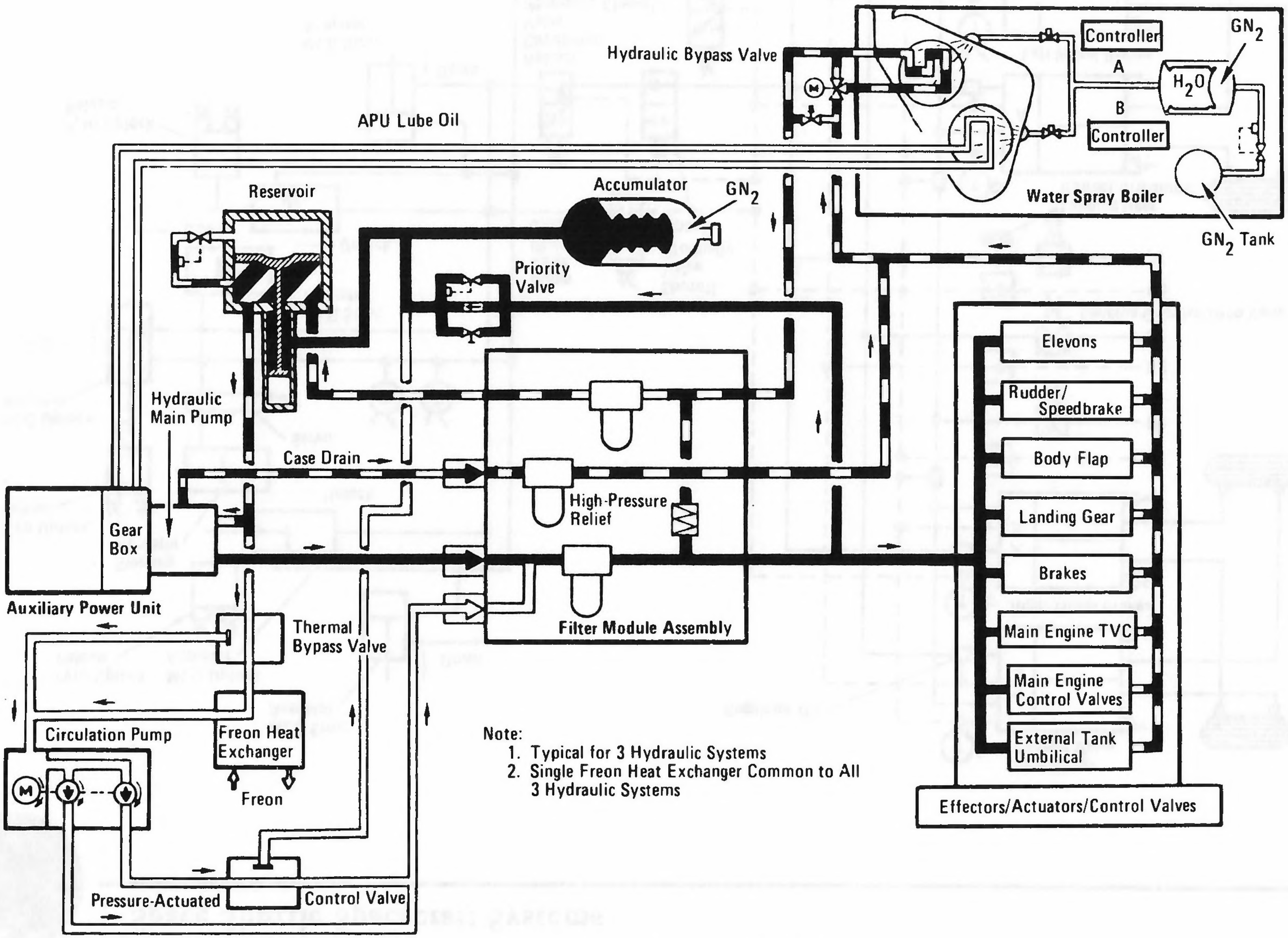
In orbit, the hydraulic system's fluids are circulated periodically by electric-motor-driven circulation pumps to absorb heat from the Freon-21 hydraulic heat exchanger and distribute it to all areas of the hydraulic systems.

The hydraulic system is restarted for deorbit, entry, and descent for actuation of the aerodynamic control surfaces (elevons, rudder/speedbrake, and body flap) during atmospheric flight, for deploying the main and nose landing gear, for actuation of the main landing gear brakes and anti-skid system, and for actuation of nose landing gear steering.

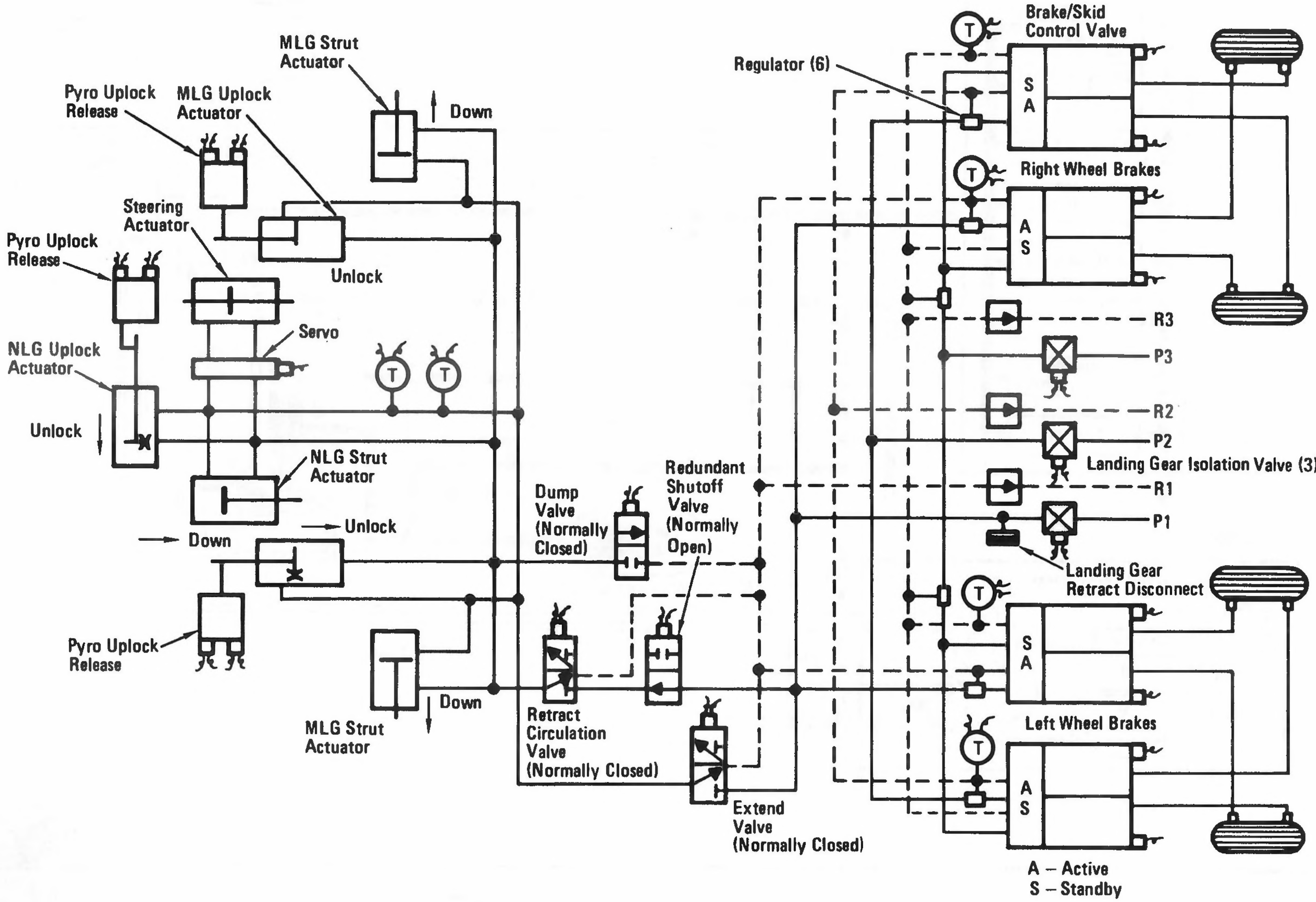
The hydraulic systems are designated 1, 2, and 3. Each consists of the components necessary to generate, distribute, control, monitor, and utilize hydraulic pressure and thermally condition the hydraulic fluid.

The hydraulic pump for each hydraulic system is a variable displacement-type pump. Each hydraulic pump has an electrically operated depressurization valve. The depressurization valve for each hydraulic pump is controlled by its respective HYD MAIN PUMP PRESS switches 1, 2, or 3 located on the flight deck display and control panel R2. When the applicable HYD MAIN PUMP PRESS switch is positioned to LOW, the respective depressurization valve is energized, which reduces the hydraulic pump discharge pressure from its normal 150,075 to 160,425 millimeters of mercury (mmHg) (2,900 to 3,100 psi) output to 25,875 to 51,570 mmHg (500 to 1,000 psi), thereby reducing the auxiliary power unit (APU) torque requirements during the start of the auxiliary power unit. (When display and control panel nomenclature is printed in all caps—e.g., HYD MAIN PUMP PRESS—it indicates that it is the exact way it appears on the panel.)

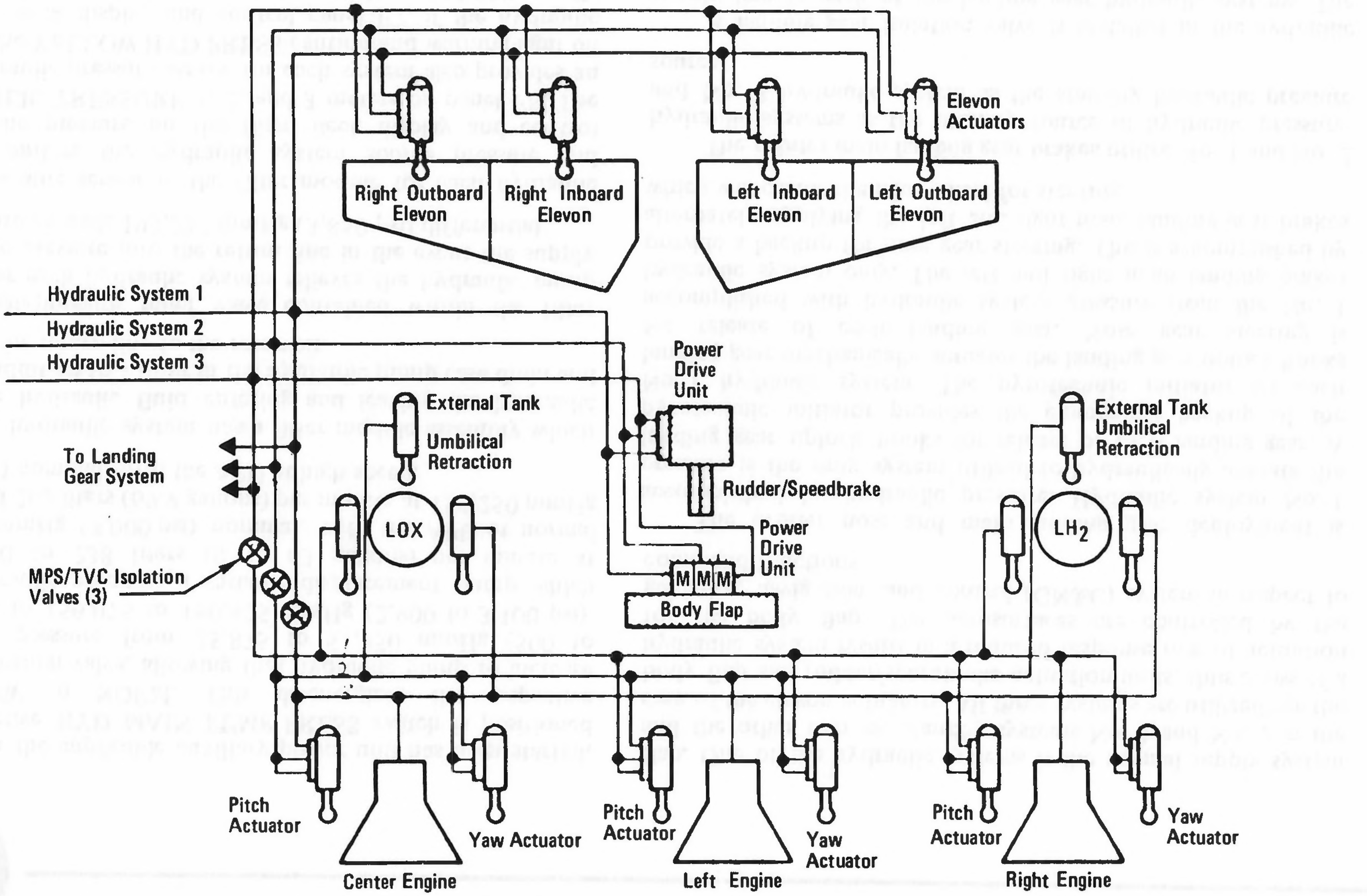
Prior to the start of the APU, the applicable hydraulic system APU/HYD READY TO START talkback indicator on the flight deck display and control panel R2 should be gray. In order for the applicable talkback indicator to indicate gray, the respective hydraulic system HYD MAIN PUMP PRESS switch must be in LOW, the respective BOILER CNTLR/PWR/HTR, BOILER CNTLR, and BOILER H_2 SUPPLY switches must be in ON position on panel R2 and the boiler ready signal must be present which consists of four parameters: boiler stream above 54°C (130°F), N_2 (nitrogen) valve open, bypass valve powered, and boiler enabled.



Hydraulic Power System Main Pump Mode



Hydraulic Actuator System – Landing Gear and Brake



Hydraulic Actuator System –
Landing Gear and Brake

When the applicable auxiliary power unit has been started, the respective HYD MAIN PUMP PRESS switch is positioned from LOW to NORM. This de-energizes the respective depressurization valve, allowing that hydraulic pump to increase its outlet pressure from 25,875 to 51,750 mmHg (500 to 1,000 psi) to 150,075 to 160,425 mmHg (2,900 to 3,100 psi). Each hydraulic pump is a variable displacement pump which provides 0 to 238 liters (0 to 63 gallons) per minute at 150,250 mmHg (3,000 psi) nominal, with the APU at normal speed; and 263 liters (69.9 gallons) per minute at 155,250 mmHg (3,000 psi) nominal, with the APU at high speed.

Each hydraulic system has a filter module assembly which filters the hydraulic fluid entering and leaving the hydraulic pump in addition to a filter in the hydraulic pump case drain and a filter in the return line to the reservoir.

A high-pressure relief valve contained within the filter module for each hydraulic system relieves the hydraulic pump supply line pressure into the return line in the event the supply line pressure exceeds 199,237 mmHg (3,850 psi) differential.

A pressure sensor in the filter module for each hydraulic system monitors the hydraulic system source pressure and displays the pressure on the flight deck display and control HYDRAULIC PRESSURE 1, 2, and 3 meters on panel F8. The same hydraulic pressure sensor for each system also provides an input to the YELLOW HYD PRESS caution and warning light on the flight deck display and control panel F7, if the hydraulic pressure of systems 1, 2, or 3 is below 142,830 mmHg (2,760 psi). The RED BACKUP caution and warning light on panel 7 will illuminate if the hydraulic pressure of systems 1, 2, or 3 is at 142,830 mmHg (2,760 psi).

The aerosurfaces (elevons, rudder/speedbrake and body flap) are powered by hydraulic pressure and the movement of the applicable aerosurface is accomplished mechanically. Each of the aerosurface actuation units have switching valves which interface with all three hydraulic systems with the exception of the body flap. One of the hydraulic systems is the normal supply system and the other two are standby systems No. 1 and No. 2 in the case of the elevon actuators. All three systems are utilized for the body flap and rudder/speecbrake actuation units, thus a loss of a hydraulic system results in a reduced response rate of actuation for the body flap. The aerosurfaces are controlled by the guidance, navigation, and control (GN&C) system in respect to command functions.

The orbiter nose and main landing gear deployment is accomplished by hydraulic pressure. Hydraulic system No. 1 pressure is the only system utilzed to hydraulically actuate the landing gear uplock hooks for release of each landing gear. A pyrotechnic initiator provides the emergency backup of the No. 1 hydraulic system. The pyrotechnic initiator on each landing gear mechanically actuates the landing gear uplock hooks for release of each landing gear. Nose gear steering is accomplished with hydraulic system pressure from the No. 1 hydraulic system only. The left and right main landing brakes provide a backup for nose gear steering. This is accomplished by alternately applying the left and right main landing gear brakes which will castor the nose wheel for steering.

The orbiter main landing gear brakes utilize No. 1 and No. 2 hydraulic systems as the primary source of hydraulic pressure, and No. 3 hydraulic system as the standby hydraulic pressure source.

A landing gear isolation valve is installed in the hydraulic supply line to each of the landing gear hydraulic systems. The No. 1 hydraulic system landing gear isolation valve, when opened, allows hydraulic pressure to the nose and main landing gear systems, the nose gear steering system, and the main landing gear brakes; and, when closed, isolates the No. 1 hydraulic system source pressure from these areas. The No. 2 or No. 3 hydraulic system landing gear isolation valves, when opened, allow access, respectively, to the main landing gear brakes, and, when closed, isolates access to the main landing gear brakes.

Each of the landing gear isolation valves is controlled by its respective HYDRAULICS LG HYD ISOL VLV switch on the flight deck display and control panel R4. The CLOSED position of the applicable switch closes that isolation valve, isolating the respective hydraulic source pressure from that landing gear system. A talkback indicator located above the respective switch on panel R4 would indicate CL (closed). The landing gear isolation valves are closed during prelaunch, boost, and entry. The OPEN position of the respective switch opens the respective isolation valve, allowing that hydraulic system source pressure to its landing gear system. The respective talkback indicator would indicate OP (open). The landing gear isolation valves are opened on-orbit to permit thermal conditioning of the landing gear hydraulic fluid and system. The GPC (general purpose computer) position permits the GPC to open landing gear isolation valve system 3 for 15 minutes, system 2 for 10 minutes, and system 1 for 5 minutes prior to landing to permit thermal conditioning of the landing gear hydraulic fluid and system.

Hydraulic system No. 1 landing gear retract/circulation valve is controlled by the HYDRAULICS LG RET/CIRC VLV switch on the flight deck display and control panel R4. The retract/circulation valve is closed normally during prelaunch, boost, and entry. When the LG RET/CIRC VLV is position to OPEN, the No. 1 hydraulic system landing gear retract/circulation valve is energized open which permits system source pressure to circulate through the retract lines through the nose and main landing gear uplock actuators and return through the extend lines (the pressure inlet port of the landing gear control valve is closed, however, this outlet port of the landing gear control valve is open to the return line) which permits the thermal conditioning of the landing gear hydraulic fluid and system. The CLOSE position closes the retract/circulation valve which prevents the thermal conditioning landing gear hydraulic fluid and system. The GPC position permits the GPC to open the retract/circulation valve on-orbit to permit thermal conditioning of landing gear hydraulic fluid and system. The retract/circulation valve will be closed in the GPC mode prior to entry.

The three Space Shuttle main engines (SSME) and their associated controllers provide the control of the individual hydraulic actuators which positions each SSME preburner oxidizer valve, main oxidizer valve, chamber coolant valve, fuel preburner oxidizer valve, and the main fuel valve. These valves are commanded open for SSME ignition and are sustained in the open position through ascent. These valves are commanded closed hydraulically at MECO (main engine cutoff). After SSME shutdown and ET (external tank) separation, these valves are sequenced open for SSME propellant dump and purge, then sequenced closed for the remainder of the mission. Hydraulic source pressure No. 1 supplies SSME 1; hydraulic source pressure No. 2 supplies SSME 2; and hydraulic source pressure No. 3 supplies SSME 3. The SSME pneumatic system provides a backup closure of these valves in the event of a loss of hydraulic source pressure.

After ET separation and SSME propellant dump and purge, the orbiter liquid oxygen and liquid hydrogen umbilical at the ET/orbiter interface are retracted and locked by three hydraulic actuators at each umbilical. Hydraulic system No. 1 source pressure is supplied to one actuator at each umbilical; hydraulic system No. 2 source pressure is supplied to a second actuator at each umbilical; and hydraulic system No. 3 source pressure is supplied to a third actuator at each umbilical.

Each SSME is provided with thrust vector control by a pitch and yaw actuator which is controlled by the ascent thrust vector control system. Each actuator is powered hydraulically which mechanically gimbals the SSME for start and launch position and for thrust vector control during the ascent. Each actuator has a switching valve which allows a primary hydraulic source pressure to power the actuator. A standby hydraulic system is available in the event of failure of the primary. The center SSME pitch actuator primary hydraulic source pressure is hydraulic system

No. 1 and the standby hydraulic supply is No. 1. The center SSME yaw actuator primary hydraulic source pressure is hydraulic system No. 3 and the standby hydraulic supply is No. 1. The left SSME pitch actuator primary hydraulic source pressure is No. 2 and the standby is No. 1. The left SSME yaw actuator primary hydraulic supply source pressure is No. 1 and the standby is No. 2. The right SSME pitch actuator primary hydraulic supply source pressure is No. 3 and the standby is No. 2. The right SSME yaw actuator primary hydraulic supply source pressure is No. 2 and the standby is No. 3. After MECO, the actuators will position the SSME's to the dump position for SSME propellant dump to minimize attitude disturbance. After propellant dump the actuators will position the SSME's to the stowed position for minimum aerodynamic interference for entry.

The hydraulic source pressure of each hydraulic system is supplied or isolated to the SSME engine hydraulic actuators and the SSME thrust vector control pitch and yaw actuators by the MPS (main propulsion system)/TVC (thrust vector control) ISOL (isolation) VLV 1, 2, and 3 switches on the flight deck display and control panel R4. When the applicable MPS/TVC ISOL VLV switch is positioned to OPEN, the corresponding hydraulic source pressure is supplied to the aforementioned areas; and when positioned to CLOSE, that hydraulic system is isolated. A talkback indicator located above the respective switch indicates OP when that valve is open or indicates CL when that valve is closed. The MPS/TVC ISOL VLV 1, 2, and 3 are open during prelaunch and ascent and are closed after main propulsion dump, and remain closed except to reposition the SSME's prior to deorbit, if required.

The return line of each hydraulic system is directed to its respective WSB. There is one WSB for each hydraulic system. The WSB provides the expendable heat sink for each orbiter hydraulic system and each of the APU lube oil systems during prelaunch, boost phase, on-orbit checkout, de-orbit, and entry through rollout and landing.

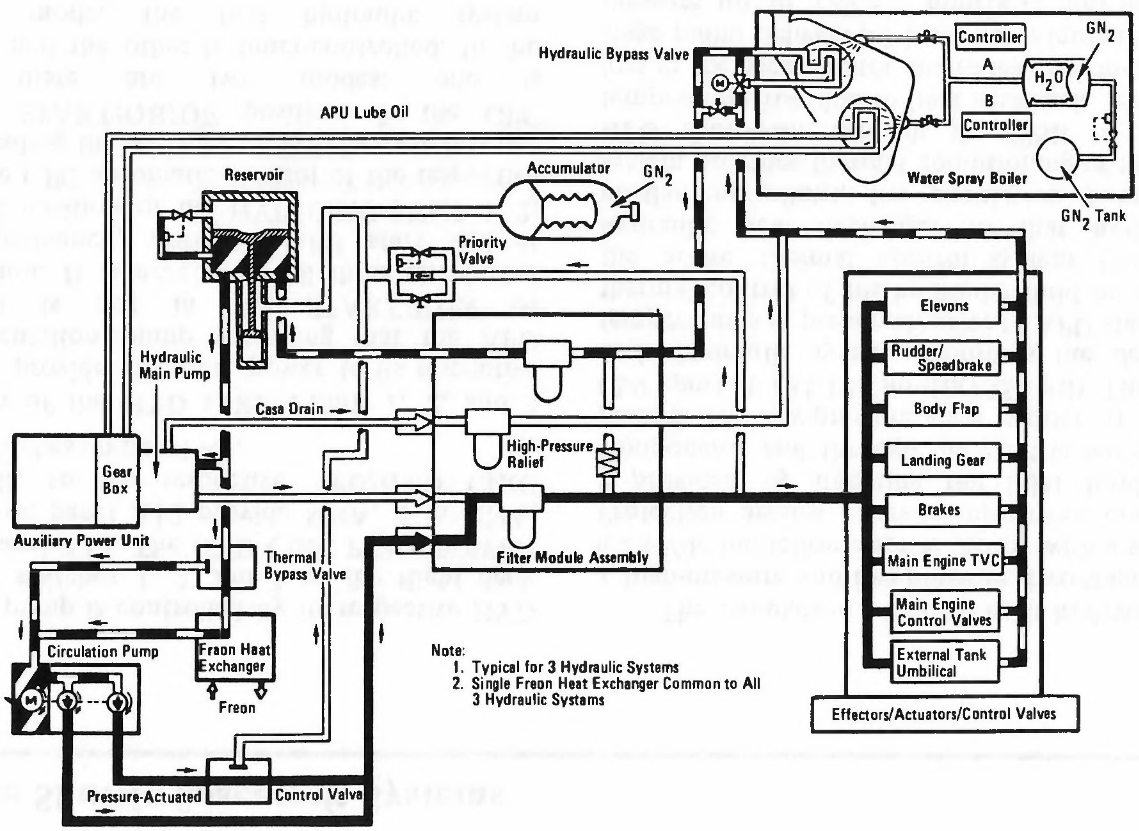
Due to the unique hydraulic system fluid flows, hydraulic fluid control valves are located in the return line of the hydraulic system to the WSB. Normally, the hydraulic system fluid flows up to 79 liters per minute (21 gallons per minute), however, the hydraulic system experiences one- to two-second flow spikes up to 238 liters per minute (63 gallons per minute). If these spikes were to pass through the WSB, pressure drop would increase nine-fold and the WSB would flow-limit the hydraulic system. To prevent this, a relief function is provided by a spring-loaded poppet valve which opens when the hydraulic fluid pressure exceeds 2,484 mmHg (48 psi) and is capable of flowing 162 liters per minute (43 gallons per minute) at 2,587 mmHg (50 psi) differential across the WSB. A temperature controller diverter valve allows the hydraulic fluid to bypass the boiler when the hydraulic fluid has increased to 98°C (210°F). At 98°C (210°F), the controller commands the diverter valve to direct the hydraulic fluid through the WSB. When the hydraulic fluid cools to 87°C (190°F), the controller again commands the diverter valve to bypass the fluid around the WSB.

A reservoir in each hydraulic system return line provides positive pressure fluid to the main hydraulic pump and the circulation pump in that hydraulic system. Within each hydraulic reservoir, there is a differential area piston (40 to 1 area ratio between the reservoir side and accumulator side) which is actuated by that hydraulic system pressure and provides the pressurized fluid to the main hydraulic pump and circulation pump of that system. The reservoir of each hydraulic system also provides for volumetric expansion and contraction. The capacity of the MIL-H-83282 hydraulic fluid (synthetic hydrocarbon, reduces fire hazard) in each reservoir is 30,321 cubic centimeters (1,850 cubic inches). The quantity of each reservoir is monitored on the flight deck display and control panel F8 as HYDRAULIC QUANTITY 1, 2, and 3 in percentages. The quantity in each reservoir is 30 liters (8 gallons). A pressure relief valve in each reservoir protects that reservoir from overpressurization and relieves at 6,210 mmHg (120 psi) differential.

A hydraulic accumulator in each hydraulic system dampens pressure impulses and minimizes the main hydraulic pump and circulation pump pressure ripples. Each accumulator is a piston type and precharged with gaseous nitrogen (GN_2) at 87,975 to 90,562 mmHg (1,650 to 1,750 psi). The GN_2 capacity of each accumulator is 243 cubic centimeters (96 cubic inches). The hydraulic fluid volume is 129 cubic centimeters (51 cubic inches).

A circulation pump in each hydraulic system is utilized to provide the thermal conditioning of the hydraulic fluid in that system. The circulation pump in each hydraulic system is electrically driven.

A pressure sensing priority valve in each hydraulic system automatically closes and traps the accumulated hydraulic pressure permitting the reservoir to supply a positive hydraulic pressure from the reservoir to the main hydraulic pump and circulation pump in altitude start conditions. A dump valve is provided for each accumulator and is utilized to depressurize the respective hydraulic reservoir for ground operations only.



Hydraulic Power System Circulation Mode

Each circulation pump is controlled by its respective HYD CIRC PUMP POWER switches 1, 2, and 3 on the flight deck display and control panel A12. The HYD CIRC PUMP POWER switches 1, 2, and 3 on panel A12 provide MNA, if in MNA; or MNB, if in MNB, to the respective APU/HYD CIRC PUMPS 1, 2, and 3 switches on panel R2.

The ON position of the HYD CIRC PUMP 1, 2, and 3 switches on panel R2 provide electrical power to its respective hydraulic system circulation pump providing that the APU START/RUN switch is not in the START/RUN or START/ORIDE position. It is noted that all three circulation pumps operate in prelaunch, prior to APU start and at postlanding. The GPC position of the HYD CIRC PUMP 1, 2, and 3 switches provide GPC automatic control of the respective circulation pump providing the APU/START/RUN switch is not in START/RUN or START/ORIDE position. In the GPC automatic mode, there are two modes: one is thermostat-controlled and the other is timer-controlled. In the thermostat-controlled mode, the first hydraulic system temperature sensor to reach below minus 17°C (0°F) is the first system to be circulated. The GPC commands that circulation pump off when all temperature sensors in that hydraulic system being circulated is minus 6°C (20°F) or when that circulation pump has been on for a good period in excess of a specified minimum run time and a second hydraulic system requires circulation. The GPC timer control consists of one temperature sensor in each hydraulic system which has excessively low and high temperature limits, which demand continuous circulation of that hydraulic system and the circulation pump run time for each pump is set to a desired value along with the delay time between pump runs and will continue to operate and cycle the three circulation pumps until the temperature high and low limits are readjusted to normal set points. The OFF position removes electrical power from the circulation pump. At postlanding, the APU/hydraulic system is shutdown. The circulation pumps are activated to circulate the heated hydraulic fluid through the WSB for cooling.

The circulation pump in each hydraulic system consists of a high-pressure and low-pressure, two-stage gear pump driven by a 28-Vdc induction electric motor with a self-contained inverter. Protection against excessive electronic component temperature is provided by directing the inlet fluid flow around these components and through the electric motor before entering the pumps. The low-pressure stage is rated at 10.9 liters per minute (2.9 gpm) at 181,125 mmHg (350 psi). The circulation pumps in each hydraulic system maintains the desired hydraulic fluid temperatures in prelaunch prior to APU start and provides orbital thermal control of the hydraulic fluid by transferring heat from the active thermal control system Freon-21 coolant loop/hydraulic heat exchanger to that hydraulic system. After postlanding/rollout, the circulation pump in each hydraulic system provides thermal conditioning of the hydraulic fluid after APU shutdown through the WSB to limit hydraulic fluid temperature rise due to heat soakback. In the event of pressure loss in the accumulator due to leakage on-orbit, the high-pressure stage pump delivers 0.3 liters per minute (0.1 gpm) at a discharge pressure up to 129,375 mmHg (2,500 psi) to repressurize the accumulator. This function is regulated by a pressure-activated control valve.

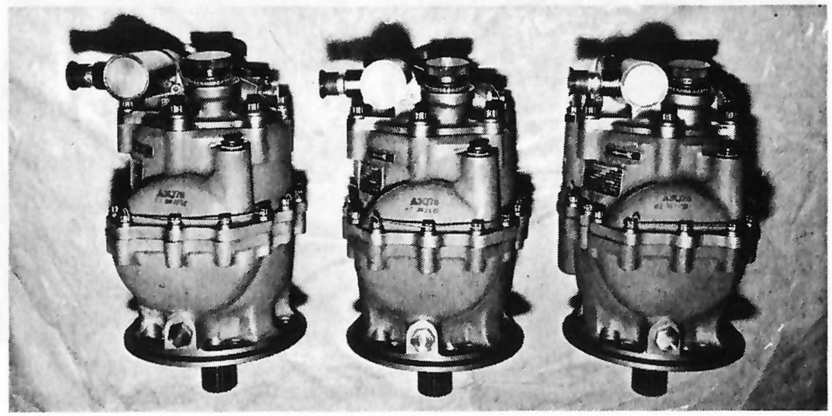
Insulation and electrical heaters are used on those portions of the hydraulic systems which are not adequately thermal conditioned by the individual hydraulic system circulation pump due to stagnant hydraulic fluid areas.

Redundant electrical heaters are installed on the body flap differential gearbox, rudder/speed brake mixer gearbox, the four elevon actuators, the aft fuselage body flap A and B seal cavity drain line, and rudder/speed brake cavity drain line.

The heaters are enabled by the HYDRAULIC HEATER switches on the flight deck display and control panel A12. There are HYDRAULIC HEATER switches A and B for the RUDDER/SPD BK, BODY FLAP, ELEVON, and AFT FUSELAGE. The AUTO A and B position of each of these switches permits the corresponding electrical Bus A or B to power the redundant heaters at each location. The thermostats in each electrical heater system cycle the heaters on/off automatically. The OFF position of the applicable switch removes electrical power from that heater system.

The various parameters of the hydraulic systems are monitored and utilized for display on the flight deck crew display and control panel and transmitted to telemetry and fault messages.

The contractors involved with the hydraulic systems are Arkwin Industries, Westbury, NY (hydraulic reservoir, filter, and control valves); Purolator Inc., Newbury Park, CA (hydraulic filter module); Parker-Hannifin Corp., Irvine, CA (hydraulic accumulator); Abex Corp., Aerospace Division, Oxnard, CA (hydraulic pump); Crissair Inc., El Segundo, CA (hydraulic check valve and flow restrictor); Hi-Temp Insulation Inc., Camarillo, CA (hydraulic blanket insulation); Bertea Corp., Irvine, CA (external tank umbilical retractor actuator, main and nose landing gear uplock actuator); Lear Siegler, Elyria, OH (hydraulic disconnect); Moog Inc., East Aurora, NY (main engine gimbal servo actuators and elevon servo actuators); Pneu Devices, Goleta, CA (hydraulic thermal control shutoff valve and electric-motor-driven circulation pump); Pneu Draulics, Montclair, CA (priority valve hydraulic reservoir); Resistoflex, Roseland, NJ (hydraulic system line connectors); Sterer Engineering and Manufacturing, Los Angeles, CA, (hydraulic solenoid shutoff valve, main engines and landing gear, nose landing gear steering and damping system, 3-way solenoid operating valve landing gear uplock and control valves); Sundstrand, Rockford, IL (rudder/speedbrake actuation unit and body flap actuation unit); Symetrics, Canoga Park, CA (hydraulic quick disconnects); Titeflex Division, Springfield, MA (hydraulic system hose); Whittaker Corp., North Hollywood, CA (hydraulic accumulator dump valve); Wright Components, Inc., Clifton Springs, NJ (hydraulic latching solenoid valve); Hamilton Standard Division of United Technologies Corp., Windsor Locks, CT (water boiler hydraulic thermal unit and ground support equipment hydraulic cart).



Hydraulic System Pumps

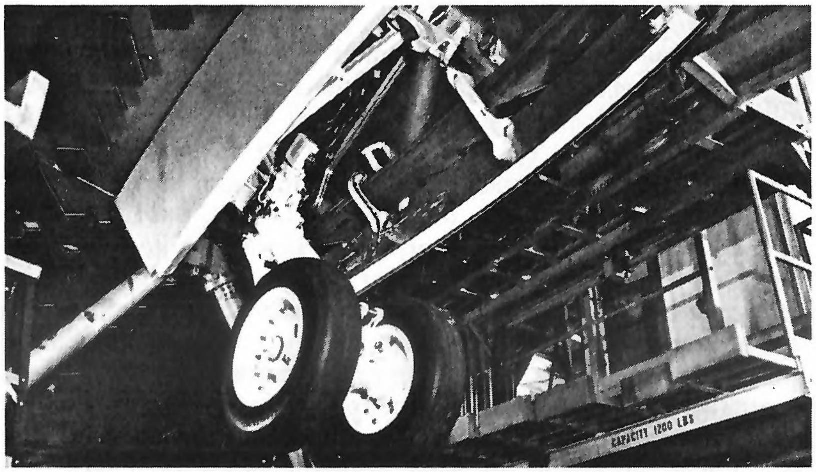
LANDING GEAR SYSTEM

The landing gear system is a conventional aircraft tricycle configuration with steerable nose gear and main left and right landing gears.

The landing gear system includes a shock strut assembly constructed of high-strength, stress-corrosion-resistant steel alloys, aluminum alloys, stainless steel, and aluminum bronze. Each gear is made up of two wheel and tire assemblies. The nose landing gear is steerable, and each of the two main landing gears has a brake assembly with antiskid protection.

The shock strut assembly of each gear is a pneudraulic shock absorber containing gaseous nitrogen and hydraulic fluid. Because the orbiter will be under zero-g conditions during space flight, a floating diaphragm separates the gaseous nitrogen from the hydraulic fluid to maintain absorption integrity.

Cadmium-titanium plating and urethane paint are applied to landing gear strut surfaces for space flight protection. The wheel hubs are forged aluminum, rivited, primed, and painted with two coats of urethane paint.



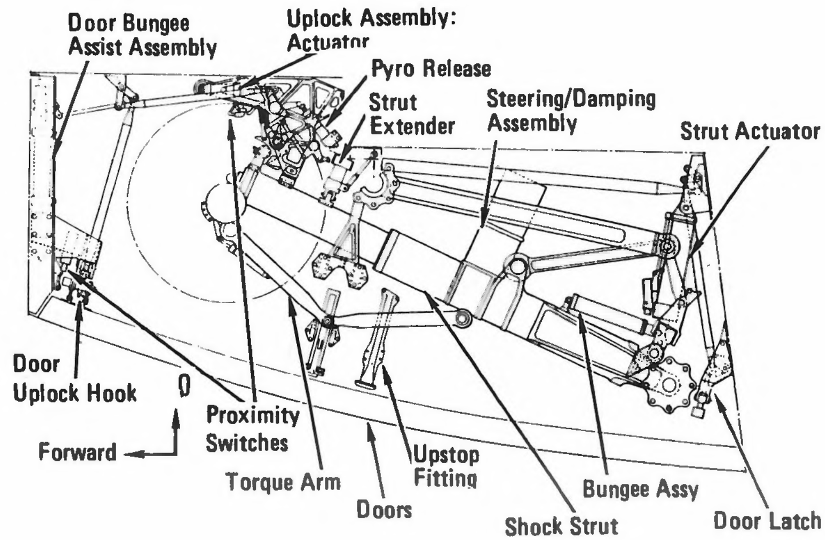
Nose Landing Gear

The landing gears, during the spacecraft's approach, are extended by the orbiter's commander or pilot, who first depresses a landing gear system "Arm" pushbutton and then a "DN" pushbutton. The gear is fully extended within ten seconds.

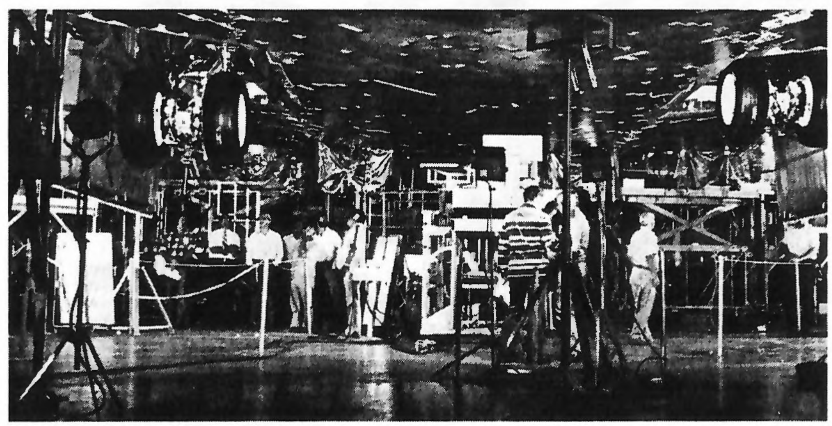
Initiation of the system hydraulically releases uplock hooks, which permit the landing gear to free-fall into the extended position. Springs, hydraulic pressure, and the gear actuators assist the free fall.

Pyrotechnic actuators may also unlock the uplock hooks if the hydraulic system malfunctions. When fully extended, the gear is locked by spring-loaded bungees.

Doors on the nose and the main right and left gears are operated through mechanical linkage attached to the door and

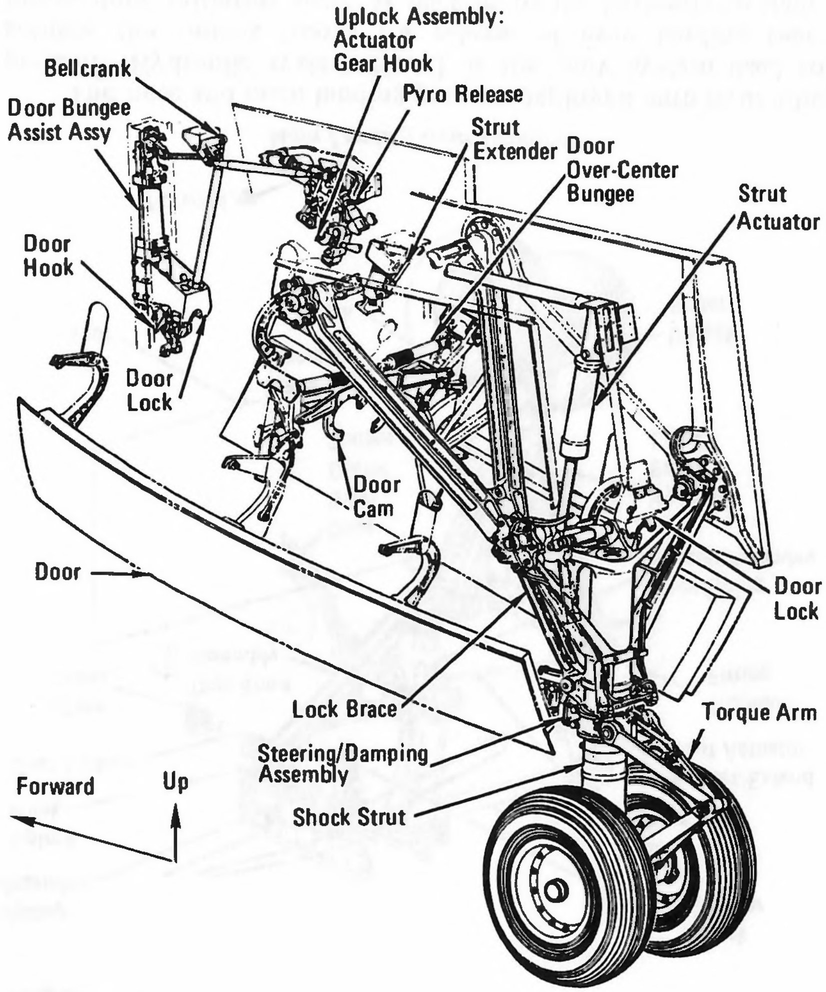


Nose Landing Gear Stowed



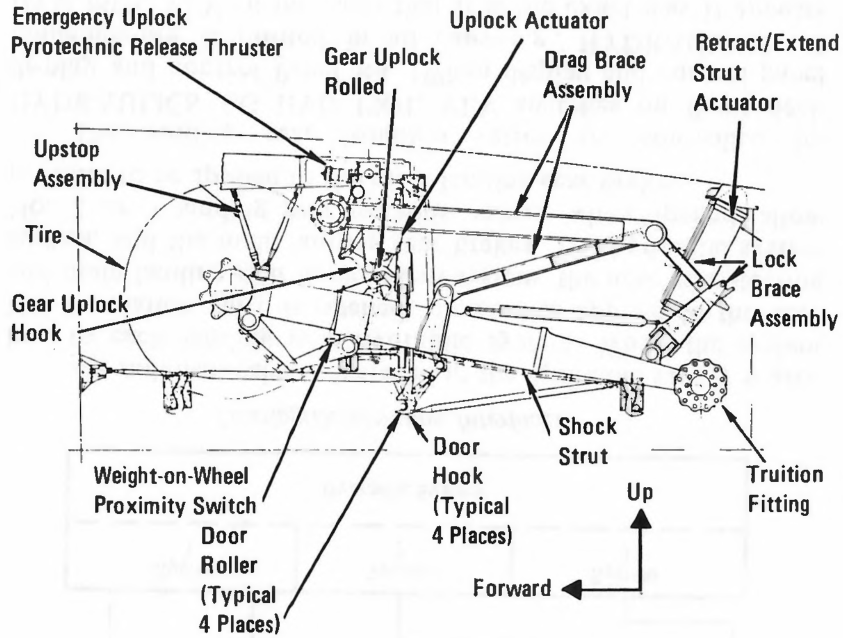
Main Landing Gear

fuselage and powered by strut camming action during gear extension. As the uplock hook is released and the gear begins its



Nose Landing Gear Deployed

descent, the doors will open. The gear strut actuators include an oil snubber to control the rate of extension and prevent damage to the downlink linkages.

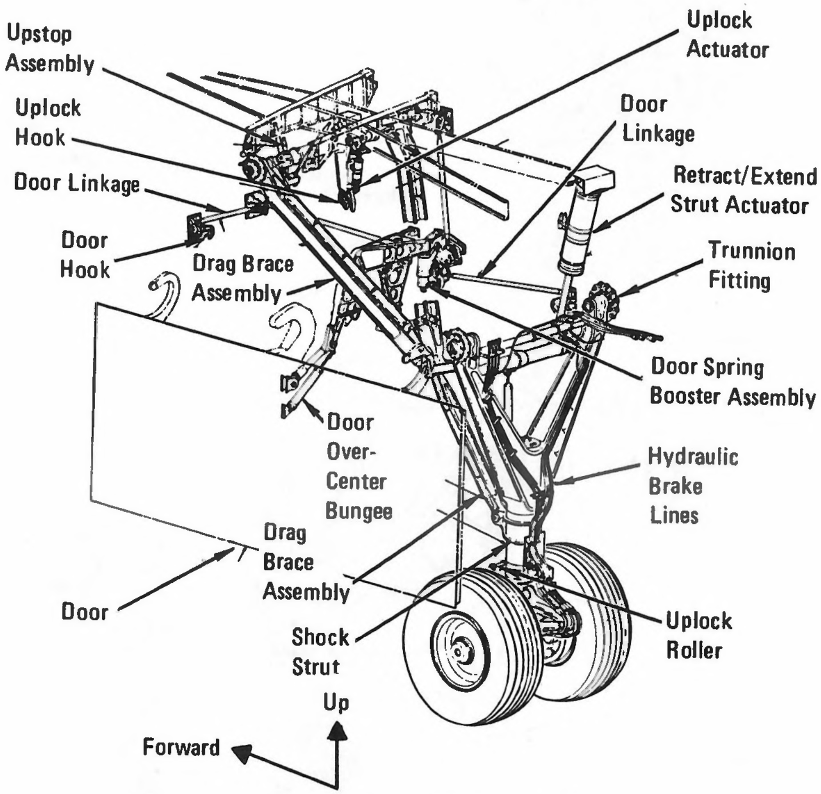


Main Landing Gear Stowed

The landing gears may not be retracted in flight. Retraction is a ground operation. For retraction, each gear is hydraulically moved forward and up. The mechanical linkage closes the doors, and the uplock hook is engaged to retain the gears in the housing. The nose gear is housed in the forward fuselage, and two doors cover the area. The right and left main gears are retracted into the wings and have only one large door covering the area.

Each landing gear door has high-temperature, reusable surface insulation (HRSI) tiles bonded to the outer surface and a thermal barrier to protect the landing gear from the high-temperature thermal loads encountered during entry.

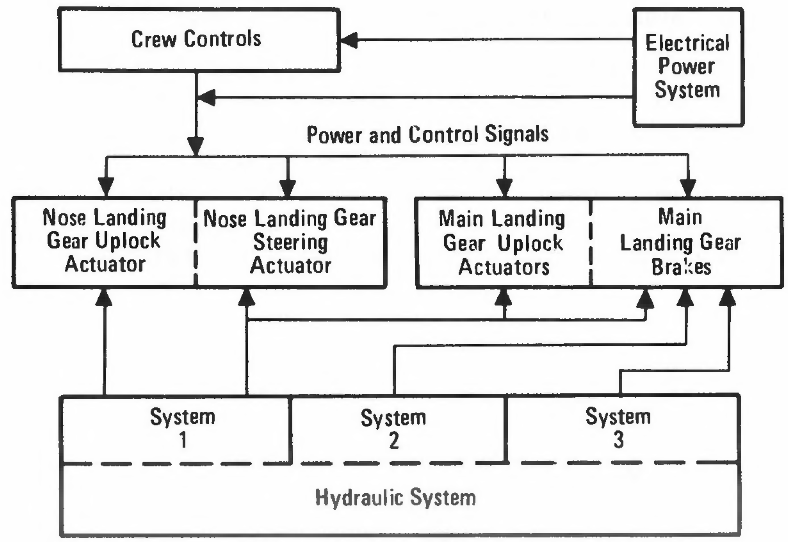
The gears are deployed only after the spacecraft has an indicated airspeed of less than 300 knots (345 mph).



Main Landing Gear Deployed

The nose and main landing gear are deployed with hydraulic pressure. Hydraulic system No. 1 is the only system used to actuate the uplock hooks for release of each landing gear. Pyrotechnic initiators serve as backup to the hydraulic system. Hydraulic system No. 1 also is the only system used to position the nose gear for steering. During rollout, the orbiter also may be steered by differential main gear brake application, which casters the nose wheel for steering.

The orbiter main landing gear brakes use hydraulic systems No. 1 and 2 as the primary sources and No. 3 as a standby source.



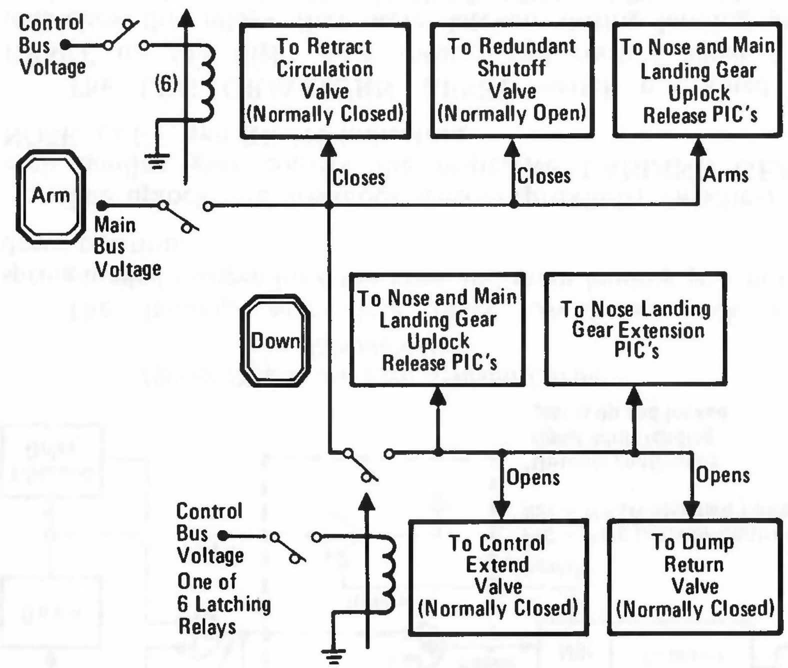
Landing/Deceleration Interfaces

An isolation valve is installed in the hydraulic supply source line to each landing gear hydraulic system. When the system No. 1 isolation valve is opened, pressure is applied to the nose and main landing gear deployment system, the nose gear steering system, and the main landing gear brakes. The hydraulic system No. 2 or 3 landing gear isolation valves, when opened, allow pressure to be applied to the main landing gear brakes.

The landing gear isolation valves are controlled by HYDRAULICS LG HYD ISOL VLV switches on flight deck display and control Panel R4. (When display and control panel nomenclature is printed in all caps–e.g., HYDRAULICS LG HYD ISOL VLV–it indicates that it is the exact way it appears on the panel.) When these switches are closed, the hydraulic sources are isolated from that landing gear system; when open, they are connected to the systems. An indicator located above each switch shows whether the valve is closed (CL) or open (OP). The isolation valves normally are closed during prelaunch, boost, and entry. The hydraulic system No. 1 isolation valve is opened on orbit to permit thermal conditioning of the landing gear deployment hydraulic systems; it is closed before entry. The GPC (general purpose computer) position of the isolation valve switches enables the computer to open the landing gear isolation valves 15, 10, and 5 minutes, for system No. 3, 2, and 1, respectively, before landing for thermal conditioning of the landing gear hydraulic brake systems.

The HYDRAULICS LG RET/CIRC VLV switch on Panel R4 controls the landing gear retract/circulation valve in hydraulic system No. 1. The retract/circulation valve normally is closed during prelaunch, boost, and entry. When it is open, hydraulic system No. 1 source pressure circulates through the retract lines to the nose and main landing gear uplock actuators for thermal conditioning of the system No. 1 one landing gear hydraulic fluid. The pressure enters through an orifice in the piston head and returns through the extend lines (the pressure inlet port of the landing gear control valve is closed; however, the outlet port of the landing gear control valve is open to the system No. 1 return line). The hydraulic fluid flow to the landing gear strut actuators dead ends at the actuators' retract ports. The GPC position of the switch permits the computer to automatically (temperature controlled) open and close the retract/circulation valve to allow on-orbit thermal conditioning of the landing gear hydraulic system number one fluid. It is noted that the retract/circulation valve would be closed in the GPC mode when the landing gear system is ARMED for deployment. Deployment controls for the nose and main landing gear are located on the flight deck display and control panel. The commander (CDR) and pilot (PLT) each has landing gear indicators and control stations. The commander's indicator and controls are on panel F6; and pilot's indicators and controls are on panel F8. At each station is a NOSE, LEFT, and RIGHT landing gear indicator and an ARM and DN (down) pushbutton switch/light indicator with guard covers.

The landing gear position indicators indicate UP when the gear is up and locked, BARBER POLE when the gear is deploying or retracting, and DN when the gear is down and locked.



Arm/Down Circuit (Simplified)

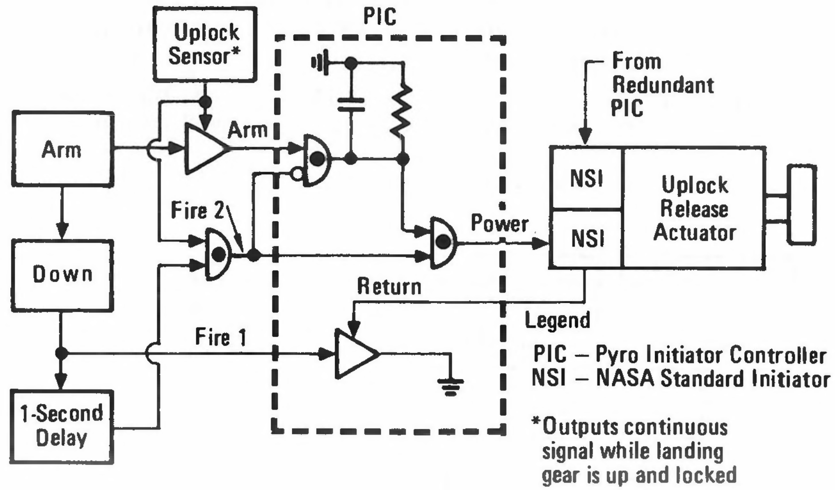
Landing gear deployment is initiated when the CDR or PLT depresses the guarded ARM pushbutton switch/light indicator, then the guarded DN pushbutton switch/light indicator. This is accomplished at least 15 seconds prior to predicted touchdown and at a speed no greater than 300 knots (345 mph).

Depressing the ARM pushbutton switch/light indicator energizes latching relays that close the hydraulic system No. 1 landing gear retract/circulation valve and the normally open redundant shutoff valve to the retract/circulation valve. It also arms the nose and main landing gear pyrotechnic initiator controller's (PIC's) and illuminates the yellow light in the ARM pushbutton switch/light indicator.

The DN pushbutton switch/light indicator is then depressed. This energizes latching relays that open the hydraulic system No. 1 landing gear control valve, permitting the fluid in hydraulic system No. 1 to flow through the inlet port to the outlet port to the landing gear extend hydraulic lines. It also opens the landing gear dump valve, allowing the landing gear retract line fluid to flow into the hydraulic system No. 1 return line and illuminating the green light in the DN pushbutton switch/light indicator.

Hydraulic system No. 1 source pressure is routed to the nose and main landing gear uplock actuators, which releases the nose and main landing gear uplock hooks and door uplock hooks. As the uplock hooks are released, the gear begins its descent, and mechanical linkage attached to the doors and fuselage is powered by landing gear strut camming action during the gear extension, which opens the landing gear doors. There are two landing gear doors for the nose gear and one door for each main landing gear. The landing gear free-falls into the extended position, and the strut actuators aid in the deployment. The hydraulic strut actuator incorporates a hydraulic fluid flow-through orifice (snubber) to control the rate of landing gear extension to prevent damage to the gear's down-lock linkages.

If hydraulic system No. 1 fails to release the landing gear—in one second after the DN pushbutton is depressed—the nose, left, and right main landing gear uplock sensors (proximity switches) will provide inputs to the PIC's for initiation of the redundant NASA Standard Initiators (NSI's) (nose, left, and right main landing gear pyrotechnic backup release system). They release the same uplock hooks as the hydraulic system. The nose landing gear, in addition, has a PIC and redundant NSI's that initiate a pyrotechnic power assist thruster strut extender two seconds after the DN pushbutton is depressed.



Uplock Release and Pyro Actuator Circuit (Simplified)

The landing gear drag brace overcenter lock and spring-loaded bungee lock the nose and main landing gear in the down position.

The uplock and downlock sensors (proximity switches) on each landing gear control the respective LANDING GEAR NOSE, LEFT, and RIGHT indicators.

The LDG GR/ARM/RN RESET switch positioned to RESET on the flight deck display and control panel A12 unlatches the relays that were latched during landing gear deployment by the LANDING GEAR ARM and DN pushbutton light/switch indicators. This is primarily a ground function, which will be performed only during landing gear deactivation.

The RESET position also will extinguish the yellow light in the ARM pushbutton switch/light indicator and the green light in the DN pushbutton switch/light indicator. In addition, the hydraulic system No. 1 landing gear dump valve is closed, the retract/circulation valve is opened only if the switch is in the OPEN position, and its redundant shutoff valve opened (deenergized) deenergizes the landing gear PIC circuits, and the landing gear control valve closes off the source pressure to the landing gear.

The nose landing gear tires are 32 by 8.8 inches and will withstand a burst pressure of not less than 3.2 times the normal inflation pressure of 15,525 mmHg (300 psi). The maximum total load per nose landing gear tire to prevent tire damage is 20,412 kilograms (45,000 pounds) and is rated at 225 knots (258 mph).

The nose landing gear shock strut has a 55-centimeter (22-inch) stroke. The maximum derotation rate to prevent gear and tire damage is approximately 9.4 degrees per second, 3.3 meters per second (11 feet per second).

The main landing gear tires are 44.5 by 16-21 inches. The normal inflation pressure is 16,301 mmHg (315 psi). The maximum tested load per main landing gear tire to prevent tire damage is 55,792 kilograms (123,000 pounds). If the orbiter touches down with a 60/40 percent load distribution on a strut's two tires, with one tire supporting the maximum load of 55,792 kilograms (123,000 pounds), then the other tire can support a load of only 37,381 kilograms (82,410 pounds). Therefore, the maximum tire load on a strut is 93,173 kilograms (205,410 pounds) with a 60/40-percent tire load distribution. The STS-1 tire is rated at 213 knots (245 mph). The operational tires will be rated at a higher velocity.

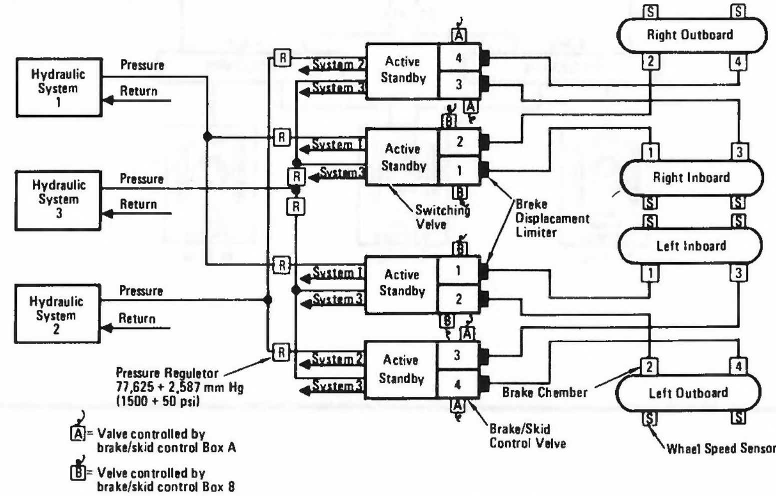
The main landing gear shock strut stroke is 40 centimeters (16 inches). The allowable main gear sink rate for a 85,276-kilogram (188,000-pound) orbiter is 2.9 meters per second (9.6 feet per second) and 1.8 meters per second (6 feet per second) with a 102,513-kilogram (226,000-pound) orbiter. With a 20-knot (23-mps) crosswind, the maximum allowable gear sink rate for a 85,276-kilogram (188,000-pound) orbiter is 1.8 meters per second (6 feet per second) and approximately 1.5 meters per second (5 feet per second) with a 102,513-kilogram (226,000-pound) orbiter.

The landing gear tires have a life of five nominal landings.

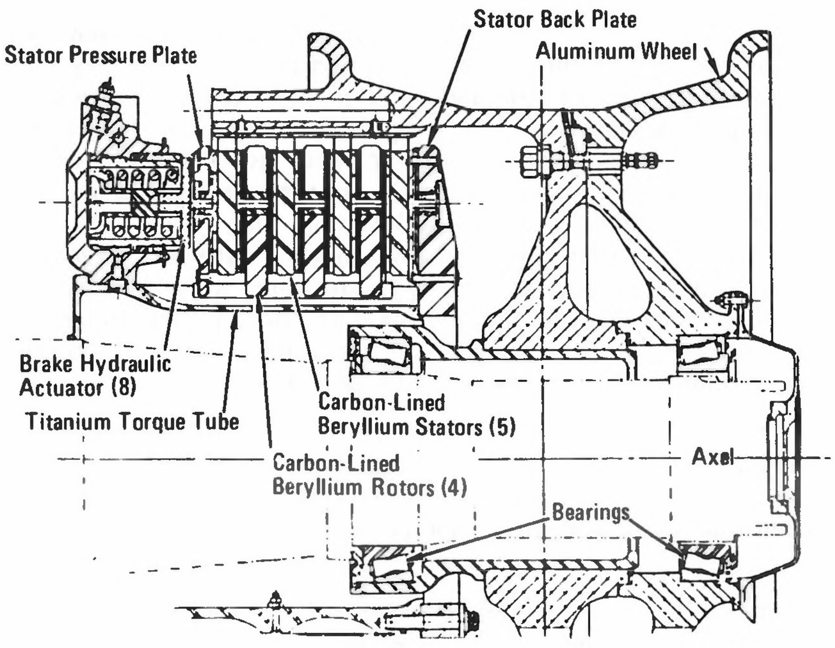
MAIN LANDING GEAR BRAKES

Each of the orbiter's four main landing gear wheels has electro-hydraulic disc brakes and an anti-skid system.

Each main landing gear wheel has a disc brake assembly consisting of nine discs, four rotors, and five stators. The carbon-lined beryllium rotors are splined to the inside of the wheel and rotate with the wheel. The carbon-lined beryllium stators are splined to the outside of the axle assembly; they do not rotate with the wheel. When the brakes are applied, eight hydraulic actuators in the brake assembly press the discs together, providing brake torque. Four of these actuators are manifolded together from hydraulic system No. 1 in a brake chamber. The remaining four actuators are manifolded together from hydraulic system No. 2. The standby hydraulic system is hydraulic system No. 3.



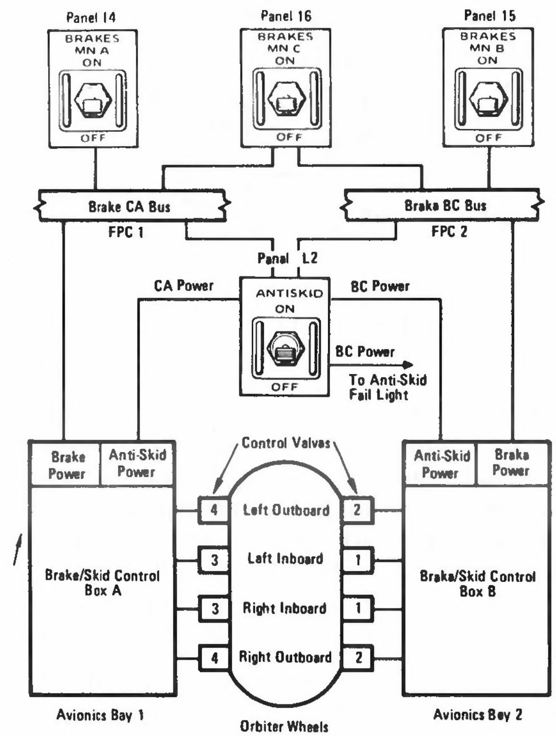
Wheel Brake Subsystem



Wheel Brake Assembly

As in the landing gear deployment, the landing gear isolation valve in hydraulic system Nos. 1, 2, and 3 must be open to allow the applicable hydraulic source pressure to the main landing gear brakes.

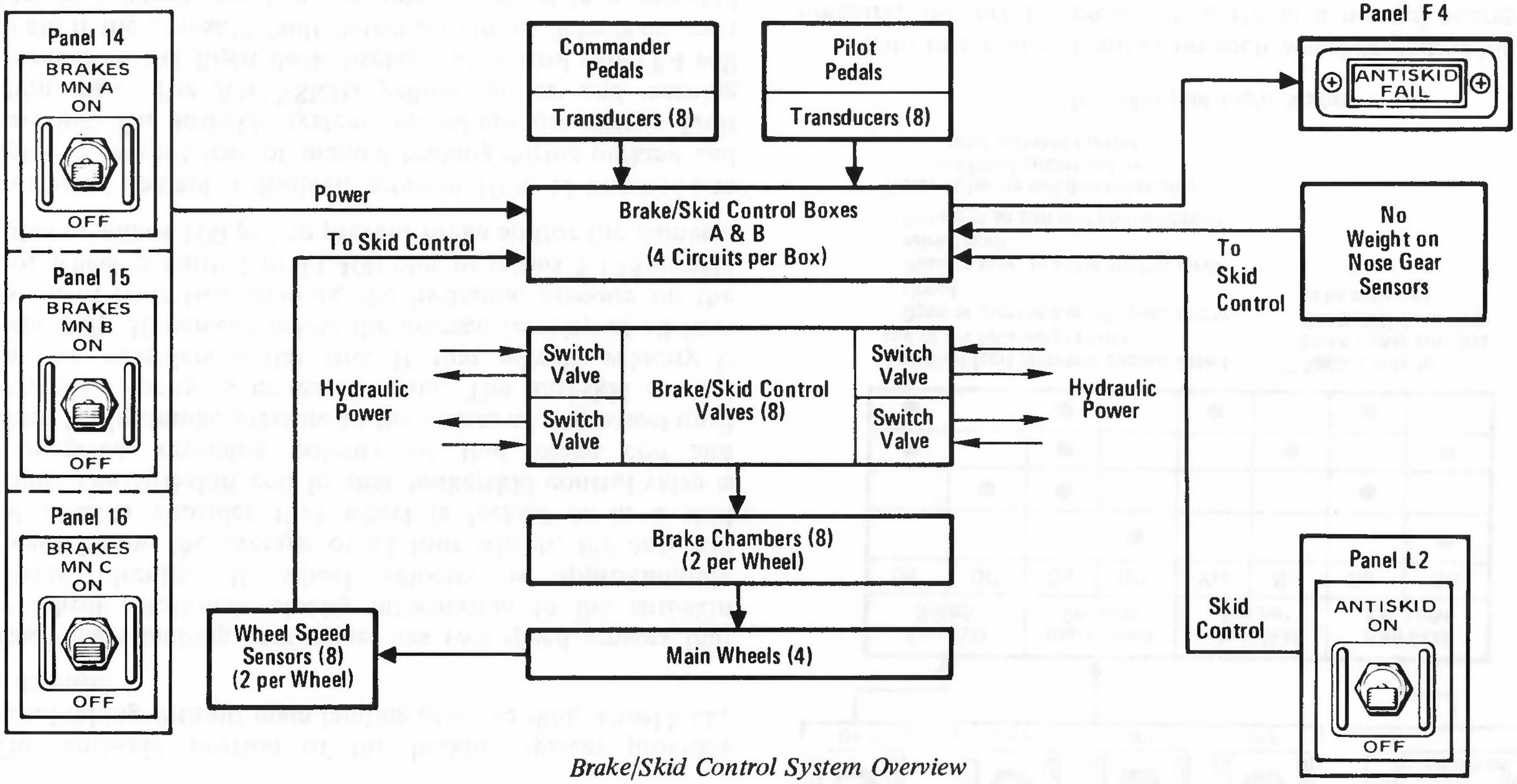
The BRAKES MNA, MNB, and MNC switches are located on the flight deck display and control panel 014, 015, and 016 and allow electrical power to the brake/anti-skid control boxes A and B. The ANTI SKID switch located on panel L2 provides electrical power for enabling the anti-skid portion of the braking system boxes A and B. The BRAKES MNA, MNB and MNC switches are positioned to ON to supply electrical power to the brake boxes A and B and to OFF to remove electrical power. The ANTI-SKID switch is positioned to ON to enable the anti-skid system and to OFF to disable the system.



Brake/Skid Control Power

Brake/anti-skid boxes A and B will not permit the main landing gear brakes to be applied until weight is sensed on one of the nose landing gear sensors (proximity switches). When weight is sensed on the nose landing gear, the brake/anti-skid boxes A and B are enabled, permitting the main landing gear brakes to become operational. No weight being sensed on the nose landing gear sensor prevents the main landing gear brakes from being applied until the main landing gear wheels touch down on the runway.

The main landing gear brakes controlled by the CDR or PLT brake pedals located on the rudder pedal assemblies at the CDR



Brake/Skid Control System Overview

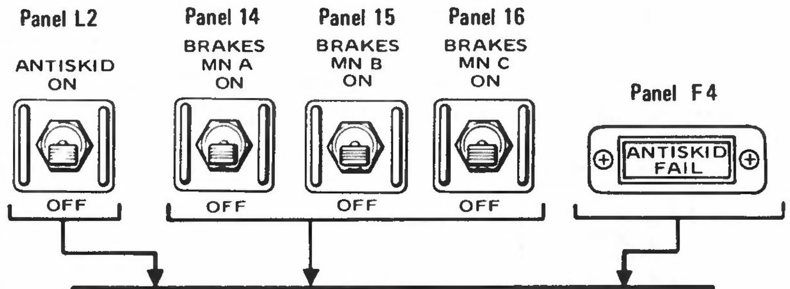
and PLT station. The pedals are adjustable by a handle. The braking commands are accomplished by the CDR or PLT initiating toe pressure on the top of the rudder pedal assembly. Each brake pedal (left and right) has four linear variable differential transducers (LVDT's). The left pedal transducer unit will output four separate braking signals through the brake/skid control boxes for braking control of the two left main wheels. The right pedal transducer unit does likewise for the two right main wheels. When toe pressure is applied to the brake pedal, the transducers transmit electrical signals of 0 to 5 Vdc to the brake/anti-skid control boxes. The pedal with the greatest toe pressure becomes the controlling pedal through electronic OR circuits. The electrical signal is proportional to the toe pressure. The electrical output energizes the main landing gear brake coils proportionally to brake pedal deflection allowing the desired hydraulic volume to be directed to the main landing gear brakes for braking action. The brake system bungee at each brake pedal provides the braking artificial feel to the crew member.

Each of the three hydraulic systems' source pressure of 155,250 millimeters of mercury (mmHg) (3,000 psi) is reduced by a regulator in each of the brake hydraulic systems to 77,625 mmHg (1,500 psi). Hydraulic system Nos. 1 and 2 are the normal hydraulic supply to the brake system, with system No. 3 as the standby. Switching valves in each brake system provide automatic switching if either of the active hydraulic systems is lost.

The anti-skid portion of the braking system provides optimum braking without main landing gear tire skid, wheel lock, or tire damage.

Each main landing gear wheel has two speed sensors that supply wheel rotational velocity information to the anti-skid monitoring circuits. If wheel velocity is approximately 30 percent below the average of all four wheels, the anti-skid control circuits consider that wheel is locked or in a skid condition. The anti-skid coil in that brake/skid control valve is then energized, reversing polarity of that brake coil and inhibiting the hydraulic pressure to the brakes of that wheel until that wheel's velocity is increased again. The anti-skid control system also considers a flat tire. If that wheel's velocity is approximately 30 percent below the average velocity of all four wheels for at least two seconds, the hydraulic pressure on the adjacent wheel is limited to 41,400 plus or minus 5,175 mmHg (800 plus or minus 100 psi) to prevent brake and/or tire damage.

Anti-skid control is disabled between 10 to 15 knots (11 to 17 mph) to prevent loss of manual braking during parking and maneuvering. The anti-skid system control circuits contain fault detection logic. The ANTI-SKID yellow caution and warning light located on the flight deck display and control panel F4 will illuminate if the anti-skid fault detection circuit detects an open or short in a wheel speed sensor, open or short in a anti-skid control valve servo coil, or a failure in an anti-skid control circuit. A failure of the aforementioned items will only deactivate the failed circuit, not the total anti-skid control. If the BRAKE POWER switches are ON and the ANTI-SKID switch is OFF, the ANTI-SKID caution/warning light will illuminate.



Anti-Skid Switch		Brake Power Switches		Anti-Skid Failure*		Anti-Skid Fail Light	
On	Off	On	Off	Yes	No	On	Off
			●				●
	●	●				●	
●		●			●		●
●		●		●		●	

*Anti-Skid fault detection circuits detect any of the following failures:
Open or short in a wheel speed sensor circuit
Open or short in a skid control servo valve circuit
Failure in an anti-skid control circuit

Note: A failure will deactivate only the failed circuit and not total anti-skid control

**Applies only to brake power switches MN B, MN C, or both to be activated

Anti-Skid Fail Light Status

With one brake chamber on each wheel providing hydraulic pressure, the orbiter crew can perform a normal energy stop; however, the braking distance will be approximately 30 to 40 percent longer.

In a normal-energy stop with an 85,276-kilogram (188,000-pound) orbiter, beginning with initial braking at 153 knots (176 mph), 871 meters (2,860 feet) is required to stop the orbiter with 77,625 mmHg (1,500 psi) of hydraulic pressure. An emergency stop with a 102,513 kilogram (227,000-pound) orbiter beginning with initial braking at 178 knots (204 mph), requires 1,330 meters (4,364 feet) to stop the orbiter with 77,625 mmHg (1,500 psi) of hydraulic pressure.

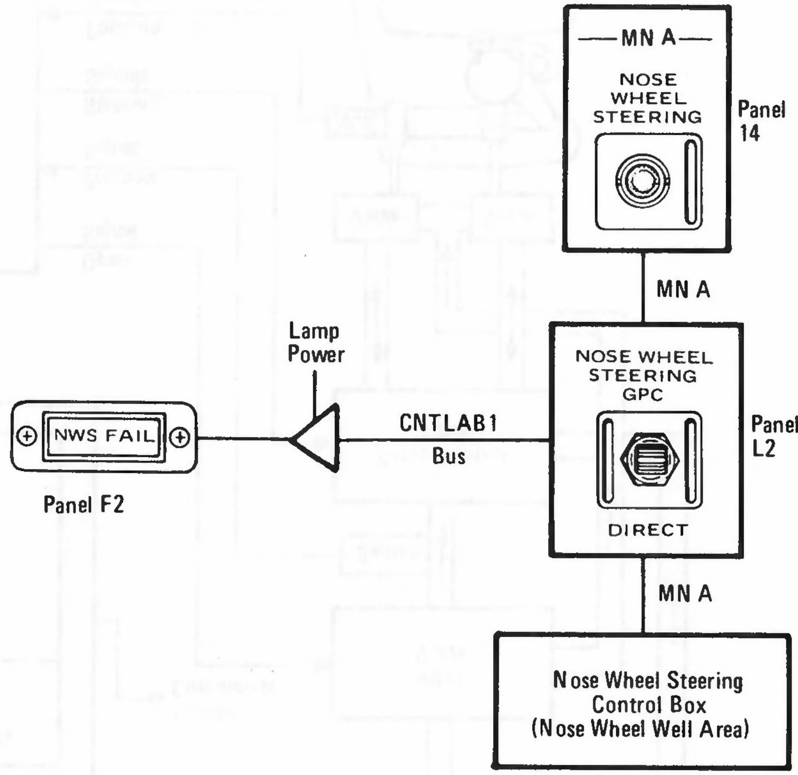
The brakes have five normal energy stop landings. One emergency stop landing will require brake refurbishment.

NOSE WHEEL STEERING

The orbiter nose wheel is steerable upon landing. The nose wheel is electro-hydraulically steerable automatically through use of the general purpose computer (GPC) and flight control system AUTO mode. The nose wheel may also be steered by use of the CDR or PLT rudder pedals in the control stick steering (CSS) mode.

Only hydraulic system No. 1 supplies hydraulic system pressure for steering the nose wheel in either of the aforementioned modes. If system No. 1 is inoperative, the CDR or PLT can apply the left and right main landing gear brakes, alternately, which will caster the nosewheel and provide nose wheel steering.

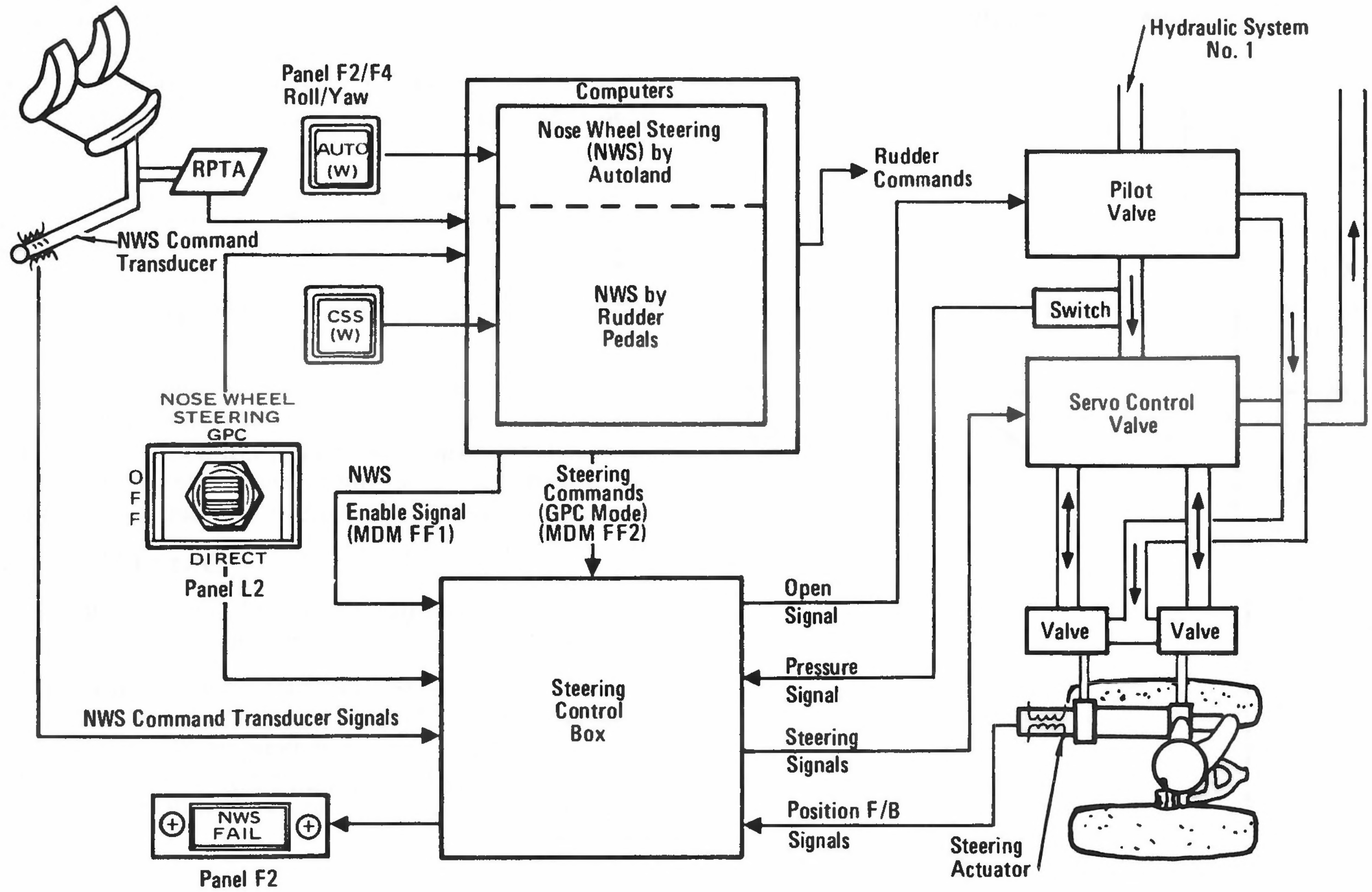
The NOSE WHEEL STEERING switch on the flight deck display and control panel L2 enables the nose wheel steering solenoid control valves, which directs hydraulic system No. 1 pressure to the nose wheel steering servo valve when the NOSE WHEEL STEERING switch is in the GPC position. The GPC position, in conjunction with the flight control system ROLL/YAW AUTO pushbutton switch/light indicator on panel F2 or F4, allows the AUTOLAND computer mode to provide steering commands to the nose wheel servo actuator when weight on the nose gear (main and nose gear proximity switches plus vehicle attitude) is sensed and the hydraulic pressure switch detects that the hydraulic pressure is at least 69,862 mmHg (1,350 psi). Position feedback is transmitted from the nose wheel steering actuator for position verification and fault detection. It



Nose Wheel Steering Power

is noted that if the hydraulic pressure falls to 51,750 plus or minus 7,762 mmHg (1,000 plus or minus 150 psi), nose wheel steering is disabled. When nose wheel steering is not activated, the nose wheel steering damping actuator prevents nose wheel shimmy and allows the nose wheel to free castor through plus or minus 10 degrees of rotation when differential braking is used.

When the NOSE GEAR STEERING switch is positioned to DIRECT, the nose wheel steering solenoid control valves are enabled, which directs hydraulic system No. 1 pressure to the nose wheel steering servo actuator. The DIRECT position, in



Nose Wheel Steering

conjunction with the flight control system ROLL/YAW CSS pushbutton switch/light indicator on panels F2 and F4 allows the CDR or PLT rudder pedals to provide steering commands directly from the steering command transducer on the rudder pedals to the servo actuator, providing hydraulic pressure to the steering actuator when weight on the nose is sensed. Position feedback is transmitted from the nose wheel steering actuator for position verification and fault detection.

When the flight control system ROLL/YAW AUTO pushbutton switch/light indicator is depressed, it illuminates a white light. When the ROLL/YAW CSS pushbutton switch/light indicator is depressed, it illuminates a white light. Only ROLL/YAW AUTO or ROLL/YAW CSS can be selected at a time, not both simultaneously.

When the NOSE WHEEL STEERING switch is in GPC or DIRECT and the nose wheel steering system fault detection logic detects loss of hydraulic pressure, open or short in the servo control valve circuitry, open or short in the position feedback, rate position error, open or short in the command transducer, broken linkage or loss of electrical power, the NWS (nose wheel steering) FAIL yellow caution and warning light on panel F2 will illuminate, and the nose wheel reverts automatically to free castor.

If nose wheel steering has failed, the NOSE WHEEL STEERING switch is positioned to OFF, extinguishing the NWS FAIL yellow light. Nose wheel steering can then be accomplished by differential braking. In differential braking, the CDR or PLT applies toe pressure to the rudder pedals, alternately to the left and right main landing gear brakes; this will castor the nose wheel for nose wheel steering. When the NOSE WHEEL STEERING switch is in the OFF position, hydraulic system No. 1 pressure to the nose wheel steering system is off, allowing the nose wheel to be in the free castor mode.

Heaters can be installed on the landing gear brake system for temperature control of the brake hydraulic system if they are required on later flights.

Various parameters of the landing gear and its associated hydraulic system are monitored and displayed on the flight deck display and control panel, and transmitted to telemetry and for fault messages.

The contractors for the landing gear are B.F. Goodrich, Troy, OH (main and nose landing gear wheel and main landing gear brake assembly and the nose/main gear tires); Bertea Corp., Irvine, CA (main landing gear hydraulic uplock actuator, main landing gear strut actuator and nose landing gear uplock actuator); Menasco Manufacturing Co., Burbank, CA (main and nose landing gear shock struts and drag brace assembly); Sterer Engineering and Manufacturing, Los Angeles, CA (nose gear steering/damping and solenoid-operated landing gear uplock control valves); Crane Co., Hydro Aire, Burbank, CA (main landing gear brake antiskid system); Eldec Corp., Lynwood, WA (landing gear proximity switch); OEA, Denver, CO (nose gear uplock release pyro thruster); Scott Inc., Downers Grove, IL (main landing gear uplock release thruster actuator).

AVIONICS SYSTEMS

The Space Shuttle avionics system controls, or assists in controlling, most of the Shuttle systems. Its functions include automatic determination of the vehicle status and operational readiness, implementation sequencing and control for the external tank and solid rocket boosters during launch and ascent, performance monitoring, digital data processing, communications and tracking, payload and system management, and guidance, navigation, and control, as well as electrical power distribution for the orbiter, external tank, and solid rocket boosters.

Automatic vehicle flight control can be used for every phase of the mission except docking, which is a manual operation by the flight crew. Manual control—referred to as the control stick steering (CSS) mode—also is available at all times as a flight crew option.

The avionics equipment is arranged to facilitate checkout, access, and replacement with minimal disturbance to the other subsystems. Almost all electrical and electronics equipment is installed in three areas of the orbiter: the flight deck, the forward avionics equipment bays in the mid deck of the orbiter crew compartment, and the aft avionics equipment bays in the orbiter aft fuselage. The flight deck of the orbiter crew compartment is the center of avionics activity, both in flight and on the ground, except during hazardous servicing operations before flight. Before launch, the orbiter avionics system is linked to ground support equipment through umbilical connections.

The Space Shuttle avionics system consists of more than 300 major electronic "black boxes" located throughout the vehicle, connected by more than 300 miles of electrical wiring. The black boxes are connected to a set of five computers through common party lines, called data busses. The black boxes offer dual or triple redundancy for every function.

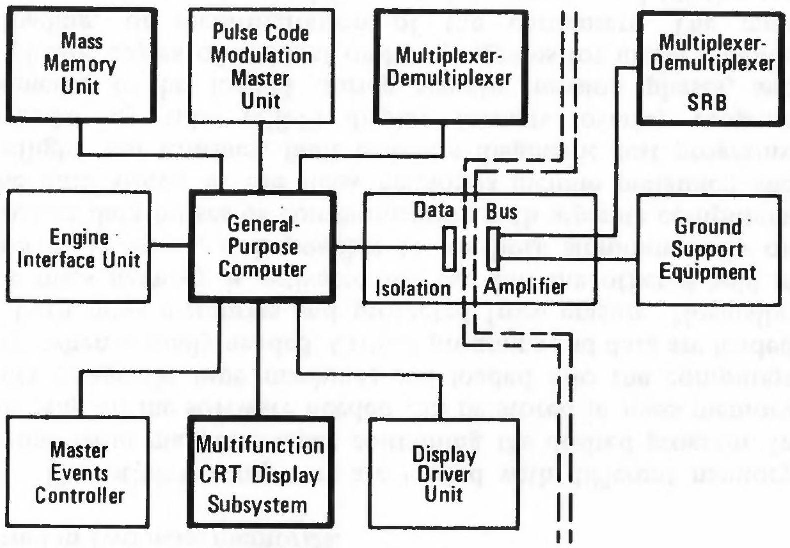
The avionics are designed to withstand multiple failures through redundant hardware and software (computer programs) managed by the complex of five computers; this is called a fail-operational/fail-safe capability. Fail-operational/fail-safe capability is provided by a combination of hardware and software redundancy. Fail-operational performance means that after a first failure in a system, redundancy management allows the vehicle to continue on its mission. Fail-safe means after a second failure, the vehicle still is capable of returning to a landing site safely.

The status of the individual avionics components is checked by a systems monitoring computer program. The status of critical vehicle functions such as the payload door position, external tank and solid rocket booster separation mechanisms, and excessive temperatures for certain areas are monitored continuously and displayed to the crew.

DATA PROCESSING SYSTEM

The orbiter relies on computerized control and monitoring for successful performance. The data processing system (DPS) through the use of various hardware components and its self-contained computer programming (software) provides this monitoring and control.

The data processing system consists of five general-purpose computers for computation and control; two magnetic-tape mass memories for large-volume bulk storage: time-shared, serial digital data busses (essentially party lines) to accommodate the data traffic between the computers and other orbiter systems; 19 multiplexer/demultiplexer units to convert and format data at various systems; three engine interface units to command the



Data Processing System

orbiter main engines; and four multi-function television (cathode ray tube–CRT) display systems so the crew can monitor and control the vehicle and payload systems.

The software stored in and executed by the orbiter computers is the most sophisticated and complex set of programs ever developed for aerospace use. The programs are written to accommodate almost every aspect of the Space Shuttle vehicle operations including vehicle checkout at the Rockwell Palmdale (CA) assembly facility, pre-launch and final countdown, turnaround activity at the Kennedy Space Center, and control during ascent, on-orbit, entry, and landing and abort or other contingency mission phases.

In-flight programs monitor the status of vehicle systems; provide consumable computations; control the opening and closing of the payload bay doors; operate the remote manipulator system; perform fault detection and annunciation, provide for payload monitoring, commanding, control, and data acquisition; provide antenna pointing for the various communications systems; and provide primary and backup guidance, navigation, and control for ascent, on-orbit, entry, landing, and abort.

These computer programs are written so they can be executed by a single computer or by all computers executing an identical program in the same time frame. The multicomputer mode is used for critical phases such as launch, ascent, entry, and abort.

The orbiter software for a major mission phase must fit into the 106,496-word central memory of each computer. Each computer consists of a central processor unit (CPU) and an input/output processor (IOP). The CPU performs the arithmetic and logical processing of data, provides control and handling of interrupts and program control of its corresponding IOP, and manages redundant systems such as sensors. The memory capacity of each computer CPU is 81,920 words. All data transmissions between the computers and vehicle systems are performed by the IOP under control of the CPU. The IOP receives data from the CPU, formats it, and relays commands to vehicle systems. The IOP also receives data from the vehicle systems and formats it for the CPU. The memory capacity of each computer IOP is 24,576 words.

General-Process Computer (IOP and CPU)

To accomplish all of the computing functions for all mission phases, approximately 400,000 words of computer memory are required. To fit the software needed into the computer memory space available, computer programs have been subdivided into nine memory groups corresponding to functions executed during specific flight and checkout phases. As an example, one memory group accommodates final countdown, ascent, and aborts; another on-orbit operations; and another the entry and landing computations. Different memory groups support checkout and ground turnaround operations and system management functions. Thus, in addition to central memory stored in the computers themselves, 34,000,000 bytes of information can be stored in two mass memories.

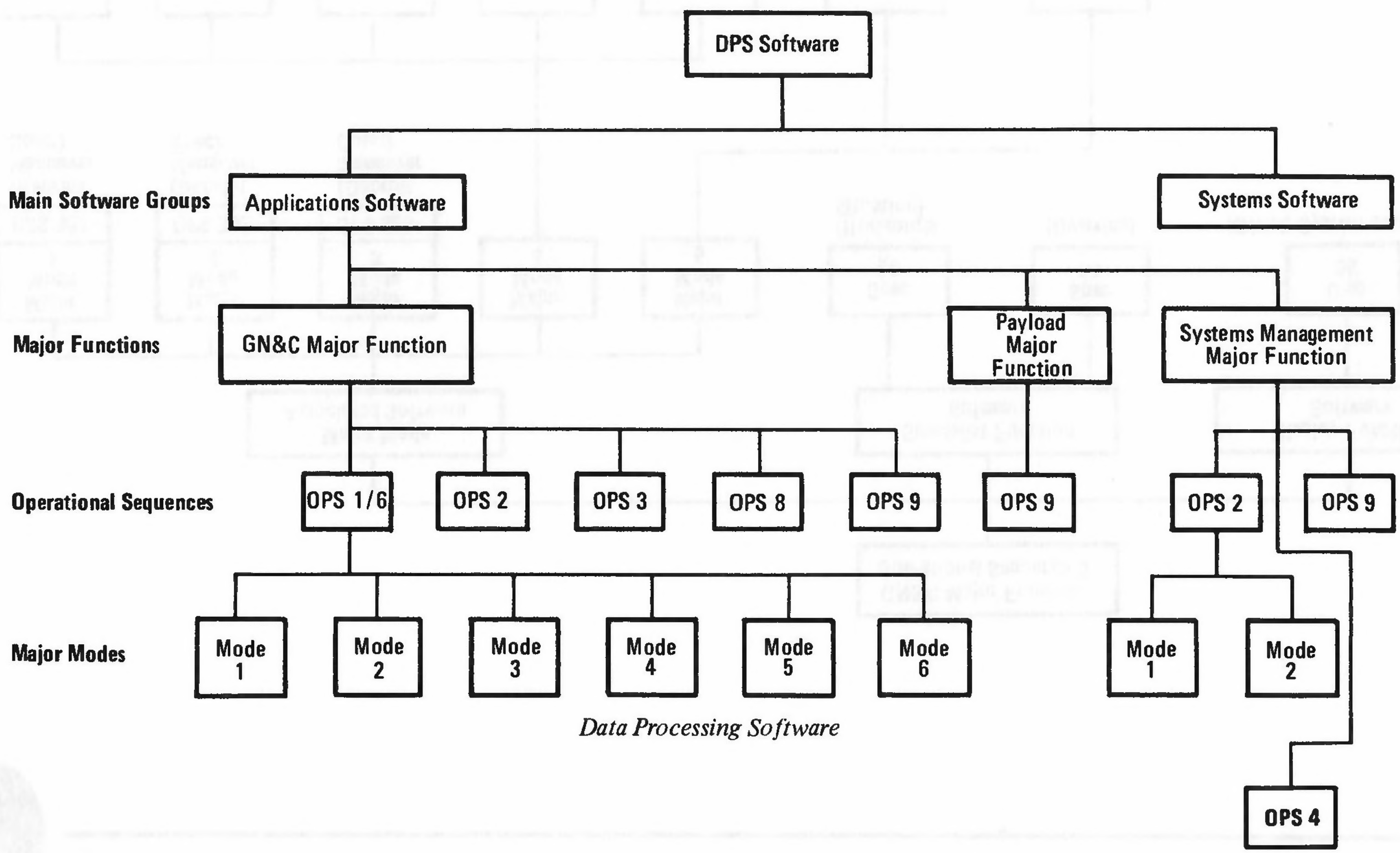
The orbiter computers are loaded with different memory groups from magnetic tapes containing the desired program. In this way all the software needed can be stored in mass memory units (magnetic tape machine) and loaded into the computers only when actually needed. Critical programs and data are loaded in both mass memories and protected from erasure. Normally, one mass memory is activated for use and the other is held in reserve. However, it is possible to use both simultaneously on separate data busses or communicating with separate computers. The data stored in the mass memories include prelaunch and preflight test routines. fault isolation diagnostic test programs, cathode ray tube (CRT) display formats, overlay program segments to be loaded during specific mission phases, and duplicate copies of resident on-line programs for initial loading, reloading, or reconfiguration of the computers. The mass memories are an advanced form of data storage and fill the gap between slow access drives of high storage capacity and discs or drums with fast access but relatively low storage capacity. In contrast to disc or drum memories, the mass memories consume power only when active.

The DPS software is divided into two major groups, called system software group and applications processing software.

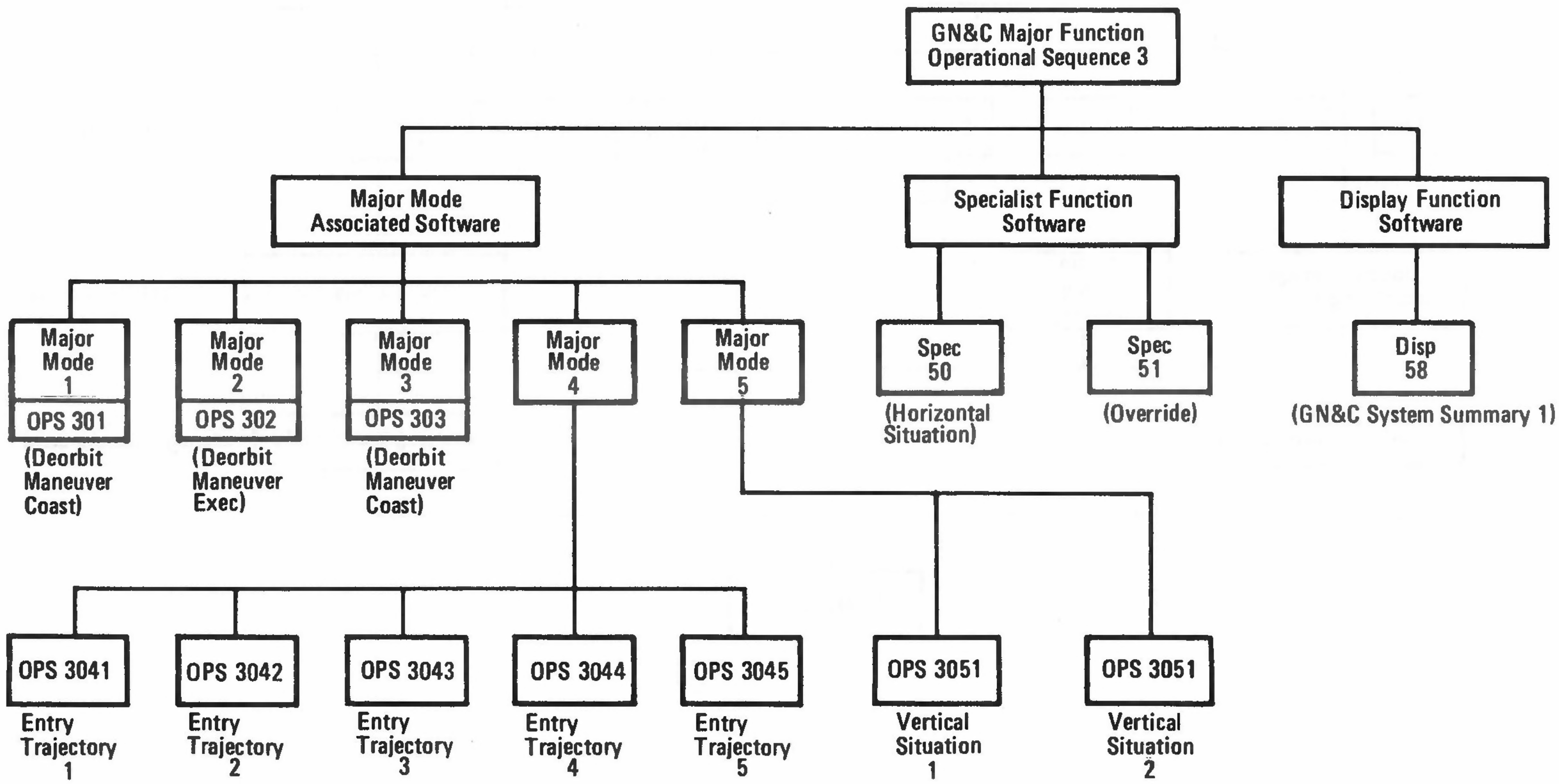
At the top is the system software group, which consists of three sets of programs: the flight computer operating program (the executive), which controls the processors, monitors key system parameters, allocates computer resources, provides for orderly program interrupts for higher priority activities, and updates computer memory; the user interface programs, which provide instructions for processing crew commands or requests; and the system control program, which initializes each computer and arranges for the multicomputer operation during flight-critical periods. The system software group programs tell the computers how to perform and how to communicate with other equipment.

The second level of memory groups is the applications processing software. This group contains (1) specific software programs for guidance, navigation, and control which are required for launch, ascent flight to orbit, maneuvering on orbit, entry and landing on a runway; (2) systems management programs which contain instructions for loading memories in the main engine computers and for checking the instrumentation system in addition to aiding in vehicle subsystem checkout and in ascertaining that crew displays and controls perform properly and update the inertial measuring unit state vectors; (3) payload processing programs which contain instructions for control and monitoring of orbiter payload systems which can be revised depending on the nature of the payload; and (4) vehicle checkout programs which are required to handle data management, performance monitoring, and special processing and display and control processing.

The two software program groups are combined to form a memory configuration for a specific mission phase. The software programs are written in HAL/S (high-order assembly language/ Shuttle) especially developed for use in real-time space applications. These programs are grouped by function and partitioned into memory configurations. When requested, memory is reconfigured from mass memory so operating sequences for the needed function can be overlaid into the main computer memory.



Data Processing Software



Operational Sequence Associated Software

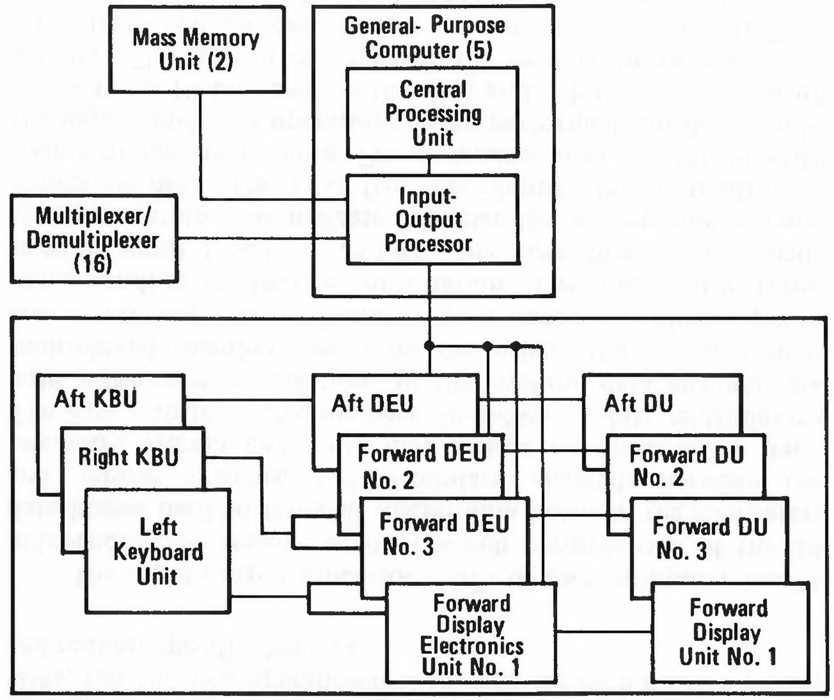
The highest level of the applications software is the OPS (operational sequences) which is required to perform part of a mission phase. Each OPS is a set of unique software required to perform phase-oriented tasks. An OPS can be further subdivided into groups called major modes, each representing a portion of the OPS mission phase. As an example, the launch phase (OPS-1) is subdivided into six major modes.

Each major mode has with it an associated CRT display which provides the flight crew with information concerning the current portion of the mission phase. The display function of OPS software presents a fixed format of data and configuration status on a CRT, which is not subject to flight crew manipulation and is used only to provide the flight crew with information.

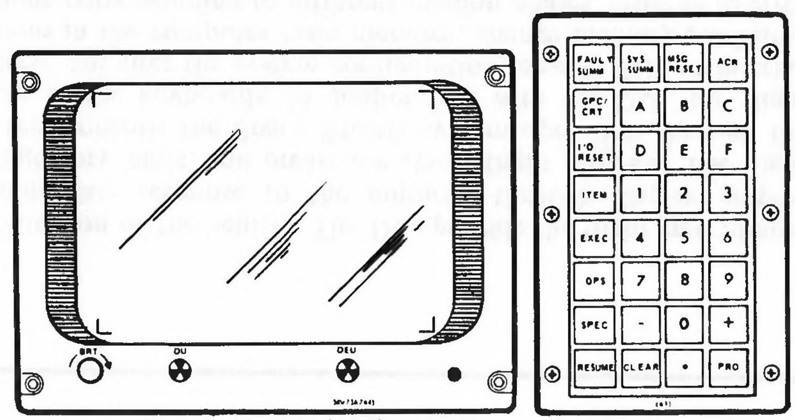
The specialist (SPEC) function of the OPS software is a block of software associated with one or more OPS which has an associated CRT display and enables the flight crew to monitor and manipulate the vehicle systems via item keyboard entries.

The multifunction CRT display system provides the flight crew with the ability to interface with and control the onboard software, observe vehicle system data, and monitor error or fault messages. The system is composed of three components: display electronics units (DEU), keyboard units (KBU), and display units (DU), which include the CRTs.

The flight crew has two keyboards, on the left and right sides of the flight deck display and control center console. There are three DU-CRTs on the flight deck forward display and control console. Each CRT is 12x17 centimeters (5x7 inches). There is also one keyboard and one DU-CRT at the aft side station flight deck display and control console. The three DU-CRTs at the forward console are connected to each of the forward center console keyboards. The DU-CRT at the aft station is connected to the keyboard at that station.



Data Processing System Hardware



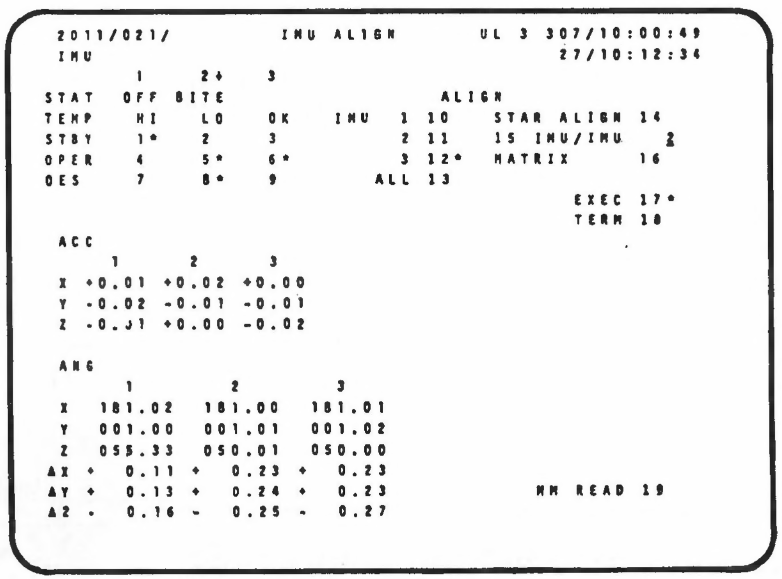
Cathode Ray Tube (CRT) **Keyboard**

Cathode Ray Tube and Keyboard

The DU uses a magnetic-deflected electrostatic-focused CRT. When supplied with deflection signals and video input, the CRT will display alphanumeric and graphic information. Characters can be flashed and the CRT brightness varied for individual characters. The CRT has a single-color (green) phosphor.

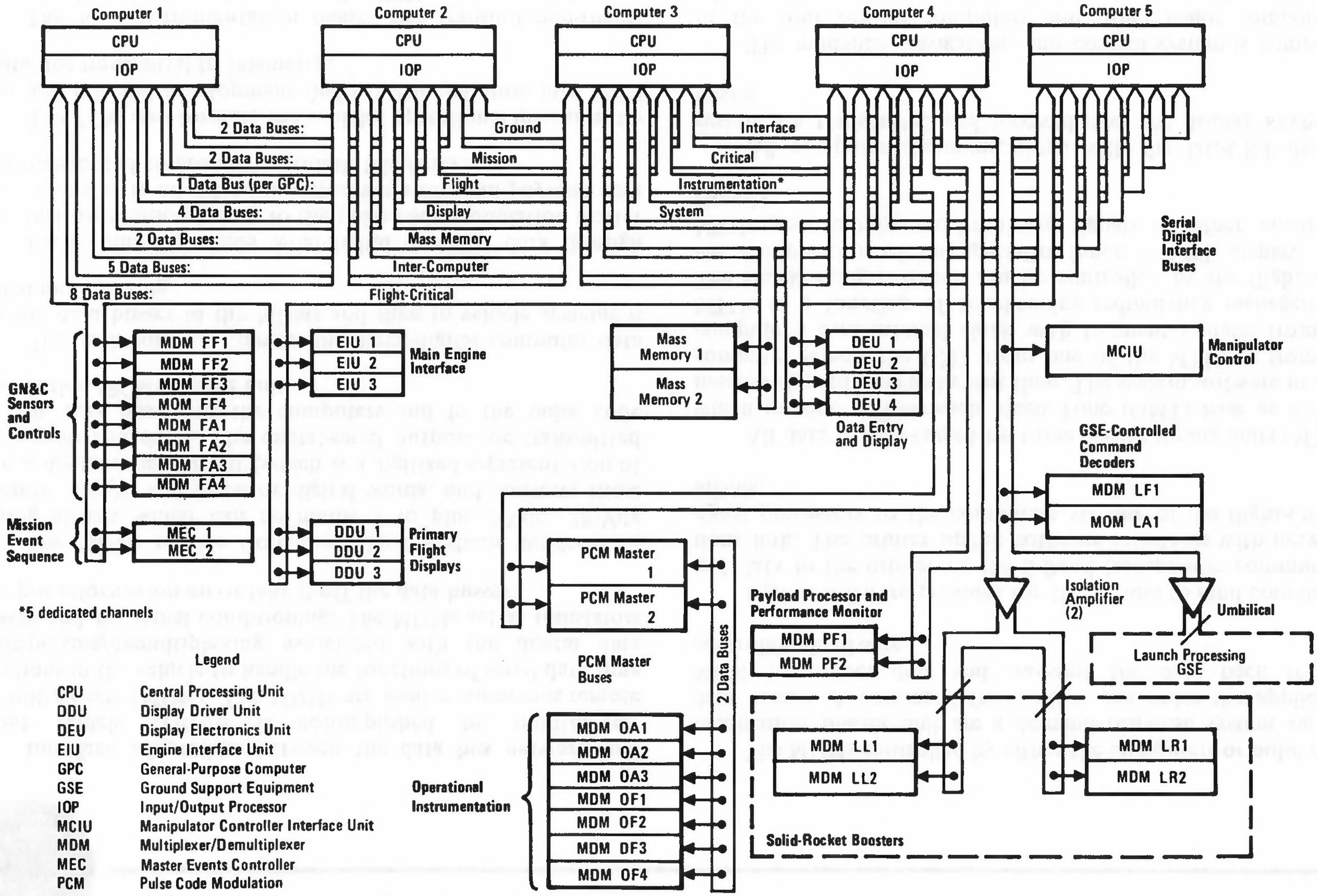
The four DEUs provide storage of display data, the computer/keyboard unit and computer/display unit interface display generation, updating, and refreshing, keyboard entry error checking, and keyboard entry echoing to the DUs.

The three keyboard units provide the crew with a controlling interface for software operations and management. Each keyboard has 32 momentary-contact (pushbutton) function and numeric keys. Using these keys, the flight crew can ask the computer more than 1,000 questions about the flight and condition of the vehicle. The DUs provide the flight crew almost immediate response to the inquiries through display graphs, trajectory plots, and prediction about flight progress. The flight crew controls the Space Shuttle system operation through the use of the keyboards. In conjunction with the DUs, the flight crew can alter the system configuration, change data or instructions in the computer main memory, change memory configurations corresponding to different mission phases, respond to error messages and alarms, request special programs to perform specific tasks, run through operational sequences for each mission phase, and request specific displays.



Cathode Ray Tube Display

The input-output processor (IOP) of each computer has 24 independent processors, each of which controls one of the 24 data busses used to transmit digital data between the computers and vehicle systems, and secondary channels between the telemetry system and units that collect instrumentation data. The data transfer technique uses time-division data multiplexing with pulse-code modulation. In this system, data channels are multiplexed together, one after the other, and information is coded on any given channel by a series of binary pulses corresponding to discrete information. The information transmission word length is 28 bits. The first three bits provide synchronization and indicate whether the information is commands or data. The next five bits identify the destination or source of the information. For command words, 19 bits identify the data transfer or operations to be performed; for data words, 16 of these 19 bits contain the data and 3 bits define the word validity. The last bit of each word format is for an odd parity error test. The 24 data busses are connected to each IOP via multiplexer interface adapters (MIAs) which receive, convert, and validate the serial data in response to discrete signals calling for available data to be transmitted or received. The data busses are organized into seven groups: intercomputer communication, display keyboard, flight critical, mass memory, payload launch/boost, and instrumentation.



Data Processing System

Interface adaptation between the data bus network and most vehicle systems is accomplished by multiplexer/demultiplexers (MDM). The MDMs are used in numerous remote locations in the vehicle to handle the functions of serial data time multiplexing/demultiplexing associated with the digital data busses and for signal conditioning. The MDMs act as translators and put information on or take it off the data busses.

The MDMs receive from the vehicle systems hundreds of analog signals which can be minus 5 to plus 5 Vdc, 28-Vdc discrete signals, and serial or digital words, and converts these into a digital-serial output (which is a digitized representation of the signals and data). The digital-serial outputs are transmitted via the data busses to the computers and to the pulse code modulation (PCM) master unit.

The technique of transferring serial-digital computer data via the data busses to the MDMs and then to vehicle systems is called multiplexing.

Each computer sends serial-digital downlist data through four instrumentation busses to the pulse code modulation master unit, where it is mixed with instrumentation and payload data and transmitted to ground downlink telemetry.

The PCM also formats the vehicles operational instrumentation and selected development flight instrumentation into serial digital for transmittal to telemetry.

The four instrumentation busses also transmit non-flight-critical orbiter system data from the PCM master unit to each computer for display on the flight crew CRTs. The PCM master unit contains a programmable read-only memory (PROM) for accessing subsystem data, a random-access memory (RAM) in which to store system data, and a memory in which data from the computers are stored for incorporation in the downlink telemetry.

The MDMs controlled by either the computers or pulse code modulation master unit are a demand response system via the data busses. A command from either can order the applicable MDM to collect data and transmit the data back to the controlling hardware.

Uplink software provides for the ground to send commands and data to the orbiter via the S-Band transponder communications link. The orbiter uplink software interfaces with network signal processors to the computers via one of the flight-critical MDMs.

All data is time-tagged by three master timing units (MTU), which provide a Greenwich Mean Time (GMT) base as well as mission elapsed time and event time. The system software in each computer selects the GMT from one of the MTUs or from the computer's own internal clock with frequent updates from the MTUs as a function of timekeeping redundancy management. The timekeeping software can be controlled by the flight crew via manipulation of the specialist function CRT display. The MTUs also supply synchronizing signals to other electronic circuits.

All computer communications with the DU-CRT display system are transmitted and received over the display keyboard busses.

The guidance, navigation, and control system is composed of the four orbiter computers and other major components which make up the primary flight control system. The computers use a program called the digital autopilot to control the vehicle through launch, ascent, on-orbit, deorbit, entry, and landing. The guidance, navigation, and control system provides automatic or manual (control stick steering) control of the vehicle in all flight phases. During launch most of the computer commands are directed to gimbal the main engines and solid rocket boosters. To circularize the orbit, in orbit, and for deorbit, the computer directs the orbital maneuvering system. At external tank separation, in orbit, and during a portion of entry, vehicle attitude control commands are directed to the reaction control system. In atmospheric flight the computers direct the orbiter aerodynamic flight control surfaces.

During critical mission phases (launch and entry), four of the computers are assigned to perform GN&C tasks, operating as a cooperative redundant set. One computer acts as a commander of a given data bus in the flight control scheme and initiates all bus transactions. The noncommander computers on the same bus listen to all incoming data that the commander requests. Thus, each response to a request by any computer is heard by all performing the same redundant operations and verified for consistently identical output. The computer redundancy management software module centers around the concept that each computer compares its outputs with the other computers in the set. If the comparison disagrees, this disagreement is displayed to the flight crew as a CRT message; however, processing continues.

Each of the computers operating in a redundant set operates in synchronized steps and cross-checks results of processing about 440 times per second. If a computer operating in a redundant set fails to meet the synchronization requirements for redundant set operations, it would be removed from the set. Each computer performs about 325,000 operations per second during critical phases of the mission.

As an example, Computers 1 through 4 are operating in a redundant set when Computer 1 fails to stay synchronized with the other three. Computer 1 software recognizes the disagreement and also recognizes that it has been voted failed by the other three computers. Computer 1, therefore, sets a self-fail vote and does not vote the other three computers failed.

All intercomputer communications other than synchronization are transmitted and received over four specific buses. Cross-strapping of the four buses to the four computers allows each computer access to the status of the data received or transmitted by the other computers, making possible the verification of identical results among the four computers. The four computers are loaded with the same software programs. Each bus is assigned to one of the four computers in the command mode and the remaining computers operate in the listen mode for that bus. Each computer has the ability to receive data with the other three computers, pass data to the others, request data from the others, and perform any other tasks required to operate the redundant set. No vehicle systems are connected to these buses.

The flight-critical buses are directed into groups of four to be compatible with the grouping of the four computers. Commands to flight deck crew flight control system (dedicated) displays and the forward GN&C system as well as the data from the forward GN&C sensors are transmitted and received over one group of buses (flight-critical FC-1 through FC-4). The data commands to the aft GN&C system, as well as the data from the aft GN&C components, are transmitted and received over FC-5 through FC-8 buses. Each bus in a group is assigned to a separate computer operating in a command mode. The computer in the command mode issues data requests and commands to the applicable vehicle systems over its assigned FC (dedicated) bus. The remaining three buses in each group are assigned to the remaining computers to operate in the listen mode. A computer operating in the listen mode can only receive data. Thus, if Computer 1 operates in the command mode on bus FC-1, it listens on the three remaining buses. In this manner, each computer commands on one bus of a flight-critical group and listens on the remaining three.

Each flight-critical bus in a group of four is commanded by a different computer. There are multiple units of each GN&C hardware item (sensors, controllers, flight control effector) and each unit is wired to a different MDM and flight-critical bus. The

MDM and bus can be assigned to another computer. The flight computer operating system in systems software in each of the redundant set computers activates a GN&C executive program and issues commands to the bus and MDM to provide a set of input data. Each MDM receives the command from the computer assigned to it, acquires the requested data from the GN&C hardware wired to it, and sends the data to all four computers.

When the sets of GN&C hardware data arrive at the computers via the MDMs and data buses, the data is generally not in the proper format, units, or form for use by flight control, guidance, or navigation. A subsystem operating program for each type of hardware processes the data to make it usable by GN&C software. These programs contain the software necessary for hardware operation, activation, self-testing, and moding. The level of redundancy varies from two to four depending on the particular unit. The software which processes data from the redundant GN&C hardware is called redundancy management. This performs two functions: selects, from redundant sets of hardware data, one set of data for use by flight control, guidance, and navigation; and detects data which is out of tolerance, identifies the faulty unit, and announces the failure to the flight crew and to the data collection software.

In the case of four hardware units, the redundancy management software utilizes three and holds the fourth in reserve and utilizes a middle value select until one of the three is bad, then uses the fourth. If one of the remaining three is lost, it would downmode to two and use the average of the two. If one of the remaining two were lost it would downmode to one and pass the only data it receives.

The three engine interface units between the computers and the three main engine controllers accept computer main engine commands, reformat them, and transfer them to each main engine controller. In return, the engine interface units accept data from the main engine controller, reformat it, and transfer it to computers and operational instrumentation. Main engine functions such as ignition, gimbaling, throttling, and shutdown are controlled by the main engine controller internally through inputs from the guidance equations which are computed in the orbiter computers.

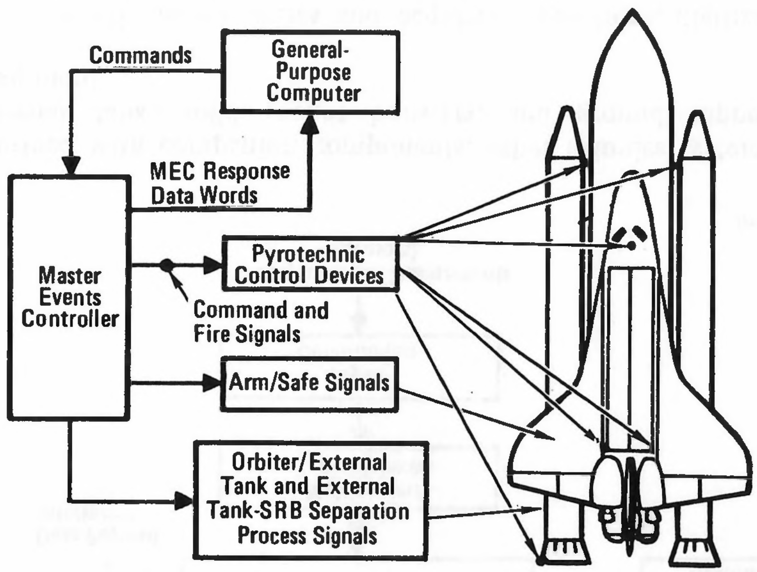
During non-critical flight periods in orbit, only one or two computers are used for GN&C tasks and another for payload operations and system management. The remaining three can be used either for payload management or deactivated on standby.

The fifth onboard computer is used as a GN&C backup in the initial development flights; however, it also provides unique functions such as system non-critical function monitoring and payload command and monitoring. The fifth computer has a separate independent software design and coding activity to protect against generic software failures in the primary computer set.

The payload in the orbiter may have up to five safety-critical status parameters hardwired, so that these parameters and others can be recorded as a part of the orbiter's system management which is transmitted and received over the two payload buses. To accommodate the various forms of payload data, the payload data interleaver integrates payload data into the orbiter avionics so it can be transmitted to ground telemetry.

The two master events controllers under computer control provide signals for arming and safing pyrotechnics, and for command and fire signals for pyrotechnics in separation processes.

Data bus isolation amplifiers are the interfacing device among GSE, the solid rocket booster MDMs, and the orbiter launch data bus. They transmit or receive multiplexed data in



Master Events Controller

any direction. The amplifiers enable multiplexed communications over the longer data bus cables which connect the orbiter and GSE. The receiving section of the amplifiers detects low-level coded signals, discriminates against noise, and decodes the signal to standard digital data at very low bit error rate; the transmit section of the amplifiers then re-encodes the data and retransmits it at full amplitude and low noise.

Data bus couplers couple the vehicle multiplexed data and control signals between the data bus and cable studs connected to the various electronic units. The couplers also provide impedance matching on the data bus, line termination, dc isolation, and noise rejection.

Each CPU is 19.05 centimeters (7-1/2 inches) high, 25.70 centimeters (10-1/8 inches) wide, and 49.53 centimeters (19-1/2 inches) long and weighs 25.85 kilograms (57 pounds). The IOPs are identical in size and weight to the CPUs.

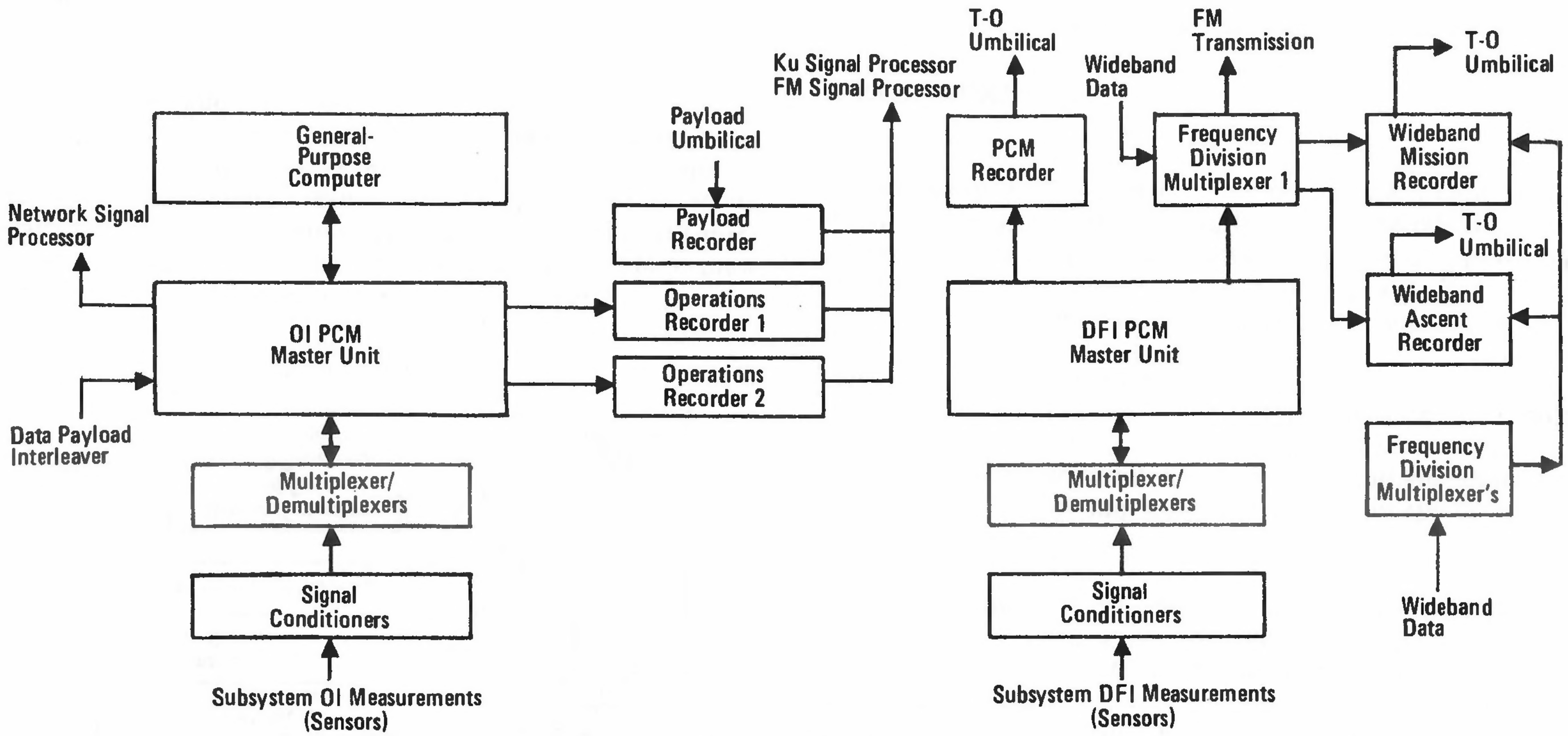
Each of the two mass memories is 19.05 centimeters (7-1/2 inches) high, 29.21 centimeters (11-1/2 inches) wide, and 38.1 centimeters (15 inches) long and weighs 9.97 kilograms (22 pounds). The MDMs are 33 by 25.4 by 17.78 centimeters (13 by 10 by 7 inches) and weigh 16.64 kilograms (36.7 pounds) each. The data bus isolation amplifiers are each 17.78 by 15.24 by 12.7 centimeters (7 by 6 by 5 inches) and weigh 3.4 kilograms (7-1/2 pounds). Each data bus coupler is 16.39 cubic centimeters (one cubic inch) in size and weighs less than 28 grams (one ounce).

The five CPUs and IOPs are located in the crew compartment mid deck avionics bays 1, 2, and 3 and are cooled by fans. The mass memories are located in the crew compartment mid deck avionics bays 2 and 3; each MM is mounted on a coldplate and cooled by a water loop.

The forward flight-critical MDMs are located in the crew compartment mid deck avionics bays 1, 2, and 3. They also are mounted on coldplates and cooled by a water loop. The aft flight-critical MDMs are located in the aft fuselage avionics bays, are mounted on coldplates, and cooled by Freon-21 coolant loops.

INSTRUMENTATION

The instrumentation system consists of transducers, signal conditioners, pulse code modulation encoding equipment, frequency multiplexing equipment, PCM recorders, analog recorders, timing equipment, and onboard checkout equipment. The instrumentation system is made up of two separable functional systems: operational instrumentation (OI) and development flight instrumentation (DFI). DFI will be used only for the initial development flights. The OI and DFI systems



Instrumentation

interface with companion components, other avionics systems, external tank, solid rocket boosters, and ground support equipment.

The OI system senses and acquires, conditions, digitizes, formats, and distributes data for display, telemetry, recording, and checkout. It provides for PCM recording, voice recording, and master timing for onboard systems. The equipment consists of two pulse code modulation master units (PCMMU's), two operations recorders, master timing unit, and various multiplexer/demultiplexers (MDM's), signal conditioners, and sensors.

The DFI system provides additional instrumentation for the initial development flights. This system senses and acquires, conditions, digitizes, formats, frequency-multiplexes, distributes, and records data. Its equipment consists of two PCMMU's, three recorders, nine frequency division multiplexers, and various MDM's, signal conditioners, and sensors.

The dedicated signal conditioners (DSC's) provide inputs to the OI and DFI from such transducer signals as frequency, voltage, current, pressure, temperature (variable resistance and thermocouple), and displacement (potentiometer), 28- and 5-volt dc discretes. The signal conditioners convert their input signals to

Area	Characteristics
Signal Conditioning	•13 DSC's conditioning approximately 1200 channels approximately 100 DFI measurements) •6 WBSC to FDM for development flights, then to MDM for operational flights •Approximately 45 high-level transducers •Distributes data to PCM data system, panel displays, C-W, flight-critical MDM's
PCM Data System	•Acquires data through 7 OI MDM's and 1 DFI MDM, approximately 3250 measurements, of which 450 are DFI •Receives and provides data to 5 computers •Accepts payload data through PDI or payload data bus (Spacelab) •Provides output data in 64 and 128-kbps formats •Provides synchronization to PDI and NSP •Provides data to NSP and T-0 umbilical
Payload Data	•Receives data directly from payload or PLSP •Provides data to GPC's and PCM
Recorders	•Stores 3 channels of engine data at 60-kbps rate •Stores interleaved voice and data at 96, 128, or 192-kbps rate •Stores payload data •Provides for playback of recorded data during and after mission
Master Timing	•Provides time reference to computers, OI/DFI PCM, DFI FDM's, display panel, and payload •Provides synchronization to instrumentation and other subsystems •Provides onboard reference for doppler measurements

OI System

Area	Characteristics
Signal Conditioning	•Approximately 2040 dedicated signal conditioning channels •Approximately 700 strain gauge channels •Approximately 670 wideband channels (includes 135 strain) •Approximately 150 transducers (high-level) •Data distributed to digital PCM and wideband FDM data systems
Digital PCM Data System	•Digitizes and transmits approximately 2570 (low-frequency) measurements •Data acquired through 6 DFI MDM's •Data transmitted via hardline (ground checkout) and telemetry link at 128 kbps •Data stored on PCM recorder
Wideband FDM Data System	•Accepts approximately 525 wideband (high-frequency) measurements for multiplexing and recording •Ascent system has 105 launch-associated measurements on all flights and approximately 280 measurements shared over three sharing programs •Provides capability via hardline (ground checkout) to 35 multiplexed outputs
Wideband Recorders	•Ascent recorder stores 7 multiplexed orbiter channels, 5 ET multiplexed channels, and 2 SRB multiplexed channels for ground playback via umbilical hardline •Mission recorder stores 28 multiplexed orbiter channels for ground playback via umbilical hardline

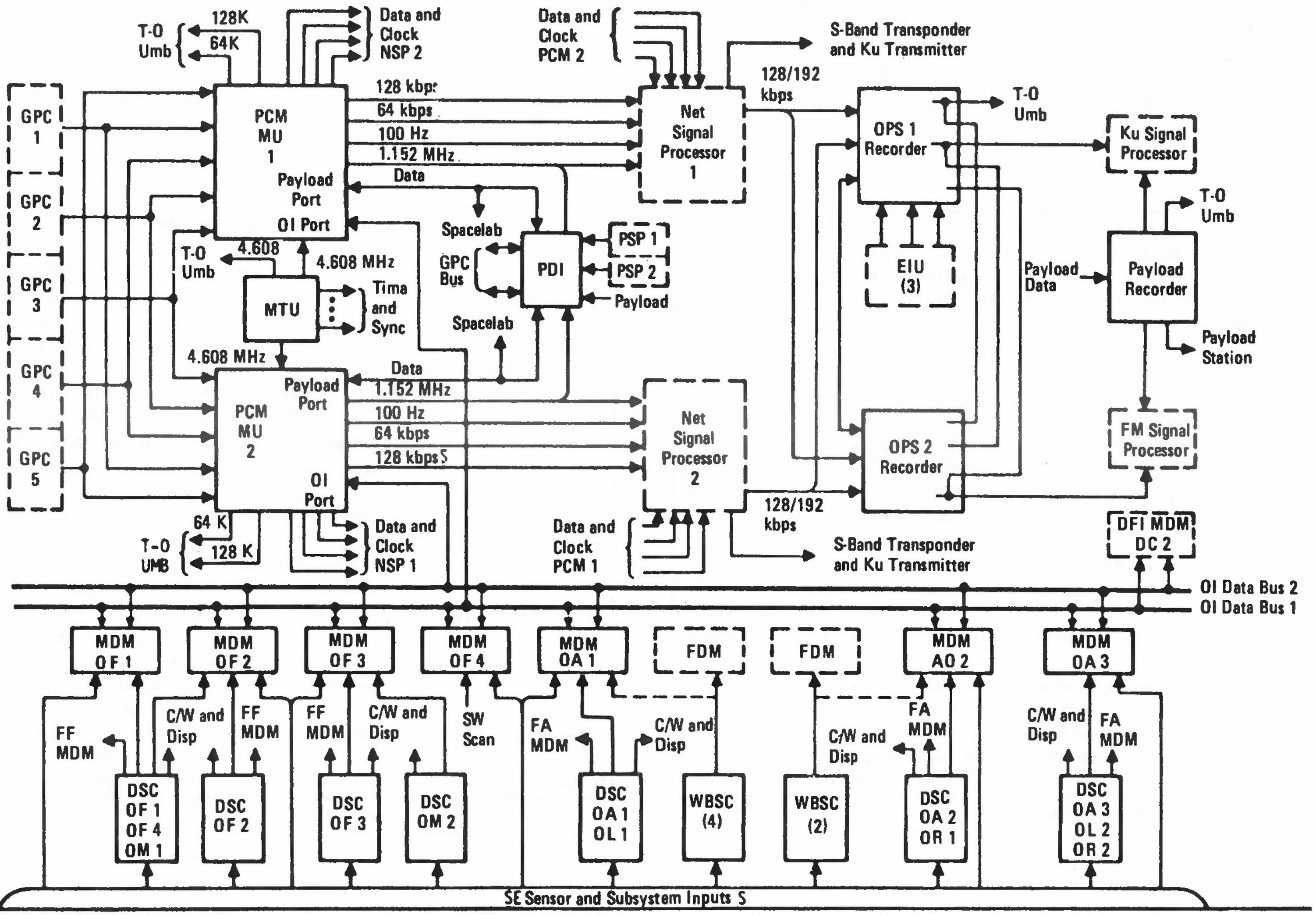
DFI System

an analog signal of 0 to 5 volts dc or to a 28- or 5-volt dc discrete output signal.

The output signals of the OI DSC's are directed to the flight deck crew displays, C/W system, and a corresponding MDM. The MDM's convert the analog signal to serial digital data (a digitized representation of the applied voltage). The MDM's send this serial digital data to a PCMMU upon request through the OI data buses. When the MDM is addressed by the PCMMU, the MDM will select, digitize, and send the requested data to the PCMMU in serial digital form.

The OI PCMMU receives digital data from the OI MDM's, in addition to computer downlist data from the onboard computers, and combines them to form the PCM telemetry for the S-band downlink. The PCMMU controls the data received from the MDM's; downlist data from the computers is under the control of the flight software. All data received by the PCMMU is stored in memory and periodically updated. The PCMMU also sends data to the onboard computers on request.

The OI PCMMU has two formatter memories: programmable read only and random access (RAM). The former is



Legend

- C/W – Caution Warning
- DFI – Development Flight Instrumentation
- DISP – Display
- DSC – Dedicated Signal Conditioner
- EIU – Engine Interface Unit
- FDM – Frequency Division Multiplexer
- GPC – General-Purpose Computer
- MU – Master Timing Unit
- WBSC – Wideband Signal Conditioner

OI System

programmed only before launch; the latter is reprogrammed several times during flight. The PCMMU will use the format memories to downlink data from the computers and data from the OI MDM's into high-rate (128 kbps) and low-rate (64 kbps) PCM telemetry data streams. These data streams are sent to the network signal processor, which then sends the 128-kbps data to the operations recorders and either the 128-kbps or 64-kbps data to the S-band transponder for transmission to the ground. In later flights, the Ku-band transmitter will be used when the orbiter is in orbit. On the ground this data is recorded again and also monitored in real time at the Mission Control Center, Houston.

Only one of the redundant OI PCMMU's and network signal processor's operates at a time. The ones used are controlled by the crew through the flight deck display and control panel.

On later flights, a payload data interleaver will accept data simultaneously from up to five attached and one detached payload, interleave it, and send it to the PCMMU. The input data is in serial digital data streams. Data temporarily stored in the PCMMU memory can be accessed by the PCMMU telemetry formatter and by the onboard computers. The payload data interleaver is programmed on board from mass memory via the computers to select specific data from each payload PCM signal and store it within its buffer memory locations.

The OI PCMMU's receive a synchronization clock signal from the master timing unit. If this signal is not present, the PCMMU provides its own timing and continues to provide timing signals to the payload data interleaver and network signal processor.

There are many different PCM telemetry formats that control the measurement groupings the PCMMU assembles into the 128-kbps and 64-kbps data streams for different mission phases. Those to be used for each mission are stored in the onboard computer's mass memory. When the ground or flight crew want to change the format, a command is sent to the computers, which will then load the desired format from mass memory into the formatter RAM of the PCMMU.

The five onboard computers are capable of data processing. They perform programmed computations and then prepare the data for telemetry transmission by means of a downlist. Four of the computers are assigned to the primary flight control system in the orbiter and provide primary downlist data. The fifth computer is assigned to the backup flight control system in the initial development flights and to systems management and payload management in later flights and provides downlist data. The computer downlist data consists of a table of values compiled in the computer. The format can be changed by ground command or by the flight crew. The downlist data is sent from the computers to the PCMMU, where it is combined with operational instrumentation data to form the PCM downlink data stream.

The network signal processor receives the PCMMU 128-kbps and 64-kbps telemetry data streams and also one or two analog voice channels from the orbiter audio central control unit. The processor converts the analog voice signals to digital voice signals, time-division multiplexes them with the PCM telemetry data, and sends the composite signal to the S-band transponder for downlink transmission. The total data rate of the composite signal will be either 192 kbps (high rate) or 96 kbps (low rate). The 128-kbps telemetry plus two 32-kbps voice channels equal the 192-kbps total. The 64-kbps telemetry plus one 32-kbps voice channel equal 96 kbps total. The downlink rate from the network signal processor can be switched by ground command at any point in the data cycle. The processor also outputs to the operations recorders either the 128-kbps telemetry or the 192-kbps composite signal. Only one processor operates at a time.

The DFI dedicated signal conditioners' output signals are directed to the DFI multiplexer-demultiplexer, which convert them into serial digital data and send them via the DFI data buses to the DFI PCMMU. The DFI PCMMU formats the data into a 128-kbps data stream and outputs it to a DFI PCM recorder and to a special frequency division multiplexer. The latter is used to transmit the data to ground stations on the DFI S-band FM transmitter and the S-band downlink. On the ground, this data is recorded again and selected parameters are sent to the Mission Control Center in Houston for real-time monitoring.

The DFI PCMMU also receives timing from the master timing unit; however, it will provide its own timing if the signal is not present. Only one of the redundant DFI PCMMU's operates at a time, the one in use selected by the flight crew.

Wideband signal conditioners provide input signals to the DFI MDM's, accommodate low level accelerometer, vibration, and acoustic signals, convert them, and send an analog output signal of 0 to 5 volts dc to the applicable frequency division multiplexer.

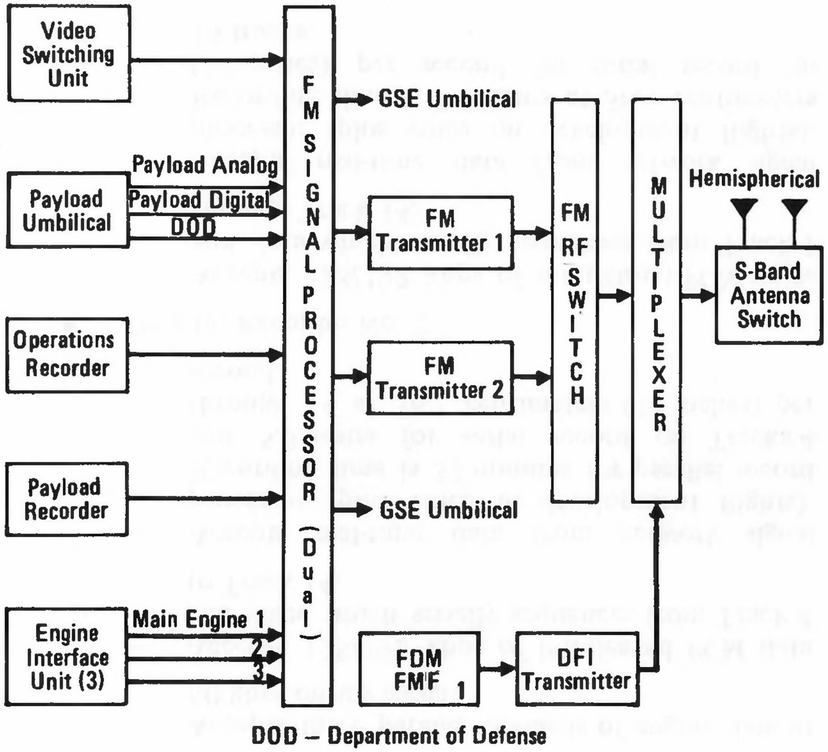
The strain gauge signal conditioners in the DFI accommodate low level signals from the strain gauges, convert them, and provide an analog output signal of 0 to 5 volts dc. Selected strain guage analog signals are sent to a corresponding MDM which converts the analog signal into a digital serial output. The digital serial output is received by the DFI data buses and transmitted to the DFI PCMMU which formats the data into the 128 kbps data stream. Other strain gauge signal conditioners send an analog signal of 0 to 5 volts dc to a corresponding FDM.

The nine DFI MDM's frequency multiplex the analog data for wideband recording to the wideband ascent recorder and wideband mission recorder. The FDM FMF-1 receives 128-kbps digital PCM data from the DFI PCMMU as well as 15 selected analog inputs from various signal conditioners. FDM FMF-1 sends this multiplexed composite signal to the DFI S-band FM transmitter for transmission to the ground stations on the S-band DFI FM downlink. This composite is also sent to the ascent and mission recorders.

MASTER TIMING UNIT. The master timing unit is a stable crystal-controlled timing source for the orbiter. It provides serial time reference signals to the onboard computers, PCMMU's, FDM's, and various time display panels. It also provides synchronization to instrumentation and other systems. It includes separate time accumulators for Greenwich mean time (GMT) and mission elapsed time (MET), which can be reset or updated from the ground via uplink through the onboard computer or by the flight crew through the use of their flight deck display and control panel keyboard and CRT (cathode ray tube) time displays.

The signal flows from the 4.608-MHz oscillators to the output of the GMT and MET accumulators. The three independent GMT and three independent MET counters operate simultaneously. Separate time accumulators are used for each GMT and MET clock, and they accumulate time in days, hours, minutes, seconds, and milliseconds. The GMT capability is 366 days, 23 hours, 59 minutes, 59 seconds, and 999.875 milliseconds. For MET, the capability is 365 days, 23 hours, 59 minutes, 59 seconds, and 999.875 milliseconds. Both can be updated and reset by ground equipment before flight or from the onboard controls by the flight crew. During flight, the GMT and MET accumulators are updated at a predetermined time by uplink and onboard computer or by voice command and entered through the flight deck display and control panel keyboard and CRT display.

FREQUENCY MODULATION SIGNAL PROCESSOR. The FM signal processor receives inputs and processes data from three sources: the three engine interface units, the video switching unit, and the recorder dump from the operations recorders. It is



Frequency Modulation Signal Processor

commanded to select one of these sources at a time for output to the S-band FM transmitter for transmission to the ground stations on the S-band FM downlink. The output is then either three channels of main engine data at 60 kbps each, real-time television from the onboard TV cameras, or operations recorder dumps (128-kbps PCM telemetry), 192-kbps composite data, or one channel of main engine data at 60 kbps.

OPERATIONS RECORDERS. There are two of these recorders (also referred to as maintenance/loop recorders) used for serial recording and dumping of digital voice and PCM data. The recorders normally are controlled by ground command, but they can be commanded by the flight crew through the flight deck display and control panel keyboard or by switches on a recorder panel. Input to the recorders is from the network signal processor, either 128-kbps PCM data or a 192-kbps composite signal which includes the 128-kbps PCM data and two 32-kbps voice channels. The network signal processor receives the PCM data from the OI PCMMU and the voice signals from the audio control center. In addition, operations recorder No. 1 receives three channels of main engine data at 60 kbps during ascent.

In the maintenance/loop mode, one is selected as a maintenance recorder while the other is designated a loop recorder. The latter records the serial output of the OI PCMMU continually for temporary storage. This data is recorded in loop fashion by switching sequentially through tracks 12, 13, and 14. The loop recorder also will record digital data from the three engine interface units at 60 kbps during ascent, but not simultaneously with the loop function. The maintenance recorder is used for permanent storage of two types of data: anomaly data from the loop recorder dump, and quick-look PCM data. When the 11th track of the 14 data tracks on the maintenance recorder is filled, it sends a signal to alert the user to interchange the loop and maintenance functions between the two recorders. The functions of recorders can be interchanged by uplink commands via the network signal processor or by the flight crew.

The operations recorders can be commanded to dump recorded data from one recorder while continuing to record real-time data on the other. The dump data is sent to the FM signal processor for transmission to the ground station via the S-band FM transmitter on the S-band FM downlink. When the ground has verified that the data they received is valid, the operations recorders can use that part of the tape to record new data.

A single recorder can store and reproduce digital and analog data both singly and in combination at many rates.

The single recorder function will be used for the first development flight; however, both recorders may be operated in a maintenance/loop function.

Recorder functions can be summarized as follows:

- Data In, Recorder No. 1:
  - Accepts three parallel channels of engine data at 60 kbps during ascent
  - Accepts 128/192 kbps of interleaved PCM data and voice which serially sequences from Track 4 to Track 14.
  - Accepts real-time data from network signal processor (plus voice in development flights). Recording time is 32 minutes for parallel record and 5.8 hours for serial record on Tracks 4 through 14 at 38.1 centimeters (15 inches) per second.
- Data In, Recorder No. 2:
  - Accepts 128/192 kbps of interleaved PCM voice and data which serially sequences from Track 1 through Track 14.
  - Accepts real-time data from network signal processor (plus voice on development flights). Recording time is 7.5 hours at 38.1 centimeters (15 inches) per second for serial record on 14 tracks.
- Data Out, Recorder No. 1:
  - In flight payback of engine interface unit data and network signal processor digital data via S-band FM transponder and in later flights via Ku-band transmitter.
  - In operational flights, in-flight playback of anomaly PCM data for maintenance recording; playback of data serially to GSE T-O umbilical.
- Data Out, Recorder No. 2:
  - In-flight playback of digital data via S-band FM transponder and in later flights via Ku-band transponder.
  - In operational flights, in-flight playback of anomaly PCM data for maintenance recording; playback of data serially to GSE T-O umbilical.
- Recorder Control:
  - Manual control from mission specialist flight deck aft station display and control panel. Uplink and onboard computer keyboard control.
  - Recorder speecs of 19.05, 38.1, 60.96, and 304.8 centimeters (7.5, 15, 24, and 120 inches) per second provided by hardware programs plug direct command.

The tape recorders contain a minimum of 731 meters (2,400 feet). of 1.27-centimeter (0.5-inch) by 1-mil magnetic tape. They operate at 60.96 centimeters (24 inches) per second in the record mode, and 304.8 centimeters (120 inches) per second in the playback mode.

PAYLOAD RECORDER. The payload recorder is not used in the initial development flights, although it will be on board. It is identical to the operations recorders in hardware and will be used to record payload data and dump in flight via the S-band transponder or in later flights the Ku-band transmitter.

The payload and recorder capabilities are:

- Data In:

- Accepts digital inputs of 64 kbps either serial or parallel up to 14 tracks.
- Accepts analog data from 1.9 kHz to 2 MHz either serial or parallel up to 14 tracks.
- Serial/parallel track programming is determined by premission payload distribution panel wiring.
- Record time is 32 minutes for parallel record or 7.5 hours for serial record at 38.1 centimeters (15 inches) per second.

• Data Out:

- In-flight playback of digital data via S-band transponder or in later flights via Ku-band transmitter.
- In-flight playback (analog/digital) data to payload distribution panel; playback of data to GSE via T-O umbilical.

• Control

- Manual control from mission specialist flight deck aft station display and control panel.
- Uplink and computer keyboard by computers. Recorder speeds of 38.1, 76.2, 152.4, and 304.8 centimeters (15, 30, 60, and 120 inches) per second provided by hardware program plug. Recorder hardwired for continuous run modes.

DFI RECORDERS. Three recorders are used in the DFI system to provide PCM and wideband recording: PCM, wideband and ascent, and wideband mission recorders. The operations of these recorders can be controlled only by the flight crew through the use of switches on the flight deck display and control panels. The DFI recorders cannot be dumped in flight; they are played back via the GSE T-O umbilical after the orbiter has landed.

The PCM recorder is used to record 128-kbps PCM data from the DFI PCMMU.

The two wideband recorders are used to record the output of the frequency division multiplexers. The wideband ascent recorder will operate during ascent only.

It will record either continuously or in one of two automatically timed intervals selected by the crew. The two intervals are a high-sample mode, which automatically turns the recorder on for 10 seconds, off for five minutes, on again for 10 seconds, etc., and a low-sample mode, which automatcially turns the recorder on for 10 seconds, off for 10 minutes, etc. The purpose of this switching is to save tape. The recorder has 14 tracks which are recorded serially, and each track will fill up in 32 minutes. This means that the recorder can be operated for a total period of about 7-1/2 hours.

All its tracks record in parallel. It will operate for only 32 minutes before all the tracks become full. When the recorder reaches the end of tape, it will shut itself off automatically and will not be used again during the flight.

The wideband mission recorder can be operated only in a continuous mode or shut off by a switch on the flight deck display and control panel. It can be operated only for a total of two hours before all the tracks become full.

EQUIPMENT LOCATION. Instrumentation equipment, except for sensors and selected dedicated signal conditioners, are located in the forward and aft avionics bays. The DFI equipment is installed in the aft avionics bays and special containers located in the forward fuselage mid-deck crew compartment and the mid fuselage. Sensors and dedicated signal conditioners are located throughout the orbiter in areas selected on the basis of accessibility, minimum harness requirements, and functional requirements. Effective use of remote data acquisition techniques was considered for optimizing equipment location. The factors which were considered in determining equipment location were weight, power, physical size, redundancy, and wire density and length to each compartment and interconnect wiring.

The abbreviation OA refers to operational, aft, OF to operational forward, OL to operational left, OR to operational right, OM to operational mid, DF to development forward, DL to development left, DR to development right, FMF to frequency multiplexer forward, FMC to frequency multiplexer center, FMR to frequency multiplexer right, DL to development left, DR to development right, and DC to development center.

COMMUNICATIONS

The S-band system is the primary means of communications between the orbiter and the ground tracking stations during the ascent, orbital, and entry phases of the flight.

There is one uplink from the ground tracking stations to the orbiter, which is phase-modulated (PM) and provides commands, voice, and a coherent ranging signal to the orbiter.

(1) **Forward Avionics Bay 1**
- DSC
- OI Data MDM
- PCM MU 1
- Payload recorder
- Payload data interleaver

(2) **Forward Avionics Bay 2**
- DSC
- OI data MDM
- PCM MU 2
- Operations Recorder 1
- Operations Recorder 2

(3) **Forward Avionics Bay 3A**
- DSC
- OI data MDM

(4) **Forward Avionics Bay 3B**
- Master timing unit

(5) **Forward RCS DSC**

(6) **Flight Deck MDM**

(7) **Aft Avionics Bay 4**
- DSC
- OI data MDM

(8) **Aft Avionics Bay 5**
- DSC
- OI data MDM

(9) **Aft Avionics Bay 6**
- DSC
- OI data MDM

(10) **OMS DSC's (4)**

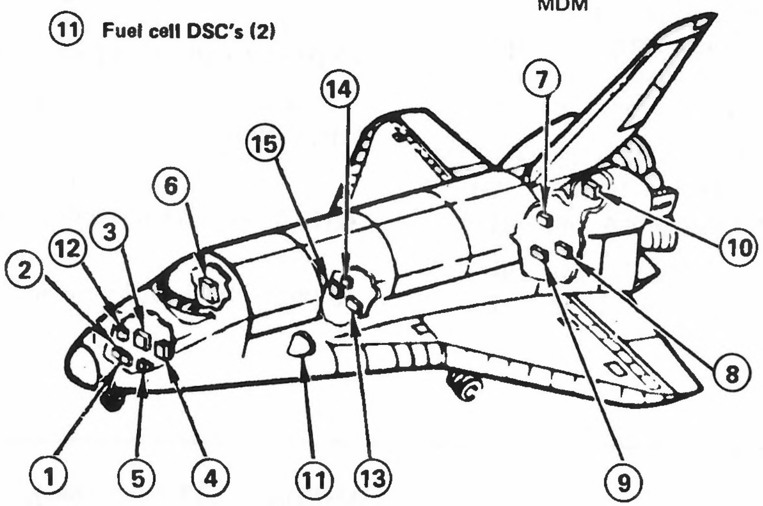
(11) **Fuel cell DSC's (2)**

(12) **Forward DFI Container**
- 3 FDM's
- PCM recorder
- Mission recorder
- Ascent recorder
- PCM MU's
- DSC's Assemblies
- 4 Wideband SCA's
- S/G SCA
- MDM

(13) **Mid Body DFI Container 1**
- 2 FDM's
- 5 DSC's
- 2 S/G SCA's
- 4 Wideband SCA's
- 2 MDM's

(14) **Mid Body DFI Container 2**
- 2 FDM's
- 4 DSC's
- 2 S/G SCA's
- 4 Wideband SCA's
- 2 MDM's

(15) **Mid Body DFI Container 3**
- FDM's
- 5 DSC's
- 2 S/G SCA's
- 4 Wideband SCA's
- 2 MDM's



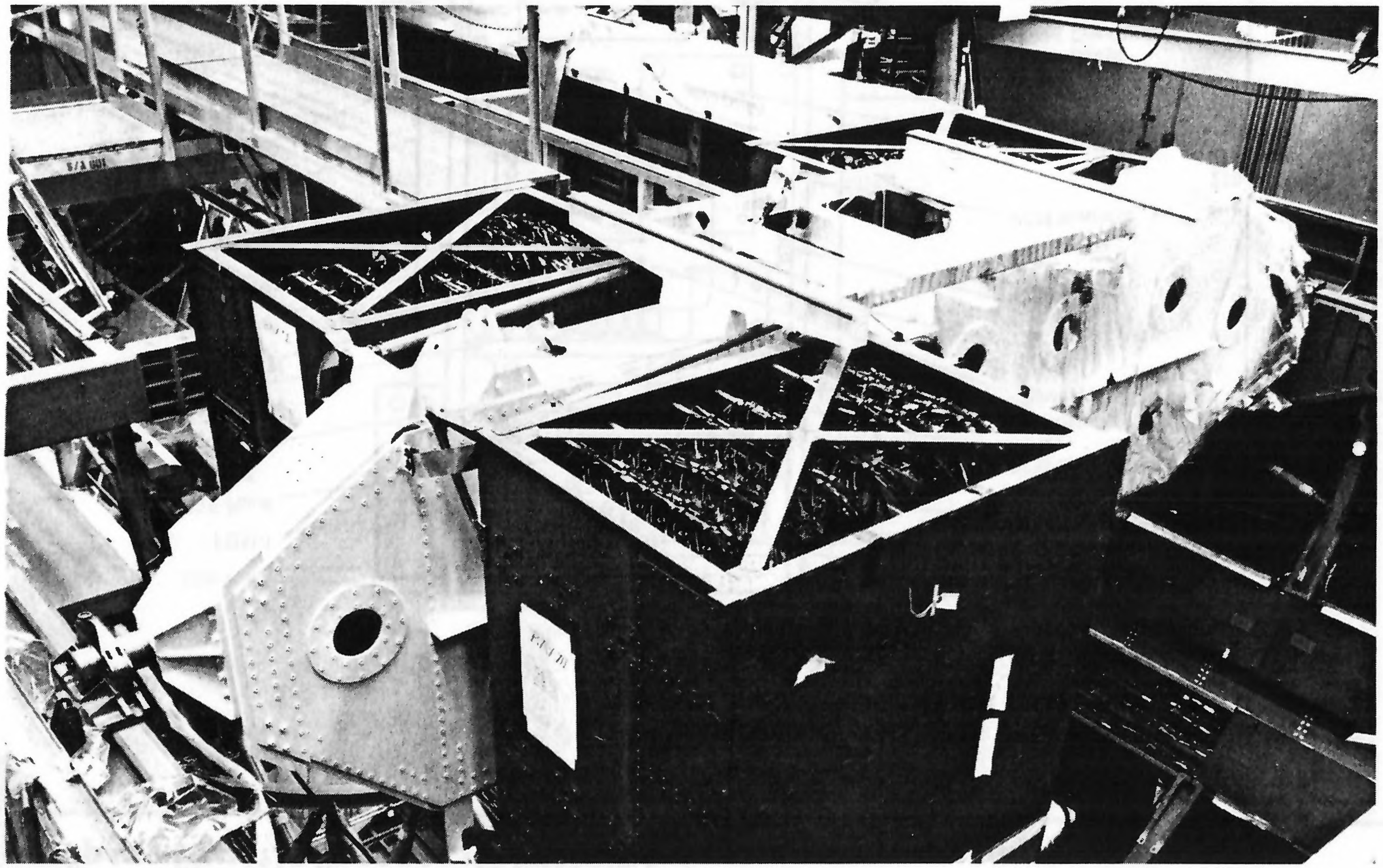
Legend

DSC	– Dedicated signal conditioner
FDM	– Frequency division multiplexer
MDM	– Multiplexer/demultiplexer
MU	– Master unit
SCA	– Signal conditioner assembly
S/G	– Strain gauge

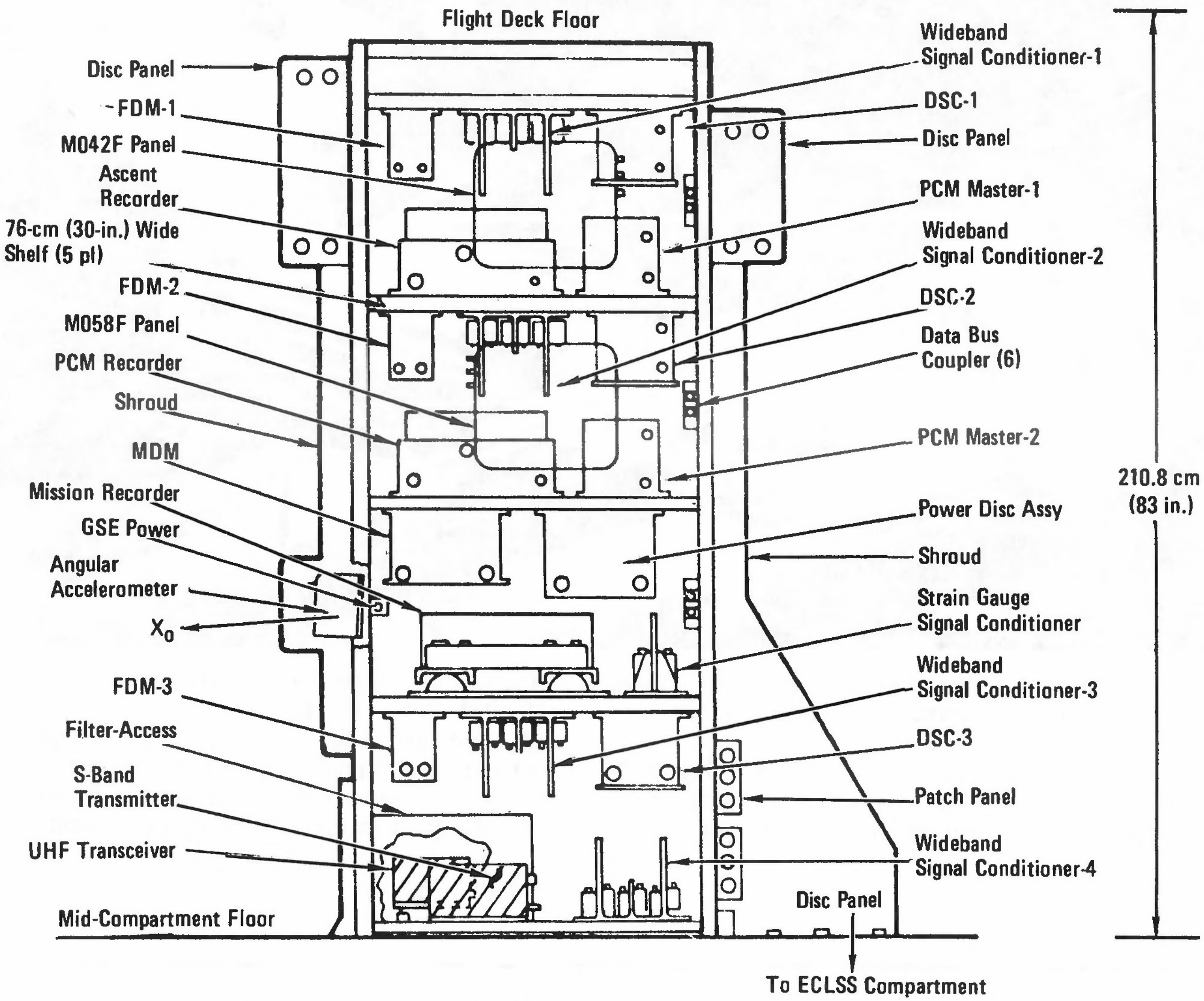
Instrumentation Equipment Location

There are three downlinks from the orbiter to the ground tracking stations. One is PM and provides voice, telemetry, two-way doppler, and a coherent ranging signal. The other two are frequency modulated (FM) because they are used for transmission of wideband data. One FM downlink provides operational real-time Space Shuttle main engine data during ascent, real-time video (television), or operations recorder dump data. The remaining FM downlink provides real-time development flight instrumentation (DFI) data.

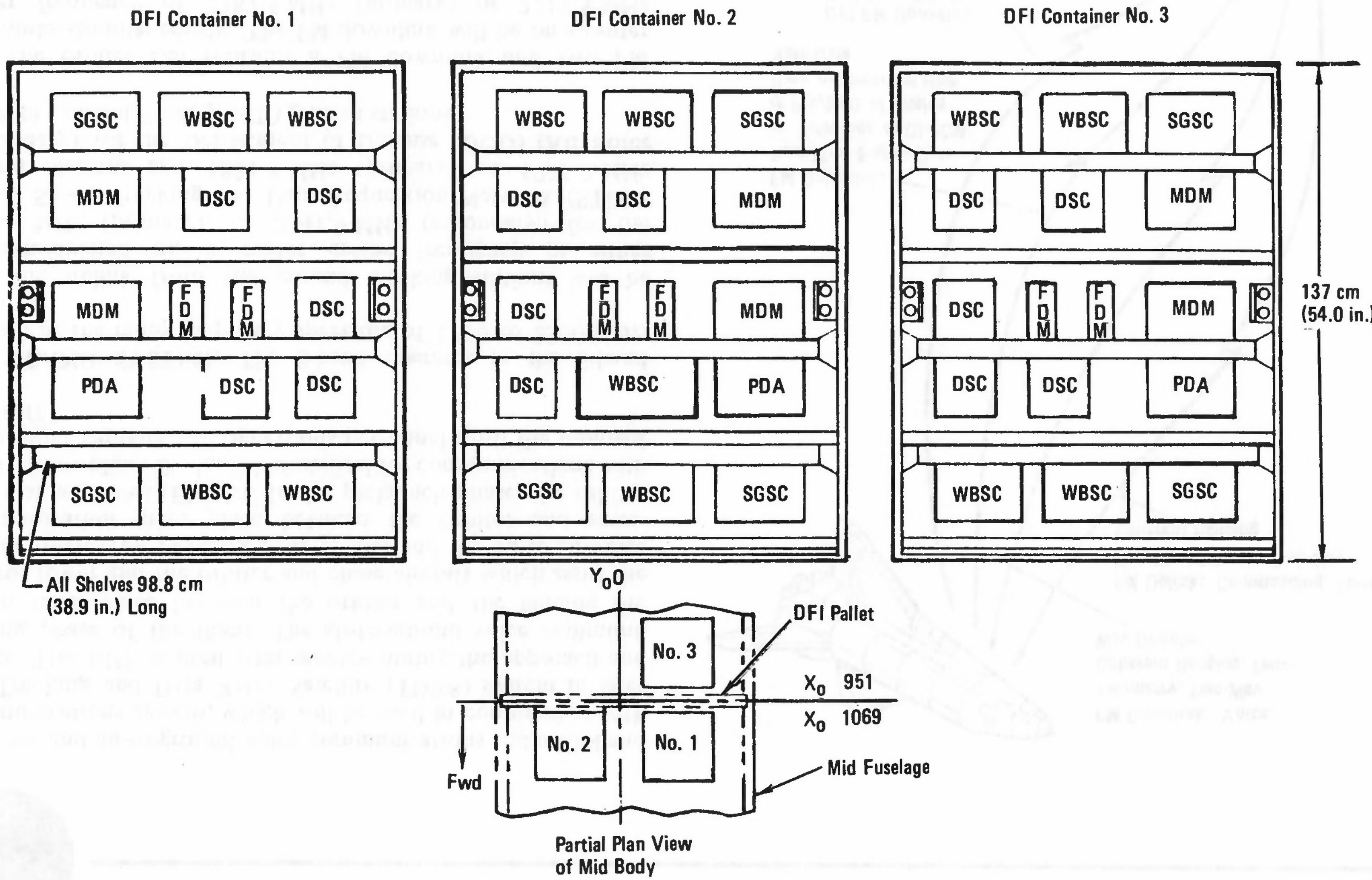
In addition to S-band, the orbiter has an ultra-high frequency (UHF) communication system which is used only for



Development Flight Instrumentation in Payload Bay



Forward DFI Container



Mid-Body DFI Containers

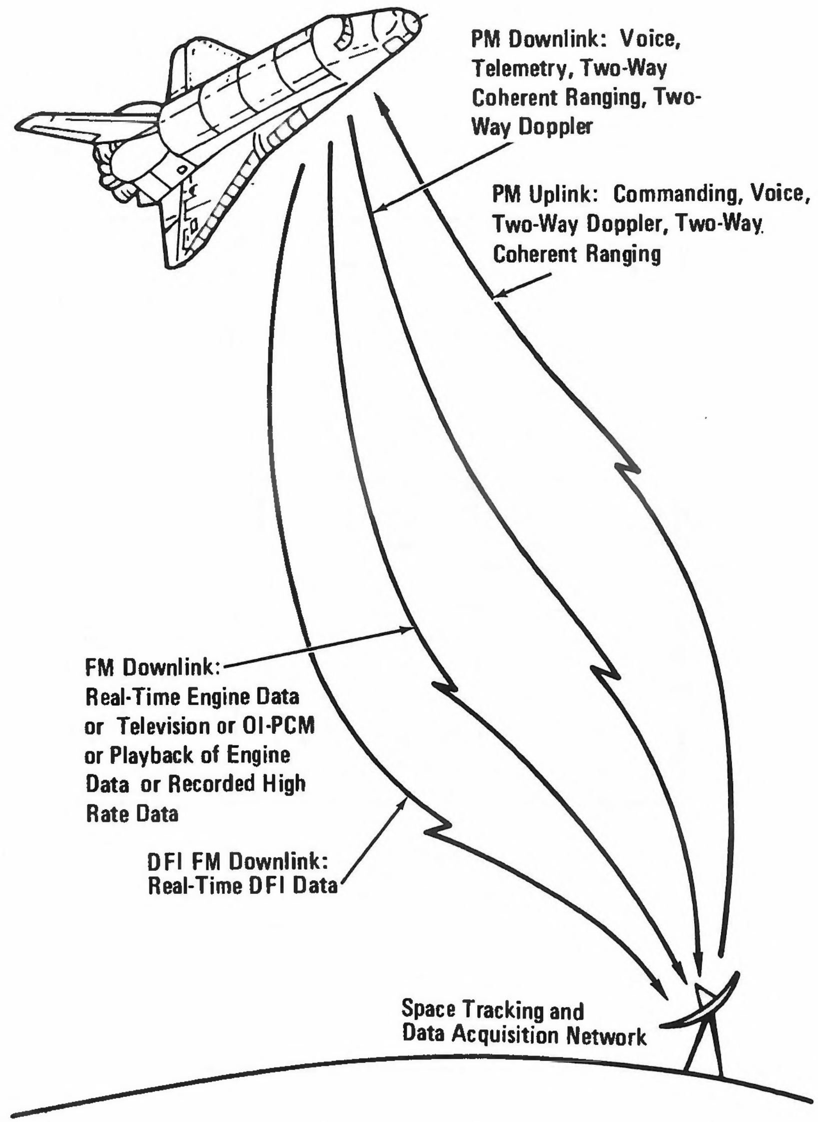
air-to-air and air-to-ground voice communications and a Ku-band communications system, which will be used in conjunction with the Tracking and Data Relay Satellite (TDRS) system in later flights. The UHF is used after reentry during the approach and landing phase of the flight. The air-to-ground voice communication takes place between the orbiter and the landing site control tower and the orbiter and chase aircraft which assist the orbiter flight crew in landing the orbiter and the air-to-air voice communication takes place between the orbiter and extra-vehicular activities (EVA's). In the prelaunch phase, the orbiter flight crew utilizes the launch umbilical for communications with the ground support and operations personnel until the moment of liftoff.

S-BAND SYSTEM. The S-band operates in the S-band portion of the radio frequency spectrum of 1700 to 2300 MHz.

The uplink from the ground tracking stations will be phase-modulated on a center carrier frequency of either 2106.4 MHz (primary), or 2041.9 MHz (secondary) for the NASA Space Tracking and Data Acquisition Network (STDN) ground stations and 1831.8 MHz (primary) or 1775.7 MHz (secondary) for the Department of Defense (DOD) (Air Force Satellite Control Facility-SCF) ground stations.

The orbiter can transmit a PM downlink and two FM downlinks simultaneously. The PM downlink will be on a center carrier frequency of 2287.5 MHz (primary) or 2217.5 MHz (secondary) for the SCF ground stations and 2287.5 MHz (primary) or 2217.5 MHz (secondary) for the NASA STDN ground tracking stations. The FM downlinks are on a center carrier frequency of 2250.0 MHz (FM operational transmitter) and 2205.0 MHz (DFI transmitter).

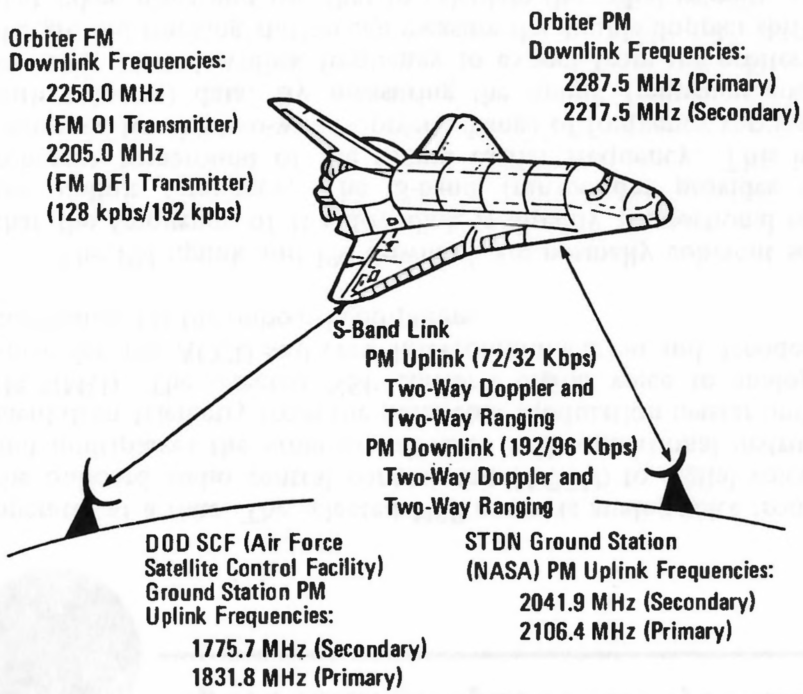
The PM uplink provides for commanding and voice transmissions from the ground stations to the orbiter. The PM downlink provides for telemetry and voice transmissions from th ›iter to ground stations.



Orbiter S-Band Communication Link

The S-band system is the only system for command reception until the Ku-band system is installed in later flights for use in conjunction with the TDRS. All MCC-H (Mission Control Center-Houston) uplink commands are multiplexed with digital voice and transmitted on the S-band PM uplink. The S-band system removes the voice and transfers the commands to the onboard orbiter computers. The computers route the commands to the intended systems.

The S-band PM uplink originates from an STDN remote station, NASA, or at an DOD SCF remote station. Each type of remote station has a choice of two uplink frequencies for data transfer. The frequency selection is under MCC-H control. The



S-Band Radio Frequencies

S-band uplink transfers the high data rate of two voice channels and commands (72 kbps–kilo bits per second) or low data rate of one voice channel and commands (32 kbps), two-way doppler, and two-way tone ranging.

The S-band PM downlink can originate from one of two S-band transponders aboard the orbiter. Each transponder can downlink on a frequency of 2217.5 MHz or 2287.5 MHz, but not both at the same time. The S-band PM downlink transfer high-data-rate telemetry plus two voice channels (192 kbps) or low-data-rate telemetry and once voice channel (96 kbps) and two-way doppler, and two-way ranging.

The S-band FM downlink can originate from three FM transmitters on board the orbiter. Two are tuned to 2250.0 MHz and one is tuned to 2205.0 MHz. The S-band FM downlinks transfer real-time engine data or television or operational instrumentation PCM (128 kbps) or playback of engine data or recorded high-rate data (192 kbps) and DFI (128 kbps).

Having two S-band uplink and downlink frequencies prevents interference when two Space Shuttle orbiters are operating at the same time. One Space Shuttle orbiter could select the high set of frequencies and the other could select the low set of frequencies.

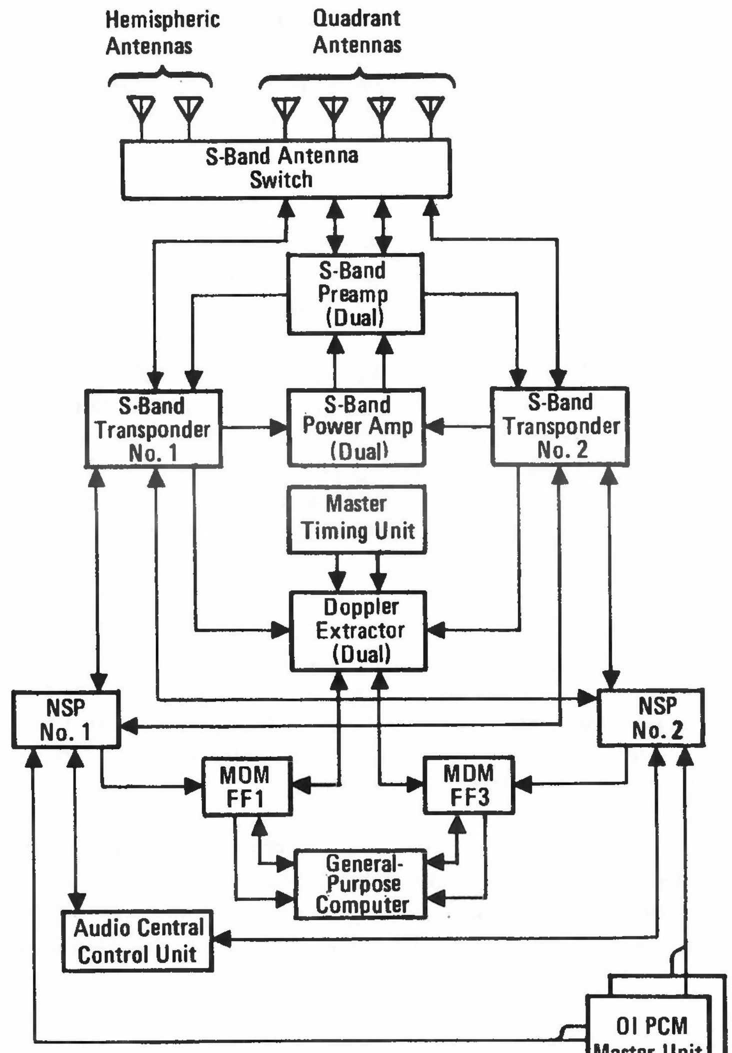
The two identical S-band transponders receive and transmit the PM uplink signal and downlink signal. Only one transponder operates at a given time; the other transponder is a redundant backup. The selected transponder transfers the uplink commands and voice to the NSP (network signal processor), receives the downlink telemetry and voice from the NSP, retransmits on a coherent frequency for the two-way doppler, and retransmits the two-way tone ranging signals.

The two onboard NSP's receive commands and transmit telemetry data to the selected S-band transponder. Only one NSP operates at a time. The selected NSP converts analog voice from the onboard audio central control unit (ACCU) to digital voice and multiplexes the same digital voice with operational instrumentation telemetry from the pulse code modulation master unit (PCMMU). The selected NSP converts digital voice to analog voice for the ACCU and crew intercommunication and decodes commands for the onboard computers.

The PM uplink and PM downlink are normally coherent so that the frequency of the downlink is directly proportional to the uplink frequency. The S-band transponder provides a coherent turnaround of the uplink carrier frequency. This is necessary for the two-way doppler (change of frequency varying with velocity) data. By measuring the uplink frequency and knowing what downlink frequency to expect from the orbiter, the ground tracking station can measure the double doppler shift that takes place and use that to calculate the radial velocity of the orbiter with respect to the ground station. These links are phase-modulated so that the S-band carrier center frequency will not be affected by the modulating wave. It would be impossible to obtain valid doppler data if the S-band carrier center frequency were affected by the modulating technique.

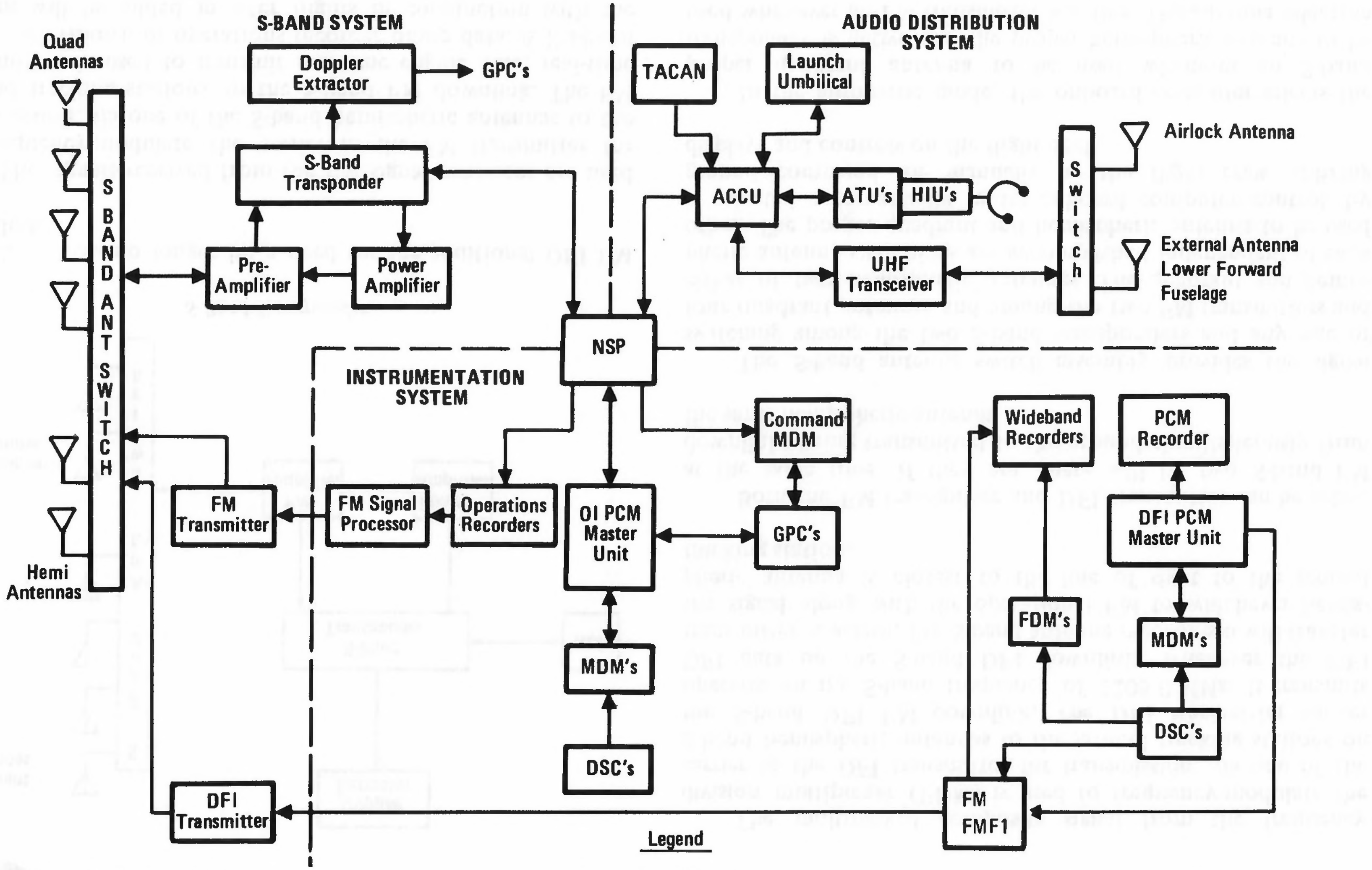
The S-band also provides a subcarrier for two-way tone ranging. The ground will uplink ranging tones at 1.7 MHz and compute vehicle slant range from the time delay in receiving the returned 1.7-MHz tones to determine orbiter range. Orbiter azimuth is determined from the ground tracking station antenna angles. A C-band skin tracking mode is also provided from the ground tracking stations to track the orbiter.

Frequency modulation is an ideal method to use for transmission of wideband data such as real-time video (television) and this is why the orbiter also has FM downlinks. There are two FM downlinks because it would be difficult to put all the wideband data on one link. After the initial development flights, the DFI equipment will be reduced greatly to a limited system



Legend:
MDM – Multiplexer/Demultiplexer
NSP – Network Signal Processor
OI – Operational Instrumentation
PCM – Pulse Code Modulation

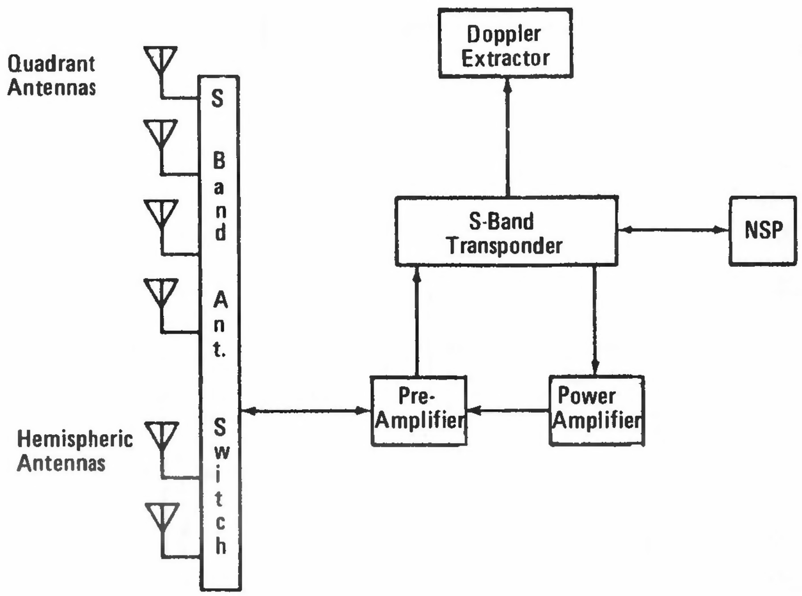
S-Band System



Legend

ACCU – Audio Central Control Unit
ATU – Audio Terminal Unit
DFI – Development Flight Instrumentation
DSC – Dedicated Signal Conditioner
FDM – Frequency Division Multiplexer
FM – Frequency Modulation
FMF – Frequency Multiplexer Forward
GPC – General-Purpose Computer
HIU – Headset Interface Unit
MDM – Multiplexer/Demultiplexer
NSP – Network Signal Processor
OI – Operational Instrumentation
PCM – Pulse Code Modulation
UHF – Ultra High Frequency

Communications System



S-Band Transponder

and there may no longer be a need for the additional DFI FM downlink.

The signals received from the FM signal processor are used to frequency-modulate the carrier in the FM transmitter for transmission, via one of the S-band hemispheric antennas to the ground tracking stations on the S-band FM downlink. The FM transmitter is used to transmit real-time engine data, real-time video (television), or operations recorder dump data. A Ku-band system will be added in later flights in conjunction with the TDRS system. This will provide an alternate system for television transmission. The frequency modulated trasmitter carrier operates on the S-band frequency of 2250.0 MHz. Whenever the FM transmitter is active, the S-band antenna mechanism will transfer the signal to whichever hemispheric antenna is closest to the most direct line of sight to the ground tracking station.

The multiplexed composite signal from the frequency division multiplexer (FDM) is used to frequency-modulate the carrier in the DFI transmitter for transmission, via one of the S-band hemispheric antennas to the ground tracking stations on the S-band DFI FM downlink. The DFI transmitter carrier operates on the S-band frequency of 2205.0 MHz. It transmits DFI data on the S-band DFI downlink. Whenever the DFI transmitter is active, the S-band antenna mechanism will transfer the signal along with the operational FM to whichever hemispheric antenna is closest to the line of sight to the ground tracking station.

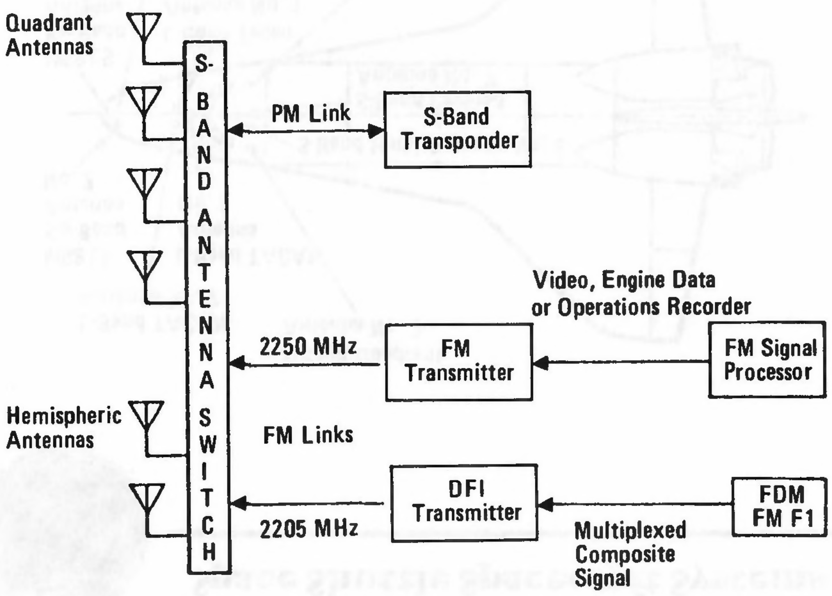
Both the FM transmitter and DFI transmitter can be active at the same time. If they are, there will be two S-band FM downlinks being transmitted to the ground simultaneously from the same hemispheric antenna.

The S-band antenna switch assembly provides the signal switching among the two S-band transponders and any one of four quadrant antennas and among the two FM transmitters and either of two hemispheric antennas. The quadrant and hemispheric antenna switchings are accomplished independent of each other. The proper quadrant and hemispheric antenna to be used is selected automatically under onboard computer control, by ground command, or manually by the flight crew utilizing displays and controls on the flight deck.

In the automatic mode, the onboard computer selects the proper quadrant antenna to be used whenever an S-band transponder is active and the proper hemispheric antenna to be used whenever an FM transmitter is active. The antenna selection is based upon the computed line of sight to the desired ground station. The antenna switching commands are sent to the switch assembly via multiplexer/demultiplexers (MDM's).

There are seven S-band antennas on the orbiter: four quadrant, two hemispherical, and one payload. The quadrant antennas are located on the forward fuselage on the outer skin approximately 90 degrees apart. In the orbiter on the flight deck looking out the front windows, the quadrant antennas are to the upper right, lower right, lower left, and upper left. The four antennas are the radiating elements for the S-band PM downlink and for receiving the S-band PM uplink. The two hemispherical antennas are located on the forward fuselage on the outer skin approximately 180 degrees apart. On the flight deck, the hemispherical antennas would be above the head (upper) and below the feet (lower). The two hemispherical antennas radiate the S-band downlink. All the antennas are covered with thermal protection system.

The basic difference between the two types of antennas is that the hemispheric antennas have a larger beam width, while the quadrant antennas have a highersantenna gain. The hemispheric antennas are so-called because there are two of them, one on top of the orbiter and one on the bottom. The quadrant antennas are so-called because there are four of them, two on each side of the orbiter, one on the upper half and one on the lower half of each side, which provides nearly total coverage in all directions.

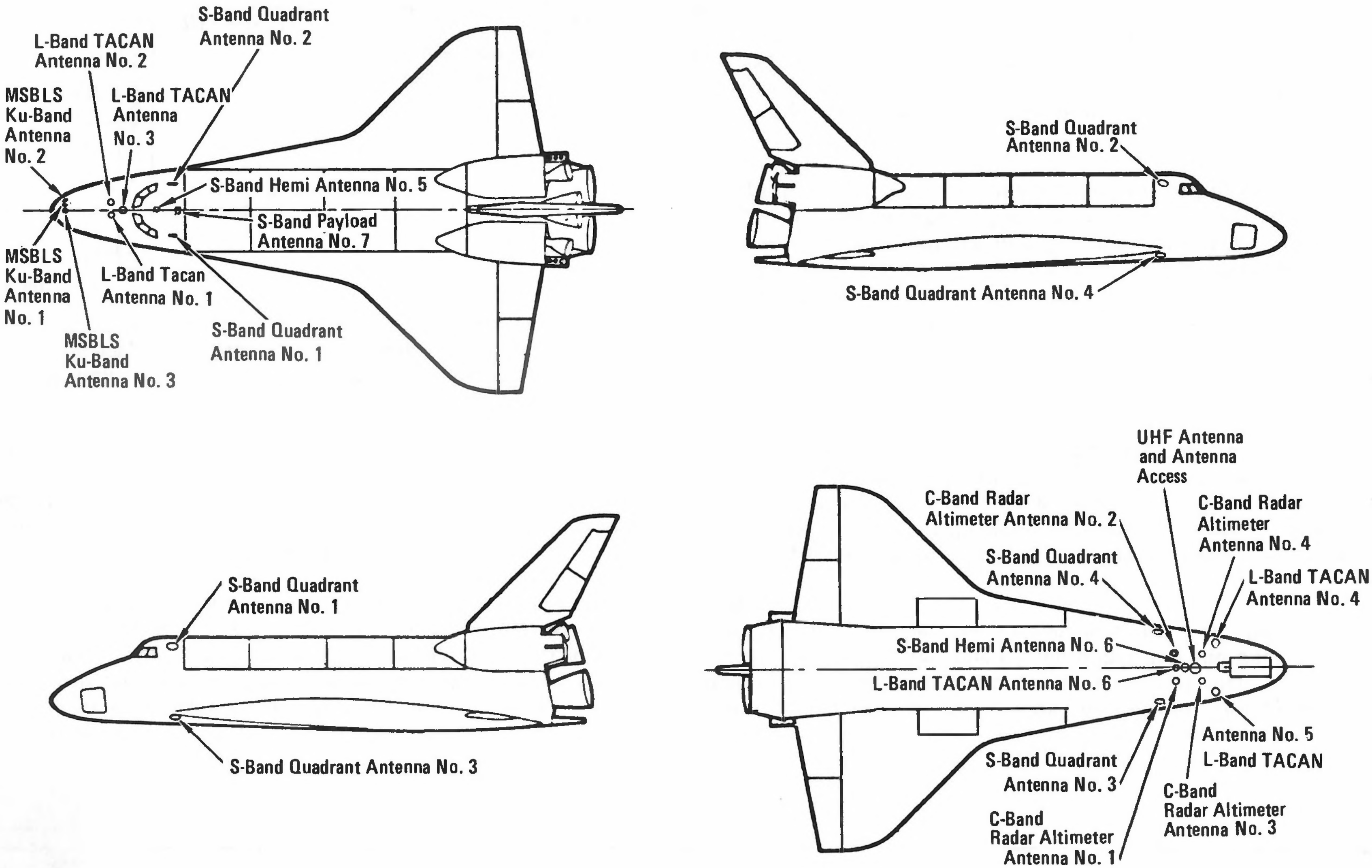


S-Band Antenna Switch

The payload antenna is located in the forward fuselage on the outer skin, just aft of the upper hemispherical antenna. It will be used as the radiating/receiving element for S-band payload forward, return links and with detached payloads. This antenna will be used in later flights. The payload antenna is also covered with thermal protection.

The S-band RF link is pulse-modulated with the high-data–rate bit stream or the low-data-rate bit stream. The high data rate contains two 32-kbps digital voice channels multiplexed with the 128-kbps telemetry on the downlink and two 32-kbps digital voice channels multiplexed with 8-kbps command on the uplink. The low data rate contains one 32-kbps digital voice channel multiplexed with 64-kbps telemetry on the downlink and one 24-kbps digital voice channel multiplexed with 8-kbps command on the uplink.

The network signal processor provides the interface to the audio distribution system with the S-band communications link. The NSP receives either one or two analog voice signals from the ACCU's for downlink transmission, depending on whether one or both of ,the air-to-ground channels are being used. The NSP converts these analog voice signals to digital voice signals, time-division-multiplexes them with PCM telemetry, and sends the composite signal to the S-band transponder. On the uplink, the NSP does just the reverse. It receives the composite signal from the S-band transponder and outputs either one or two analog voice signals to the ACCU. This composite uplink signal also has ground commands, which the NSP sends to the onboard computers.

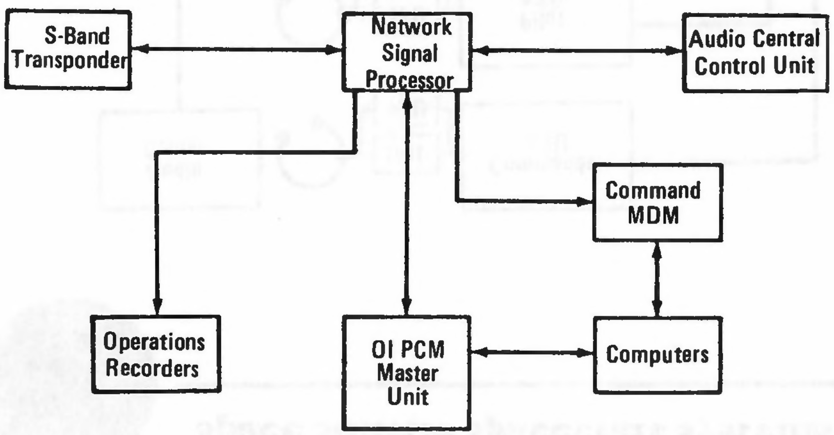


Orbiter Antennas

A UHF (ultra-high-frequency) system will be in operation during ascent, on orbit, and entry as an alternative to S band for voice and commands. The UHF system is an analog voice system.

AUDIO SYSTEM. It is through the audio distribution system that the flight crew can communicate with each other (intercom) as well as with ground support personnel. The system also provides for reception of caution and warning (C/W) aural tone signals in the event of an equipment malfunction or emergency, provides for reception of TACAN (tactical air navigation) identification signals from the ground tracking station, and, in later flights, provides for a communication with attached payload (Spacelab), payload bay, and docking ring.

The audio distribution system interfaces with the S-band communications system and in later flights with Ku-band communications system via the NSP to provide transmission and reception of external air-to-ground voice signals. It also interfaces with the UHF communication systems via the UHF transceiver for transmission and reception of external air-to-ground and air-to-air voice signals. It also interfaces with the launch umbilical for voice communications prior to and up to liftoff.



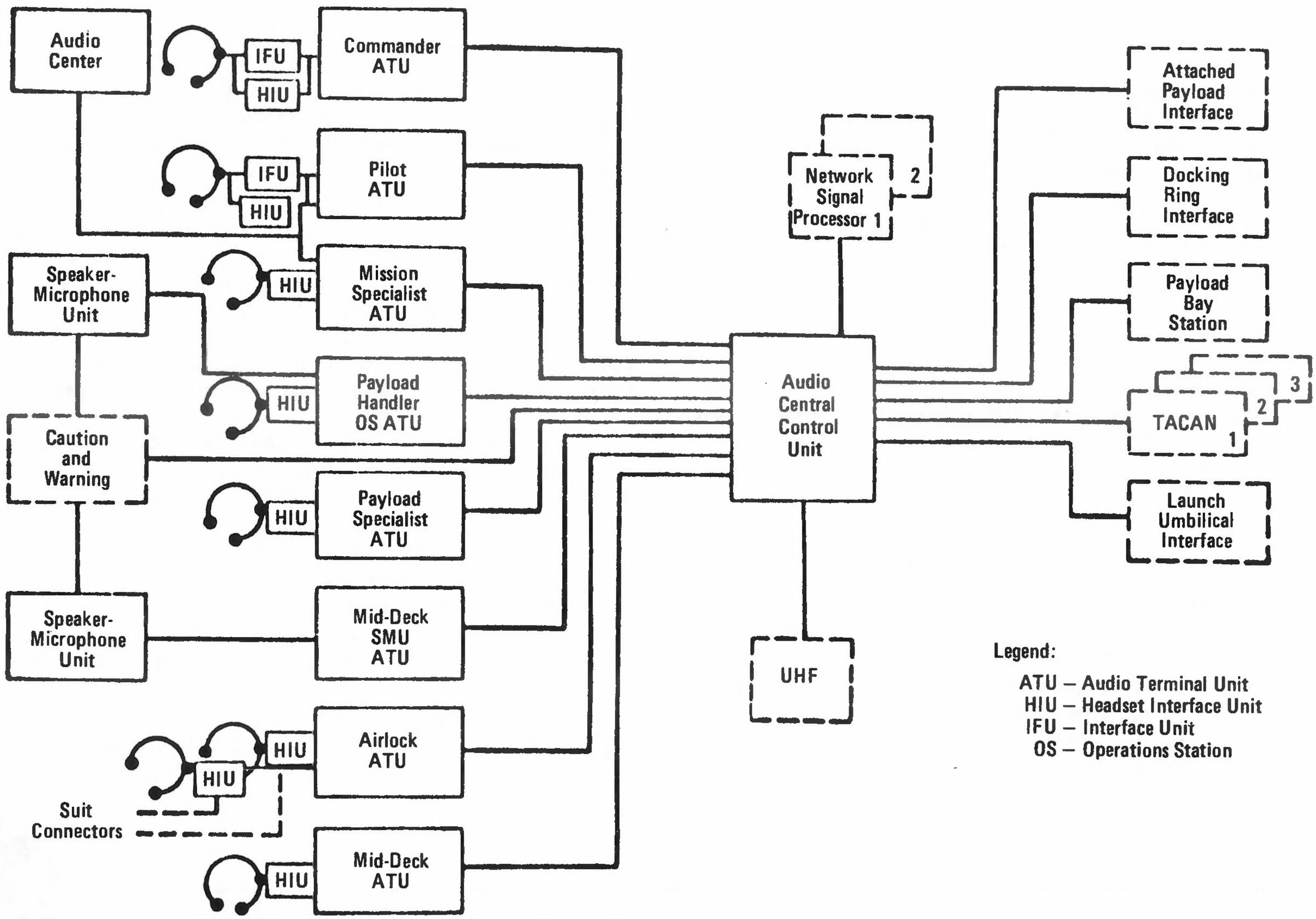
Network Signal Processor

During launch and reentry in the development flights, the flight crew members will be wearing ejection escape suits and will communicate with each other, and with the ground, through the helmets of their suits. When in orbit, they can, at their option, wear either the lightweight head set (LWHS) or a communications carrier assembly (CCA) headset. Regardless of the headset they use, they will also use a headset interface unit and a 1.21-meter (4-foot) crew communication umbilical (CCU).

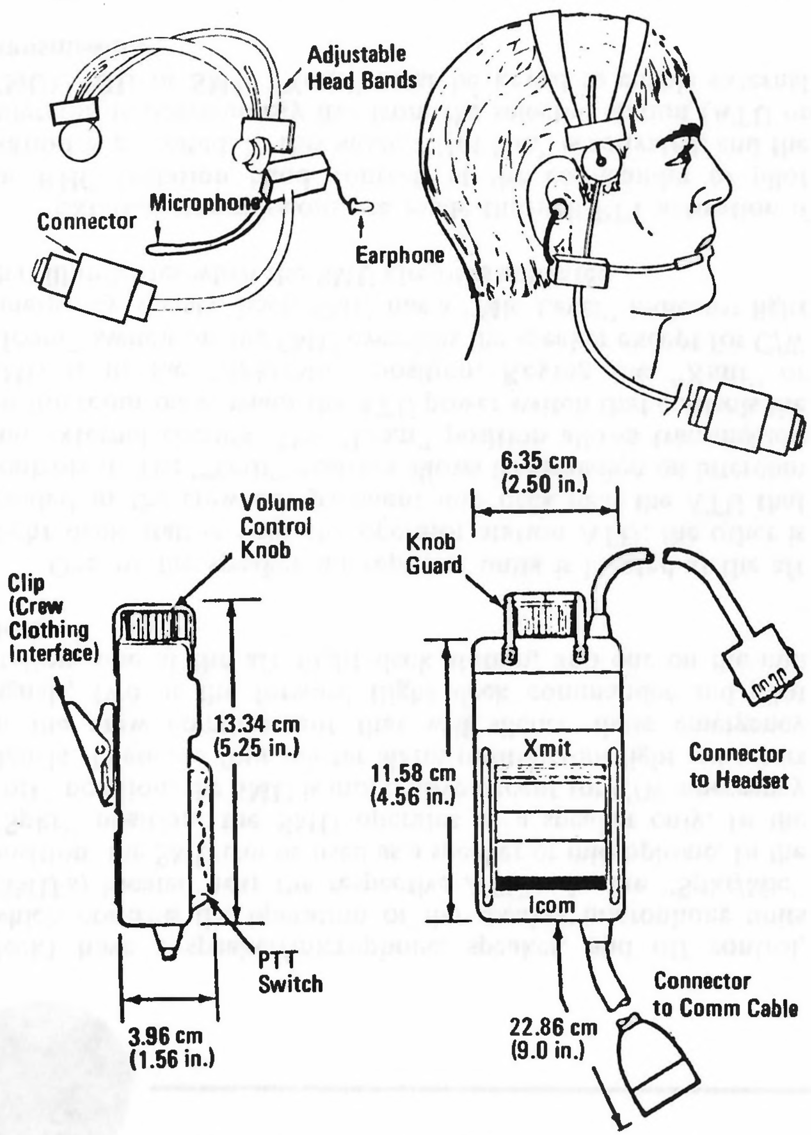
The headset interface unit (HIU) is 6.35 centimeters (2-1/2 inches) by 12.7 centimeters (5 inches) and clips onto the belt of the flight suit. The unit has a volume control knob and a rocker-type switch with three positions. The transmit position allows access to intercom and external circuits. The intercom position allows access to intercom only. The center (springloaded) position is the off position. The volume control knob acts in series with other volume controls on audio terminal units. The top of the HIU connects to the headset and the bottom to the CCU which, in turn, is connected to a CCU jack. Thus, the crew member headsets are electrically connected to the audio distribution system.

The audio terminal units (ATU's) are control panels used by the crew members to select audio channel, volume, and mode of voice communication such as "PTT" (push to talk), "VOX" (voice activated), or "Hot" (live microphone). Eight ATU's are available for use by the flight crew members. Two are located on the flight deck display and control panel forward station, three are located at the aft flight deck control and display panel station, two are located in the mid deck of the crew compartment, and one is located in the mid deck airlock for use during extra-vehicular activity (EVA).

Common to all ATU's are channel power switches, channel volume control thumbwheels, mode select knob, VOX sensitivity



Audio Distribution System



Headset Interface Unit

knob, page switch, and power switch. The channel power switches select access to any of five available channels for transmission or reception. The five available channels are two air to ground, one air to air, and two intercom. The switch positions are "T/R" (transmit and receive), "Rcv" (receive only), and off. The channel volume control thumbwheels provide volume control of the voice signal. The mode select knob allows selection of the "PTT/H ot," "PTT/VOX," and "VOX/VOX" method of transmission activation. The first set of initials indicates the method of external transmission activation, and the second set indicates the method of intercom activation. The VOX sensitivity knob controls the sensitivity of the microphones when operating by voice. The page switch enables a page to be transmitted to all other ATU's. The power switch in the "Aud/Tone" position provides all ATU active functions, including the reception of C/W tone signals; the "Aud" position is the same except it prevents reception of C/W tone signals at that ATU and the off position shuts off power to the ATU.

Features not common to all ATU's are the control knob, TACAN, and master volume control. Four of the eight ATU's have a two-position control knob that allows control of that ATU to be transferred to another ATU. Control of the commander's ATU can be switched to the pilot's ATU and vice versa. Control of the mission specialist ATU can be switched to the payload specialist ATU, and control of the airlock ATU can be switched to the mid deck ATU's. The commander and pilot only have ATU's for reception of TACAN ground identification signals of the TACAN ground station call letters in morse code, repeated every 40 seconds. The TACAN is a polar coordinate type of system which provides distance and bearing information to the station selected. The system operates in a band of frequencies from 962 to 1213 MHz. The master volume control on five of the eight ATU's (commander, pilot, operator station, airlock, and one mid deck) controls the volume of all incoming signals simultaneously and acts in series with other volume controls. Two of the ATU's (operator station and one in the mid deck) have a speaker/microphone, speaker, and off control, which controls the operation of the speaker microphone units (SMU's) located near the respective ATU's. In the "Spkr/Mic" position, the SMU can be used as a speaker or microphone. In the "Spkr" position, the SMU operates as a speaker only. In the "off" position, the SMU is inoperative except for C/W emergency signals. There are four master alarm pushbuttom light indicators in the crew compartment that will silence these emergency signals, two at the forward flight deck commander and pilot station, one at the aft flight deck station, and one on the mid deck.

One of the speaker microphone units is located at the aft flight deck station near the operator station ATU; the other is located in the crew compartment mid deck near the ATU that controls it. The "Xmit" position allows transmission on intercom and external circuits. The "Icom" position allows transmission on intercom only, when the ATU power switch that controls the SMU is in the "Spkr/Mic" position. Keying the "Xmit" or "Icom" switch on the SMU overrides the speaker except for C/W emergency signals. Each SMU has a "Mic Level" indicator light that illuminates when the SMU circuit is activated.

External transmissions are made through PTT activation if an RHC (rotation hand control) at the commander or pilot station is activated. In this mode, "Hot Mic" is activated, and the intercom is continuously live from the selected station (ATU or SMU). HIU or SMU "Xmit" must be keyed to enable external transmissions.

The audio control central unit located in the crew compartment mid deck forward avionics bay is the heart and most important component of the audio distribution system. All audio signals coming from or going to the ATU's (air-to-ground, air-to-air, C/W, TACAN identification, and intercom) are channeled through the ACCU. It acts as a central switchboard to identify, switch, and distribute all incoming and outgoing audio signals. The ACCU interfaces with all eight ATU's, the network signal processor, the UHF transceiver, the C/W, the three TACAN sets, and the launch umbilical. All of the audio signal circuits in the ACCU are electronically duplicated to provide internal redundancy. The ACCU has redundant power sources.

The audio center panel is located on the flight deck aft station near the mission specialist station. It provides operations recorder selection, electrical interface with the docking ring, payload bay, and attached payloads (not used on initial development flight), and UHF channel access selection. The two rotary knobs on the panel select various audio signals to be sent to the operation recorders via the NSP. Any two voice channels can be recorded at the same time. One knob selects the Channel 1 audio signal and the other knob selects the Channel 2 audio signal to be recorded. The audio center panel controls the communications with the docking ring, payload bay, and attached payloads (Spacelab) with on-off toggle switches which electrically connect that particular area to the audio distribution system. The audio center panel has three toggle switches that provide access to transmission and reception of UHF signals. One switch is for air-to-air channels and one is for each of the two air-to-ground channels. Each switch has T/R (transmission/reception) and off position. If all three switches are on the off position, UHF operations are not possible.

The UHF control panel located in the forward flight deck station overhead commander panel controls the UHF communication link. The transmit frequency switch selects one of the two UHF frequencies, 296.8 MHz primary or 259.7 MHz secondary, for external transmission. The antenna switch selects the UHF antenna on the lower forward fuselage external skin of the orbiter or the airlock antenna located in the forward payload bay. The airlock antenna is used only for air-to-air communications during EVA. The UHF antenna on the lower forward fuselage of the orbiter has a thermal protection covering. The squelch switch permits the UHF signal to be suppressed. A five-position rotary knob on the UHF control panel selects any of the following modes of UHF transmission: EVA–transmissions made on one frequency selected by "Xmit Freq" switch and reception is on the other frequency; Off–removes all electrical power; Simplex–transmission and reception are both made on the frequency selected by the "Xmit Freq" switch; Simplex + G Rcv–same as Simplex except that reception of the UHF guard (emergency) frequency (243.0 MHz) is also possible; G T/R–transmission and reception are both made on the UHF guard (emergency) frequency. When the rotary knob is in any position other than off, it activates power to the UHF channel access switches on the audio center panel and activates power to the UHF transceiver. All external UHF transmissions are sent to and received from the UHF transceiver via the ACCU. All three of the UHF frequencies (296.8 MHz, 259.7 MHz, and 243.0 MHz) are pre-set in the UHF transmitter and cannot be altered by the flight crew.

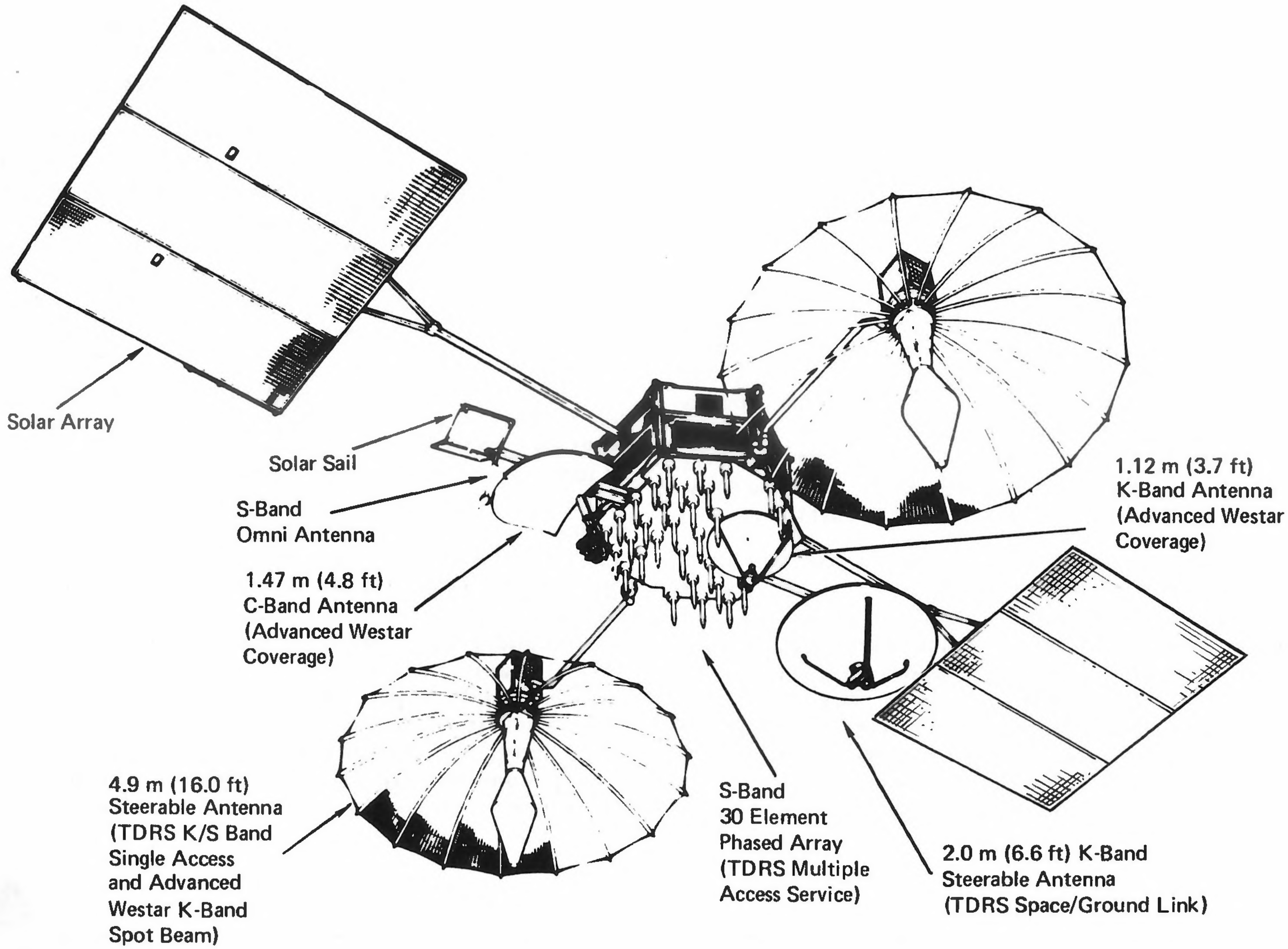
During EVA's in later flights, the UHF system will be used during the orbital phase of the flight. The UHF system will be used for air-to-air voice communications with the flight crew member who is outside the orbiter using the UHF antenna in the airlock. If there should be any need for EVA in the initial development flights, voice communications will be performed with the communications umbilical.

TRACKING AND DATA RELAY SATELLITE SYSTEM. In later flights of the Space Shuttle, a TDRS system will be placed into operation. In this system, the TDRS satellites will relay data to and from the orbiter, unmanned earth-orbiting spacecraft, and the earth station at NASA's White Sands (New Mexico) Test Facility. TDRS offers simultaneous data relay services for up to 32 user spacecraft in orbits up to approximately 2693 nautical miles (3100 statute miles) through 85 to 100 percent of each spacecraft's orbit. Today's earth stations provide only an average of 15 percent orbital tracking for each revolution as a result of line-of-sight acquisition and loss of signal line of sight to other ground stations in respect to the spacecraft. The TDRS satellite communications payload operates as a repeater, relaying signals to and from the earth station and user spacecraft. The TDRS incorporates three frequency bands, adding ,the high capacity K-band to the conventional S- and C-band. The TDRS, however, will require much stronger signals to travel the longer distances from the orbiter to the TDRS and ground. In the later flights, a Ku-band system installed in the orbiter payload bay will provide the stronger signals.

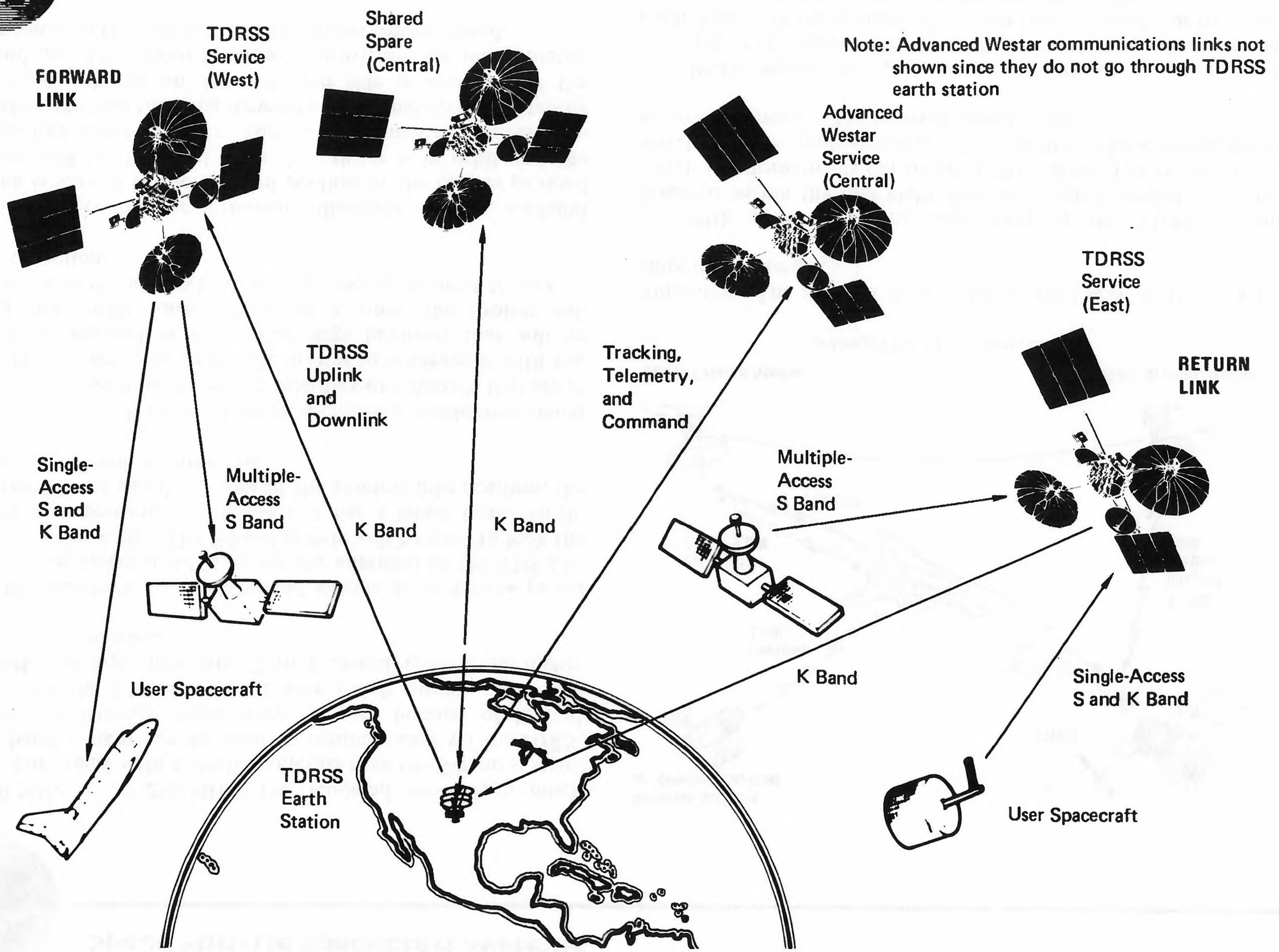
The TDRS was awarded in December 1976 by NASA to Western Union Space Communications Inc., a subsidiary of the Western Union Corp. TRW's Defense and Space Systems Group, Redondo Beach, California, is building the satellites and ground support equipment for Western Union Space Communications Inc. Electronic hardware is jointly supplied by Harris Electronics System Division, in Melbourne, Fla., and TRW, who also performs integration and test of the total earth station. TRW is also developing software for the overall system and will integrate this software with the earth station and the satellites.

The TDRS system will consist of four specialized communications satellites in geosynchronous orbit and a ground terminal at White Sands, New Mexico. One of these satellites will be used by Western Union as a domestic communications satellite and one will be a spare. The remaining two will provide the NASA tracking, command, and data relay services. Two of the four satellites are stationed over the Atlantic Ocean and two are stationed over the Pacific ocean. All TDRS satellites will be placed into earth orbit from the Space Shuttle with an Inertial Upper Stage (IUS) now being developed for the USAF by Boeing Aerospace. The TDRS includes advanced redundancy and reliability features to meet a 10-year-service requirement.

KU-BAND SYSTEM. The orbiter KU-band system operates in the Ku-band portion of the RF spectrum, which is



TDRSS/Advanced Westar Shared Satellite



Linking Four Identical and Interchangeable Satellites With Earth Station

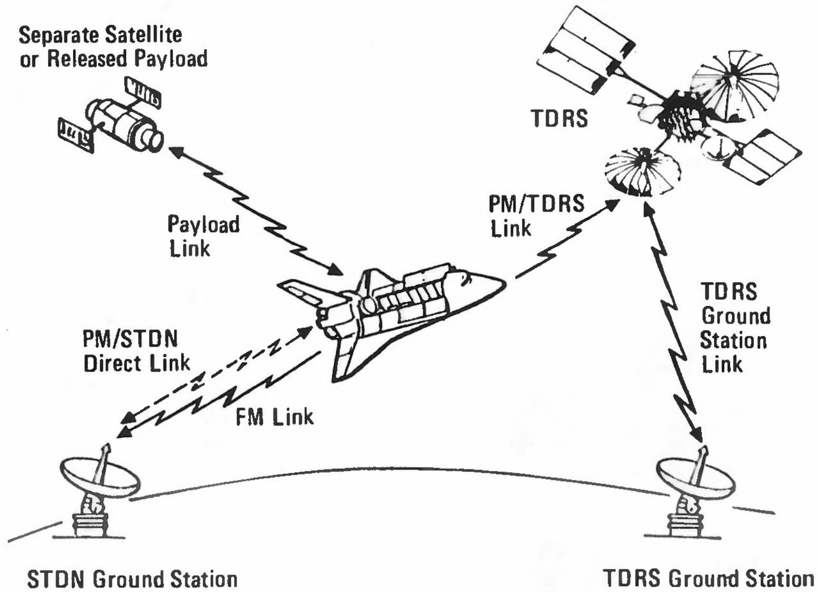
15,250 MHz to 17,250 MHz. The Ku-band provides a much higher gain signal with a smaller antenna than the S-band system. The S-band system can be used to communicate via theTDRS, but the low-data-rate mode must be used because of limited power since the S-band does not have a high enough signal gain to handle the high data rate. With Ku-band system, the higher data rates can be used.

One drawback of the Ku-band system is its narrow pencil beam, which makes it difficult for the antennas on the TDRS to lock on to the signal. The S-band system will be used to lock the antenna into position first because it has a larger beam width. Once the S-band signal has locked the antenna into position, the Ku-band signal will be turned on.

The orbiter Ku-band system includes a rendezvous radar which will be used to skin-track satellites or payloads that are in orbit. This makes it easier for the orbiter to rendezvous with any satellite or payload in orbit. For large payloads that will be carried into orbit, one section at a time, the orbiter will rendezvous with the payload that is already in orbit to add on the next section.

The 91-centimeter (36-inch) diameter orbiter Ku-band antenna is stowed in the forward portion or the orbiter payload bay and will be deployed after the orbiter is in orbit and the payload bay doors are open. The Ku-band antenna is located on the right-hand side (looking forward). The capability of installing a Ku-band antenna on the left-hand side is available. If the Ku-band antenna cannot be stowed, provisions are incorporated to jettison it so that the payload bay doors can be closed.

The Ku-band antenna is gimbaled, which permits it to acquire the TDRS for communications acquisition or radar search for other space hardware. The Ku-band system is first given the general location of the space hardware from the orbiter



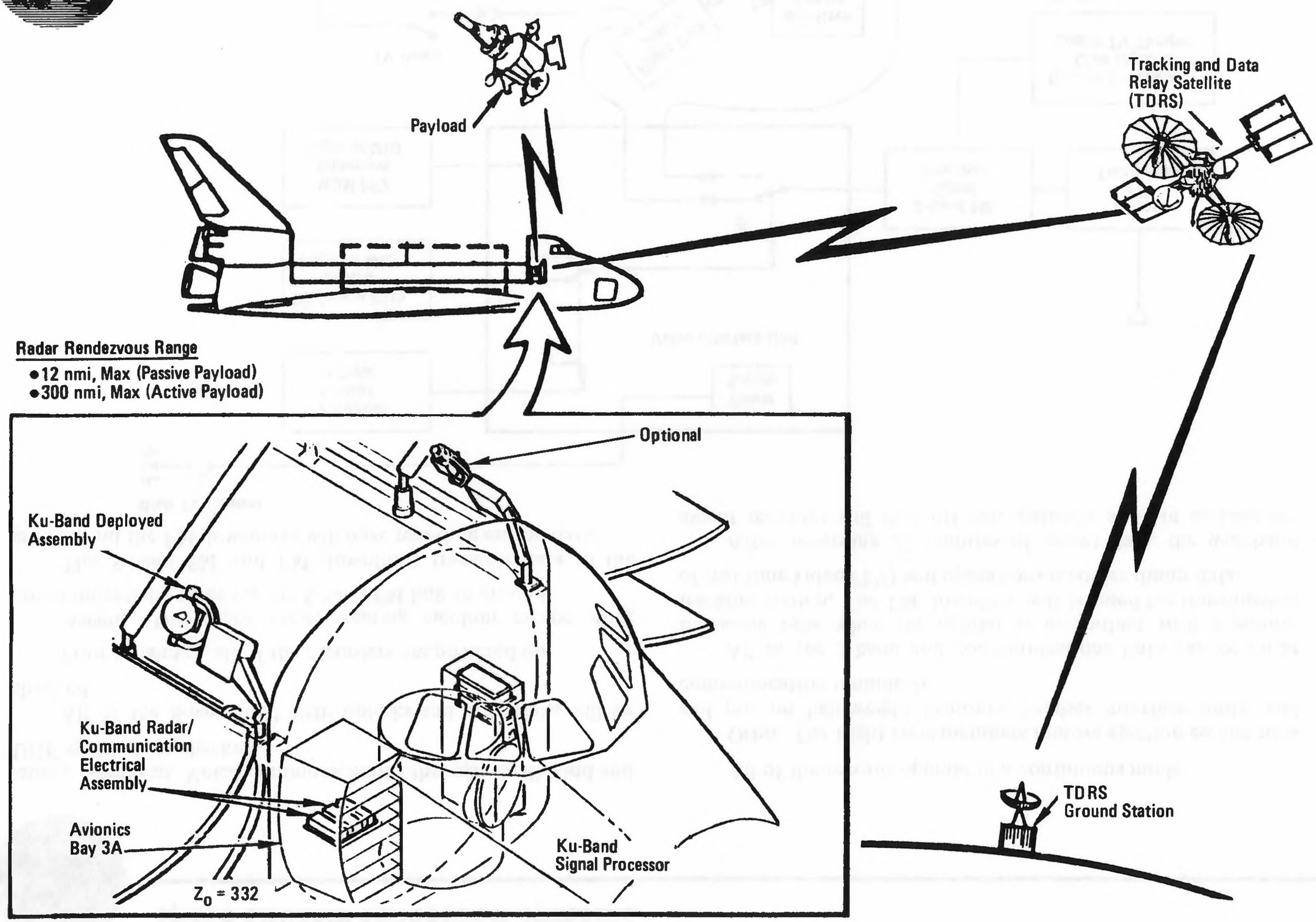
S-Band/TDRS Communications

computers. The antenna then makes a spiral scan of the area to pinpoint the target.

With communications acquisition, if the TDRS is not detected within the first eight degrees of spiral conical scan, the search is automatically expanded to 20 degrees. The entire TDRS search requires approximately three minutes. The scanning stops when an increase in the received signal is sensed.

Radar search for space hardware may use a wider spiral scan, up to 60 degrees. Objects may be detected by reflecting the radar beam off the surface of a target (passive mode) or by using the radar to trigger a transponder beacon on the target (active mode).

OPERATIONS SEQUENCE–Prelaunch. The crew, wearing ejection escape suits, will communicate by voice through the



Ku-Band Radar Communication System

launch umbilical. Voice communications through the S-band and UHF systems are checked.

All of the S-band and UHF uplinks and downlinks will be checked.

Prior to launch, all of the recorders are powered on.

Ascent. The flight crew, wearing ejection escape suits, communicate by voice via the S-band PM link to ground.

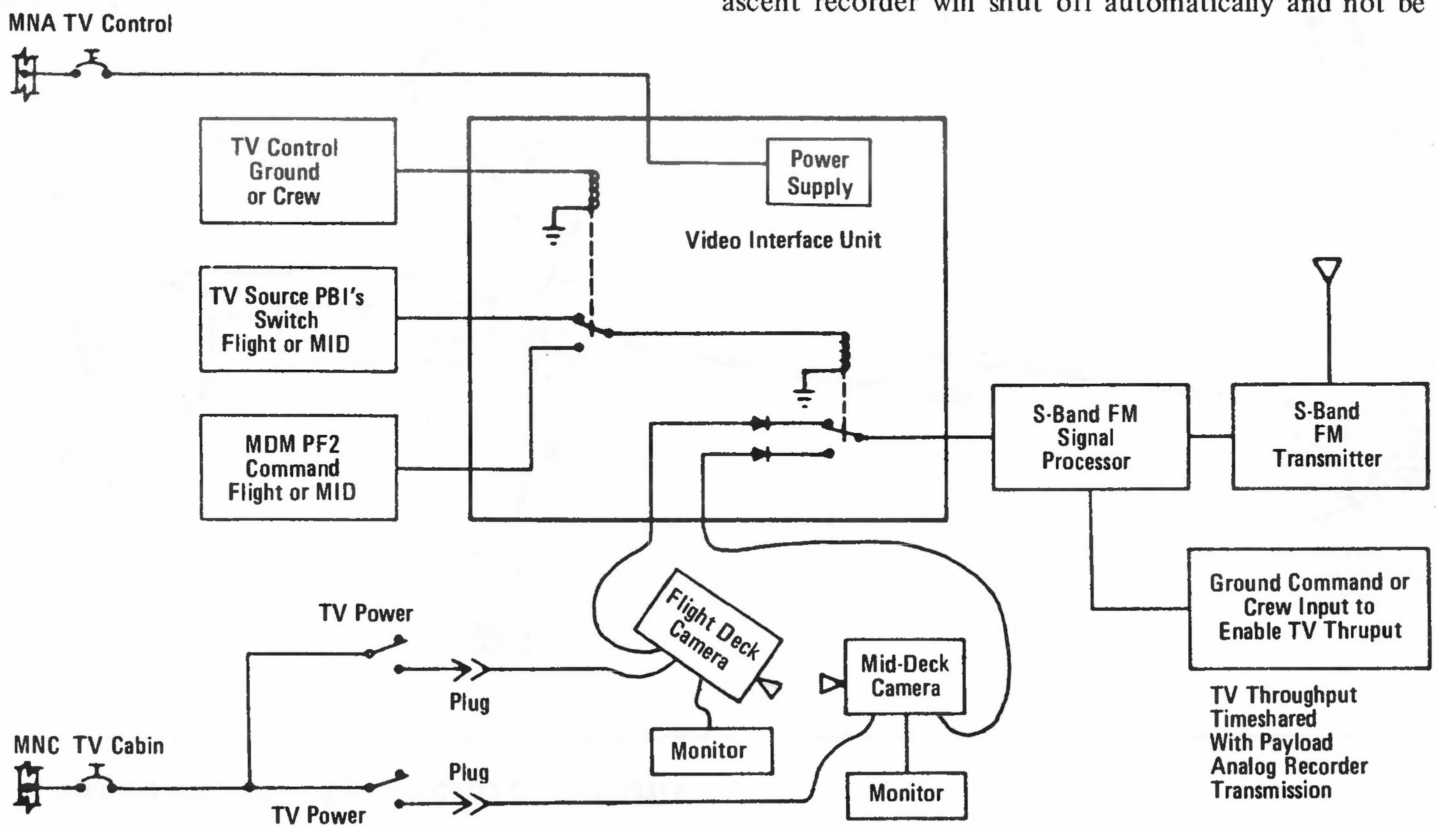
The S-band PM and FM downlinks transmit data to the ground and the FM downlinks will have real-time engine data.

All of the records operate in a continuous mode.

Orbit. The flight crew members remove ejection escape suits and put on lightweight headsets, headset interface units, and communication umbilicals.

All of the S-band and communications links can be on at the same time when the orbiter is in contact with a ground tracking station. The FM downlink will be used for transmission of real-time video (TV) and operations recorder dump data.

After recording 32 minutes of ascent data, the wideband ascent recorder will shut off automatically and not be used for



Television System

the rest of the flight. The PCM unit will be placed into a high or low sample rate mode of operation, and the wideband mission recorder will be shut off by the flight crew. The wideband mission recorder will remain off for most of the flight except when the flight crew will turn it on and back off again. The operations records will continue to operate under control by ground commands. They will be dumped whenever possible over each ground station via the FM system.

Reentry and Landing. The flight crew again don the ejection escape suits.

The UHF system will be used during approach and landing phase for air-to-ground voice communications with the landing site control tower and for air-to-air voice communications with the chase aircraft.

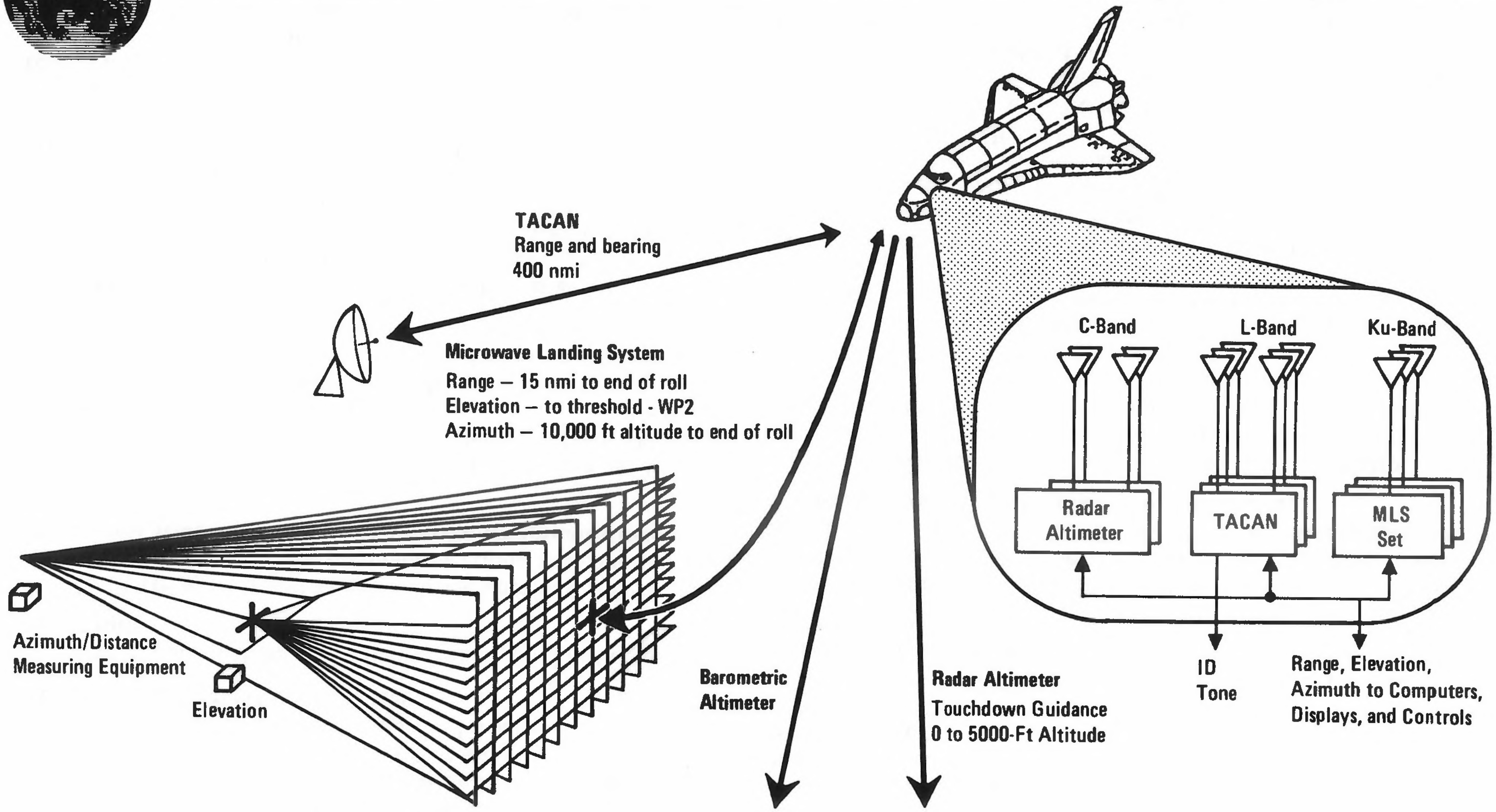
The wideband mission recorder will be turned on by the crew before reentry to use up the remainder of its tape. The PCM unit will be put into continuous mode of operation by the flight crew, and the operations recorders will continue recording. After landing, all of the instrumentation recorder will be played back by hardline via the T-O umbilical to retrieve the data.

TELEVISION. A color television system is available in the orbiter crew compartment. The TV system in the initial development flight test consists of refurbished Apollo-Soyuz Test Project equipment. For later development flight tests and subsequent flights it will be a newly developed closed-circuit system under development by RCA (Radio Corp. of America).

The interim TV system comprises two camera/lens assemblies with their monitors and a video interface unit (VIU). The two cameras are identical. One camera is for the flight deck and the other is for the mid deck. Either camera may be used in either location. The camera provides, through the monitor cables, both power and signal to its view finder monitor (VFM). This (black and white) monitor is used to assist the camera operator in locating the appropriate field for camera operations. The VIU contains the interface between the television cameras and the RF transmission equipment as well as the command logic for enabling the crew or the ground to select which camera is to be used. The flight crew utilizes the TV source pushbutton indicators on the flight deck display and control panel to select the flight deck or mid deck camera. The ground commands via S-band perform the same functions through a multiplexer/demultiplexer (MDM). The TV camera signal selected for transmission will be sent to the S-band FM signal processor for transmission to the ground on a time-shared basis with the recorder dumps.

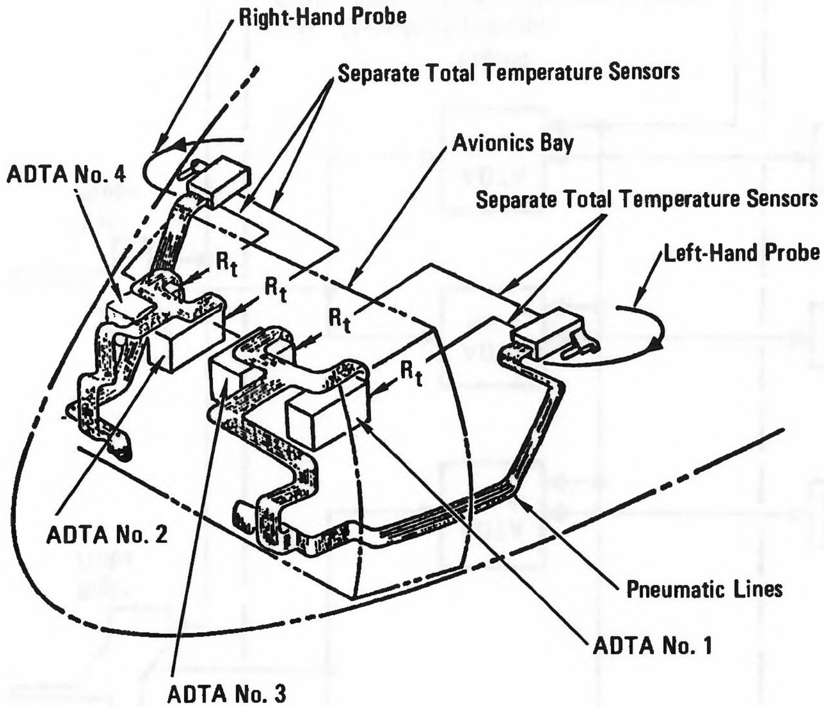
A cue light on the TV camera serves a dual purpose. First, the light is on if the camera is selected as the active camera. Second, the light will blink on/off if an overtemperature condition is sensed by circuits within the camera.

The contractors involved with the instrumentation and communication systems are Aydin, Vector Division, Newton, PA (wideband frequency division multiplexing); Communications Components, Costa Mesa, CA (UHF antenna); Conrac Corp., West Caldwell, NJ (mission timer event timer, ground command interface logic box, FM signal processor); Eldec Corp., Lynwood, WA (dedicated signal conditioner); Endevco, San Juan Capistrano, CA (piezoelectric accelerometer, acoustic pickup piezoelectric–acoustic and vibration); Gulton Industries. Costa Mesa, CA (accelerometer linear flow frequency, vibration, acoustic); Harris Corp., Electronic Systems Division, Melbourne, FL (pulse code modulation master unit); Harris Corp., Electronic Systems Division, Baltimore, MD (payload data interleaver); Hughes, El Segundo, CA (Ku-band radar, communication system deployable antenna and electrical assembly); K-West, Westminster, CA (wideband signal conditioner, strain guage signal conditioner); Magnavox, Ft. Wayne, IN (UHF 'receiver-transmitter mount, UHF receiver-transmitter); Micro Measurements, Romulus, MI (strain gauge), RDF Corp., Hudson, MH (sensors, transducers); Rosemount, Inc., Eden Priarie, MN (transducer, sensors); Radio Corp. of America, Astro-Electronics



Navigation Aids

Division, Princeton, NJ (closed circuit television); Spectran, La Habra, CA (sensors); Sperry Rand Corp., Flight Systems Division, Phoenix, AZ (multiplexer/demultiplexer); Statham Instruments, Oxnard, CA (transducers); Systron-Donner, Concord, CA (accelerometer); Teledynamics Division of Ambac Industries, Fort Washington, PA (S-band transmitter DFI, FM transmitter, S-band transceiver); Telephonics Division, Instruments Systems Corp., Huntington, NY (orbiter audio distribution system); TRW Systems, Electronic Systems Division, Redondo Beach, CA (S-band payload interrogator, S-band network equipment, network signal processor, payload signal processor); Transco Products, Venice, CA (S-band switch); Watkins Johnson, Palo Alto, CA (C-band radar altimeter antenna, L-band TACAN, S-band quad antenna, S-band hemiantenna, S-band payload antenna, S-band power amplifier); Wavecom, Northridge, CA (S-band multiplexer, DFI); Westinghouse Electric Corp., Systems Development Division, Baltimore, MD (master timing unit); AIL, Huntington, NY (S-band preamplifier assembly), AVCO, Wilmington, MA (Ku-band MSBLS antenna, Ku-band waveguide); Teledyne-Microwave, Mountain View, CA (S-band switch assem-



Air Data System

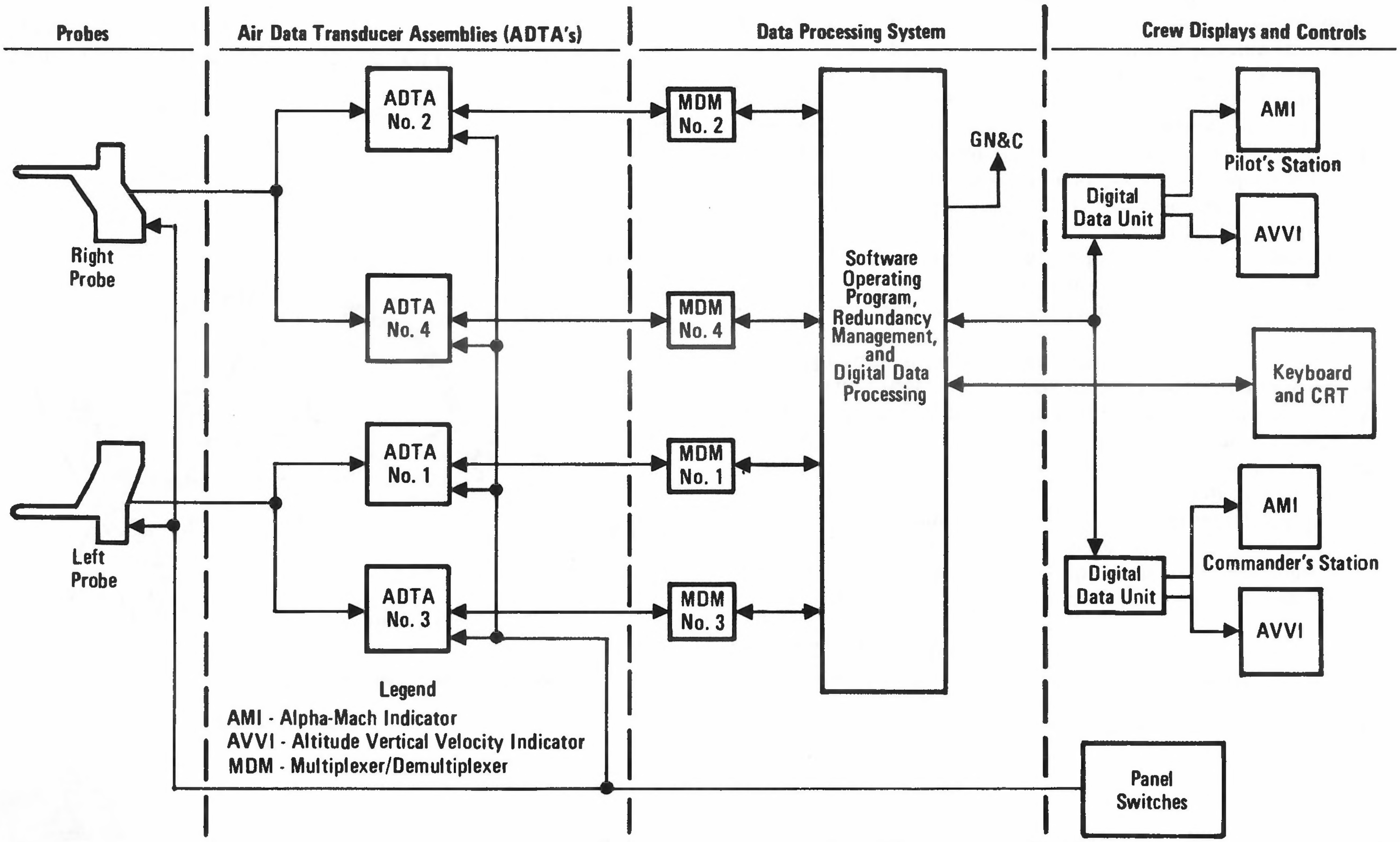
bly); Teledyne-Electronics, Neubury Park, CA (S-band FM transmitter); RCA, Government Communications Systems, Camden, NJ (extravehicular activity communications system).

NAVIGATION AIDS

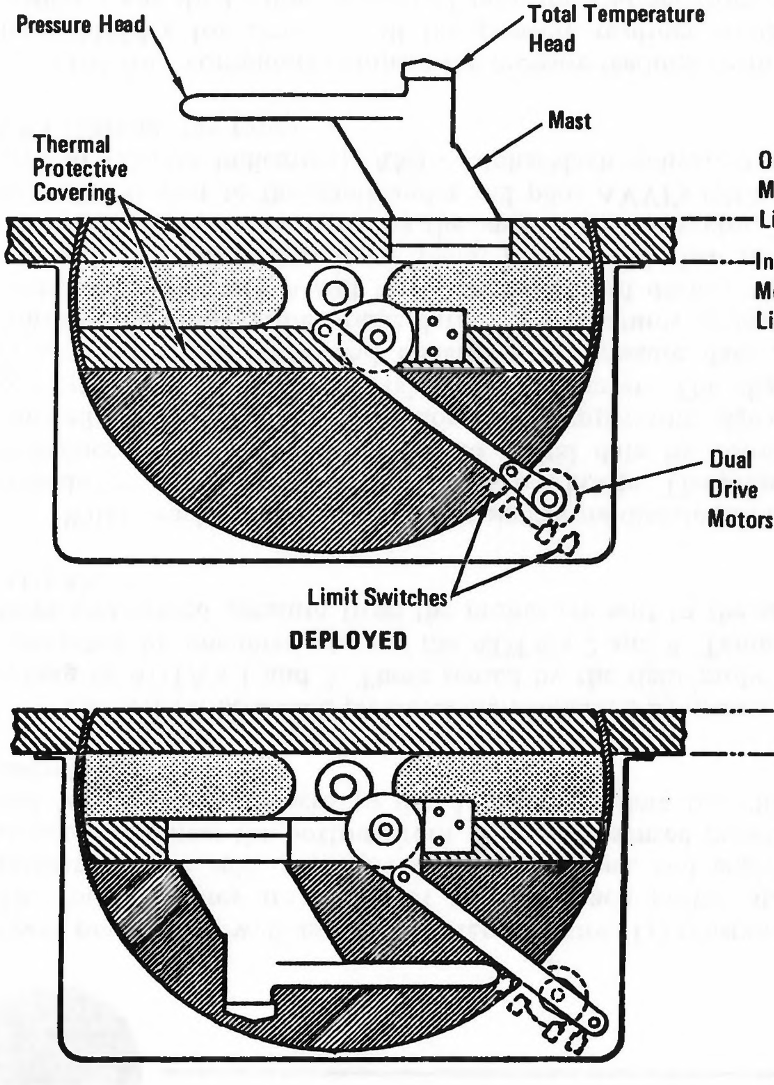
The navigation system used during entry consists of the inertial measurement units (IMU's) and navigation aids: TACAN (tactical air navigation), MSBLS (microwave scan beam landing system), air data system, and radar altimeter. The three IMU s maintain an inertial reference and provide velocity changes until MSBLS is acquired. Navigation-derived air data is needed during entry as inputs to the guidance, flight control and flight-crew-dedicated displays. Such data is collected from below Mach 3 through landing. The navigation-derived data is used as a backup to TACAN, which supplies range and bearing measurements and is available beginning at an altitude of approximately 44,196 meters (145,000 feet). TACAN is used until MSBLS acquisition or until landing if MSBLS is not available.

AIR DATA SYSTEM. The air data system provides information on movement of the orbiter in the air mass (flight environment) during entry. There are two air data probe assemblies, each consisting of a probe, an actuator, and dual drive motors. The probes are stowed during ascent, orbit, and the initial entry heat load environment. The probes are then deployed independently by the flight crew from the flight deck display and control panel when the orbiter's velocity is below Mach 3. The probes are located on the lower left and lower right side of the forward fuselage nose area.

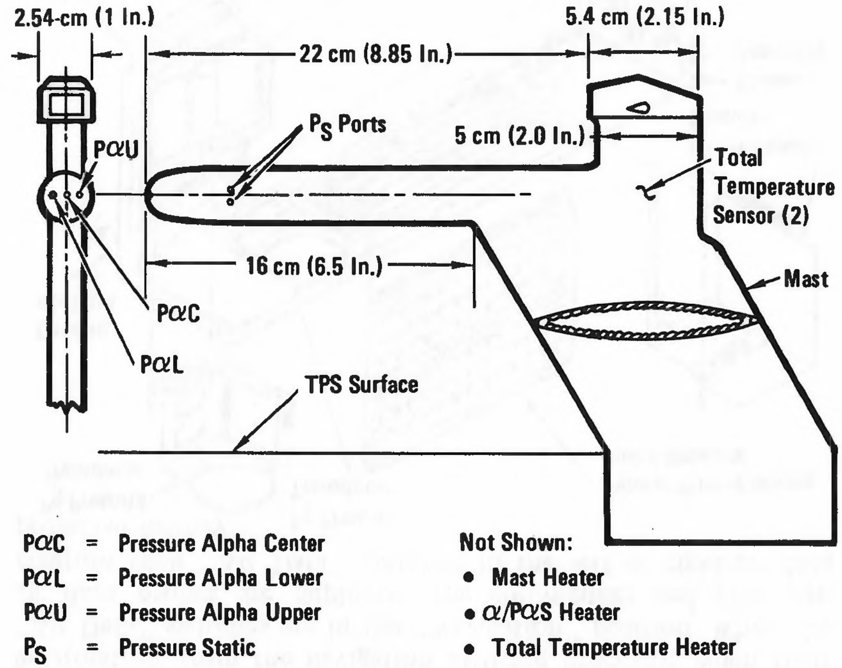
The air data system senses air pressures related to the spacecraft movement through the atmosphere for updating the navigation state vector in altitude, guidance in steering and speedbrake command calculations, flight control for control law computations and for display on the commander's and pilot's AMI's (alpha Mach indicators) and the commander's and pilot's AVVI's (altitude vertical velocity indicators). The AMI's display essential flight parameters relative to the spacecraft travel in the air mass such as angle of attack (alpha), acceleration, Mach/velocity, and knots equivalent airspeed. The AVVI's display such essential flight parameters as radar altitude, barometric altitude, atltitude rate, and altitude acceleration. Prior to deployment of the air data system probe, the AVVI's would receive their inputs from the navigation attitude processor. The AMI's would receive their inputs from the navigation attitude processor and IMU's prior to air data system probe deployment. The AMI acceleration indicator remains on the navigation attitude processor from the IMU's, as does the AVVI's altitude acceleration indicator. The AVVI radar altitude receive their information from the radar altimeters when the spacecraft is down to 5000 feet in altitude.



Air Data Interfaces



Air Data Probe



Air Data Probe Details

Each probe is independently deployed by two motors mechanically linked to an actuator. The flight crew controls the deployment from switches on the flight deck display and control panel. The motors, through mechanical gear reductions, are linked to an actuator which drives the probe to the deployed state. When the probe is fully deployed, limit switches in the probe assembly remove power from the electrical motors. If deicing of the probe is required, heaters are turned on automatically upon deployment. Deployment time is 15 seconds for two-motor operation and 30 seconds for single-motor operation. The probe mechanism (except for the probe itself) has thermal protection covering in both stowed and deployed positions.

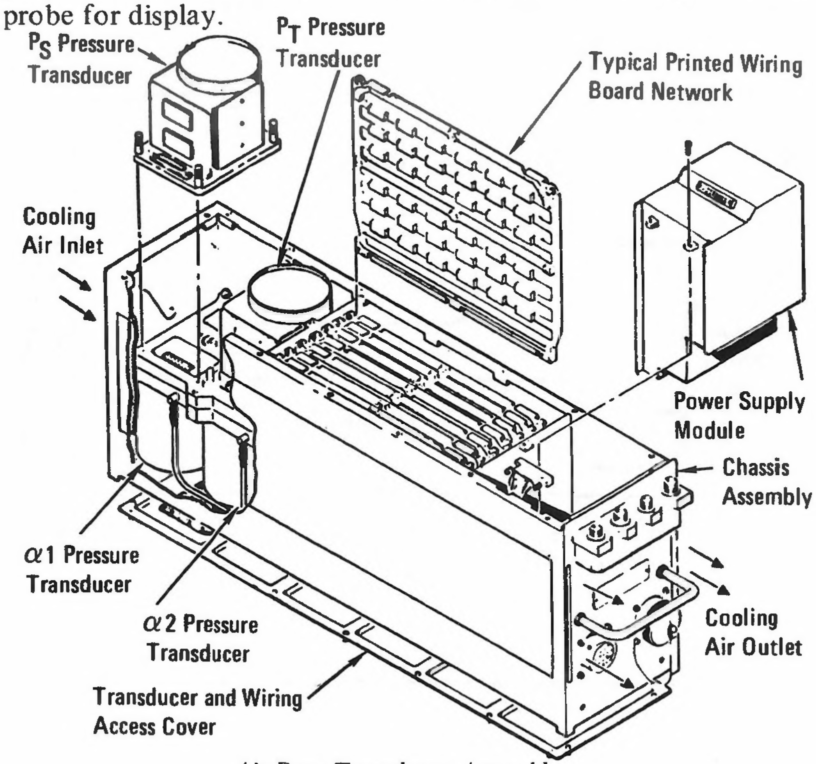
Each probe senses four pressures (static pressure, total pressure, angle of attack upper pressure, and angle of attack lower pressure) as well as two total temperature (T_t) resistances. The four pressures are sensed at ports on each probe: static pressure at the side, total pressure at the front, and angle of attack lower near the bottom front. The probe-sensed pressures and temperatures are sent to the ADTA's (air data transducer assemblies).

The left probe-sensed pressures are connected by pneumatic tubing to ADTA's 1 and 3. Those sensed by the right probe are connected by pneumatic tubing the ADTA's 2 and 4. Temperatures and sensed pressure from the probes are sent to the same ADTA's.

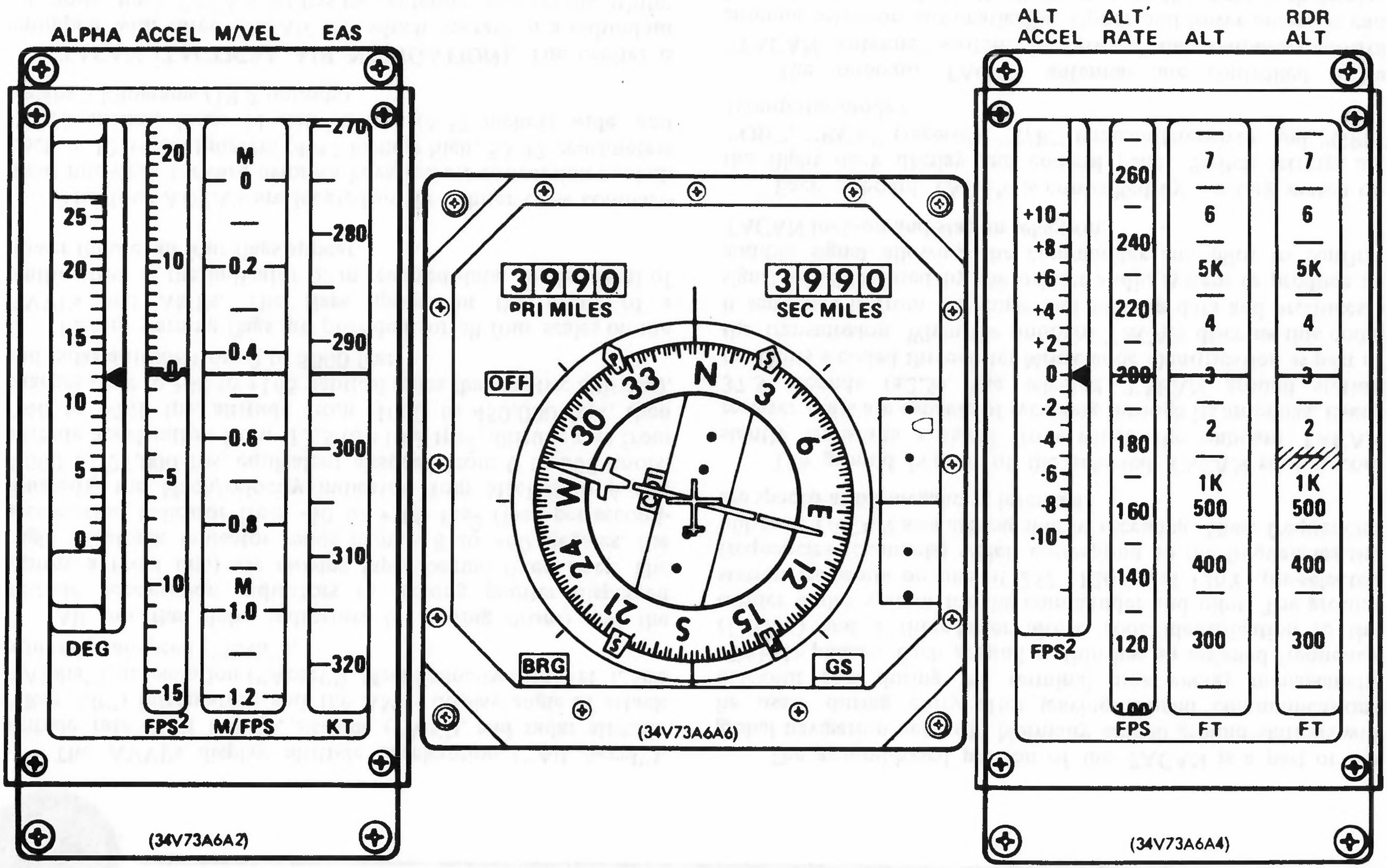
Within each ADTA, the pressure signals are directed to four transducers and the temperature signal to a bridge. The pressure transducer analogs are converted to digital data by counters controlled by the digital processor. The temperature signal is converted by an analog/digital (A/D) converter. The digital processor corrects errors and linearizes the pressure data and converts the temperature bridge data to temperatures in degrees centigrade. This data is sent to the digital output device, which converts the signals into serial digital format, and then to the onboard computers for updating the navigation state vector. The data also is sent to the commander and pilot AVVI's (altitude/vertical velocity indicators), AMI's (alpha/Mach indicator), and CRT (cathode ray tube).

The four computers compare the pressure readings from the four ADTA's for error. If all the pressure readings compare within a specified value, one set of pressure readings from each probe is summed and averaged and sent to the software. If one or more pressure signals of a set of probe pressure readings fail, the failed set data flow from that ADTA to the averager is stopped and the software will receive data from the other ADTA of that probe. If both sets from a probe fail, the software operates on data from the two ADTA's connected to the other probe. The best total temperature from all four ADTA's is sent to the software. A fault detection would illuminate the "Air Data" red caution/warning light, backup C/W alarm light, and master alarm light and sound the audible tone and a fault message on the CRT. A communication fault will illuminate the "SM Alert". Prior to the air data probe deployment, the commander and pilot AVVI's and AMI's receive information from the navigation attitude processor.

The commander and pilot AVVI's and AMI's receive inertial information from the navigation attitude processor when their "Air Data" switches are in the "Navigation" position. When the air data probes are deployed, the commander and pilot can position their "Air Data" switches to the left or right air data probe for display.



Air Data Transducer Assembly



AMI (left), HSI (center), and AVVI (right)

The AVVI's display altitude acceleration ("Alt Accel"), altitude rate ("Alt Rate"), altitude ("Alt"), and radar altitude ("Rdr Alt") information and the AMI's display angle of attack ("Alpha"), acceleration ("Accel"), Mach/velocity ("M/Vel"), and equivalent airspeed ("EAS").

All but the alpha indicators (a moving drum) and the altitude acceleration indicators (a moving pointer displayed against a fixed line) are moving tapes behind fixed lines. The angle of attack indicator reads from -18 to +60 degrees, the acceleration indicator from -50 to +100 fps^2 (feet per second-squared), the Mach/velocity indicator from Mach 0 to 4 and 4,000 to 27,000 fps; equivalent airspeed from 0 to 500 knots, altitude acceleration from -13.3 to +13.3 fps^2, altitude rate from -740 to +740 fps, altitude from -1000 to 450,000 feet, then changes scale to +40 to +165 nautical miles (barometric altitude), and radar altitude from 0 to 5000 feet.

Failure warning flags are provided for all four scales on the AVVI's and AMI's. The flags appear in the event of a malfunction in the indicator or in received data. In the event of power failure, all four flags appear.

The four ADTA's are located in the orbiter crew compartment mid-deck forward avionics bays and are convection cooled. Each is 12.36 centimeters (4.87 inches) high, 53.97 centimeters (21.25 inches) long, 11 centimeters (4.37 inches) wide, and weighs 8 kilograms (19.2 pounds).

TACAN (TACTICAL AIR NAVIGATION). The orbiter is equipped with three TACAN sets which operate in a redundant set mode. Each TACAN set has two antennas, one on the orbiter lower forward fuselage and one on the orbiter upper forward fuselage. The TACAN's are the airborne portion of the global navigation system for military and civil aircraft operating at L-band (1-gigahertz) frequencies. The TACAN sets are used as an external navigation aid in the orbiter during the entry phase and RTLS (return-to-launch-site) abort.

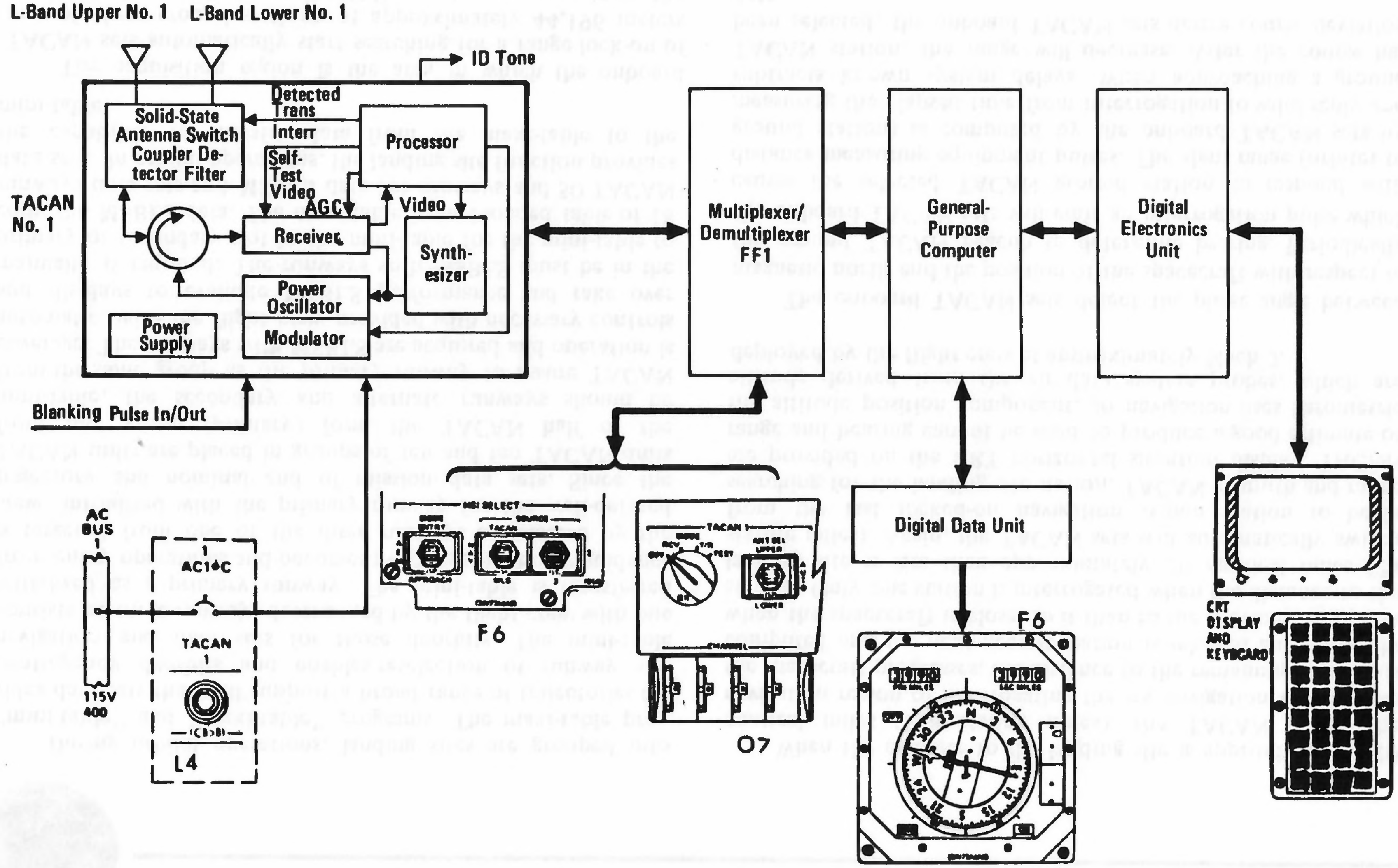
The ground-based portion of the TACAN is a part of the global navigation network. Normally several ground stations will be used during entry after leaving L-band communications blackout and during the terminal area energy management (TAEM) phases. Each ground station has an assigned frequency (L band) and a three-letter Morse code identification to the orbiter audio system for the commander and pilot. The ground station transmits on one of 252 (126X and 126Y) pre-selected frequencies (channels) which correspond to the frequencies the onboard TACAN sets are capable of receiving. These frequencies are spaced at 63-megahertz intervals.

The ground beacon of the selected TACAN station constantly transmits a signal from which the onboard TACAN receiver units are capable of receiving through its antennas. Every 37.5 seconds (±2.5), the selected TACAN ground station transmits a coded three-letter Morse code identification as part of the transmission. When the onboard TACAN discerns this code, it separates it from the range and bearing data and produces a signal which is used by the orbiter audio system to produce an audible signal allowing the commander and pilot to confirm TACAN lock-on and station selection.

Each onboard TACAN is controlled by a rotary switch on the flight deck display and control panel. Switch settings are "Off", "RCV" (receive), "T/R" (transmit/receive), and "GPC" (computer mode).

The onboard TACAN antennas are controlled by a "TACAN Antenna" switch. In "Auto," the computers control antenna selection automatically. Upper and lower antennas can be selected manually by the flight crew on the flight deck display and control panel.

In the GPC mode, the onboard computers control TACAN channel selection automatically. Ten TACAN ground stations are programmed into the software, divided into three geometric regions: the acquisition region (three stations), the navigation region (six stations), and the landing site region (one station).



TACAN Receiver-Transmitter

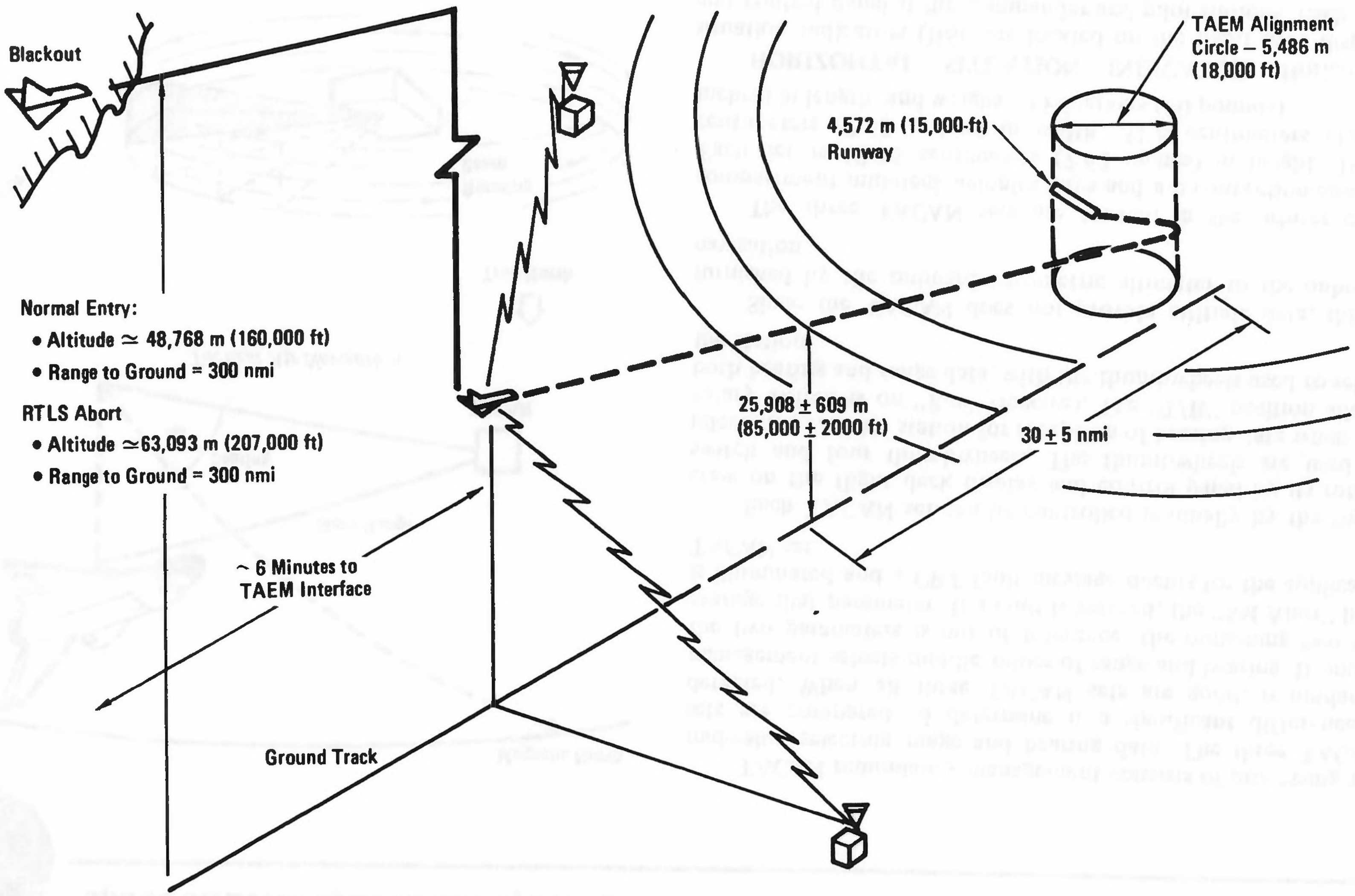
During orbital operations, landing sites are grouped into "mini-table" and "maxi-table" programs. The maxi-table provides data sets that will support a broad range of trajectories for contingency deorbits and enables reselection of runway and navigation and data sets for those deorbits. The mini-table consists of three runways determined by the flight crew with one initialized as a primary runway. The mini-table is transferred from entry operations and becomes unchangable. Entry guidance is targeted from one of the three runways as selected by the crew, initialized with the primary runway for the well-defined trajectory and nominal end of mission data sets. Since the TACAN units are placed in groups of ten and ten TACAN units from one group (primary) form the TACAN half of the mini-table, the secondary and alternate runways should be from the same group as the primary runway to assure TACAN coverage. The runways with MSBLS are acquired and operation is automatic, with the flight crew provided with necessary controls and displays to evaluate MSBLS performance and take over manually if required. The runways with MSBLS must be in the primary or secondary slot in the mini-table for the mini-table to copy the MSBLS data. The maxi-table is an I-loaded table of 18 runways data sets and MSBLS data for runways and 50 TACAN data sets. In orbital operations, the landing site function provides the capability to transfer data from the maxi-table to the mini-table.

The acquisition region is the area in which the onboard TACAN sets automatically start searching for a range lock-on of one of three ground stations, at approximately 44,196 meters (145,000 feet). After one TACAN acquires a range lock, the other two will lock on to the same ground station. If TACAN has not updated navigation at 42,672 meters (140,000 feet) altitude, the right-most event sequence green light on the flight deck display and control eyebrow panel will flash for five seconds and remain on. When at least two TACAN sets lock on, TACAN range and bearing are used by navigation to update state vector until MSBLS selection and acquisition.

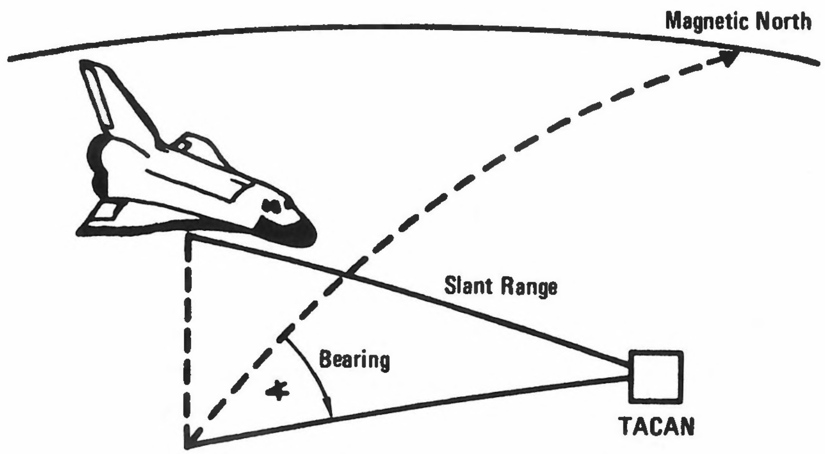
When the distance to the landing site is approximately 120 nautical miles (138 statute miles), the TACAN begins the navigation region of interrogating the six navigation stations. As the spacecraft progresses, the distance to the remaining stations is computed and the next nearest station is selected automatically when the spacecraft is closer to it than to the previous locked-on station. Only one station is interrogated when the distance to the landing site is less than approximately 20 nautical miles (23 statute miles). Again, the TACAN sets will automatically switch from the last locked-on navigation region station to begin searching for the landing site station. TACAN azimuth and range are provided on the CRT horizontal situation display. TACAN range and bearing cannot be used to produce a good estimate of the altitude position component, so navigation uses barometric altitude derived from the air data system probes, which are deployed by the flight crew at approximately Mach 3.

The onboard TACAN sets detect the phase angle between magnetic north and the position of the spacecraft with respect to the ground TACAN beacon to determine bearing. Periodically the onboard TACAN sets will emit an interrogation pulse which causes the selected TACAN ground station to respond with distance measuring equipment pulses. The slant range (orbiter to ground station) is computed by the onboard TACAN sets by measuring the elapsed time from interrogation to valid reply and subtracts known system delays. When approaching a ground TACAN station, the range will decrease. After the course has been selected, the onboard TACAN sets derive course deviation data.

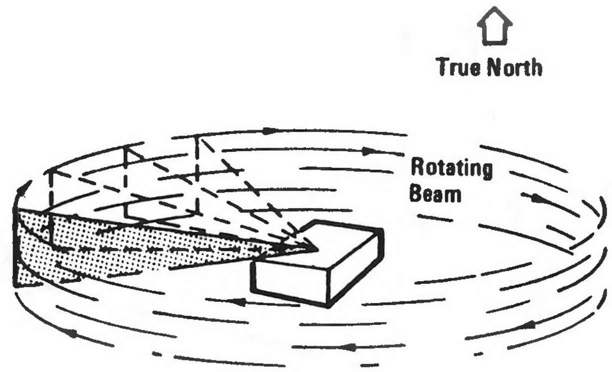
The range and bearing data are used in the entry phase by navigation to update the state vector position components, area navigation for display on the flight deck display and control panel HSI's (horizontal situation indicators) after the data is transformed during TACAN operation, and for display of raw TACAN data on the CRT.



TACAN Appr--ch Geometry



Tactical Air Navigation



- TACAN station transmits RF burst when beam points to true north
  - Aircraft receiver times interval from burst to beam impingement and derives bearing

TACAN Operation

TACAN redundancy management consists of processing and mid-value selecting range and bearing data. The three TACAN sets are compared to determine if a significant difference is detected. When all three TACAN sets are good, redundancy management selects middle values of range and bearing. If one of the two parameters is out of tolerance, the remaining two will average that parameter. If a fault is verified, the "SM Alert" light is illuminated and a CRT fault message occurs for the applicable TACAN set.

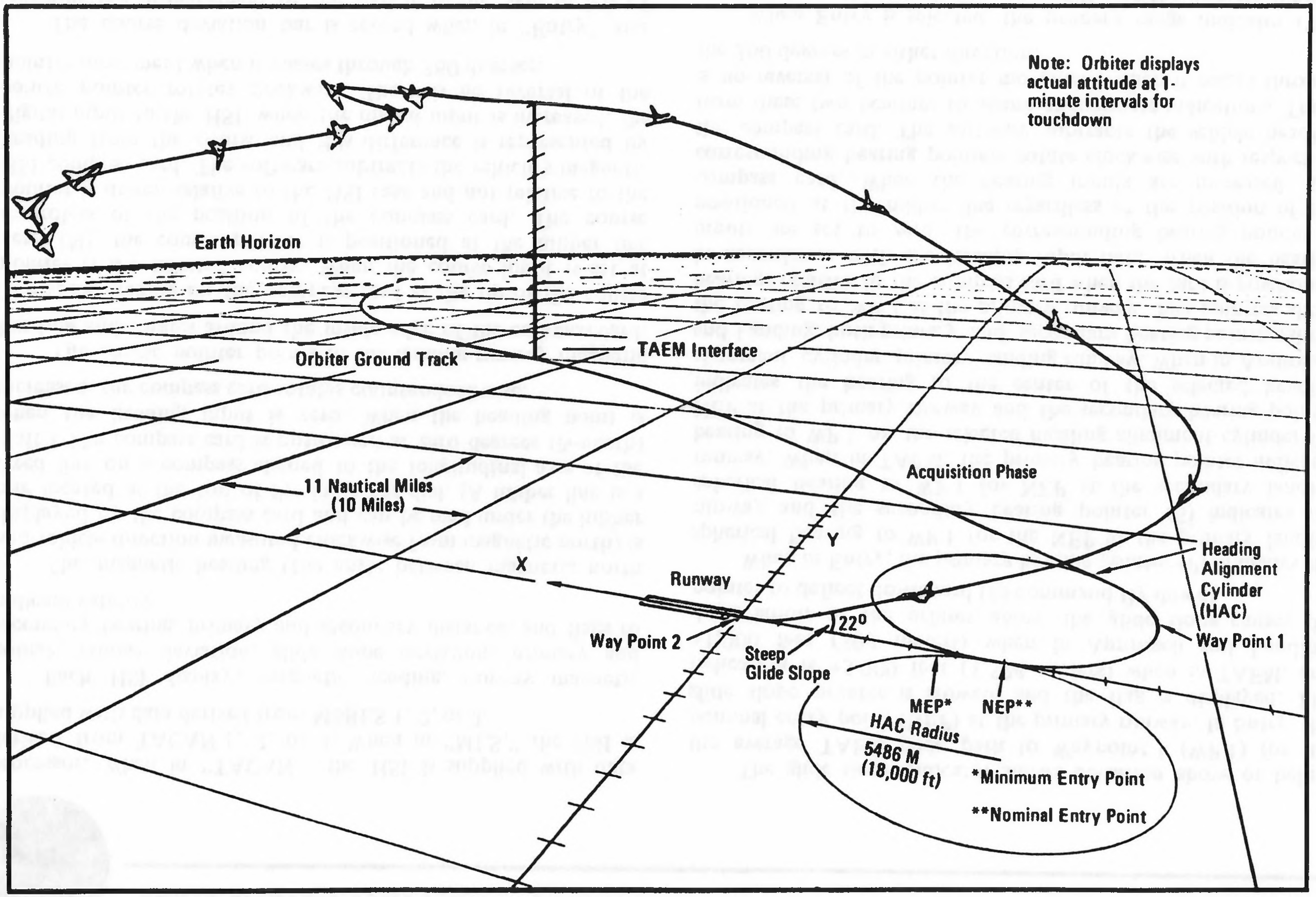
Each TACAN set can be controlled manually by the flight crew on the flight deck display and control panel by its rotary switch and four thumbwheels. The thumbwheels are used to select the TACAN station for reception of bearing data when the rotary switch is on "Rcv" (receive). The "T/R" position allows both bearing and range data, with the thumbwheels used to select the station.

Since the TACAN does not provide altitude data, this is furnished by the onboard barometric altimeter to the onboard navigation.

The three TACAN sets are located in the orbiter crew compartment mid-deck avionics bays and are convection cooled. Each set is 19.35 centimeters (7.62 inches) in height, 19.35 centimeters (7.62 inches) in width, 31.8 centimeters (12.53 inches) in length, and weighs 13 kilograms (30 pounds).

HORIZONTAL SITUATION INDICATOR. Horizontal situation indicators (HSI) are located on the flight deck display and control panel at the commander and pilot stations. Each HSI displays a pictorial view of the spacecraft's location with respect to various navigation points during entry through rollout and during RTLS (return-to-launch-site) abort.

Three switches are associated with each HSI. One switch selects the mode ("Entry," "TAEM," or "Approach") and the other two select the source of data ("Nav," "TACAN," or "MLS" from source 1, 2, or 3). When in the "Nav" position, that HSI is supplied with data derived from navigation attitude



Entry Flig' 'Profile

processor. When in "TACAN," the HSI is supplied with data derived from TACAN 1, 2, or 3. When in "MLS," the HSI is supplied with data derived from MSBLS 1, 2, or 3.

Each HSI displays magnetic heading, runway magnetic course, course deviation, glide slope deviation, primary and secondary bearing, primary and secondary distance, and flags to indicate validity.

The magnetic heading (the angle between magnetic north and vehicle direction measured clockwise from magnetic north) is displayed by the compass card and can be read under the lubber line located at the top of the indicator dial. (A lubber line is a fixed line on a compass aligned to the longitudinal axis of the craft.) The compass card is positioned at zero degrees (N-north) when the heading input is zero. When the heading point is increased, the compass card rotates counterclockwise.

The course pointer points to the landing runway magnetic heading and rotates around the inside edge of the compass card.

The scale reading on the compass card at the tip of the course pointer is the selected course. When the course input is set at zero (N), the course pointer is positioned at the lubber line regardless of the position of the compass card. The course pointer is driven relative to the HSI case and not relative to the HSI compass card. The software subtracts the vehicle's magnetic heading from the course and this difference is represented by digital input to the HSI; when the digital input is increased, the course pointer rotates clockwise. There is no reversal of the pointer movement when it passes through 360 degrees.

The course deviation bar is zeroed when in "Entry" and indicates the displacement of the vehicle to the right or left of the extended runway centerline +10 degrees when in TAEM and +2.5 degres when in Approach and Landing. When the course deviation input is zero, the deviation bar is aligned with the ends of the course pointer. An orbiter deviation off course to the left causes the deviation bar to deflect to the right (command to fly right).

The glide slope indicator shows deviation above or below the average TAEM glide path to Waypoint 1 (WP 1) for the nominal entry point (NEP) at the primary runway. In Entry, the glide slope pointer is stowed, and the flag is displayed. The indication is +5,000 feet (1,524 meters) when in TAEM, and +1,000 feet (304 meters) when in Approach and Landing. A deviation of the orbiter above the glide slope causes the pointer to deflect downward (to command fly down).

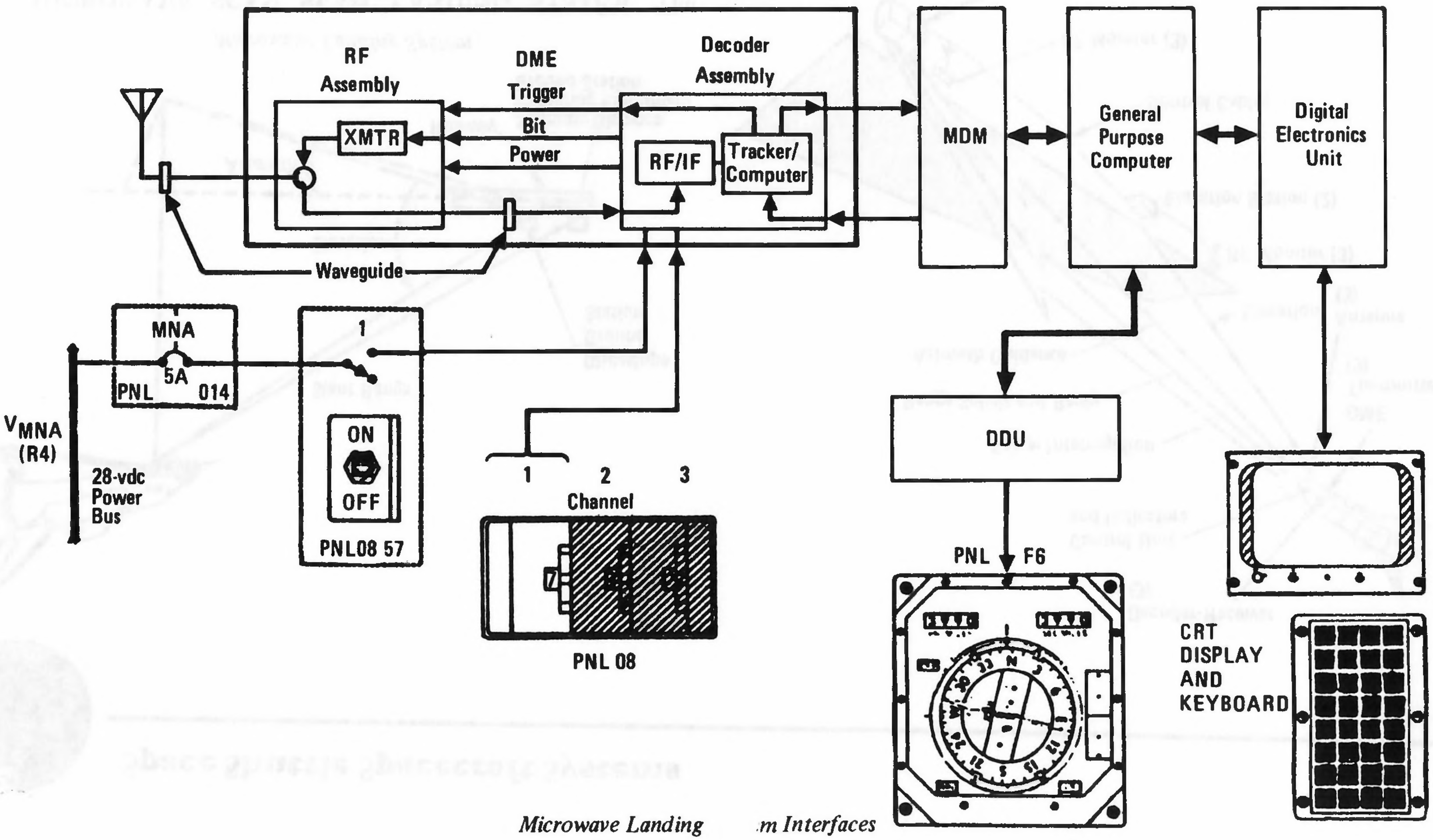
When in Entry, the primary bearing pointer (P) indicates the spherical bearing to WP 1 for the NEP at the primary landing runway and the secondary bearing pointer (S) indicates the spherical bearing to WP 1 for NEP at the secondary landing runway. When in TAEM, the primary bearing pointer indicates bearing to WP 1 on the selected heading alignment cylinder for NEP at the primary runway and the secondary bearing pointer indicates the bearing to the center of the selected heading alignment cylinder primary landing runway. When in Approach and Landing, both primary and secondary bearing pointers show the bearing to WP 2 at the primary runway. The pointers show bearings relative to the compass card when the card is positioned in accordance with the heading input data. When the bearing inputs are set to zero, the corresponding bearing pointer is positioned at the lubber line regardless of the position of the compass card. When the bearing inputs are increased, the corresponding bearing pointers rotate clockwise with respect to the compass card. The software subtracts the vehicle heading from these two bearings to assure appropriate indications. There is no reversal of the pointer movement when it passes through the 360 degrees in either direction.

When Entry is selected, the primary range indicator ("Pri Miles") shows the spherical range to WP 2 on the primary runway via WP 1 for NEP. The secondary range ("sec miles") indicator shows spherical range to WP 2 on the secondary runway via WP 1 for NEP.

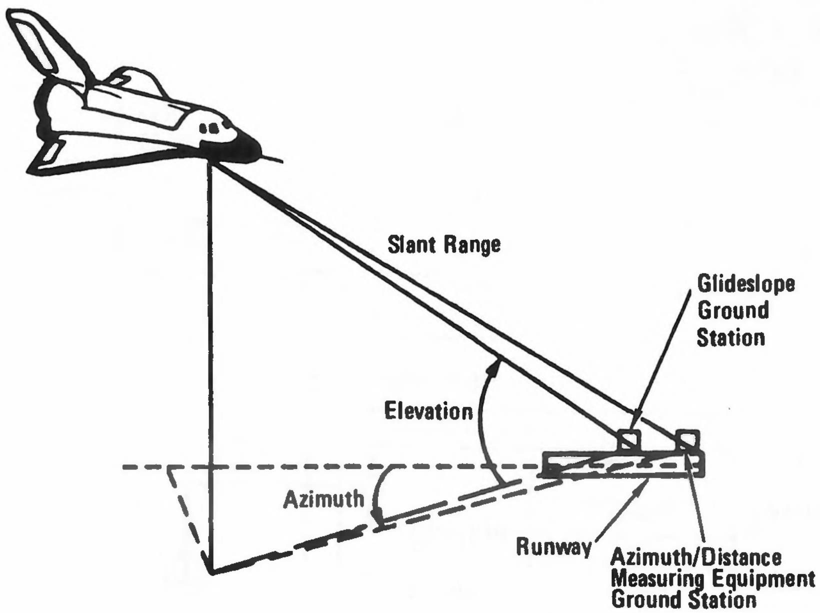
When in TAEM, the primary range is the horizontal distance to WP 2 on the primary runway via WP 1 for NEP and the secondary range is the horizontal distance to the center of the selected heading alignment cylinder for the primary runway. When in Approach and Landing, both primary and secondary range indicators show horizontal distance to WP 2 on the primary runway. The range displays are on the magnetic wheels in the upper right and left corners of the HSI face. Each display ranges from 0 to 3999 nautical miles. Both indicators use the same range data.

The bearing ("Brg") flag indicates that the heading, primary or secondary bearing, is invalid; the "CI" flag indicates that the course deviation display is invalid; and the glide slope ("GS") flag indicates that the glide slope deviation display is invalid.

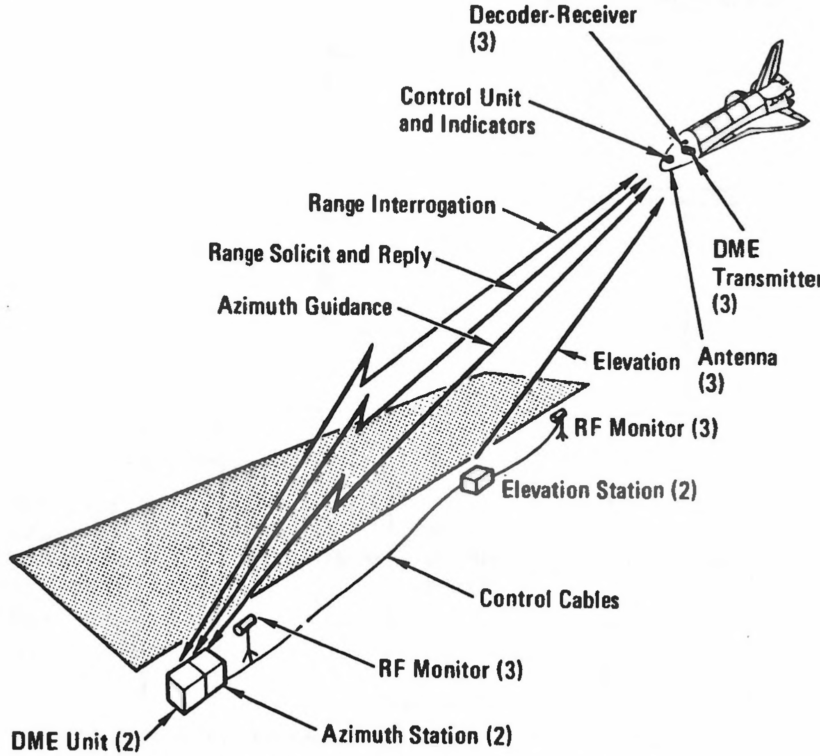
Red and white diagonally striped flags are used to obscure the range indicators whenever their displays are invalid. An "Off" flag indicates that power is off or less than 18 volts.



Microwave Landing ...m Interfaces



Microwave Landing System

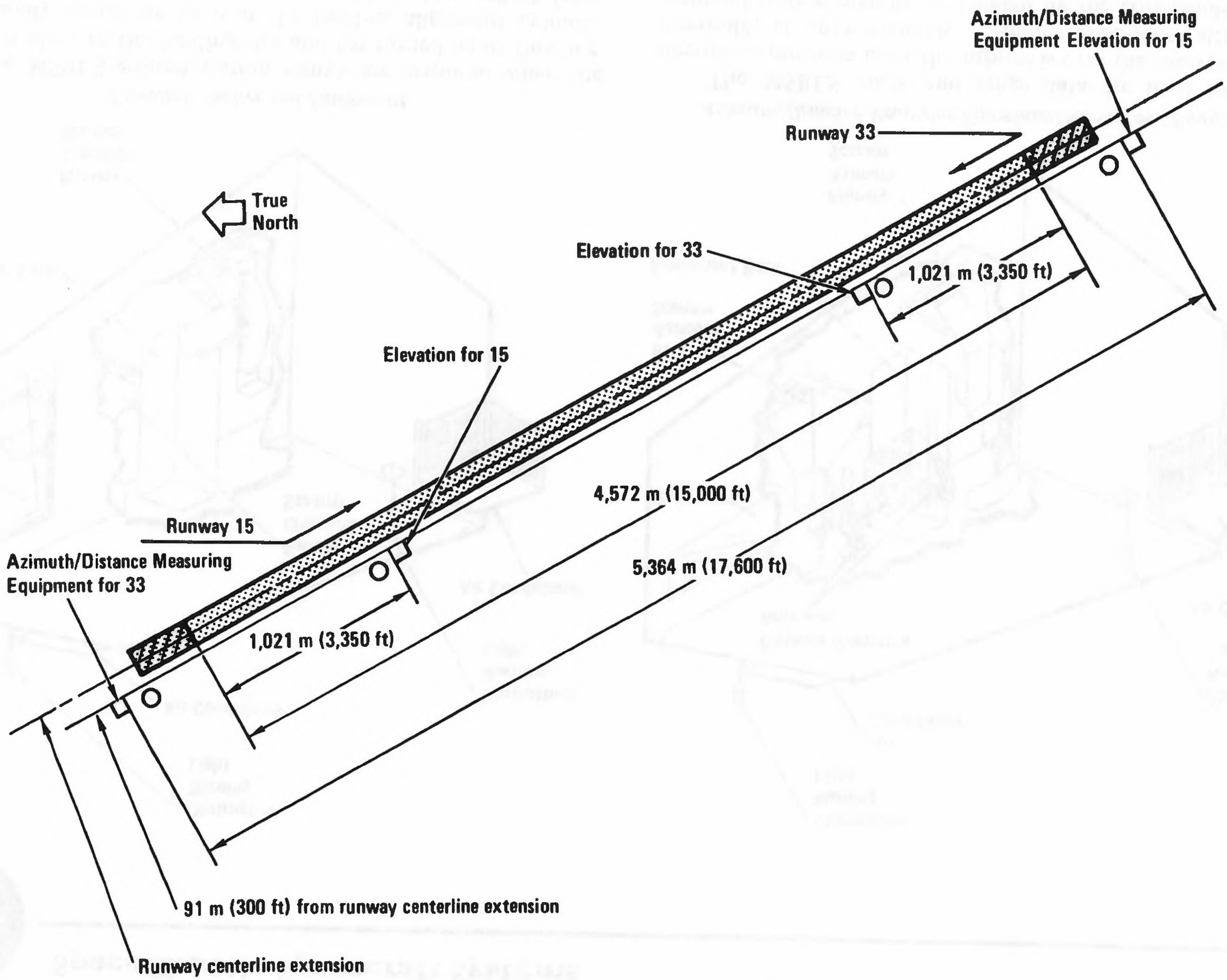


MSBLS Major Components and RF Links

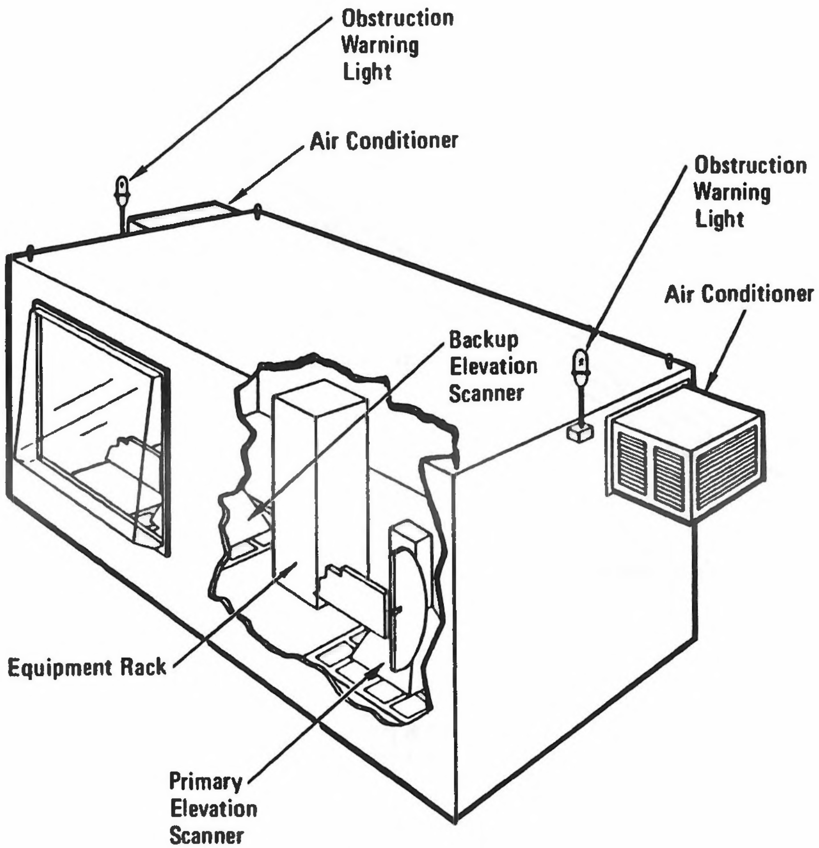
MICROWAVE SCAN BEAM LANDING SYSTEM. The MSBLS or MLS (microwave landing system) is an airborne Ku-band receiver-transmitter landing and navigation aid with decoding and computational capabilities. When the channel (specific frequency) associated with the targeted runway approach is selected, the airborne portion of the MSBLS receives elevation, azimuth, and range data from the ground station. MSBLS is used during TAEM and approach and landing phases of the flight and return-to-launch-site (RTLS) abort.

The orbiter is equipped with three independent MSBLS sets. Each consists of a Ku-band antenna RF assembly and a decoder assembly. Each Ku-band receiver-transmitter with its decoder and data computation capabilities determines the elevation angle, azimuth angle, and range of the orbiter with respect to the MSBLS ground station. The MSBLS provides highly accurate three-dimensional position information to the orbiter to compute steering commands which maintain the orbiter on its proper flight trajectory. The three Ku-band antennas on the orbiter are located on the upper forward fuselage nose area. The MSBLS and decoder assembly are located in the crew compartment mid-deck avionics bays and are convection cooled.

The ground portion of the MSBLS consists of two shelters: an elevation shelter and an azimuth/DME (distance measuring equipment) shelter. The elevation shelter is located near the projected touchdown point and the azimuth/DME shelter near the far end of the runway. Both ends of the runway are instrumented to enable landing in either direction.



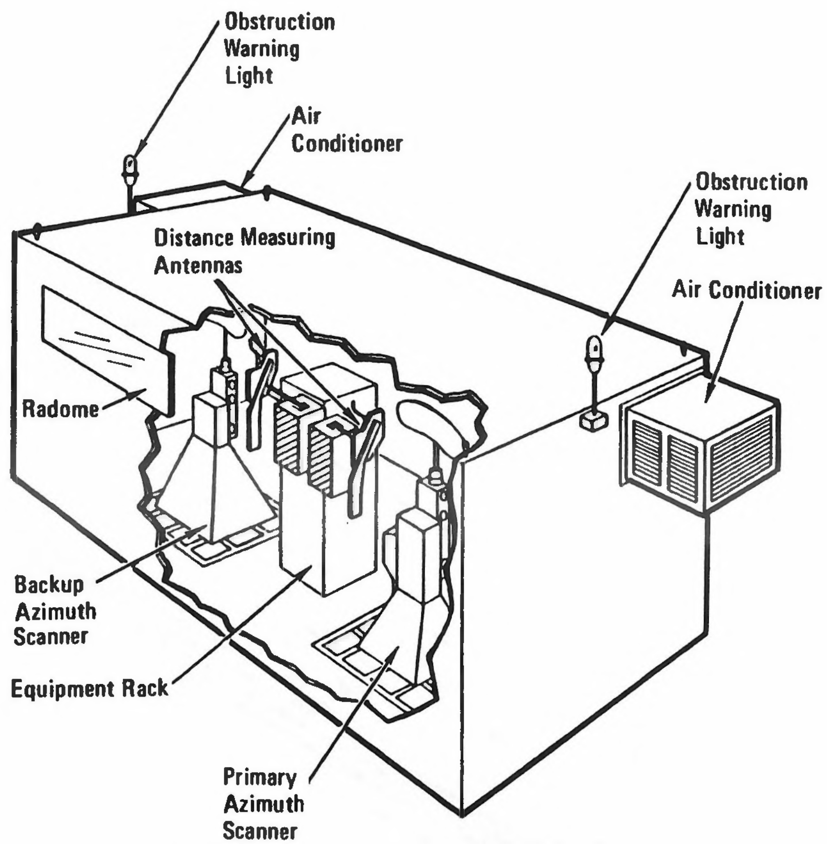
MSBLS Instrument Locations at KSC



Elevation Shelter and Equipment

The MSBLS ground station signals are acquired when the orbiter is close to the landing site and has turned on its final leg. This usually occurs on or near the heading alignment cylinder about 8 to 12 nautical miles (9 to 13 statute miles) from touchdown and at an altitude of approximately 5486 meters (18,000 feet).

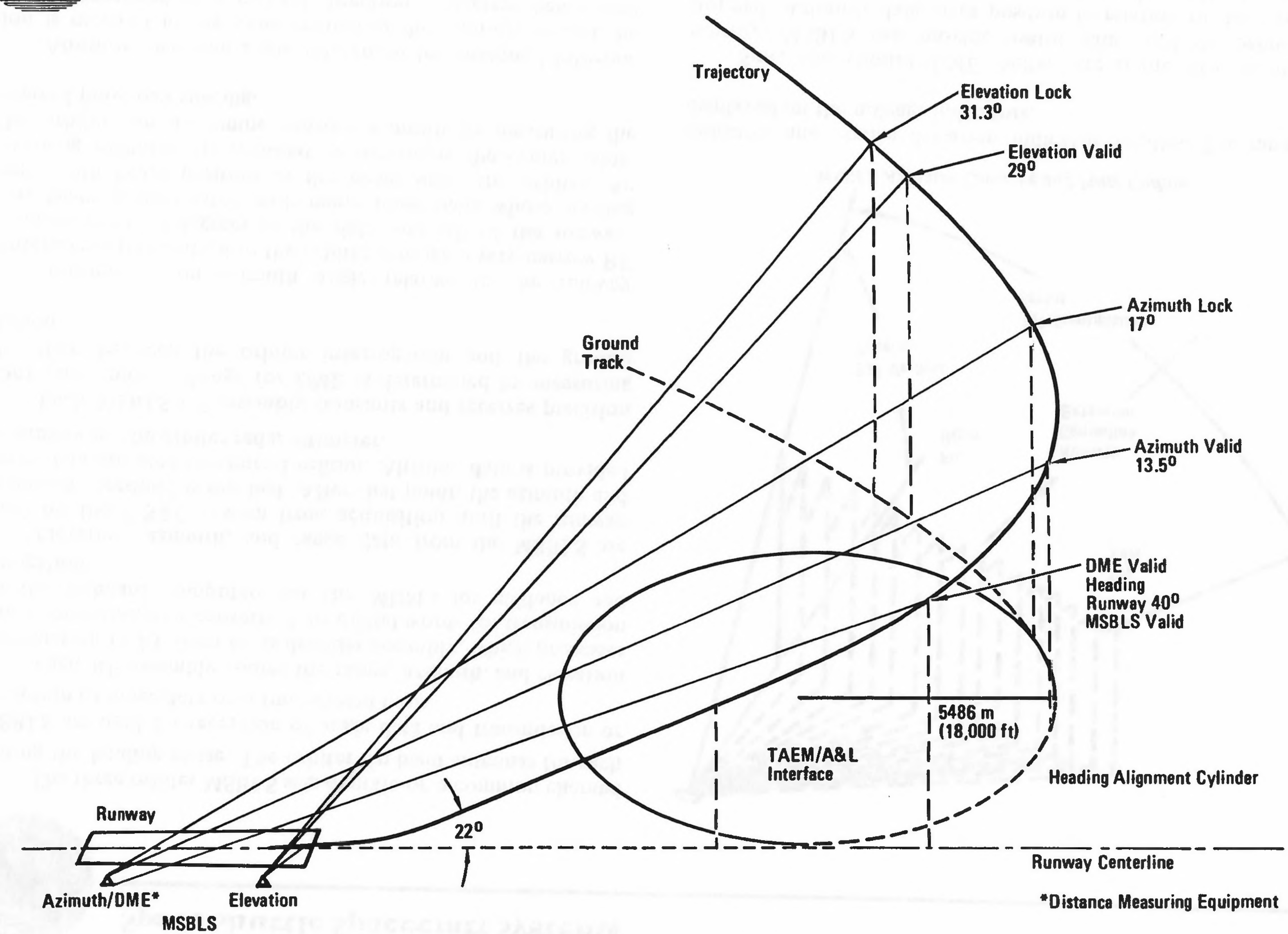
Final tracking occurs at the TAEM-"Autoland" interface at approximately 3048 meters (10,000 feet) altitude and eight nautical miles (9 statute miles) from the azimuth/DME station.



Azimuth/Distance Measuring Equipment Shelter and Equipment

The MSBLS angle and range data are used to compute steering commands until the orbiter is over the runway approach threshold, at approximately 30-meter (100-foot) altitude. The autoland system may be overridden by the commander or pilot.

The HSI's show deviations from the selected glide slope and azimuth. When the orbiter is over the runway threshold, the radar altimeter is used to provide elevation (pitch) guidance. Azimuth/DME data is used during the landing rollout.



Microwave Scan Beam Landing System

The three orbiter MSBLS sets operate on a common channel during the landing phase. The orbiter Ku-band antennas for each MSBLS are used for reception of angle data and transmission or reception of range data on a time-shared basis.

Each RF assembly routes the range, azimuth, and elevation information in RF form to its decoder assembly, which processes the information and converts it to digital words for transmission to the onboard computers via the MDM's for guidance and navigation.

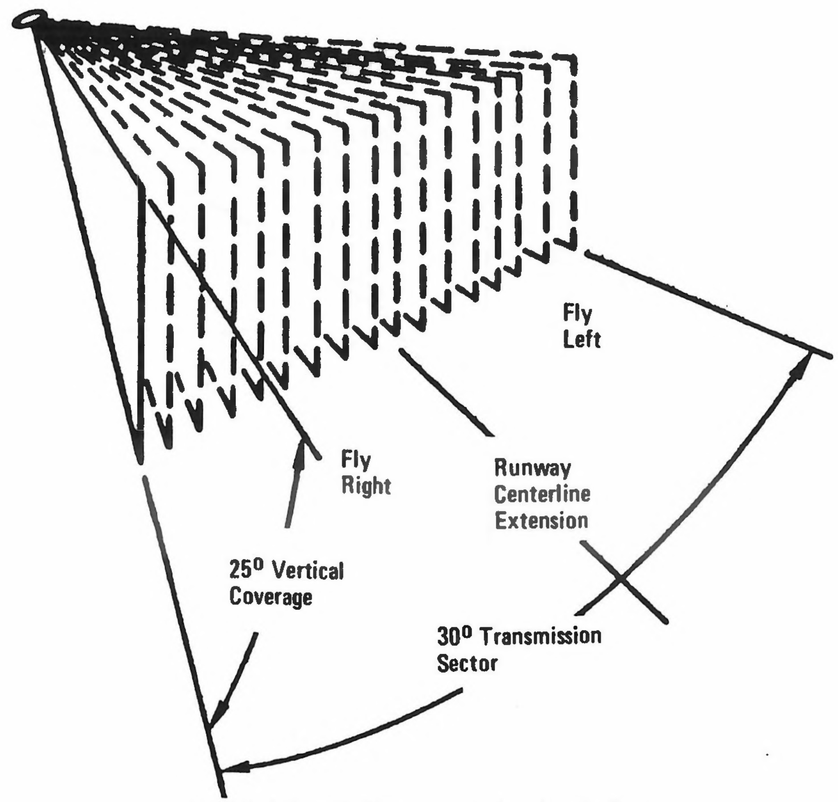
Elevation, azimuth, and range data from the MSBLS are used by the GN&C system from acquisition until the runway approach threshold is reached. After that point, the azimuth and range data are used to control rollout. Altitude data is provided separately by the orbiter radar altimeter.

Each MSBLS RF assembly transmits and receives precision slant range pulses. Range for DME is determined by measuring the time between the orbiter interrogation and the ground station reply.

Information on azimuth angle relative to the runway centerline is transmitted to the orbiter through a very narrow RF beam scanned 15 degrees to the right and left of the runway. This beam is modulated with many pulse pairs whose spacing varies with beam position as the beam scans the orbiter. An averaging operation is required to determine the center value. The orbiter can determine runway azimuth by measuring the received pulse pair spacing.

Absolute elevation angle referenced to horizontal information is received in the same manner as the azimuth, except the beam is scanned in a vertical direction, 3 degrees below and 27 degrees above the runway plane.

In addition, the position of the orbiter with respect to the runway is displayed on the HSI's. Elevation and azimuth are shown relative to a computer-derived glide slope on glide slope

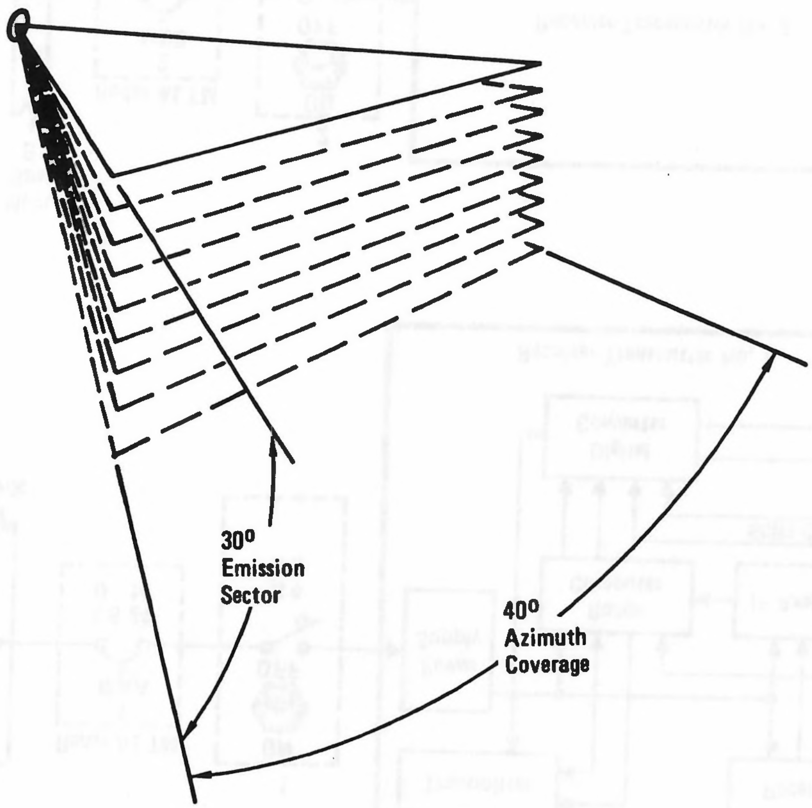


MSBLS Azimuth Coverage and Pulse Coding

indicator and course deviation indicator needles. The range is displayed on the mileage indicators.

Since the azimuth/DME shelters are at the far ends of the runway, MSBLS can provide useful data until the orbiter is stopped. Azimuth data gives position in relation to the runway centerline and the DME gives distance from the orbiter to end of runway.

Each MSBLS set has channel select thumbwheels on the flight deck display and control panel which allow the flight crew



MSBLS Elevation Coverage and Pulse Coding

to manually select the channel for the ground station at the selected runway.

Redundancy management mid value selects azimuth and elevation angles for processing of Nav data. The three MSBLS sets are ,compared to determine if a significant difference is detected in one against the other.

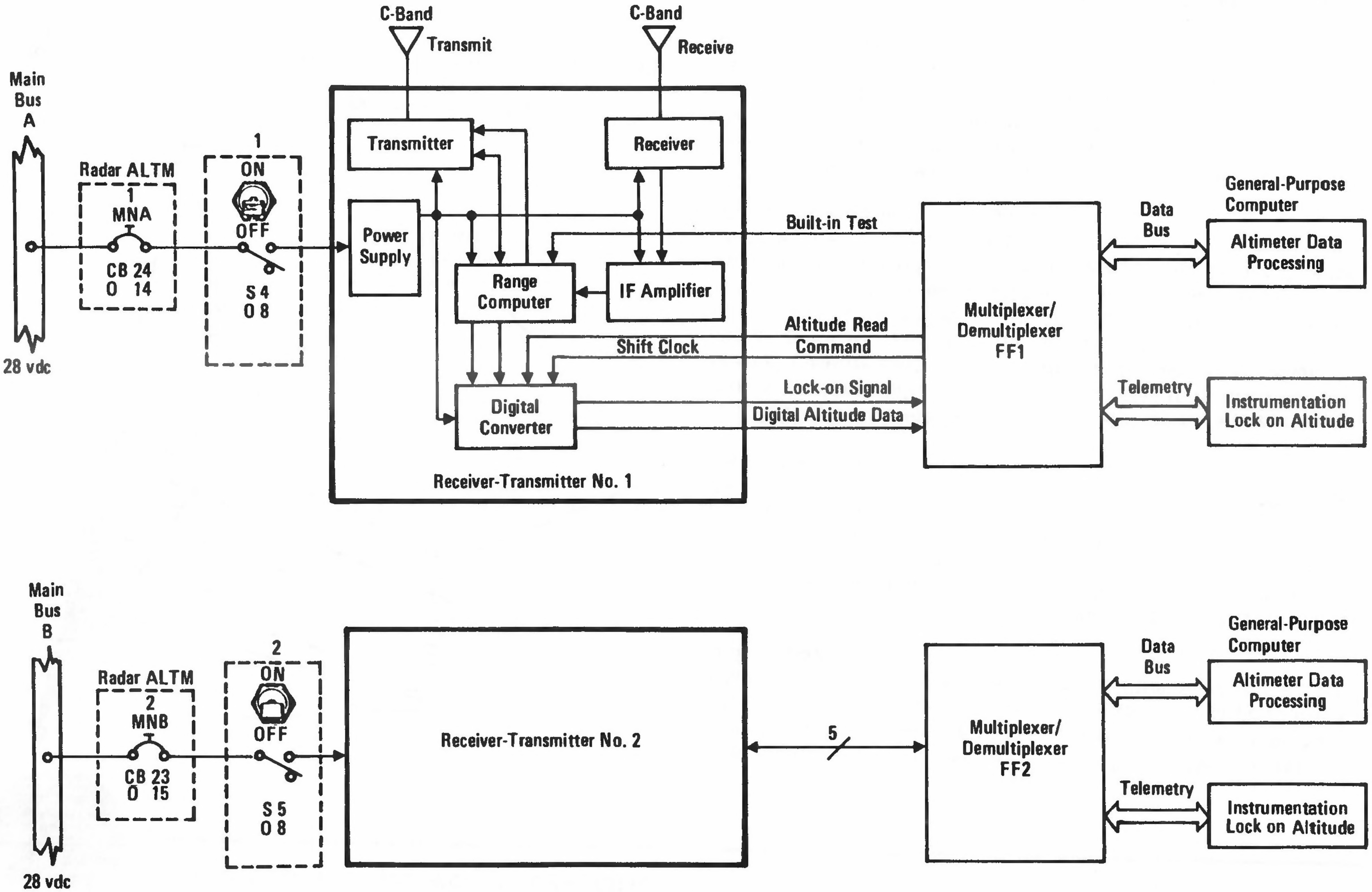
When data from all three MSBLS sets are valid, redundancy m-- -gement selects middle values of three ranges, azimuths, and three elevations. With two good MSBLS sets, the two ranges, azimuth, and elevations are averaged. With only one good MSBLS set, its range, azimuth, and elevation are passed for display. When a fault is detected the "SM Alert" light is illuminated, and a CRT fault message shown.

Each MSBLS decoder assembly is 20.9 centimeters (8.25 inches) in height, 12.7 centimeters (5 inches) in width, 41 centimeters (16.158 inches) in length, and weighs 7.9 kilograms (17.5 pounds). The RF assembly is 17.78 centimeters (7 inches) in height, 8.89 centimeters (3.50 inches) in width, 26 centimeters (10.25 inches) in length, and weighs 2.72 kilograms (6 pounds).

RADAR ALTIMETER. The radar altimeter is a low-altitude terrain tracking and altitude-sensing system. It is based on the precise measurement of time required for an electromagnetic energy pulse to travel from the orbiter to the nearest object on the ground below and return during altitude rate changes up to 609 meters per second (2000 fps). This enables tracking of mountain or cliff sides ahead of adjacent to the orbiter if these obstacles are nearer than the ground below, and provides warning of rapid changes in absolute altitude.

The radar altimeter is the primary sensor of the "Autoland" system and for touchdown guidance after the orbiter has crossed the runway threshold from an altitude of 30 meters (100 feet) down to touchdown. Before that point, radar altimeter data is displayed on the crew's AVVI's (altitude/vertical velocity indicators) up to 5000 feet.

The orbiter contains two independent radar altimeter systems, each with a transmitting and receiving antenna. The systems are independent and can operate simultaneously without affecting each other. The four C-band antennas are located on the lower forward fuselage. The two receiver/transmitters are located in the mid-deck forward avionics bays and are convection cooled.

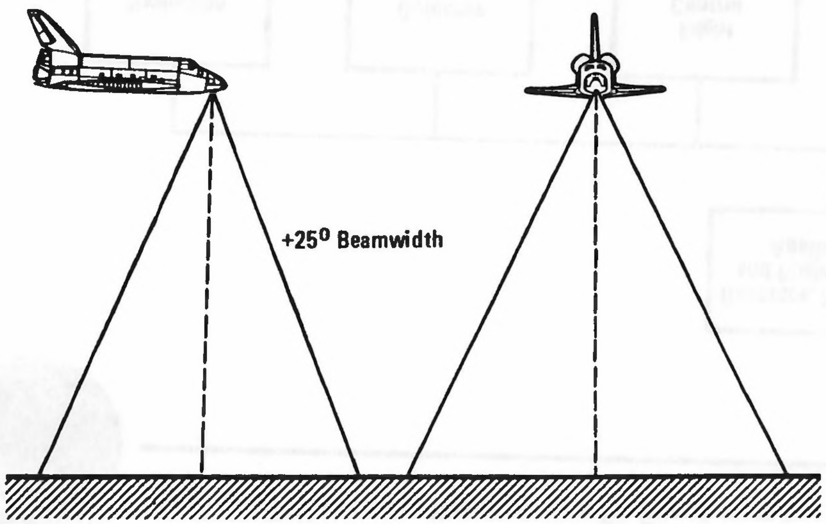


Radar Altimeter

Altitude range signals are analog voltages proportional to the elapsed time required for return of the ground pulse, which is a function of height or distance to the nearest terrain. The onboard computers process the data for automatic flight control of orbiter altitude if in the autoland mode; otherwise, the data is used only for display on the AVVI's.

Since there are only two radar altimeters aboard the orbiter, the altitude data from the two units are averaged in the redundancy management for use in the autoland mode. The MSBLS elevation and range information then can be used down to 30 meters (100 feet).

Both commander and pilot stations have switches for selecting radar altimeter 1 or 2 for display on his AVVI. The display scale ranges from 0 to 5000 feet and the altitude is displayed on moving tapes. Above 5000 feet, the scale will be pegged. Below 5000 feet, the indicator will show actual altitude,



Radar Altimeter Antenna Beamwidth Coverage

orbiter to ground. At 1500 feet the RA indicator changes scale. The "RA Off" flag will appear if there is a loss of power, loss of lock, data good-bad, or after three communication faults.

Each radar altimeter receiver-transmitter is 7.9 centimeters (3.13 inches) in height, 18.82 centimeters (7.41 inches) in length, 9.7 centimeters (3.83 inches) in width, and weighs 2 kilograms (4.5 pounds).

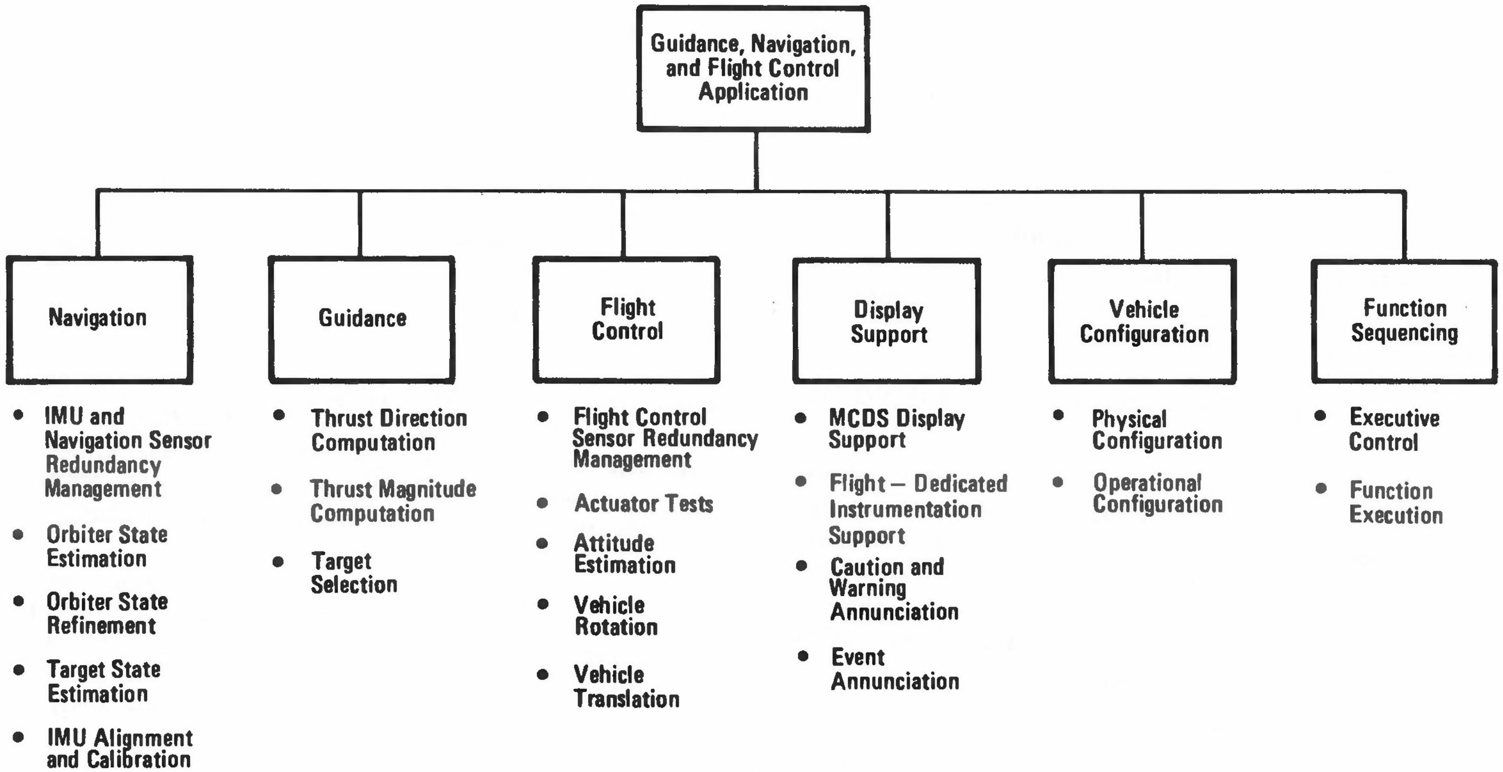
The contractors involved are; s AIL Culter Hammer, Farmingdale, NY (MSBLS); AiResearch Manufacturing Co., Garrett Corp., Torrance, CA (air data transducer assemblies); Hoffman Electronics Corp., Navigation Communication System Division, El Monte, CA (TACAN); Honeywell Inc., Minneapolis, MN (radar altimeter); Rosemount Inc., Eden Prairie, MN (air data sensor probes).

GUIDANCE, NAVIGATION, AND CONTROL

The guidance, navigation, and control (GN&C) system responds to software commands to provide vehicle control and also furnishes sensor and controller data to the GN&C software so that it can compute the commands.

The orbiter's five flight computers are organized into a redundant set of four which form the primary flight system, with the fifth computer used as the backup flight system in the initial development flight tests. The fifth computer operates independently of the other four.

These computers interface with various systems. The orbiter's flight forward and flight aft multiplexer/demultiplexers (MDM's) and data bus serve as the conduit for signals going to and from such units as the master timing unit, the sensors that provide velocity and attitude information, orbiter propulsion systems, aerodynamic surfaces, and displays and controls.



GN&C Structure

The GN&C system consists of two modes of operation: auto and manual (control stick steering). In the automatic mode, the primary flight system (PFS) essentially allows the computers to do all the flying, with the flight crew selecting the various operational sequences. The crew may control the vehicle in the control stick steering mode by using hand controls such as the rotation hand control (RHC), translation hand control (THC), speedbrake/thrust controller (SBTC), and rudder pedals; however, the commands issued by the crew must pass through and be issued by the computers.

There are no direct mechanical linkages by which the flight crew can manipulate the various propulsion systems or aerodynamic aerosurfaces; thus, the orbiter is an entirely digitally controlled, fly-by-wire vehicle.

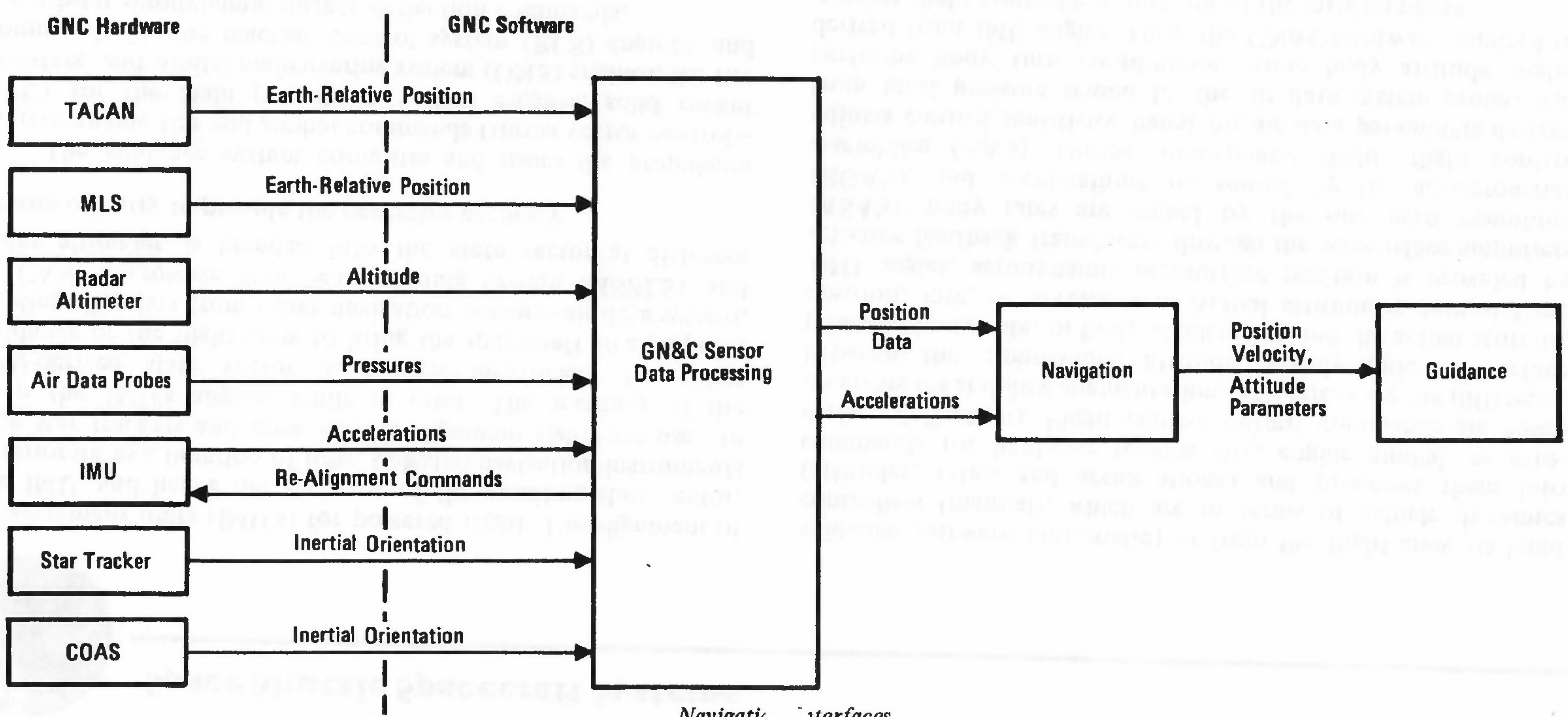
The multi-function cathode ray tube (CRT) display system presents GN&C and system status information to the flight crew and allows the crew extensive interaction with the computers via keyboard units on the flight deck display and control panel.

The mass memory units contain the primary and backup flight software to be loaded into the computers and communicate directly with the computers.

The computers also transmit pulse code modulation (PCM) telemetry data to the ground stations for monitoring of the internal state of the computers' status and operational information of all subsystems and to report progress of the flight.

The ground launch processing system (LPS) interfaces with the orbiter before launch via the launch umbilical, which provides direct access to the onboard computers' memory. In addition to performing vital ground launch sequencing tasks, the LPS, using a general memory read/write capability, can monitor and modify the onboard computer core (such as updating launch pad initial conditions).

The navigation system maintains an accurate estimate of the vehicle position and velocity, referred to as a state vector. From position and velocity, other parameters (acceleration, angle of attack) are calculated for use in guidance and for display to the crew. The current state vector is mathematically determined by integrating the equation of motion for coasting flight and by using the acceleration of the vehicle as sensed by the inertial



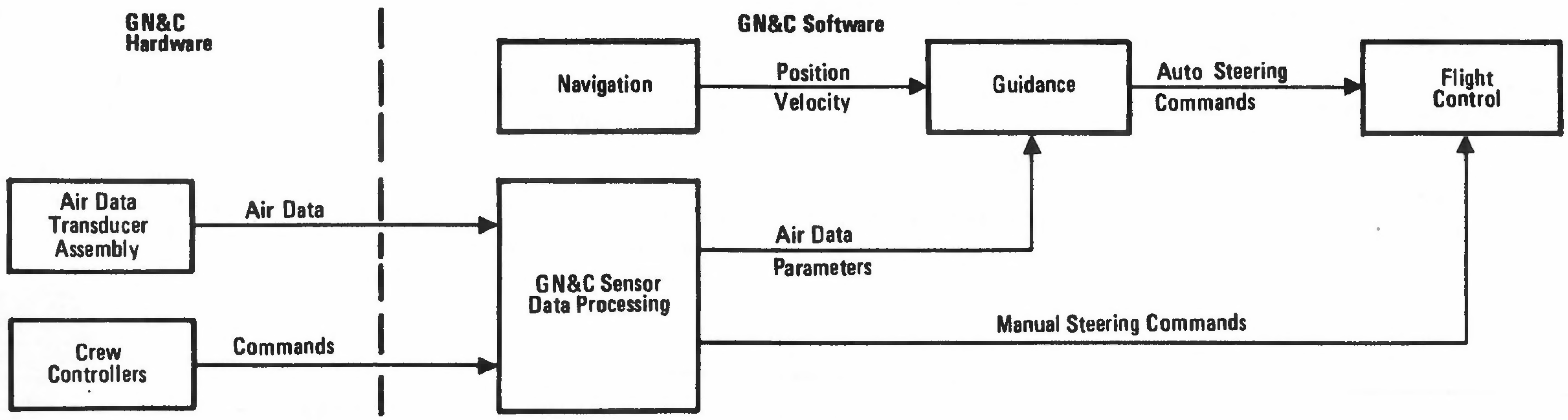
Navigati- -terfaces

measurement units (IMU's) for powered flight. The alignment of the IMU, and hence the accuracy of the resulting state vector, deteriorate as a function of time. Celestial navigation instruments (the star trackers and crew optical alignment sight) are used to keep the IMU's aligned while in orbit. The accuracy of the IMU-derived state vector is, however,insufficient for either guidance or the flight crew to bring the spacecraft to a pinpoint landing. So, data from other navigation sensors–air data system, TACAN, microwave scan beam landing system (MSBLS), and radar altimeter–is blended into the state vector at different phases of entry to provide the necessary accuracy.

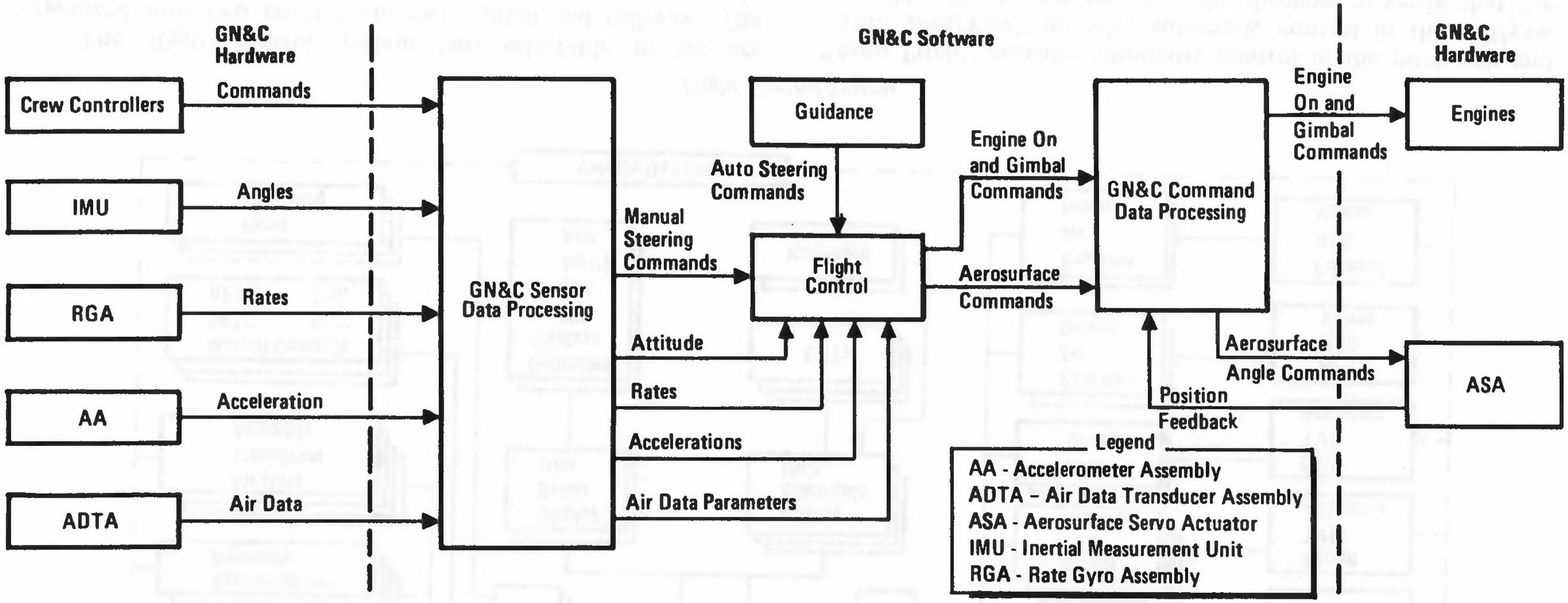
The guidance system computes and issues the propulsion system engine fire and gimbal commands (thrust vector control–TVC) for the main propulsion system engines, solid rocket boosters, and orbital maneuvering system (OMS) engines, the fire commands for the reaction control system (RCS) engines, and the orbiter aerodynamic surface deflection commands.

Flight control includes attitude processing and a digital autopilot which provides steering, thrust vector, attitude, and aerosurface control. Flight control receives commands from guidance software (automatic) or from the flight crew via hand controllers (manual), which are in terms of vehicle dynamics (attitudes, rates, and accelerations) and processes them into commands for hardware (engine fire, engine gimbal, or aero-surface deflection). Flight control output commands are based on errors for stability augmentation. The errors are the difference between the commanded attitude, aerodynamic aerosurface position, body rate, or body acceleration and the actual attitude, position, rate, or acceleration. Actual attitude is derived from IMU angles; aerodynamic aerosurface position is provided by actuator feedback transducers through the aerosurface amplifiers (ASA's); body rates are sensed by the rate gyro assemblies (RGA's); and accelerations are sensed by the accelerometer assemblies (AA's). During atmospheric flight, flight control adjusts control sensitivity based on air data parameters derived from local pressure sensed by the air data system probes and performs body turn coordination using body attitude angles derived from IMU angles. Thus, the GN&C hardware required to support flight control is a function of the mission phase.

Steering commands used by the flight control software are augmented by the guidance software or manually commanded



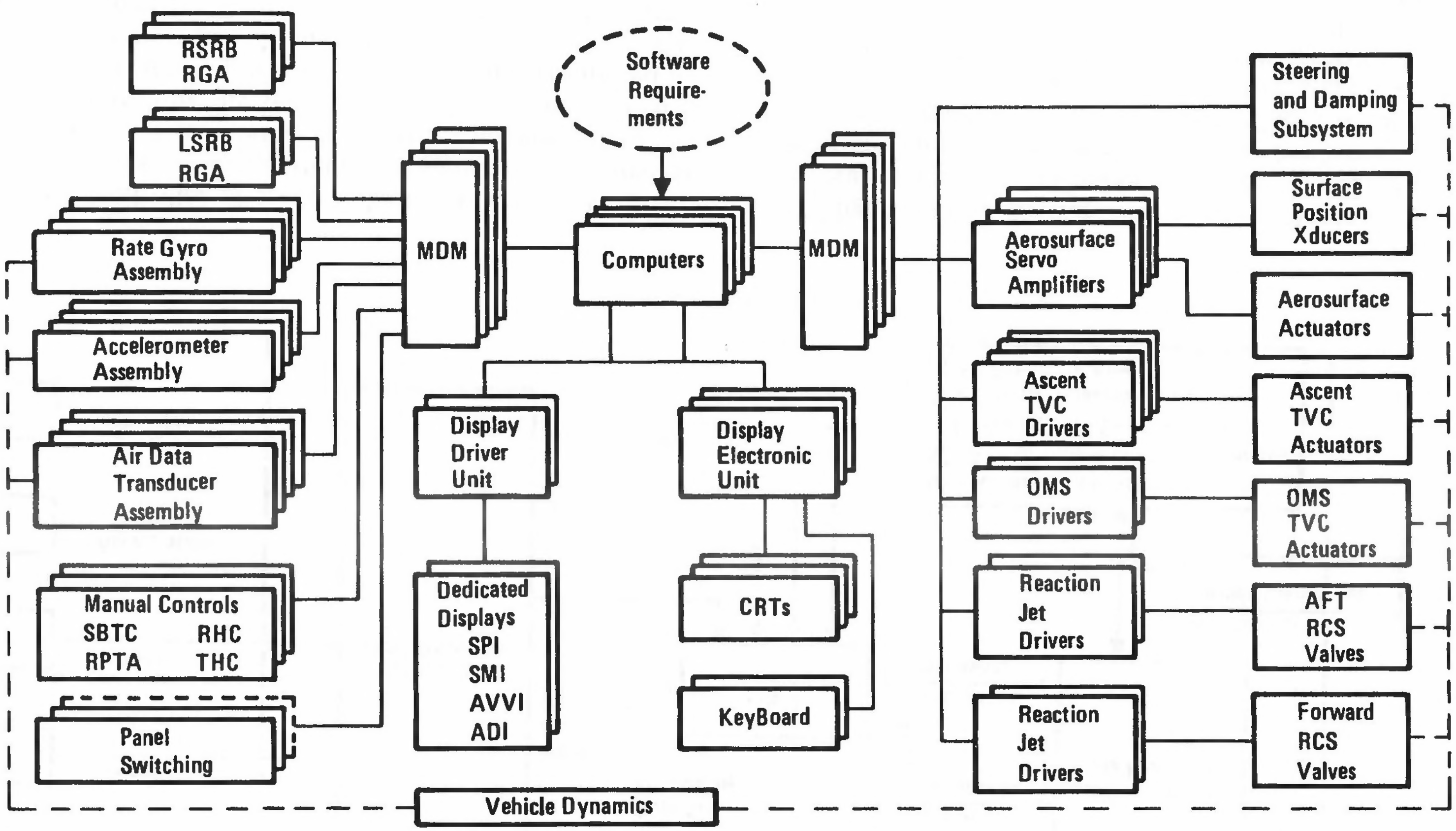
Guidance Interfaces



Flight Control Interfaces

using the hand controller or speedbrake/thrust controller. When flight control software uses the steering commands computed by guidance software, it is termed automatic guidance; when the flight crew is controlling the vehicle via the hand controllers it is called control stick steering (CSS). The commands computed by guidance are those required to get from the current state (position and velocity) to a desired state (specified by target conditions, attitude, airspeed, runway centerline). The steering commands take the form of translational and rotational rates and accelerations. Guidance receives the current state from navigation software. The desired state or targets are part of the initialized software load and can be changed in flight.

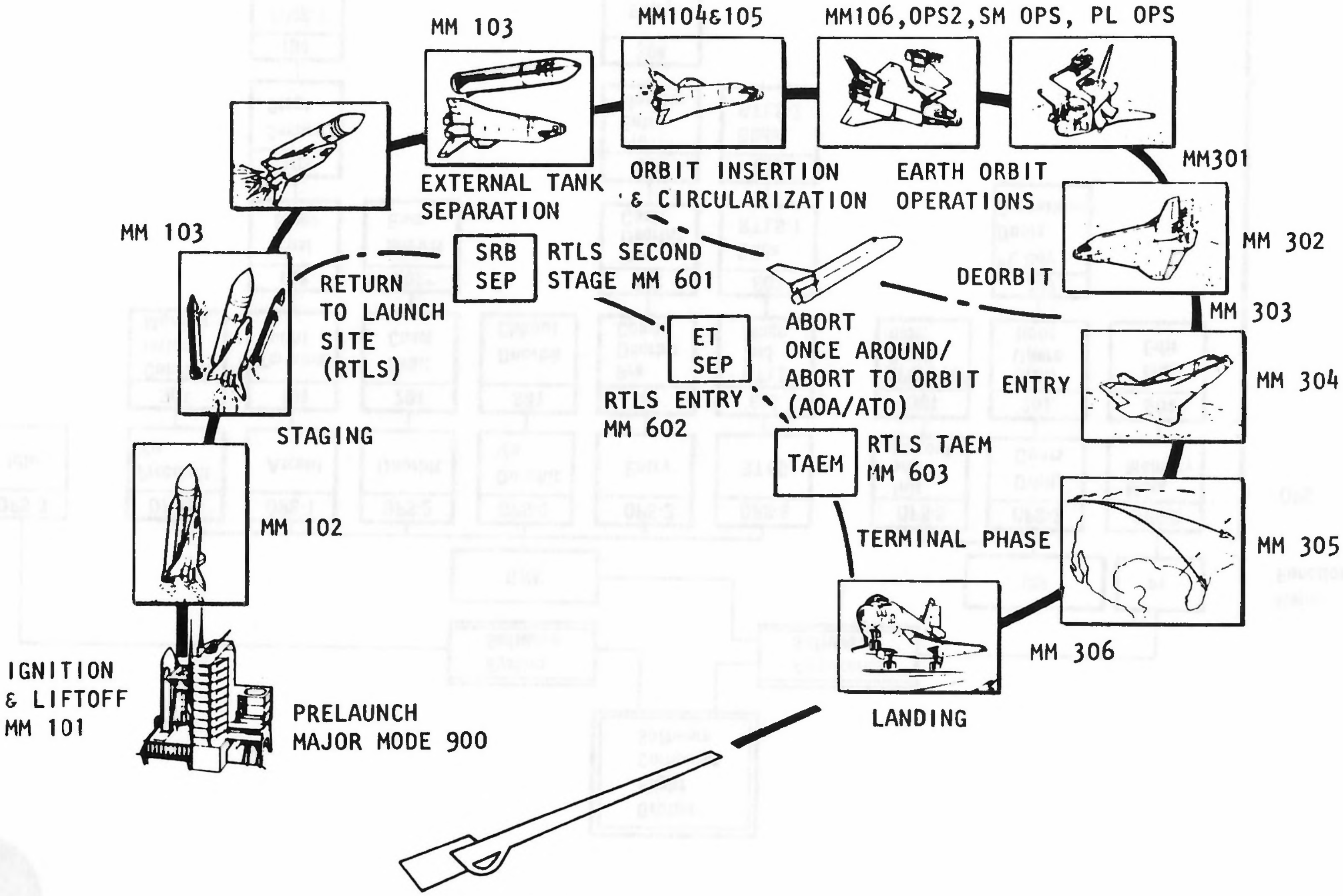
Flight control software controls the vehicle via the selection of operational sequences/major modes (OPS/MM) in the onboard computers; this is accomplished via the keyboards at the flight deck commander and pilot stations. There is an operational sequence/major mode for prelaunch, first-stage ascent, second-stage ascent, orbit insertion OMS-1, orbit insertion OMS-2, orbit coast, maneuver execute, orbit checkout, deorbit maneuver, deorbit maneuver coast, entry guidance, and TAEM (terminal area energy management) guidance. In addition, the flight control system consists of two selectable modes–Auto and CSS–via pushbutton light indicators on the flight deck at the commander and pilot stations.

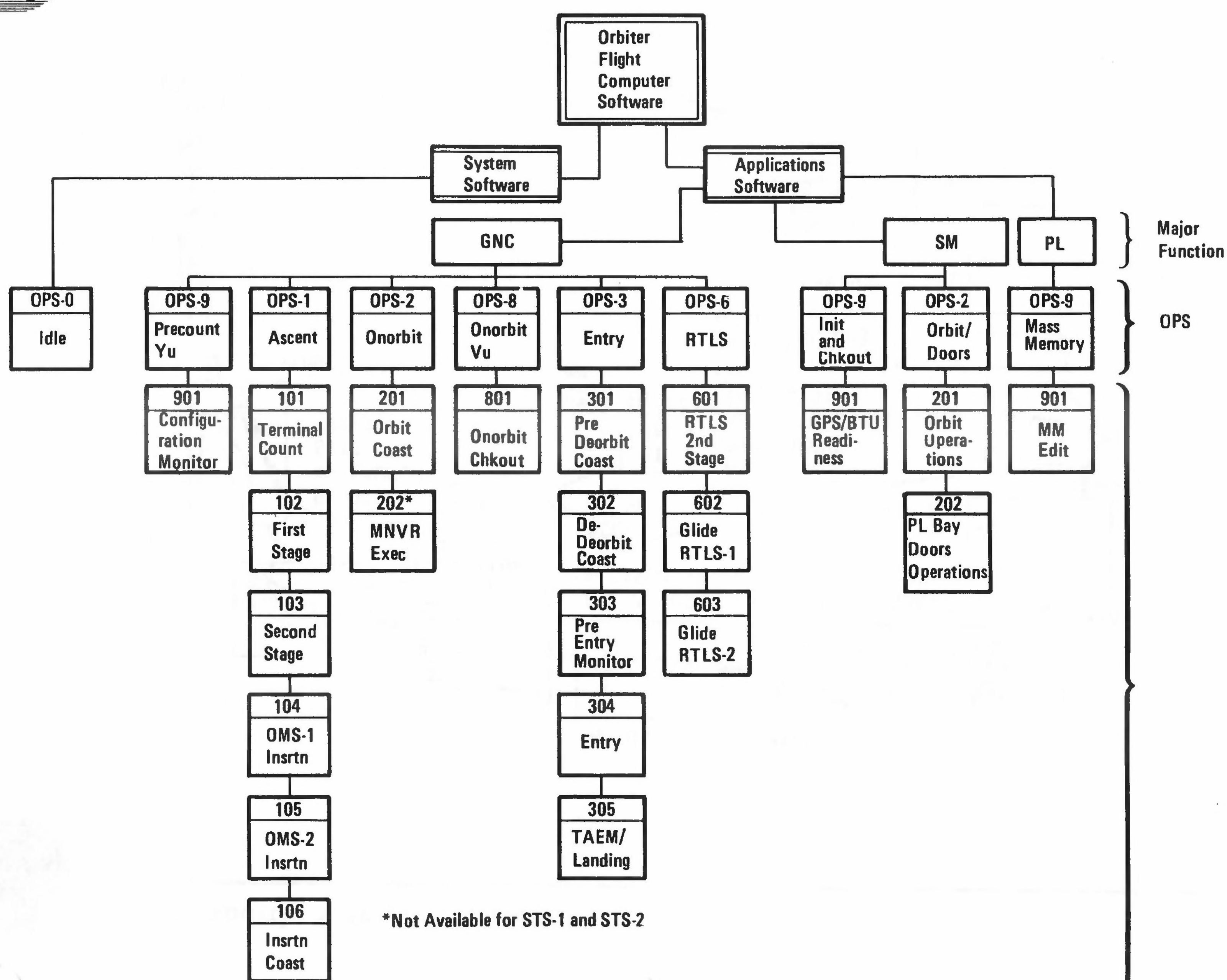


Flight Control System

The flight control system two selectable modes are organized into two control channels: pitch and roll/yaw. The pushbutton will illuminate a white light to indicate the mode selected. Each pushbutton is a triple redundant, momentary contact, nonlatching switch.

In the Auto mode, the GN&C provides automatic control of the vehicle through selection of the operational sequence major mode via the onboard keyboard, thus the flight control system, "Auto Pitch" provides automatic control in the pitch axis and "Auto Roll/Yaw" provides automatic control in the roll/yaw axes. The flight crew monitors the displays to verify that the vehicle is following the correct trajectory. In the GN&C Auto system, the computers process the vehicle motion sensors to obtain the dynamic state of the vehicle. The flight control system processes the flight control laws (equations) in response to guidance commands and commands the flight control system effectors.


MM 103
MM104&105
MM106,OPS2,SM OPS, PL OPS
EXTERNAL TANK SEPARATION
ORBIT INSERTION & CIRCULARIZATION
EARTH ORBIT OPERATIONS
MM301
MM 103
SRB SEP
RTLS SECOND STAGE MM 601
MM 302
RETURN TO LAUNCH SITE (RTLS)
DEORBIT
ET SEP
ABORT ONCE AROUND/ ABORT TO ORBIT (AOA/ATO)
MM 303
ENTRY
MM 304
RTLS ENTRY MM 602
STAGING
TAEM
RTLS TAEM MM 603
MM 102
TERMINAL PHASE
MM 305
MM 306
IGNITION & LIFTOFF MM 101
PRELAUNCH MAJOR MODE 900
LANDING



Orbiter Flight Co...ol Software

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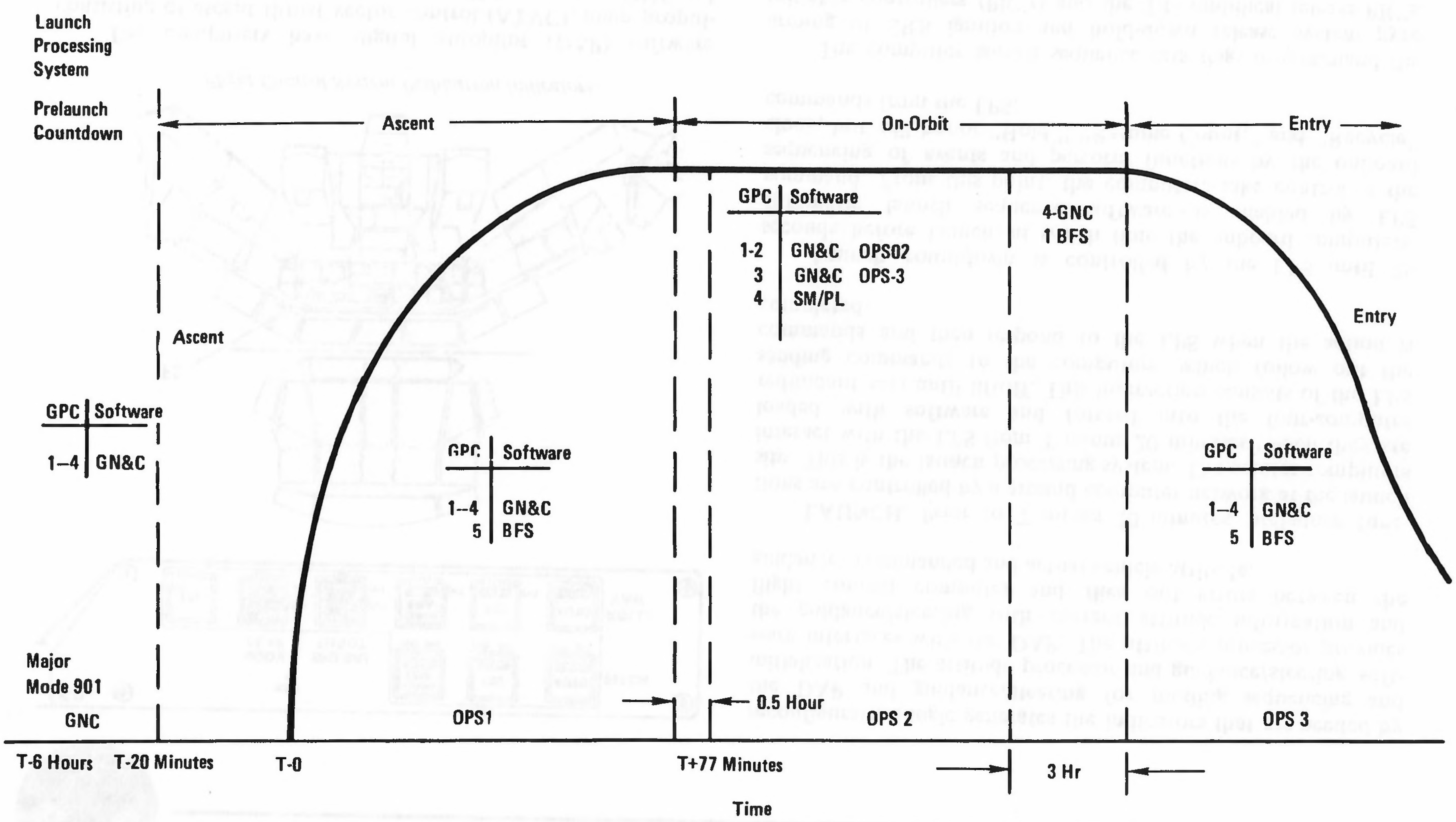
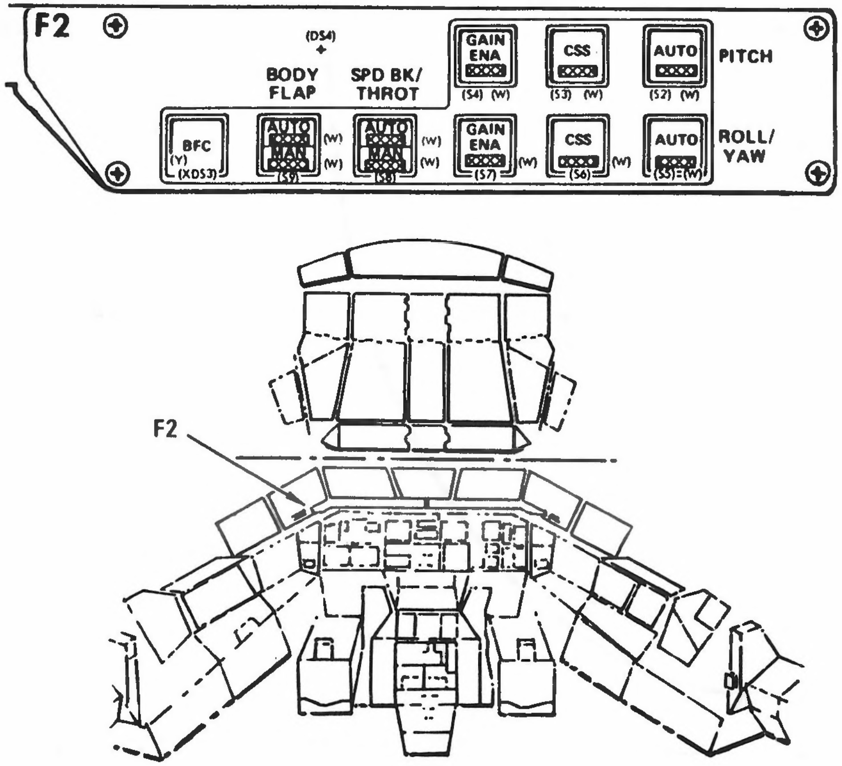


Orbiter Computer Configurati 'ominal Mission Scenario



Flight Control System Pushbutton Indicators

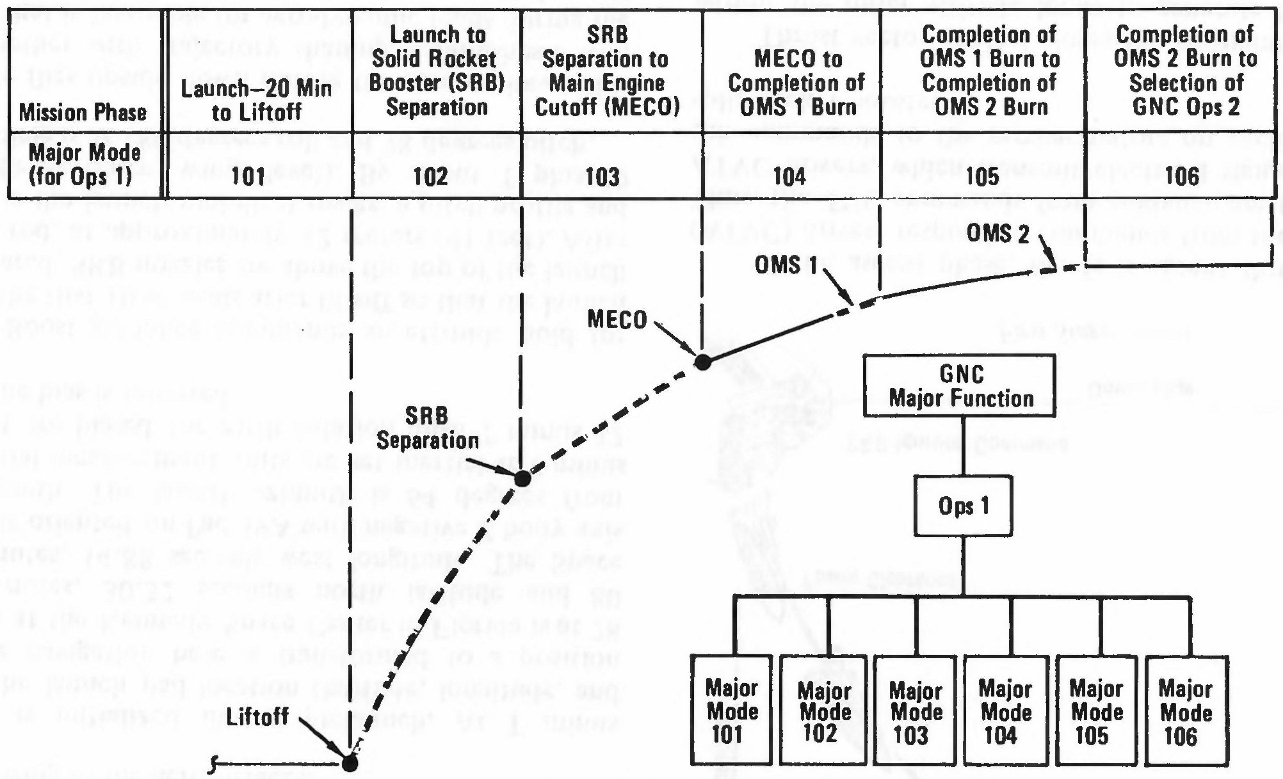
The computers have digital autopilot (DAP) software consisting of ascent thrust vector control (ATVC), main propulsion system (MPS) command processor, SRB processor, OMS and RCS processors, aerosurface control, and reconfiguration logic.

The DAP software generates acceleration profiles, trims, elevon load relief schedule, and scheduled gains as functions of earth relative velocity magnitude, mission elapsed time, and vehicle mass. The DAP reconfiguration logic is the bookeeper of the flight control software. In response to control mode changes, changes in failure status, and the occurrence of events, the reconfiguration logic generates the indicators that are needed by the DAP and guidance/steering for moding sequencing and initialization. The attitude processor and guidance/steering software interfaces with the DAP. The attitude processor provides the guidance/steering with current attitude information and flight control computes and flies out errors between the guidance–commanded and actual vehicle attitude.

LAUNCH. Prior to T minus 20 minutes, prelaunch functions are controlled by a ground computer network at the launch site. This is the launch processing system. The orbiter computers interact with the LPS from T minus 20 minutes (when they are loaded with software and formed into the four-computer redundant set) until liftoff. This interaction consists of the LPS sending commands to the computers, which follow out the commands and then respond to the LPS when the action is completed.

Launch countdown is controlled by the LPS until 25 seconds before launch, at which time the onboard computers' automatic launch sequence software is enabled by LPS command. From this point, the computers take control of the sequencing of events and perform functions by the onboard clock, but will honor "Hold," "Resume Count," and "Recycle" commands from the LPS.

The computer launch sequence sets flags to command the arming of SRB ignition and hold-down release system pyro initiator controllers (PIC's) and the T-O umbilical release PIC's. After a delay, the SRB ignition PIC voltages are monitored for acceptable levels. The hold-down release system PIC's and the T-O umbilical release system PIC's are monitored by the LPS. The computer launch sequence logic initiates a countdown "Hold" if the SRB ignition PIC voltages fall below an acceptable level at any time prior to issuance of the main engine start commands. If the SRB ignition PIC voltages are not acceptable, after the start commands are issued, the engines are shut down.



Ascent Major Modes

The computer launch sequence also controls certain critical main propulsion system valves and monitors the engine-ready indications from the main engines. The MPS start commands are issued by the onboard computers at T minus 3.8 seconds (staggered start – engine three, engine two, engine one – all approximately within one-fourth of a second) and the sequence monitors the thrust buildup of each engine. All three engines must reach the required 90 percent thrust within this time period or an orderly shutdown is commanded up to 4.6 seconds and safing functions are initiated.

Normal thrust build-up to the required 90 percent thrust level will result in the engines being commanded to the liftoff p[illegible]on at T-O, start of the onboard timer, termination of LPS polling, issue a Fire 1 command to arm the SRB's, and start a 2.64 second timer which allows the vehicle base bending load modes to initialize (movement of approximately 76 centimeters [30 inches] measured at the tip of the external tank-with movement towards the external tank).

When the 2.64 second timer has timed out, the Fire 2 command is issued which causes SRB ignition, T-O umbilical release, SRB hold-down release commands, reset of the onboard master timing unit to T-O, start of the onboard event timer, reset of the mission event timer, and modes the computers.

The DAP ascent configuration logic, upon ascent initialization, places the main engine nozzles in the start position, pla[illegible] the SRB nozzles in the null position, and places the orbiter aerosurfaces in the launch position. Upon receipt of the command to prepare main engines for liftoff, the engines are commanded to the launch position.

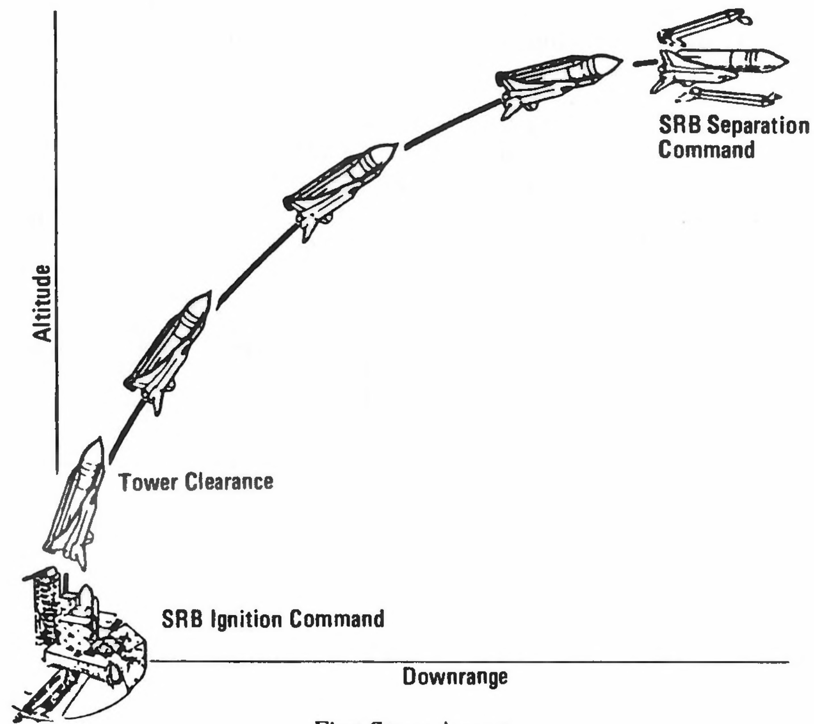
There are no guidance functions performed during prelaunch. Also, there are no active flight control functions, except for the commands to gimbal the engines for test and launch positions and slewing of the aerosurfaces.

Navigation is initialized during prelaunch. At T minus eight seconds, the launch pad location (latitude, longitude, and attitude) of the navigation base is transformed to a position vector. Pad 39A at the Kennedy Space Center in Florida is at 28 degrees, 36 minutes, 30.32 seconds north latitude and 80 degrees, 36 minutes, 14.88 seconds west longitude. The Space Shuttle vehicle is oriented on Pad 39A with negative Z body axis (tail) pointed south. The launch azimuth is 64 degrees from north. The inertial measurement units are set inertial at T minus 20 minutes, but are biased for earth rotation until T minus 12 seconds, when the bias is removed.

ASCENT. Boost guidance commands an attitude hold for approximately the first 10 seconds after liftoff so that the launch pad tower is cleared, SRB nozzles are above the top of the launch tower lightning rod, at approximately 12 meters (41 feet). After the vehicle clears the launch umbilical tower; a pitch profile and a roll begins (heads down, wings level). By about T plus 20 seconds, the vehicle is at 180 degrees roll and 78 degrees pitch.

The vehicle flies upside down during the ascent phase. This orientation, together with trajectory shaping, establishes a trim angle of attack that is favorable for aerodynamic loads during the region of high dynamic pressure (max q) and yet results in a net positive load factor.

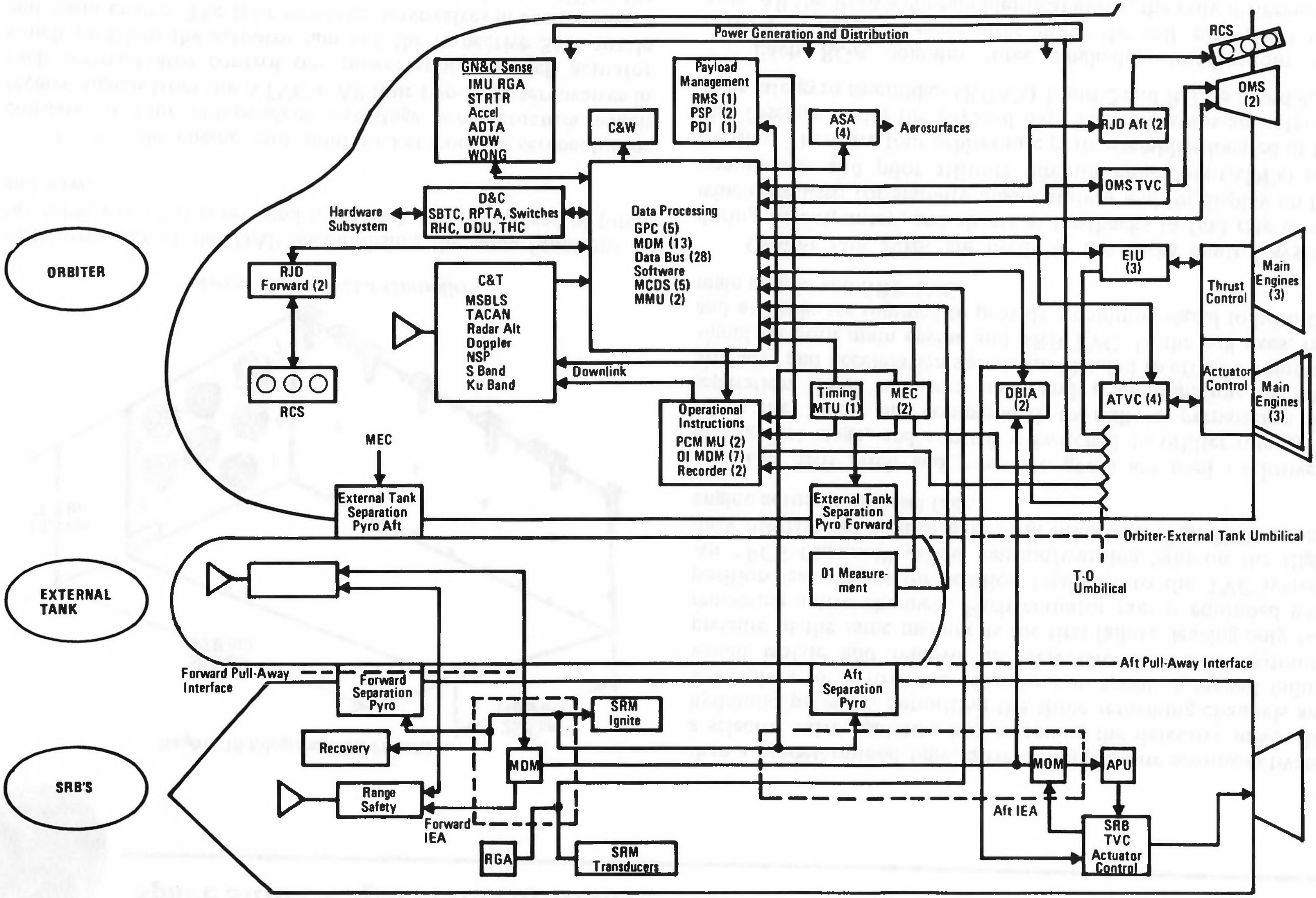
Thrust vector control is the hub of the wheel of flight control.



First Stage Ascent

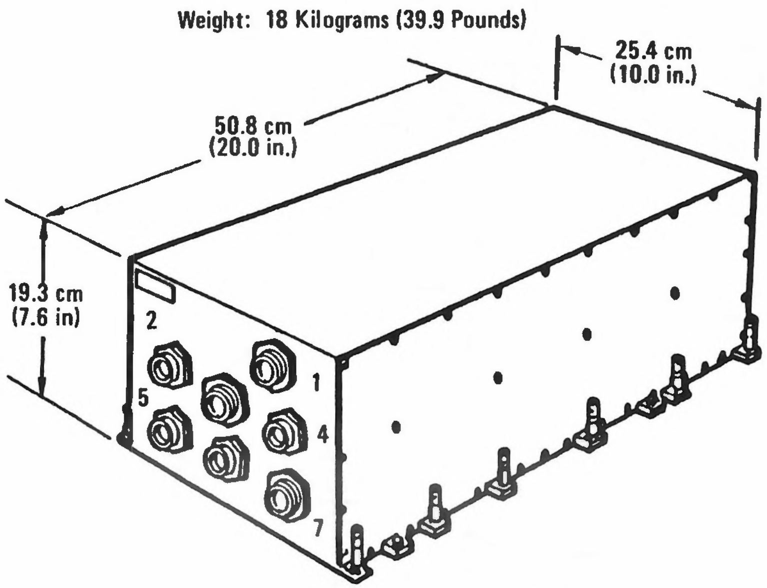
In the ascent phase, the four ascent thrust vector control (ATVC) drivers respond to commands from the guidance system Thus, the TVC commands from guidance are transmitted to the ATVC drivers, which transmit electrical signals proportional to the commands to the servoactuators on each main engine and solid rocket booster.

Thrust vector control closes the acceleration and rate loops within the outer attitude loops to generate body axis attitude error rates which eventually are flown out by the main engines and SRB's. The main propulsion system processor of the DAP converts body axis and attitude error signals that are generated in TVC, into pitch and yaw engine bell deflection commands. The



Power Generation and Distribution
RCS
GN&C Sense
IMU RGA
STRTR
Accel
ADTA
WOW
WONG
C&W
Payload Management
RMS (1)
PSP (2)
PDI (1)
ASA (4)
Aerosurfaces
RJD Aft (2)
OMS (2)
OMS TVC
Hardware Subsystem
D&C
SBTC, RPTA, Switches
RHC, DDU, THC
Data Processing
GPC (5)
MDM (13)
Data Bus (28)
Software
MCDS (5)
MMU (2)
EIU (3)
Thrust Control
Main Engines (3)
ORBITER
RJD Forward (2)
C&T
MSBLS
TACAN
Radar Alt
Doppler
NSP
S Band
Ku Band
Downlink
RCS
Timing MTU (1)
MEC (2)
DBIA (2)
ATVC (4)
Actuator Control
Main Engines (3)
Operational Instructions
PCM MU (2)
OI MDM (7)
Recorder (2)
MEC
External Tank Separation Pyro Aft
External Tank Separation Pyro Forward
Orbiter-External Tank Umbilical
EXTERNAL TANK
01 Measure-ment
T-0 Umbilical
Aft Pull-Away Interface
Forward Pull-Away Interface
Forward Separation Pyro
Aft Separation Pyro
SRM Ignite
Recovery
SRB'S
MDM
MDM
APU
Range Safety
Forward IEA
Aft IEA
SRB TVC Actuator Control
RGA
SRM Transducers

Avionics



Ascent Thrust Vector Controller

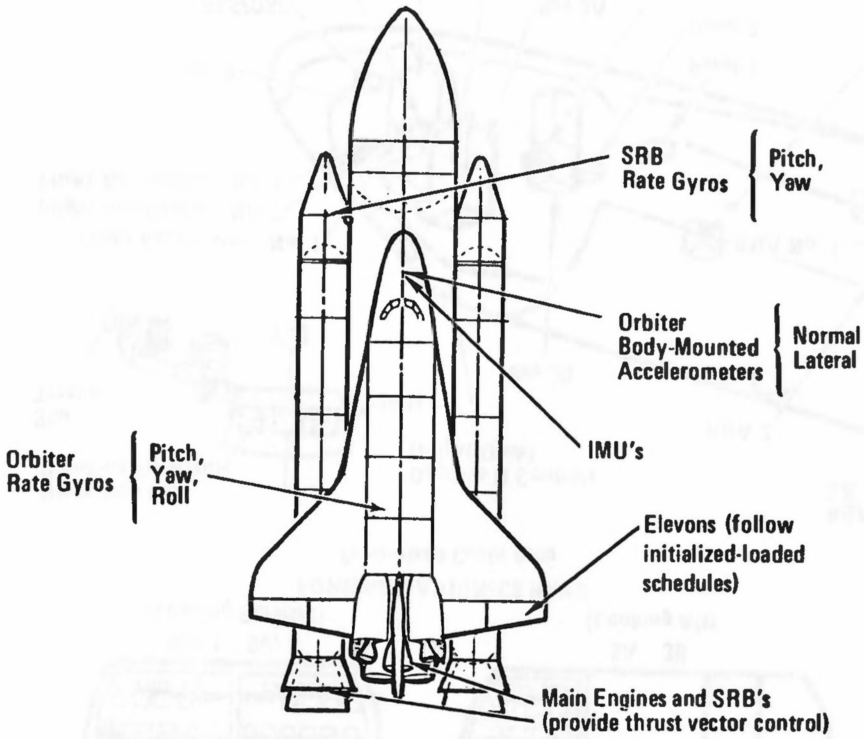
SRB processor of the DAP accomplishes the same functions as the MPS, except it is referred to as rock and tilt instead of pitch and yaw.

Each main engine and solid rocket booster servoactuator consists of four independent two-stage servoactuators which receive signals from the ATVC's. All four two-stage servovalves in each servoactuator control one power spool in each actuator which positions the actuator ram and the respective SRB nozzle and main engine. The four two-stage servovalves in each actuator provide a force summed majority arrangement to position the power spool. With the four identical flight control system commands to the four two-stage servovalves, the actuator force sum action prevents a single erroneous command from affecting power ram motion. If the erroneous command persists for more than a predetermined time, differential pressure sensing activates a selector valve, isolating and removing the defective servovalve hydraulic pressure, permitting the three remaining channels and servovalves to control the actuator ram spool. A second failure would isolate and remove the defective servovalve hydraulic pressure in the same manner as the first failure, leaving only two remaining active channels. Each actuator ram is equipped with position transducers for position feedback to the TVC system An "FCS Channel" yellow caution/warning light on the flight deck display and control panel will illuminate if an SRB or main engine actuator channel fails.

The SRB pitch and yaw rate gyros are used exclusively during first stage, and control is switched to orbiter rate gyros when the SRB's are commanded to null in preparation for separation. Pitch and yaw axes and a combination of rate, attitude, and acceleration signals are blended to effect a common signal to both main engine and SRB TVC. In the roll axes, rate and attitude are summed to provide a common signal to both the main engine and SRB TVC.

Orbiter rate gyros are used by the flight control system during ascent, entry, and aborts as feedbacks to find rate errors which are used for stability augmentation and for display on the commander and pilot attitude director indicators (ADI's) rate needles. There are four orbiter rate gyro assemblies located in the mid fuselage under the payload bay. The rate gyros are referred to as rate gyro assemblies (RGA's) 1 and 2 and RGA's 3 and 4.

Each RGA contains three single-degree-of-freedom rate gyros positioned to sense rates about the roll, pitch, and yaw axes. All the RGA's contain identical gyros; the only difference is in the manner in which each gyro is positioned within the RGA. Each gyro has three axes. A motor forces the gyro to rotate about the spin axes. When the vehicle rotates about the gyro input axis, a torque results in a rotation about the output axis.



Ascent Flight Control Hardware

An electrical voltage proportional to the angular deflection about the output axis and representing vehicle rate about the input axis, is generated and sent through the flight aft multiplexer/demultiplexers to the computers and RGA subsystem operating program (SOP). Selected rates are then sent to the ADI rate needles. The data is also sent to the display electronic unit CRT's.

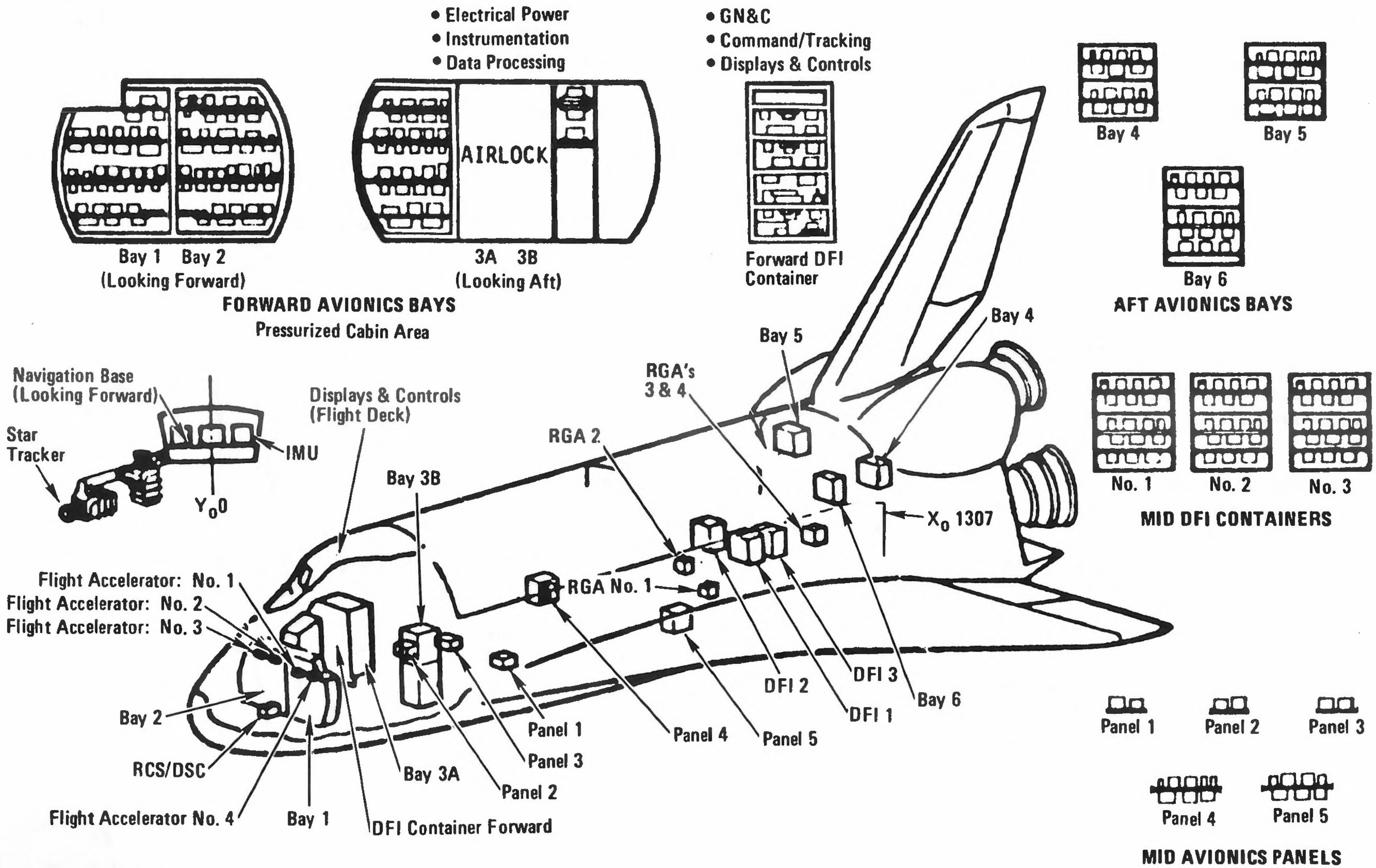
The RGA's are mounted on coldplates for cooling by the Freon-21 coolant loops. The RGA's require a five-minute warmup time.

SRB rate gyros are used by the flight control system during first-stage ascent as feedback to find rates from liftoff to two to three seconds prior to SRB-external tank separation. There are three rate gyros on each SRB. They are mounted on the forward ring in the forward skirt of the SRB-external tank attach point. The SRB rate gyros contain only two gyros, for sensing rates in the pitch and yaw axes.

SRB rate gyro data is sent through the orbiter forward MDM's to the computers. The flight control system DAP uses this data during first stage ascent. The SRB gyros are switched out of the loop two or three seconds before SRB separation and their yaw and pitch rate data is replaced by orbiter RGA pitch and yaw rate data.

Orbiter accelerometer assemblies (AA's) are used during ascent and entry for feedback to find acceleration errors, which are used for stability augmentation, elevon load relief during first-stage ascent, and during TAEM (terminal area energy management) and approach and landing phases. The accelerations transmitted to the forward MDM's are voltages proportional to the sensed acceleration. They are multiplexed and sent to the computers, where an AA operating program conditions and scales the signals from the selected normal and lateral AA's. The rates are then sent to flight control. The four body-mounted accelerometer assemblies are located in the crew compartment mid deck avionics bays 1 and 2.

Each assembly contains two single-axis accelerometers positioned so that one senses lateral acceleration and the other senses normal acceleration. Lateral (left and right) acceleration is sensed along the Y axis and normal (vertical) accleration is sensed along the Z axis (yaw and pitch, respectively). The accelerometers use a pendulum light beam and photo diode to provide a reading of acceleration. Data is also sent to the display electronics unit CRT's and ADI error needles during entry. The AA's are convection cooled and require a five-minute warmup time.



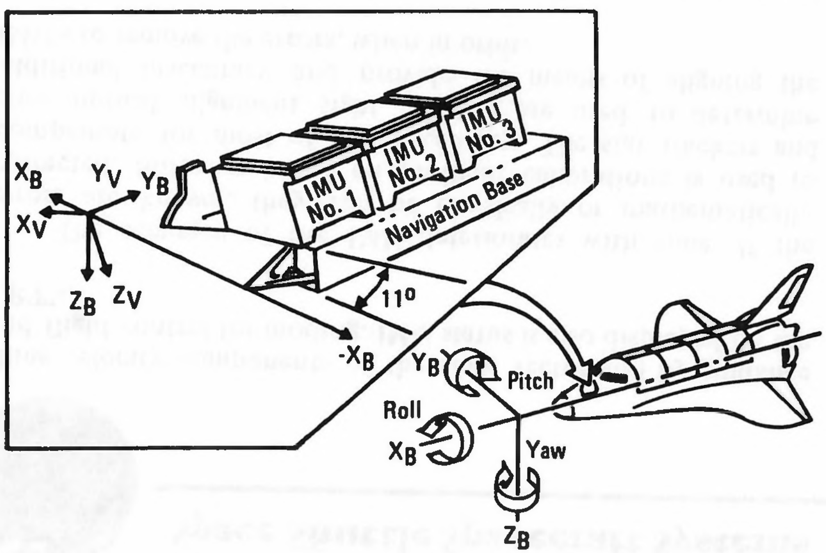
Equipment Installation

An "RGA/Accel" red caution/warning light on the flight deck display and control panel informs the flight crew of an RGA or AA failure.

The commander and pilot ADI's display attitude, rate, and error data.

ORBITER INERTIAL MEASUREMENT UNITS. These contain two accelerometers to sense accelerations and two gyros which provide an inertially stabilized four-gimbal platform (called cluster) that can obtain vehicle attitude with respect to inertial space. There are three IMU's mounted on a navigation base inside the crew compartment forward of the flight deck display and control panel.

Each IMU gyro has two output axes. Angular motion about any axis perpendicular to the spin axis causes rotation about a third orthogonal axis, which contain components of rotation about the two output axes. The appropriate gimbals are torqued to null the gyro rotation. The gimbal torquing in response to vehicle motion results in an inertially stabilized platform. In addition to torquing in response to vehicle motion, the gyros may be pulse torqued (small angles) or slewed (large angles) by software which results in gimbal torquing to position or reposition the platform. Each gimbal has coarse (one-speed) and fine (eight-speed) resolvers. The resolver readouts are sent to the forward MDM's and then to the computers.



Orbiter IMU's

The two IMU accelerometers are a pendulous mass anchored to a case. Acceleration produces relative motion between the mass and the case. The motion is sensed and voltage is applied to counteract the motion. The voltage required is proportional to the acceleration. The voltage is electronically converted to pulses which are accumulated. The accumulated data pulses represent velocity and are sent to the computers via the forward MDM's. One accelerometer senses acceleration along two axes, X and Y, and the other senses delta velocity along a Z axis. These X, Y, and Z axes are not vehicle axes.

The three IMU's are initially aligned so that their cluster orientations are skewed relative to one another, which results in only one IMU being exposed to "gimbal flip" conditions at any one time. The gimbal sequence is outer roll, pitch, inner roll, and azimuth. The inner roll gimbal is redundant to the outer roll. By maintaining the pitch gimbal perpendicular to the yaw-azimuth gimbal and by mechanizing two roll gimbals, "gimbal lock" is avoided, but "gimbal flip" becomes a possibility with pure yaw motion at a pitch of 90 to 270 degrees.

The selected data from the IMU operating program is used by flight control for coordination and guidance for steering commands. Attitude angles are displayed on the ADI's and horizontal situation indicators (HSI's). The selected delta veloci-ies are used by navigation to determine the three position a·

three velocity components of the state vector and by guidance and flight control for moding. IMU status is also displayed on the CRT's.

The accuracy of the IMU deteriorates with time. If the errors are known, they can be physically or mathematically corrected. Software based on preflight calibrations is used to compensate for most of the inaccuracy. The star trackers and crew optical alignment sight (COAS) are used to determine additional inaccuracy and provide the means of aligning the IMU's to remove the errors, when in orbit.

Each IMU is cooled by cabin air drawn into and circulated within the casings by IMU fans. Critical temperatures are maintained by automatic heater circuits within each IMU. An IMU red caution/warning light on the flight deck display and control panel informs the flight crew of an IMU failure.

DEDICATED DISPLAY INSTRUMENTS. These are located on the orbiter crew compartment flight deck display and control panels. The dedicated display software performs the processing required to accept, prepare, and output guidance, navigation, and control data to the dedicated display indicators. The dedicated display indicators are the ADI's, HSI's, alpha-Mach indicators (AMI's), attitude vertical velocity indicators' (AVVI's), and surface position indicators (SPI's). These are two sets of ADI's, HSI's, AMI's and AVVI's at the flight deck commander and pilot stations. There is only one SPI. Another ADI is located on the flight deck aft flight display and control panel.

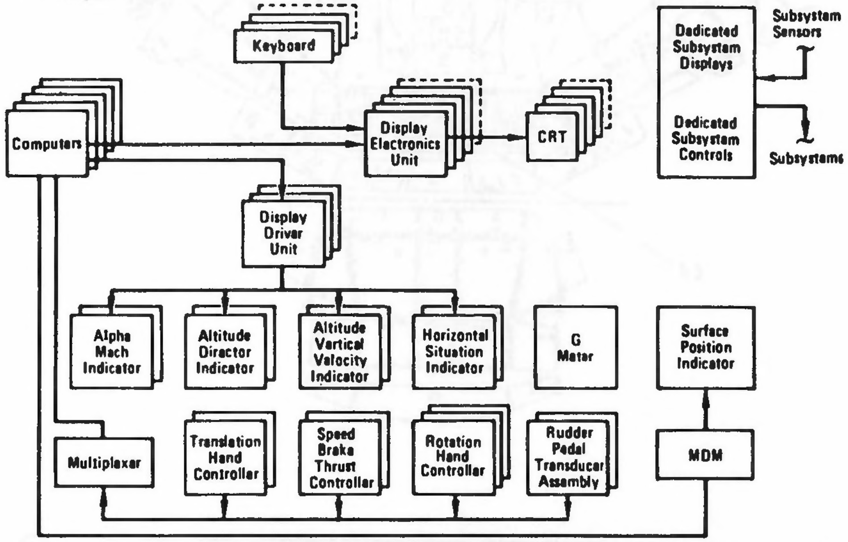
The display driver unit (DDU) is an electronics unit that connects the computers and the primary flight displays. The DDU receives data signals from the computers and decodes them to drive the dedicated displays. The unit also provides dc and ac power for the ADI's and RHC's. The DDU contains logic for setting flags on the dedicated instruments for such items as data dropouts and failure to synchronize. The orbiter contains three DDU's, one at the commander's station, one at the pilot's station, and one at the aft station.

The three ADI's indicate the vehicle's roll, pitch, and yaw attitude via a gimbaled ball, attitude errors via three needles, and attitude rates via three pointers. The angles displayed on the three ADI's are generated in the attitude processor from IMU data and indicate the orbiter's roll, pitch, and yaw via the gimbaled ball. The attitude error needles are superimposed on the top face of the ADI ball and provides attitude error information in roll, pitch, and yaw. The needles are driven in first- and second-stage ascent by attitude errors generated in guidance. The rate pointers are positioned on the indicator outside the ADI ball and provide rate information from the rate gyro assemblies. The ADI also provides an "Off" flag to indicate when the attitude display may be invalid. The needles and pointers have a stored (out of view) position which is also used to indicate invalid conditions.

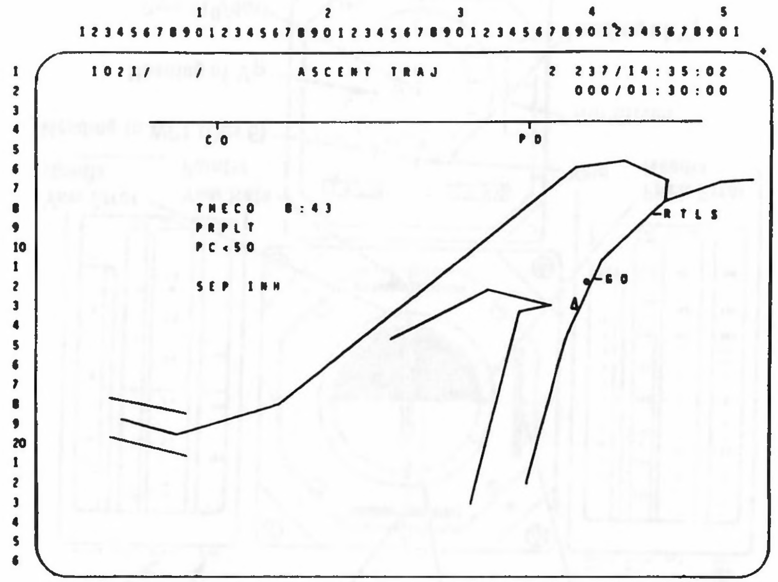
In first-stage ascent, the ADI's monitor the roll, pitch, and yaw of the vehicle and ensure that attitude rates remain within their limits. During second-stage ascent, the ADI's confirm that second-stage guidance has initiated the monitor hold for external tank separation.

FIRST-STAGE ASCENT. During first-stage ascent, it is sometimes necessary to relieve loads on elevon hinges that result from high shears. This is performed automatically by a load relief logic which drives the elevons to the appropriate positions to relieve the aerodynamic adverse hinge moment loads. This logic also holds the body flap and rudder/speedbrake in place during ascent. The SPI displays the position of the aerosurfaces.

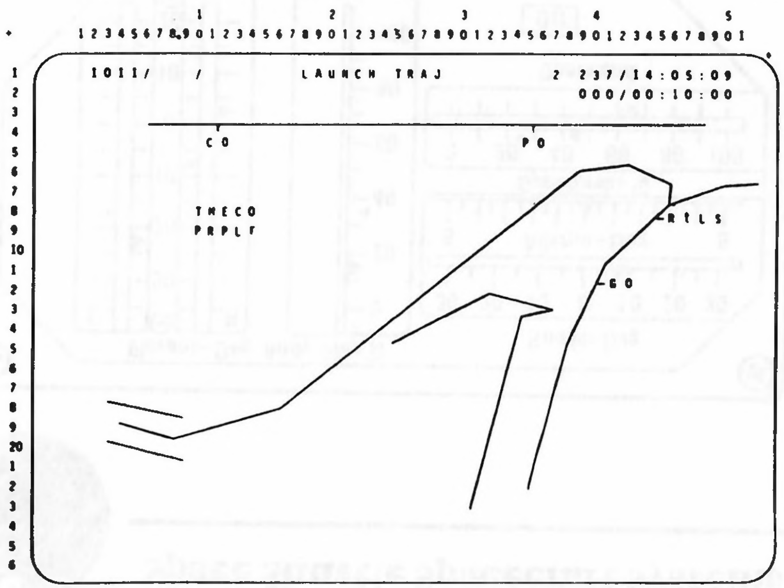
During first-stage ascent, the pre-stored guidance program works open-loop in that it selects an initialized-loaded roll, pitch, and yaw command based on vehicle relative velocity. An initialized-loaded table is accessed by software and commands



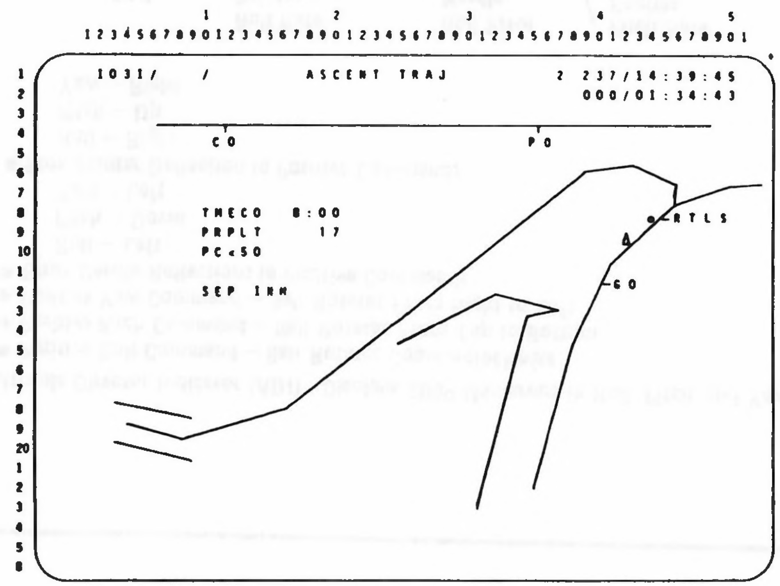
Display and Control System



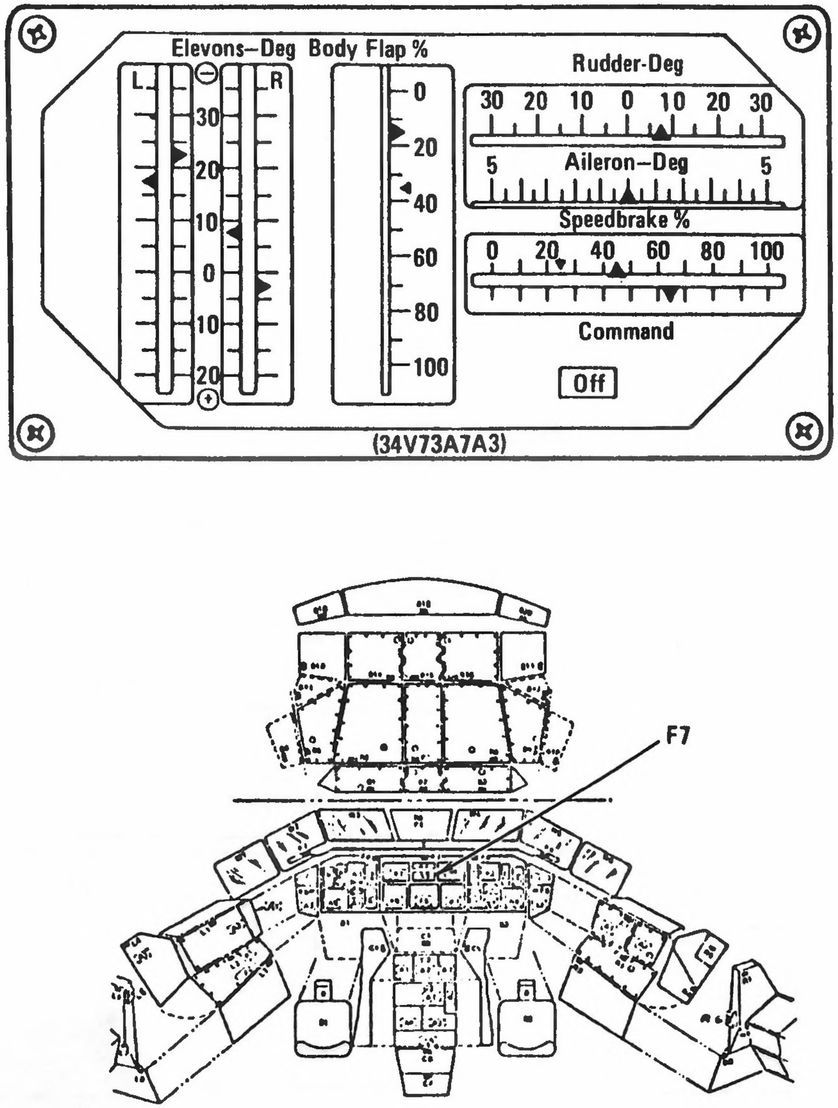
Ascent Trajectory Typical Display



Launch Trajectory Typical Display



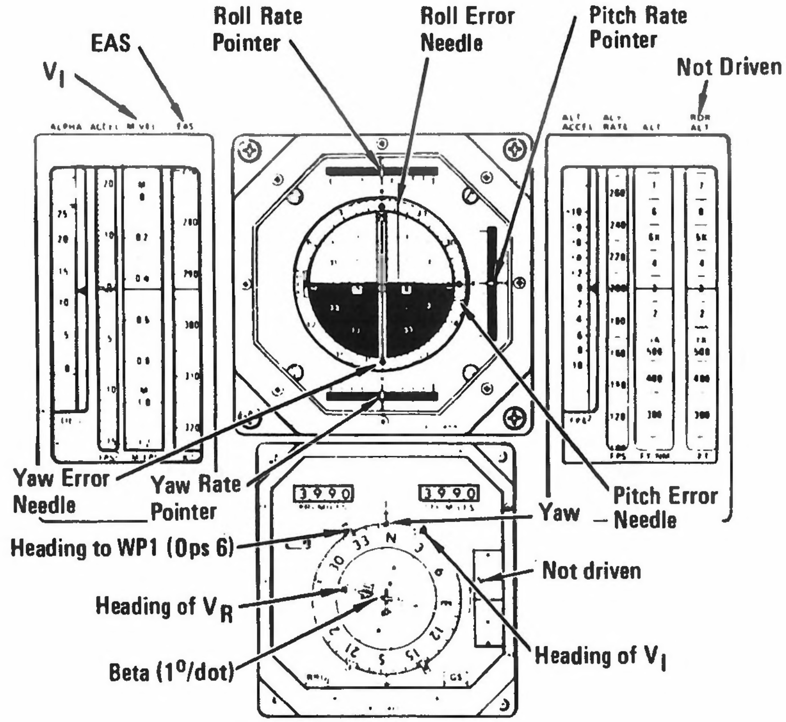
Ascent Trajectory Typical Display



Surface Position Indicator

Attitude Director Indicator (ADI) - Displays 360° Maneuvers in Roll, Pitch, and Yaw

- Positive Roll Command – Ball Rotates Counterclockwise
- Positive Pitch Command – Ball Rotates From Top to Bottom
- Positive Yaw Command – Ball Rotates From Right to Left
- Error Needle Reflections to Positive Commands
  - Roll – Left
  - Pitch – Down
  - Yaw – Left
- Rate Pointer Deflection to Positive Commands
  - Roll – Right
  - Pitch – Up
  - Yaw – Right



Dedicated Displays - First Stage Ascent

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the vehicle attitude needed to obtain the desired SRB separation attitude and orbit inclination.

Guidance also is responsible for issuing throttle commands to the main engines. These commands also are initialized-loaded. There are four initialized-loaded slots for throttle settings and they are referenced to relative velocity. The main engines are at 100-percent thrust from liftoff to approximately 3.5 seconds; they are throttled to 65 percent at 10-percent thrust/second to limit dynamic pressure (maximum q). The engines are held at 65 percent for approximately 24 seconds and at approximately 63 seconds from liftoff are throttled back up to 100 percent at 10-percent thrust/second to maintain constant 3 g's in the initial development flights. The thrust profile reduces heating and vehicle loads during maximum dynamic pressure.

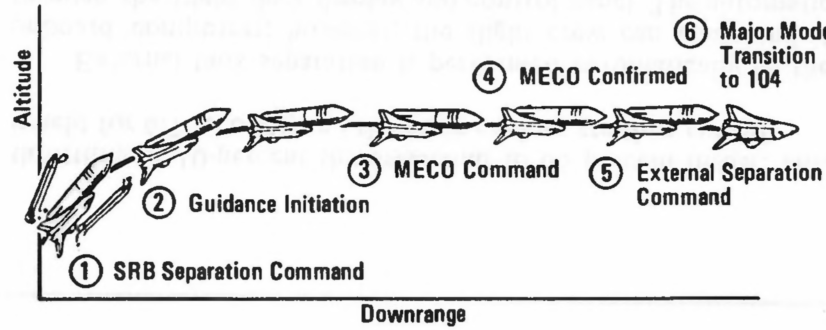
Navigation programs in first-stage ascent are responsible for providing accurate knowledge of the vehicle state vector through the use of IMU-sensed velocity changes and a mathematical model of the earth's gravitational forces. This function can be used to aid in driving the CRT's predictor.

Boost guidance is responsible for performing the "table lookup" of the appropriate roll, pitch, and yaw command depending on relative velocity. The table used depends on how many main engines are thrusting. If one engine fails, guidance automatically recognizes the failure and lofts the trajectory, commanding the remaining two main engines to 100-percent thrust for the remainder of first-stage ascent in the initial development flights. Each main engine controller is used during the ascent phase to monitor the operation of an engine, issue inhibit commands to prevent a second engine from automatically shutting down when one has shut down, monitor the state of the flight deck crew displays and switches from the switch processor, and issues appropriate commands. The controllers also monitor GN&C software for the proper time to check the main propulsion system liquid oxygen and liquid hydrogen low level sensors, monitor GN&C software for proper time of main engine cutoff (MECO), and issue the closure of the main propulsion system liquid oxygen and liquid hydrogen prevalves after engine shutdown.

SRB separation is normally performed automatically by the onboard computers; however, the flight crew can command separation through switches on the flight deck display and control panel. The automatic sequence is initiated by the software in the computers when the chamber pressure of both SRB's is below 2587 millimeters of mercury (mmHg) (50 psi).

At SRB separation command, a three-axis attitude hold is commanded for four seconds. When the SRB separation command is received, the SRB nozzles are positioned to null and the flight control system is switched to the orbiter rate gyro assemblies. Four seconds after SRB separation, second-stage (main engine) guidance takes over. In addition, if a main engine shutdown is detected, the failed engine is positioned to null and trim changes are commanded on the remaining engines.

SECOND-STAGE GUIDANCE. Second-stage guidance is completely changed from first stage. It is a closed-loop function that computes each cycle where the vehicle should be so that an initialized-loaded MECO target can be reached. Commands are generated to place the vehicle in a desired position with a desired



Second-Stage Ascent

velocity and with minimal fuel usage. The desired position and velocity are determined from a set of initialized-loaded conditions. The parameters defining MECO target are target-velocity, flight-path angle, target radius from earth center, and desired orbital plane. The MECO target conditions are selected primarily to ensure correct placement of the external tank splashdown.

The first phase of second-stage ascent extends from the SRB separation command until the earliest time at which an abort once around (AOA) is possible with a single engine failure. This assumption causes a lofted trajectory during the first phase so that MECO conditions can be reached with the use of only two engines.

The second phase of second-stage ascent begins at the earliest time at which an AOA is possible with a single engine failure and continues until an acceleration of 3 g's is reached. The remainder of the second phase assumes use of three engines.

The third phase begins when thrust acceleration reaches 3 g's. At this point, a 3-g limiting software program begins to adjust the throttle command so that acceleration does not exceed 3 g's. If the vehicle is sufficiently heavy, or if main engine performance is degraded, the 3-g limit may not be reached before MECO. In this case, the third phase would not be entered, and the second phase would continue until MECO. Also, if an engine failure occurred during the third phase, the acceleration would decrease, and the second phase would be re-entered.

The hardware utilized in second-stage ascent is the same as that used in first stage except that the body-mounted accelerometers are not used and the elevons are held in place.

Second-stage navigation is exactly the same as for first stage.

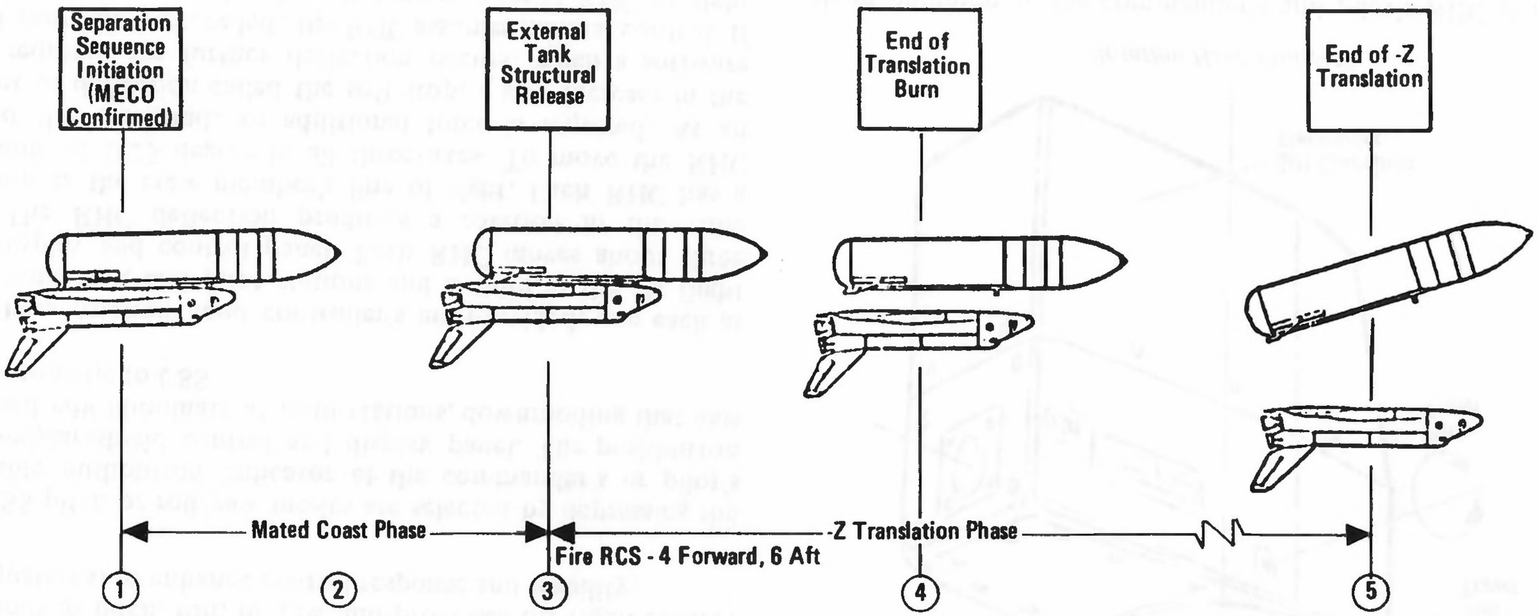
At MECO minus ten seconds, the MECO sequence begins and three seconds later the main engines are commanded to begin throttling at 10-percent thrust/second to 65 percent thrust. This is held for 6.7 seconds, and the main engines are then cut off.

External tank separation is performed automatically by the onboard computers; however, the flight crew can command it through the flight deck display and control panel. The automatic sequence is initiated by the GN&C moding, sequencing, and control when the main engine operations sequence has determined that all of the engines have shut down and the "MECO confirmed" flag is set.

The main engine operations sequence is checked and if set and shutdown commands have been issued for that engine, its prevalves are closed after a delay. A "MECO confirmed" flag is set after it has been confirmed that all engines have entered the shutdown phase. Also, a flag is set for the external tank separation sequence after the prevalves for all engines have been commanded closed. This is necessary because shutdown times may differ.

First, the computer operation program determines the mode of separation or if the separation is to be manually inhibited. It then arms the umbilical plate pyro initiator controllers, arms and fires the external tank tumble system after all the engine prevalves have been commanded closed, closes the feedline disconnect valves, gimbals the main engine nozzles to the proper position, deadfaces the external tank-orbiter interface, and unlatches and retracts the umbilical plates.

The sequence then arms the structural separation pyro controllers, performs some limit tests on certain body rates, and tests for feedline disconnect valve closure. If any test is not satisfied, the separation is inhibited and can occur only if the out-of-tolerance parameter comes back within tolerance or if the flight crew elects to continue the separation by overriding the inhibit. When either of these conditions is satisfied, the structural separation pyro initiators are fired.



External Tank Separation

After initiation of the orbiter-external tank separation sequence, there is an approximate 11-second mated coast and then the orbiter and external tank separate. The external tank tumble system is activated approximately two seconds after MECO to produce a tumble rate of 10 to 50 degrees per second after separation. The external tank is on a suborbital trajectory that results in an impact location in the Indian Ocean. External tank breakup nominally will occur during its entry into the earth's atmosphere at approximately 56,388 meters (185,000 feet) altitude.

At MECO, orbiter attitude commands (roll, pitch, and yaw) are frozen, and body rate damping is maintained during the coast period by the RCS. Just prior to external tank structural release, the RCS is inhibited, then reenabled immediately after external tank structural separation to an inertial attitude hold, followed by an RCS minus-Z translation maneuver. Four forward and six aft RCS jets are used to achieve a translation of 1.2 meters per second (four feed per second) vertically to ensure orbiter clearance from the arc to the rotating external tank. The orbiter continues to coast away from the external tank in the inertial altitude hold mode to obtain additional vertical clearance. When the orbiter has gained a velocity of 1.2 meters per second (four feet per second) vertically, the separation is flagged complete and the ground is responsible for issuing a go/no go for the impending OMS-1 thrusting sequence. OMS-1 time of ignition is set for MECO plus two minutes and is normally accomplished with the two OMS engines.

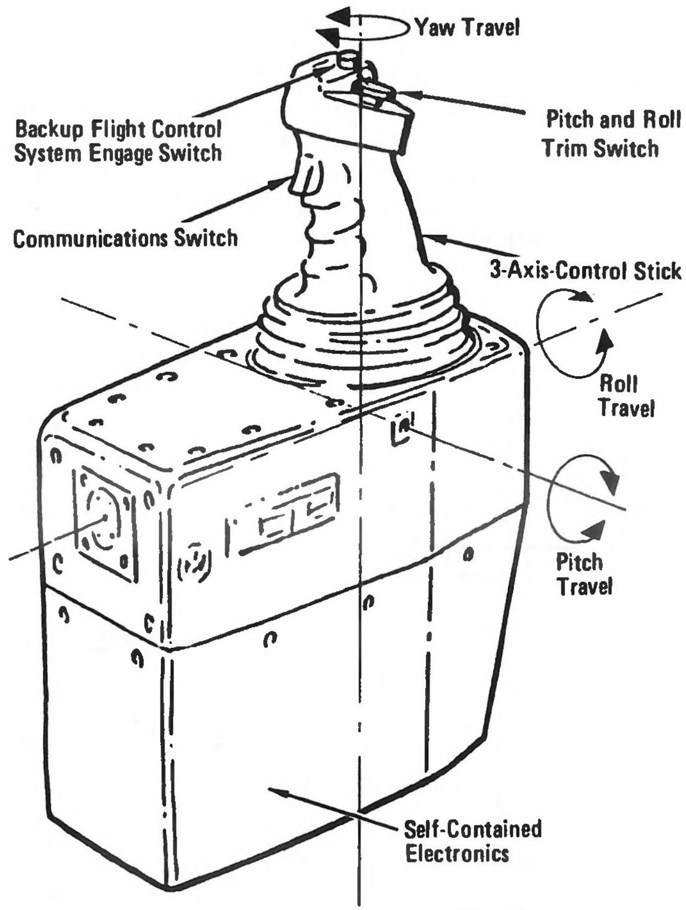
At launch and during the pre-MECO phase, the OMS engine nozzles are stowed to protect the nozzles from aerodynamic and thermal loads during ascent.

CONTROL STICK STEERING (CSS). This mode is similar to the automatic mode, except the flight crew commands three-axis motion using the rotation hand control. The computers process the commands from the RHC and motion sensors and the flight control system interprets the RHC motions (fore and aft, right and left, clockwise and counter-clockwise) as rate commands in pitch, roll, or yaw and processes the flight control law (equations) to enhance control response and stability.

CSS pitch or roll/yaw modes are selected by depressing the applicable pushbutton indicator at the commander's or pilot's eyebrow/glareshield control and display panel. The pushbutton depressed will illuminate at both stations, downmoding that axis from automatic to CSS.

Three rotation hand controller's are provided, one each at the commander, and pilot stations and a third at the aft flight deck display and control panel. Each RHC moves about three axes. The RHC deflection produces a rotation in the same direction as the crew member's line of sight. Each RHC has a deadband of 0.25 degree in all three axes. To move the RHC beyond the deadband, an additional force is required. At an amount of deflection called the soft stop, a step increase in the force required for further deflection occurs. When a software detent position is exceeded, the RHC assumes manual control. If a malfunction occurs in the left (commander's) RHC or right (pilot's) RHC, it illuminates a red left or right RHC caution/warning light on the display and control panel.

The aft RHC is used only in orbit, for rendezvous and docking with an unmanned vehicle. Software performs the necessary transformations on the aft RHC commands to make



Rotation Hand Control

them common to the commander's and pilot's RHC in rotation of the same direction relative to the crew member's line of sight.

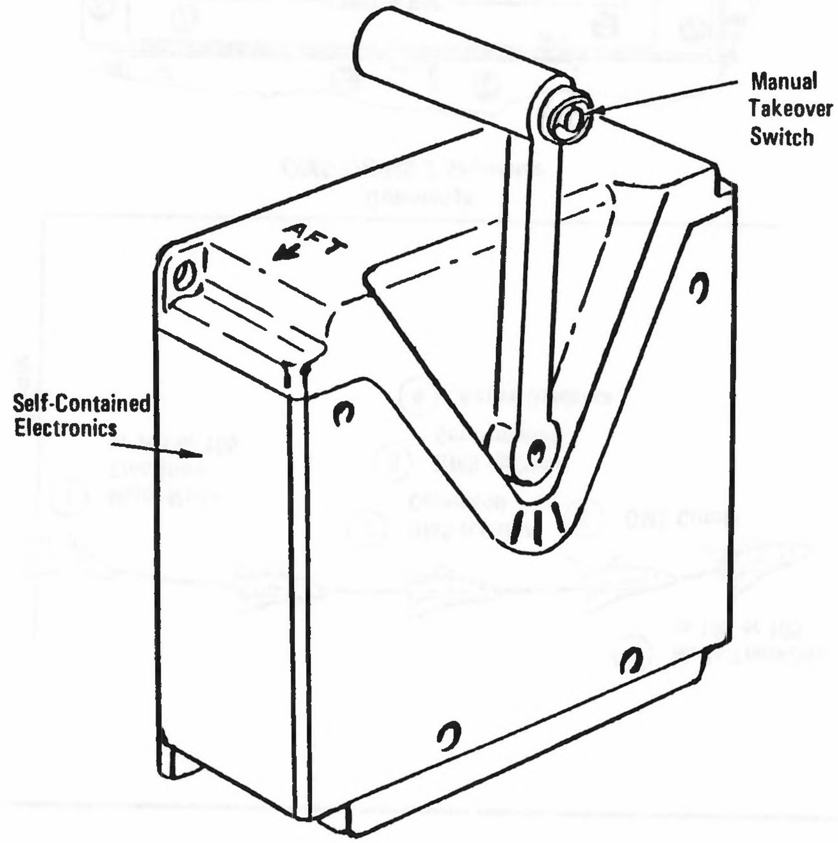
Each RHC contains nine transducers: three redundant transducers sense pitch deflection, three sense roll deflection, and three sense yaw deflection. The transducers produce an electrical signal proportional to the deflection of the RHC. The three transducers are called Channels 1, 2, and 3; the channel selected by redundancy management provides the command. Each channel is powered by a separate power supply in the associated display driver unit.

MANUAL THRUST VECTOR CONTROL. This capability is provided during first- and second-stage ascent. This is accomplished by substituting the inputs from the RHC for the automatic commands from guidance to gimbal the main engines and SRB's during first stage or the main engines during second stage. The DAP remains active to process the flight crew's input. This is referred to in this mission phase as manual thrust vector control (MTVC). MTVC is available at liftoff (SRB ignition command plus 0.365 second). One of the CSS pushbuttons at the glareshield/eyebrow panels must be depressed before MTVC is available.

Once MTVC is activated, the vehicle is in a rate-command/attitude-hold mode in all axes. When the RHC is in detent, with MTVC selected, the vehicle is in attitude hold. The DAP holds the vehicle in the attitude it had when the RHC was in detent. A rate command equal to zero replaces the rate command generated in guidance and control steering. The attitude error for the DAP is computed by integrating the rate gyro assemblies measured rates. However, there are limits on vehicle rates and attitude errors. If the limits are exceeded when attitude hold is requested by placing the RHC in detent, attitude hold will not be initiated until the rates and errors are within limits.

When the RHC is removed from detent, a rate command proportional to the amount of deflection replaces the rate command previously generated. The attitude error is zeroed. The larger the deflection of the RHC, the larger the command. The flight control system compares these commands with inputs from the rate gyro assemblies and accelerometer assemblies (what the vehicle is actually doing–motion sensors) and generates control signals to produce the desired rates. When the commander or pilot release the RHC, it returns to center and the vehicle will maintain its present attitude (zero rates).

MANUAL THROTTLING. Throttling of the main engines also is possible through use of the speedbrake/thrust control (SBTC) at the pilot station. Manual throttling is accomplished by depressing the takeover switch on the SBTC. The automatic thrust command is frozen until the "Man" (manual) white light in the "Spd Bk/Throt" pushbutton at either station is illuminated. This



Speed Brake Thrust Control

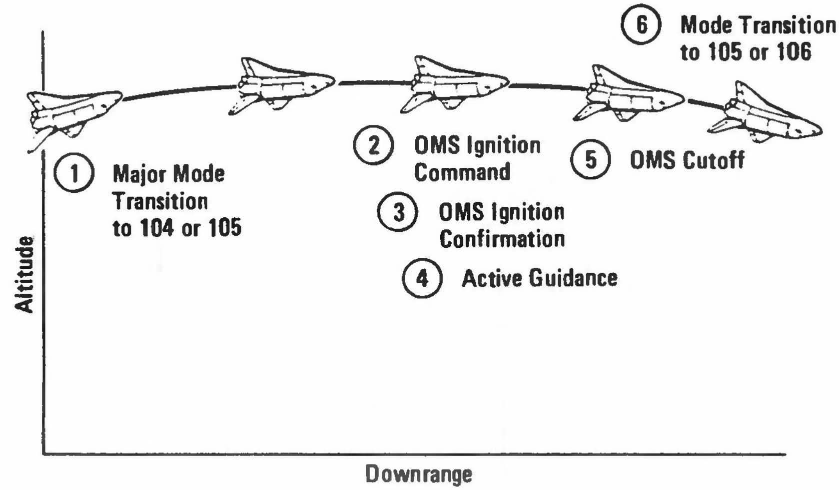
means the SBTC has matched the frozen automatic command. The takeover switch on the SBTC is then released and the pilot has control of throttling. This method prevents undesirable transients in throttle commands during takeover.

At the forward setting, thrust level is the highest. Rotating the SBTC back decreases the thrust. Each SBTC contains three transducers (Channels 1, 2, and 3) which produce a voltage proportional to deflection. Redundancy management selects the output.

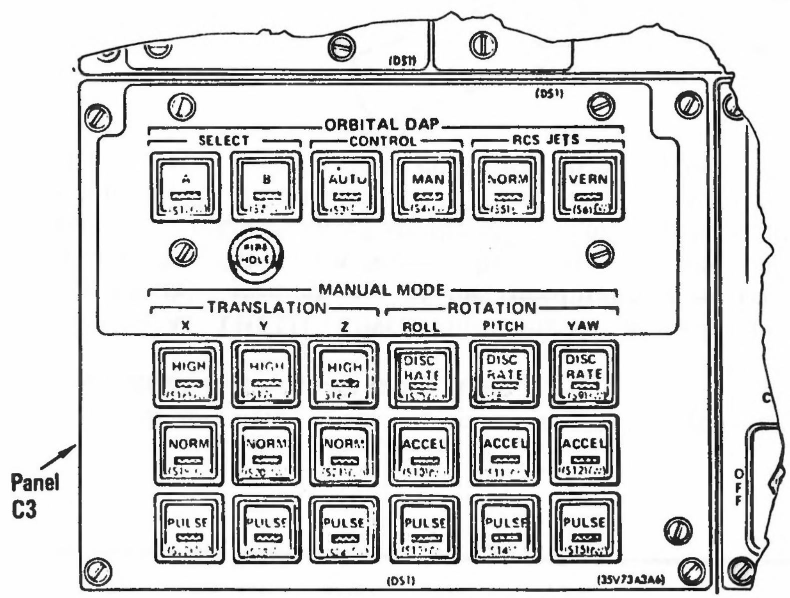
TRANSITION DAP. The transition DAP flight control mode controls the orbiter in response to automatic or manual commands during insertion and deorbit. The effectors used to produce control forces and moments on the orbiter are the two OMS engines and the 38 primary RCS engines. At main engine cutoff, rotation and translation control of the spacecraft is provided by commands issued to the forward and aft RCS jets. The forward and aft RCS also provide attitude control and three-axis translation during external tank separation, on-orbit maneuvers, and roll control for a single OMS engine operation. The OMS provides propulsive and three-axis control for orbit insertion, orbit circularization, orbit transfer, and rendezvous. Failure of a single OMS engine will not preclude a nominal orbit insertion.

Normally, the first OMS thrusting period raises the low elliptical orbit following external tank jettison and the second OMS thrusting places the spacecraft into a circular orbit as designated for that mission. For orbital maneuvers utilizing the OMS, any delta velocities greater than 1.8 meters per second (6 feet per second) utilize the two OMS engines.

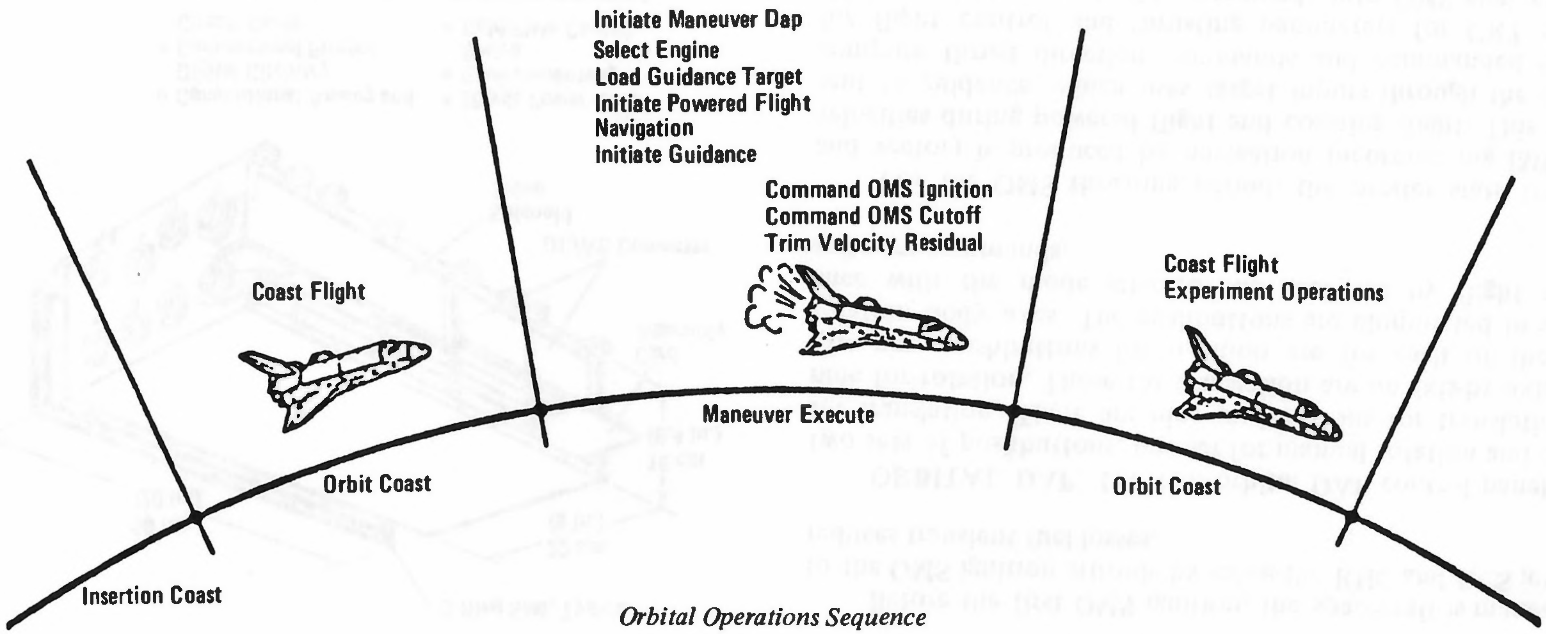
Orbit insertion guidance, navigation, and flight control software, through the transition DAP, controls the external tank/orbiter separation maneuvering, OMS-1 and OMS-2 ignition attitude, OMS thrusting commands, OMS engine gimbaling for



OMS-1/OMS-2 Sequence



Orbital Digital Autopilot Pushbuttons



Orbital Operations Sequence

thrust vector control, and RCS thrusting commands in conjunction with the use of the "Orbital DAP" control panels.

Two "orbital DAP" pushbuttons provide automatic and manual modes to the transition DAP program. The pushbuttons are illuminated by software commands in accordance with the mode selected and accepted by flight control software.

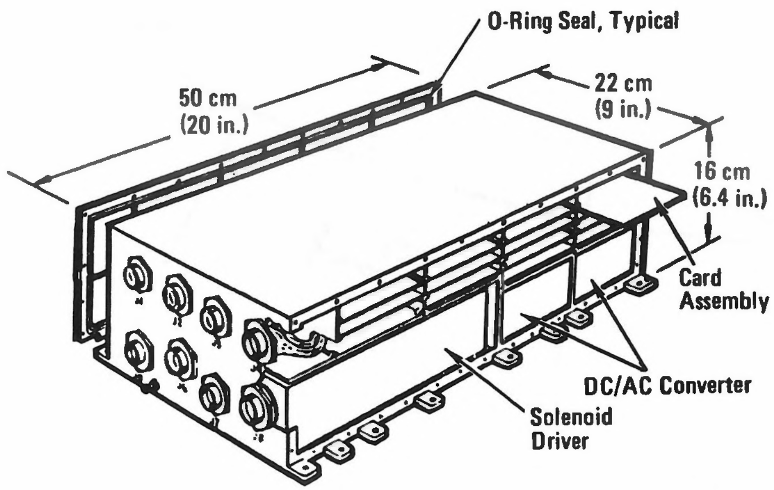
Navigation incorporates IMU delta velocities during powered flight and coasting flight to produce orbiter state (position and vector).

The transition DAP reconfiguration logic controls the moding, sequencing, and initialization of the control law modules and sets gains, deadbands, and rate limits. The steering processor is the interface between the guidance or manual steering c ands and the transition DAP. The steering processor generates commands to the RCS processor, which generates the RCS jet commands required to produce the commanded spacecraft translation and rotation using attitude and rotational rate signals, or translation or rotation acceleration commands. The OMS processor generates OMS engine gimbal actuator thrust vector control commands to produce desired spacecraft/engine relationship for commanded thrust direction.

When in the automatic DAP mode, the external tank separation module initialized by the external tank separation sequencer compares Z delta velocities from the DAP attitude processor with an initialized-loaded desired Z delta velocity. Before this value is reached, the transition DAP and steering processor sends commands to the RCS jet selection logic which uses a table look-up technique for the primary RCS jets and commands 10 RCS jets to fire in the plus Z direction. When the desired Z delta velocity is reached, the translation comman set to zero. Rotation commands are permitted during the external tank separation sequence.

The RCS jet selection module receives both RCS rotation and translation commands and, using the table look-up technique, outputs 75 discretes to the primary RCS jets to turn the 38 jets on or off.

The RCS reaction jet driver forward and aft assemblies provide the turn-on/turn-off jet selection logic signals to the RCS jets. There is also a driver redundancy management program that permits only "good" RCS jets to be turned on. The "RCS Jet" yellow caution/warning light on the flight deck display and control panel indicates a failed on, failed off, or leaking RCS jet.



- Conventional Analog and Digital Circuitry
- Conventional Printed Circuit Cards
- 28-vdc Power Only
- Environmentally Sealed
- Cold-Plate-Cooled

	Weight, kg (lb)	Power (LRU Only), W	
		Normal Quiescent	Maximum
Aft Unit	15.0 (33.0)	27.4	62.6
Forward Unit	13.8 (30.4)	20.3	41.4

Reaction Jet Driver

The 19 RCS jets thrusting in the plus or minus Z direction provides Z translation and roll or pitch rotation control and are considered independent of other axes. The 12 RCS jets thrusting in the plus or minus Y direction provide Y translation, and yaw rotation only. The seven RCS jets thrusting in the plus or minus X direction provide X translation only.

Before the first OMS ignition, the spacecraft is maneuvered to the OMS ignition attitude by using the RHC and RCS jets; this reduces transient fuel losses.

ORBITAL DAP. The two orbital DAP control panels have two sets of pushbuttons, one set for manual rotation and one set for translation. There are nine pushbuttons for translation and nine for rotation. Those for translation are on axis-by-axis basis. The nine pushbuttons for rotation are for each of the three rotation body axes. The pushbuttons are illuminated in accordance with the mode selected and accepted by flight control software commands.

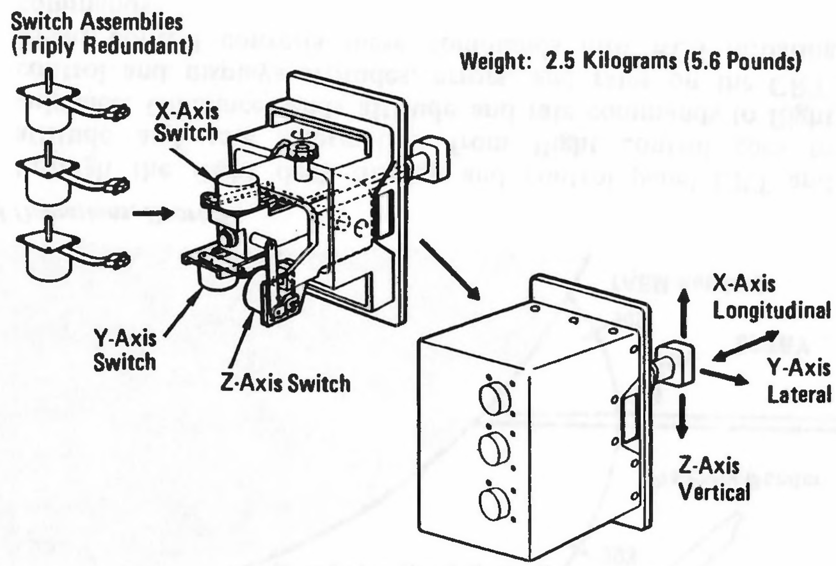
For the OMS thrusting period, the orbiter state (position and vector) is produced by navigation incorporating IMU delta velocities during powered flight and coasting flight. This state is sent to guidance, which uses target inputs through the CRT to compute thrust direction commands and commanded attitude for flight control and thrusting parameters for CRT display. Flight control converts the commands into OMS engine gimbal angles for an automatic thrusting period. OMS thrust vector control for normal two-engine thrusting is entered by depressing the "Orbital DAP Auto" pushbutton with both RHC's within software detents. OMS manual TVC for both OMS engines is entered by depressing the "Orbital Man DAP" pushbutton or by moving the commander or pilot RHC out of any one of the three detents. The manual RHC rotation requests are converted into gimbal angles. OMS thrust in either case is applied through the spacecraft center of gravity.

Automatic thrust vector control for one OMS engine is identical to that for two OMS except that the RCS processor is responsible for roll control. Single-OMS engine thrust also is through the spacecraft's center of gravity, except when pitch or yaw rate commands are non-zero. If the left or right OMS engine fails, an "OMS TVC" red light on the flight deck display and control panel will illuminate.

Since an OMS cutoff is based on time rather than velocity, a velocity residual may exist following OMS cutoff. The residual is zeroed using the RCS via the THC.

TRANSLATION HAND CONTROLS. There are two translation hand controls, one at the commander's station and one at the aft flight deck station. These are used for manual control of translation along the longitudinal (X), lateral (Y), and vertical (Z) vehicle axes. The commander's THC is active during orbit insertion, in orbit, and during deorbit. The aft flight deck station THC is active only during the orbital phase. Each THC contains six three-contact switches, one each in the plus and minus directions for each of the three axes. Moving the THC to the right commands translation along the plus Y axis, closes three-switch contacts (referred to as Channels 1, 2, and 3), and the selected redundancy management channel provides the command. The aft THC is used when the flight crew is using the aft flight deck station.

In the transition DAP, the commander's THC is active and totally independent of the flight control orbital DAP push-buttons or the RHC position or status. Whenever the commander's THC is out of detent, plus or minus X, Y, or Z, translation acceleration commands are sent directly to the RCS



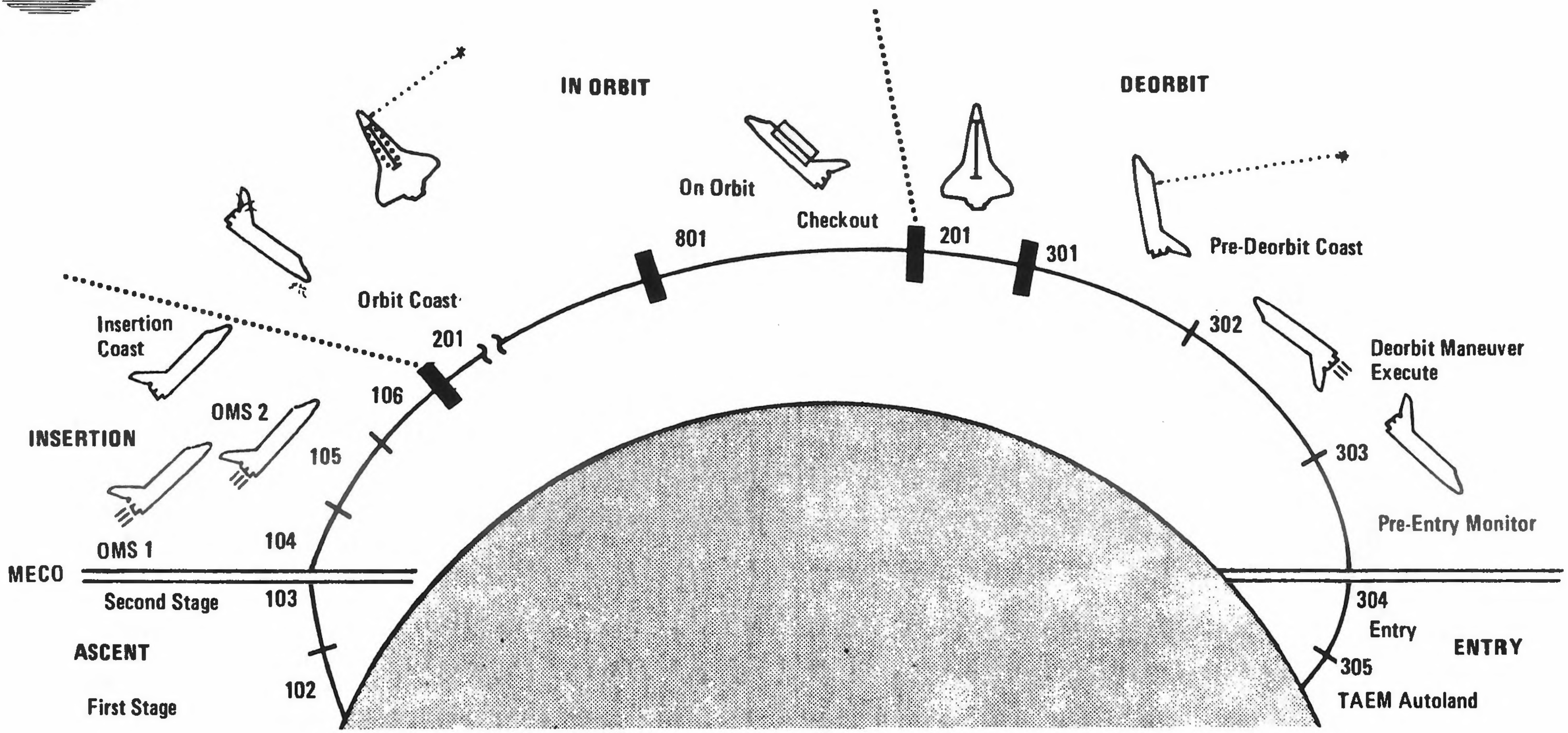
Translation Hand Control

jet selection logic for continuous RCS jet firings. Rotational commands may be sent simultaneously with translation commands within the limits of the RCS jet selection logic; if both plus X and minus Z translations are commanded simultaneously, plus X translation is given priority.

The attitude processor computes attitude from IMU gimbal angles and attitude errors from attitude and commanded attitude and sends these to the ADI's. The orbiter rate gyro assemblies provide body rates to flight control and ADI's.

OMS-2 is entered manually by the flight crew after OMS-1 and is similar to OMS-1 except for the different orbital parameters. Insertion is entered after the OMS-2 thrusting period. An orbital coast is achieved at this point.

ORBITAL OPERATIONS. Orbital guidance, navigation, and flight control software for the initial development flight provides



Orbit Insertion, Orbital Operations, Deorbit

simple coasting flight. In later development and subsequent flights, the orbital software includes rendezvous targeting, proximity operations, precision and rendezvous navigation, orbital maneuver guidance and flight control, and additional universal pointing and navigation display.

Navigation incorporates a drag model during coasting flight on IMU delta velocities during powered flight together with a gravity model to produce the orbiter state. This state is maintained primarily by navigation for entry; the only user in orbit is the local-vertical/local-horizontal function of universal pointing. For automatic rotations, attitude requests are input through the flight deck display and control panel CRT and attitude and rate information from flight control goes to guidance. Guidance sends attitude and rate commands to flight control and displays attitudes, errors, and rates on the CRT. Flight control converts these commands into RCS thrusting commands.

The orbital digital autopilot also can be operated in automatic or manual modes. "Auto" or "Man" pushbuttons are used to mode the RCS for atuomatic or manual rotation operation. The "Orbital DAP Select" A or B pushbuttons select the values the DAP will use from the DAP configuration parameter limits CRT display software loads. The values of attitude deadband, rate deadband, and vehicle change in rotation rate due to minimum impulse RCS thrusting are a function of DAP A or B selection and jet selection "Norm" (normal-primary) or "Vern" (vernier). Vernier selection prohibits all translation commands.

The jet selection logic uses a table look-up technique for the primary jets similar to the transition DAP. Unlike the transition DAP, the orbital jet selection logic checks for high or low rotation mode for both pitch and yaw for the forward and aft RCS jets. There are special tables for the high plus Z translation mode not used by the transition DAP. Angular rate increments are predicted for each group of primary jets and for each specific vernier jet. The primary jet predictions are based on one nominal inertia matrix and center of gravity. The vernier predictions are based on nominal or one of five different payload configuration (with the manipulator arm extended) inertia matrixes. The rotation compensation logic uses a combination threshold set on the DAP specialist display and counter (or accumulator) in determining when to command RCS jets on for rotation compensation.

The "Auto" pushbutton permits the automatic RCS rotation and movement of any RHC out of detent (any axis) downmodes the DAP from automatic to manual as will depressing the "Man" pushbutton. All active RHC's must be in detent when an "Auto" pusbutton is depressed in order to upmode to automatic RCS rotation mode. The control pushbuttons are illuminated by software commands in accordance with the mode selected and accepted by flight control software.

Automatic rotation commands are supplied by the universal pointing processor. The universal pointing processor via the operational sequence display provides: three-axis automatic maneuver, tracking local-vertical/local-horizontal about any body vector, rotating about any body vector at the DAP discrete rate, and stopping any of these options and commanding attitude hold. The parameters of these maneuvers are displayed in current attitude, required attitude, attitude error, and body rates. Either total or DAP attitude errors may be selected for the display and the ADI error needles.

The automatic maneuver option calculates a commanded vehicle attitude and angular rate or to hold a vehicle attitude. The desired inertial commanded and rotation attitude is input into the operational sequence display in pitch, yaw, and roll. When the maneuver option is selected, universal pointing sends the required attitude increment and body rate to flight control, and flight control performs the maneuver when the DAP is in automatic.

The automatic rotation option calculates a rotation about a desired body axis. This option is used for passive thermal control, also known as barbecue. Pitch and yaw body components of the desired rotational axis are first input. The orbiter is maneuvered automatically or manually so that the rotational axis is oriented properly in inertial space. When the rotational option is selected, universal pointing will calculate the required body attitude and send it to flight control. Flight control performs the maneuver if the DAP is in automatic.

The local-vertical/local-horizontal automatic option calculates the attitude necessary to maintain LVLH with a desired body orientation. It calculates the attitude necessary to track the center of the earth with a given body vector. Pitch and yaw body components of the desired pointing vector are input first. Omicron (roll angle about the pointing vector) is then input. When the LVLH option is selected, the required LVLH attitude and the maneuver to get to that attitude are calculated and sent to flight control. When the LVLH attitude is reached, universal pointing will calculate an attitude and send it to flight control. Flight control performs the maneuver if the DAP is in automatic

The automatic stop/attitude hold option cancels universal pointing processing of the automatic maneuver, rotation, or LVLH options. When the stop/attitude hold option is selected, universal pointing will cancel the processing of the maneuver options and send the current attitude to flight control. When flight control is in automatic, attitude hold will be initiated about the current attitude.

Manual rotation commands are controlled by the three RHC's and either of the two orbital DAP control panels.

The rotation discrete rate pushbuttons select the "DAP-load" rates in effect (one for normal and one for vernier) designating specific rotational rates in the manual rotation mode. These body rates are commanded as long as the RHC is between the detent and the software softstop for the axis selected. Attitude hold is commanded when the RHC is in the detent, and the attitude reference used is the attitude sensed at the time the RHC reached the detent. Free drift exists on the affected axis between the softstop and the detent when the RHC is returning to the detent after having been past the softstop.

The "Rotation Accel" pushbutton changes the mode to acceleration (continuous thrusting) when depressed and the RHC moved out of the affected axis detent in the manual rotation mode. This mode is available any time the RHC is past the software stop, regardless of mode selected. Free drift is in effect when the RHC is in the detent of the axis or axes selected.

The "Rotation Pulse" pushbuttons enable a one-time rotation (open-loop) rate command in accordance with the value selected in the manual rotation moding. Body rate increments are set for both primary and vernier RCS jets as well as control acceleration available for each body axis for both primary and vernier RCS jets. Each time the RHC is out of detent, a command is generated at a predetermined body rate about a selected axis or axes. It is used in the calculation of the time required for the orbiter to reach the commanded rate increment (or rate increment sum), hence the time the command is active to the RCS jet selection logic. This rate is commanded each time the RHC is moved out of detent and is cumulative for multiple RHC commands. Free drift is in effect when the RHC is below the softstop.

Rotation options may be mixed as they are made axis by axis. Requests for changes in rotation options with the manual mode pushbuttons are not recognized when the RHC is out of detent.

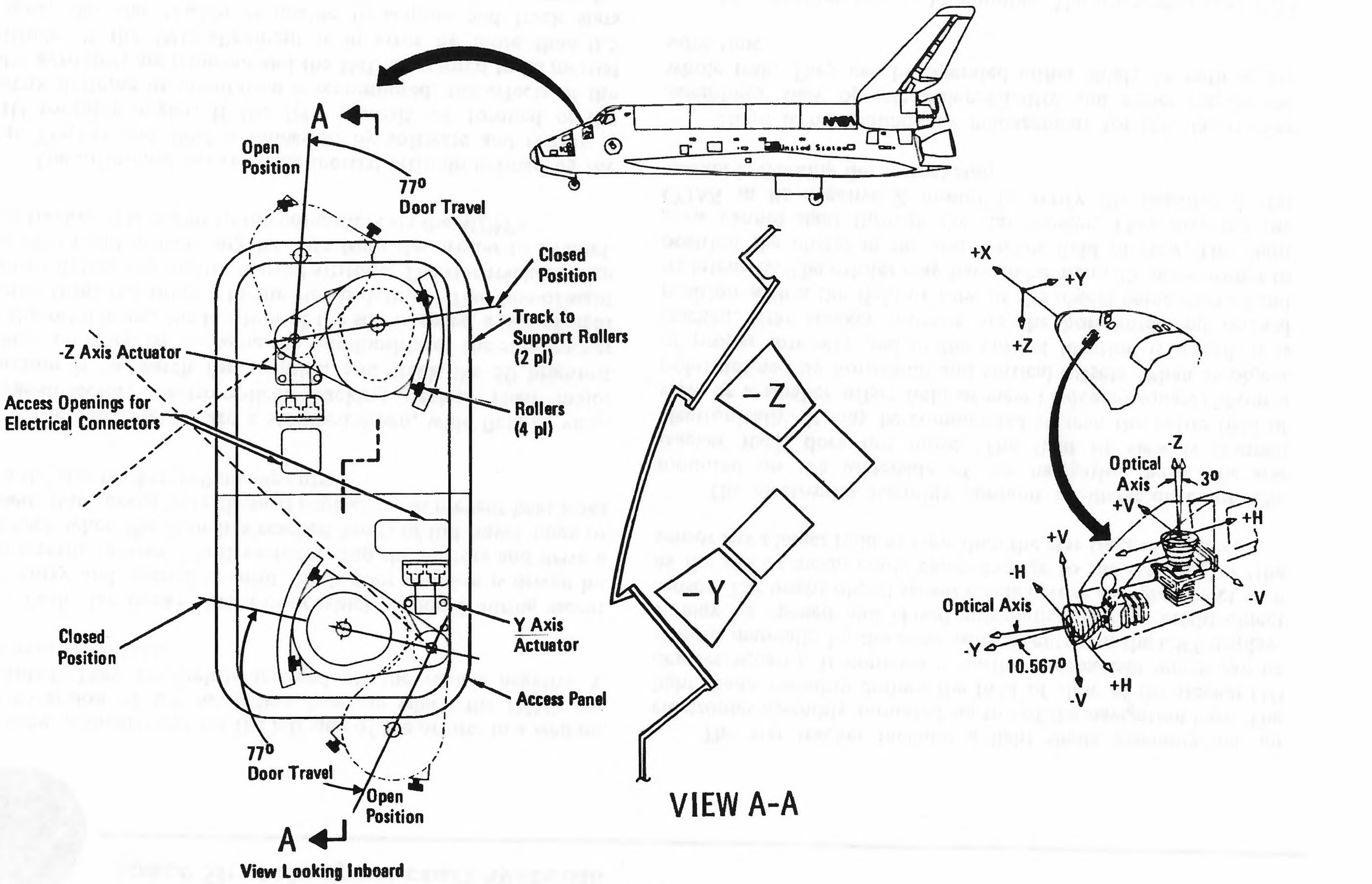
Manual translation commands are controlled by the two THC's and either of the two "Orbital DAP" control panels.

There are two translation acceleration options: high and normal. The high mode commands all RCS jets in that axis to thrust, whereas in the normal mode, only selected RCS jets thrust. The moding is a function of the jet selection logic software and cannot be changed by the flight crew. Applicable RCS jets will thrust continuously as long as the THC is out of detent in one or more axes.

The "Translation Pulse" pushbuttons select a specific delta velocity in feet per second each time the THC is moved out of the detent for the axis or axes concerned in the manual translation mode. The commands are cumulative, and the incremental delta velocity is the value set on the DAP specialist display.

The attitude processor combines commanded attitude with the reference from the IMU to form attitude errors, which are sent to the ADI along with attitude. Body rates from flight control also are sent to the ADI.

STAR TRACKERS. The two star trackers are used in orbit to align the IMU's. The star trackers are located just forward of


Open
Position
A
77°
Door Travel
Closed
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Support Rollers
(2 pl)
-Z Axis Actuator
Access Openings for
Electrical Connectors
Rollers
(4 pl)
Closed
Position
Y Axis
Actuator
Access Panel
77°
Door Travel
Open
Position
A
View Looking Inboard
+X
+Y
+Z
-Z
-Y
VIEW A-A
Z
Optical
Axis
3°
+V
+H
+V
-H
-V
Optical Axis
-Y
10.567°
+H
-V

Star Tr- -ers

the crew compartment on the left side of the orbiter in a well on the extension of the navigation base on which the IMU's are mounted. They are slightly inclined off the vehicle negative Y and negative Z axes.

Each star tracker has a door which is closed during ascent and entry and opened in orbit. Each door rotation is driven by two electric motors. Limit switches stop the motors and drive a talkback when the door has reached limits of full travel, open or closed. Both doors have thermal protection to prevent heat leaks into the star tracker well during entry.

The star trackers are a strapped-down, wide field of view, image-dissector, electro-optical tracking device. Their major function is to search for, acquire, and track the 50 brightest navigation stars. By knowing the relationship of the star tracker to the orbiter and the location of the star in space, a line-of-sight vector from the orbiter to the star is defined. Two line-of-sight vectors define the orbiter inertial attitude. The star trackers align the IMU's and provide angular data from the orbiter to a target. Star tracker data is sent to the computers via the MDM's.

The difference between the inertial attitude defined by the star Tracker and IMU is processed by software and results in IMU torquing angles. If the IMU gimbals are torqued or the matrix defining its orientation is recomputed, the effects of the IMU gyro drift are removed and the IMU is restored to its inertial attitude. If the IMU alignment is in error by more than 0.5 degree, the star tracker is unable to acquire and track stars correctly. This is because the angles the star tracker is given for searching are based on the current knowledge of the orbiter attitude, which is based on IMU gimbal angles. If that attitude is greatly in error, the star tracker may acquire and track the wrong star. In this case, the crew optical alignment sight must be used to re-align the IMU's to within 0.5 degree and then the star trackers used to re-align the IMU's more precisely.

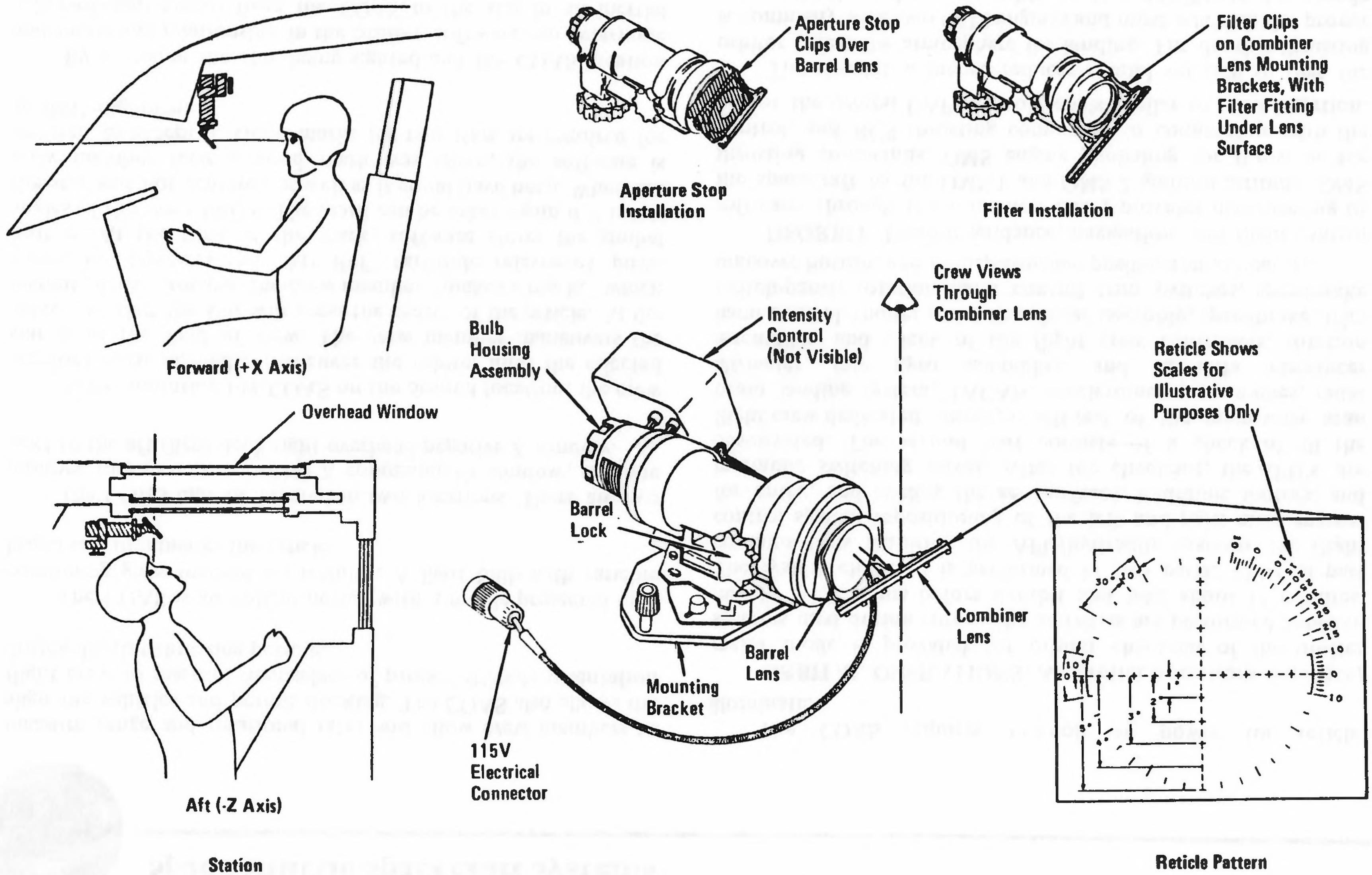
The star tracker includes a light shade assembly and an electronics assembly mounted on top of the navigation base. The light shade assembly defines the field of view of the tracker (10 degrees square). It contains a shutter mechanism which can be opened manually by the crew using an entry on the CRT display. It may be opened and closed automatically by a bright object sensor. The bright object sensor reacts before a bright object such as the sun or moon could cause damage to the star tracker (the sensor has a larger field of view than the star tracker shutter).

The electronics assembly contains an image dissector tube mounted on the underside of the navigation base. The star tracker itself does not move. The field of view is scanned electronically. It may be commanded to scan the entire field of view or a smaller offset field of view (1 degree square) about a point defined by horizontal and vertical offsets. When an object of proper intensity and in the correct location is sensed, it is tracked. Star tracker outputs are the horizontal and vertical position within the field of view of the object being tracked and its intensity. The orbiter may have to be manually maneuvered to position the object in the star tracker field of view. The flight crew cannot sight through the star tracker. They may use the COAS in its negative Z mount to verify the negative Z star tracker is tracking the correct star.

There is no redundancy management for the star tracker assemblies; they operate independently and either can do the whole task. They can be operated either singly or both at the same time.

The warmup time is 15 minutes. The door-open time with two electric motors is 60 seconds and with one motor is 120 seconds.

CREW OPTICAL ALIGNMENT SIGHT. This is used as a backup to the star trackers for aligning the IMU's. A secondary use of the COAS is as the primary optical docking device to



Crewman Opti[illegible]lignment Sight

measure range and rotational rates and allow crew members to align the vehicles and permit docking. The COAS also allows the flight crew to reassure themselves of proper attitude orientation during deorbit thrusting periods.

The COAS is an optical device with a reticle projected on a combining glass focused on infinity. A light bulb with variable brightness illuminates the reticle.

The COAS may be located in two locations. There are two mounts, one over the positive X commander's window, and one next to the aft flight deck right overhead negative Z window.

After mounting the COAS on the desired location, the crew member must manually maneuver the orbiter until the selected star is in the field of view. The crew member maneuvers the orbiter so that the star will cross the center of the reticle. At the instant of the crossing, the crew member "makes a mark," which means he depresses the "Att Ref" (attitude reference) pushbutton. At the time of the mark, software stores the gimbal angles of the three IMU's. The mark can be taken again if it is felt the star was not centered as well as it could have been. When the crew member feels a good mark was taken, the software is notified to accept it. Good marks for two stars are required for an IMU alignment.

By knowing the star being sighted and the COAS location and mounting relationship in the orbiter, software can determine a line-of-sight vector from the COAS to the star in an inertial coordinate system. Line-of-sight vectors to two stars define the attitude of the orbiter in inertial space. This attitude can be compared to the attitude defined by the IMU's and, if the IMU's are in error, they can be realigned to the more correct orientation by the COAS sightings.

An additional COAS is used at the aft flight deck station to check the alignment of the payload bay doors.

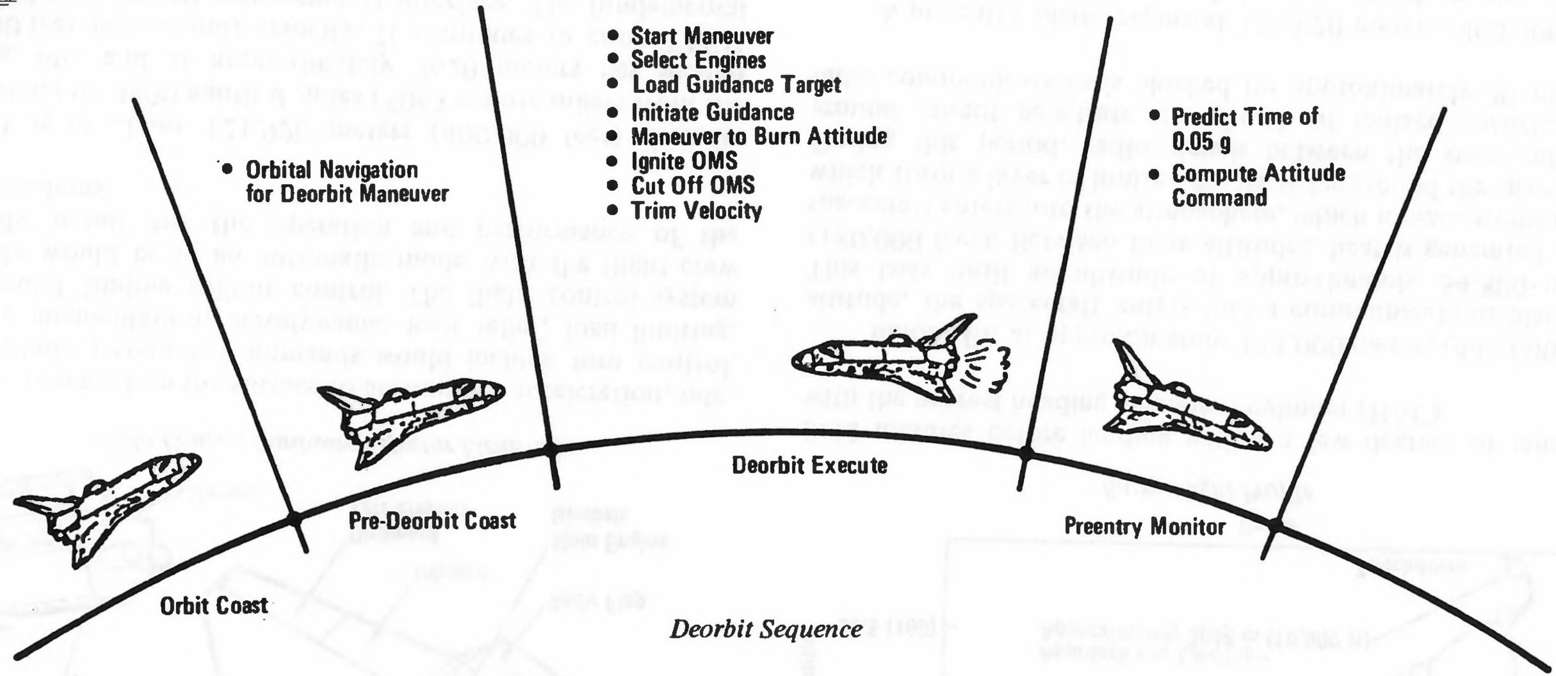
The COAS requires 115-volt ac power for reticle illumination.

ORBITAL OPERATIONS. An orbital operations sequence/ major mode is provided for orbital checkout of the orbiter systems used during entry. The activities are performed approximately 5-1/2 hours before deorbit and take about 15 minutes. The system checkout is performed in two parts. The first part contains tests requiring the APU/hydraulic systems, the flight control system repositioning of the left and right main engines for entry, and cycling the aerosurfaces, hydraulic motors, and hydraulic switching valves. After the checkout, the APU's are deactivated. The second part consists of a check of all the flight crew dedicated displays; self-test of the microwave scan beam landing system, TACAN, accelerometer assemblies, radar altimeter, rate gyro assemblies, and air data transducer assemblies; and check of the flight crew controllers, rotation hand control, rudder pedal transducer assembly, speedbrake, trim switch-panel, rotation hand control trim switches, speedbrake takeover button, and mode/sequence pushbutton indicators.

DEORBIT. Deorbit guidance, navigation, and flight control software, through the transition DAP, provides maneuvering of the spacecraft to the OMS-1 and OMS-2 ignition attitude, OMS thrusting commands, OMS engine gimbaling for thrust vector control, and RCS thrusting commands in conjunction with the use of the orbital DAP control panels similar to orbit insertion.

The deorbit sequence reduces orbital velocity so that the orbiter enters the atmosphere for landing. The deorbit thrusting is nominally with two OMS engines and must establish the proper entry velocity and range conditions. It is possible to downmode to one OMS engine or plus X aft RCS jets for deorbit. The orbiter is positioned to a retrograde, tail-first thrusting attitude for deorbit.

Before deorbit, the flight crew manually orients the spacecraft to point the spacecraft negative Z axis at the sun. Approximately two hours before deorbit, the ECLSS high-load flash evaporator is activated for cooling of the Freon-21 coolant

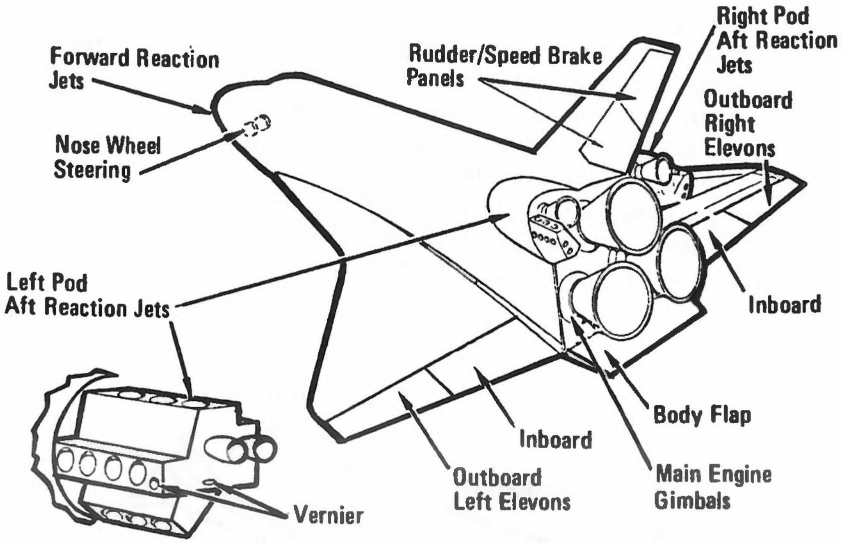


Deorbit Sequence

loops as the ECLSS radiators are deactivated and the payload bay doors are closed. The high-load flash evaporator cools the ECLSS Freon-21 coolant loops until the ECLSS ammonia boilers are activated by the computers at approximately 42,672 meters (140,000 feet) altitude during entry. The orbiter IMU's are aligned, the star trackers deactivated, and the star tracker doors closed. At 50 minutes before deorbit (in the initial development flights), the flight crew takes their seats and dons helmets. The spacecraft is then manually maneuvered utilizing the RCS jets to the deorbit attitude (retrograde). About 30 minutes before deorbit, the OMS preparations for the deorbit thrusting are accomplished. They consist of OMS thrust vector control gimbal checks, OMS system data checks, landing gear hydraulics, and APU reactivation (they remain operating throughout entry and landing rollout). The OMS deorbit thrusting consists of two OMS thrusting periods. At the completion of the thrusting, the flight crew manually maneuvers the spacecraft to orient the nose first for the entry sequence utilizing the RCS jets. The spacecraft hydraulic fluid system thermal conditioning also begins, if required.

ENTRY THROUGH LANDING. Guidance, navigation, and flight control software for the entry phase is initiated by the flight crew approximately five minutes before entry interface (EI), and the DAP processes the flight control laws in response to automatic or manual commands and feedback sensor signals that indicate the dynamic state of the spacecraft. The entry interface is the point at which aerodynamic forces are sensed.

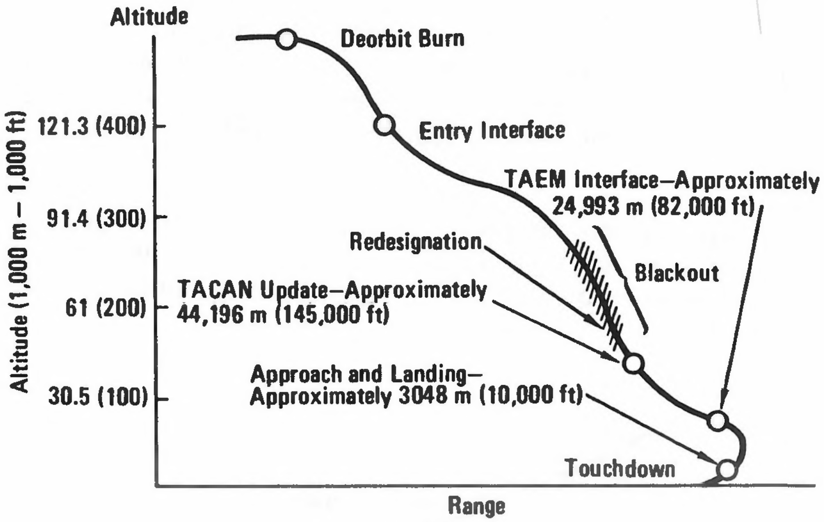
RCS jets (pitch, roll, and yaw) are commanded and actuators for aerosurfaces (elevons, body flap, rudder/speedbrake) are commanded once sufficient aerodyna-



Flight Control Hardware Effector Elements

pressure is sensed on the surface to accomplish acceleration, rate, and attitude response. Commands would include turn control, stability augmentation, aerodynamic load relief, load limiting, and ground landing rollout control. The flight control system normally would be in an automatic mode with the flight crew primarily monitoring the operation and performance of the GN&C systems.

EI is at about 121,920 meters (400,000 feet) altitude, approximately 4400 nautical miles (5063 statute miles) from the landing site, and at approximately 7620 meters per second (25,000 feet per second) velocity. It continues to entry-TAEM (terminal area energy management) interface. The fundamental guidance requirement during entry is to reach the TAEM interface at approximately 25,298 meters (83,000 feet) altitude, 762 meters per second (2500 feet per second) velocity, and on range 52 nautical miles (59 statute miles) from the landing runway and at a time interval between 34:45 minutes and



Entry Flight Profile

5:32 minutes before landing within a few degrees of tangency with the nearest heading alignment cylinder (HAC).

Before EI, at approximately 134,000 meters (440,000 feet) altitude, the spacecraft enters into a communications blackout. This lasts until an altitude of approximately 54,800 meters (180,000 feet). Between these altitudes, heat is generated as the spacecraft enters into the atmosphere, which ionizes atoms of air, which form a layer of ionized gas particles around the spacecraft. During this period, radio signals between the spacecraft and ground cannot penetrate this sheath of ionized particles, and radio communications is blocked for approximately 20 minutes.

A pre-entry phase begins at 121,920 meters (400,000 feet) altitude in which the spacecraft is maneuvered to zero degrees roll and yaw (wings level) and a predetermined angle of attack for entry. In the initial development flight tests, the angle of attack is 40°; in subsequent flights, it will be between 38° and 28°. The flight control system issues the commands to the roll, yaw, and pitch RCS jets for rate damping in attitude hold for entry into the earth's atmosphere until 0.176 g is sensed which corresponds to a dynamic pressure of 517 millimeters of mercury (mmHg) per square meter (10 pounds per square foot), approximately the point where the aerosurfaces become active.

The forward RCS jets are inhibited at 121,920 meters (400,000 feet) altitude. The aft RCS jets maneuver the spacecraft until a dynamic pressure of 517 mmHg per square meter (10 pounds per square foot) is sensed which is when the orbiter's ailerons become effective and the aft RCS roll jets are deactivated. At a dynamic pressure of 1035 mmHg per square meter (20 pounds per square foot), the orbiter's elevators become effective and the aft RCS pitch jets are deactivated. The orbiter's speedbrake is used below Mach 10 to induce a more positive downward elevator trim deflection. At Mach 3.5, the rudder becomes activated and the aft RCS yaw jets are deactivated at 13,716 meters (45,000 feet).

In the entry phase, the RCS commands roll, pitch, and yaw. Lights on the commander's flight deck display and control panel station are used to indicate the presence of an RCS command from the flight control system to the RCS jet selection logic; however, this does not indicate an actual RCS jet thrusting command. The minimum light-on duration is extended to allow the light to be seen even for minimum-impulse RCS jet thrusting commands. After the roll and pitch aft RCS jets are deactivated, the roll indicator lights are used to show that three or more yaw RCS jets have been requested. The pitch indicator lights are used to show elevon rate saturation.

Entry guidance must dissipate the tremendous amount of energy the orbiter posseses when it enters the earth's atmosphere ensuring the orbiter does not burn up (entry angle too steep) or skip out of the atmosphere (entry angle too shallow) and at TAEM interface the orbiter is capable of reaching the desired t down point.

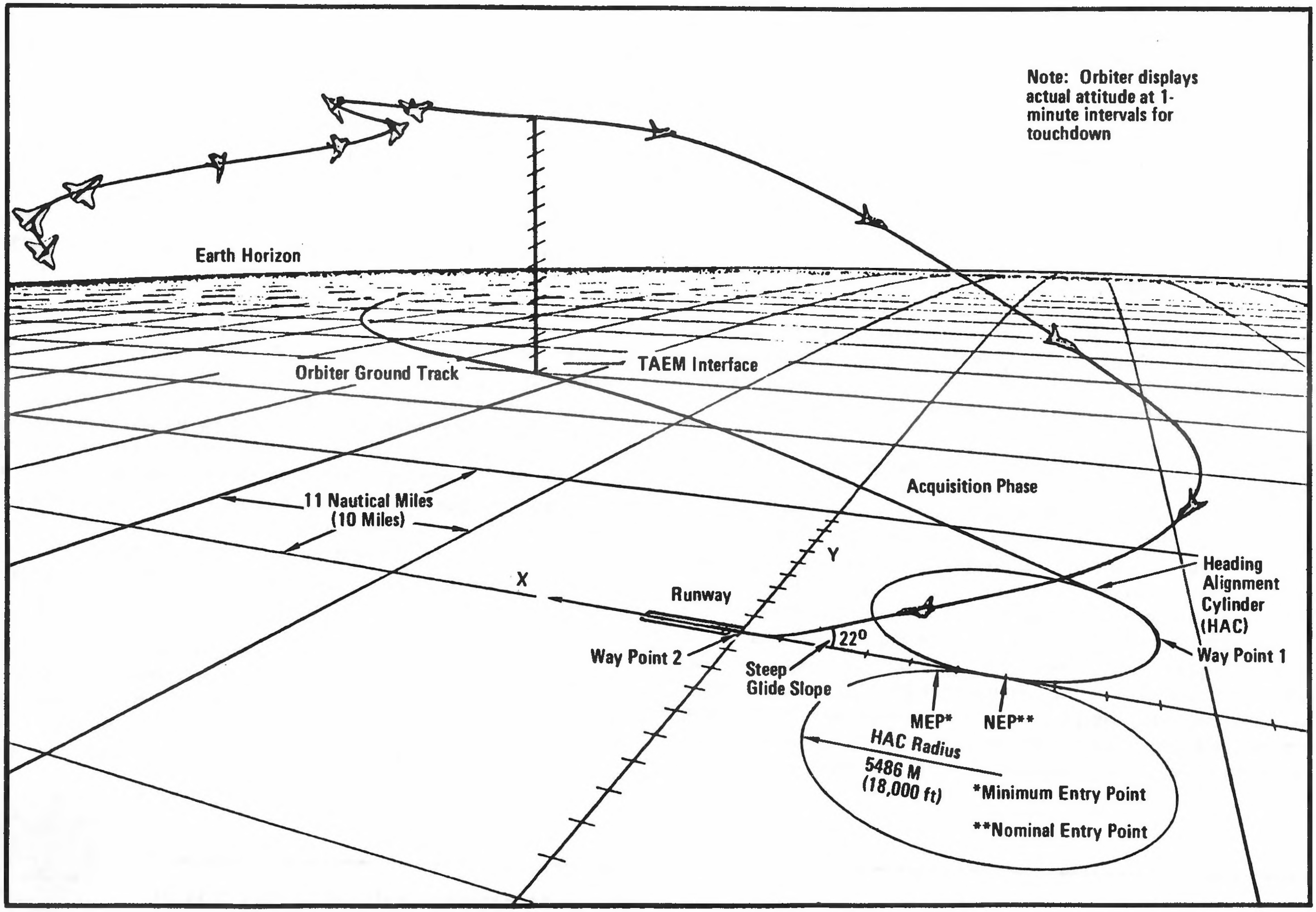
During entry, energy is dissipated by the atmospheric drag on the orbiter's surface. Higher atmospheric drag levels enable faster energy dissipation with a steeper trajectory. Normally, the angle of attack and roll angle enable the atmospheric drag of any flight vehicle to be controlled. However, for the orbiter, angle of attack was rejected because it creates surface temperatures above the design specification. The angle of attack schedule used during entry is initialized-loaded as a function of relative velocity leaving roll angle for energy control. Increasing the roll angle decreases the vertical component of lift, causing a higher sink rate and energy dissipation rate. Increasing the roll rate does raise the surface temperature of the orbiter, but not nearly as drastically as an equal angle of attack command.

If the orbiter is low on energy (current range-to-go much greater than nominal at current velocity) entry guidance will command lower than nominal drag levels. If the orbiter has too much energy (current range-to-go much less than nominal at the current velocity) entry guidance will command higher than nominal drag levels to dissipate the extra energy.

Roll angle is used to control crossrange. Azimuth error is the angle between the plane containing the orbiter's position vector and the heading alignment cylinder tangency point and the plane containing the orbiter's position vector and velocity vector. When the azimuth error exceeds an initialized-loaded number, the orbiter's roll angle is reversed.

Thus descent, rate and down ranging are controlled by bank angles, the steeper the bank angle, the greater the descent rate and the greater the drag, conversely the minimum drag altitude is wings level. Cross-range is controlled by bank reversals.

The entry thermal control phase is designed to keep the backface temperatures within the design limits. A constant heating rate is established until below 5791 meters per second (19,000 feet per second).



Entry Flight Profile

The equilibrium glide phase transitions the orbiter from the rapidly increasing drag levels of the temperature control phase to the constant drag level of the constant drag phase. The equilibrium glide flight is defined as flight in which the flight path angle, the angle between the local horizontal and the local velocity vector, remains constant. Equilibrium glide flight provides the maximum downrange capability. It lasts until the drag acceleration reaches 33 ft/second squared.

The constant drag phase begins at 33 ft/second squared. In the initial development flight the angle of attack is initially 40 degrees but it begins to ramp down in this phase to approximately 36 degrees by the end of this phase.

The transition phase is where the angle of attack continues to ramp down, reaching the approximate 14 degree angle of attack at the entry TAEM interface, approximately 25,298 meters (83,000 feet) altitude, 762 meters per second (2500 feet per second), Mach 2.5 and 52 nautical miles (59 statute miles) from the landing runway. Control is then transferred to TAEM guidance.

It is noted, that during the aforementioned entry phases, the orbiter's roll commands keep the orbiter on the drag profile and control cross range.

Terminal area energy management guidance in entry begins at approximately 25,298 meters (83,000 feet) altitude and 762 meters per second (2500 feet per second) and terminates at the A/L (approach and landing) guidance phase capture zone, which begins at approximately 3048 meters (10,000 feet) altitude, at Mach 0.9. The spacecraft attains subsonic velocity at approximately 14,935 meters (49,000 feet) at a range of over 22 nautical miles (25 statute miles) from the landing site.

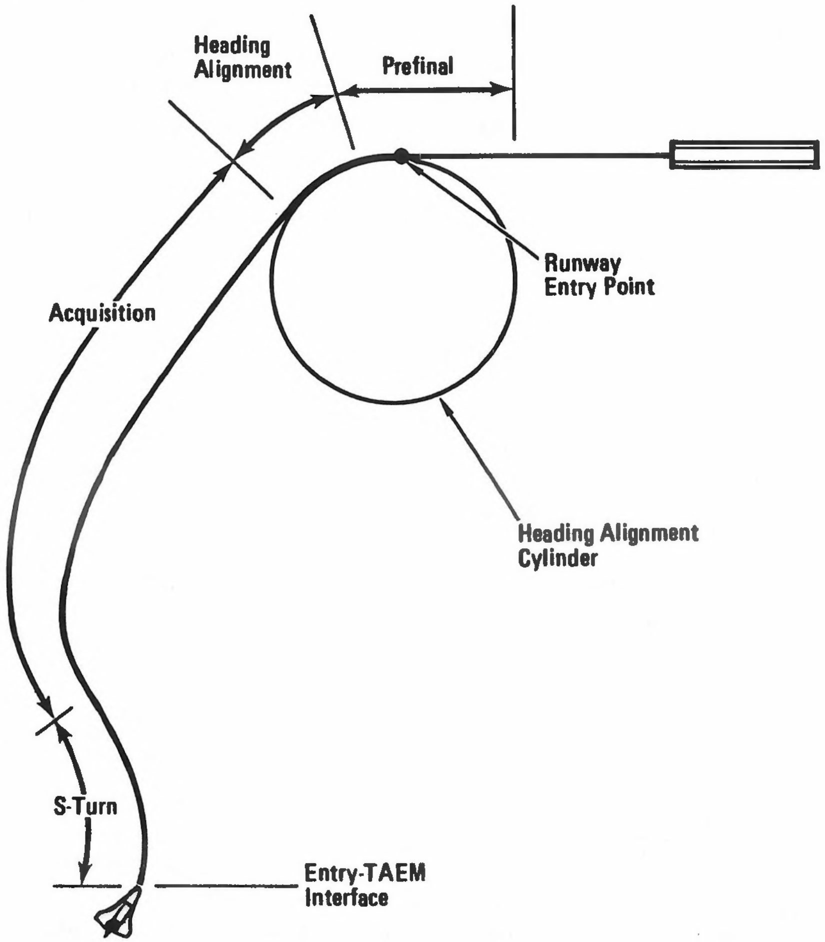
TAEM guidance steers the orbiter to the nearest of two 'ing alignment cylinders whose radius are 5486 meters (18,000 feet), which are located tangent to and on either side of the runway centerline on the approach end. In TAEM guidance, excess energy is dissipated with an S turn and the speedbrake can be utilized to modify drag, L/D (lift/drag) ratio, and the flight path angle in high energy conditions. This increases the ground track range as the orbiter turns away from the nearest HAC until sufficient energy is dissipated to allow a normal A/L guidance phase capture, which begins at 3048 meters (10,000 feet) altitude at the nominal entry point (NEP). The orbiter also can be flown near the velocity for maximum lift over drag or wings level for the range stretch case, which moves the A/L guidance phase to the MEP (minimum entry point).

At TAEM acquisition, the orbiter is turned until it is aimed at a point tangent to the nearest HAC and continues until it reaches the point, WP-1 (waypoint one). At WP-1 the TAEM heading alignment phase begins in which the HAC is followed until landing runway alignment plus or minus 20 degrees has been achieved.

In the TAEM prefinal phase, the orbiter leaves the HAC, pitches down to acquire the steep glide slope, increases airspeed, and banks to acquire the runway centerline and continues until on the runway centerline, on the outer glide slope and on airspeed.

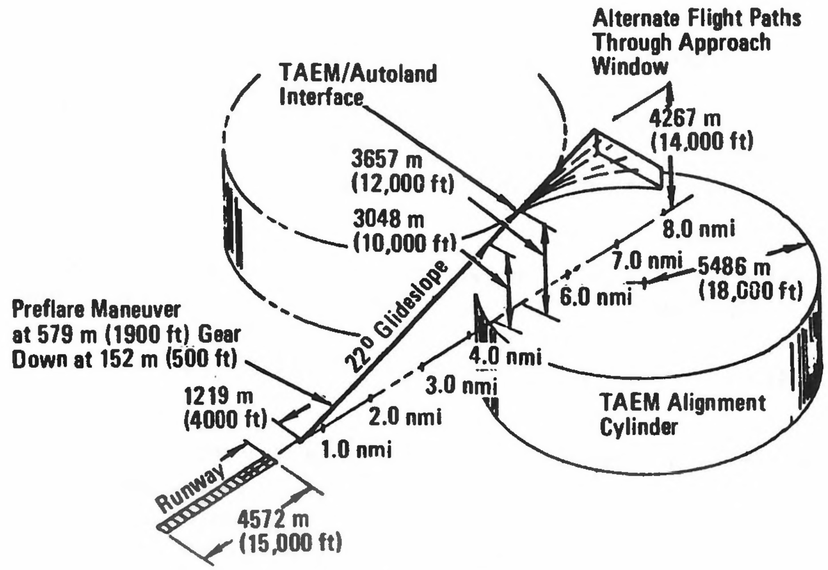
The A/L guidance phase begins with the completion of the TAEM prefinal phase and ends when the craft comes to a complete stop on the runway. The A/L interface airspeed requirement at 3048 meters (10,000 feet) altitude is 290 plus or minus 12 knots, equivalent airspeed (EAS), 6.9 nautical miles (7.9 statute miles) from touchdown.

Autoland guidance is initiated at this point to guide the orbiter to the minus 20° glide slope (which is over seven times that of a commercial airliner's approach) aimed at a target approximately 0.86 nautical mile (one statute mile) in front of the runway. The descent rate in the later portion of TAEM

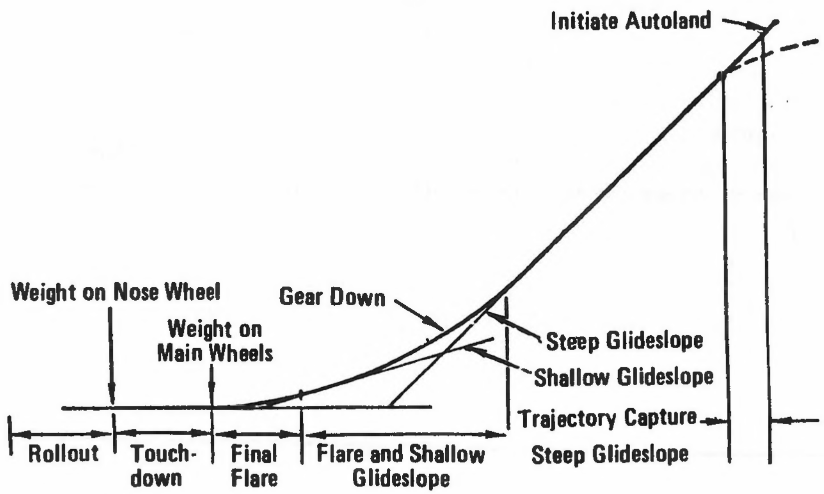


TAEM Guidance Phases

A/L is greater than 3048 meters per minute (10,000 feet per minute–approximately 20 times higher rate of descent than a commercial airliner's standard three-degree instrument approach angle). The steep glide slope is tracked in azimuth and elevation and the speedbrake is positioned as required.



TAEM/Autoland



Autoland Profile

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At 533 meters (1750 feet) above ground level a pre-flare maneuver is started to position the spacecraft for a 1.5^{o} glide slope in preparation for landing with the speedbrake positioned as required. The flight crew deploys the landing gear at this point.

The final phase reduces the sink rate of the vehicle to less than 2.7 meters per second (9 feet per second). After the spacecraft crosses the runway threshold–WP-2 in the Autoland mode navigation uses the radar altimeter vertical component of position in the state vector for guidance and navigation computations, this is from an altitude of 30 meters (100 feet) to touchdown. Touchdown occurs approximately 762 meters (2500 feet) past the runway threshold at a speed of 184 to 196 knots (213 to 226 mph).

In the automatic mode, the orbiter is essentially like a missile; the flight crew monitors the instruments to verify that the vehicle is following the correct trajectory. The onboard computers execute the flight control laws (equations). If the vehicle diverges from the trajectory, the flight crew can take over at any time by switching to CSS. The orbiter can fly to a landing in the automatic mode (only landing gear extension and braking action on the runway are required by the flight crew); however, in the initial development flight tests, the automatic mode is used in entry until 9,756 meters (32,000 feet) altitude, and the flight crew will then select CSS for the remainder of the flight. In the subsequent flights, the automatic mode will be utilized for the entire entry through landing rollout.

The navigation system used from entry to landing consists of the IMU's and navigation aids (TACAN, air data system, MSBLS, and radar altimeter). The three IMU's maintain an inertial reference and provide delta velocities until MSBLS is ac ed.

Navigation-derived air data–obtained after deployment of the two air data probes below Mach 3–is needed from entry through landing as inputs to the guidance, flight control, and crew display. TACAN provides range and bearing measurements and is available at approximately 44,196 meters (145,000 feet) altitude and will nominally accept the data into the state vector before 39,624 meters (130,000 feet) altitude. TACAN is used until MSBLS acquisition which provides range, azimuth, and elevation and is expected to occur at approximately 5,486 meters (18,000 feet) altitude. Radar altimeter data is available at approximately 2743 meters (9000 feet).

TACAN acquisition and operation is completely automatic. The crew is provided with the necessary controls and displays to evaluate the TACAN system performance and to take over if required. When the distance to the landing site is approximately 120 nautical miles (138 statute miles), the TACAN begins the navigation region of interrogating six navigation stations. As the spacecraft progresses, the distance to the remaining stations and the next nearest station is computed and the next nearest station will be selected automatically when the spacecraft is closer to it than it is to the previous locked-on station. Only one station is interrogated when the distance to the landing site is less than approximately 20 nautical miles (23 statute miles). Again, the TACAN's will automatically switch from the last locked-on navigation region station to begin searching for the landing site station. TACAN azimuth and range are provided on the CRT horizontal situation display. TACAN range and bearing cannot be used to produce a good estimate of the altitude position component, so navigation uses barometric altitude derived from the air data system probes.

MSBLS acquisition and operation is completely automatic; the flight crew provided with controls and displays to evaluate system performance and to take over if required. MSBLS acquisition is expected to occur at approximately 5,486 mete

(18,000 feet) and approximately eight nautical miles (9.2 statute miles) from the runway. The range and azimuth measurements are provided by a ground antenna located at the end of the runway and to the left of the runway centerline. Elevation measurements are provided by a ground antenna to the left of the runway centerline approximately 800 meters (2624 feet) from the runway threshold.

During entry, the commander and pilot ADI.s (attitude director indicators) becomes a two-axis ball displaying body roll and pitch attitudes with respect to local vertical/local horizontal. These are generated in the attitude processor from IMU data. The roll and pitch error needles each display the body roll and pitch attitude error with respect to entry guidance commands by using the bank guidance error and the angle-of-attack error generated from the accelerometer assemblies. In atmospheric flight, the roll attitude error and the normal acceleration error are displayed on the roll and pitch error needles, respectively. The sideslip angle is displayed on the yaw error needle. The roll and pitch rate needles display stability roll and body rates by using stability roll rate, rate gyro rate, and pitch rate. The yaw rate needle displays stability yaw rate. After main landing gear touchdown, the yaw error with respect to runway centerline and nose gear slapdown pitch rate error are displayed on the roll and pitch error needles. During rollout, the pitch error indicator indicates pitch error rate.

During entry, the commander and pilot HSI's (horizontal situation indicators) display a pictorial view of the spacecraft's location with respect to various navigation points. During entry, navigation attitude processor provides the inputs to the HSI until after leaving the communications blackout at approximately 44,196 meters (145,000 feet) altitude. Then TACAN is acquired and accepted for HSI inputs at approximately 39,624 meters (130,000 feet) altitude until MSBLS acquisition at approximately 5,486 meters (18,000 feet), and approximately 8 nautical miles (9.2 statute miles) from the runway.

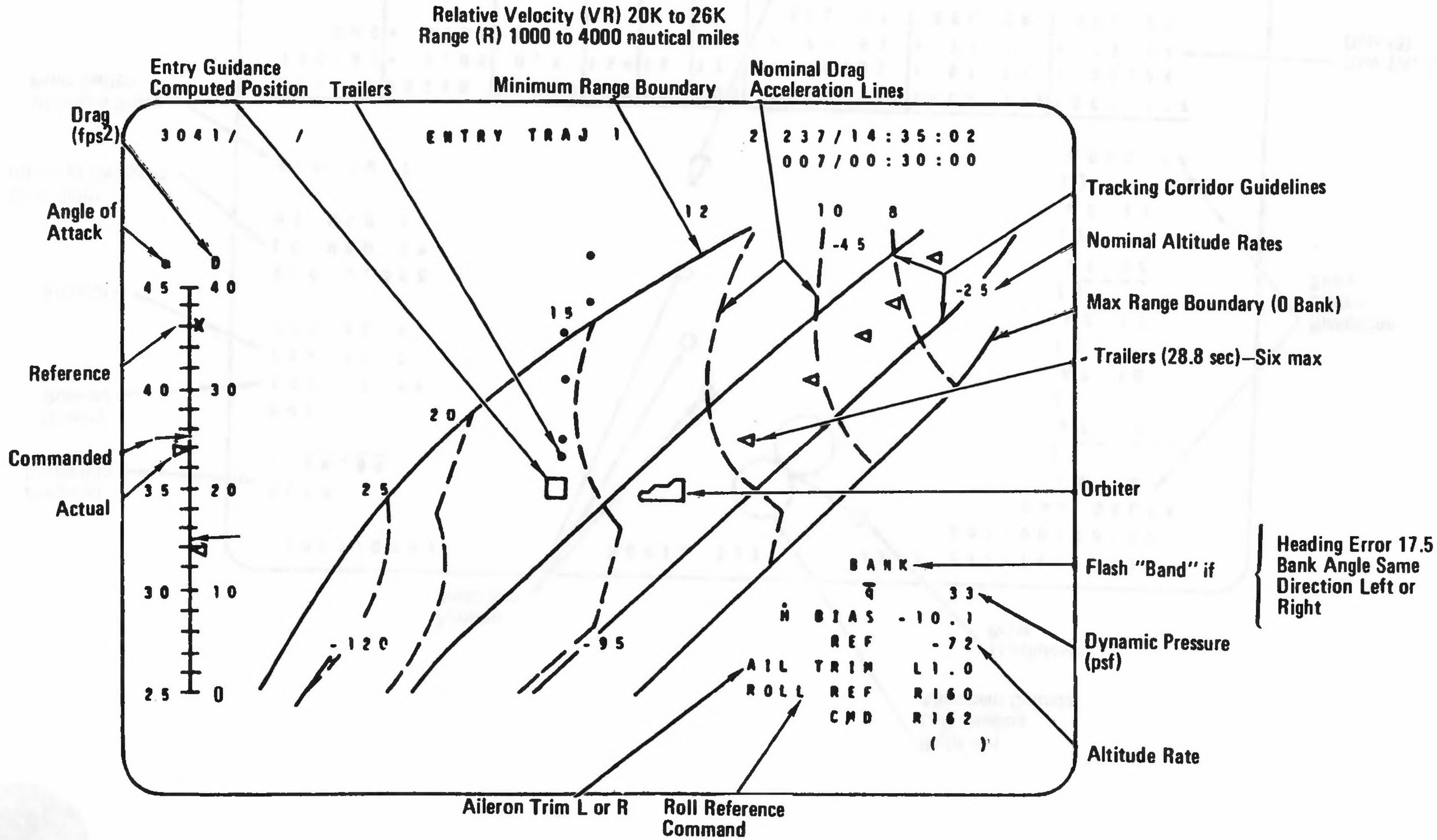
When the approach mode and MLS source are selected for the commander and pilot HSI, data from the microwave scan beam landing system replaces TACAN data. MSBLS azimuth, elevation, and range are used from acquisition until runway threshold is reached and azimuth and range are used to control rollout.

At an altitude of 2743 meters (9000 feet), radar altitude one or two can be selected to measure the nearest terrain within the beamwidth of the altimeters. They provide this indication to the AVVI radar, altitude, meter display from 5000 feet to 0.

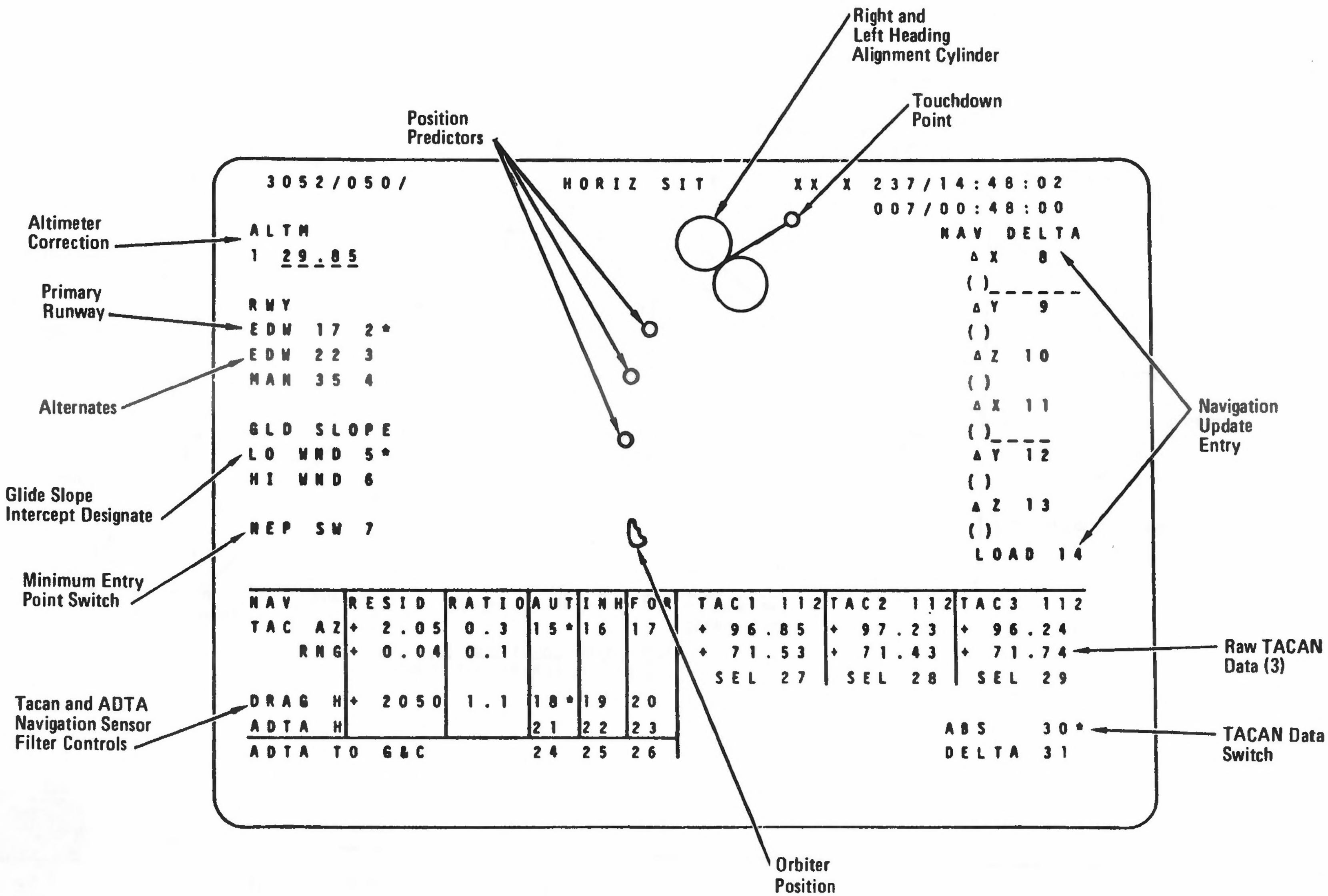
Below Mach 3, the left and right air data system probes are deployed by the flight crew. This system senses air pressures related to the spacecraft movement through the atmosphere for updating the navigation state vector in altitude, guidance in steering and speed-brake command calculations, flight control for control law computations, and for display on the AMI's (alpha Mach indicators) and AVVI's (altitude, vertical velocity indicators).

The AMI's display essential flight parameters relative to the spacecraft's travel in the air mass such as angle of attack, (alpha), acceleration, Mach/velocity, and knots equivalent airspeed. The source of data during the AMI's is determined by the position of the air data select switch. Prior to air data system probe deployment, the AMI's receive their inputs from the navigation attitude processor. When the air data probes are deployed, the left or right air data system provides the inputs to the AMI's except for the acceleration indicator, which remains on the navigation attitude processor.

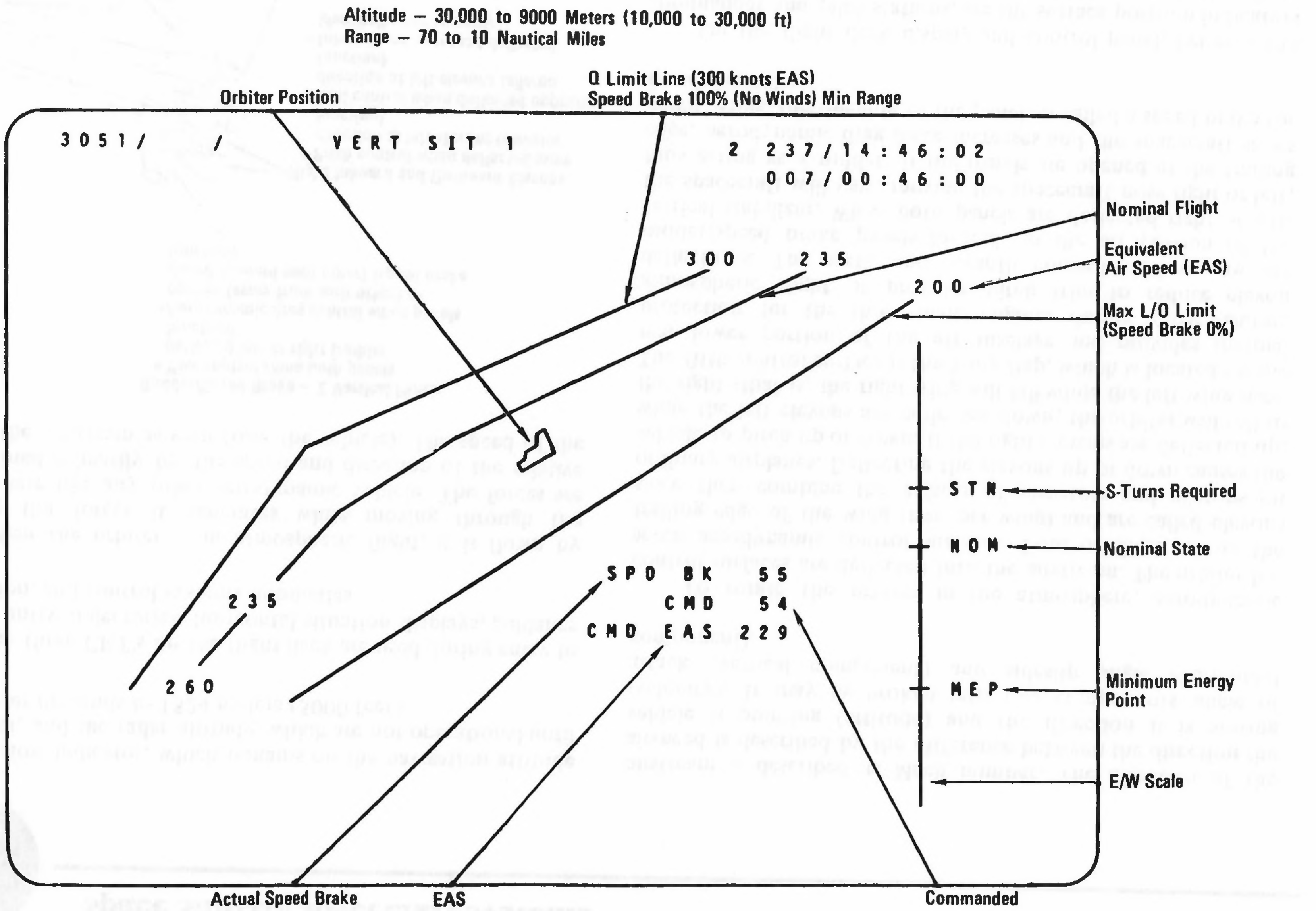
The AVVI's display radar altitude, barometric altitude, altitude rate, and altitude acceleration. The data driving the AVVI's is determined by the air data select switch. When the air data probes are deployed, the right or left air data system provides the inputs to the AVVI's, except for the altitude



Entry Trajectory



Horizontal Situation Display

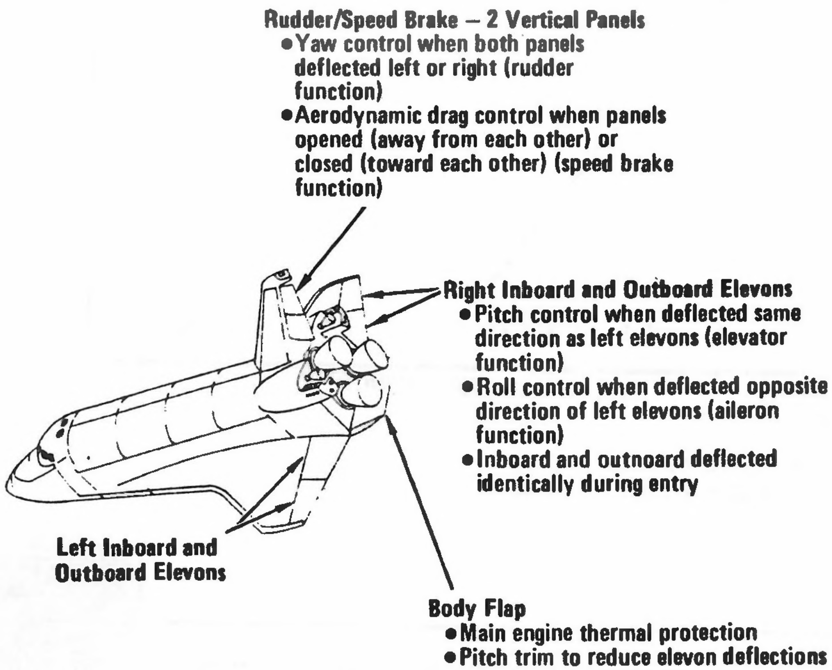


Vertical Situation Display

acceleration indicator, which remains on the navigation attitude processor, and the radar altitude, which are not operational until the orbiter descends to 1524 meters (5000 feet).

The three CRT's on the flight deck are used during entry to display entry trajectories, horizontal situation displays, guidance navigation, and control systems summaries.

When the orbiter is in atmospheric flight, it is flown by varying the forces it generates while moving through the atmosphere like any other aerodynamic vehicle. The forces are determined primarily by the speed and direction of the relative wind (the airstream as seen from the vehicle). The speed of the



Aerodynamic Surfaces

airstream is described as Mach number. The direction of the airspeed is described by the difference between the direction the vehicle is pointing (attitude) and the direction it is moving (velocity). It may be broken into two components–angle of attack (vertical component) and sideslip angle (horizontal component).

To rotate the orbiter in the atmosphere, aerodynamic control surfaces are deflected into the airstream. The orbiter has seven aerodynamic control surfaces. Four of these are on the trailing edge of the wing (two per wing) and are called elevons since they combine the effects of elevators and ailerons on ordinary airplanes. Deflecting the elevons up or down causes the vehicle to pitch up or down. If the right elevons are deflected up, while the left elevons are deflected down, the orbiter will roll to the right–that is, the right wing will fall while the left wing rises. The fifth control surface is the body flap, which is located on the rear lower portion of the aft fuselage and provides thermal protection for the three main engines during entry. During atmospheric flight, it provides pitch trim to reduce elevon deflections. The sixth and seventh control surfaces are the rudder/speed brake panels located on the aft portion of the vertical stabilizer. When both panels are deflected right or left, the spacecraft will yaw, moving the spacecraft nose right or left, thus acting as a rudder. If the panels are opened at the trailing edge, aerodynamic drag force increases and the spacecraft slows down. Thus, the opening of the panels is called a speed brake for speed control.

On the flight deck display and control panel, between the commander and pilot stations, are the surface position indicators which display the position of each aerodynamic control surface. The positions of the aerodynamic surfaces are displayed on moving pointer displays. Elevon positions are shown in degrees from plus 20°, full down, to minus 35°, full up. Body flap position is 0 to 100 percent. Rudder position is left 30°, right 30°. Aileron position is left 5°, right 5°. Speed brake position and command are 0 to 100 percent. An off flag is provided in each indicator to indicate power loss, erroneous input signals, or failure in any display channel.

The commander and pilot can select the flight control system modes of operation with pushbutton light indicators on the flight deck display and control glareshield/eyebrow panel. There are two modes–automatic and CSS (control stick steering). The flight crew can select separate modes for pitch and roll/yaw (roll and yaw must be in the same mode). The body flap and speed brake have an automatic and manual mode selection. Each pushbutton is a triple redundant, momentary contact, non-latching switch.

Automatic pitch or automatic roll/yaw are selected by depressing the applicable pushbutton at either the commander or pilot station. When depressed, that pushbutton will illuminate at both stations. Automatic pitch provides automatic control in the pitch axis and the automatic roll/yaw provides automatic control in the roll/yaw axes. During entry, the automatic mode uses the RCS jets until dynamic pressure permits the aerosurfaces to become effective; the aft RCS jets and spacecraft aerosurfaces are then used together until dynamic pressure becomes sufficient for aerosurface control only.

Control in the pitch axis is provided by the elevons, speed brake, and body flap. The elevons provide control to guidance normal acceleration commands, control of pitch rate during slapdown (landing) for nose wheel load protection, and static load relief after slapdown for main landing gear wheel and tire load protection. The speed brake provides control to guidance surface deflection (open/close, increase/decrease velocity) command. The body flap provides control to null elevon deflection. Pushbuttons at the commander and pilot flight deck display and control eyebrow panel illuminate to indicate automatic speed brake or body flap control.

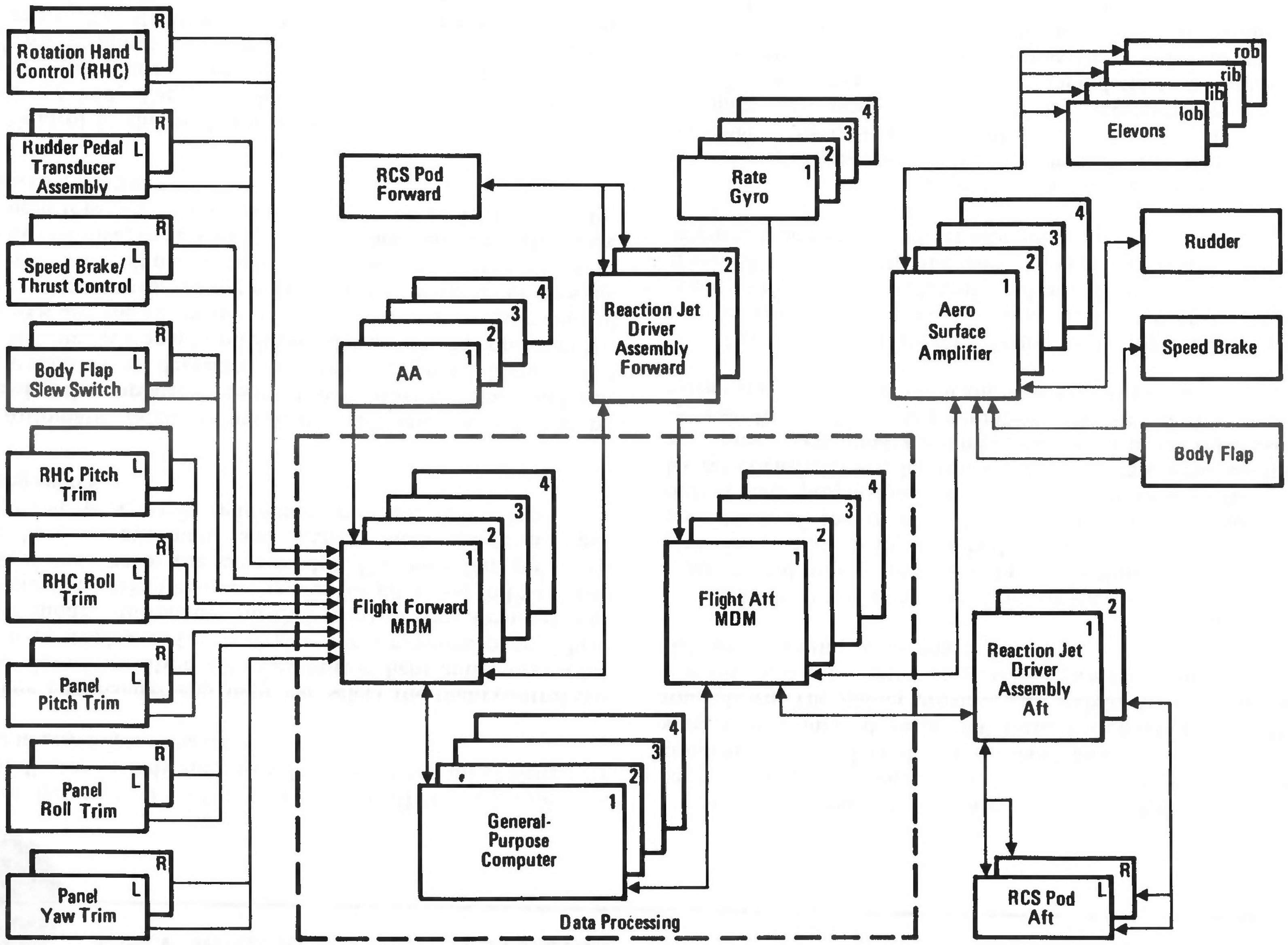
Control in the roll and yaw axes is provided by the elevons and rudder. The elevons provide control to guidance bank angle command during TAEM and autoland and control to guidance wings-level command during flat turns 1.5 meters (5 feet) above touchdown. The rudder provides yaw stabilization during TAEM and autoland and control to guidance yaw rate command during flat turn and subsequent phases.

The three rate gyro assemblies of the flight control system measure and supply output data proportional to the orbiter's attitude rates about its three body axes. The three accelerometer assemblies of the flight control system measure and supply output data proportional to the orbiter's normal (vertical) and lateral (right/left) accelerations. The three rate gyro assemblies and three accelerometer assemblies are used in the flight control system for stability augmentation because of the orbiter's marginal stability in its pitch and yaw axes at subsonic speeds.

The three IMU's are an all-attitude stabilized platform that also measures and supplies output data proportional to the spacecraft's attitude (rotation) and acceleration (velocity). The three IMU's augment the flight control system's rate gyro assemblies and accelerometer assemblies.

The rate gyro assembly pitch rate (rotation) and the accelerometer assembly normal acceleration (velocity) are used to generate elevon (elevator) deflection commands. The rate gyro assembly yaw rate (rotation) and the accelerometer assembly lateral acceleration generate the rudder deflection required for directional stability. The rate gyro assembly roll rate (rotation) generates the elevon (aileron) deflection command required for lateral (roll) stability. The speed brake and body flap positions generate the elevon deflection required for trim near neutral to maximize roll effectiveness of the elevons.

When the orbiter is in the automatic pitch and roll/yaw mode, the flight crew manual control stick steering commands



Flight Control System Hardware

are inhibited. In the CSS mode, the crew flies the orbiter by deflecting the RHC and rudder pedals. The flight control system interprets the RHC motions as rate commands in pitch, roll, or yaw and controls the RCS jets and aerosurfaces. The larger the deflection, the larger the command. The flight control system compares these commands with inputs from rate gyros and accelerometers (what the vehicle is actually doing–motion sensors) and generates control signals to produce the desired rates. If the flight crew releases the RHC, it will return to center, and the orbiter will maintain its present attitude (zero rates). The rudder pedals position the rudder during atmospheric flight; however, in actual use, because flight control software performs automatic turn coordination, the rudder pedals are not used until the wings are leveled prior to touchdown.

The CSS mode is similar to the automatic mode except the flight crew can issue three-axis commands affecting spacecraft motion. They are augmented by the feedback from the same spacecraft motion sensors, except for the normal acceleration (velocity) accelerometer assemblies, to enhance control response and stability. CSS pitch or roll/yaw modes are selected by depressing a pushbutton at either the commander or pilots station. When depressed, that PBI will illuminate to show CSS control in that axis.

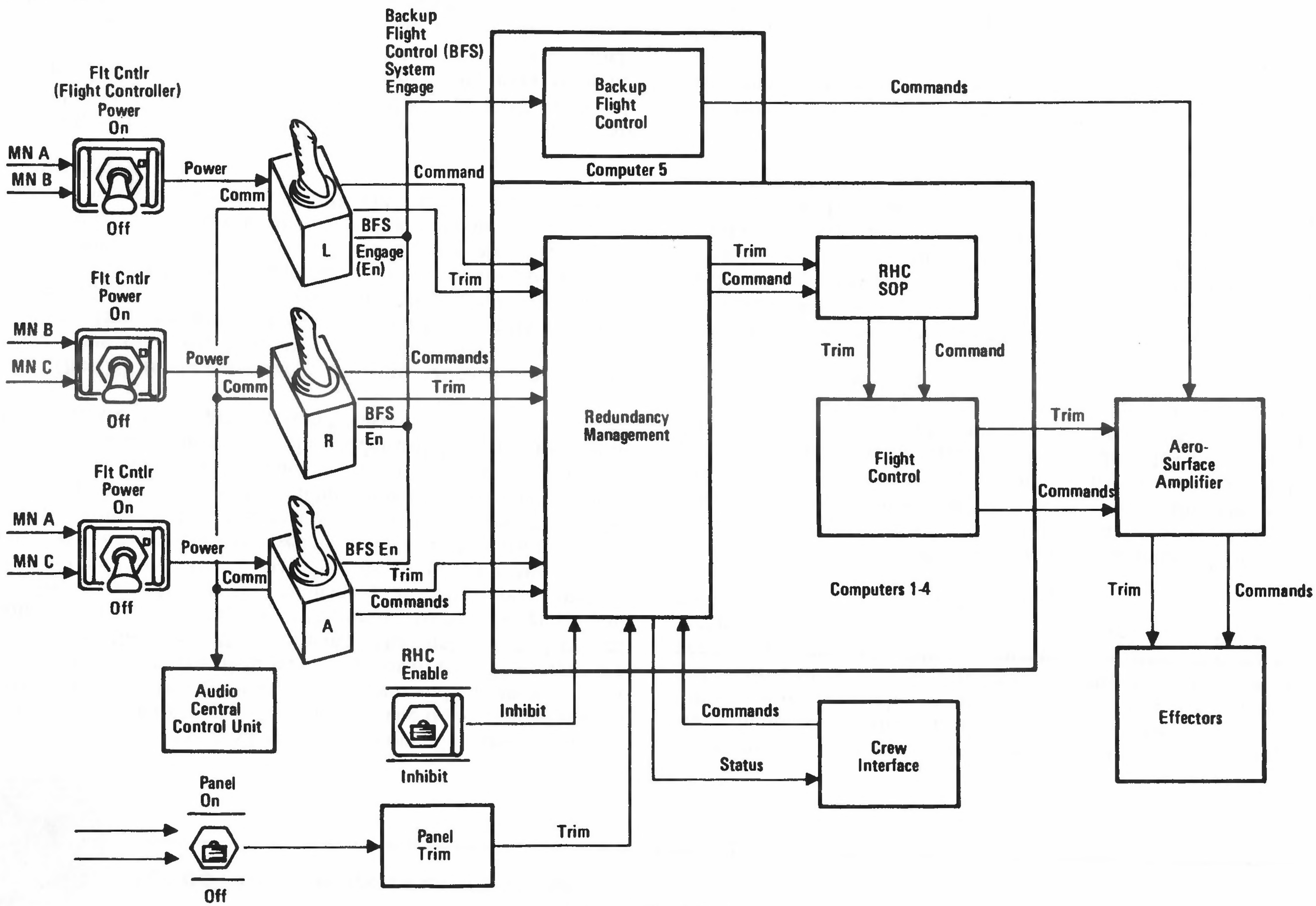
When the orbiter is in the CSS mode, flight crew inputs are provided by the commander or pilot RHC. The RHC commands are processed by the computers together with data from the motion sensors. The flight control module processes the flight control laws and provides commands to the flight control system, which positions the aerosurfaces in atmospheric flight.

Control in the roll/yaw axis is provided by the elevon (ailerons) and the rudder. The elevons provide augmented control to the RHC control. The rudder interface between the roll/yaw channel automatically positions the rudder for coordinated turns. A 'der pedal transducer assembly (RPTA) is provided at the commander and pilot's stations. The two rudder pedal assemblies are connected to their respective RPTA's. Because of the roll/yaw interface, rudder pedal use should not be required until just before touchdown. An artificial feel is provided in the rudder pedal assemblies. The RPTA commands are processed by the computers and the flight control module commands the flight control system to position the rudder.

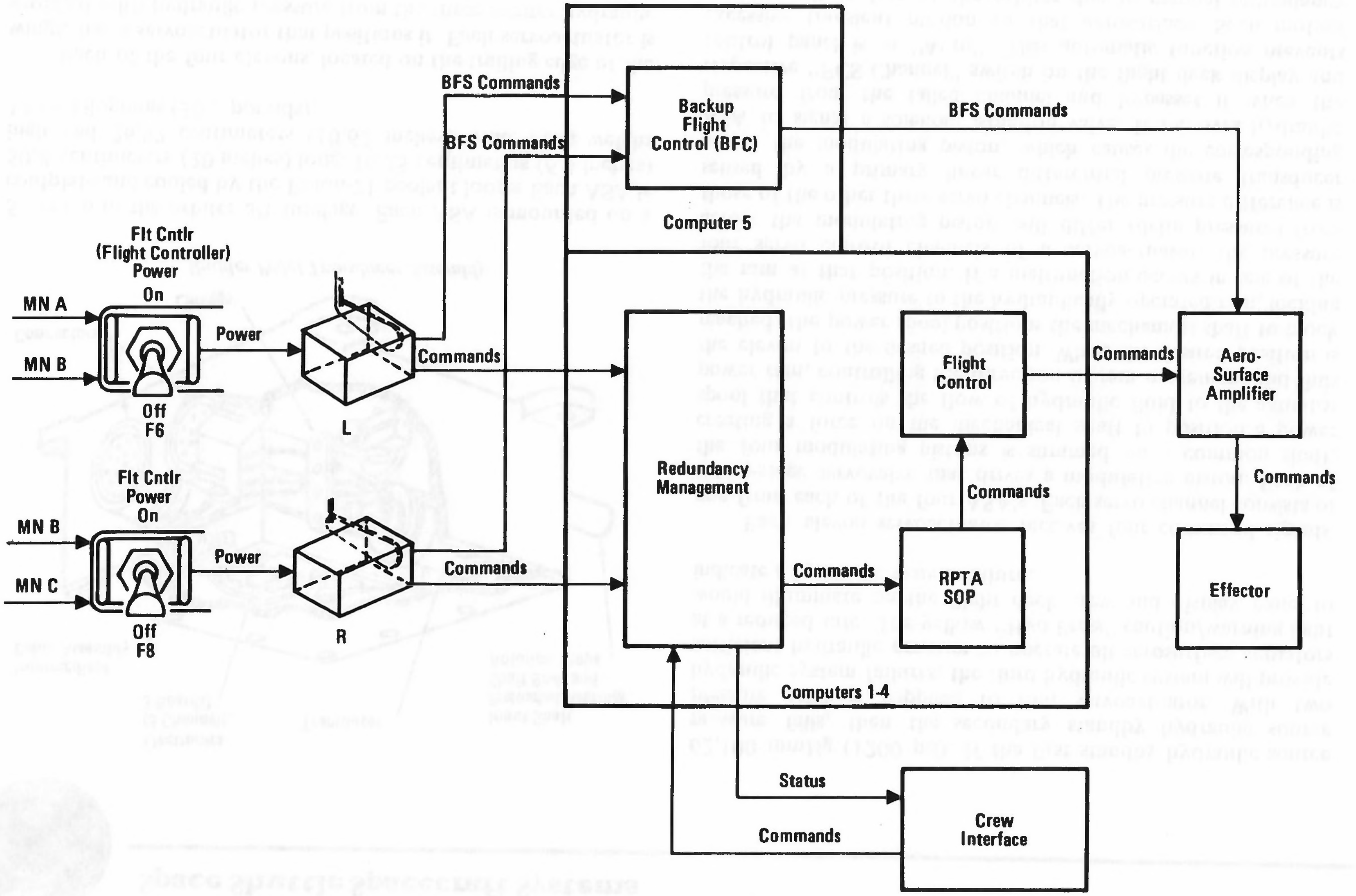
In the CSS mode the commander and pilot's RHC trim switch, in conjunction with the "Trim Enable/Inhibit" switch, activates or inhibits the RHC trim switch. When the RHC trim switch positioned forward or aft adds a trim rate to the RHC pitch command; positioning it left or right adds a roll trim. Redundancy management processes the two contacts in each position and enables or inhibits the use of the RHC trim in the flight control system software.

The "Gain Ena" (enable) pitch and roll/yaw pushbuttons at the commander and pilot's stations are used only in the CSS mode. It is intended for use as a last resort only because of degradation of off-nominal flight conditions such as bending, instabilities, large aerodynamic coefficient errors, and angle-of-attack errors. Depressing the CSS pushbutton restores nominal gain structure in that axis.

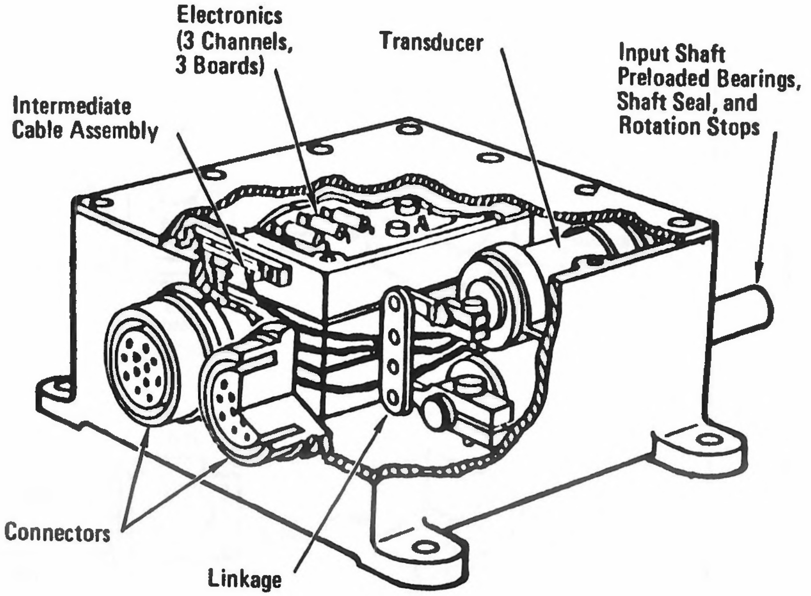
Aerosurface servo amplifier's (ASA's) are electronic devices that receive the aerosurface commands during atmospheric flight from the flight control system software and electrically position hydraulic valves in the aerosurface actuators which cause the deflections. The ASA's receive position feedback from the aerosurface, which is summed with the position command to provide a servo loop closure for one of the four independent servo loops associated with the elevons, rudder, and speed brake. The body flap utilizes three servo loops. The path from an ASA to its servovalve in the actuators and the feedback sensor to an ASA is called a flight control channel; thus there are four flight control channels. The four ASA's are located in avionics bays



Rotation Hand Control



Rudder Pedal Tr ·cer Assembly



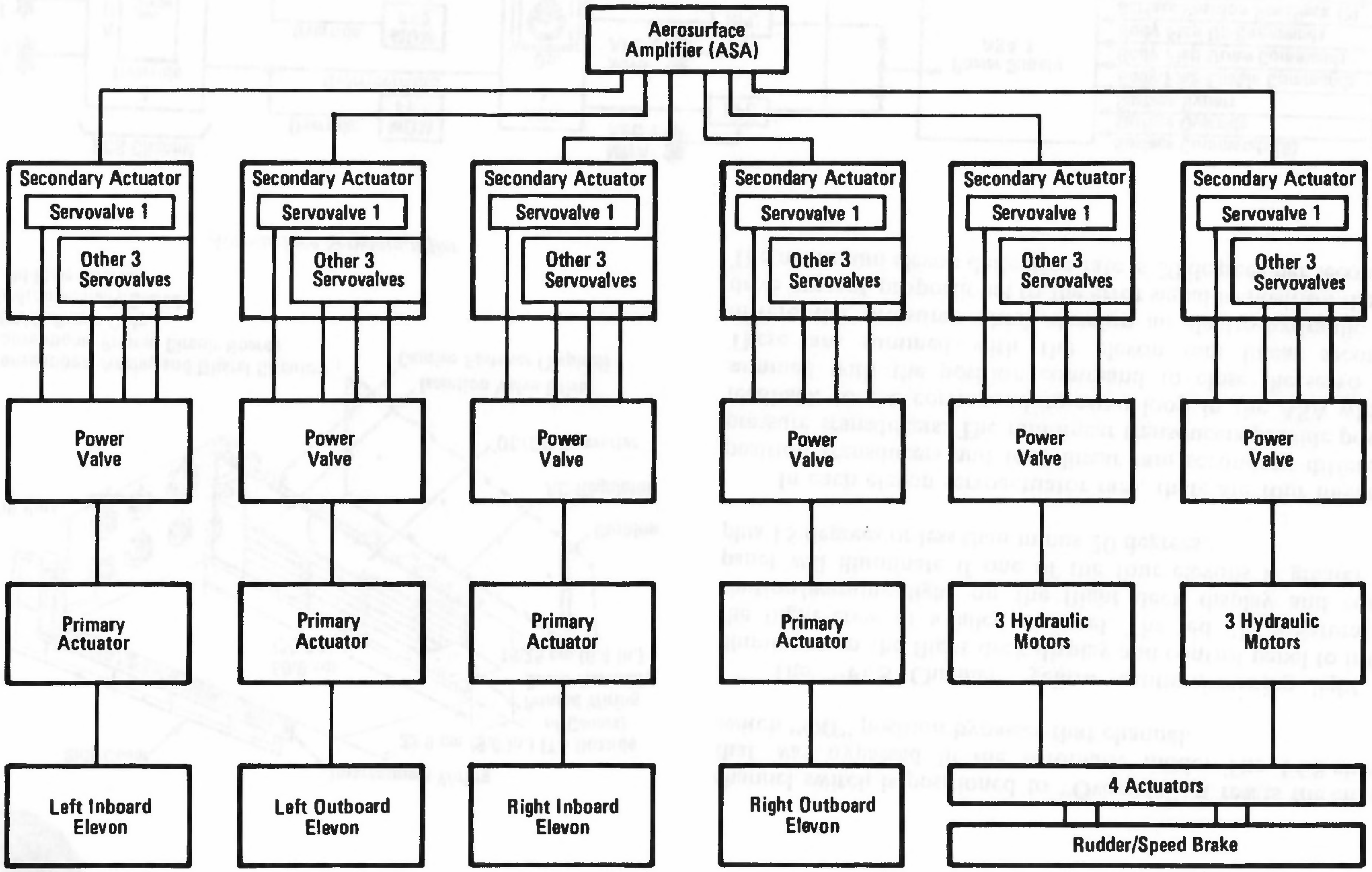
Rudder Pedal Transducer Assembly

5, and 6 in the orbiter aft fuselage. Each ASA is mounted on a coldplate and cooled by the Freon-21 coolant loops. Each ASA is 50.8 centimeters (20 inches) long, 16.25 centimeters (6.4 inches) high and 26.97 centimeters (10.62 inches) wide. Each weighs 13.69 kilograms (30.2 pounds).

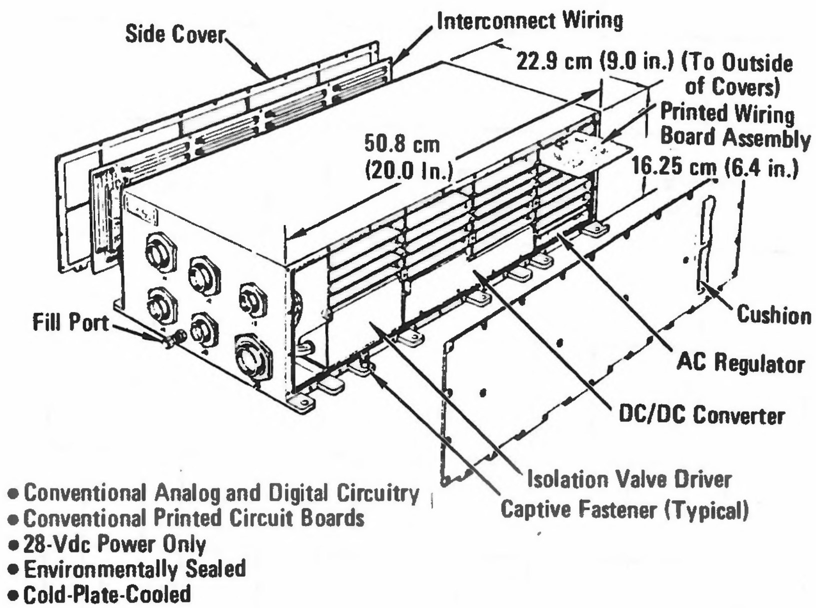
Each of the four elevons, located on the trailing edge of the wings, has a servoactuator that positions it. Each servoactuator is supplied with hydraulic pressure from the three orbiter hydraulic systems. A switching valve controls which hydraulic system becomes the source of hydraulic pressure for that servoactuator. The valve allows a primary source of pressure (P1) to be supplied to that servoactuator. If the primary hydraulic pressure source fails, the switching valve allows the first standby hydraulic pressure (P2) to supply that servoactuator. Failure means a decrease of pressure between 77,625 mmHg (1500 psi) and 62,100 mmHg (1200 psi). If the first standby hydraulic source pressure fails, then the secondary standby hydraulic source pressure (P3) is supplied to that servoactuator. With two hydraulic system failures, the third hydraulic system will provide sufficient hydraulic pressure to operate all aerosurface actuators at a reduced rate. The yellow "Hyd Press" caution/warning light would illuminate on the flight deck crew and display panel to indicate a hydraulic system failure.

Each elevon servoactuator receives four command signals, one from each of the four ASA's. Each servo channel consists of a two-stage servovalve that drives a modulating piston. Each of the four modulating pistons is summed on a common shaft, creating a force on the mechanical shaft to position a power spool that controls the flow of hydraulic fluid to the actuator power ram, controlling the direction of ram movement and thus the elevon to the desired position. When the desired position is reached, the power spool positions the mechanical shaft to block the hydraulic pressure to the hydraulically operated ram, locking the ram at that position. If a malfunction occurs in one of the four servo control channels of a servoactuator, the pressure across the modulating piston will differ (delta pressure) from those of the other three servo channels. The pressure difference is sensed by a primary linear differential pressure transducer across the modulating piston, which causes the corresponding ASA to signal a solenoid isolation valve. It removes hydraulic pressure from the failed channel and bypasses it when the respective "FCS Channel" switch on the flight deck display and control panel is in "Auto". This automatic function prevents excessive transient motion to that aerosurface. Such motion could result in loss of the orbiter due to manual redundancy being too slow.

The four "FCS Channel" swtiches control the ASA channels. The switch for each channel controls that channel for the elevons, rudder/speed brake, and body flap, except for channel 4, which has no body flap commands. When an FCS



Aerosurface Amplifier

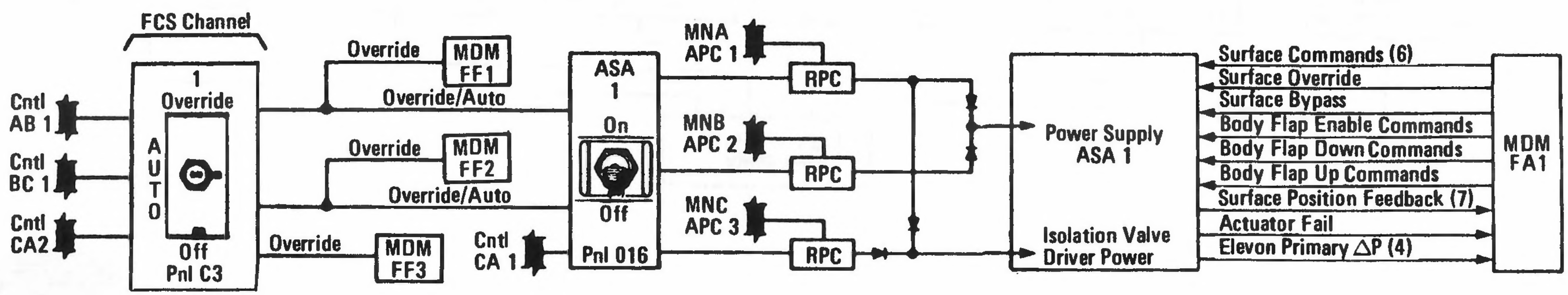


Aerosurface Servoamplifier

channel switch is positioned to "Override", it resets the channel that was bypassed in the automatic mode. The FCS channel switch "Off" position bypasses that channel.

The "FCS Channel" yellow caution/warning light will illuminate on the flight deck display and control panel to inform the flight crew of a failed channel. The red "FCS Saturation" caution/warning light on the flight deck display and control panel will illuminate if one of the four elevons is greater than plus 15 degrees or less than minus 20 degrees.

In each elevon servoactuator ram, there are four linear ram position transducers and four linear ram secondary differential pressure transducers. The ram linear transducers provide position feedback to the corresponding servo loop in the ASA which is summed with the position command to close the servo loop. These are summed with the elevon ram linear secondary differential pressures which develop an electro-hydraulic valve drive current proportional to the error signal to position the ram. The maximum elevon deflection rate is 20 degrees per second.



Typical for four ASA's

Aerosurface Amplifier Control

The rudder/speed brake is located on the trailing edge of the orbiter's vertical stabilizer. The rudder/speed consists of upper and lower segments. One servoactuator positions both segments together as a rudder. Another servoactuator opens the segments at the flared end of the rudder to function as a speed brake.

The rudder and speed brake servoactuator receives four command signals from the four ASA's. Each of the four servo channels consists of a two-stage servovalve which functions similar to that of the elevons. The exception is that the power spool for the rudder controls the flow of hydraulic fluid to three rudder reversible hydraulic motors, and the power spool for the speed brake controls the flow of hydraulic fluid in the three speed brake hydraulic reversible motors. Each rudder hydraulic motor receives hydraulic pressure from only one of the orbiter's hydraulic systems and each speed brake hydraulic motor receives hydraulic pressure from only one of the orbiter's hydraulic systems. Each hydraulic motor has a hydraulic brake. When the motor is supplied with hydraulic pressure, that motor's brake is released. When the hydraulic pressure is blocked to that hydraulic motor, the hydraulic brake is applied holding that motor and the corresponding aerosurface at that position.

The three hydraulic motors provide the output to the rudder differential gearbox. The gearbox is connected to a mixer gearbox, which drives rotary shafts, which drive four rotary actuators, which position the rudder segments for rudder position.

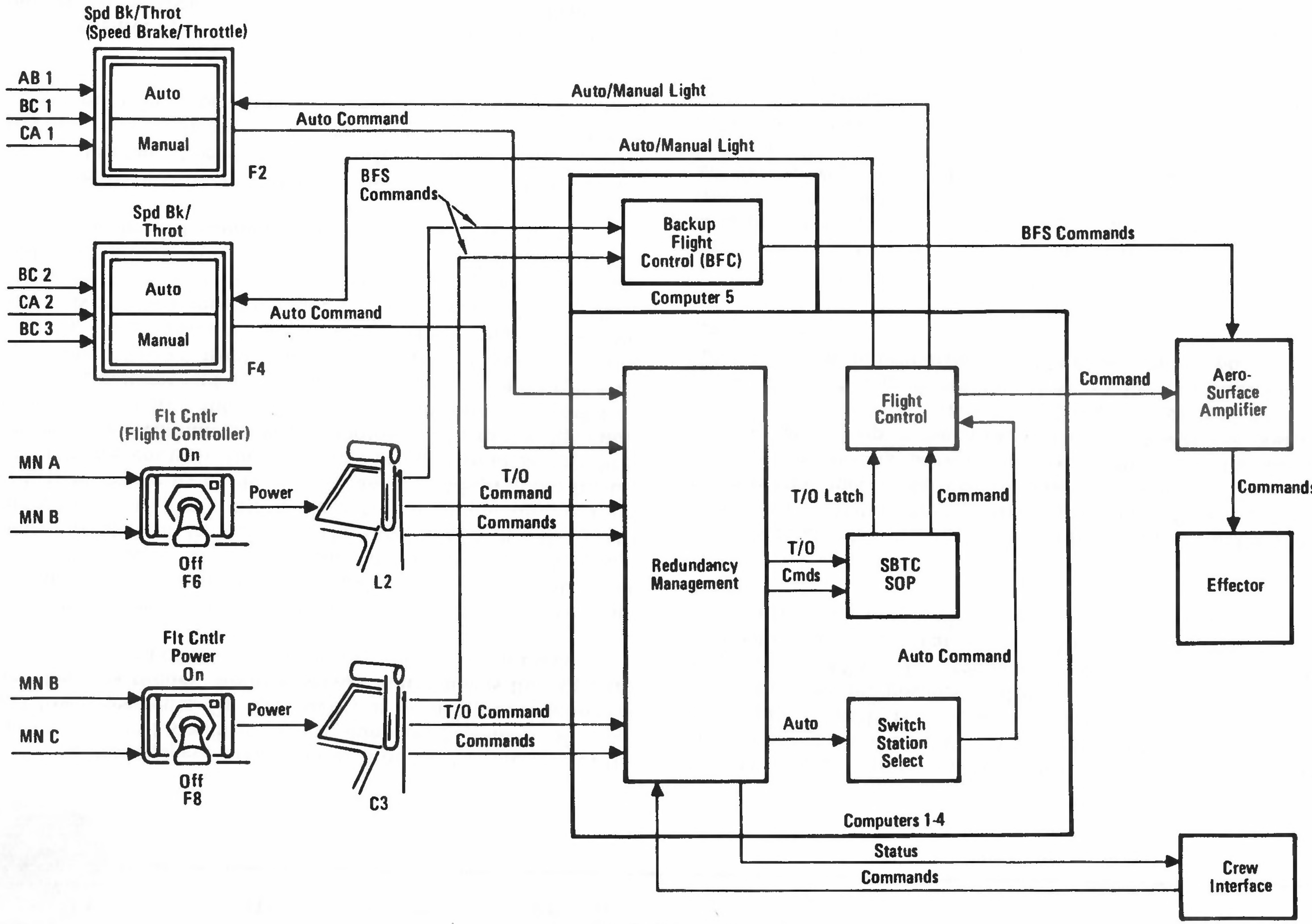
The three speed brake hydraulic motors provide the power output to the speed brake differential gearbox. The differential gearbox is connected to the same mixer gearbox as that of the rudder, which drives rotary shafts, which drive the same four rotary actuators involved with the rudder. Within each of the four rotary actuators, are planetary gears which provide the blend of positioning the rudder and opening the rudder flared e

There are four rotary position transducers on the rudder differential gearbox output and one differential linear position transducer in each rudder servoactuator. The rotary position transducers provide position feedback to the corresponding servo loop in the ASA. The feedback is summed with the linear differential pressures that develop the electro-hydraulic valve drive current proportional to the error signal to position the rudder.

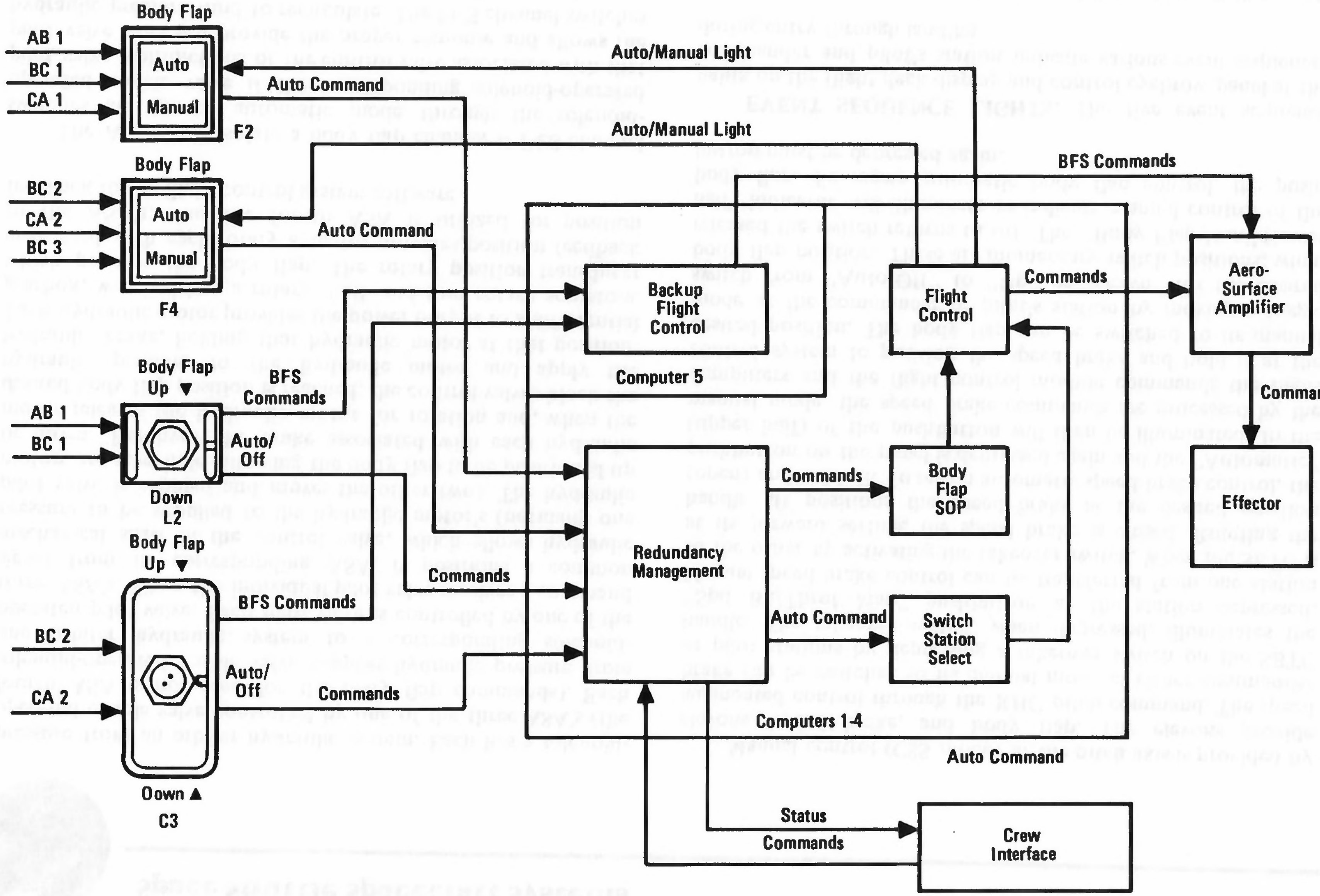
There are also four rotary position transducers on the speed brake differential gearbox output and one differential linear pressure transducer in each speed brake servoactuator. The rotary position transducers provide position feedback to the corresponding servo loop in the ASA. It is summed with the position command to close the servo loop. These are summed with the linear differential pressures that develop the electro-hydraulic valve drive current proportional to the error signal to position the speed brake.

If a malfunction occurs in one of the four rudder servoactuator channels or one of the four speed brake servoactuator channels, the corresponding linear differential pressure transducer will cause the corresponding ASA to signal a solenoid isolation valve, which removes pressure from the failed channel and bypasses it, if that FCS channel switch is in "Auto." The FCS channel switch override and off position functions the same as for the elevons and the FCS channel caution/warning light. The "Hyd Press" caution/warning light will indicate a hydraulic failure. The rudder deflection rate is a maximum of 14 degrees per second. The speed brake deflection rate is approximately 10 degrees per second. If two of the three hydraulic motors fail in the rudder or speed brake, the corresponding deflection rate is reduced approximately 50 percent.

Three servoactuators at the lower aft end of the fuselage are used to position the body flap. Each is supplied with hydra



Speed Probe/Thrust Control


Body Flap
AB 1
BC 1
CA 1
Auto
Manual
F2
Auto/Manual Light
Auto Command
Auto/Manual Light
Body Flap
BC 2
CA 2
BC 3
Auto
Manual
F4
Auto Command
BFS Commands
Backup Flight Control
Computer 5
Flight Control
Commands
Aero-Surface Amplifier
Command
Effector
Body Flap Up
Down
L2
BFS Commands
AB 1
BC 1
Auto/Off
Body Flap SOP
Commands
Body Flap Up
Down
C3
BC 2
CA 2
Auto/Off
BFS Commands
Commands
Commands
Redundancy Management
Auto Command
Switch Station Select
Computers 1-4
Auto Command
Status
Commands
Crew Interface

Bod n

pressure from an orbiter hydraulic system. Each has a solenoid-operated enable valve controlled by one of the three ASA's (the fourth ASA is not used for the body flap commands). Each solenoid-operated enable valve supplies hydraulic pressure from one orbiter hydraulic system to a corresponding solenoid-operated pilot valve. Each pilot valve is controlled by one of the three ASA's. When the individual pilot valve receives a command signal from its corresponding ASA, it positions a common mechanical shaft in the control valve, which allows hydraulic pressure to be supplied to the hydraulic motor's (normally one pilot valve is enabled and moves the other two). The hydraulic motors are reversible, allowing the body flap to be positioned up or down. The hydraulic brake associated with each hydraulic motor releases the hydraulic motor for rotation and, when the desired body flap position is reached, the control valves block the hydraulic pressure to the hydraulic motor and apply the hydraulic brake, holding that hydraulic motor at that position. Each hydraulic motor provides the power output to a differential gearbox, which drives a rotary shaft, and four rotary actuators, which position the body flap. The rotary position transducer associated with each rotary actuator provides position feedback to the ASA's; thus the fourth ASA is utilized for position feedback to the flight control system software.

The ASA's will isolate a body flap channel if FCS channel switches are in the automatic mode through the solenoid-operated enable valve if the corresponding solenoid-operated pilot valve malfunctions or the control valve associated with that pilot valve does not provide the proper response and allows the hydraulic pressure fluid to recirculate. The FCS channel switches and FCS channel caution/warning light would function the same as for the elevons. If the hydraulic system associated with the hydraulic motor fails, the remaining two hydraulic motors will position the body flap and the "Hyd Press" caution/warning light will illuminate. The body flap deflection rate is approximately 4.5 degrees per second.

Manual control (CSS mode) in the pitch axis is provided by elevons, speed brake, and body flap. The elevons provide augmented control through the RHC pitch command. The speed brake can be switched to its manual mode at either commander or pilot stations by depressing a takeover switch on the SBTC handle. The takeover switch, when depressed, illuminates the "Spd Bk/Throt Man" pushbutton at the station depressed. Manual speed brake control can be transferred from one station to the other by activating the takeover switch. When the SBTC is at its forward setting, the speed brake is closed. Rotating the handle aft positions the speed brake at the desired position (open) and holds it. To regain automatic speed brake control, the pushbutton on the panel is depressed again and the "Automatic" (upper half) of the pushbutton will then be illuminated. In the manual mode, the speed brake commands are processed by the computers and the flight control module commands the flight control system to position the speed brake and hold it at the desired position. The body flap can be switched to its manual mode at the commander or pilot's station by moving a toggle switch from "Auto/Off" to "Up" or "Down" for the desired body flap position. These are momentary switch positions; when released the switch returns to off. The "Body Flap Man" (lower half) indicator will illuminate to indicate manual control of the body flap. To regain automatic body flap control, the push-button must be depressed again.

EVENT SEQUENCE LIGHTS. The five event sequence lights on the flight deck display and control eyebrow panel at the commander and pilot's station indicate various event sequences during entry through landing.

The left-most event light is a red light that will illuminate when the spacecraft normal acceleration reaches 2 g's and will flash if the normal acceleration is 2.5 g's and above.

The second-most event light is an amber light. It will illuminate if the speed brake is out of the required position by 10 percent or more between Mach 10 and 0.9, if the air data probes are not deployed between Mach 2.5 and 1, or the speed brake or body flap are out of position plus or minus 0.5 degree at or below 152 meters (500 feet).

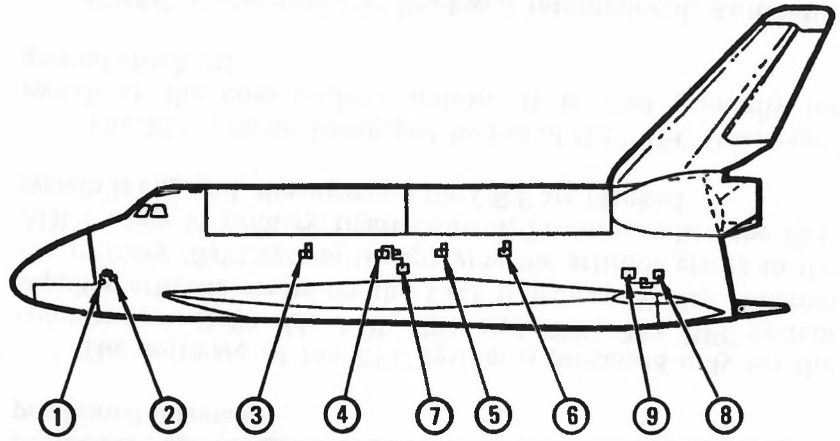
The third light from the left is not implemented on the initial development flight, but is on the subsequent flights. It is a green light and will illuminate when the spacecraft altitude is below the upper flare altitude limit of 838 meters (2750 feet) and the lower flare altitude of 533 meters (1750 feet). At the lower flare altitude, the light flashes for three seconds and then is turned off.

The fourth light from the left is not implemented on the initial development flight but is on the subsequent flights. It is a red light and flashes when the spacecraft reaches gear-down altitude of 91 meters (300 feet) or below until all three landing gear uplock signals are false (gear down and locked). It illuminates for ten seconds after it flashes.

The right-most light is a green light and will flash for five seconds and remain on from any of the following events: the spacecraft is below 42,672 meters (140,000 feet) altitude and TACAN range, TACAN azimuth, or drag altitude data are lost; the spacecraft is below 25,908 meters (85,000 feet) down to the altitude at which Mach 0.75 [9753 meters (32,000 feet)] is reached and TACAN range or azimuth are lost; the spacecraft is below Mach 0.75 and TACAN range or azimuth, MLS range or azimuth (if acquired), barometric altitude, and radar altimeter below 152 meters (500 feet) altitude are invalid.

ORBITER VENT AND PURGE SYSTEM. The vent and purge system is controlled exclusively through software. There are no dedicated manual controls or displays on the crew compartment flight deck display and control panel.

The orbiter's vent and purge system comprises 18 active ports and is divided into six groups: left and right ports one and two, left and right port three, left and right port five, left and right ports four and seven, left and right port six, and left and right ports eight and nine.



Vent No.*	Compartment Vented	Vent Door Subsystem
1	Forward RCS	Forward
2	Forward Fuselage	
7	Wing	Payload Bay and Wing
4	Mid Fuselage	
5	Mid Fuselage	Payload Bay
3	Mid Fuselage	
6	Mid Fuselage	
8	OMS Pod	Aft
9	Aft Fuselage	

*LH and RH

Orbiter Vent Doors

All vent ports have a purge position with the exception of left and right vent ports three, four, five, and seven.

The vent and purge system is used for equalizing the pressure across the outer surface of the orbiter and to permit molecular venting of orbiter cavities and insulation blankets to achieve the required low internal blanket pressure. The purge position must maintain a positive pressure in the orbiter's payload bay area to prevent contamination and to vent any residue in the payload bay area while in the ground turnaround phase.

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The sequencing of the active ports is by the software program in the onboard redundant set computers. The ports are cycled to open, close, or purge position as required in each mission phase. Positioning of the active ports is performed by the software based on mission time or mission events during ascent, entry, aborts, and by keyboard entry on the crew compartment flight deck display and control panel in orbit. Vent port status is displayed on the flight deck display and control panel CRT's in orbit.

When a cue is received from the computer launch sequence to configure the vent ports to a launch configuration, vent ports one and two and eight and nine are commanded to the open position, and all other vent ports are closed. The status of the vent ports is transmitted to the computer launch sequence to determine that the vent ports have achieved the launch configuration within the specified time. The orbiter is launched with the vent ports in this configuration, and at T plus 10 seconds all vent doors are commanded open. In a nominal mission, the vent ports will remain open until the flight crew closes the ports with a keyboard entry on the flight deck display and control panel prior to deorbit. During entry, at approximately 24,384 meters (80,000 feet) altitude, the computers automatically open the fuselage vents. The vents should be fully open at 21,336 meters (70,000 feet) altitude and remain open until weight on the nose gear at landing, at which time they will go to purge.

If, during the launch phase–T minus ten to T minus zero–a launch abort has occurred, the vent door system is reconfigured to the prelaunch configuration by the launch processing system.

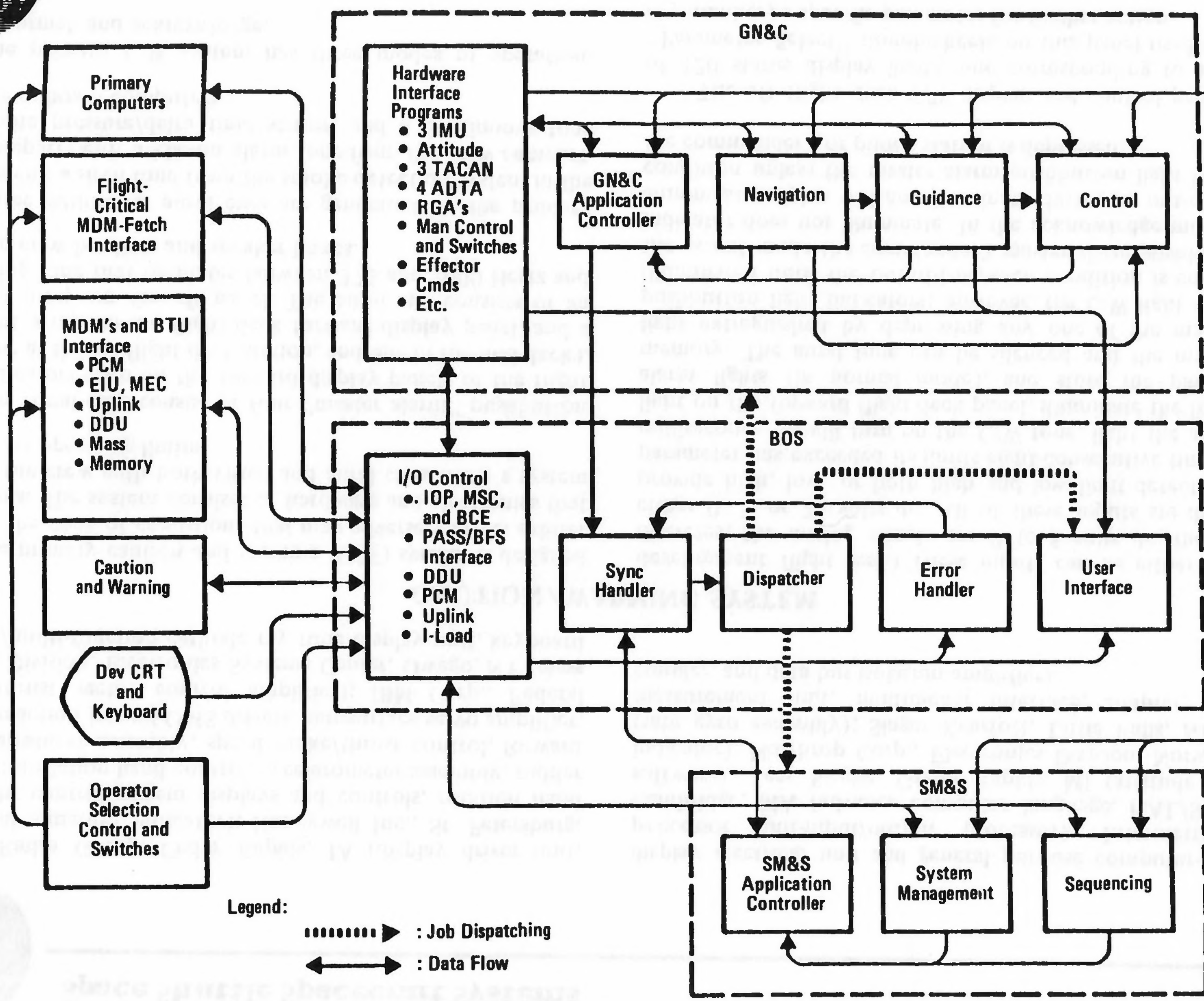
BACKUP FLIGHT CONTROL SYSTEM. The backup flight control (BFC) system is engaged by depressing the BFC switch located near the top of the commander or pilot's RHC or by depressing the BFC pushbutton on the commander or pilot's flight deck display and control eyebrow panel. The fifth computer is allocated as the backup flight control system in the initial development flights. It has a separate independent software design and coding activity to protect against generic software failures in the primary flight control system computer set. It also provides unique functions such as non-critical monitoring and payload command and monitoring.

The BFC system is distinct from the primary flight system. The BFC system is used only in an extreme emergency. It utilizes control laws (equations) similar to the primary flight control system; however, it is a non-redundant mechanization. The BFC is constrained in a single-string flight control system to avoid generic problems that may exist with the primary computers or primary system software. It operates concurrently with the primary flight system, processing the same commands and sensor data; however, its output is inhibited. When engaged, the BFC system augments the CSS mode by using the commander's RHC. Thus there is no manual thrust vector control or manual throttling capability during first- and second-stage ascent and, during atmospheric flight, the speedbrake would be positioned by using the SBTC and the body flap would be positioned manually.

The software of the BFC system is processed only for the commander's (left) side ADI, HSI, and RHC. The BFC system supplies attitude errors on the CRT trajectory display, whereas the primary flight system is supplying the attitude errors to the ADI's when in primary flight control; however, when the BFC system is engaged, the errors on the CRT are blanked.

The BFC can be disengaged by use of the "BFC Disengage" switch at the commander's station. It is used primarily for ground checkout.

GN&C contractors are: Rockwell International, Autonetics Group, Anaheim, CA (driver module controller, master event sequence controller, backup flight control system); Ball Brothers Research, Boulder, CO, (star tracker); Rockwell International,


GN&C
Primary Computers
Flight-Critical MDM-Fetch Interface
MDM's and BTU Interface
• PCM
• EIU, MEC
• Uplink
• DDU
• Mass
• Memory
Caution and Warning
Dev CRT and Keyboard
Operator Selection Control and Switches
Hardware Interface Programs
• 3 IMU
• Attitude
• 3 TACAN
• 4 ADTA
• RGA's
• Man Control and Switches
• Effector Cmds
• Etc.
GN&C Application Controller
Navigation
Guidance
Control
BOS
I/O Control
• IOP, MSC, and BCE
• PASS/BFS Interface
• DDU
• PCM
• Uplink
• I-Load
Sync Handler
Dispatcher
Error Handler
User Interface
SM&S
SM&S Application Controller
System Management
Sequencing
Legend:
: Job Dispatching
: Data Flow

BFS Flow D[illegible]am

Collins Radio Group, Cedar Rapids, IA (display driver unit, horizontal situation indicator); Honeywell Inc., St. Petersburg, FL (flight control system displays and controls, rotation hand control, translation hand control, accelerometer assembly, rudder pedal transducer assembly, speed brake/thrust control, forward and aft reaction jet and OMS drivers, aerosurface servo amplifier, ascent thrust vector control amplifier); IBM Corp., Federal Systems Division, Electronics Systems Center, Owego, NY (mass memory/multi-function cathode ray tube display unit, keyboard display electrical unit and general purpose computer-computer processor unit-input/output processor); Intermetrics Inc., Cambridge, MA (advance computer language, HAL/S, avionics software); Lear Siegler, Grand Rapids, MI (attitude direction indicator); Northrop Corp., Electronics Division, Norwood, MA (rate gyro assembly); Singer Kearfott, Little Falls, NJ (inertial measurement unit, multiplexer interface, adapter, data bus coupler, and data bus isolation amplifier).

CAUTION / WARNING SYSTEM

The primary caution and warning (C/W) system is designed to warn the crew of conditions that may adversely affect orbiter operations. The system consists of hardware and electronics that provide the crew with both visual and aural cues when a system exceeds set operating limits.

The visual cues consist of four "master alarm" pushbutton light indicators (two on the forward display panels of the flight deck, one at the aft flight deck station, and one in the mid-deck), a 40-light array on the flight deck forward display panel, and a 120-light array on the aft panel. The aural cue consists of an alternating tone that oscillates between 375 and 1000 Hertz and is sent to crew headsets and speaker boxes.

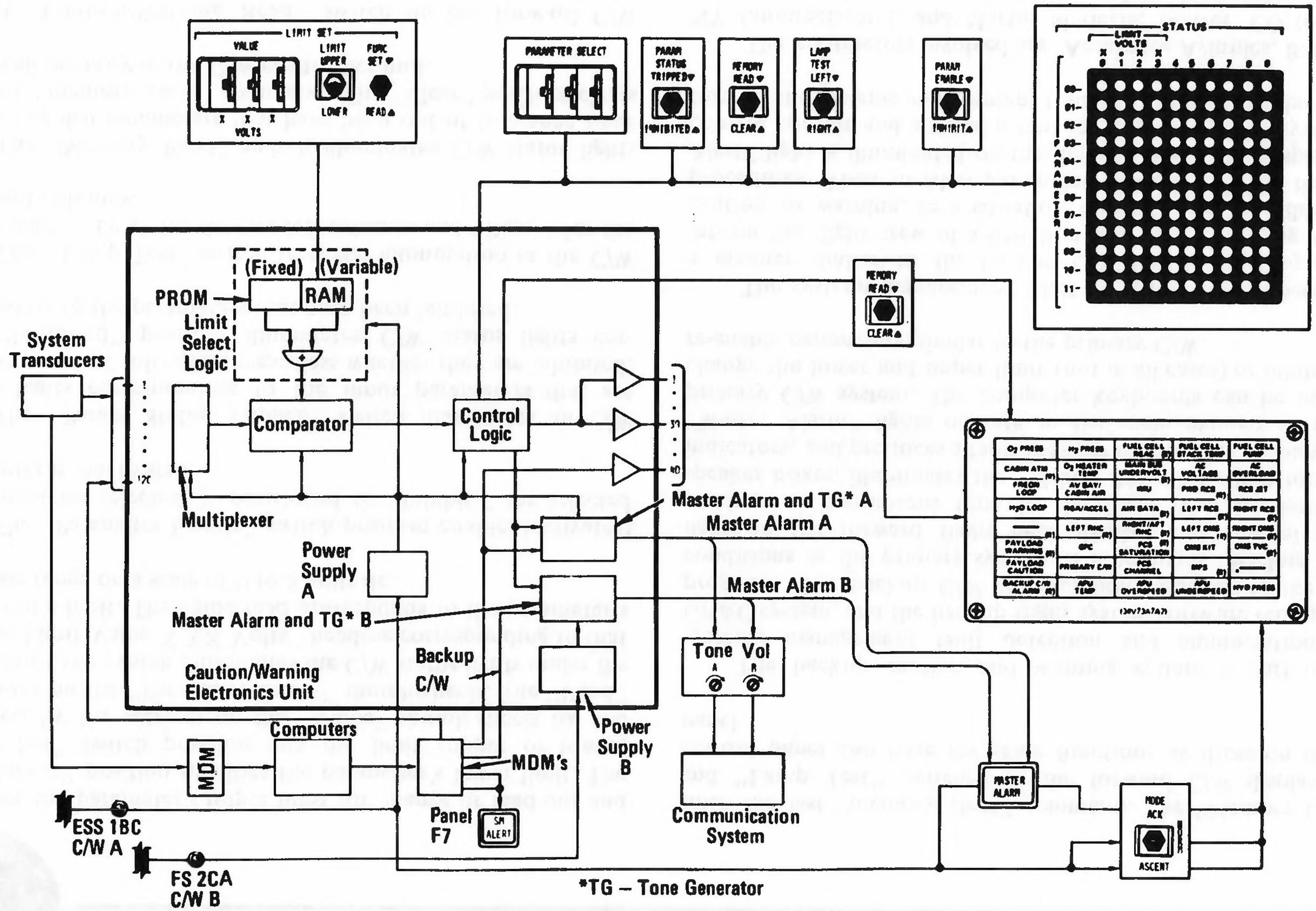
Three additional aural cues are generated by the primary C/W system: a siren tone from the smoke detection system in the crew compartment, a klaxon alarm tone from the crew compartment delta pressure/delta time sensor, and a continuous tone from the onboard computers.

The primary C/W system has three modes of operation: ascent, normal, and acknowledge.

The system receives 120 inputs directly from transducers, through signal conditioners, or from the flight forward multiplexer/demultiplexers. These inputs are fed into a multiplexing system. (Some of these 120 inputs will not be used on the initial development flight test.) These inputs can be either analog or discretes; the analog signals are 0 to 5 volts dc; the discretes either 0, 5, or 28 volts dc. All of these inputs are designed to provide high, low, or both high and low limit detection. If the parameter has exceeded its limits eight consecutive times for 100 milliseconds, it will turn on the C/W tone, light the appropriate light on the forward flight deck panel, illuminate the four master alarm lights (in normal mode), and store the parameter in memory. The aural tone can be silenced and the master alarm light extinguished by depressing any one of the master alarm pushbutton light indicators; however, the C/W light will remain illuminated until the out-of-tolerance condition is corrected. In the ascent mode the commander's master alarm pushbutton light indicator does not illuminate. In the acknowledge mode, the 40 annunciator lights will not illuminate during an out-of-tolerance condition unless the master alarm pushbutton light indicator at the commander's or pilot's station is depressed.

The aft flight deck C/W display and control panel consists of 120 status display lights, one corresponding to each input. "Parameter Select" thumbwheels on this panel used to identify (by number) a specific parameter for further action.

The "Limit Set" switch grouping is used to change limits or to read out a parameter's limits. The "Value" thumbwheels are to select the lower or upper limits. The "Limit Upper" position



Caution and Warning System Hardware

specifies the parameter's upper limit for change or read-out and the "Lower" position specifies the parameter's lower limit. The "Func Set" switch position sets the limit (upper or lower) specified by the settings on the "Value" thumbwheels for the parameter on the "Parameter Select" thumbwheels. The "Read" position of the switch illuminates the C/W status lights under the "Status Limit Value X.XX Volts" heading corresponding to that parameter's limit. The value read corresponds to the parameter's full-scale range on a scale of 0 to 5 volts dc.

The "Parameter Enable" switch position enables (activates) the parameter selected; if positioned to "Inhibit," the selected parameter is inactivated.

The "Param Status Tripped" switch illuminates all C/W status lights corresponding to the input parameters that are currently out of tolerance, regardless whether they are inhibited. The "Inhibited" position illuminates C/W status lights corresponding to the parameters that have been inhibited.

The "Lamp Test" switch tests the illumination in the C/W status lights; "Left" for the five left columns and "Right" for the five right columns.

The "Memory Read" switch illuminates C/W status lights for all enabled parameters that have been out of tolerance since the last "memory clear" command. The "Clear" position clears the recall memory in the C/W electronics unit.

A "Caution/Warning Read" switch on the forward C/W display and control panel illuminates lights on the forward C/W panel from enabled parameters that have been out of tolerance since the last "memory clear" command. The "Memory Clear" and "Lamp Test" switches on the forward C/W display and control panel also have the same functions as those on the aft panel.

The backup caution and warning system is part of the systems management fault detection and annunciation, the GN&C system, and the backup flight system software (computer) programs. The backup C/W annunciation responds to the same conditions as the primary system. It illuminates "Backup C/W" light on the forward flight deck display and control panel, produces a continuous tone in the flight crew headsets and speaker boxes, illuminates the "Master Alarm" pushbutton light indicators, and produces a fault message on the CRT display. The "Master Alarm" lights operate in the same manner as in the primary C/W system. The computer keyboards can be used to change the lower and upper limit (not in all cases) or inhibit and re-enable parameters, similar to the primary C/W.

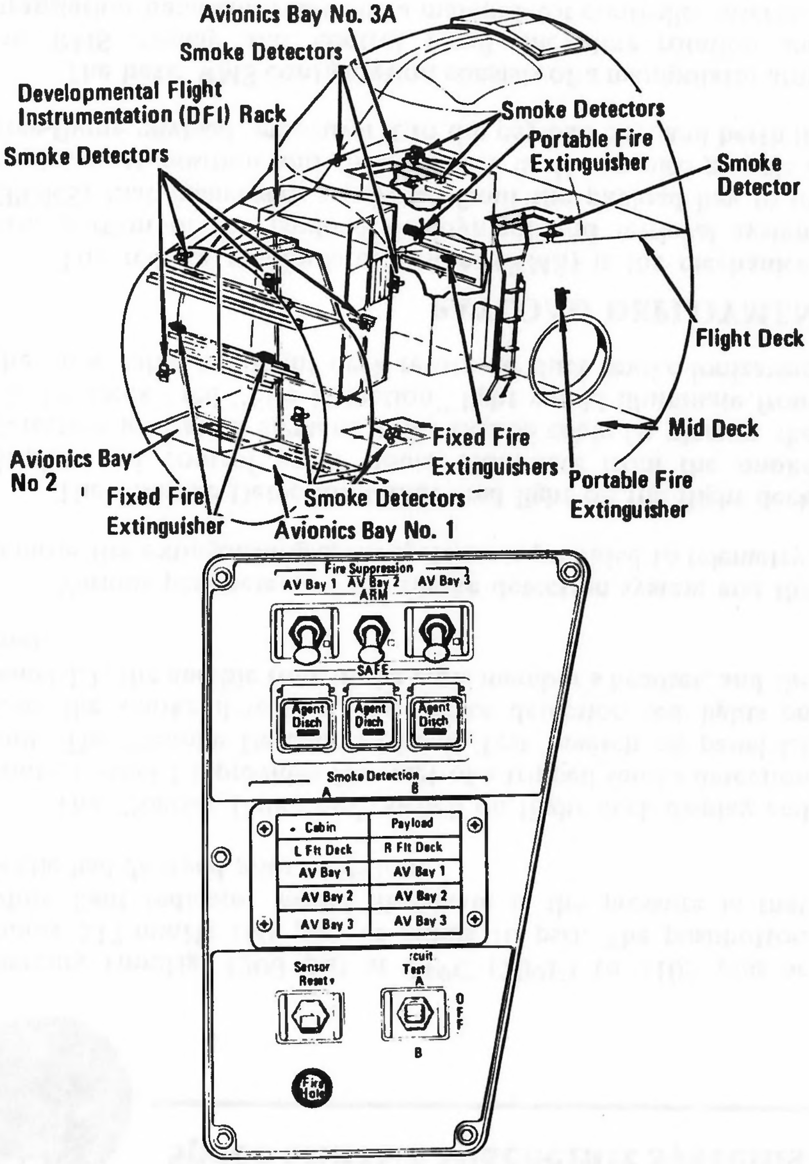
The systems management Alert program, which operates in a manner similar to the backup C/W system, is designed to inform the flight crew of a situation that may be leading up to a caution or warning, or a situation that may require additional procedures. When an Alert parameter exceeds its limits the "SM Alert" light is illuminated on the flight deck forward display and control console and a signal is sent to the primary C/W system to turn on the systems management tone and the CRT displays.

The contractors involved are: Aerospace Avionics, Bohemia, NY (annunciators), and Martin Marietta, Denver, CO (caution and warning electronics and caution and warning status display, limit module).

SMOKE DETECTION AND FIRE SUPPRESSION

Smoke detection and fire suppression capabilities are provided in the crew cabin avionics bays, in the crew cabin and, for the operational flights, the payload bay.

Ionization detection elements, which sense levels of smoke concentrations or rate of concentration change, trigger alarms and provide smoke-concentration-level intelligence to the



Smoke Detection and Fire Suppression System

performance-monitoring system on the crew cabin flight deck display and control panel. There are two ionization detection elements each in crew cabin avionics bays 1, 2, and 3A. There are three ionization detection elements in the crew cabin, one in the ECLSS cabin fan plenum outlet, which is located beneath the crew cabin mid deck floor, and one each in the crew cabin left and right flight deck return air ducts. There are also ionization detectors in Spacelab when it is in the payload bay.

Fire suppression in the crew cabin avionics bays 1, 2, and 3A is by remote-controlled fire extinguishing agents in each bay. For the remaining areas there are portable extinguishers.

The two smoke ionization detector elements (A and B) in crew cabin avionics bays 1, 2, or 3A will trip if the smoke concentration is greater than 2000 plus or minus 200 micrograms per cubic meter (cubic foot) or if the rate of smoke increase is 22 micrograms per cubic meter (cubic foot) for eight consecutive counts in 20 seconds. The trip signal will illuminate the applicable "Av Bay" red "Smoke Detection" A or B light on flight deck display and control panel L1. A siren tone will sound in the crew members' headsets.

Avionics bays 1, 2, and 3A each have a Freon-1301 (bromotrifluoromethane) extinguishing agent bottle. To activate the bottle in the applicable bay, the applicable "Fire Suppression Av Bay" switch on the flight deck display and control panel is positioned to "Arm" on panel L1. The applicable "Fire Suppression Av Bay Agent Disch" (discharge) guarded push-button white light indicator is depressed on the panel. The pushbutton white light indicator activates the corresponding pyro initiator controller (PIC), which drives a pyrotechnic valve to discharge the Freon-1301 fire extinguishing agent from the bottle. The pushbutton is held until the white light in the pushbutton illuminates, which indicates that fire extinguishing agent bottle pressure has decreased from 10,350 millimeters mercury (mmHg) (200 psi) at 21°C (70°F) to 3105 plus or minus 517 mmHg (60 plus or minus 10 psi). The pushbutton white light indicator would illuminate if the pressure in that bottle had decayed prior to its use.

The "Smoke Detection" switch on flight deck display and control panel L1 provides for reset of a tripped smoke detection unit. The "Smoke Detection Circuit Test" switch on panel L1 tests the smoke detectors, the smoke detection red lights on panel L1, the audible tone in the crew member's headset, and the siren.

Various parameters of the smoke detection system and the remote fire extinguishing agent system are provided to telemetry.

The "Smoke Detection Cabin" red light on the flight deck display and control panel would illuminate from the smoke detection ionization element in the ECLSS cabin fan plenum the "L Flt Deck" red "Fire Detection" light would illuminate from the crew cabin left flight deck return air duct smoke ionization element, and the "R Flt Deck" red "Fire Detection" light would illuminate from the right flight deck return air duct.

Portable hand-held fire extinguishers are available in the crew cabin. The extinguishing agent is Halon-1301 (monobromotrafluoromethane). Halon-1301 minimizes the major hazards of a conflagration: smoke, heat, oxygen depletion, and formation of pyrolysis products such as carbon monoxide.

Two of the fire extinguishers are located on the crew module cabin mid deck and two on the flight deck. The fire extinguisher nozzles can fire through fire hole ports in the display and control panels in the event of fire in back of the display and control panels.

The contractors involved with the smoke detection and fire suppression system are Brunswick Celesco, Costa Mesa, CA (smoke detectors and remote control fire extinguishing agent); J.L. Products, Gardena, CA (arming fire pushbutton); Metalcraft Inc., Baltimore, MD (portable fire extinguishers). 378

PAYLOAD DEPLOYMENT AND RETRIEVAL SYSTEM

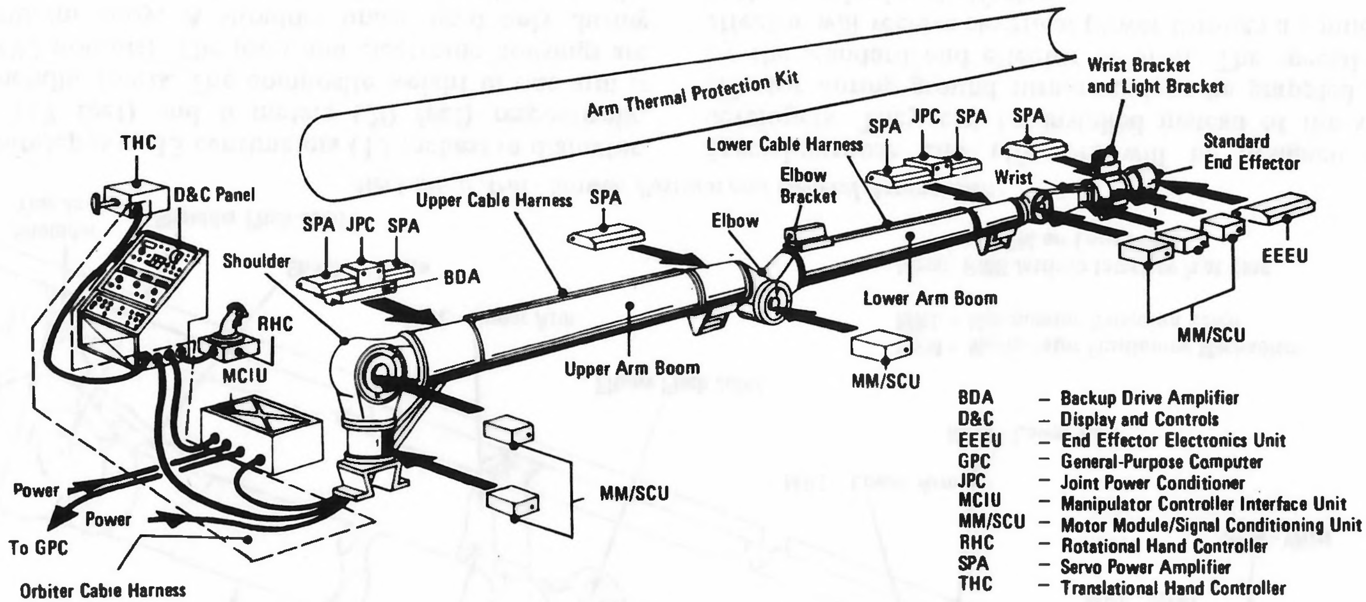
The remote manipulator system (RMS) is the mechanical arm portion of the payload deployment and retrieval system (PDRS) that maneuvers a payload from the payload bay to its deployment position and then releases it. It can also grapple a free-flying payload, maneuver it to the payload bay, and berth it.

The basic RMS configuration consists of a manipulator arm, an RMS display and control panel (including rotation and translation hand controls), and a manipulator controller interface unit which interfaces with the orbiter computer. Most missions will require only one manipulator arm, which normally will be installed on the port (left) side longeron of the orbiter payload bay. It also can be installed on the starboard (right) side if needed. A two-arm installation also can be used.

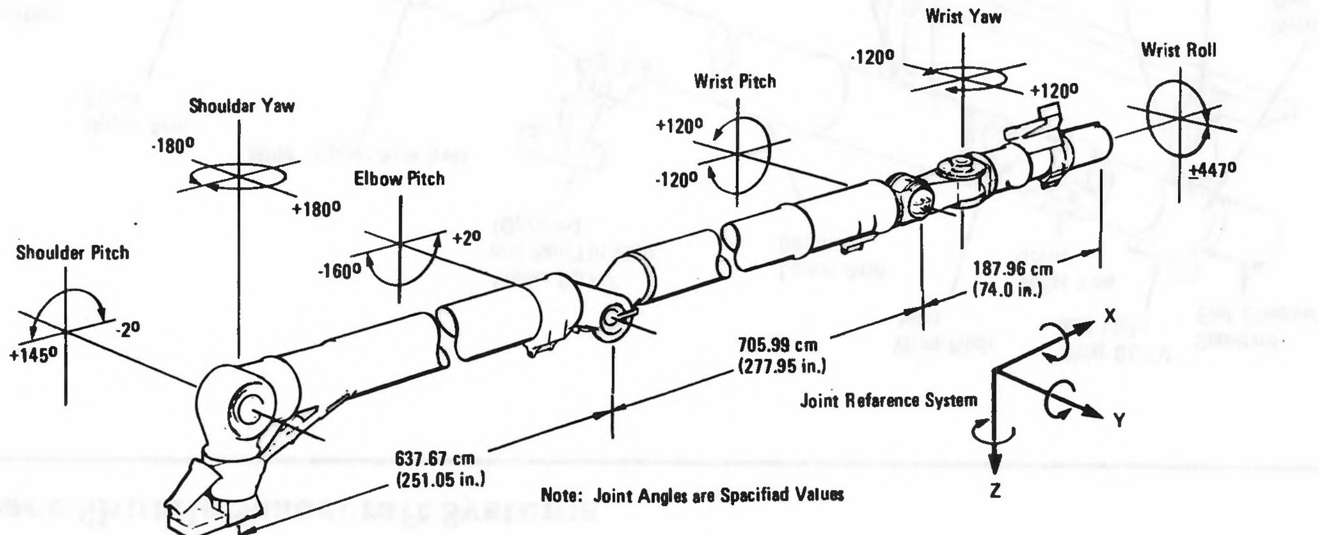
When both manipulator arms are installed, they can be operated only one at a time, since only a single software package (computer programs) and a single set of display and control panel hardware are provided.

The manipulator arm is 15 meters, 7.62 centimeters (50 feet, 3 inches) in length, 38 centimeters (15 inches) in diameter, and has six degrees of freedom. In conjunction with handling aids, it can remove and install a 4.5-meter (15-foot diameter), 18-meter (60-foot) long, 29,484-kilogram (65,000-pound) payload.

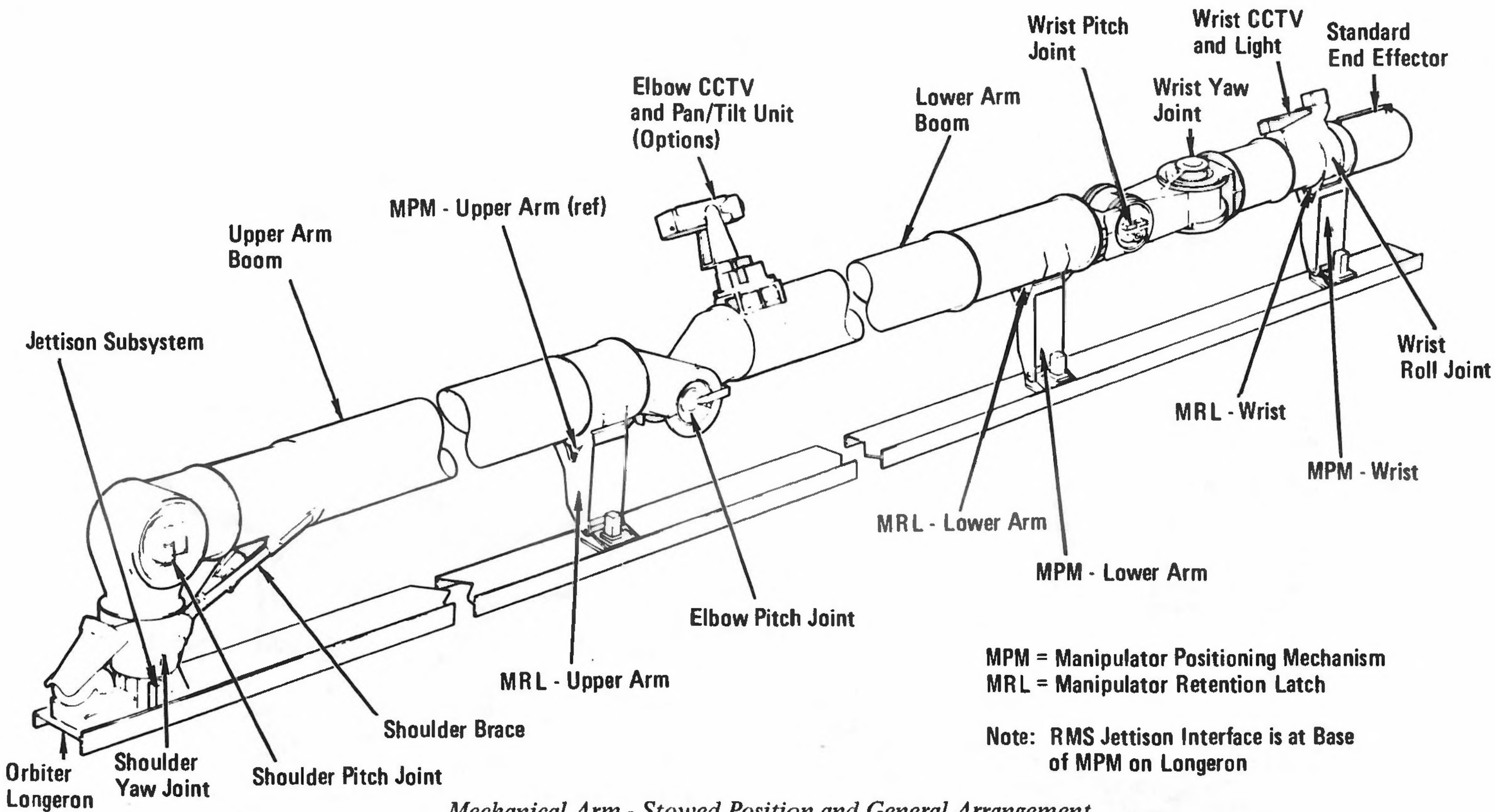
The RMS arm consists of joint housings, electronics housing, arm booms, and shoulder brace. There are two booms: the upper, which connects the shoulder and elbow joints, and the lower, which connects the elbow and wrist joints. The booms are



RMS Components



Mechanical Arm-Stowed Position and Movement Configuration

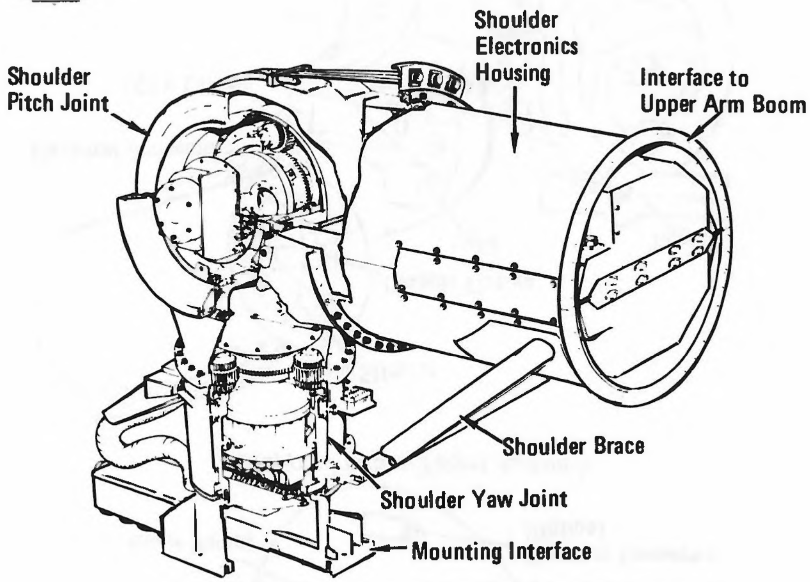


Mechanical Arm - Stowed Position and General Arrangement

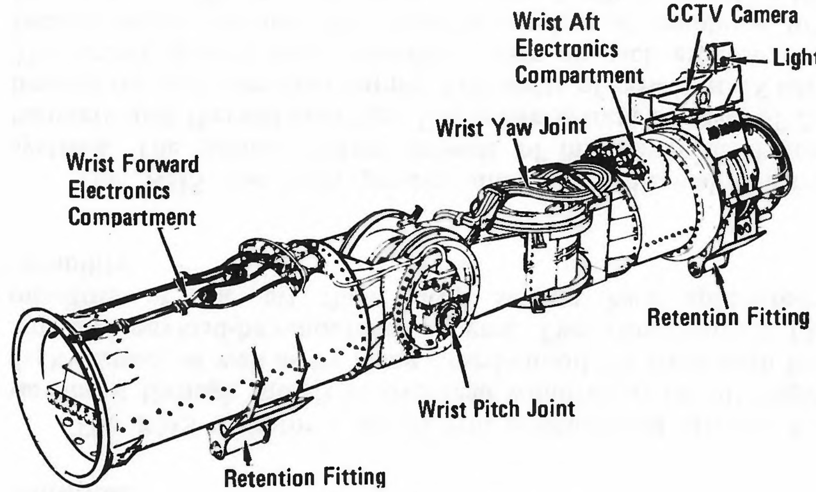
made of graphite/epoxy, 33 centimeters (13 inches) in diameter, by 5 meters (17 feet) and 6 meters (20 feet) respectively, attached by metallic joints. The composite weight in one arm is 42 kilograms (93 pounds). The joint and electronic housings are made of aluminum alloy. A shoulder brace, used only during launch, minimizes high pitch axis moment loading on the shoulder pitch gear train. The shoulder brace is unlatched by a switch located on the aft flight deck display and control panel.

The RMS can operate with standard or special-purpose end effectors. The standard end effector can grapple a payload, keep it rigidly attached as long as required, and then release it. Special-purpose end effectors will be designed by payload developers. They can be installed instead of the standard end effector during ground turnaround or be grappled and released by the standard end effector in orbit. The special-purpose end effector will receive electrical power through a connector located in the standard end effector.

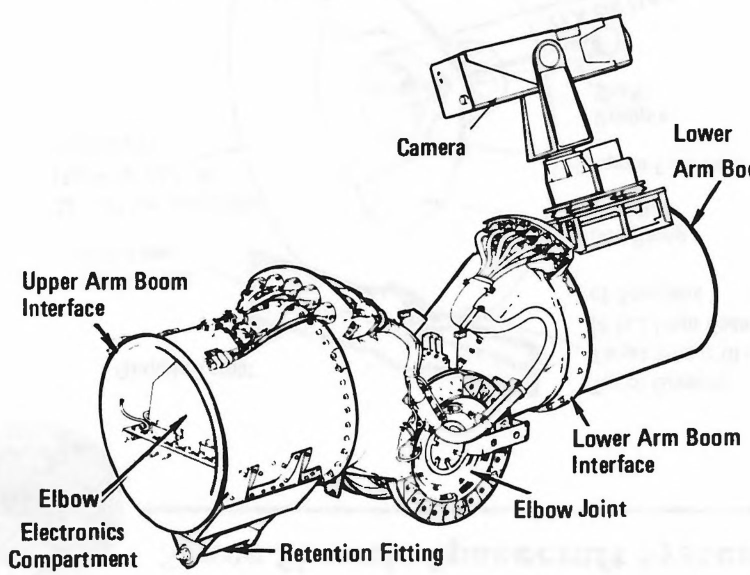
The standard end effector has two functions: capture/release and rigidize/derigidize. Capture/release is accomplished by rotating an inner cage assembly containing three wire snares to open and close around the payload-mounted standard grapple fixture. A switch on the back of the RMS rotation hand control (RHC) commands capture or release. Rigidize/derigidize is



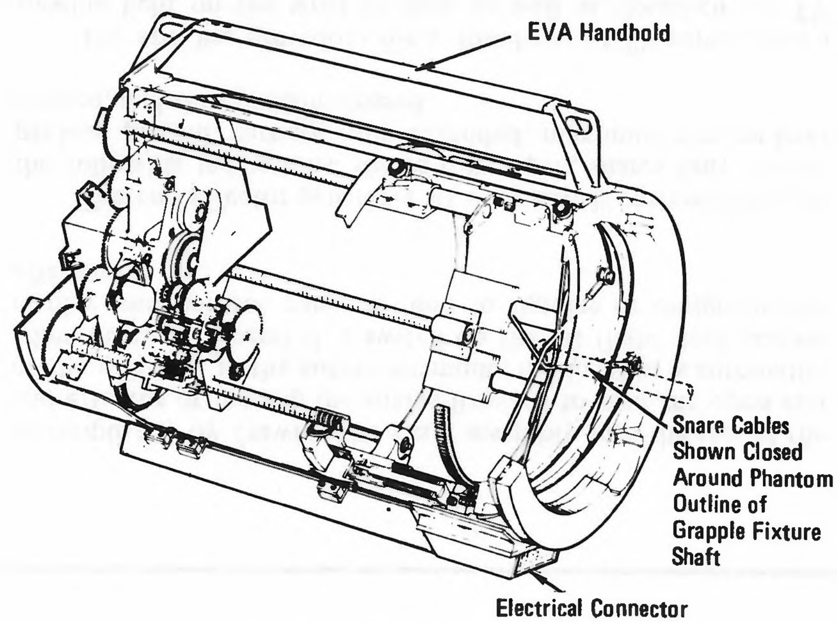
Overall Configuration of the Shoulder Joint



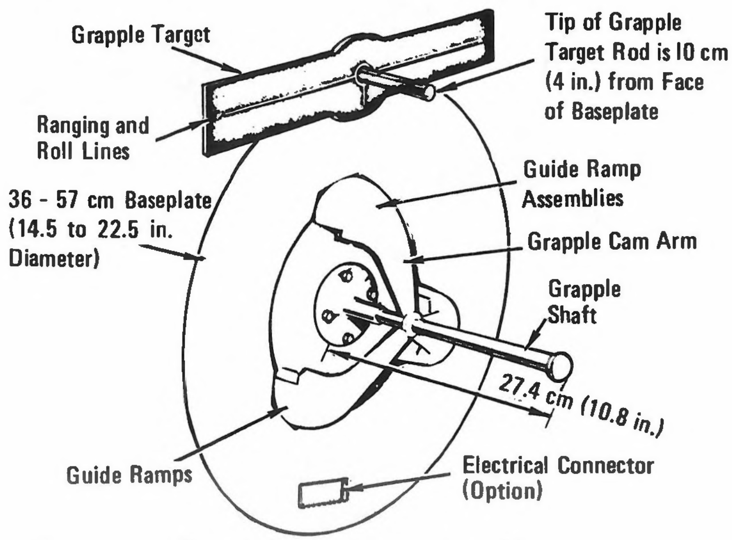
Overall Configuration of Wrist Joint



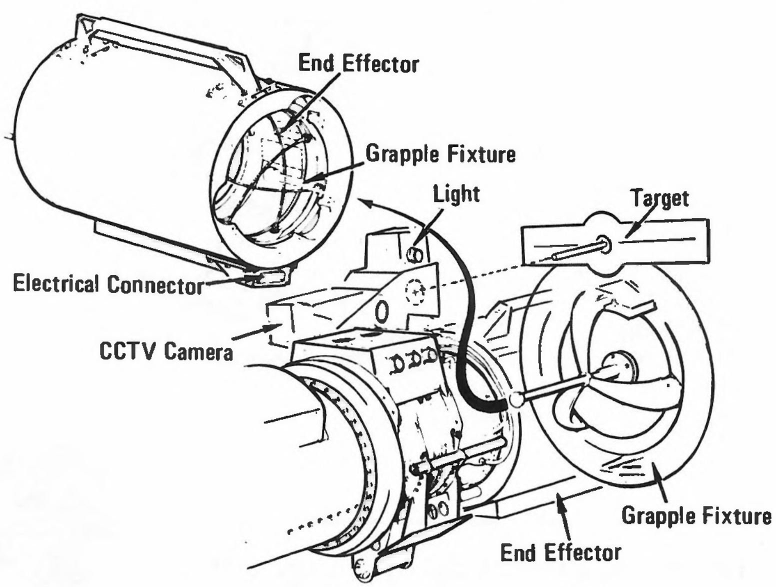
Overall Configuration of the Elbow Joint



Standard Snare Type End Effector



Grapple Fixture/Target Assembly



End Effector/Grapple Fixture Interface

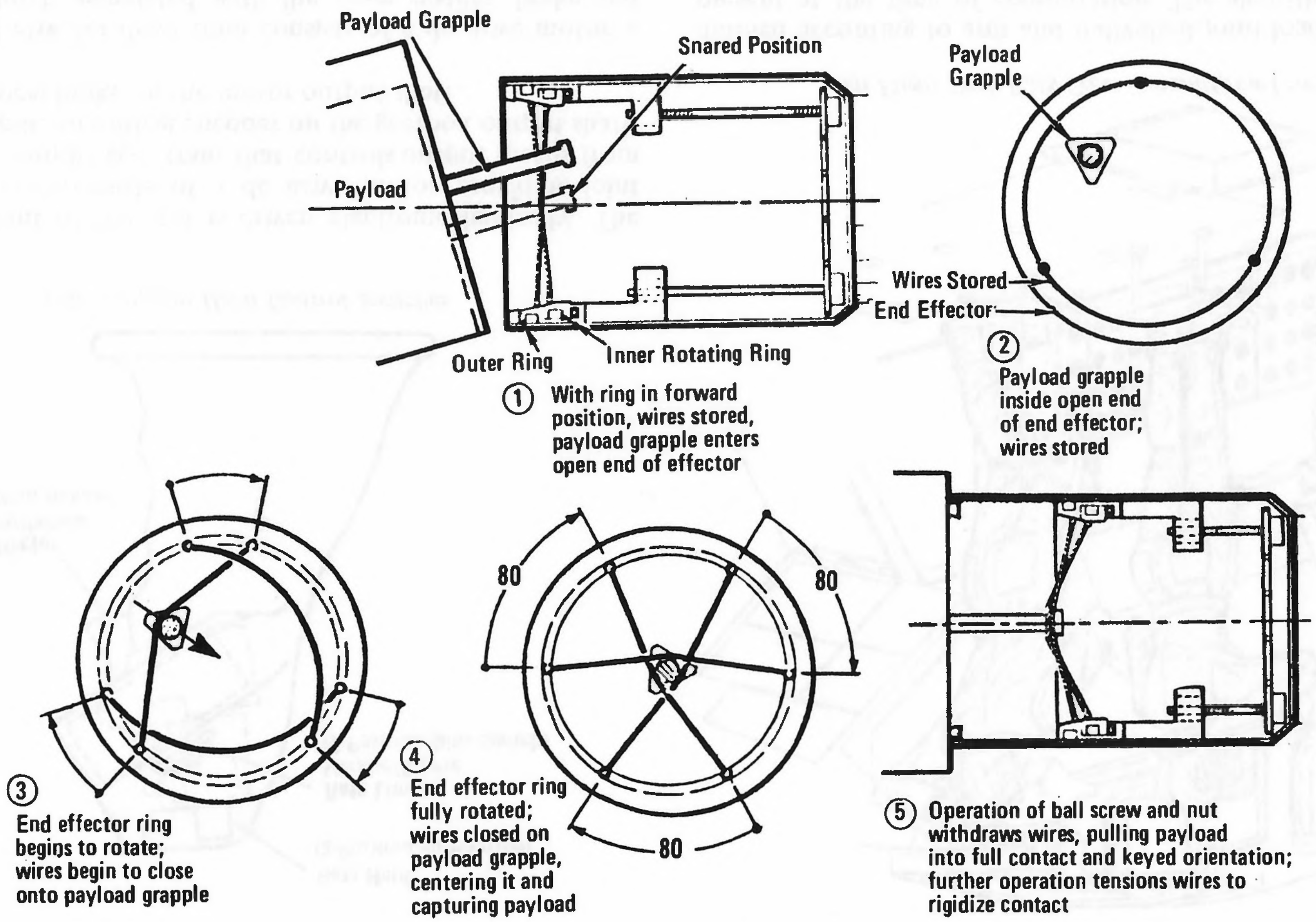
accomplished by drawing the snare assembly into the rear of the end effector or moving the snares forward toward the open end of the effector. In the automatic mode, rigidization is automatic; when manually operated, a switch on the aft flight deck station display and control panel is used to rigidize or derigidize the effector.

The end effector generates six data signals corresponding to the following indications: snares fully open, snares fully closed, payload present, carriage fully extended, maximum tension level crossed, and zero tension crossed.

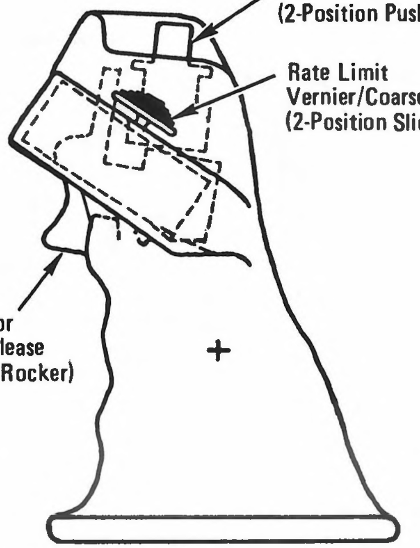
The arm has provisions for a closed-circuit TV camera and a viewing light on the wrist section, as well as closed-circuit TV camera and a pan and a tilt unit at the elbow lower arm transition.

The RMS operator controls arm position and attitude by viewing it through the aft or overhead windows at the aft flight deck station, as well as by using closed-circuit TV from both the arm and payload-bay-mounted cameras. Two closed-circuit TV monitors at the aft flight deck station have split-screen capability.

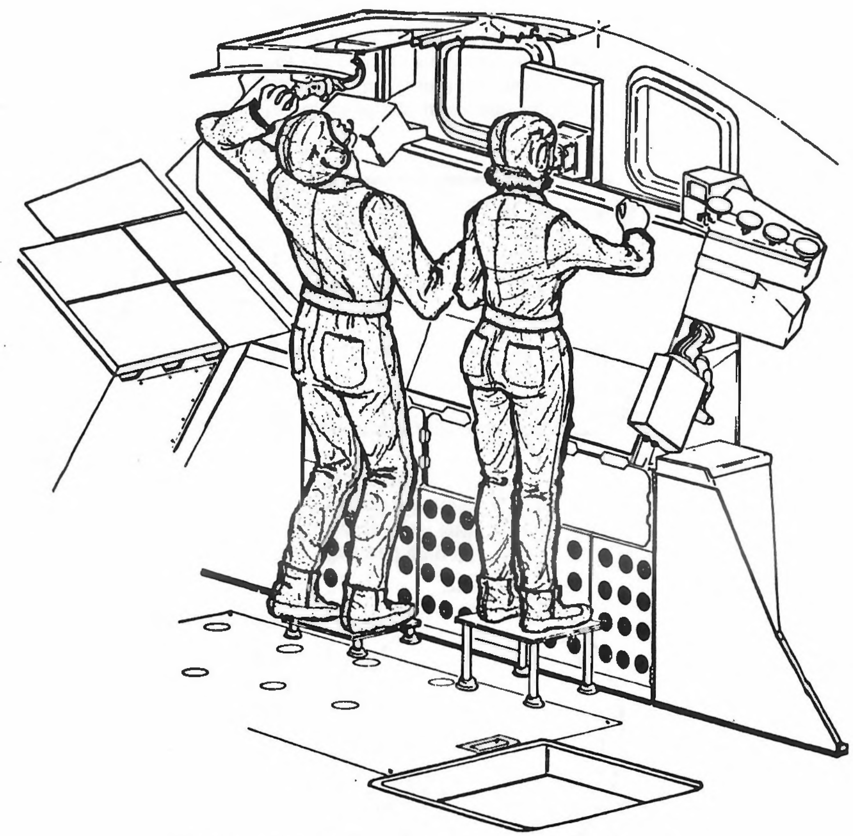
The RMS has both passive and active thermal control systems. The passive system consists of multilayer insulation blankets and thermal coatings. The active system consists of 26 heaters on each arm that supply 520 watts of power at 28 vdc. The heater system uses redundant buses on each arm, so if a failure occurs on one, the other is capable of supplying full heater power. The heaters operate automatically to maintain the temperature within the joints above -25°C (-14°F). Heater circuits are individually switched off as the corresponding temperature reaches 0°C (32°F). Twelve temperature thermisters per arm monitor the temperatures, which can be displayed at the aft flight deck station.



Snare Capture and Rigidization Sequence



RMS Rotation Hand Control Switches

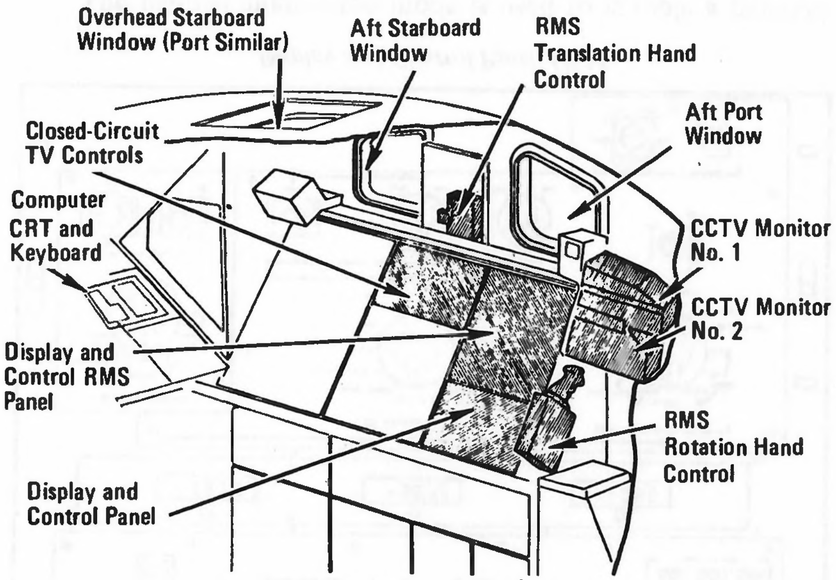


Aft Flight Deck RMS Crew Station/Crew Interface

Every joint of the arm is driven electromechanically. The joint drive train consists of a dc drive motor providing joint actuation, an output gear train that controls output speeds from the motor input, an optical encoder on the gearbox output shaft, and a mechanical brake on the motor output shaft.

The end effector drive train consists of a dc drive motor, a brake and clutch associated with the snare system, brake and clutch associated with the rigidization carriage and a differential unit. A spring mechanism is used for backup release.

The joint motor tachometers are the prime means of motion sensing, augmented by optical encoders. Tachometer data is supplied to control algorithms, which convert input drive commands to an output rate demand resolved for each joint of the arm. The algorithms output this rate demand within limits defined according to arm and individual joint loading conditions present at the time of computation. The algorithms supply the rate demand to control either end effector speed or position. The maximum attainable commanded velocity for the end effector and individual joints is limited by arm loading conditions, as is the maximum torque that can be applied to an individual joint under certain conditions. The aspect of arm control is provided by end effector velocity, joint rate, and motor current limiting within the software system under normal operating conditions. Joint velocity is limited during software-supported control modes

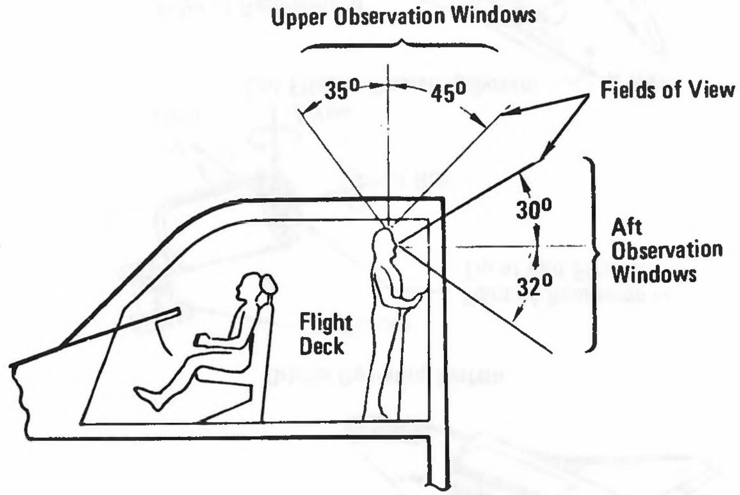


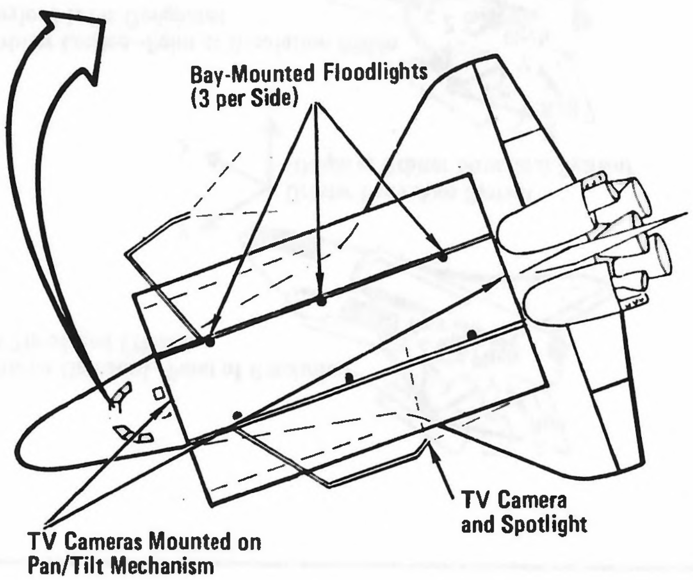
RMS Controls and Displays

by specifying a rate limit for each joint by the software system. Current limiting by the computer occurs during capture/rigidization operations. When the capture command is detected, the software commands zero current to all joint servos, except for the wrist roll joint servo; thus, for a short period there is a "limp" arm, except for the wrist roll joint. This is to allow for constrained motion adjustment during deployment.

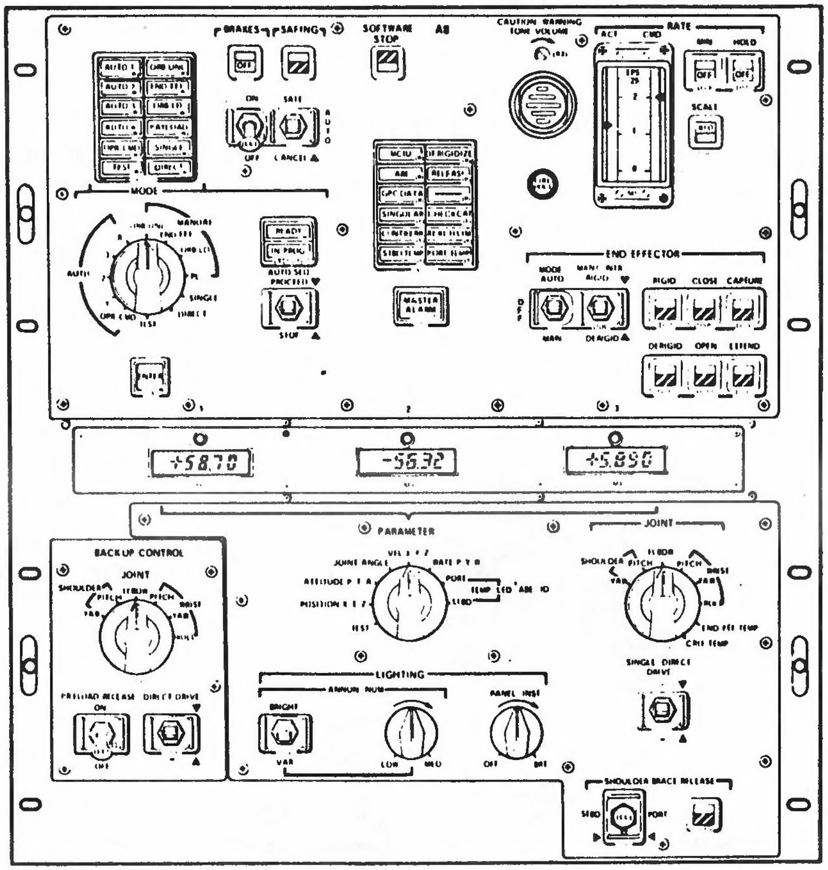
Normal braking is accomplished by motor deceleration, while the joint brakes are used for emergency or driving contingency operations only. Backdriving occurs when the payload or moving arm transmits kinetic energy into the drive train.

The RMS can be operated in any one of five different modes: automatic, manual augmented, manual single-joint drive, direct drive, and manual backup drive.



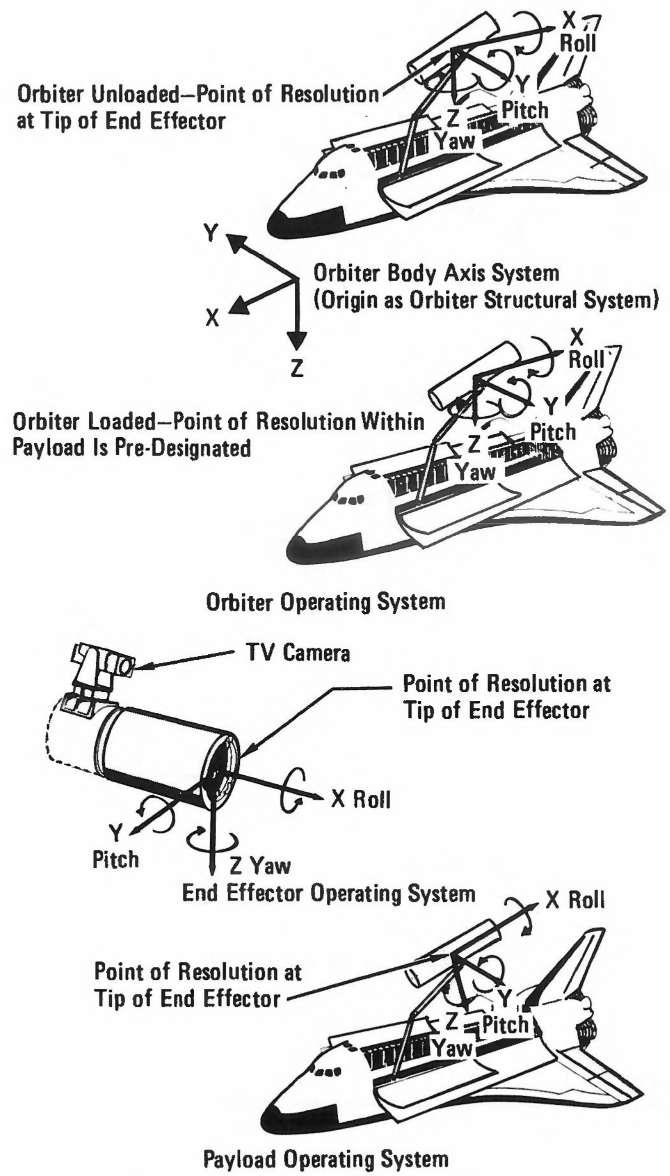


Payload Bay Television Cameras and Floodlights



Display and Control Panel A8A1

The manual augmented mode is used to grapple a payload, maneuver it into or out of the payload retention fittings or handling aids, and grapple or stow a special-purpose end effector in orbit. The manual augmented mode enables the operator to direct the end-point of the arm using two 3-degree-of-freedom hand controllers to control end effector translation and rotation rate. The control algorithms process the hand controller signals into a rate demand to each joint of the arm. The operator can



Control Coordinate Operating Systems

carry out manual augmented control of the arm using any four coordinate systems: orbiter, end effector, payload, or orbiter loaded.

When the manual orbiter mode is selected, rate commands through the aft flight deck station RMS translation hand control (THC) result in motions at the tip of the end effector which are parallel to the orbiter-referenced coordinate frame and compatible with the up/down, left/right, in/out direction of the THC. Commands from the aft flight deck station RHC result in rotation at the tip of the end effector, which are also about the orbiter-referenced coordinate frame.

The manual end effector mode is to maintain compatibility at all times between rate commands at the THC and RHC and the instantaneous orientation of the end effector. The end effector mode is used primarily for grappling operations in conjunction with a wrist-mounted CCTV camera which is oriented with the end effector coordinates and rolls with the end effector. The CCTV scene presented on the television monitor has viewing axes which are oriented with the end effector coordinate frame. This results in compatible motion between the rate commands applied at the hand controllers and movement of the background image presented on the television monitor. Up/down, left/right, in/out motions of the THC results in the same direction of motion of the end effector as seen on the television monitor, except that the background in the scene will move in the opposite direction. Therefore, the operator must remember to use a "fly to" control strategy and apply commands to the THC and RHC that are toward the target area in the television scene.

The manual orbiter loaded mode is to enable the operator to translate and rotate a payload about the orbiter axis with the point of resolution of the resolved rate algorithm being at a predetermined point within the payload, normally the center of geometry. This allows for pure rotations of the payload, which is useful for berthing operations.

There are two types of automatic modes, preprogrammed and operator commanded. The preprogrammed auto mode can store up to 20 automatic sequences in the computer, four of which can be assigned for selection at the aft flight deck station.

In the automatic modes, the payload is maneuvered to different locations for data taking according to a preprogrammed sequence.

Each automatic sequence is made up of a series of positions and attitudes of the end effector which define a trajectory of motion. The series may have from one to 199 points to define the trajectories. Pauses may be preprogrammed into the trajectory at any point. These will automatically cause the arm to come to rest, from which it may be able to proceed with the automatic sequence through the auto sequence "Proceed/Stop" switch on the aft flight deck station display and control panel. The operator can use the "Stop" position to halt the automatic sequence. This will bring the arm to rest; the switch is positioned to "Proceed" to resume the automatic sequence. When the last point in the sequence is reached, the computer will terminate the movement of the arm and enter a position hold mode. The speed of the end effector between points in a sequence is governed by the individual joint rate limits set in the RMS software.

The operator-commanded automatic mode moves the end effector from its present position and orientation to a new one defined by the operator to the computer via the keyboard and RMS cathode ray tube (CRT) display. After the data is keyed in, the RMS software verifies that the acquired position and orientation are "legal" with respect to arm configuration and reach envelope. The outcome of this check is displayed on the CRT. After the check, a "Ready" light will be displayed and the operator can execute the automatic sequence by placing the automatic sequence switch to "Proceed." The end effector will move in a straight line to the required position and orientation and then enter the hold mode. The operator can stop and start the sequence through the automatic sequence switch.

The single-joint drive control mode enables the operator to move the arm on a joint-by-joint basis with full computer support, thereby enabling full use of joint drive characteristics on a joint-by-joint basis. The operator supplies a fixed drive signal to the control algorithms via a toggle switch at the aft flight deck station. The algorithms supply joint rate demands to the selected joint while holding position on the other joints. The single-joint drive mode is used to stow and unstow the arm and drive it out of joint travel limits.

Direct-drive control is a contingency mode. It bypasses the manipulator control interface unit (MCIU), computer, and data buses to send a direct command to the motor drive amplifier (MDA) via hardwires. The direct-drive mode is used when the MCIU or computer has a problem that necessitates arm control by the direct drive mode to maneuver the loaded arm to a safe payload release position or to maneuver an unloaded arm to the storage position. The operator must place the brake on and select direct drive on the mode select switch. Since this is a contingency mode, full joint performance characteristics are not available. Computer-supported displays may or may not be available, depending on the fault that necessitated use of direct drive.

Backup drive control is a contingency mode used when the prime channel drive modes are not available. The backup is a degraded joint-by-joint drive system. It meets the fail-safe requirement of the RMS by using only the drive train of the prime channel.

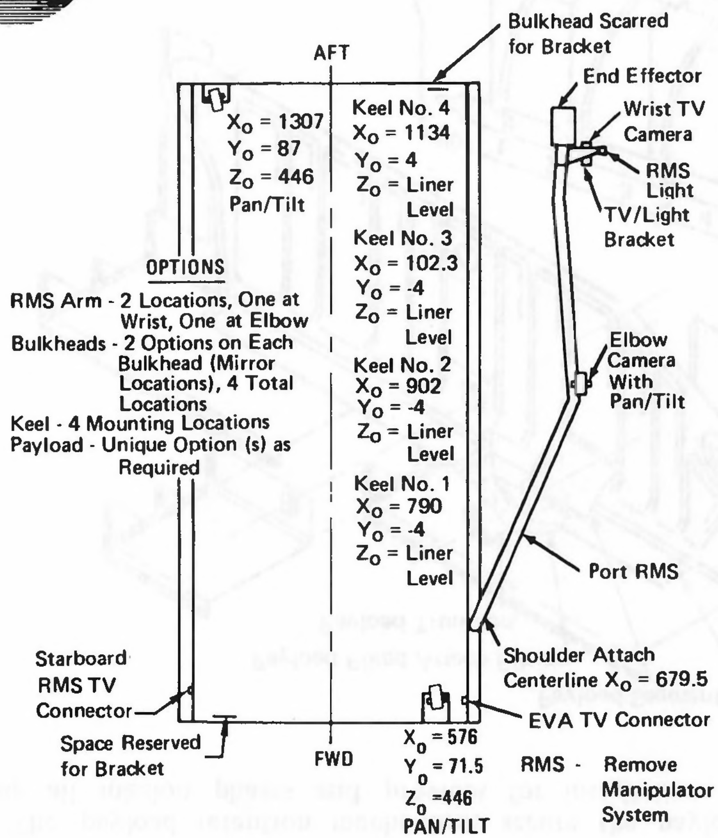
Safing and braking are the two methods available for bringing the arm to rest. Safing can be accomplished by the operator from the aft flight deck station or by the MCIU in receipt of certain failure indications. Operator-initiated safing is sent on hardwires to the input latches, setting them to zero and thus resulting in zero current to each joint independent of computer commands.

The RMS has a built-in test capability to detect and display critical failures. It monitors the arm based electronics (ABE), display and controls, and the MCIU software checks in the computer monitor computations. Failures are displayed on the aft flight deck station panel and on the CRT and also are available for downlinking through orbiter telemetry.

All of the major systems of the ABE are monitored by built-in test equipment. The MCIU checks the integrity of the communications link between itself and the ABE, display and control, and the orbiter computer. It also monitors end effector functions, thermistor circuit operation, and its own internal consistency. The computer checks cover an overall check of each joint's behavior through the consistency check, encoder data validity, and end effector behavior, as well as the proximity of the arm to reach limits, soft stops, and singularities.

The caution/warning annunciators are located on the aft flight deck station display panel. There are six caution annunciators (port temperature, starboard temperature, reach limit, singularity, control error, and check CRT) and five warning annunciators (release, derigidize, ABE, GPC data, and MCIU). A "Master Alarm" light and an audio signal attract the flight crew member's attention whenever a fault condition is detected.

A jettisoning system is installed within the Rockwell-provided manipulator positioning mechanism in the event the RMS cannot be stowed. Three floodlights are installed on each side of the payload bay. A portion of the orbiter closed circuit television (CCTV) system supports the payload deployment retrieval operations. The payload deployment retrieval operator uses the payload bay TV cameras, the remote manipulator arm cameras, the TV monitors, and the TV controls and displays to assist in all phases of the payload deployment retrieval system operations. There are five TV cameras available and they can be positioned in the following locations, depending upon mission



Remote Manipulator System

needs: arm wrist, arm elbow, forward port bulkhead, forward starboard bulkhead, aft port bulkhead, aft starboard bulkhead and keel (one of four predetermined positions).

The wrist TV camera is mounted on the roll joint of the arm; the elbow TV camera is mounted on the lower arm boom next to the elbow joint. The payload bay bulkhead TV camera brackets are attached to the aft and forward bulkheads. The keel camera bracket is mounted to the bottom of the payload bay. The TV monitors and the displays and controls are mounted on the aft flight deck display and control panel station.

The TV cameras used for payload deployment and retrieval operations are identical and, therefore, interchangeable. They are black and white cameras. The cameras have a pan/tilt unit, which provides plus or minus 170^{o} in pan and tilt, except when used on the arm's wrist or in the payload keel.

There are two black and white monitors. The monitors' electronic crosshairs have both vertical and horizontal components at the electrical center of the image. They are used to align the cameras with targets and sighting aids. The crosshairs are also used to align overlays with the monitor image. Alphanumerics are available on the monitors. The pan and tilt angles are displayed in degrees and tenths of degrees when the monitors display full scene images. The alphanumerics can be turned off. Each monitor can display two images simultaneously. The right or left half of the monitor will display the center half of the selected camera scene when the split screen mode is used.

Spar Aerospace Limited, Toronto, Canada, is the prime contractor to the National Research Council for development of the RMS for NASA. CAE Electronics Ltd, Montreal is responsible for the displays and controls in the orbiter. RCA Ltd, Montreal is responsible for the electronic interfaces, provides servo amplifiers and power conditioners. Dilworth, Secord, Meagher and Assoc. Ltd (DSMA), Toronto is responsible for the end effector.

PAYLOAD RETENTION MECHANISMS

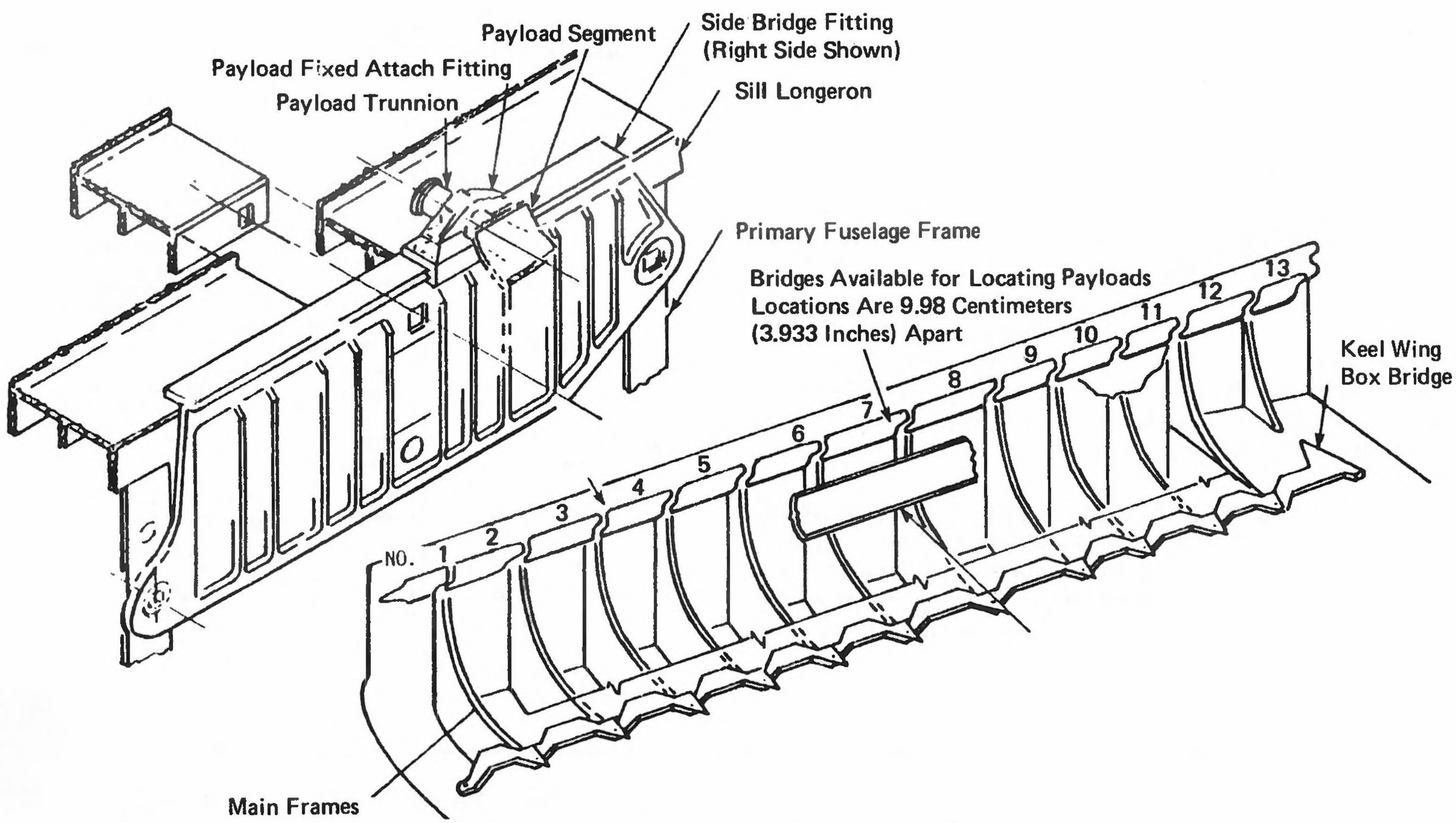
Nondeployable payloads are retained by passive retention devices, whereas ,deployable payloads are secured by motor-driven, active retention devices.

Payloads are secured in the orbiter payload bay by means of the payload retention system or are equipped with their own unique retention systems.

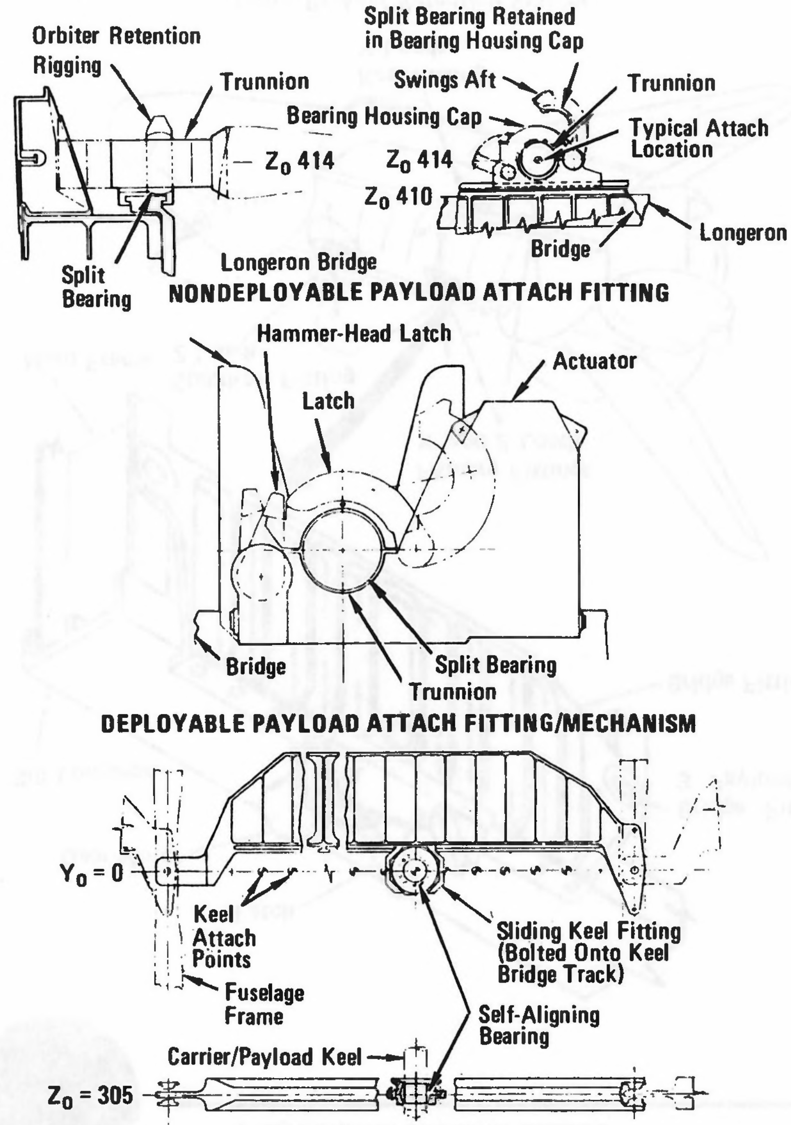
The orbiter payload retention system provides three-axis support for up to five payloads per flight. After the initial orbiter development flights, the payload bay will be modified to accommodate attach fittings for five payloads.

The payload retention mechanisms secure the payloads during all mission phases and provides for installation and removal of the payloads when the orbiter is either horizontal or vertical.

Attachment points in the payload bay are in 9.9-centimeter (3.933-inch) increments along the left- and right-side longerons and along the bottom centerlinė of the bay. Of the potential 172



Payload Retention



Standard Attach Fittings for Payloads

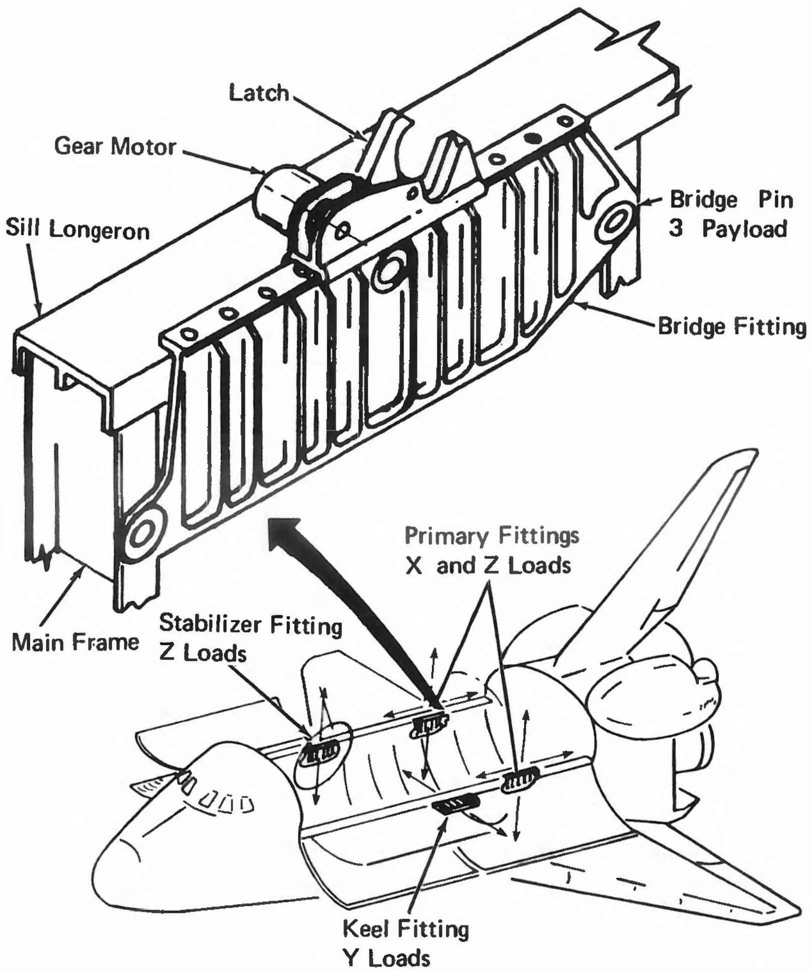
attach points on the longerons, 45 are unavailable because of the proximity of spacecraft hardware. The remaining 127 may be used for carrier/payload attachment; of these, 111 may be used for deployable payloads. Along the centerline keel, 104 attach points are available, any of which may be used for deployable payloads. There are 13 longeron bridges per side and 12 keel bridges available per flight. Only the bridges required for a particular flight are flown. The bridges are not interchangeable because of main frame spacing, varying load capability, and subframe attachments.

The longeron bridge fittings are attached to the payload bay frame at the longeron level and at the side of the bay. Keel bridge fittings are attached to the payload bay frame at the bottom of the payload bay.

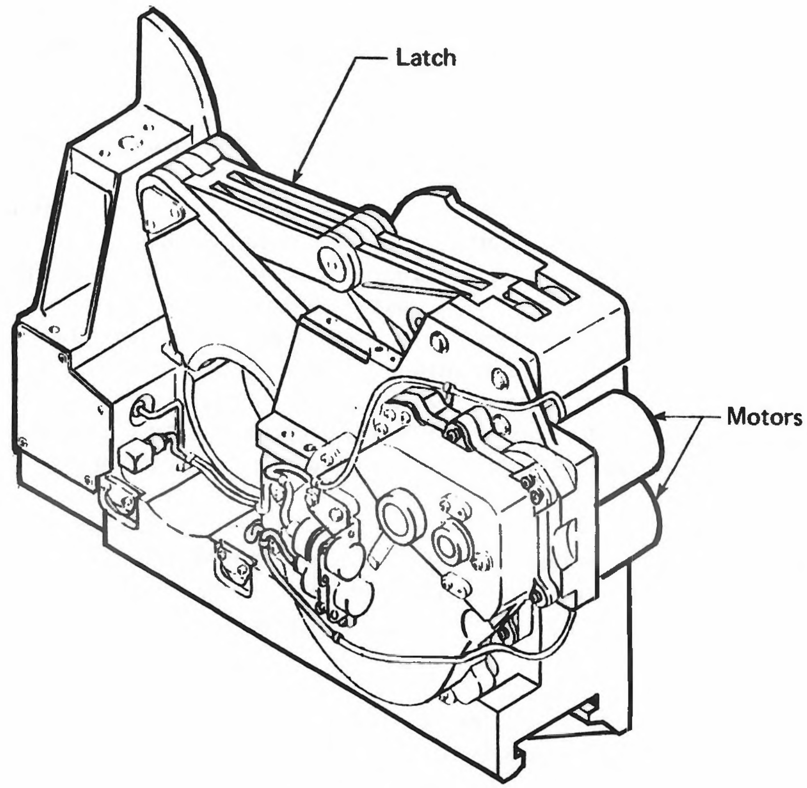
The payload trunnions are the interfacing portion of the payload with the orbiter retention system. The trunnions that interface with the longeron are 8.2 centimeters (3.25 inches) in diameter and 17.78 or 22.22 centimeters (7 or 8.75 inches) long, depending upon where they are positioned along the payload bay. The keel trunnions are 7.62 centimeters (3 inches) in diameter and vary in length from 10.16 to 29.21 centimeters (4 to 11.5 inches), depending upon where they fit in the payload bay.

The orbiter/payload attachments are the trunnion/bearing/journal type. The longeron and keel attach fitting have a split, self-aligning bearing for nonrelease-type payloads in which the hinged half is bolted closed. For on-orbit deployment and retrieval payloads, the hinged half fitting releases or secures the payload by latches that are driven by dual redundant electric motors.

Payload guides and scuff plates are used to assist in deploying and berthing payloads in the payload bag. The payload is constrained in the X direction by guides and in the Y direction



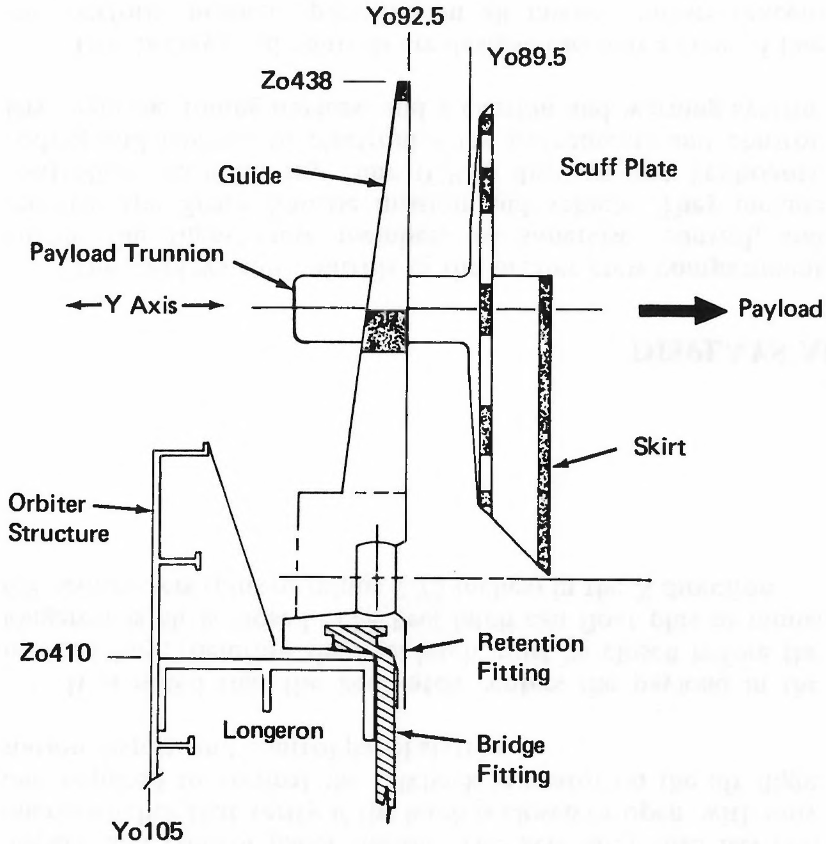
Active Payload Retention System



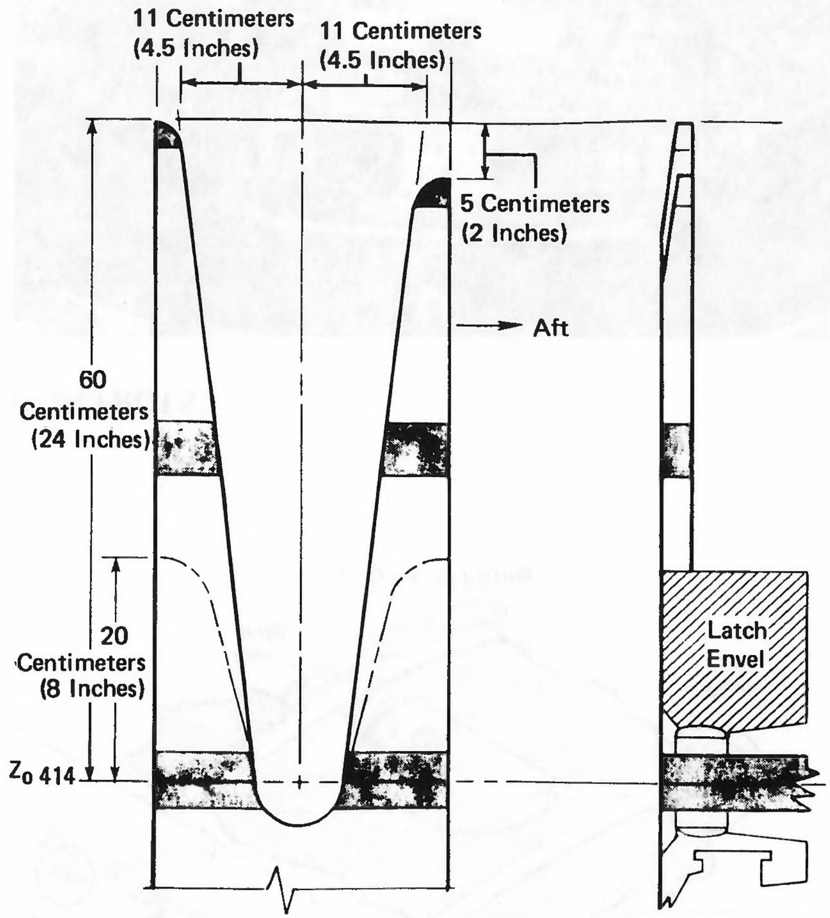
Payload Retention Latch

by scuff plates. The guides are mounted to the inboard side of the payload latches and interface with the payload trunnions and scuff plates. The scuff plates are attached to the payload trunnions and interface with the payload guides.

The guides are V shaped with the forward part of the V being 5.08 centimeters (2 inches) taller than the aft part. This difference enables the operator monitoring the berthing or deployment operations through the aft bulkhead TV cameras to better determine when the payload trunnion has entered the guide. The top of the forward portion of the guide is 60.96 centimeters (24 inches) above the centerline of the payload trunnion when it is all the way down in the guide. The top of the guide has a 22.86-centimeter (9-inch) opening. These guides are mounted to the 20.32-centimeter (8-inch) guides that are a part of the longeron payload retention latches.



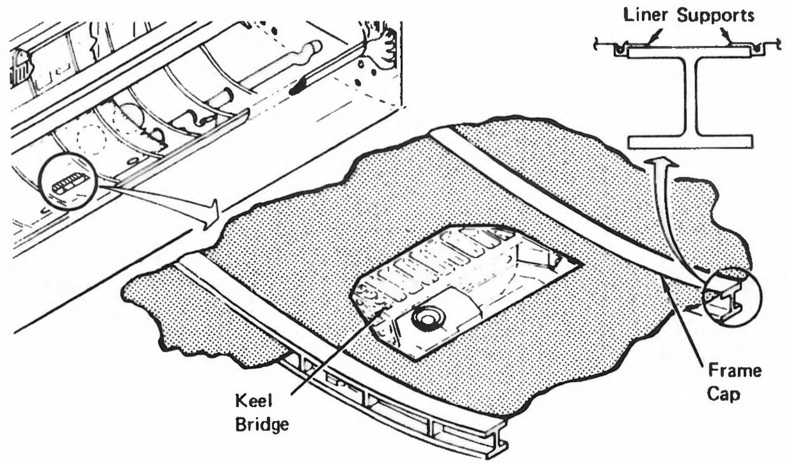
Orbiter Payload Guide and Trunnion/Scuff Plate (Nominal)



Orbiter Active Latch Guide

The payload scuff plates are mounted to the payload trunnions or the payload structure. There are normally three or four longeron latches and a keel latch for on-orbit deployment and retrieval of payloads. These latches are controlled by dual redundant electric motors with either or both motors releasing or latching the mechanism. The operating time of the latch is four seconds with both motors operating or eight seconds with one motor operating. The latch/release switches on the aft flight deck display and control panel station control the latches. Each longeron latch has two microswitches sensing the ready-to-latch condition. Only one is required to control the ready-to-latch talkback indicator on the aft flight deck display and control panel station. Each longeron latch also has two microswitches to indicate latch and two to indicate release. Only one of each is required to control the latch or release talkback indicator on the aft flight deck display and control panel station. The keel latch also has two microswitches that sense when the keel latch is closed with the trunnion in it. Only one of the switches is required to operate the talkback indicator on the aft flight deck display and control panel station. The keel latch also has two microswitches that verify if the latch is closed or open, with only one required to control the talkback indicator on the aft flight station display and control panel station.

It is noted that the keel latch centers the payload in the payload bay; therefore the keel latch must be closed before the longeron latch is closed. The keel latch can float plus or minus 6.9 centimeters (plus or minus 2.75 inches) in the X direction.

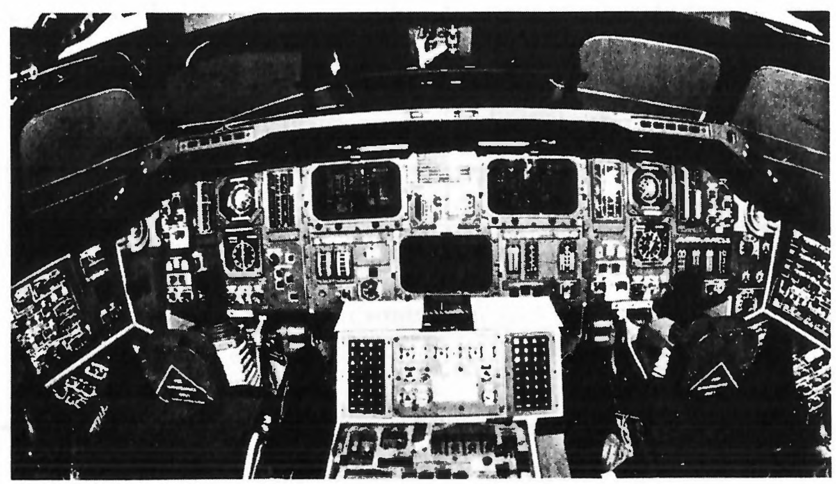


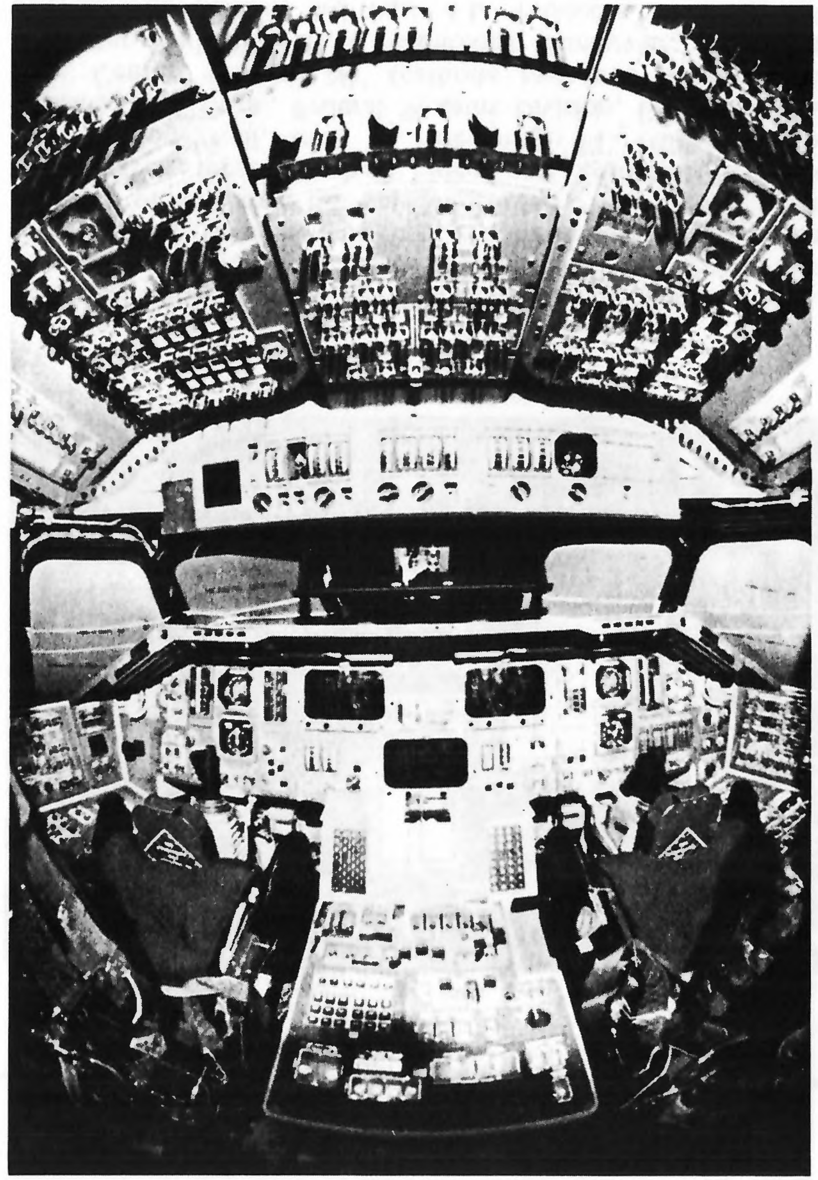
Active Keel Fitting

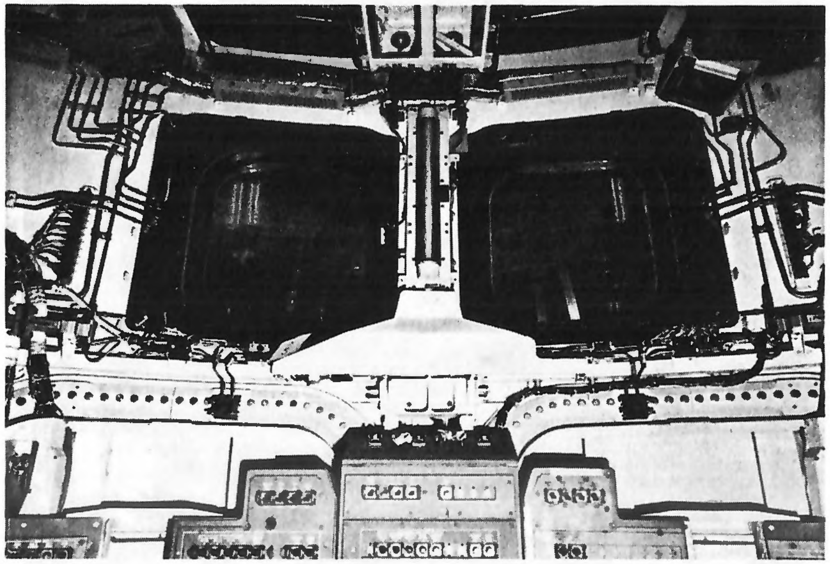
DISPLAYS AND CONTROLS

The displays and controls in the orbiter crew compartment enable the flight crew members to supervise, control, and monitor the Space Shuttle mission and vehicle. They include controllers, cathode ray tube (CRT) displays and keyboards, coding and conversion electronics for instruments and controllers, lighting, timing devices, and a caution and warning system.

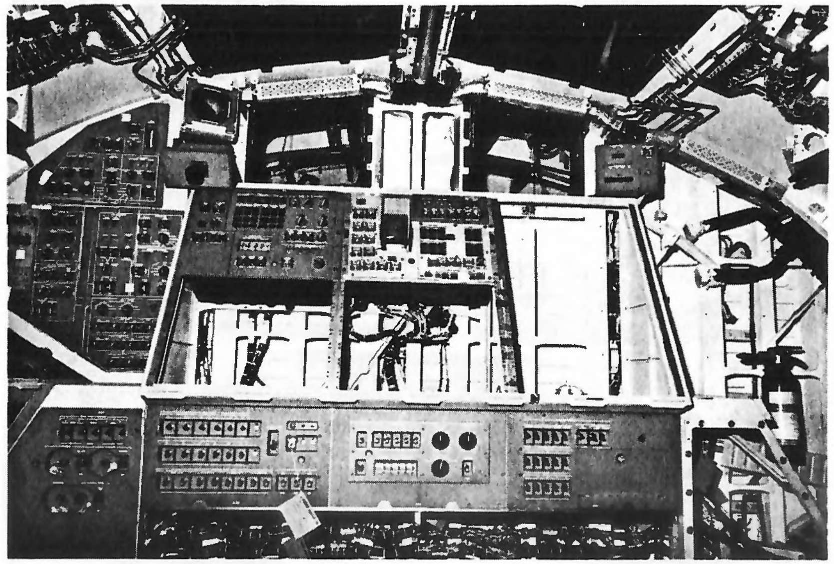
The displays and controls are designed so that a crew of two can perform normal operations in all mission phases (except payload operations). They are designed to enable a safe return to earth from either commander's or pilot's seat; flight-critical displays and controls are accessible from the forward flight deck station from launch to orbital operations and from deorbit to landing rollout.


Forward Displays and Controls





Aft Station Displays and Controls



Aft Station Displays and Controls

All controls are protected against inadvertent activation. Toggle switches are protected by wicket guards, and lever lock switches are used wherever inadvertent activation would be detrimental to flight operations or could damage equipment. Cover guards are used on switches where inadvertent actuation would be irreversible.

All displays and controls are provided with dimmable floodlighting in addition to integral meter lighting.

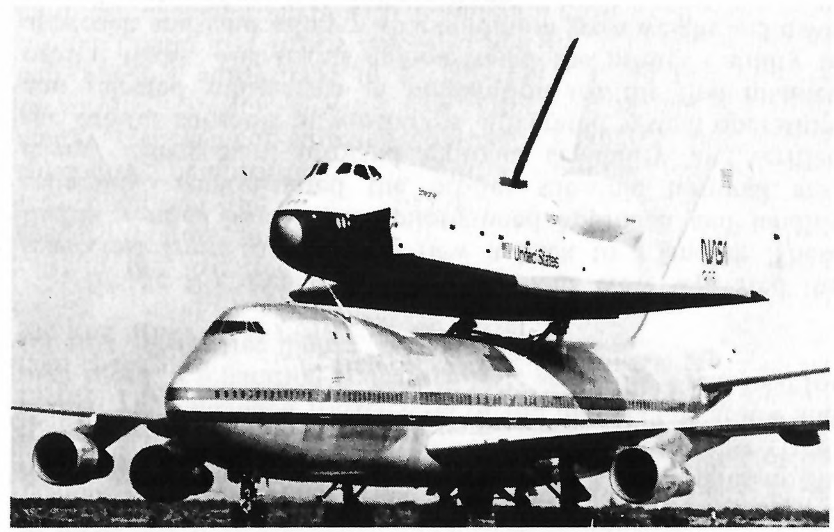
The contractors involved are: Abbott Transistor, Los Angeles, CA (transformers); Aerospace Avionics, Bohemia, NY (propellant quantity indicator and annunciators); Aiken Industries, Mechanical Product Division, Jackson, MI (thermal circuit breakers); Applied Resources, Fairfield, NJ (rotary switch); Bendix Corp. . Teterboro, NJ (surface position, alpha-Mach, altitude, vertical velocity indicators); Bendix Corp., Davenport, IA (accelerometer indicator); Conrac Corp., West Caldwell, NJ (mission and event timer); Edison Electronics Division of McGraw Edison, Manchester, NH (digital select thumbwheels, toggle switches); Eldec Corp., Lynwood, WA (tape meter); Honeywell, Inc., St. Petersburg, FL (flight control system); IBM Corp., Federal Systems Division, Electronic Systems Center, Owego, NY (cathode ray tube display unit, computer keyboard); ILC Technology, Sunnyvale, CA (cabin interior and exterior lighting); J.L. Products, Gardena, CA (pushbutton switch); Lear Siegler, Grand Rapids, MI (attitude direction indicator); Martin Marietta, Denver CO (C/W status display, limit module); Weston Instruments, Newark, NY (event indicator, electrical indicator meter); Collins-Rockwell, Cedar Rapids, IA (display driver unit, horizontal situation indicator); U.S. Radium Inc., Parisippany, NJ (integrally-illuminated panels); Betatronix, Hauppauge, NY (potentiometers).

SHUTTLE CARRIER AIRCRAFT

The NASA Shuttle carrier aircraft (SCA) is a Boeing 747 (100 series) purchased from American Airlines on June 17, 1974. It was modified to ferry orbiters to and from various Shuttle facilities and to transport and release Orbiter 101 for the ALT program.

Modifications to the basic 747 aircraft included removal of interior equipment (passenger seats, galleys, etc.); changes to air conditioning ducts, electrical wiring, and plumbing; installation of higher-thrust engines (JT9D-7AHW) and the 747-200 series rudder ratio-changer; and alteration of the longitudinal trim system to permit two degrees more nose-down trim. Other changes included relocation and installation of antennas, addition of bulkheads and doublers in the fuselage main deck, addition of structural doublers and tip fins to the horizontal stabilizers to improve directional stability with the orbiter on top of the aircraft, and addition of one forward and two aft support assemblies for attachment of the orbiter. The modifications increased the basic weight of the aircraft by approximately 1270 kilograms (2800 pounds).

The Orbiter's mated location on the 747 was based on consideration of static stability and control, structural modification, weight, and performance. Center-of-gravity limits for the 747 with the orbiter mated are 15 percent of the 747's mean aerodynamic chord (MAC) for the forward limit and 33 percent MAC for the aft limit. Longitudinal stability is similar to that of the basic 747; ballast must be added so that the center of gravity limits are not exceeded. The ballast is carried in standard 747 cargo containers in the forward cargo compartment. The mated



Boeing 747 Shuttle Carrier

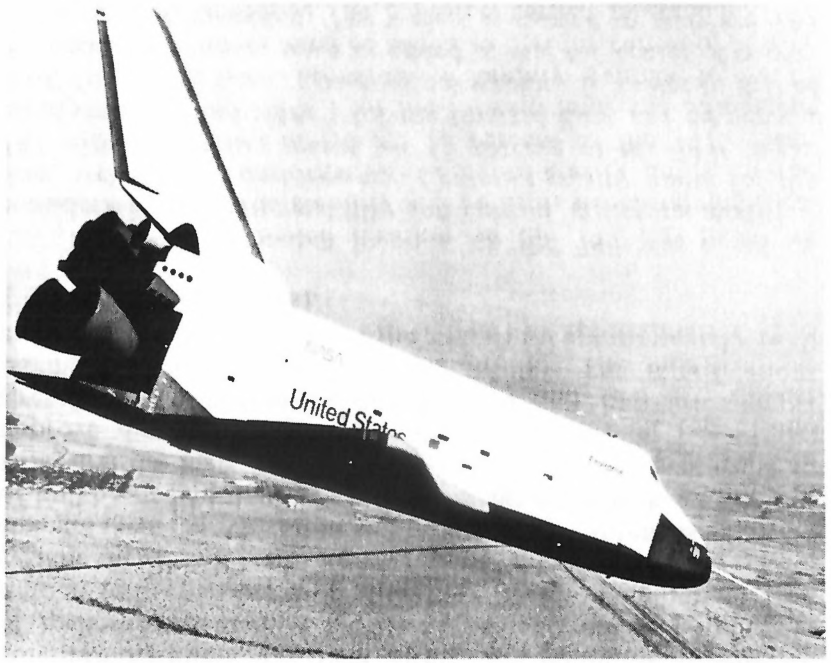
configuration allows the 747 center of gravity to shift approximately 3 meters (10 feet) upward.

For the ferry flight configurations, the tail cone fairing is installed on the orbiter to decrease aerodynamic drag and buffet, and aerosurface control locks are added to the orbiter's elevons. The orbiter is unmanned and the orbiter systems inert. A bailout system also is installed in the 747.

Some modifications to the 747 SCA are removable. These include support struts for the orbiter, horizontal tip fans, and associated cabling and umbilicals.

ORBITER APPROACH AND LANDING TEST PROGRAM

The nine-month-long Approach and Landing Test (ALT) program in 1977 at NASA's Dryden Flight Research Center, Edwards Air Force Base, proved that America's newest spacecraft, the Enterprise (OV-101), is truly a "magnificent flying machine and its aerodynamic flight controls handle crisply and accurately."



Enterprise Approaches Edwards AFB Runway in ALT Flight

Two NASA astronaut crews–Fred Haise and Gordon Fullerton and Joe Engle and Dick Truly–took turns flying the 68,040 kilogram (150,000-pound) spacecraft to free-flight landings and described it as "magnificent, precise, excellent, outstanding."

The ALT program involved a total of 13 flights (five unmanned "captive," three manned captive, and five free flights), as well as various ground tests.

The ground tests included taxi tests of the mated 747-orbiter to determine structural loads and responses and assess the mated capability in ground handling and control characteristics up to flight takeoff speed. The taxi tests also validated 747 steering and braking with orbiter attached. A ground test of orbiter systems followed the unmannned captive tests. All orbiter systems were activated as they would be in atmospheric flight; this was the final preparation for the manned captive flight phase.

In the unmanned captive flights, the orbiter–its systems inert–was attached atop the 747 Shuttle carrier aircraft. These flights assessed the structural integrity and performance handling qualities of the mated craft.

The three manned captive flights which followed included an astronaut crew aboard the orbiter operating its flight control systems while the orbiter remained perched atop the 747. These flights were designed to exercise and evaluate all systems in the flight environment in preparation for the orbiter release (free) flights. They included flutter tests of the mated craft at low and high speed, a separation trajectory test, and a dress rehearsal for the first orbiter free flight.

In the five free flights the astronaut crew separated the spacecraft from the 747 and flew it back to a landing. These flights verified the orbiter pilot-guided approach and landing capability, demonstrated the orbiter subsonic terminal area energy management autoland approach capability, and verified the orbiter subsonic airworthiness, integrated system operation, and selected subsystems in preparation for the first manned orbital flight. The flights demonstrated the orbiter's ability to approach and land safely with a minimum gross weight and using several center-of-gravity configurations.

For all of the captive flights and the first three free flights the Orbiter was outfitted with a tail cone covering its aft section to reduce aerodynamic drag and turbulence. The final two free flights were without the tail cone and the simulated engines–three main and two orbital maneuvering–were exposed aerodynamically.

The final phase of the ALT program prepared the spacecraft for four ferry flights. Fluid systems were drained and purged, the tail cone was re-installed, and elevon locks were installed. The forward attachment strut was replaced to lower the orbiter's cant from six to three degrees. This reduces drag of the mated vehicles during the ferry flights.

After the ferry flight tests, Orbiter 101 was returned to the NASA hangar at Dryden FRC and modified for vertical ground vibration tests at the Marshall Space Flight Center, Huntsville, AL, and then was ferried to Huntsville. At the Marshall Space Flight Center, the Enterprise was mated with the external tank and solid rocket boosters and subjected to vertical ground vibration tests. These tested the mated configuration's critical structural dynamic response modes, which were assessed against analytical math models used to design the various element-element interfaces.

The Enterprise then was ferried to the Kennedy Space Center, where it was mated with the external tank and solid rocket boosters and transported via the Mobile Launch Platform to Launch Complex 39A. At Launch Complex 39A, it served as a practice and launch complex fit check verification tool representing the flight vehicles.

Finally, the Enterprise was ferried back to Edwards AFB, CA, and returned via overland transport to Rockwell's Palmdale final assembly facility. Certain components will be refurbished for use on flight vehicles being assembled at Palmdale. It will eventually be used as a practice and fit check verification tool at Vandenberg AFB, CA.

SPACE SHUTTLE CHRONOLOGY

Date	Event
1972	
Aug. 9	Rockwell receives authority to proceed, Space Shuttle orbiter
1974	
June 4	Start structural assembly crew module (Orbiter 101)
July 17	Start long-lead fabrication (MPTA-098)
Aug. 26	Start structural assembly aft fuselage (Orbiter 101)
1975	
Jan. 6	Start long-lead fabrication aft fuselage (STA-099)
March 27	Mid fuselage on dock, Palmdale (Orbiter 101)
March 27	Start long-lead fabrication aft fuselage (Orbiter 102)
May 23	Wings on dock, Palmdale–less elevons, seals, and main gear doors (Orbiter 101)
May 27	Vertical stabilizer on dock, Palmdale (main fin box only) (Orbiter 101)
June 24	Start structural assembly (MPTA-098)
Aug. 25	Start final assembly and closeout system installation (Orbiter 101)
Sept. 5	Aft fuselage on dock, Palmdale (Orbiter 101)
Oct. 17	Space Shuttle main engine first mainstage test at NSTL
Oct. 31	Lower forward fuselage on dock, Palmdale (orbiter 101)

Date	Event
Nov. 17	Start long-lead fabrication crew module (Orbiter 102)
Dec. 1	Upper forward fuselage on dock, Palmdale (Orbiter 101)
Dec. 20	First Space Shuttle main engine 60-second duration test, NSTL
Dec. 31	1/4-scale model ground vibration test facility construction complete, Downey, Building 288
1976	
Jan. 16	Crew module on dock, Palmdale (Orbiter 101)
Jan. 23	MPTA-098 truss on dock, Downey
Feb. 16	Start fabrication forward fuselage (STA-099)
March 3	Payload bay doors on dock, Palmdale (Orbiter 101)
March 12	Complete final assembly and closeout system installation, Palmdale (Orbiter 101)
March 15	Start functional checkout (Orbiter 101)
March 17	Complete pre-mate MPTA test structure, Downey, and deliver to Palmdale
April 2	Crew escape system test sled on dock, Downey
April 3	Complete assembly and deliver MPTA structure on dock, Lockheed Test Site, Palmdale
April 22	Body flap on dock, Palmdale (Orbiter 101)
May 3	Complete MPTA-098 proof load test set-up, Lockheed test site, Palmdale
June 14	Start aft fuselage assembly (STA-099)
June 24	Complete MPTA-098 proof load test, Lockheed test site and on dock, Palmdale
June 25	Complete functional checkout (Orbiter 101)
June 28	Start assembly crew module (Orbiter 102)
June 28	Start horizontal ground vibration tests and proof load tests (Orbiter 101)
June 29	MPTA-098 truss assembly, Palmdale, Bldg. 294 to Bldg. 295
June 30	Space Shuttle main engine dummy set on dock, Palmdale (Orbiter 101)
July 8	MPTA-098 on dock, Downey, without truss assembly
July 12	Start installation secondary structure (MPTA-098)
Aug. 2	Start final assembly forward fuselage (STA-099)
Aug. 2	Start carrier aircraft modification
Aug. 20	Complete horizontal ground vibration tests and proof load tests (Orbiter 101)
Aug. 23	Start Delta F modification (Orbiter 101)
Aug. 27	Reaction control system/orbital maneuvering system pods (simulated), approach and landing tests, on dock, Palmdale (Orbiter 101)
Aug.	Start overland roadway construction from Palmdale to DFRC
Sept. 10	Complete Delta F modifications (Orbiter 101)
Sept. 13	Start preparations for first rollout (Orbiter 101)
Sept. 13	Start assembly aft fuselage (Orbiter 102)
Sept. 17	Rollout (Orbiter 101)
Sept. 17	Complete on-stand construction, NSTL
Sept. 20	Start Delta F retest (Orbiter 101)
Oct. 1	Start final assembly, vertical stabilizer (STA-099)
Oct. 1	Start final assembly, wing (STA-099)
Oct. 15	Mid fuselage on dock, Palmdale (STA-099)
Oct. 26	Escape system test assembly sled ship from Downey to Holloman, NM
Oct. 29	Deliver ejection seats (Orbiter 101)
Oct. 29	Complete Delta F retest (Orbiter 101)
Oct. 31	Solid rocket booster 1/4-scale model (burnout configuration) on dock, Downey
Nov. 4	Complete 747 Shuttle carrier aircraft modification, rollout, Boeing
Nov. 18	Start escape system sled test, Holloman, NM
Nov. 26	Complete integrated checkout (Orbiter 101)
Nov. 28	Complete orbiter transporter strongback
Dec. 7	Tail cone fairing on dock, Palmdale

Date	Event
Dec. 10	Complete overland roadway construction, Palmdale to DFRC
Dec. 13	Start assembly upper forward fuselage (Orbiter 102)
Dec. 17	External tank 1/4-scale model on dock, Downey
1977	
Jan. 3	Start assembly vertical stabilizer (Orbiter 102)
Jan. 14	Complete post-checkout (Orbiter 101)
Jan. 14	Boeing 747 Shuttle carrier aircraft delivered to DFRC
Jan. 25	Complete aft fuselage assembly on dock, Palmdale (STA-099)
Jan. 28	Simulated crew module on dock, Palmdale (STA-099)
Jan. 31	Solid rocket booster 1/4-scale model (liftoff configuration) on dock, Downey
Jan. 31	Orbiter 101 transported overland from Palmdale to DFRC (36 miles)
Jan. 31	Mass simulated Space Shuttle Main Engines on dock, Palmdale (Orbiter 101)
Feb. 7	Orbiter 101 Shuttle carrier aircraft mate start
Feb. 10	Mid fuselage on dock, Palmdale (Orbiter 102)
Feb. 15	Complete Orbiter 101 Shuttle carrier aircraft mated ground vibration test and taxi tests
Feb. 18	Conduct first inert captive flight, DFRC (2 hr. 5 min.) (Orbiter 101)
Feb. 22	Conduct second inert captive flight, DFRC (3 hr. 13 min.) (Orbiter 101)
Feb. 25	Conduct third inert captive flight, DFRC (2 hr. 28 min.) (Orbiter 101)
Feb. 28	Conduct fourth inert captive flight, DFRC (2 hr. 11 min.) (Orbiter 101)

Date	Event
March 2	Conduct fifth inert captive flight, DFRC (1 hr. 39 min.) (Orbiter 101)
March 16	Wings on dock, Palmdale (STA-099)
March 21	Orbiter 1/4-scale model on dock, Downey
April 1	Lower forward fuselage on dock, Palmdale (STA-099)
April 6	Upper forward fuselage on dock, Palmdale (STA-099)
April 6	Vertical stabilizer on dock, Palmdale (STA-099)
May 20	Nose landing gear doors on dock, Palmdale (STA-099)
May 26	Aft payload bay doors on dock, Palmdale (STA-099)
May 27	Complete systems installation/final acceptance, MPTA-098, transport from Downey to Seal Beach
May 31	Solid rocket booster 1/4-scale model, burnout and maximum q configuration, on dock, Downey
May 31	Solid rocket booster 1/4-scale model, liftoff configuration, on dock, Downey
May 31	Body flap on dock, Palmdale (STA-099)
June 3	Ship MPTA-098 from Seal Beach to NSTL
June 7	Complete integrated checkout and hot fire ground test, DFRC (Orbiter 101)
June 18	Conduct first manned captive active flight, Orbiter 101–747, DFRC (55 min. 46 sec.)
June 23	Deliver first Space Shuttle main engine to NSTL (MPTA-098)
June 24	Deliver MPTA-098 to NSTL
June 28	Conduct second manned captive active flight, Orbiter 101–Shuttle carrier aircraft, DFRC, (1 hr. 2 min.)

Date	Event
July 5	Start long-lead fabrication aft fuselage, Orbiter 103
July 8	Deliver second main engine to NSTL (MPTA-098)
July 14	Deliver third main engine to NSTL (MPTA-098)
July 18	Conduct two-minute firing of solid rocket booster at Brigham City, Utah, Thiokol (2.4 million pounds of thrust)
July 22	Deliver forward payload bay doors, on dock, Palmdale (STA-099)
July 26	Conduct third manned captive active flight, Orbiter 101–Shuttle carrier aircraft, DFRC (59 min. 50 sec.)
Aug. 12	Conduct first free flight, ALT, tail cone on, DFRC (5 min. 21 sec.) (Orbiter 101)
Aug. 26	Deliver wings on dock Palmdale (Orbiter 102)
Sept. 7	Lower forward fuselage on dock Palmdale (Orbiter 102)
Sept. 10	Deliver external tank MPTA-098 (Martin Marietta) to NSTL
Sept. 13	Conduct second free flight, ALT, tail cone on, DFRC (5 min. 28 sec.) (Orbiter 101)
Sept. 23	Conduct third free flight, ALT, tail cone on, DFRC (5 min. 34 sec.) (Orbiter 101)
Sept. 30	Complete mate vertical stabilizer, Palmdale (STA-099)
Oct. 12	Conduct fourth free flight, ALT, first tail cone off, DFRC (2 min. 34 sec.) (Orbiter 101)
Oct. 26	Conduct fifth free flight, ALT, final tail cone off, DFRC (2 min. 1 sec.) (Orbiter 101)
Oct. 28	Lower forward fuselage on dock Palmdale (Orbiter 102)
Nov. 4	Start forward reaction control system development tests, White Sands Test Facility, N.M.
Nov. 4	Deliver aft fuselage on dock Palmdale (Orbiter 102)
Nov. 7	Forward reaction control system on dock Palmdale (STA-099)
Nov. 7	Start final assembly and closeout system installation Palmdale (Orbiter 102)
Nov. 15	First ferry flight test, DFRC (3 hr. 21 min.) (Orbiter 101)
Nov. 16	Second Ferry flight test, DFRC (4 hr. 17 min.) (Orbiter 101)
Nov. 17	Third ferry flight test, DFRC (4 hr. 13 min.) (Orbiter 101)
Nov. 18	Fourth ferry flight test, DFRC (3 hr. 37 min.) (Orbiter 101)
Dec. 9	Complete approach and landing flight tests including ferry flights (Orbiter 101)
Dec. 12	Start removal for mated vertical ground vibration test modification at DFRC (Orbiter 101)
Dec. 13	Complete propellant load testing NSTL (MPTA-098)
Dec. 31	Deliver Space Shuttle main engine envelope/ electrical simulators, on dock Palmdale (Orbiter 102)
1978	
Jan. 10	Vertical stabilizer on dock Palmdale (Orbiter 102)
Jan. 18	Second solid rocket booster firing, Thiokol
Feb. 10	Complete final assembly STA-099, Palmdale
Feb. 14	STA-099 on dock, Lockheed facility, Palmdale
Feb. 17	Crew module on dock, Palmdale (Orbiter 102)
Feb. 24	Body flap on dock Palmdale (Orbiter 102)
March 3	Complete modification for mated vertical ground vibration test, DFRC (Orbiter 101)

Date	Event
March 6	Upper forward fuselage on dock, Palmdale (Orbiter 102)
March 10	Ferry Orbiter 101 atop Shuttle carrier aircraft from DFRC to Ellington Air Force Base, TX (approximately 3 hours, 38 minutes)
March 13	Ferry Orbiter 101 atop Shuttle carrier aircraft from Ellington AFB to Marshall Space Flight Center, Huntsville, AL (approximate 2 hours)
March 19	Aft payload bay doors on dock, Palmdale (Orbiter 102)
March 31	External tank for mated vertical ground vibration test arrives at MSFC from Martin Marietta, Michoud, LA
March 31	Operational readiness date, solid rocket booster refurbishment and subassembly, Kennedy Space Center, FL
April 14	Complete ground vibration test modification at MSFC; deliver Orbiter 101 for mated vertical ground vibration test
April 21	First static firing, MPTA-098, NSTL (2.5 seconds; stub nozzles)
April 23	Complete final assembly and closeout system installation, ready for power-on (Orbiter 102)
April 24	Start pre-combined systems test (Orbiter 102)
April 28	Forward payload bay doors on dock, Palmdale (Orbiter 102)
May 19	Second static firing MPTA-098, NSTL (15 seconds, 70% thrust; stub nozzles)
May 19	Start forward reaction control system thermal tests, WSTF
May 26	Start development test aft reaction control system, WSTF
May 26	Complete forward reaction control system structure (Orbiter 102)
May 26	Upper forward fuselage mate, Orbiter 102

Date	Event
May 30	Start test Orbiter 101/external tank mated vertical ground vibration test, MSFC
May 31	Loaded solid rocket boosters (2) arrive at MSFC for mated vertical ground vibration test
June 15	Third static firing, MPTA-098, NSTL (50 seconds, 90% thrust, last 5 seconds 70% thrust; stub nozzles)
July 3	Deliver left-hand orbital maneuvering system pod to WSTF
July 7	Fourth static firing, MPTA-098, NSTL (No. 2 engine 70% thrust for 90 seconds, simulated one engine out at 90 seconds, then No. 1 and No. 3 to 90% thrust—then to 70% thrust, then to 90% for 100 seconds; stub nozzles)
July 7	Complete mate forward and aft payload bay doors (Orbiter 102)
July 13	Complete forward reaction control system development tests, WSTF
July 13	Reconfigure from boost to launch, mated vertical ground vibration test, MSFC (Orbiter 101)
July 15	Deliver solid rocket boosters (2) empty to MSFC for mated vertical ground vibration test.
July 21	First firing development test, orbital maneuvering system, WSTF
July 31	Operational readiness date, Orbiter Processing Facility Bay 1, Orbiter landing Facility and Hypergolic Maintenance Facility, KSC
Aug. 11	Complete forward reaction reaction control system (Orbiter 102)
Aug. 11	Complete test preparation STA-099, Lockheed facility, Palmdale
Aug. 11	Complete forward reaction control system thermal test, WSTF
Aug. 14	Start coefficient tests STA-099, Lockheed facility, Palmdale

Date	Event
Aug. 31	Operational readiness date, Vertical Assembly Building High Bay 3 and 4, KSC
Sept. 8	Start orbital maneuvering system left-hand development test, WSTF, N.M.
Sept. 20	Start launch configuration test liftoff configuration, Orbiter 101/external tank/solid rocket boosters mated vertical ground vibration test at MSFC
Sept. 20	Start acoustic test, forward reaction control system, WSTF
Sept. 25	Start pre-combined system test (Orbiter 102)
Sept. 29	Complete coefficient tests, STA-099, Lockheed facility, Palmdale
Sept. 30	Operational readiness date, mobile launch platform, KSC
Oct. 19	Third solid rocket booster firing, Thiokol
Nov. 11	Complete forward reaction control system acoustic test, WSTF
Nov. 15	Complete orbital maneuvering system development test, WSTF
Nov. 15	Complete aft reaction control system development test, WSTF
Nov. 30	Operational readiness date, Pad A, KSC
Dec. 9	Start orbital maneuvering system Phase I qualification tests, WSTF
Dec. 15	Start aft reaction control system Phase I qualification tests, WSTF
Dec. 15	Complete pre-combined system test, Palmdale (Orbiter 102)
1979	
Jan. 2	Start long-lead fabrication crew module (Orbiter 099)
Jan. 30	Start orbiter/external tank/solid rocket booster burnout mated vertical ground vibration test MSFC
Jan. 31	Deliver Shuttle carrier aircraft to DFRC for ferry operations
Jan. 31	Start left-hand orbital maneuvering system Phase I qualification tests, WSTF
Jan. 31	Mission Control Center Houston/Goldstone Ready for operational flight test early operations
Feb. 3	Complete combined systems test, Palmdale (Orbiter 102)
Feb. 16	Airlock on dock, Palmdale (Orbiter 102)
Feb. 17	Fourth solid rocket booster firing, Thiokol
Feb. 26	Complete mated vertical ground vibration test program at MSFC (Orbiter 101)
Feb. 28	Operational readiness date, Shuttle landing site, DFRC (Edwards AFB) Runway 23, for first manned orbital flight
March 5	Complete post-checkout, Palmdale (Orbiter 102)
March 8	Complete closeout inspection, final acceptance, Palmdale (Orbiter 102)
March 8	Orbiter 102 transported overland from Palmdale to DFRC (38 miles)
March 9	Shuttle carrier aircraft/Orbiter 102 test flight at NASA DFRC
March 17	Space Shuttle main engine 2005, first flight engine delivered to NSTL for acceptance test firings
March 20	Ferry flight, Shuttle carrier aircraft/Orbiter 102 from DFRC to Biggs Army Air Base, El Paso, TX (3 hr 20 min.)
March 22	Ferry flight, Shuttle carrier aircraft/Orbiter 102 from Biggs Army Air Base to Kelly AFB, San Antonio, TX (1 hr, 39 min.)
March 23	Ferry flight Shuttle carrier aircraft/Orbiter 102 from Kelly AFB to Eglin AFB, FL (2 hr. 12 min.)

Date	Event
March 24	Ferry flight Shuttle carrier aircraft/Orbiter 102 from Eglin AFB, to KSC (1 hr., 33 min.)
March 30	Space Shuttle main engine 2007, flight engine delivered to NSTL for acceptance test firing
April 6	Complete Phase I qualification tests, aft reaction control system, WSTF
April 10	Ferry flight, Shuttle carrier aircraft/Orbiter 101 from MSFC to KSC (1 hr., 52 min.)
April 16	Space Shuttle main engine 2006, flight engine delivered to NSTL, for acceptance test firing
April 18	Complete left-hand orbital maneuvering system Phase I qualification, WSTF
May 1	Orbiter 101/external tank/solid rocket boosters mated on mobile launch platform transported to Launch Complex 39A from VAB at KSC
May 4	Start forward reaction control system Phase I qualification tests, WSTF
May 4	Fifth static firing, MPTA-098 NSTL, flight nozzles 1.5 seconds
May 10	Deliver right-hand orbital maneuvering system/ reaction control system from McDonnell Douglas, St. Louis, to KSC (Orbiter 102)
May 15	Deliver left-hand orbital maneuvering system/ reaction control system from McDonnell Douglas to KSC (Orbiter 102)
May 30	Deliver external tank used in mated vertical ground vibration test from MSFC to Martin Marietta for refurbishment
June 12	Fifth static firing, MPTA-098 NSTL, flight nozzles (54 seconds, early cutoff, accelerometer filters)
June 15	First solid rocket booster qualification firing, Thiokol, UT, 122 seconds; nozzle extension severed at end of run as in actual mission; full cycle gimbal
June 21	Start assembly crew module (Orbiter 099)
July 2	Six static firing, MPTA-098 NSTL, flight nozzles (19 seconds, early cutoff—main fuel valve rupture)
July 23	Orbiter 101, external tank, solid rocket boosters transported on mobile launch platform from Launch Complex 39A to VAB at KSC
Aug. 1	Start long-lead fabrication crew module (Orbiter 103)
Aug. 6	Complete limit test (STA 099), Lockheed facility, Palmdale
Aug. 10	Ferry flight, Shuttle carrier aircraft/Orbiter 101, KSC to Atlanta (1 hr, 55 min.)
Aug. 11	Ferry flight, Shuttle carrier aircraft/Orbiter 101, Atlanta to St. Louis (1 hr, 50 min.)
Aug. 12	Ferry flight, Shuttle carrier aircraft/Orbiter 101, St. Louis to Tulsa (1 hr, 35 min.)
Aug. 13	Ferry flight, Shuttle carrier aircraft/Orbiter 101, Tulsa airport to Denver, (2 hr)
Aug. 14	Ferry flight, Shuttle carrier aircraft/Orbiter 101, Denver to Hill Air Force Base, Ogden, UT (1 hr, 30 min.)
Aug. 15	Ferry flight, Shuttle carrier aircraft/Orbiter 101, Ogden to Vandenberg Air Force Base (VAFB) (2 hr, 20 min.)
Aug. 16	Ferry flight, Shuttle carrier aircraft/Orbiter 101, VAFB to DFRC (1 hr, 10 min.)
Aug. 23	Orbiter 101/Shuttle carrier aircraft demate, DFRC
Aug. 27	Start long-lead fabrication, crew module (Orbiter 103)
Aug. 31	Complete OMS Phase II qualification tests, WSTF
Aug.	Second SRB qualification firing, Thiokol
Sept. 5	Complete forward RCS Phase I qualification tests, WSTF

Date	Event
Sept. 12	Start forward RCS Phase II qualification tests, WSTF
Sept. 21	Start aft RCS Phase II qualification tests, WSTF
Oct. 5	Complete setup and thermal tests (STA 099), Lockheed Facility, Palmdale
Oct. 24	Sixth static firing, MPTA-098, NSTL, flight nozzles (scrubbed, hydrogen detector oversensitive)
Oct. 30	Move Orbiter 101 from DFRC overland to Rockwell Palmdale facility (38 miles)
Nov. 2	Start OMS left-hand pod Phase II qualification, WSTF
Nov. 3	Complete Orbiter 102 APU hot fire tests, Orbiter Processing Facility, KSC
Nov. 4	MPTA-098 static firing, NSTL (10 seconds, flight nozzles, SSME LOX turbine, seal cavity pressure, high-cutoff steerhorn failure)
Nov. 7	Deliver STA-099 from Lockheed facility, Palmdale to Rockwell Palmdale for rework as second operational orbiter (redesignated OV-099)
Nov. 12	Complete qualification test OMS engine at WSTF
Dec. 7	Demate payload bay door, Palmdale, OV-099
Dec. 17	Sixth static firing, MPTA-098, NSTL, 554 seconds (340 seconds at 100% rated power level, then 90% at 385 seconds, 80% at 450 seconds, 70% at 505 seconds; 1 engine shutdown—other 2 continued at 70% until 554 seconds). Pogo and gimbaling tests accomplished; stub nozzles
Dec. 14	Complete demate body flap, Palmdale, OV-099
Dec. 16	Orbiter integrated test start Orbiter 102, KSC
Dec. 21	Complete demate elevons, Palmdale, Orbiter 099
1980	
Jan. 14	Complete orbiter integrated test, Orbiter 102, KSC
Jan. 18	Vertical stabilizer on dock, Fairchild, NY, for rework, Orbiter 099
Jan. 25	Body flap on dock, Downey, Orbiter 099
Jan. 25	Payload bay doors on dock, Rockwell, Tulsa, OK, for rework, Orbiter 099
Jan. 28	Start instrumentation removal and prepare mid fuselage for modification, Palmdale, Orbiter 099
Feb. 1	Complete aft fuselage demate Palmdale, Orbiter 099 for rework on dock, Downey
Feb. 1	Elevons on dock, Grumman, NY, for rework, Orbiter 099
Feb. 1	Complete aft RCS, Phase II qualification tests, WSTF
Feb. 4	Start instrumentation removal and prepare wing for modification, Palmdale, Orbiter 099
Feb. 8	Demate forward RCS module, Palmdale, Orbiter 099
Feb. 14	Final qualification firing SRB, Thiokol, Utah
Feb. 15	Complete demate upper forward fuselage, Palmdale, Orbiter 099
Feb. 18	Complete left-hand OMS Phase II qualification test, WSTF
Feb. 20	Complete forward RCS module qualification test, WSTF
Feb. 23	Upper forward fuselage on dock, Downey, for rework, Orbiter 099

Date	Event
Feb. 28	Seventh static firing, MPTA-098, NSTL, 555 seconds (No. 2 engine planned shutdown at 520 seconds, throttle down to 70% from 100%), pogo and gimbaling tests; stub nozzles
March 3	Start detail fabrication, crew module, Orbiter 103
March 20	Eighth static firing, MPTA-098, NSTL, 539 seconds (started at 100%, two engines throttled to 70%, then up to 100%, two other engines throttled to 70%, then up to 100%; all three engines throttled to simulate 3-g mission profile, then to 70% and shutdown; simultaneous pogo and gimbal tests; stub nozzles
March 21	Forward RCS module on dock, Downey, for rework, Orbiter 099
March 31	Complete 1/4-scale model tests, Downey
April 16	Ninth static firing, MPTA-098, NSTL (No. 2 engine, 4.6 seconds, shutdown due to discharge. Overtemperature on high-pressure fuel turbo pump; No. 1 and No. 3 shutdown at 6 seconds; stub nozzles)
May 12	Pyro shock test, Orbiter 102 KSC
May 30	Complete preparation lower forward fuselage modification, Palmdale, Orbiter 099
May 30	Ninth static firing, MPTA-098, NSTL (46 seconds into run, throttled to 70% for 18 seconds, then to 100%, to 95% at 375 seconds, to 85% at 400 seconds, 83% at 430 seconds, 75% at 460 seconds; No. 3 shutdown at 480 seconds, remaining two to 60% at 490 seconds; No. 2 shutdown at 565 seconds; No. 1 shutdown by engine cutoff sensors at 575 seconds). Pogo and gimbal tests; stub nozzles
June 1	Orbiter 102, Engine 2005 SSME fired for 520 seconds, NSTL
June 1	Start fabrication / assembly wings, Orbiter 103,
June 5	Orbiter 102, Engine 2006 SSME fired for 520 seconds at NSTL
June 16	Orbiter 102, Engine 2007 SSME fired for 520 seconds at NSTL
June 20	Start fabrication lower forward fuselage, Orbiter 103
July 12	Tenth static firing, MPTA-098, NSTL, shutdown 105 seconds into firing due to burn through in Engine No. 3 fuel preburner, 102% thrust, flight nozzles
July 28	Start detail fabrication aft fuselage, Orbiter 104
Aug. 3	Complete installation SSME's, Orbiter 102, KSC
Sept. 1	Start body flap modification Downey, Orbiter 099
Sept. 29	Start assembly crew module, Orbiter 103
Oct. 1	Start fabrication/assembly mid fuselage, Orbiter 103
Oct. 1	Start fabrication/assembly mid fuselage, Orbiter 104
Oct. 9 & 10	Removal of SSME's from Orbiter 102, KSC for modifications
Oct. 11 & 12	Installation of orbital maneuvering system/ reaction control system pods, Orbiter 102, KSC
Nov. 3	Start initial systems installation crew module Downey, Orbiter 099
Nov. 3	External tank mated to solid rocket boosters in Vehicle Assembly Building, KSC, for STS-1
Nov. 3	Eleventh static firing MPTA-098, NSTL, shutdown 20 seconds into firing due to burn through in Engine No. 2 nozzle, 102% thrust, flight nozzles

Date	Event
Nov. 4	Structural integrity test aft fuselage, Orbiter 102, KSC
Nov. 5	External tank mated to solid rocket boosters at KSC
Nov. 8, 9 & 10	Reinstallation of SSME's, Orbiter 102, KSC
Nov. 10	Start assembly aft fuselage, Orbiter 103
Nov. 14	Complete modifications, Orbiter 102, KSC
Nov. 16	Complete thermal protection system installation, Orbiter 102, KSC
Nov. 21	Complete wing modification, Palmdale, Orbiter 099
Nov. 21	Complete modification lower forward fuselage, Palmdale, Orbiter 099
Nov. 24	Transfer Orbiter 102 from Orbiter Processing Facility to Vehicle Assembly Building, KSC
Nov. 26	Mating of Orbiter 102 to external tank and solid rocket boosters in Vehicle Assembly Building, KSC
Dec. 4	Space Shuttle vehicle power-up in Vehicle Assembly Building (Orbiter 102), KSC
Dec. 4	Eleventh static firing MPTA-098, NSTL stub nozzles, 591 seconds simulated flight profile, started at 100% then to 65% at 37 seconds, then to 102% at 65 seconds, then to 65% at 438 seconds. Engine No. 2 shutdown at 442 seconds, remaining two to 65% at 508 seconds, then shutdown gimbal test
Dec. 8	Start initial system installation aft fuselage, Orbiter 103
Dec. 12	Complete rework aft fuselage, Downey, Orbiter 099
Dec. 15-18	Space Shuttle interface test in Vehicle Assembly Building (Orbiter 102), KSC

Date	Event
Dec. 19	Start preparations for Space Shuttle rollout and ordnance installation (Orbiter 102), KSC
Dec. 29	Transfer Space Shuttle aboard mobile launch platform from Vehicle Assembly Building to Launch Complex 39A (Orbiter 102)
1981	
Jan. 5	Emergency egress test
Jan. 17	Twelfth static firing MPTA-098, NSTL, flight nozzles, 625 seconds, 100% thrust, simulated abort mission profile, No. 1 engine shutdown at 239 seconds, remaining two shutdown at 625 seconds, POGO and gimbal tests. External tank test without anti-geyser line to verify feasibility of eventually removing it from later external tank versions.
Jan. 22	External tank LH_2 load, KSC, Orbiter 102
Jan. 23	Auxiliary power unit confidence run, Orbiter 102, KSC, each APU serially run for two minutes
Jan. 24	External tank LO_2 load, KSC, Orbiter 102
Jan. 29	Hypergolic load, reaction control system, orbital maneuvering system, KSC, Orbiter 102
Feb. 2	Start initial system installation forward reaction control system module, Downey, Orbiter 099
Feb. 2	Met countdown demonstration test simulations, KSC, Orbiter 102
Feb. 4	Start series of countdown demonstration tests for flight readiness firing, Orbiter 102, KSC
Feb. 13	Elevons on dock, Palmdale, Orbiter 099
Feb. 20	Flight readiness firing (20 second firing of all three SSME's, Orbiter 102, KSC

Date	Event
Feb. 24, 25, & 26	Start mission verification tests, Orbiter 102, KSC
March 2	Start fabrication/assembly, payload bay doors, Orbiter 103
March 11	Launch verification tests, Orbiter 102, KSC
1981 (3rd Q)	
April 9	Conduct STS–1, Orbiter 102, launch KSC, land Edwards AFB, CA, Runway 23 dry lakebed
	Forward payload bay doors on dock, Downey, Orbiter 099
	Aft payload bay doors on dock, Downey, Orbiter 099
	Start fabrication elevons, Orbiter 104
	Start fabrication vertical stabilizer, Orbiter 103
	Start fabrication/assembly wings, Orbiter 104
	Start mate elevons, Palmdale, Orbiter 099
	Crew module on dock, Palmdale, Orbiter 099
	Upper forward fuselage on dock, Palmdale, Orbiter 099
1981 (4th Q)	
	Forward payload bay doors on dock, Palmdale, Orbiter 099
	Start fabrication crew module, Orbiter 104
	Aft payload bay doors on dock, Palmdale, Orbiter 099
	Complete body flap modification, Downey, Orbiter 099
	Body flap on dock, Palmdale, Orbiter 099
	Conduct STS-2, Orbiter 102, Launch KSC, land Edwards AFB, CA, runway 23 dry lakebed
	Aft fuselage on dock Palmdale, Orbiter 099
	Deliver modification kits for Orbiter 102, STS-3, KSC

Date	Event
1982 (1st Q)	
	Forward reaction control system module on dock, Palmdale, Orbiter 099
	Start detail fabrication lower forward fuselage, Orbiter 104
	Complete airframe modifications, Palmdale, Orbiter 099
	Start initial system installation crew module, Downey, Orbiter 103
	Start pre–combined system test power–on, Palmdale, Orbiter 099
	Start assembly aft fuselage, Orbiter 104
	Start initial system installation aft fuselage, Orbiter 103
	Conduct STS–3, Orbiter 102, launch KSC, land Edwards AFB, CA, dry lakebed, Runway 23
	Start fabrication upper forward fuselage, Orbiter 104
1982 (2nd Q)	
	Start initial system installation upper forward fuselage, Orbiter 103
	Orbital maneuvering system/reaction control system pods, on dock, Palmdale, Orbiter 099
	Mid fuselage on dock, Palmdale, Orbiter 103
	Complete pre–combined system test, Palmdale, Orbiter 099
	Start assembly crew module, Orbiter 104
	Deliver Ku–Band communication/rendezvous radar system, Orbiter 102, KSC
	Complete main propulsion test program, MPTA–098, NSTL MS
	Elevons on dock, Palmdale, Orbiter 103
	Wings on dock, Palmdale, Orbiter 103
	Start initial systems installation forward reaction control system module, Orbiter 103
	Start initial systems installation wings, Orbiter 103

Date	Event
1982 (3rd Q)	
	Conduct STS-4, Orbiter 102, launch KSC, land Edwards AFB, CA, concrete Runway 04
	Complete combined system test, Palmdale, Orbiter 099
	Payload bay doors on dock, Downey, Orbiter 103
	Lower forward fuselage on dock, Palmdale, Orbiter 103
	Complete standby period, MPTA-098, NSTL, MS
	Orbiter 102, KSC, available for modification to STS-5
	Upper forward fuselage on dock, Palmdale, Orbiter 103
	Start initial system installation payload bay door, Downey, Orbiter 103
	Start fabrication forward reaction control system module, Orbiter 104
	Complete post checkout, Palmdale, Orbiter 099
	Vertical stabilizer on dock, Palmdale, Orbiter 103
	Complete configuration inspection, rollout, Palmdale, Orbiter 099
	Orbiter 099, on dock, KSC
1982 (4th Q)	
	Start assembly payload bay doors, Orbiter 104
	Orbital maneuvering system/reaction control system pods on dock, KSC, Orbiter 099
	Start final assembly and closeout installation, Palmdale, Orbiter 103
	Aft fuselage on dock, Palmdale, Orbiter 103
	Payload bay doors on dock, Palmdale, Orbiter 103
	Complete acceptance at KSC, Orbiter 103, modifications for STS-5
	First operational flight, launch and land KSC, Orbiter 102
1983 (1st Q)	
	Start fabrication assembly vertical stabilizer, Orbiter 104
	Crew module on dock, Palmdale, Orbiter 103
	Body flap on dock, Palmdale, Orbiter 103
	Start fabrication orbital maneuvering system/reaction control system pods, Orbiter 104
	Orbital maneuvering system/reaction control system pods on dock, Palmdale, Orbiter 103
	Forward reaction control system module on dock, Palmdale, Orbiter 103
1983 (2nd Q)	
	Start initial system installation aft fuselage, Orbiter 104
	Deliver Orbiter 101 to Vandenberg AFB, CA, for facility verification
	Complete final assembly and closeout installation, Palmdale, Orbiter 103
	Start pre-combined systems test power-on, Palmdale, Orbiter 103
	Start fabrication/assembly body flap, Orbiter 104
	Start initial system installation crew module, Orbiter 104
1983 (3rd Q)	
	Mid fuselage on dock, Palmdale, Orbiter 104
	Complete pre-combined systems test, Palmdale, Orbiter 103
	Start initial systems installation upper forward fuselage, Orbiter 104

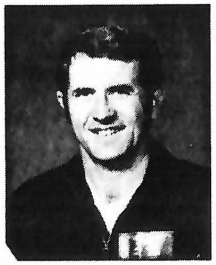
Date	Event
	Start initial systems installation lower forward fuselage, Orbiter 104
	Start initial system installation mid fuselage, Orbiter 104
	Elevons on dock, Palmdale, Orbiter 104
	Wings on dock, Palmdale, Orbiter 104
1983 (4th Q)	Start initial system installation forward reaction control system module, Orbiter 104
	Complete combined systems test, Palmdale, Orbiter 103
	Payload bay doors on dock, Downey, Orbiter 104
	Vertical stabilizer on dock, Palmdale, Orbiter 104
	Start initial system installation payload bay doors, Orbiter 104
	Lower forward fuselage on dock, Palmdale, Orbiter 104
	Complete post checkout, Palmdale, Orbiter 103
	Orbital maneuvering system/reaction control system pods on dock, KSC, FL, Orbiter 103
	Upper forward fuselage on dock, Palmdale, Orbiter 104
	Start initial system installation vertical stabilizer, Orbiter 104
	Complete configuration inspection, ready for acceptance, Palmdale, Orbiter 103
	Orbiter 103, on dock, KSC
1984 (1st Q)	Start mate lower forward fuselage, Palmdale, Orbiter 104
	Start final assembly and closeout installation, Orbiter 104
	Aft fuselage on dock, Palmdale, Orbiter 104
	Payload bay doors on dock, Palmdale, Orbiter 104
1984 (2nd Q)	Body flap on dock, Palmdale, Orbiter 104
	Orbital maneuvering system/reaction control system pods on dock, Palmdale, Orbiter 104
	Crew module on dock, Palmdale, Orbiter 104
1984 (3rd Q)	Forward reaction control system module on dock, Palmdale, Orbiter 104
	Complete final assembly and closeout installation, Palmdale, Orbiter 104
	Start pre-combined systems test power-on, Palmdale, Orbiter 104
1984 (4th Q)	Complete pre-combined systems test, Palmdale, Orbiter 104
1985 (1st Q)	Complete post checkout, Palmdale, Orbiter 104
	Orbital maneuvering system/reaction control system pods on dock, Orbiter 104, KSC
	Complete configuration inspection, ready for acceptance, Palmdale, Orbiter 104
	Orbiter 104 on dock, KSC
	Deliver Quarter-Scale models to JSC, TX

Astronaut Crews

JOHN W. YOUNG, veteran of four space flights, will be the spacecraft commander for the Space Shuttle orbiter's first flight into space. He has logged 533 hours and 33 minutes in space flight on the Gemini 3 and 10 missions and the Apollo 10 and 16 flights to the moon. A graduate of Georgia Institute of Technology in aeronautical engineering, Young entered U.S. Naval service and after a year of destroyer duty he was accepted and completed flight training. He is a graduate of the Navy's Test Pilot School and was stationed at the Naval Air Test Center for three years prior to entering the Astronaut Corps in 1962. He retired from the Navy in 1976. Young was assigned responsiblity for the Space Shuttle Branch of the Astronaut Office in 1973, and in 1975 was named as chief of the Astronaut Office. Young is a Fellow of the American Astronautical Society (AAS), Associated Fellow of the Society of Experimental Test Pilots (SETP) and the American Institute of Aeronautics and Astronautics (AIAA). He has two NASA Distinguished Service Medals, two NASA Exceptional Service Medals, the JSC Certificate of Commendation, the Navy Astronaut Wings, two Navy Distinguished Service Medals, three Navy Distinguished Flying Crosses, the Georgia Tech Distinguished Alumni Award (1965) and the Distinguished Service Alumni Award (1972), the SETP Iven C. Kincheloe Award, the AAS Flight Achievement Award, and the AIAA Haley Astronautics Award. Young was born in San Francisco, Calif., Sept. 24, 1930, is married and has two children. He is 5'9" in height, weighs 165 pounds, and has green eyes and brown hair.

ROBERT L. CRIPPEN, one of the research pilots selected for the U.S. Air Force's Manned Orbiting Laboratory, will be the spacecraft pilot for the first Space Shuttle orbiter flight into space. He has logged more than 400 hours of flying time—most of it in jet-powered aircraft—as a U.S. Navy pilot and astronaut. A graduate of the University of Texas in aerospace engineering, Crippen entered naval service (he presently is a commander) and was a carrier pilot. He completed the U.S. Air Force's Aerospace Research Pilot School at Edwards AFB and remained as an instructor until he was selected for the Manned Orbiting Laboratory program in 1966. He transferred to the NASA Astronaut Office in 1969 and was a crew member of the Skylab Medical Experiments Altitude Test—a 56-day simulation of the Skylab mission. He was a member of the support crew for Skylab 2, 3, and 4, and the ASTP mission. He has been awarded the NASA Exceptional Service Medal and the JSC Group Achievement Award. Crippen was born in Beaumont, Tex., Sept. 11, 1937, is married and has three children. He is 5'10" in height, weighs 160 pounds, and has brown hair and eyes.

JOE H. ENGLE, Shuttle spacecraft commander, earned his astronaut wings as an Air Force pilot of the X-15. Three of his 16 flights in the high-speed, high-altitude research craft exceeded 50 miles in altitude. He was spacecraft commander on the Shuttle ALT flights. Engle graduated from the University of Kansas with a BS in aeronautical engineering and entered the Air Force (he presently is a colonel). Following duty with fighter aircraft squadrons, he was accepted and graduated from the USAF Experimental Test Pilot School and the Air Force Aerospace Research Pilot School. As an Air Force test pilot, he was assigned to the X-15 and during his career has flown more than 130 different types of aircraft. He has been awarded the NASA Exceptional Service Medal, the Distinguished Flying Cross, was named the Air Force Association's Outstanding Young Officer (1964), was selected by the U.S. Junior Chamber of Commerce as one of the Ten Outstanding Young Men in America, and received the AIAA's Lawrence Speery Award. He is a member of SETP. Engle was born in Abilene, Kan., Aug. 26, 1932, is married and has two children. He is 6' in height, weighs 155 pounds, and has hazel eyes and blond hair.

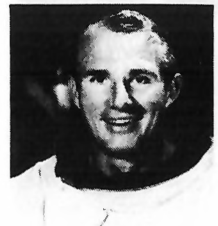


RICHARD H. TRULY, Shuttle spacecraft pilot, was an orbiter pilot during the successful Approach and Landing Test program, and as a naval pilot and astronaut has logged nearly 5000 hours in jet aircraft. He graduated from the Georgia Institute of Technology in aeronautical engineering and entered naval flight training. Following service as a carrier pilot, Truly completed the USAF Aerospace Research Pilot School at Edwards and was subsequently assigned there as an instructor. In 1965 he was selected as a research pilot on the Manned Orbiting Laboratory program and in 1969 was assigned to the NASA Astronaut Office. Truly was a member of the Skylab support crew and served in a similar capacity for the ASTP flight. He has been awarded two NASA Exceptional Service Medals. Truly was born in Fayette, Miss., Nov. 12, 1937, is married and has three children. He is 5'8" in height, weighs 150 pounds, and has brown hair and eyes.

Astronaut Crews

JACK R. LOUSMA, Shuttle spacecraft pilot, has 1427 hours and 9 minutes of space flight as crew member of Skylab 3. He also has spent 11 hours and 2 minutes in extravehicular activities. Lousma graduated from the University of Michigan with a degree in aeronautical engineering. He entered naval flight training and served as a Marine Corps pilot (he presently is a Lt. Col. in the USMC). He was selected an astronaut in 1966 and served as a support crew member for the Apollo 9, 10, and 13 missions. He was a backup pilot for ASTP. Lousma also has an aeronautical engineering degree from the U.S. Naval Postgraduate School and an honorary doctorate of astronautical science from the University of Michigan. He has the NASA Distinguished Service Medal, the JSC Certificate of Commendation, the Navy's Distinguished Service Medal and Navy Astronaut Wings, the City of Chicago Gold Medal, the Robert J. Collier Trophy, the Marine Corps Aviation Association's Exceptional Achievement Award, the FAI's V.M. Komarov Diploma, the Dr. Robert H. Goddard Memorial Trophy, the AIAA Octave Chanute Award, and the AAS Flight Achievement Award. Lousma was born in Grand Rapids, Mich., Feb. 20, 1936, is married and has three children. He is 6' in height, weighs 195 pounds, has blond hair and blue eyes.

C.G. (GORDON) FULLERTON, Shuttle spacecraft pilot, served as the Enterprise's pilot on the Approach and Landing Test program. A graduate of California Institute of Technology with both bachelor and master of science degrees in mechanical engineering, Fullerton entered active duty with the U.S. Air Force (he is presently a lieutenant colonel). He has served as both an interceptor and bomber pilot and is a graduate of the USAF Aerospace Research Pilot School. He was a test pilot at Wright-Patterson Air Force Base prior to being selected for the Manned Orbiting Laboratory program. Fullerton transferred to the Astronaut Office in 1969, and served on the support crews for Apollo 14 and 17. He is a member of SETP and an honorary member of the National World War II Glider Pilots Assn. Fullerton has been awarded the NASA Exceptional Service Award, the JSC Group Achievement Award, and the USAF Commendation and Meritorious Service medals. He was born in Rochester, N.Y., Oct. 11, 1936. Fullerton is married, and has two children. He is 6' in height, weighs 165 pounds, and has blond hair and blue eyes.



VANCE D. BRAND, Shuttle spacecraft commander, has logged 217 hours and 28 minutes in space flight as command module pilot of the Apollo-Soyuz Test Project. A graduate of the University of Colorado in business (1953) and aeronautical engineering (1960), Brand was commissioned a naval aviator and served as a Marine Corps fighter pilot until 1957. He was with the Marine Reserve and Air National Guard until 1964. He joined Lockheed Aircraft as a flight test engineer in 1960, and following completion of the Navy's Test Pilot School was assigned to Palmdale, Calif., as an experimental test pilot on the F-104. He was selected as an astronaut in 1966, and was a crew member of the Apollo 8 thermal-vacuum chamber program. He was a support crewman on Apollo 8 and 13, and was backup pilot for Apollo 15 and the Skylab 3 and 4 missions. Brand is a Fellow, American Astronautical Society, Associate Fellow of AIAA, and a member of SETP. He has the NASA Distinguished and Exceptional Service Medals, the JSC Certificate of Commendation, the Richard Gottheil Medal, the Wright Brothers International Manned Space Flight Award, the VFW National Space Award, the FAI Yuri Gagarin Gold Medal, the AIAA Special Presidental Citation and the Haley Astronautics Award, the AAS's Flight Achievement Award, and the University of Colorado's Alumnus of the Century award. Brand was born in Longmont, Colo., May 9, 1931, is married and has four children. He is 5'11" in height, and weighs 175 pounds. He has blond hair and gray eyes.

Space Shuttle Management

JOHN F. YARDLEY is associate administrator of Space Transportation Systems, NASA Headquarters. He was appointed associate administrator of Space Flight in May 1974 (now Space Transportation Systems). Before joining NASA, Mr. Yardley was vice president and general manager of the Eastern Division, McDonnel Douglas Astronautics Company, St. Louis, Mo. He joined McDonnel Aircraft Corporation in 1946 as a structural engineer. Other positions he held with McDonnell Douglas were project engineer for Mercury spacecraft design, launch operations manager for Mercury and Gemini spacecraft, and Gemini technical director. From 1968 to 1972, he was McDonnel Douglas Corporate vice president and general manager of Skylab. He received the NASA Public Service Award for his contributions to the Mercury program in 1963 and Gemini in 1966. He was awarded the Spirit of St. Louis Medal by the American Society of Mechanical Engineers and the Space Flight Award by the American Astronautical Society. He is a fellow of the American Institute for Aeronautics and Astronautics and the American Astronautical Society, and a member of the National Academy of Engineering. Mr. Yardley received a BS degree in aeronautical engineering from Iowa State College in 1943 and an MS degree from Washington University in St. Louis, Mo., in 1950.

L. MICHAEL WEEKS is deputy associate administrator of Space Transportation Systems Acquisition at NASA Headquarters, a position he has held since November 1979. In this position, he will play a key role in the development and acquisition of NASA's Space Transportation System, including the Space Shuttle, its upper stages, associated ground facilities, and equipment and system improvements. From 1975 to 1979, he was the manager of Advanced Systems Development for the Reentry and Environmental Systems Division of General Electric. Before that, he was vice president and general manager of the Missile and Space Division of LTV Corporation and was associated with the Perkin-Elmer Corporation in NASA's Space Telescope program. Weeks began his career at McDonnel Aircraft Corporation in St. Louis, Mo., on the Mercury program. He helped create the U.S. Air Force Manned Orbital Laboratory concept when he was with the Aerospace Corporation. He is an associate fellow of the American Institute of Aeronautics and Astronautics, and a member of the National Securities Association, the American Defense Preparedness Association, and the Technical Marketing Society. He received his BS degree from Iowa State University and an MS in applied mechanics from Washington University, St. Louis, Mo.

DAVID R. BRAUNSTEIM is deputy associate administrator of management in the Office of Space Transportation Systems at NASA Headquarters. He is responsible for planning and implementing management systems in both NASA Headquarters and field centers that are related to overall operation of Space Transportation Systems. Before joining NASA, he was director of program management in the Defense Advanced Rsearch Projects Agency (DARPA). Before that, he participated in the president's executive interchange program on a one-year assignment at Rockwell International's Financial Management and Marine Systems Division. From 1966 to 1974, he worked for the U.S. Navy Department in Washington, D.C. From 1965 to 1966, he was a marine engineer in Machinery Systems Division's Marine Engineering Laboratory in Annapolis, Md. He began his civil service career at the New York Naval Shipyard in 1958. He received a BSE from the University of Michigan and an MBA from George Washington University.

DR. MYRON S. MALKIN is Space Shuttle program director for NASA Headquarters. Named to this post in April 1973, he heads overall design, development, and testing of spacecraft. Dr. Malkin joined NASA after serving as Deputy Assistant Secretary of Defense for technical evaluation for almost one year. He was president of NUS Corporation, an engineering consulting firm, from 1969 to 1971. Before that, he held the position of program manager for Titan II and Minuteman III. He was general manager of the Manned Orbiting Laboratory program at General Electric from 1961 to 1969. He received BS, MS, and PhD degrees from Yale University.

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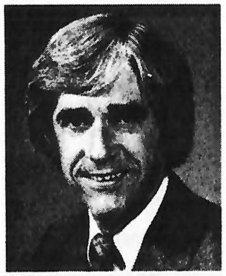
Space Shuttle Management

AARON COHEN is manager of the Space Shuttle Orbiter Project Office for Johnson Space Center. He is responsible for design, development, production, and testing of the orbiter and for providing direction to organizations within JSC, other NASA centers, and contractors to ensure orbiter technical and schedule compliance. He joined NASA at JSC in 1962 as a member of the Apollo Spacecraft Program Office and subsequently held varied executive posts on the program. He was appointed Command and Service Module (CSM) manager in 1970, directing CSM efforts on both Apollo and Skylab programs until his appointment to the Space Shuttle post in 1972. Cohen has earned two NASA Exceptional Service Awards, the NASA Certificate of Commendation, and the NASA Distinguished Service Medal. He has a BS in mechanical engineering from Texas A&M and an MS in applied mathematics from Stevens Institute of Technology.

DONALD K. (DEKE) SLAYTON is manager for the development flight tests of the Space Shuttle program at Johnson Space Center. He was previously manager of Approach and Landing Test, Space Shuttle Program Office, at JSC and was responsible for planning and implementing the test program. Slayton was docking module pilot in the joint Apollo-Soyuz Test Project in July 1975. He was director of Flight Crew Operations at JSC from November 1963 until he was restored to full flight status and eligible for manned space flights. Named as one of the original Mercury astronauts in April 1969, Slayton was removed from flight status due to a heart condition. He is a former Air Force and Air National Guard pilot and graduated from the Air Force Test Pilot School in 1955. Slayton served as a test pilot from 1956 until being named to the astronaut corps in 1959. He received a BS degree in aeronautical engineering from the University of Minnesota.

GLYNN LUNNEY is manager of the Shuttle Payload Integration and Development Program Office at Johnson Space Center. From 1976 to 1977, Lunney was temporarily assigned to NASA Washington Headquarters as deputy associate administrator for Space Flight. As Apollo spacecraft program manager, Lunney was also responsible for JSC's Skylab activities; his leadership in the Apollo-Soyuz Test Project earned him a second NASA Distinguished Service Medal. His first was awarded for service as chief of the Flight Director's Office at JSC during the Apollo program from 1968 to 1972. Lunney joined NASA in 1958 as an aerodynamics research engineer after graduating (BS in aeronautical engineering) from the University of Detroit. He was assigned to the NASA Space Test Group at Langley Research Center, Va., in 1959 and in 1962 was named head of the Mission Logic and Computer Hardware Section at JSC. He headed the Flight Dynamics Branch before being named chief of the Flight Director's Office. He also has been awarded NASA's Outstanding Performance Award and the Exceptional Service Medal. He received the AIAA Lawrence Sperry Award in 1970 and the Authur S. Fleming Award in 1974.

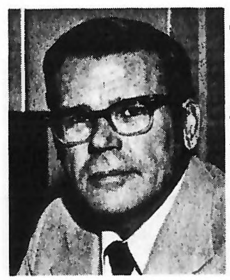
ROBERT F. THOMPSON is Space Shuttle program manager for the Johnson Space Center. He is responsible for the management and integration of all elements of the program. Thompson was appointed to this position in 1970 after serving as manager of the Skylab program through the conceptual design and development phases. He joined NASA's predecessor organization, NACA, in 1947 and was selected as one of the early members of the Space Task Group, the nucleus of JSC. He was chief of the Landing and Recovery Division for Mercury, Gemini, and early phases of the Apollo program before managing the early Skylab effort. He is recipient of NASA's Outstanding Leadership, Exceptional Service, and Distinguished Service Medals. Thompson graduated from Virginia Polytechnic Institute with a BS in aeronautical engineering.



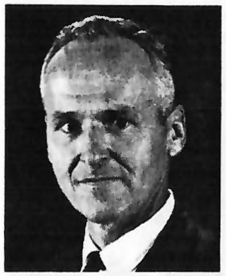
JERRY C. BOSTICK is deputy manager of Shuttle Transportation Systems Operations at Johnson Space Center, where he shares responsibilities with the manager for the planning of Space Transportation Systems operations and for the integration of all payloads and payload carriers that will fly aboard the Space Shuttle. His responsibilities include defining standard payload interfaces with the Shuttle, analyzing payload requirements vis-a-vis Shuttle performance, manifesting payloads, and managing Space Transportation Systems planning efforts associated with the Department of Defense activities. From 1976 to 1979, he was manager of the Payload Deployment and Retrieval Systems Office. From 1975 to 1976, he was chief of the Technical Planning Office at JSC. From 1974 to 1975, he was with the Office of Energy Programs. He was assistant executive secretary to the Office of the Administrator at NASA Headquarters and served as staff assistant to the associate deputy administrator from 1973 to 1974. He served as an aerospace engineer at the Mission Planning and Analysis Division of JSC from 1962 to 1965, as a Flight Dynamics Officer from 1965 to 1966, as head of Flight Dynamics Officer Section from 1966 to 1968, and as chief of the Flight Dynamics Branch from 1968 to 1973. He joined NASA in 1962 at Langley Research Center, Hampton, Va., as an aerospace engineer. He received a BS in civil engineering from Mississippi State University in 1962.

RICHARD G. SMITH is director of NASA's John F. Kennedy Space Center. He was appointed to this position September 1979. Smith became a member of the rocket research and development team at Redstone Arsenal, Ala., in June 1951. He transferred to NASA in July 1960, when the Development Operations Division of the Army Ballistic Missile Agency became the nucleus for establishing the George C. Marshall Space Flight Center. He held various assignments in the former Guidance and Control Laboratory and in the Systems Engineering Office before being appointed deputy manager and later manager of the Saturn program. In January 1974, Smith became director of Science and Engineering, where he served until he was named deputy director of the MSFC in 1974. In August 1978, he accepted a one-year assignment as deputy associate administrator of Space Transportation Systems at NASA Headquarters. He served as director of the Skylab task force appointed by the NASA administrator to represent NASA before and after the reentry of Skylab. He received the NASA Medal for Exceptional Service in 1969 and the NASA Medal for Distinguished Service in 1973 for his contributions to the Apollo lunar landing program and the Skylab program. He received his BS in electrical engineering from Auburn in 1951.

KENNETH S. KLEINKNECHT is Orbiter 102 vehicle manager for NASA's Johnson Space Center. He directs the orbiter activities at Kennedy Space Center relating to installation and modification of vehicle systems as well as the planning and scheduling of work to be performed while the vehicle is in the orbiter processing facility. Kleinknecht was named to this position in August 1979. In 1977, he was NASA Headquarters deputy associate administrator of the Space Transportation Systems (European Operations) to the European Space Agency in Paris, France. He joined NASA (formerly NACA) in 1942 and has served as project engineer, head of the Operations Engineering Section, aeronautical research scientist at NASA's Flight Research Center's Space Task Group, technical assistant to the director at JSC, manager of Project Mercury, deputy manager of the Gemini program, manager of the Apollo Command and Service Modules program, manager of the Skylab program, director of Flight Operations, and assistant manager of the orbiter project. He is a recipient of NASA's Medal for Outstanding Leadership, the John J. Montgomery Award of the National Society of Aerospace Professionals, the NASA Exceptional Service Medal, two NASA Distinguished Service Medals, and the W. Randolph Loveland II Award of the American Astronautical Society. He is a fellow of the American Astronomical Society and associate fellow of the American Institute of Aeronautics and Astronautics. He has a BS in mechanical engineering from Purdue University.



GEORGE F. PAGE is the director of Shuttle Operations at NASA's John F. Kennedy Space Center. He is responsible for the management and technical direction of the test, checkout, launch, and landing operations of the Space Shuttle at KSC. This responsibility includes the combined orbiter, external tank, and solid rocket boosters as they are assembled and tested for launch. Since joining NASA in June 1963 as a spacecraft test conductor for the Gemini program, Page has served as chief spacecraft test conductor for Gemini and Apollo launch operations, chief of Spacecraft Operations Division for Apollo, Skylab, and Apollo-Soyuz Test Project (ASTP) launch operations, director of the Expendable Vehicle Operations Directorate, and director of Cargo Operations. Before joining NASA, he served for five years as a launch operations engineer with General Dynamics Corporation and for six years as a flight test engineer with Westinghouse Electric Corporation. He has received two NASA Exceptional Service Medals for his part in the Apollo 8 and 11 operations and NASA's Distinguished Service Medal for his role in the 1975 ASTP operations. He received a BS degree in aeronautical engineering from Pennsylvania State University in 1952.

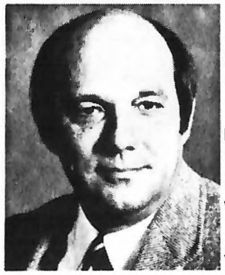


DR. ROBERT H. GRAY is manager of the Space Shuttle Projects Office at Kennedy Space Center, a position he has held since 1973. He is responsible for operations planning, facilities preparation, and ground equipment acquisition for launch, landing, and refurbishment of the Space Shuttle Transportation System, which includes the Space Shuttle, Spacelab, upper stages, and other Shuttle payloads. Earlier, Dr. Gray was KSC deputy director of Launch Operations and director of Unmanned Launch Operations, directing more flights (178) than any engineer in the free world. He joined NASA in 1958 after three years as the Vanguard launch director and deputy manager of the Vanguard Group at Cape Canaveral for the Naval Research Laboratory. Gray was named chief of the Goddard Space Flight Center Field Projects Branch in 1959, a post he held until joining the KSC organization in 1965. Honors accorded Gray include the Navy's Outstanding Performance Award for the Vanguard program and, from NASA, the Distinguished Service Award and Exceptional Service Medal. Gray graduated from Allegheny College, Pa., with a BS in physics; he received an honorary doctorate of science from Allegheny in 1968.

GEORGE B. HARDY is manager of the Solid Rocket Booster project, Space Shuttle program, for Marshall Space Flight Center. He served earlier as manager of Program Engineering and Integration for Skylab, assistant manager of the Saturn IB Launch Vehicle project, and deputy project manager for the Saturn I-IB project. Hardy began his professional career in 1952 with E.I. du Pont in Georgia; he moved to the Redstone Arsenal in 1958 and transferred to MSFC in 1962 as a project engineer. He graduated from the Georgia Institute of Technology in 1952 with a BS in civil engineering.

ROBERT E. LINDSTROM has been manager of the Shuttle Projects Office at Marshall Space Flight Center since March 1974, after serving as deputy manager for the preceding two years. From 1970 to 1972, he was deputy director of MSFC's Process Engineering Laboratory. Before 1960, he was with the Army Ballistic Missile Agency as a Saturn project engineer and as project engineer for the Jupiter C vehicle that launched Explorer I. He joined MSFC in 1960 as manager of the Saturn I-IB program. Lindstrom left government employment in 1963 to serve in top posts in industry but rejoined MSFC in 1970. He holds numerous awards, including NASA's Exceptional Service Medal and the Director's Commendation Certificate. He received a BS in ceramic engineering from the University of Illinois.

JAMES B. ODOM is manager of the External Tank project, Space Shuttle program, at Marshall Space Flight Center. Odom began his professional career in 1955 with Chemstrand Corporation, Decatur, Ala. He moved in 1956 to the Army Ballistic Missile Agency and in 1959 joined the organization that became MSFC in 1960. He has been associated with earth satellite programs, lunar unmanned probes, and the Apollo program. Odom graduated from Auburn University with a BS in mechanical engineering in 1955.



JAMES R. (BOB) THOMPSON JR. is manager of the Space Shuttle Main Engine project at Marshall Space Flight Center. He served earlier as chief of MSFC's Man/Systems Integration Branch Astronautics Laboratory. Thompson joined the propulsion research development team at MSFC in 1963, where he was responsible for component design and performance analysis of the engine system on Saturn launch vehicles. He is a graduate of Georgia Institute of Technology (1958) and the University of Florida (1963) with a BS in aeronautical engineering and an MS in mechanical engineering.

OTHA C. JEAN is manager of the Spacelab Payloads Project Office at NASA's Marshall Space Flight Center. He was appointed to this position in February 1976. He is responsible for defining and managing the first two Spacelab payloads to be carried into earth orbit. He had previously served as deputy director of the center's Program Development Directorate. He joined the rocket research and development team at Redstone Arsenal in 1957 and transferred to NASA's MSFC in July 1960, where he worked in the Aero-Astrodynamics Laboratory as its deputy director for five years. He graduated from Middle Tennessee University with a BS in mathematics in 1950 and an MS in education in 1951.

JOHN W. THOMAS is manager of the Spacelab Program Office at NASA's Marshall Space Flight Center (MSFC). He was appointed to this position in December, 1980, after serving for over a year as deputy manager of this office. Thomas began his professional career in 1960 as an engineer in the Field Support Division of the Army Ballistic Missile Agency at Redstone Arsenal, transferring to NASA in December 1961. At MSFC, Thomas served in various engineering and technical management positions with increasing responsibilities. He was involved in the development and certification of the H-1 rocket engine and in mission planning for the Skylab space station, and, in 1974, he was appointed project engineer for the structural and mechanical system of Spacelab. In 1979, he became deputy chief of the Spacelab Systems Division in the Systems Analysis and Integration Laboratory, where he served until he was appointed chief engineer for the Spacelab Program in 1976. He subsequently served as deputy manager of the Spacelab Program Office before assuming his present duties. Thomas received the NASA Exceptional Service Award in 1974 for his contributions to the Skylab Program. He is a registered professional engineer in the State of Alabama, having received a BS in mechanical engineering from Auburn University, Auburn, Alabama in 1960.

DR. FRIDTJOF A. SPEER has been manager of the Space Telescope Project Office at NASA's Marshall Space Flight Center since February 1980. He was previously the manager of the Space Sciences Project Office, which had management responsibility for the high-energy astronomy observatory (HEAO) and the gravitational red-shift space probe experiment. He served from March 1955 to June 1960 as chief of the Flight Evaluation Branch of the Guided Missile Development Division and the Development Operations Division of the Army Ballistic Missile Agency at Redstone Arsenal, Ala. He transferred to NASA in July 1960, when the Development Operations Division became the nucleus for the George C. Marshall Space Flight Center. Dr. Speer directed the Flight Evaluations and Operations Studies Division of the Aero-Astrodynamics Laboratory, then became manager of the Mission perations Office in charge of the center's participation in launch and flight operations for the Saturn launch vehicle during the Apollo program. He received NASA's Exceptional Service Medals in January and September 1969 for his work on the Apollo 8 and 11 programs. Dr Speer received the Holger N. Toftoy Award from the Alabama section of the American Institute of Aeronautics and Astronautics in February 1978 for his outstanding management of the HEAO program. In July 1978, he was awarded the NASA Outstanding Leadership Medal by the NASA administrator for the HEAO-1 program. In October 1979, the American Astronomical Society presented him the W. Randolph Lovelance II Award for his significant contribution to space science and technology. Dr. Speer is an associate fellow of the American Institute of Aeronautics and Astronautics. He received an MA and PhD in physics from the Technical University in Berlin. A native of Germany, he became a U.S. citizen on November 2, 1960.

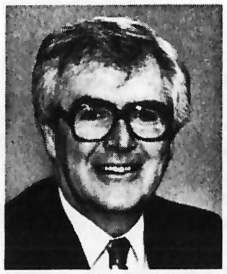
Rockwell

Space Shuttle Management

GEORGE W. JEFFS has been president of North American Aerospace Operations since 1978. He has led the development, design, engineering, and construction of the Space Shuttle orbiter and has directed the integration of the Space Shuttle vehicle. In April of 1976, he was given the additional responsibility for the major launch vehicle propulsion engines, advanced space engines, solid rockets, water-jet engines, and laser programs at Rockwell's Rocketdyne Division. Previously, he was Space Division president, responsible for the Space Shuttle program and the Navstar Global Positioning System. While serving as Space Division vice president and Apollo program manager, Mr. Jeffs directed nine Apollo lunar missions, three Skylab missions, and the joint U.S.A.-U.S.S.R. Apollo-Soyuz test project. Mr. Jeffs is a member of the National Academy of Engineering and has received NASA's Distinguished Service medal for his contributions to the space program. He has received numerous other NASA awards and presidential and congressional commendations. He is a fellow of both the American Institute of Aeronautics and Astronautics and the American Astronomical Society. He is also a recipient of the National Management Association's Golden Knight of Management. Mr. Jeffs received a BS in aeronautical engineering from the University of Washington in 1945 and an MS in aeronautical engineering in 1947.

CHARLES H. FELTZ is president of the Space Transportation System Development and Production Division. He is responsible for the company's design, development, fabrication, and test activities for the Space Shuttle program, as well as the program's field site operations. He is also responsible for orbiter subcontractor management and integration operations. He previously served as vice president and assistant to the president for Rockwell's North American Aerospace Operations and was responsible for technical management of the corporation's Space Systems Group, Rocketdyne Division, Energy Systems Group, and Aircraft Group. Between 1974 and 1976, he served as vice president and technical assistant to the Space Division president and chaired the government-industry Wide Body Advisory Group, which assessed the design of the Space Shuttle orbiter. In addition, he was responsible for Space Shuttle hardware development and generating a balanced test program for the Space Shuttle. From 1970 to 1974, he was the vice president and chief program engineer for the Space Division. In 1961, Mr. Feltz joined the Apollo program as chief program engineer and was later named deputy program manager in charge of the early design studies for the Apollo lunar landing configuration. He later directed the development, fabrication, test, and operation of the Apollo Command and Service Module program. Mr. Feltz was involved in the definitive work on the P-51 and B-25 aircraft and in the development of the B-45, F-86, and the F-100. He became chief engineer and program manager of the X-15 in 1955. He has served on the Panel on Space Vehicles for the NASA Research and Technology Advisory Council since 1974, was presented the Institute for the Advancement of Engineering Outstanding Engineer Merit Award in 1978, was awarded NASA's Johnson Space Center Certificate of Appreciation in 1969, and was presented NASA's Public Service Award in 1969. Mr. Feltz has been employed by Rockwell International and its predecessor, North American Aviation, for the past 40 years.

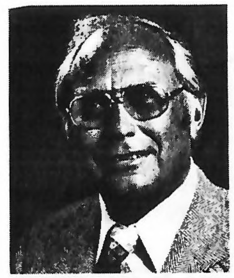
RICHARD SCHWARTZ is executive vice president for the Space Transporation System Development and Production Division. He provides technical expertise and leadership for engineering, design, fabrication, and test requirements of the space vehicle systems. From 1974 to 1979, he served as vice president and program manager of the Navstar Global Positioning System at the Space Division. In 1973, he was appointed vice president and program manager of Space Systems and Applications, responsible for division efforts on advanced programs ranging from manned and unmanned spacecraft projects and studies to work in the insulation and energy fields. Before this, Schwartz was the chief engineer and program vice president for the Saturn S-II in charge of design, development, and evaluation of the S-II stages. He also played a key role in the application technique and repair of both honeycomb and spray-on foam insulation. During the Apollo program, he was involved with integration testing of the early J-2 engines and was instrumental in the design of the S-II pogo suppression system. He later participated with NASA in analyzing alternative Space Shuttle configurations to reduce pogo effects.



WILLIAM C. STRATHERN is Vice President and General Manager of Space Operations and Satellite Systems Division. His most recent position was Vice President of Business Development and Strategic Planning at Rockwell's Aerospace Operations. Before that, he was Executive Vice President of Rockwell's Space Systems Group. Strathern has held numerous executive positions with Rockwell in the more than 25 years he has been with the company. In 1973, he was President of Program Management and Marketing of the Government Telecommunications Division of Rockwell in Dallas, Texas; and in 1977, he was Vice President and General Manager of the Collins Government Avionics Division in Cedar Rapids, Iowa. Strathern joined the Collins division in 1955 following Air Force service during the Korean War, in which he was an electronics instructor at airborne and ground communications and navigation schools. First, he was an avionics field service engineer on Strategic Air Command programs and was later named Manager of Field Operations. In 1961, he was appointed Sales Manager for Government Marketing in Washington, D.C. Later, he became Director of Government Field Representation, responsible for all Rockwell Collins field representatives assigned to government projects. In 1968, Strathern was appointed Vice President of Government Sales and in 1972 was named Vice President of Government Operations in Washington, D.C. Strathern holds membership in the Institute of Electrical and Electronics Engineers, the National Contract Management Association, and the Radio Club of America. He belongs to the Air Force Association, the Association of the U.S. Army, the National Aviation Club, and the American Marketing Association.

Rockwell
Space Shuttle Management

SEYMOUR (SY) RUBENSTEIN is vice president and program manager for the Space Transportation System Development and Production Division. Appointed to this position in June 1980, Rubenstein is responsible for the orbiter development program. He was previously vice president and deputy program manager from 1979 to 1980. He became vice president of engineering and chief engineer in 1977 after serving as associate chief program engineer for Space Shuttle avionics. Rubenstein transferred to the division from Rockwell's Autonetics Group in 1973 as director of Shuttle Avionics Systems. He received a BS in electrical engineering from the Massachusetts Institute of Technology and an MS in electrical engineering from New York University. He received an MBA from California State University at Fullerton. Rubenstein joined the Autonetics Division in 1961. In 1966 he joined the Strike Avionics Systems of Autonetics where he became chief of Digital Systems and later chief of Guidance Navigation Control. In 1970, he was named program manager for Information Systems of the company's NARISCO group.

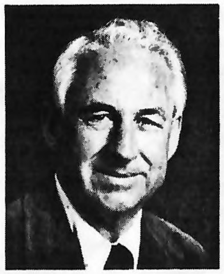


EDWARD SMITH has been vice president of engineering for the Space Transportation System Development and Production Division, responsible for the engineering of the orbiter vehicles, since June 1977. From April 1976 to June 1977, he was Space Shuttle orbiter vice president and program manager. From April 1974 to 1976, he was vice president of Engineering, serving as Space Shuttle chief engineer responsible for division wide engineering. He was Shuttle chief engineer from 1972 to 1974. With the company for more than 20 years, Smith came to the Space Division in 1966 as a senior project engineer on the Apollo program; he was named Apollo chief engineer in 1970. He received the NASA Public Service Award and the Certificate of Appreciation for his contributions to the Apollo program. Smith graduated from UCLA with a BS in mechanical engineering.

THOMAS O'MALLEY is vice president of Space Transportation System Development and Production at Kennedy Space Center Launch Operations. He is responsible for prelaunch and postlaunch testing and Space Shuttle system preparations during test and operational flights. He was named to this assignment in January 1970, after serving more than two years as the division's director of Apollo Spacecraft Operations at KSC. He joined the company in 1967, following nine years with General Dynamics in assignments including operations manager, for all space launchings at Cape Canaveral Air Force Station's Eastern Test Range. O'Malley has received two NASA Public Service Awards for work on the Apollo and Skylab programs. He has a BS in mechanical engineering from Newark College of Engineering.

ROLAND L. BENNER is vice president and program manager of Shuttle System Integration for the Production Division, a position he has held since January 1980. He was previously program manager for Space Shuttle System Integration from 1978 to 1980 and director of Space Shuttle System Integration from 1973 to 1978. He was assistant chief engineer for Space Shuttle Program Engineering from 1972 to 1973. Before that, Benner held the positions of assistant chief engineer, assistant program manager (spacecraft manager), and chief program engineer on Apollo. Benner also served as assistant project engineer and project engineer on the X-14 program. He received his BS in 1949 from Villanova University, Philadelphia, Pa.

Rockwell
Space Shuttle Management



ALBERT MARTIN became chief program engineer for the Space Transportation System Development and Production Division in January 1980. Martin is responsible for directing the engineering activities of the orbiter design, development, test, and evaluation phase. He was chief engineer for Space Shuttle Integration from 1978 to 1980. Previously, he served as chief project engineer for Space Shuttle Integration from 1972 to 1978. During the Apollo lunar landing program, Martin was director of S-II Launch Operations at KSC, where he received the NASA Public Service Award. In 1971, he returned to California and became chief engineer of Rockwell's Planetary Programs. Martin joined Rockwell in 1951 as a test engineer following graduation from the University of Texas, where he received both BS and MS degrees in aeronautical engineering.

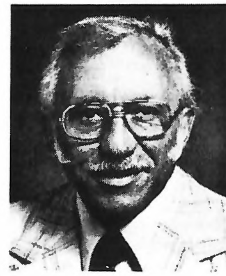
ROY BEAT is division director of Material and Subcontract Management for the Space Transportation System Development and Production Division, a position he has held since December 1978. Beat manages all activities necessary to procure, store, and control all material required to produce an end product within cost and schedule limitations. In addition, he has been director of Material, manager of Contracts/Pricing for Finance and Administration, director of Business Operations, manager of Contracts/Pricing on the Space Shuttle program, director of Business Management, director of Central Procurement, and director of Shuttle Major Subcontract Management. Beat attended both the University of Detroit and the University of Wichita. He is also a member of the National Management Association and the Purchasing Management Association.

C. E. (ED) KINDELBERGER is division director of Assurance Management for the Space Transportation System Development and Production Division. He has worked with Rockwell for more than 40 years in various departments. During World War II, Kindelberger worked on the AT-6 and B-24 programs. He later served as group engineer on the P-51 series aircraft, including the two-place TP-51. He also served as group engineer on the F-86 series aircraft, responsible for mockup fabrication and engineering-manufacturing procurement liaison. He directed engineering efforts as project engineer on the aircraft and latter portion of the F-107 fighter contract, and was assigned to the Design Support Group on the B-70 program. Kindelberger became manager of the Spacecraft Trainer during the Paraglider program in 1963. Later, he became manager of Apollo Quality Control until 1966, when he was appointed director of Quality and Reliability Assurance on the Apollo CSM and ASTP programs.

DANIEL BROWN has been division director of Production Operations and Test for the Space Transportation System Development and Production Division since 1975. He is responsible for production operations, test, and checkout of the spacecraft, and related facilities at Downey and Palmdale, as well as related traffic transportation activities. He has held key roles in building spacecraft, missiles, and aircraft. Brown joined Rockwell in 1969 as director of Manufacturing Operations and Test. Previously, he was with Martin-Marietta Corporation, where he was a quality control engineer, system propulsion engineer, general supervisor of Manufacturing Installation and Checkout, and chief of Program and Production Control. During his service with Martin-Marietta, Brown held manufacturing assignments in Denver, Colo., Orlando, Fla., Vandenberg AFB, Ca., and Little Rock, Ark.

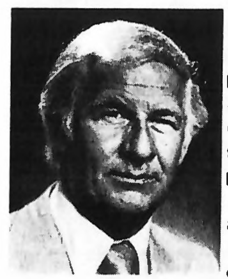
Rockwell
Space Shuttle Management

BRUCE A. GERSTNER is chief program engineer of Space Shuttle System Integration at the Space Transportation System Development and Production Division. Appointed in January 1980, Gerstner is the administrator of system integration contracts in engineering. Previously, he was chief project engineer of Integration from 1978 to 1980 and project manager in System Engineering from 1975 to 1978. He attended the College of the City University of New York, where he received a BS degree in mechanical engineering in 1948.



DONALD H. CARTER is director of Material for the Space Operations and Satellite Systems Division. Appointed to this position in June 1979, he manages all material activities. Carter previously held key positions in material functions involving the Apollo and Space Shuttle programs. Carter attended UCLA, where he received a BA degree in accounting.

RAY F. LARSON chief program engineer for the Space Transportation System Development and Production Division since 1980, is responsible for directing the design and supporting the fabrication of Increment 3 production orbiter vehicles, including the planning, organization, and integration of engineering activities. Larson served as vice president of Satellite Systems Advanced Programs from 1979 to 1980 and as vice president of Space Advanced Programs from 1976 to 1978. He was vice president and program engineer on the ASTP Command Service Module program from 1974 to 1976 after serving more than four years as the division's vice president for Assurance Management. Larson joined Space Division in 1954 and worked on missle programs and study projects until being assigned to the Apollo program in 1962. He later became an Apollo assistant program manager, responsible for management of division work on the spacecraft for the Apollo 9, 12, 13, 14, and 15 missions. He received the NASA Public Service Award in 1975 for his work on the ASTP program. He also serves as a consultant and chairman of the Subcommittee on Space Systems, NASA Space Systems Technology Advisory Committee, and as a consultant to the National Academy of Sciences on the Naval Studies Board Space Panel. Larson attended Wayne State College, Neb., and earned a BS degree at the U.S. Naval Academy. He also took engineering courses at USC and did postgraduate work at the California Institue of Technology.

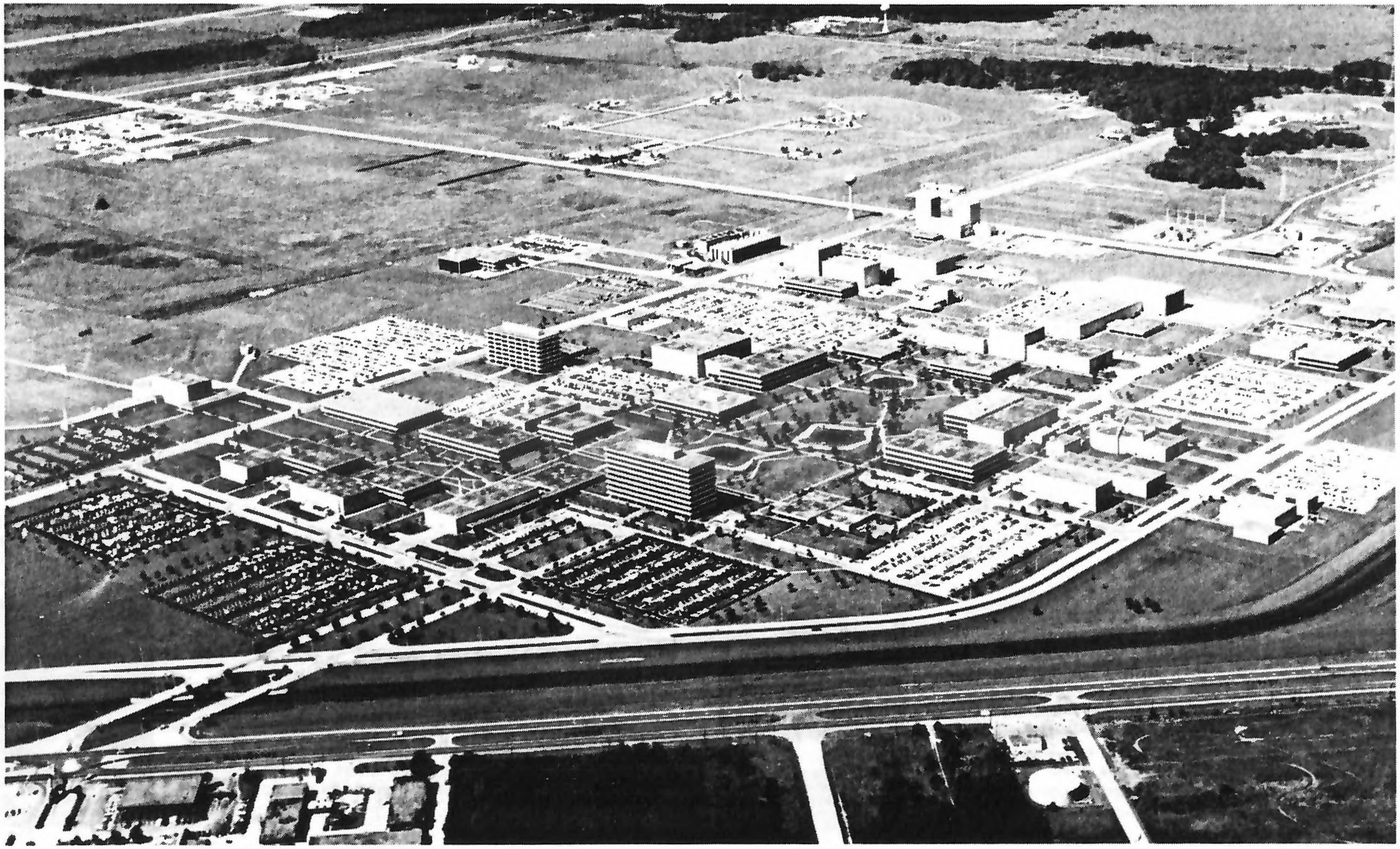


HAROLD E. (HAL) EMIGH is programs director for Rockwell's Space Operations and Satellite Systems Division. Named to this post in 1977, Emigh has responsibility for program direction of all Rockwell Space Transportation System payload and cargo integration and is division manager of this business segment. He joined Rockwell in 1958 and has served as project manager for the Paraglider program and as project manager for Advanced Reentry/Recovery Systems. A graduate of Northrop University with a BS degree in aeronautical engineering, Emigh served twice with the U.S. Navy in communications during World War II and the Korean conflict.

DANIEL H. JENSEN is Orbiter 099 manager for the Space Transportation System Development and Production Division, responsible for the production and delivery of spacecraft. In addition, he interfaces with the NASA flight operations functions for STS-1. He was named to this position in November 1979. Previously, he was director of Mated Vertical Ground Vibration Test Operations. He also served as director of the Approach and Landing Test program. Jensen joined Rockwell in 1962 as a senior test project engineer. Since then, he was spacecraft manager at Rockwell's Launch Operations, Fla., for the Apollo lunar landing flights and manager of Acceptance Test and Checkout on the Skylab and Apollo-Soyuz Test Project spacecraft. Jensen has received NASA achievement awards for his contributions to the Apollo and Skylab programs. He has a BS degree in aerodynamics from the University of Michigan.

PAUL A. BARKER is Director Material, responsible for the direction of Material Procurement. This includes all purchases required to be made except for major subcontracts. He has held this position sine 1974. Prior to this position, he held the positions of Manager, General Supervisor, Supervisor, and Subcontract Administrator. Prior to joining Rockwell International, he was assistant project manager at Pratt and Whitney Aircraft and subcontractor administrator at Raytheon.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (NASA)

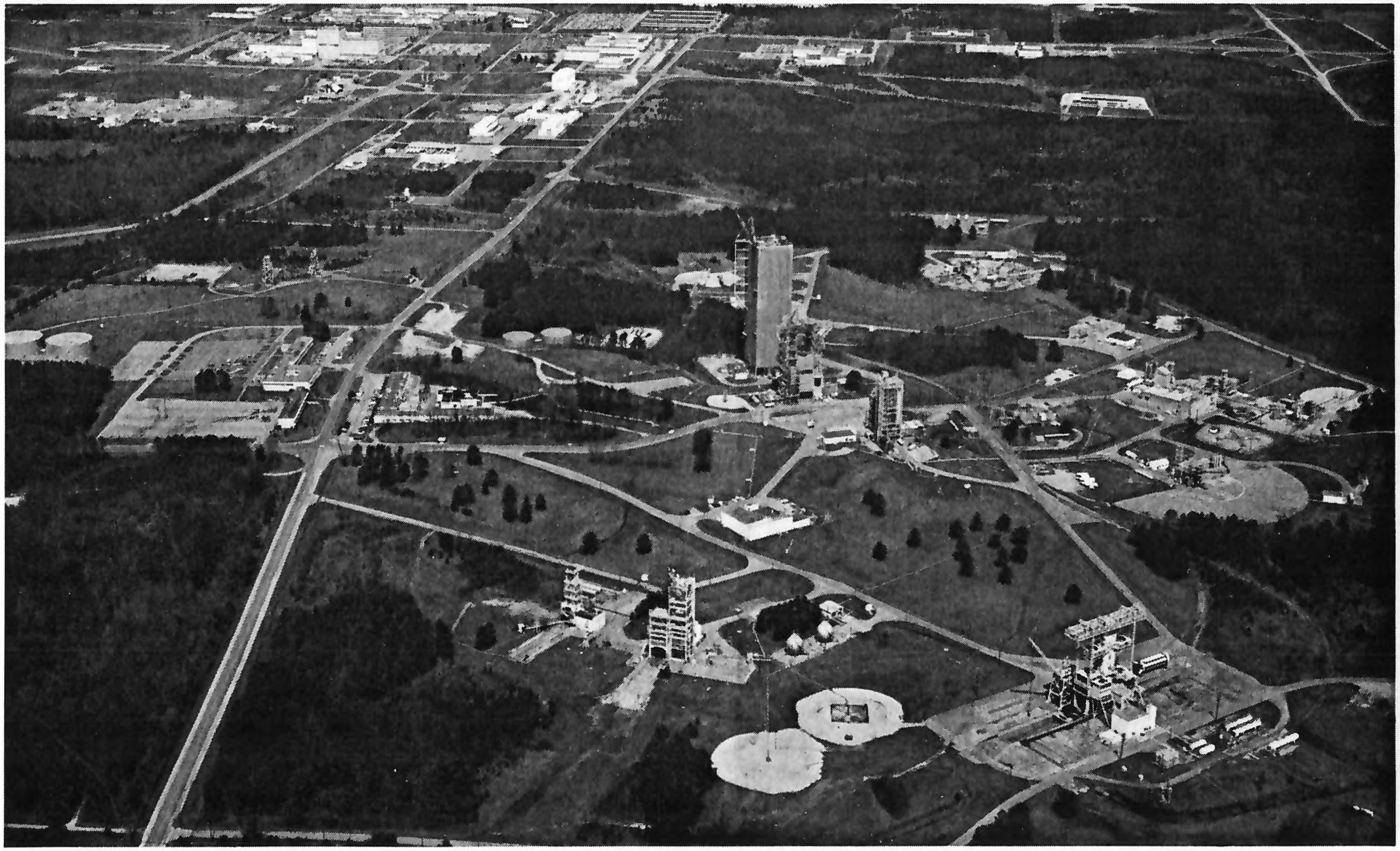


Lyndon B. Johnson Space Center (JSC), Houston, TX

The Johnson Space Center is the lead center for overall Space Shuttle program development and is responsible for program control, overall systems engineering, and Space Shuttle systems integration and also is responsible for design and development of the orbiter vehicle and the Shuttle carrier aircraft.

The Johnson Space Center will also provide the facilities for the Shuttle Avionics Integration Laboratory (SAIL), thermal-vacuum tests, thermal protection system (TPS) tests, simulators, trainers, and mockups.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (NASA)

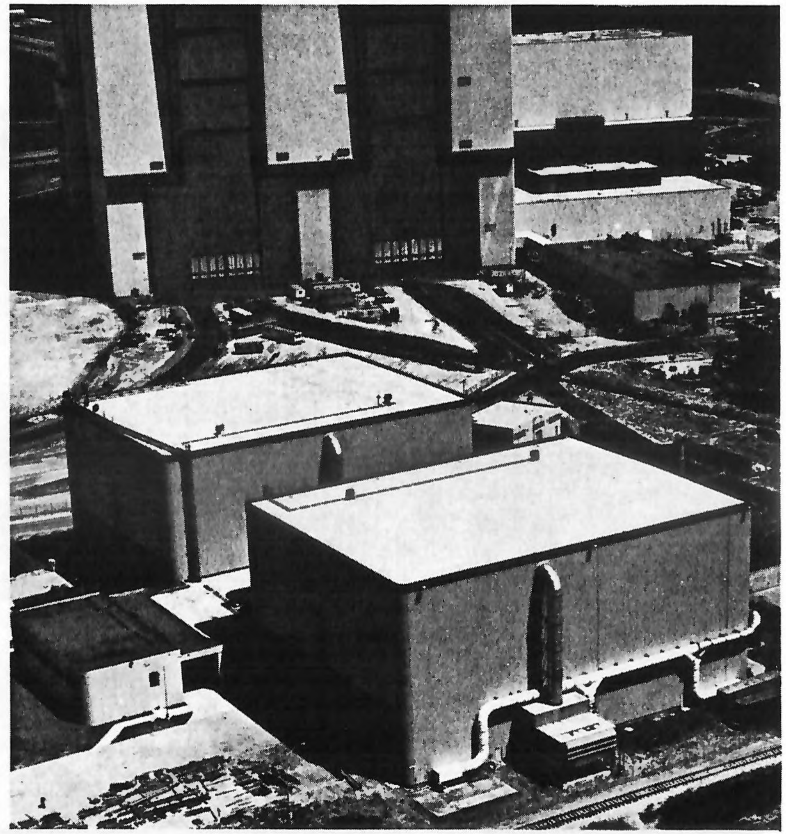


Marshall Space Flight Center (MSFC), Huntsville, AL

The Marshall Space Flight Center is responsible for managing the development of the solid rocket boosters, the Space Shuttle main engines, the external tank, and Spacelab.

The mated (Orbiter 101, external tank and solid rocket booster) vertical ground vibration test was conducted at the Marshall Space Flight Center.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (NASA)



Kennedy Space Center (KSC), FL

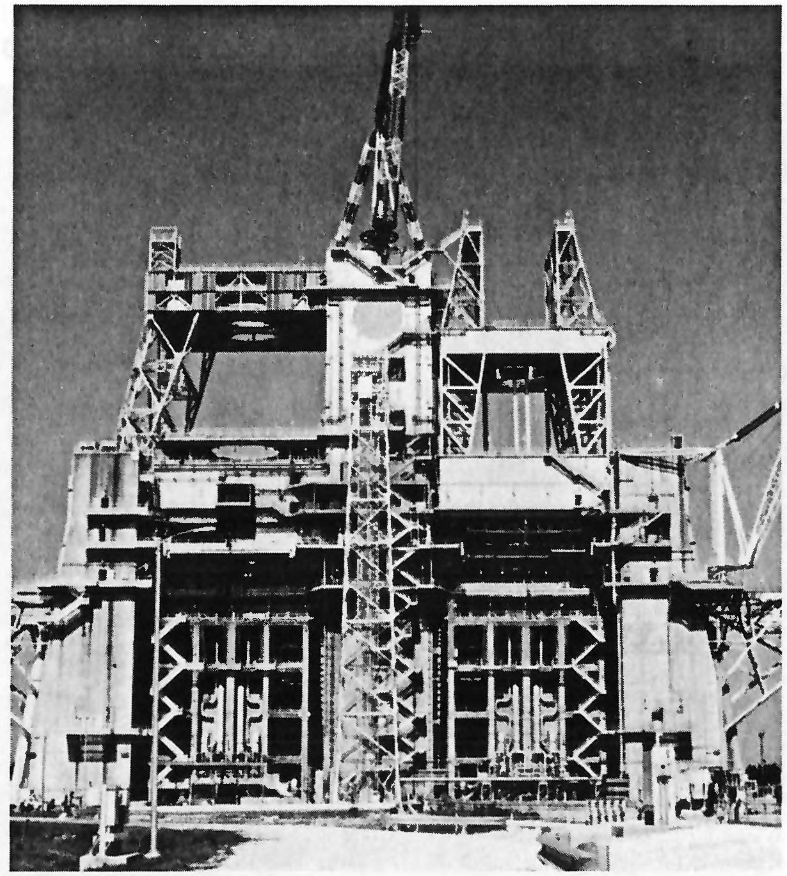
The Kennedy Space Center is responsible for the design and development of the Space Shuttle launch, landing, and refurbishment facilities and launch operations for launches in easterly azimuths (equatorial orbits).

The Vehicle Assembly Building (VAB), the Launch Control Center, the mobile launcher platforms, and Launch Pad 39A were modified to support the Space Shuttle vehicle and Launch Pad 39B will be modified to support it. The orbiter is ferried by the Shuttle carrier aircraft to KSC, the external tank is shipped from Michoud, LA, by barge to KSC and the solid rocket boosters are shipped via rail from Thiokol in Utah. The Space Shuttle is mated in the VAB and transported via the mobile launcher platform to the launch pad.

The new Orbiter Processing Facility (OPF), located near the northwest corner of the VAB, will accommodate and process two orbiters simultaneously. A new landing facility for the orbiter is located approximately one-and-five-tenths mile north and west of the VAB. The landing runway is 4572 meters (15,000 feet) long and 91 meters (300 feet) wide. The runway designation is 33 from southeast to northwest and 15 from northwest to southeast.

The Space Shuttle flight readiness firing test and initial development fligh· ·est operations will be conducted at KSC.

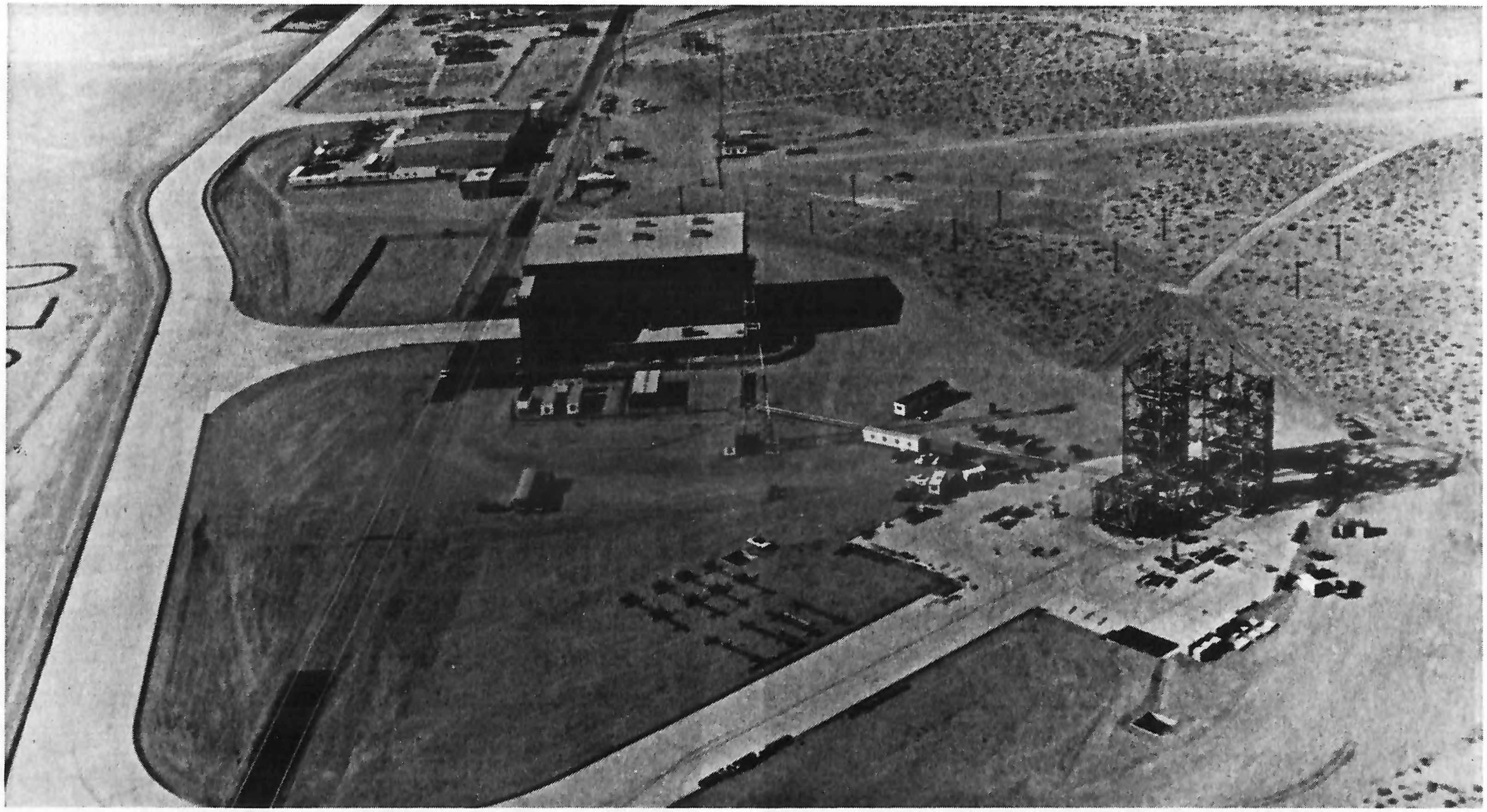
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (NASA)



National Space Technology Laboratory (NSTL), Bay St. Louis, MS .

The National Space Technology Laboratory is utilized for the initial Space Shuttle main engine tests as well as the main propulsion test article (MPTA), external tank, and Space Shuttle main engine cluster test firings.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (NASA)



Hugh L. Dryden Flight Research Center (DFRC), Edwards AFB, CA

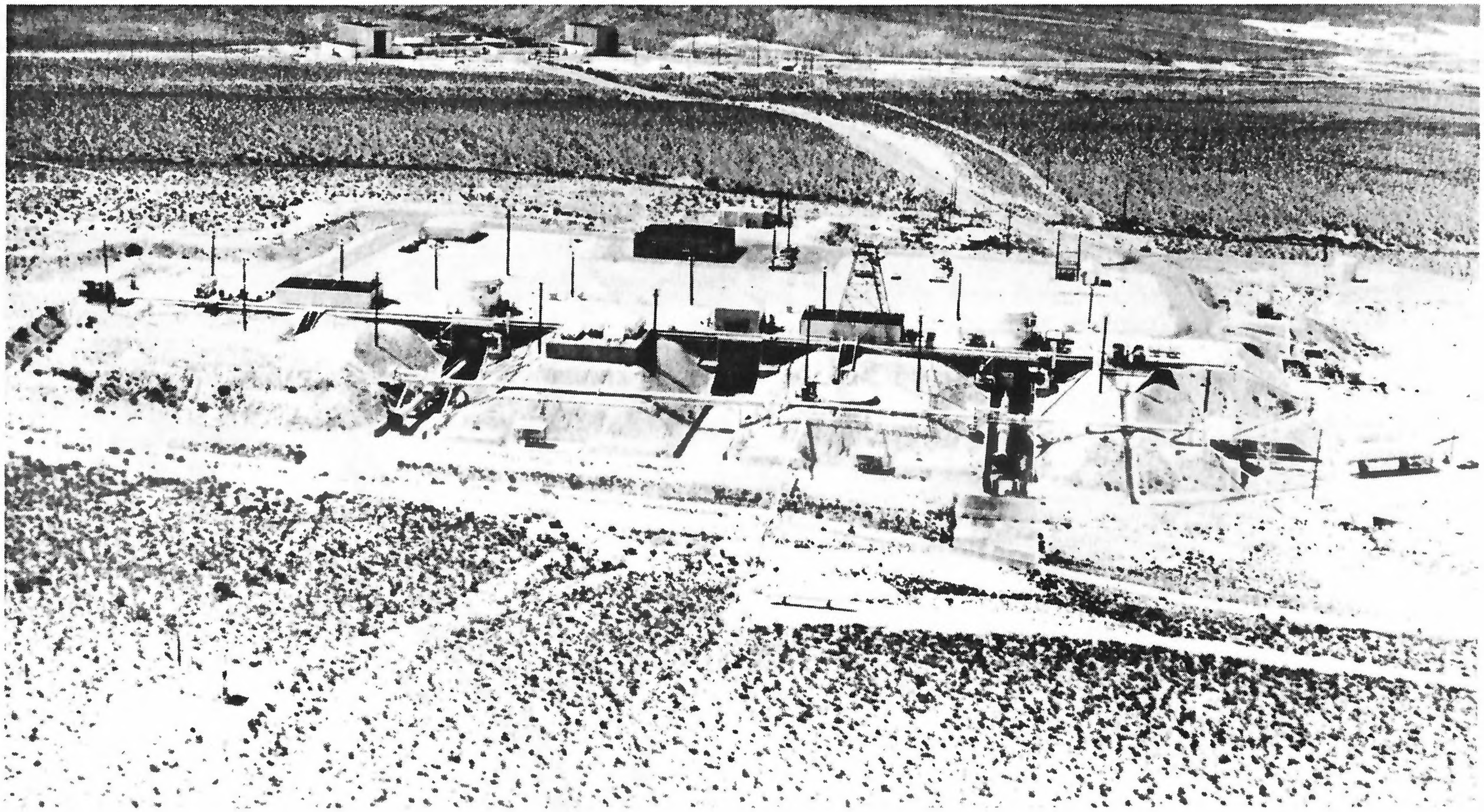
The Dryden Flight Research Center was used to support the Orbiter 101 approach and landing test (ALT) program. The structure in the right foreground is used to mate and demate the orbiter from the Shuttle carrier aircraft. DFRC also will be used as the landing site for the initial development flight tests of Orbiter 102 as well as an alternate landing site in future missions.

UNITED STATES AIR FORCE

Holloman Air Force Base (HAFB), NM

Holloman Air Force Base was used for the escape system sled tests. These tests involved the orbiter ejection panels and ejection seats that were provided for the commander and pilot in Orbiter 101 for the approach and landing test program and the initial development flight tests of Orbiter 102.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (NASA)

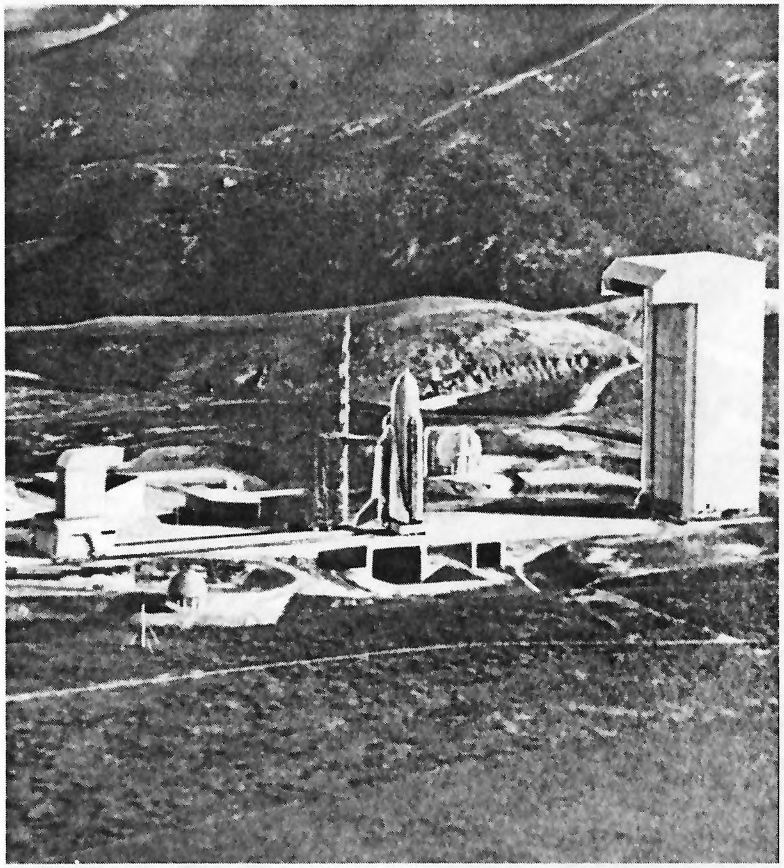


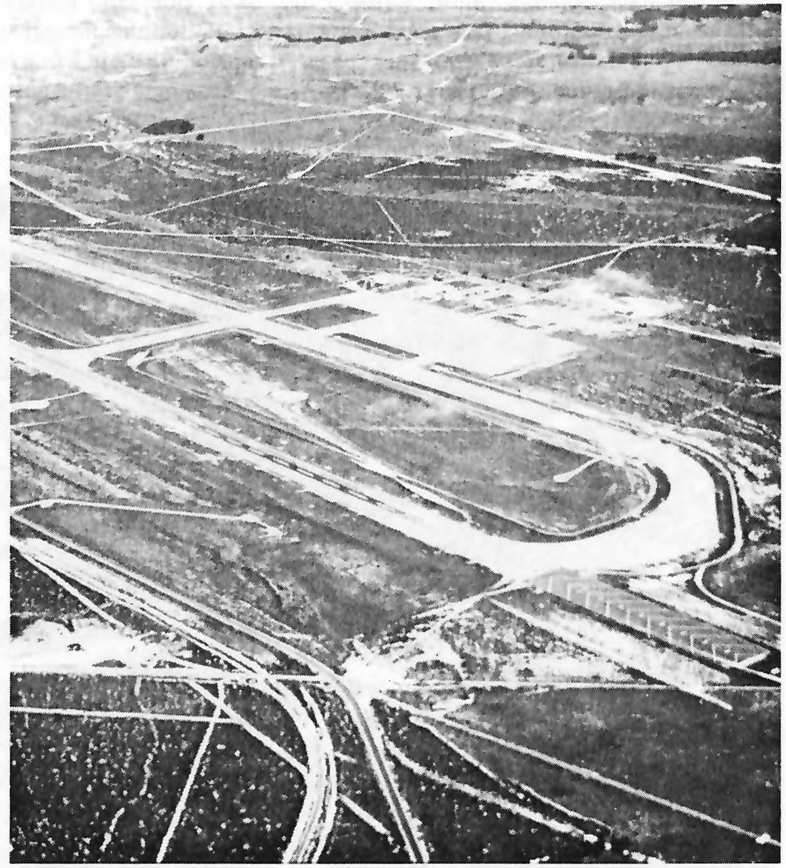
White Sands Test Facility (WSTF), NM

The NASA White Sands Test Facility is used for the testing of the orbital maneuvering system/reaction control system engines and systems and the forward reaction control system module engines and systems.

The Northrup Strip at White Sands may be used as an alternate landing site.

UNITED STATES AIR FORCE





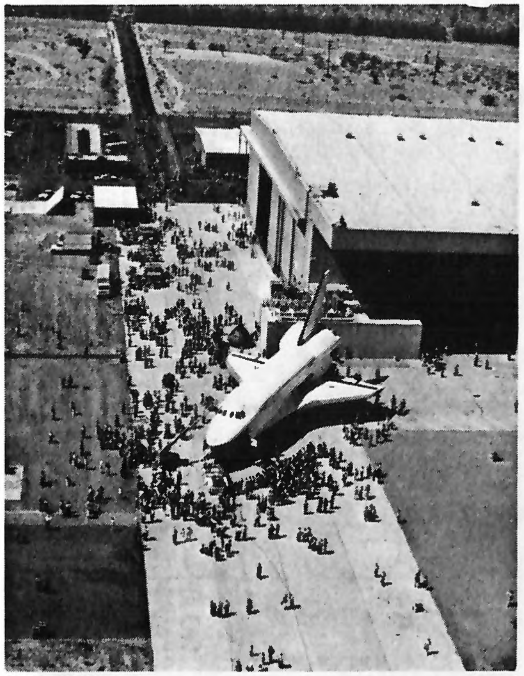
Vandenberg Air Force Base (VAFB), CA

Vandenberg Air Force Base will be used for Space Shuttle southern launches (polar and near-polar azimuths). The Space Shuttle facilities at Vandenberg AFB will be provided by the Department of Defense.

Rockwell International Corporation

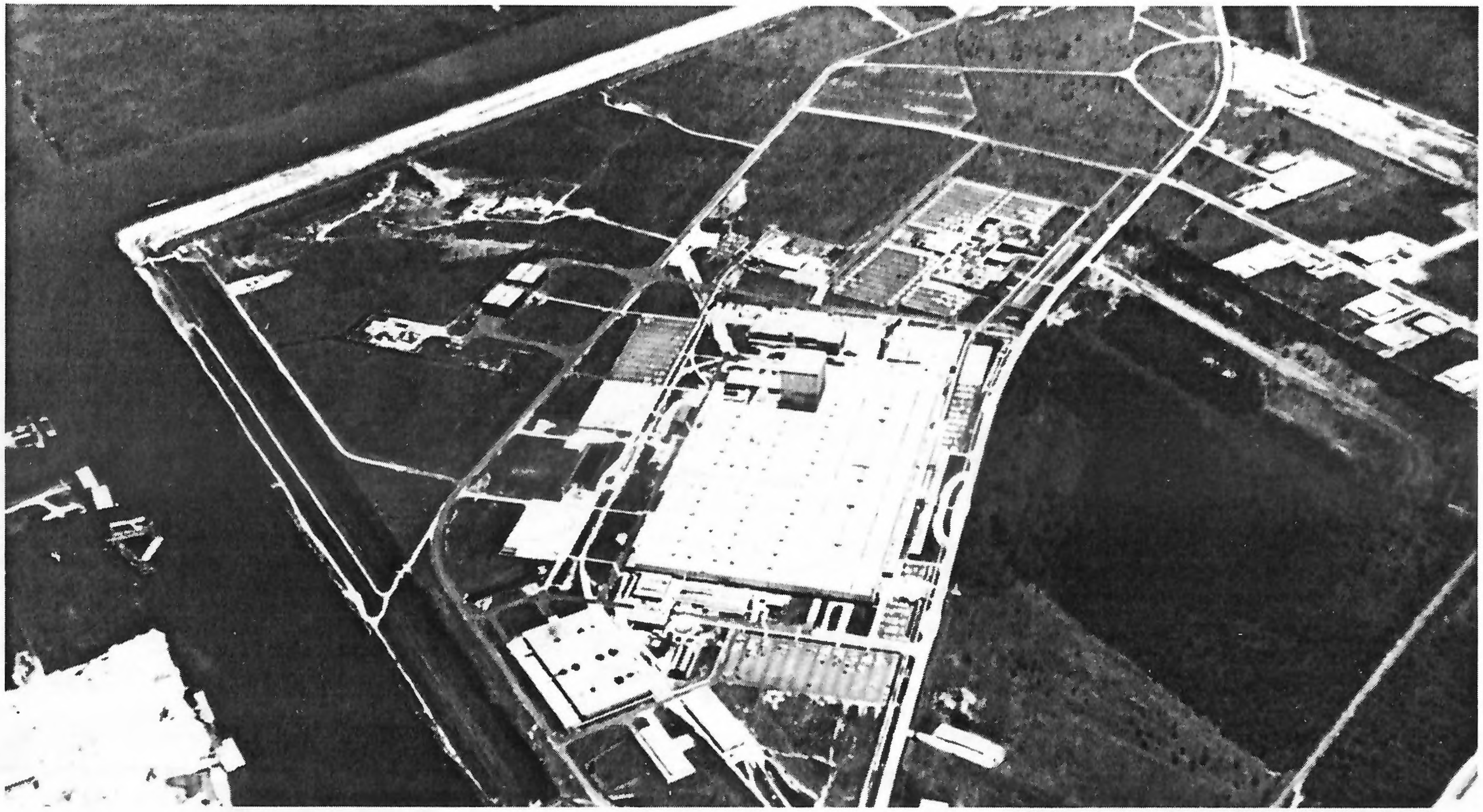
The STS Development and Production Division of Rockwell International's facilities at Downey, Calif., is responsible to the Johnson Space Center for the design, development, test and evaluation (DDT&E) of the Space Shuttle orbiter. The orbiter DDT&E calls for the fabrication and testing of two orbiter spacecraft (Orbiter 101 and 102), a structural test article (STA) and a main propulsion test article (MPTA). In addition to the DDT&E contract, the division is responsible for the integration of the overall Space Transportation System. The division is also converting the STA to an operational orbiter (Orbiter 099) in addition to building two additional orbiters (103 and 104). The Downey facility provides engineering and manufacturing of the Space Shuttle orbiter crew module, forward fuselage and aft fuselage, and forward reaction control system module, as well as Shuttle integration.

The Downey facility also maintains a Flight Simulation Laboratory, an Avionics Development Laboratory (ADL), and Flight Control Hydraulics Laboratory (FCHL). These laboratories have supported the total orbiter system verification process by accomplishing early testing on breadboard/prototype hardware and preliminary releases of software provided by NASA. These tests provide the foundation for orbiter testing at the Palmdale facility and the final hardware/software verification testing at the Shuttle Avionics Integration Laboratory (SAIL) located at the Johnson Space Center (JSC) in Houston, TX. Both the Flight Simulation Laboratory and ADL start with system hardware design development, proceed into flight software evaluation, and end in a complete orbiter mission simulation using the "iron bird" of the FCHL.



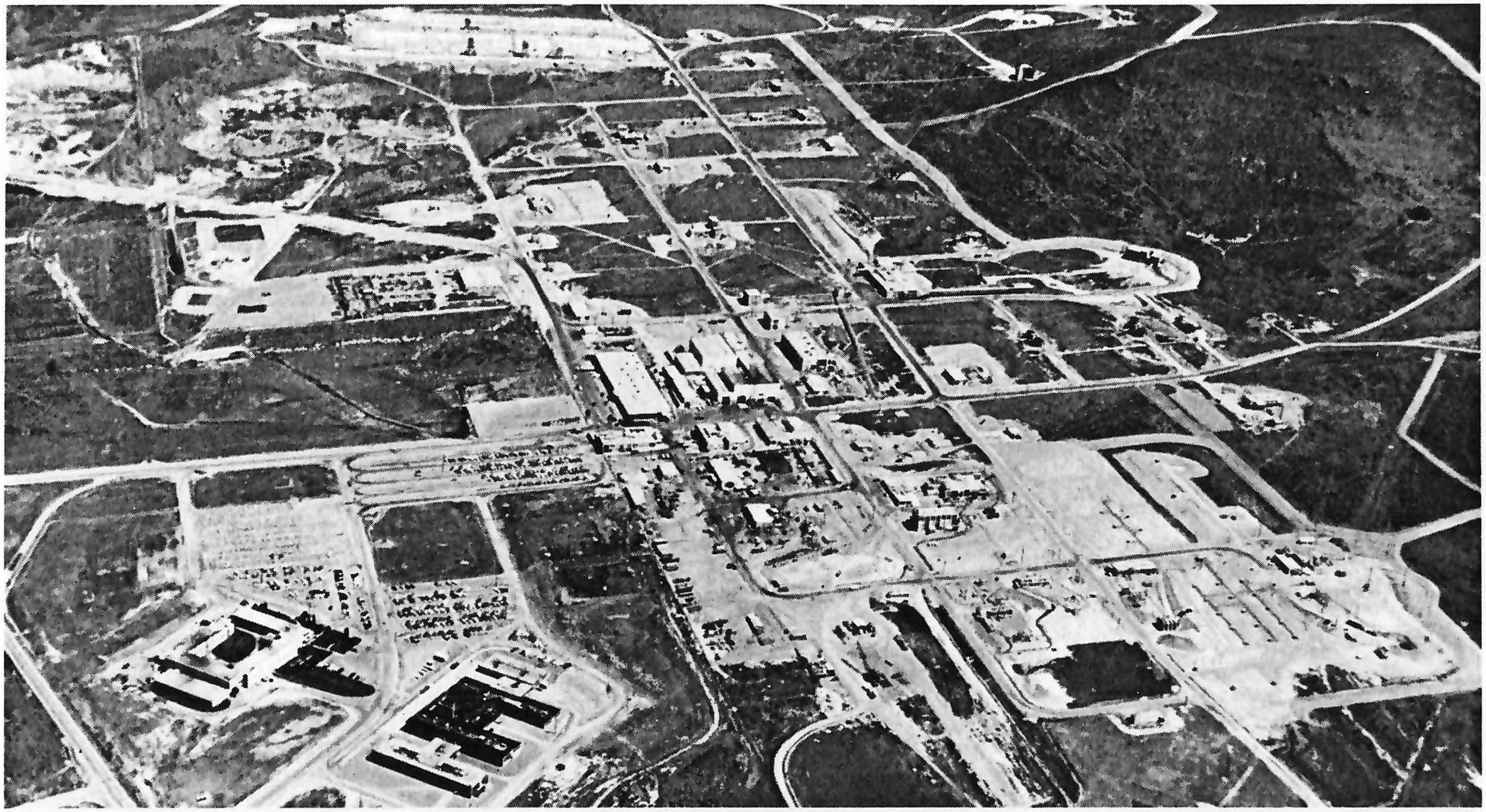
The U.S. Air Force Plant No. 42 at Palmdale, CA (Site 1) provides Space Operations with the facilities for the final assembly, test and checkout of the orbiters.

The Rocketdyne Division of Rockwell International, located in Canoga Park, CA, is responsible to the Marshall Space Flight Center for the design, development, manufacturing, testing, and assembly of the Space Shuttle main engines. The Rocketdyne facility at Santa Susana, CA, also is used for the Space Shuttle main engine test program.



Martin Marietta, Michoud, LA, Facility

Martin Marietta is responsible to the Marshall Space Flight Center and is using the Michoud facility in Louisiana for the design, development, manufacturing, testing, and assembly of the external tank.



Thiokol Chemical Corporation, Brigham, City, UT

Thiokol is responsible to the Marshall Space Flight Center for the design, development, manufacturing, testing, and assembly of the solid rocket booster motors. The solid rocket boosters, which are separated during launch of the Space Transportation System, will descend to the ocean and be towed from the recovery area to a land return. The solid rocket booster motor casings will be returned to Thiokol for refurbishment.

McDonnell Douglas Astronautics, Huntington Beach, CA

McDonnell Douglas is responsible to the Marshall Space Flight Center for the design, development, manufacturing, testing, and assembly of the solid rocket booster structure.

United Space Boosters, Incorporated, Sunnyvale, CA

United Space Boosters is responsible to the Marshall Space Flight Center for the solid rocket booster checkout, assembly, launch, and refurbishment except for the solid rocket booster motors.

Space Transportation System Glossary

AA	Accelerometer assembly
ABE	Arm-based electronics
ACCU	Audio control central unit
ACE	Automatic checkout equipment
ACK	Acknowledge
ADI	Attitude director indicator
ADS	Air data system
ADTA	Air data transducer assembly
AGL	Above ground level
AGS	Anti-gravity suit
A/L	Approach and landing
ALC	Aft load controller
ALT	Approach and landing test
AMC	Aft motor controller
AMI	Alpha Mach indicator
AOA	Abort once around
APC	Aft power controller
ARPCS	Atmospheric revitalization pressurization control system
ARS	Atmospheric revitalization system
ASA	Aerosurface servo amplifier
ASI	Augmented spark igniter
ASS	Airlock support system
ATO	Abort to orbit
AUX	Auxiliary
BFC(S)	Backup flight control (system)
CCA	Communications carrier assembly
CCTV	Closed circuit television
CCU	Crew communications unit
CCV	Chamber coolant valve
CDF	Confined detonating fuse
CDR	Commander
CI	Course (deviation display) invalid
COAS	Crew optical alignment sight
CPU	Control processor unit
CRT	Cathode ray tube
CSS	Control stick steering
C/W	Caution and warning
DA	Distribution assembly
DAP	Digital autopilot
DBC	Data bus coupler
DBIA	Data bus isolation amplifier
DBN	Data bus network
DDU	Display driver unit
D&C	Displays and Controls
DEU	Display electronics unit
DFI	Development flight instrumentation
DFRC	Dryden Flight Research Center
DK	Display keyboard
DME	Distance measuring equipment
DP/DT	Delta pressure/delta time
DPS	Data processing system
DSC	Dedicated signal conditioner
DU	Display unit
EAFB	Edwards Air Force Base
EAS	Equivalent airspeed
ECLSS	Environmental control and life support system
EES	Escape ejection suit
EI	Entry interface
EIU	Engine interface unit
EMU	Extravehicular mobility unit
EPDU	Electrical power distribution unit
EPS	Electrical power system
ESA	European Space Agency
ESRO	European Space Research Organization
ESS	Essential
ESVS	Escape suit ventilation system
ET	External tank or elapsed time
ETR	Eastern Test Range
EVA	Extravehicular activity

FC(S)	Flight control (system)
FCL	Freon coolant loop
FCOS	Flight computer operating system
FDM	Frequency division multiplexer
FES	Flash evaporator system
GH_2	Gaseous hydrogen
GN_2	Gaseous nitrogen
G&N	Guidance and navigation
GN&C	Guidance, navigation, and control
GMT	Greenwich mean time
GO_2	Gaseous oxygen
GPC	General purpose computer
GS	Glideslope
G/S	Guided steering
GSE	Ground support equipment
H_2	Hydrogen
HAC	Heading alignment cylinder
HAL/S	High order assembly language/Shuttle
HIU	Headset interface unit
H_2O	Water
HPFT	High-pressure fuel turbopump
HPOT	High-pressure oxygen turbopump
HRSI	High-temperature, reusable, surface insulation
HSI	Horizontal situation indicator
I	Initialized (computer program)
IEA	Integrated electronic assembly
IMU	Inertial measurement unit
IOP	Input-output processor
IUS	Inertial upper stage
IVA	Intra-vehicular activity
JSC	Johnson Space Center
KBU	Keyboard unit
LCA	Load control assembly
LDEF	Long-duration exposure facility
LH_2	Liquid hydrogen
LiOH	Lithium hydroxide
LO_2	Liquid oxygen
LPFT	Low-pressure fuel transducer
LPOT	Low-pressure oxygen transducer
LPS	Launch processing system
LRSI	Low-temperature, reusable, surface insulation
LVLH	Local vertical, local horizontal
LWHS	Lightweight headset
MAC	Mean aerodynamic chord
MCC	Main combustion chamber
MCC-H	Mission control center–Houston
MCDS	Multi-function CRT display system
MCIU	Manipulator controller interface unit
MDA	Motor drive amplifier
MDF	Mild detonating fuse
MDM	Multiplexer/demultiplexer
ME	Main engine
MECO	Main engine cutoff
MEP	Minimum entry point
MET	Mission elapsed time
MFV	Main fuel valve
MIA	Multiplex interface adapter
MLG	Main landing gear
MLP	Mobile launch platform
MLS	Microwave landing system
MM	Major mode or mass memory
MMC	Mid-motor controller
MMH	Monomethyl hydrazine
MMS	Multi-mission modular spacecraft
MMU	Master measurement unit
MOV	Main oxidizer valve
MPC	Mid-power control
MPL	Minimum power level

MPS	Main propulsion system
MPTA	Main propulsion test article
MSBLS	Microwave scan beam landing system
MSFC	Marshall Space Flight Center
MTU	Master timing unit
MTVC	Manual thrust vector control
N_2	Nitrogen
NASA	National Aeronautics and Space Administration
NEP	Nominal entry point
NH_3	Ammonia
NLG	Nose landing gear
N_2O_4	Nitrogen tetroxide
NSI	NASA standard initiator
NSP	Network signal processor
NSTL	National Space Technology Laboratories
O_2	Oxygen
OFI	Operational flight instrumentation
OI	Operational instrumentation
OMS	Orbital maneuvering system
OPF	Orbiter processing facility
OPOV	Oxidizer preburner oxidizer valve
OPS	Operational sequence
OV	Orbiter vehicle
PBI	Pushbutton indicator
PBS	Protective breathing system
PCA	Pneumatic control assembly
PCM	Pulse-code modulation
PCMMU	Pulse-code-modulation master unit
PDRS	Payload deployment and retrieval system
PDU	Power drive unit
PFCS	Primary flight control system
PRFLT	Preflight
PIC	Pyro initiator controller
PL	Payload
PLBD	Payload bay door
PLT	Pilot
PM	Phase modualtion
POS	Portable oxygen system
PRI	Primary
PROM	Programmable read-only memory
PRSD	Power reactant storage and distribution system
PSI	Pounds per square inch
PTC	Passive thermal control
RA	Radar altimeter
RAM	Random access memory
RCC	Reinforced carbon carbon
RCS	Reaction control system
RGA	Rate gyro assembly
RHC	Rotation hand controller
RJD	Reaction jet driver
RM	Redundancy management
RMS	Remote manipulation system
RPC	Remote power controller
RPL	Rated power level
RPTA	Rudder pedal transducer assembly
RS	Redundant set (computers)
RSS	Range safety system
RTLS	Return to launch site
SBTC	Speedbrake thrust control
SCA	Shuttle carrier aircraft
SCF	Satellite control facility
SEC	Secondary
SGLS	Space ground link system
SIP	Strain isolation pod
SM	System management
SMU	Speaker microphone unit
SOP	Software operating program
SPI	Surface position indicator
SRB	Solid rocket booster

Space Transportation System Glossary

SSME's	Space Shuttle main engines
SSUS	Solid spinning upper stage
ST	Star tracker
STA	Structural test article
STS	Space transportation system
TACAN	Tactical air navigation
TAEM	Terminal area energy management
TDRS	Tracking and Data Relay Satellite
THC	Translation hand control
TLM	Telemetry
TPS	Thermal protection system
TVC	Thrust vector control
T/W	Thrust to weight
UHF	Ultra-high frequency
VAB	Vehicle Assembly Building
Vdc	Volts direct current
VFM	View finder monitor
VIU	Video interface unit
W/B	Wide band
WBSC	Wide-band signal conditioner
WCLS	Water coolant loop subsystem
WONG	Weight on nose gear
WOW	Weight on wheels
WP	Way point
WSB	Water spray boiler
WSTF	White Sands Test Facility
WTR	Western Test Range

Space Transportation System Contractors

An experienced team of aerospace firms is working on the Space Shuttle System. Many of these firms, together with the system or component they produce, are listed in the following pages.

Contractor	Location	System/Component
Abbott Transistor	Los Angeles, CA	Transformer, displays and controls
Abex Corp., Aerospace Div.	Oxnard, CA	Pump, hydraulic, variable delivery
Aeroceanic Castings	Bell Gardens, CA	Orbiter castings
Aerodyne Controls Corp.	Farmingdale, NY	Oxygen, hydrogen check valve (fuel cell and environmental control life support system) Valve, pressure relief (water) Orbiter castings
Aeroflex Laboratories	Plainview, NY	Line assembly vibration isolation mounts, feedline (main propulsion system)
Aerojet General, Aerojet Liquid Rocket Co.	Sacramento, CA	Orbital maneuvering system engines
Aeroquip Corp., Marman	Los Angeles, CA	For environmental control/life support system: V-band coupling Coupling No. 10T-bolt, 40 degree Band clamp Band clamp, light weight Retaining strap Forty degree flange Flexible air duct
Aerospace Avionics	Bohemia, NY	Annunciator assembly (caution/warning system) General: Annunciator display general requirements Annunciator, performance monitoring Annunciator, special Annunciator, fire warning Annunciator, assembly event sequence Annunciator, assembly computer status Annunciator, single event Annunciator, control assembly Indicator, quantity propellant Floodlight, incandescent overhead Light dimmer Battery charger and power supply (environmental control/life support system)

Space Transportation System Contractors

Contractor	Location	System/Component
Aerospace Research Associates	West Covina, CA	Cable attenuator assembly for crew escape system, Orbiters 101 and 102
Aiken Industries, Mechanical Products Div.	Jackson, MI	Circuit breakers, thermal
		Circuit breakers, three phase
AIL, Cutler Hammer	Farmingdale, NY	Microwave scan beam landing set navigation set
	Milwaukee, WI	Circuit breaker, remote control (Orbiter 101)
	Huntington, NY	Preamplifier assembly
Airco Inc.	Industry, CA	Cylinders, hydrogen and oxygen, Orbiter 101
Aircraft Engineering Corp.	Paramount, CA	Main propulsion test article structure and platform, orbiter overland transporter, strongback orbiter overland transporter, strongback for MSFC
Air Industries	Garden Grove, CA	Nuts and bolts
Aircraft Instruments Co.	Montgomery, PA	Indicator position (Orbiter 101)
AiResearch Manufacturing Co., Garrett Corp.	Torrance, CA	Air data transducer assembly and computer
		Safety valve, cabin air pressure
		Solenoid valve, shutoff, air
		Ground coolant unit (circulate Freon through orbiter heat exchanger during ground operations, checkout, preflight, postflight)
Airite Div., Sargent Industries	El Segundo, CA	Helium receiver (spherical), surge pressure relief of gaseous helium during actuation of valves and external tank disconnects (main propulsion system)
Allen Bradley	Milwaukee, WI	Potentiometer (controls and displays)
American Aerospace	Farmingdale, NY	Current, sensor ac, dc
		Current level detector
American Airlines	Tulsa, OK	Operations support and maintenance of Boeing 747 Shuttle carrier aircraft
Ametek Calmec	Pico Rivera, CA	2-inch shutoff valve, liquid hydrogen (main propulsion system)
		4-inch disconnect, liquid hydrogen, orbiter to tank recirculation and replenishment system (main propulsion system)
		2-inch disconnect, gaseous hydrogen/gaseous oxygen, orbiter to tank pressurization system (main propulsion system)
Ametek Straza	El Cajon, CA	Liquid hydrogen and liquid oxygen 8-inch fill and drain assembly (main propulsion system)
		Liquid hydrogen recirculation and replenishment 2-inch and 4-inch line assembly (main propulsion system)
		Gimbal joint pressure (main propulsion system)

Space Transportation System Contractors

Contractor	Location	System/Component
Amex System	Lawndale, CA	Coax cable, high-temperature (flight instrumentation)
AMI	Colorado Springs, CO	Operational flight crew and passenger seats
Anemostat Products	Scranton, PA	Cabin air diffuser
		Cabin air diffuser
Apex Mills	Los Angeles, CA	Separators - passive thermal control
Applied Resources	Fairfield, NJ	Switch, rotary
Arkwin Industries	Westbury, NY	Hydraulic reservoir, bootstrap
		Valve, 3-way, 2-position hydraulic control
		Oleophobic filter hydraulic canister
Arrowhead Products, Div. of Federal Mogul	Los Alamitos, CA	Space Shuttle main engine feedlines, 12- to 17-inch diameter liquid oxygen and liquid hydrogen (main propulsion system)
		Connector, flexible purge gas (main propulsion system)
		Environmental control/life support system:
		Coupling sleeve
		Duct flex air
		Coupling sleeve flex
		Clamp cushion flex sleeve
		Flexible connector
		Flexible connector
		Convoluted bellows long flexible connector drain system
Astech	Santa Ana, CA	Heat shield (main propulsion system)
Autonetics Group, Rockwell International	Anaheim, CA	Shuttle avionics test set—Avionics Development Laboratory, Downey
		Shuttle avionics test set—Johnson Space Center
		Driver module controller
		AC bus sensor
		Load control assemblies (forward No. 1-3, aft No. 1-3)
		Master event controller
		Backup flight control system, Orbiter 101, 102
Avco	Wilmington, MA	Ku-band antenna (microwave scan beam landing system)
		Ku waveguide assembly
	Nashville, TN	Manufacturing (crew module bulkheads)
	Lowell, MA	Elevon ablator
Aydin, Vector Div.	Newtown, PA	Wideband frequency division multiplexing unit

Space Transportation System Contractors

Contractor	Location	System/Component
Ball Brothers Research Corp.	Boulder, CO	Star tracker and light shade
		Active keel actuator
		Payload retention latch
Barry Controls	Burbank, CA	Vibration mount assembly kit
Beech Aircraft Corp., Boulder Division	Boulder, CO	Power reactant storage assembly
		Orbiter Freon coolant servicing unit
		Gaseous hydrogen and gaseous oxygen valve box unit (ground support equipment)
Bell and Howell	Pasadena, CA	Magnetic tape recorder
Bell Industries	Gardena, CA	Terminal boards, modular
Bendix Corp.	Sidney, NY	High density connector, data processing software
		Connector, electrical
	Teterboro, NJ	Indicator/surface position
		Indicator/alpha mach
		Indicator/altitude/vertical velocity
	Franklin, IN	Connector, triax, 93 ohm (electrical power distribution system)
	Davenport, IA	Accelerometer indicator (senses "g" forces in vehicles and displays on display & control panel) (backup)
Beldo Steel Corp.	Orlando, FL	Canister to carry payloads from checkout facilities to orbiter*
B.F. Goodrich Co.	Troy, OH	Main/nose landing gear wheel and main landing gear brake assembly
		Tires, main and nose gear
Bertea Corp.	Irvine, CA	Main landing gear uplock actuator, hydraulic
		Main landing gear strut actuator
		Nose landing gear uplock actuator
		External tank umbilical retractor actuator (retracts external tank feedline into orbiter at separation; composed of linear hydraulic actuators with integral control valves and locking devices)
Betatronix	Hauppauge, NY	Potentiometers
Boeing Aerospace Co.	Houston, TX	Sneak circuit analysis
	Seattle, WA	Carrier aircraft modification
		Tail cone fairing
		Load measurement system (3 load sensor cells and signal conditioner; used in ALT to measure and display to 747 and orbiter crews the flight loads during taxi, takeoff, cruise, separation, and landing)

*Separate contract from NASA

Space Transportation System Contractors

Contractor	Location	System/Component
Bomar/TIC	Newbury Park, CA	Variable transformer (displays and controls) (Orbiter 101)
Brunswick	Lincoln, NE	Filament wound tank, developmental program
		Pressure storage tanks for orbital maneuvering system, main propulsion system, reaction control system, atmospheric revitalization pressure control system
Brunswick Celesco	Costa Mesa, CA	Smoke detection
		Fire suppression system
Brunswick-Circle Seal	Anaheim, CA	Check valve (purge, vent, and drain)
		Check valve, water
		Check valve, 1-inch helium repressurization line (main propulsion system)
		Valve 3/8 inch relief liquid hydrogen, (main propulsion system)
		Engine isolation dual check valve (main propulsion system)
		Servicing check valve
		Water relief valve
Brunswick-Wintec	El Segundo, CA	Fuel line filter (auxiliary power unit)
		Filter, windshield (purge, vent, and drain)
		Filter, coolant return
		Filter assembly, cryo
		Filter, ammonia inline
		Filter, helium (main propulsion system)
		Filter, water
Bussman Div. of McGraw Edison	St. Louis, MO	Seal, bulkhead window conditioning system
		General-purpose fuse holder
		Fuse, general-purpose microminiature, axial load
		Fuse, dc limiter high current
		Fuse holder, dc, boltdown
Calspan	Buffalo, NY	Wind tunnel (hypersonic) tests, jet plume effects
		Ascent and entry rates
		Heat/pressure distribution
Carleton Controls	East Aurora, NY	Atmospheric pressure control system
		Valve, shutoff ram air inlet (Orbiter 101)
		Regulators, hydrogen and oxygen, cryo
		Airlock support component system

Space Transportation System Contractors

Contractor	Location	System/Component
Celesco Industries	Canoga Park, CA	Transducer pressure absolute, high level
		Discrete pressure transducer
Charles Stark Draper	Cambridge, MA	Software (guidance, navigation, and control)
Chem Tric	Rosemont, IL	Generator, silver/ion (environmental control/life support system—Orbiter 101)
Coast Metal Craft	Compton, CA	Metal flex hoses (main propulsion system)
		Bulkhead pen lines
		Flex line assembly propulsion
Collins Radio Group, Rockwell International	Cedar Rapids, IA	Display driver unit
		Horizontal situation indicator
Columbus Aircraft Div., Rockwell International	Columbus, OH	Body flap structure
		Manufacturing (detail parts and subassembly, forward and aft fuselage tooling, nose gear doors)
Communications Components	Costa Mesa, CA	Antenna, UHF (approach and landing test), Orbiter 101
Conrac Corp.	West Caldwell, NJ	Engine interface unit (main propulsion system)
		Mission timer
		Event timer
		Ground command interface logic box (enables crew to select either crew control or NASA ground control of 250 communications functions)
Consolidated Controls	El Segundo, CA	Fuel isolation valve (auxiliary power unit)
		Unidirectional/bidirectional shutoff valve (fuel cell, environmental control/life support system)
		Solenoid valve, hydrogen and oxygen, cryo
		High-pressure helium valves and low-pressure vernier engine manifold valves, dc (reaction control system)
		Valve, hydraulic and oxygen primary flow control pressure (main propulsion system)
		Pressurant flow control valve, hydrogen and oxygen flow from orbiter main engines for external tank main propulsion system
		Regulator, 750/20 psia, helium (main propulsion system)
		Regulator valve, 850 psig, helium (main propulsion system)
		Helium regulator pressure, 750 psia, (main propulsion system)
		Water solenoid latching valve

Space Transportation System Contractors

Contractor	Location	System/Component
Contractors Cargo	South Gate, CA	Tractor for overland transport, Palmdale to DFRC
Convair Aerospace Div., General Dynamics	San Diego, CA	Mid fuselage, including mid fuselage glove fairing
Corning Glass	Corning, NY	Windshield and windows and side hatch window
		Thermal protection system Macor machinable glass ceramic
Cox and Co.	New York, NY	Heater, water, relief valve, vent nozzle, and port
		Heater assembly tank, fuel and lube lines (auxiliary power unit)
		Heater, water boiler steam vent line
		Heater, oxygen purge lines
		Heater, strop, water relief valve
		Heater electrical set (reaction control system)
Crane Co., Hydro Aire	Burbank, CA	Main landing gear brake anti-skid hydraulic modules, transducers, control box, and wheel sensor
Crissair Inc.	El Segundo, CA	Check valve, hydraulic
		Flow restrictor, hydraulic
Curtiss Wright	Caldwell, NJ	Payload bay door actuation system (power drive units, rotary actuators, drive shaft, torque tubes, couplings)
		Radiator deploy actuator and latch mechanism
Datum Inc.	Anaheim, CA	Multi-channel, closed-loop structural test
Deutsch	Banning, CA	General-purpose connector, electrical
		Bulkhead feedthrough
Descent Controls Inc.	Costa Mesa, CA	Descent control and lines
Dynamics Corp.	Scranton, PA	Cabin diffuser
Eckel Valve	San Fernando, CA	Check valves, purge (vertical stabilizer, forward reaction control system, and plenum)
Edcliff Instruments	Monrovia, CA	Position transducer, landing gear and rudder (Orbiter 101, ALT)
Edison Electronics Div., McGraw Edison	Manchester, NH	Digital select thumbwheel switch
		Toggle switches
Eldec Corp.	Lynwood, WA	Dedicated signal conditioner (subsystem pressure, temperature, etc., to multiplexer/demultiplexer)
		Tape, meter
		Proximity switch (landing gear operation)

Space Transportation System Contractors

Contractor	Location	System/Component
Electronics Association	West Long Branch, NJ	Analog computer system (Rockwell simulator)
Electronic Resources Inc., Tasker Industries	Los Angeles, CA	Coax cable, special, external temperature (communication tie, links)
Ellanef	Corona, NY	Hatch latch actuator
		Air data sensor probe actuator
		Star tracker door (consists of shutter-type door, electro-mechanical actuator, door supports)
		Manipulator retention latch actuator
Endevco	San Juan Capistrano, CA	Piezoelectric accelerometer (flight instrumentation, vibration-acoustic data)
		Acoustic pickup, piezoelectric (development flight instrumentation, acoustic data, Orbiter 101)
		Piezoelectric accelerometer (flight instrumentation, vibration-acoustic data)
ESA (European Space Agency)	Paris, France	Spacelab and U-shaped pallets*
Ex-Cell-O Corp., Div. of Cadillac Controls	Costa Mesa, CA	Hatch attenuator, main ingres/egress hatch
Explosive Technology	Fairfield, CA	Interseat energy transfer and sequencer, pyro (crew escape system, Orbiter 101s and 102)
		Severence system, pyro, ejection panel (Orbiters 101 and 102)
Fairchild Republic	Farmingdale, NY	Vertical tail
Fairchild Stratos	Manhattan Beach, CA	12-inch prevalves, shutoff propellant (main propulsion system)
		1-1/2 inch disconnect, liquid oxygen overboard bleed (main propulsion system)
		8-inch fill and drain valve, propellant (main propulsion system)
		Ammonia boiler subsystem (rejects heat during reentry)
		1-inch disconnect pneumatic, helium and gaseous nitrogen (main propulsion system)
		1-inch shutoff valve, liquid oxygen and liquid hydrogen relief (main propulsion system)
		Regulator, pressure, helium (series redundant), forward and aft (reaction control system)

*Memorandum of understanding signed by NASA and European Space Research Organization (ESRO), now the European Space Agency, effective 5/30/75

Space Transportation System Contractors

Contractor	Location	System/Component
		Cryogenic fluid and gas supply disconnects between orbiter power reactant storage and distribution system fill, drain, and vent lines and ground support equipment)
		Coupling, hypergolic servicing, nitrogen tetroxide/hydrazine (orbital maneuvering system/reaction control system)
		Cap, liquid hydrogen/liquid oxygen, 8-inch orbiter disconnect (main propulsion system)
		Coupling helium fill disconnect flight half
Ford Aerospace and Communications Corporation*	Palo Alto, CA	Ground communications system to support Space Shuttle at Vandenberg AFB
G&H Technology Co.	Santa Monica, CA	Connector, cryo
General Electric	Valley Forge, PA	Waste collector subsystem
	Waterford, NY	Thermal protection system room-temperature vulcanizing adhesive
George A. Fuller Co., Div. of Northrop Corp.	Chicago, IL	Fabrication and erection of mating-demating device for orbiter and 747 carrier aircraft at Dryden Flight Research Center**
Globe Albany	Auburn, ME	Nomex felt (thermal protection system)
Grimes Manufacturing	Urbana, OH	Floodlight, overhead, incandescent (Orbiter 101)
Grumman Corp.	Bethpage, NY	Wing (includes main landing gear doors, elevons, wing box glove)
Gulton Industries	Costa Mesa, CA	Accelerometer, linear low frequency (flight instrumentation, vibration-acoustic data)
		Differential pressure transducer, hydraulic actuators
		Transducer, pressure, pogo (main propulsion system)
		Transducer, sound pressure level
		Transducer, cabin acoustic
Hamilton Standard Div., United Technologies Corp.	Windsor Locks, CT	Freon coolant loop (includes fuel cell heat exchanger, Freon to water interchanger, Freon loop 1 and 2 inlet sensor, development instrumentation package, Freon pump package)
		Atmospheric revitalization subsystem (includes cabin fan assembly and debris trap, CO_2 absorber and temperature control avionics cooling assembly, humidity control heat exchanger assembly, development instrumentation signal conditioner and avionics fan, secondary coolant pump and accumulator, primary coolant pump and accumulator)
		Water boiler, hydraulic thermal control unit
		Hydraulic cart (ground support equipment)

*Separate contract from USAF
**Separate contract from NASA

Space Transportation System Contractors

Contractor	Location	System/Component
		Flash evaporator system (removes heat generated by orbiter systems from two coolant loops by turning water from fuel cells into steam and venting to space)
		Water management control panel
		Life support system for Space Shuttle extravehicular activity*
		Fabrication and field support of Space Shuttle EVA mobility unit suits*
Harris Corp., Electronics Systems Div.	Melbourne, FL	Pulse code modulation master unit
	Baltimore, MD	Payload data interleaver
Hartman Electric Manufacturing Div. of A-T-O	Mansfield, OH	General-purpose power contactor
Haveg Industries, Inc.	Winooski, VT	Wire, general-purpose
Hayden Switch and Instrument	Waterbury, CT	Limit switches, hermetic seal (electrical power and distribution system)
Hexcell Aerospace	Dublin, CA	Attenuator pads (energy absorbers for inner and outer crew escape panels during deployment) (crew escape system, Orbiters 101 and 102)
Hi-Temp Insulation Inc.	Camarillo, CA	Hydraulic blanket insulation (hydraulic system)
		Fibrous bulk insulation-passive thermal control
Hoffman Electronics Corp. NavCom Systems Div.	El Monte, CA	TACAN (tactical air navigation)
Holloway Corp.	Titusville, FL	Construction of solid-rocket booster recovery and disassembly at Cape Canaveral Air Force Station*
Honeywell Inc.	St. Petersburg, FL	Flight control system (displays and controls):
		Rotation hand control
		Accelerometer assembly
		Rudder pedal transducer assembly
		Speed brake thrust control
		Translation hand control
		Forward and aft reaction jets and orbital maneuvering system drivers
		Aerosurface servo amplifier
		Ascent thrust vector control amplifier
	Minneapolis, MN	Radar altimeter
	McLean, VA	Central data processing for Space Shuttle launch processing system, Kennedy Space Center*

*Separate contract from NASA

Space Transportation System Contractors

Contractor	Location	System/Component
Hoover Electric	Los Angeles, CA	Umbilical door latch actuator
		Umbilical door actuator
		Umbilical door centerline latch actuator
		Orbiter payload door latch actuators (2 electromechanical rotary actuators to latch payload bay doors closed; one to extend and retract radiators and operate latches to hold radiators in retracted position)
		Actuator, payload bay door centerline latch
Hughes	El Segundo, CA	Ku-band radar/communication system, deployable antenna and electrical assembly
Hydraulic Research, Textron	Valencia, CA	Servo actuator, elevon-electro command, hydraulics, Orbiter 101
IBM Corp., Federal Systems Div., Electronics Systems Center	Owego, NY	Mass memory/multi-function cathode ray tube display unit, keyboard display electrical unit subsystem
		General-purpose computer processor unit and input-output processor
		Space Shuttle Mission Control Operations data processing complex at JSC*
		Orbiter computer programs*
		Shuttle data processing complex programming at JSC*
		Launch processing system at KSC*
	Gaithersburg, Md.	Space Shuttle orbiter data processing hardware maintenance*
ICI United States Inc.	Wilmington, DE	Thermal protection system alumina mat
ILC Technology	Sunnyvale, CA	Cabin interior lighting
		Orbiter floodlight system, payload bay floodlight, overhead docking floodlight, forward bulkhead floodlight, and floodlight electronic assemblies
Intermetrics Inc.	Cambridge, MA	Advanced computer programming language, HAL/S (high-order assembly language/Shuttle)
		Avionics software
ITT Cannon	Santa Ana, CA	Connector, power
		High-density connectors
		Rectangular connector
		Connector, coax TNC
		Connector, coax TNC bulkhead
		Connector, coax HN
		Connector, coax bulkhead
	Phoenix, AZ	Bulkhead feedthrough

*Separate contract from NASA

Space Transportation System Contractors

Contractor	Location	System/Component
J. C. Carter Co.	Costa Mesa, CA	Service coupling (auxiliary power unit)
Jet Electronics	Grand Rapids, MI	Indicator attitude, backup, Orbiter 101
J. L. Products	Gardena, CA	Crew compartment failure warning and corrective control: Arming fire switch, pushbutton Pushbutton switch Relay
J.P. Stevens Co.	Los Angeles, CA	Thermal protection system Quartz thread
Kaiser Electronics	San Jose, CA	Heads-up display Orbiter 102, STS-5, Orbiter 099 and subsequent
K-West	Westminster, CA	Wideband signal conditioner, accelerometer/acoustic
		Strain gage signal conditioner, stresses
		Ullage pressure signal conditioner for external tank (monitors and controls external tank ullage pressure, liquid oxygen and liquid hydrogen tanks)
		Differential pressure transducer and electronics, propellant head pressure in main feed and fill lines (main propulsion system)
Labarge	Santa Ana, CA	Wire, general-purpose
Leach Relay	Los Angeles, CA	General-purpose latching relay
		Relay, 10 amp
Lear Siegler	Grand Rapids, MI	Attitude direction indicator
	Elyria, OH	Hydraulic disconnect supply, 1/2-inch
		Coupling, heat exchanger test point (ground support equipment)
		Test point and nitrogen coupling
		Coupling, flight half
	Los Angeles, CA	Disconnect, flight half—air, gaseous nitrogen, purge
Life Systems Inc.	Cleveland, OH	Separator, hydrogen/water unit
Lockheed-California Co.	Burbank, CA	Ejection seats, including drogue and personnel chute (crew escape system, Orbiters 101 and 102)
		Orbiter structural static and fatigue testing
Lockheed Electronics	White Sands, NM	Support services*
Lockheed Missiles and Space Co., Inc.	Sunnyvale, CA	Reusable surface insulation, high temperature and low temperature and high temperature reusable surface insulation
		Fibrous refractory composite insulation (FRCI) -8 and -12
		Space telescope support systems module* (14-ft. diameter, 43-ft. long cylindrical unit houses optics, sensors, and support systems; systems engineering, analysis, integration, verification of complete assembly and support of ground and flight operations

*Separate contract from NASA

Space Transportation System Contractors

Contractor	Location	System/Component
Los Angeles Aircraft Div., Rockwell International	Los Angeles, CA	Manufacturing of aft fuselage upper truss thrust structure
		Diffusion bonding
		Crew module panels
		1/4-scale ground vibration test model
		Tool fabrication
Magnavox	Ft. Wayne, IN	Receiver transmitter mount, UHF
		Receiver/transmitter, UHF
Malco Microdot Corp.	Pasadena, CA	Connector, coax bulkhead
Marquardt Co.	Van Nuys, CA	Reaction control system thrusters, 870-lb thrust
CCI Corp.		Reaction control system thrusters, 24-lb thrust
Marshall Space Flight Center (NASA)	Huntsville, AL	Solid rocket booster integration and final assembly
Martin Marietta	Denver, CO	Caution and warning electronics, status display, limit module
		Pyro initiator controller
		Reaction control system fluel and oxidizer tanks (forward and aft)
	New Orleans, LA	External Tank*
		Checkout, control, and monitor subsystem*
	Vandenberg Air Force Base, CA	Definition and planning of acquisition of ground systems to support Shuttle operations at VAFB**
Marvin Engineering	Inglewood, CA	Manufacturing (crew module skins and ejection panels)
McDonnell Douglas Astronautics Co.	St. Louis, MO	Orbital maneuvering system/reaction control system aft propulsion pod
	Huntington Beach, CA	Solid rocket booster structure*
		Assembly, checkout, launch operations, and refurbishment, solid rocket boosters*
		Spinning solid upper stage** (SSUS)
	Huntsville, AL	Spacelab integration
Megatek	Van Nuys, CA	Cryo seals, line flange (main propulsion system)
Menasco Manufacturing Co.	Burbank, CA	Main/nose landing gear shock struts and brace assembly
Merco Manufacturing Co.	Anaheim, CA	Manufacturing (crew module star tracker panels)

*Separate contract from NASA
**Separate contract from Department of Defense

Space Transportation System Contractors

Contractor	Location	System/Component
Metal Bellows Co.	Chatsworth, CA	Potable and waste water tanks Metal bellows assembly Flex metal tube, convoluted Flex metal tube assembly RCS flexible line assembly Propellant flex line assembly, OMS/RCS
Metalcraft Inc.	Baltimore, MD	Portable fire extinguishers
Micro Measurements	Romulus, MI	Strain gage
3M Co., Inc.	St. Paul, MN	Thermal protection system AB312 fibers
Modular Computer Systems	Ft. Lauderdale, FL	Data acquisition system (Rockwell laboratories) Central data subsystem of launch processing system (mini-computer)*
Moog, Inc.	East Aurora, NY	Main engine gimbal servo actuator Elevon servo actuators, Orbiter 102 and subsequent
Motorola	Scottsdale, AZ	Communications test set (ground support equipment)
Networks Electronics Corp., U.S. Bearings Div.	Chatsworth, CA	Hatch latch links, main hatch
Northrop Corp., Electronics Div.	Norwood, MA	Rate gyro assembly
OEA	Denver, CO	Thruster assembly, pyro, nose gear uplock release Guillotine assembly pyro crew escape system, Orbiters 101 and 102 Thruster assembly ejection panel jettison, Orbiters 101 and 102
Parker Hannifin	Irvine, CA	17-inch disconnects, liquid hydrogen and liquid oxygen, orbiter to external tank feed system (main propulsion system) 8-inch disconnect, liquid hydrogen and liquid oxygen, orbiter to ground fill and drain (main propulsion system) Accumulator, hydraulic Cryogenic pressure relief valve Orbital maneuvering system and reaction control system propellant isolation valve tank manifold interconnect lines, 64 valves on each orbiter, plus additional valves, payload delta V kit Valve, inline liquid hydrogen/liquid oxygen emergency relief (main propulsion system)

*Separate contract from NASA

Space Transportation System Contractors

Contractor	Location	System/Component
		AC motor valve, valve isolation, propellant (reaction control system)
		Hypergolic couplings
		Valve, pressure relief (reaction control system)
		Dual check valves, manually operated (main propulsion system)
		Valve, manually operated (orbital maneuvering system/reaction control system)
Perkin Elmer Corp., Optical Technology Div.	Danbury, CT	Space telescope assembly, fine pointing assembly and associated controls*
Pneu Devices	Goleta, CA	Shutoff valve, emergency thermal control, hydraulic
		Circulation pump, hydraulics, electric motor driven
Pneu Draulics	Montclair, CA	Priority valve reservoir primary, hydraulic
Porter Seal	Glendale, CA	Seal plate, air data sensor unit
Pratt & Whitney Div., United Technologies Corp.	E. Hartford, CT	Fuel cell power plant
Pressed Steel Tank Co.	Milwaukee, WI	Ammonia storage module tank, mid fuselage (Orbiter 101)
Pressure System Inc.	Los Angeles, CA	Hydrazine fuel tank, (auxiliary power unit)
Purolator Inc.	Newbury Park, CA	Hydraulic filter module assembly
		Helium fill connect, 5/8-inch
		Helium fill connect, 1/4-inch
		Hydrogen, oxygen fill and vent disconnects (EPS)
RDF Corp.	Hudson, NH	Temperature sensor/transducer, general
		Temperature resistance transducer (probe-type)
		Temperature resistance transducer (probe-type)
		Cryo temperature transducer
		Thermocouple reference junction (thermal protection system thermocouple signal conditioner)
		Transducer temperature tip
Regent Jack Manufacturing Co.	Downey, CA	Forward orbiter jacks
Remtech, Inc.	Huntsville, AL	Functional material

*Separate contract from NASA

Contractor	Location	System/Component
Resistoflex	Roseland, NJ	Hydraulic systems: Line connector, dynatube Line connector, 90^{0} dynatube Line connector, tee dynatube Line connector, bulkhead dynatube Line connector, jam nut-type dynatube Line connector, female/male adapter dynatube
Rocketdyne Div., Rockwell International	Canoga Park, CA	Space Shuttle main engines* Valve quad check (orbital maneuvering system/reaction control system)
Rosemount Inc.	Eden Prairie, MN	Probe system, air data sensor Probe air data sensor, flight boom Sensor temperature probe Temperature transducer Sensor temperature surface, general Indicator, angle of attack (backup) (Orbiter 101) Indicator, angle of slideslip (backup) (Orbiter 101)
Radio Corp. of America, Astro-Electronics Div.	Princeton, NJ	Closed-circuit television, orbiter and remote manipulator systems*
R.V. Weatherford	Glendale, CA	Shunt
Santa Fe Textiles	Santa Ana, CA	Thermal protection system Inconel 750 wire spring and fabric sleeving
Scheldahl	Northfield, MN	Cover materials and inner layers - passive thermal control
Scott, Inc.	Downers Grove, IL	Main landing gear uplock release thruster actuator
Sealectro	Mamaroneck, NY	Connector, coaxial SMA series
Sentran Co.	Santa Barbara, CA	Vacuum sensors (located in upper and lower wing area, vertical stabilizer, and upper and lower sections of fuselage to measure aerodynamic pressure at reentry and compartment pressures)
Simmonds Precision Instruments	Vergennes, VT	Sensors and electronics, point level, liquid oxygen and liquid hydrogen (main propulsion system)
Simmonds Precision, Motion Controls Div.	Caldwell, NJ	Forward and aft vent doors (electrical, mechanical actuators, torque tubes, push/pull rods, bell cranks, and mechanism support for opeaning and closing vent/purge doors for all ground operations and flight operations)

*Separate contract from NASA

Space Transportation System Contractors

Contractor	Location	System/Component
Singer Kearfott	Little Falls, NJ	Inertial measurement unit Multiplexer interface adapter Data bus coupler Data bus isolation Maintenance, modification, and operations support, JSC simulation complex* Spacelab simulator*
Skipper and Co.	Cerritos, CA	Chemical processing (Rockwell facility)
Snap Tite	Northridge, CA	Coupling, quick-disconnect, high-pressure
Space Ordnance Systems Div., Trans Technology Corp.	Saugus, CA	Cartridge assembly detonator (frangible nut, tail cone separation, orbiter-carrier aircraft separation, Orbiter 101, and orbiter-external tank separation) Pressure cartridge Frangible nut, orbiter-external tank aft separation, 3/4-inch Sequencing assembly pyro, crew escape system, Orbiters 101 and 102 Gas generator assembly, pyro crew escape system, Orbiters 101 and 102 Initiator assembly, pyro panel jettison, Orbiters 101 and 102 Booster cartridge, detonator assembly Cartridge, forward separation bolt, pyro, centering mechanism
Spar Aerospace	Toronto, Canada	Remote manipulator system for Space Shuttle Orbiter (arrangement between NASA and National Research Council of Canada, supported by RCA, Ltd., and CAE of Montreal, Canada)*
Spectran Instrument	La Habra, CA	Radiometer (thermal protection system) Calorimeter, standard Sensor, radiant heat flux Sensor, combination heat flux Sensor, temperature thermocouple (measures main engine plume temperature on thermal protection system)
Speedring Manufacturing Co.	Cullman, AL	Window retainers
Sperry Rand Corp., Flight Systems Div.	Phoenix, AZ	Multiplexer/demultiplexer Automatic landing Indicator, airspeed/mach, backup (4-inch round) (Orbiter 101) Indicator, altimeter, barometric (Orbiter 101)

*Separate contract from NASA

Space Transportation System Contractors

Contractor	Location	System/Component
SSP Products Inc.	Burbank, CA	Exhaust duct assembly (auxiliary power unit)
Statham Instruments	Oxnard, CA	Pressure transducers (low, medium, high), general systems
		Pressure transducer, cryo
Sterer Engineering & Manufacturing	Los Angeles, CA	Nose gear steering and damping system
		Valve, selector 3-way, solenoid-operated, landing gear uplock and control
		Shutoff valve, solenoid-operated, main engine hydraulic and hydraulic landing gear
Sterling Transformer Corp.	Brooklyn, NY	Transformer, power displays and controls, 115/26 volt (Orbiter 101)
Sundstrand Corp.	Rockford, IL	Auxiliary power unit
		Rudder-speed brake actuation unit and drive shaft
		Actuation unit, body flap
		Hydrogen recirculation pump assembly (main propulsion system)
Sundstrand Data Control	Redmond, WA	Heater thermostat (auxiliary power unit)
		Thermal switch
Symetrics	Canoga Park, CA	Hydraulic quick disconnects
		Fluid disconnect
		Fluid disconnects (environmental control life support system) Freon
		Quick disconnects, water boiler, water fill vent
Systron-Donner	Concord, CA	Accelerometer, angular, 3-axis (development flight instrumentation)
Subsidiary of Systron-Donner: Seaton Wilson Inc.	Burbank, CA	Water and coolant system:
		Coupling, 2-inch, half, quick-disconnect female
		Coupling, 2-inch, half, quick-disconnect male
Tavis Corp.	Mariposa, CA	Flow meter, Freon
		Transducer, pressure very low range
Tayco Engineering	Long Beach, CA	Fuel cell, excess water dump nozzle
		Urine, waste water, oxygen, and nitrogen waste dump
Teledynamics Div., Ambac Industries	Fort Washington, PA	S-band transmitter (development flight instrumentation)
		Frequency modulation
		S-band transceiver

Space Transportation System Contractors

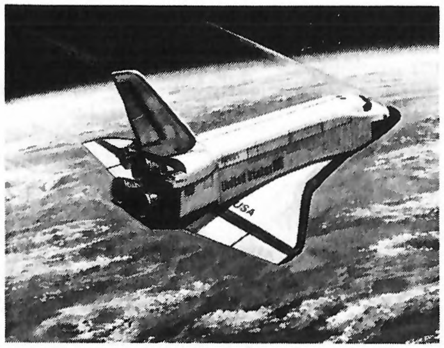
Contractor	Location	System/Component
Teledyne Kinetics	Solano Beach, CA	DC power contractor Limit switches, hermetic seal (electrical power and distribution)
Teledyne Thermatics	Elm City, NC	Cable coax prototype Wire, general-purpose Wire, TSP71 ohm Wire, T/C Wire, TFE insulated
Teledyne McCormack	Hollister, CA	Initiator assembly, pyro (crew escape system, Orbiters 101 and 102)
Telephonics Division, Instruments Systems Corp.	Huntington, NY	Orbiter audio distribution system (voice and tonal signals)
Thiokol Chemical Corp., Wastach Div.	Brigham City, UT	Solid rocket booster motors*
Times Wire and Cable	Wallingford, CT	Coax cable Coax cable
Titeflex Div.	Springfield, MA	Flex hose, low-pressure, windshield purge Flex line coolant loop (water coolant) Hose, high/low pressure (hydraulic system) Hose, swivel assembly (hydraulic system) Flexible stainless steel hose
Torrington Co.	Torrington, CT	Needle bearing (environmental control/life support system)
TRW Systems, Electronic Systems Div.	Redondo Beach, CA	S-band payload interrogator S-band network equipment Network signal processor Payload signal processor Materials processing in Spacelab program* Tracking data relay satellites (TDRSS)*
Transco Products	Venice, CA	S-band switch
Tulsa Div., Rockwell International	Tulsa, OK	Cargo bay doors Ground support equipment/parts off-load
United Space Boosters, Inc.	Sunnyvale, CA	Assembly, checkout, launch operations, and refurbishment of solid rocket boosters* including recovery ship operation*

*Separate contract from NASA

Space Transportation System Contractors

Contractor	Location	System/Component
U.S. Bearing	Chatsworth, CA	Spherical bearing, orbiter-external tank
U.S. Radium	Parsippany, NJ	Lighting panel overlay
Vacco Industries	El Monte, CA	Pressure relief valve, inline potable water
Vought Corp.	Dallas, TX	Leading edge structural subsystem and nose cap, reinforced carbon-carbon
		Radiator and flow control assembly system
		Manufacturing (crew module skins and bulkheads)
Waltham Precision Instruments	Waltham, MA	8-day clock, windup (Orbiter 101)
Watkins Johnson	Palo Alto, CA	Antennas:
		C-band radar altimeter
		UHF air traffic control voice
		L-band TACAN (tactical air navigation)
		Antenna S-band (approach and landing test) Orbiter 101
		S-band, quad antenna
		S-band, hemi antenna
		S-band, payload antenna
Wavecom	Northridge, CA	S-band multiplexer (development flight instrumentation)
Westinghouse Electric Corp., Aerospace Electrical Div.	Lima, OH	Remote power controller
		Electrical system inverters, dc-ac
Westinghouse Electric Corp., Systems Development Div.	Baltimore, MD	Master timing unit
Weston Instruments	Newark, NJ	Event indicator
		Electrical indicator meter, round scale, vertical scale
Whittaker Corp.	North Hollywood, CA	Dump valve, manually operated, hydraulic accumulator (ground)
		Regulator, 750/20 psia, helium (main propulsion system)
Wright Components Inc.	Clifton Springs, NJ	2-way pneumatic solenoid valve (main propulsion system)
		3-way solenoid valve helium (main propulsion system)
		Valve, latching, solenoid-operated, hydraulic (hydraulic system and main propulsion system)
		Fuel pump seal cavity drain catch bottle relief (auxiliary power unit)
Xebec Corp.	Kansas City, MO	Automated circuit
Xerox Corp.	El Segundo, CA	Digital computer (Rockwell simulator)

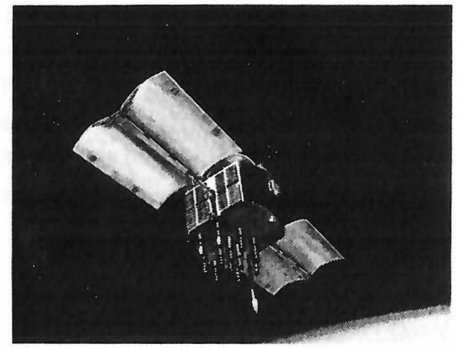
*Separate contract from NASA



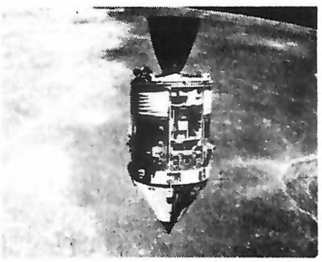
Shuttle Orbiter

The STS Development and Production Division of Rockwell International Corporation, Downey, CA, is in the second phase of a design, development, test, and evaluation (DDT&E) contract with NASA's Johnson Space Center to build two Space Shuttle orbiters and associated test articles. In addition, NASA has given the division authority to proceed with the manufacture of two additional orbiters and the conversion of the structural test article orbiter to a space flight configuration orbiter. The division also has contractual responsibilities for integration of the entire Space Shuttle Transportation System.

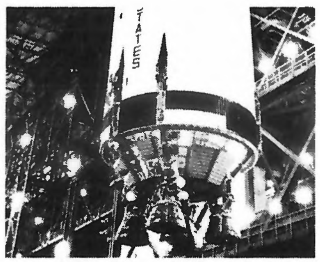
The Space Operations and Satellite Systems Division of Rockwell International Corporation, Seal Beach, CA, is under contract to design and develop the Global Positioning System satellites (Navstar) under a Department of Defense contract managed by a joint program office at the U.S. Air Force's Space Division, Los Angeles, CA.



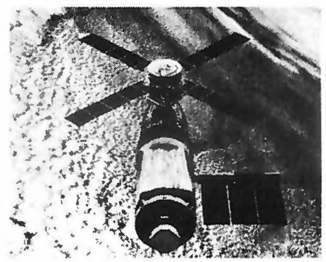
GPS Navstar



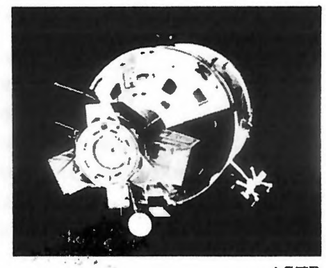
Apollo CSM



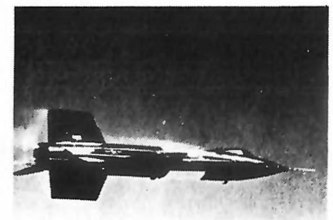
Saturn S-II



Skylab



ASTP



X-15

The Rockwell International Corporation, Downey, CA, facility was the prime contractor for design, development, fabrication, and test of the Apollo Command and Service Modules (spacecraft) used in the successful NASA Apollo lunar landing program, the three Skylab missions, the Apollo-Soyuz Test Project (ASTP), and also designed and fabricated the docking module and docking system used in ASTP.

Rockwell International's, Seal Beach, CA, facility designed and built the Saturn Second Stage (S-II) of the Saturn V launch vehicle used in the lunar landing program as well as the first flight of Skylab when it boosted the laboratory into earth orbit. In the mid-1950s, Rockwell International designed and built the three X-15 experimental aircraft used by NASA in the 1960s for space flight research.

Additional information may be obtained by contacting Rockwell International's Downey, CA, facility, Public Relations personnel:

R. E. Barton, Manager of Public Relations
Bus.: 213/922–1217
Home: 213/377–8962

R. V. Gordon, Media Relations Executive
Bus.: 213/922–1217

W. F. Green, Technical Communications
Bus.: 213/922–2066
Home: 213/331–6773

Or National Aeronautics and Space Administration (NASA) Public Information news centers:

Headquarters, Washington, D.C. *202/755–3090*

Johnson Space Center *713/483–5111*

Marshall Space Flight Center *205/453–0034*

Dryden Flight Research Center *805/258–3311*

Kennedy Space Center *305/867–2468*

Contacts for planning and information concerning payload use of the STS by potential customers should be made to Space Transportation Systems Operations Office, MO, NASA, Washington, D.C., 20546, telephone 202/755—2344 or Space Transportation Systems User Service Center, Space Operations and Satellite Systems Division, Rockwell International, 12214 Lakewood Boulevard, Downey, CA, 90241, telephone 213/922—3344 or 3345.

Metric Conversion Table

	Multiply	By	To Obtain
Distance:	Inches	2.54	Centimeters
	Feet	0.3048	Meters
	Meters	3.281	Feet
	Kilometers	3281	Feet
	Kilometers	0.6214	Statute Miles
	Statute Miles	1.6093	Kilometers
	Nautical Miles	1.852	Kilometers
	Nautical Miles	1.1508	Statute Miles
	Statute Miles	0.8689	Nautical Miles
	Statute Miles	1760	Yards
Liquid Measure, Weight:	Gallons	3.785	Liters
	Liters	0.2642	Gallons
	Pounds	0.4536	Kilograms
	Kilograms	2.205	Pounds
	Metric Ton	1000	Kilograms
	Short Ton	907.2	Kilograms

	Multiply	By	To Obtain
Pressure:	Pounds/Sq Inch	70.31	Grams/Sq Cm
Thrust:	Pounds	4.448	Newtons
	Newtons	0.225	Pounds
Velocity:	Feet/Sec	0.3048	Meters/Sec
	Meters/Sec	3.281	Feet/Sec
	Meters/Sec	2.237	Statute mph
	Feet/Sec	0.6818	Statute mph
	Feet/Sec	0.5925	Nautical mph
	Statute Miles/Hr	1.609	Km/Hr
	Nautical Miles/Hr (Knots)	1.852	Km/Hr
	Km/Hr	0.6214	Statute mph
Volume:	Cubic Feet	0.02832	Cubic Meters