SUMMARY OF THE ORBITER MECHANICAL SYSTEMS

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ABSTRACT

Major mechanical systems of the Orbiter space vehicle are summarized with respect to general design details, manner of operation, expected performance, and, where applicable, unique features. A synopsis of data obtained during the five atmospheric flight tests of spacecraft OV-101 and status of the systems for the first orbital spacecraft STS-1 are presented.

INTRODUCTION

The NASA Orbiter manned spacecraft is the next generation of space vehicles being designed for reuse and to reduce the cost and increase the effectiveness of using space for commercial, scientific, and defense needs. The primary design and operations goal for the Orbiter is to provide routine access to space. The general configuration and details of the Orbiter are shown in figure 1. The successful operation and accomplishment of the NASA mission objectives, using this new generation of spacecraft, is dependent on the proper functioning of numerous mechanical systems.

This paper is devoted to a review of Shuttle mechanical systems, including separation systems, the crew escape system, aerothermal seals, pressure seals and thermal barriers, payload bay door mechanisms, and the landing deceleration system. These systems have in common certain mechanical design aspects but obviously involve many other engineering disciplines, both within the system and in interfaces between the mechanical system and other Shuttle systems. Vehicle dynamics and aerodynamics are particularly important to the design of the separation, crew escape, and landing deceleration systems. Each of these systems also includes pyrotechnic components. The aerothermal seals, the pressure seals, and the thermal barriers derive requirements from aerodynamic heating and static pressure environments. The aerothermal seals are unique in that seals must be maintained between moving surfaces during the high aerodynamic heating pulse of atmospheric entry.

Design of all mechanical systems for the Shuttle is based on maximum use of existing aircraft and spacecraft technology. This approach minimizes direct development costs and schedule risk and reduces the need for extensive systems-level tests to establish reliability. For example, the ejection seat used in the crew escape system is one previously qualified for use in another aircraft, and most of the components of the landing deceleration system are of conventional design and materials.

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New technology is applied only where it results in significant weight savings, as in the use of the carbon/beryllium brake, or where no solid technology base exists, as in the aerothermal seals.

Another unusual factor governing the design and development of Shuttle mechanical systems is the limited life requirement. Each Orbiter vehicle has a planned design life of 100 missions, with appropriate margins. However, for components which are easily changed out between flights, the cost of replacement as frequently as every fifth mission is traded against the cost of developing a component with full 100-mission capability. For example, the Orbiter tires and brakes are designed for 5 to 10 missions, and their routine replacement is scheduled in the normal turnaround procedure. Lifetimes as short as five design missions result in lightweight components but reduce the margin between the design five-use and emergency one-use capability. This reduction dictates a realistic assessment of the design and emergency system requirements; such an assessment is the basis for extensive systems analysis and careful attention to the conditions imposed in limited test programs.

The mechanical systems are described independently. For each system, key design requirements and the impact on systems design of changing requirements are discussed. Early trade-offs and system design options are reviewed with consideration of critical design issues and primary design drivers. Interesting design and development problems are treated primarily to indicate the technical evolution of current Shuttle systems designs. The application of new technology is discussed as appropriate during the presentation of each mechanical system.

AEROTHERMAL SEALS AND REMOTELY OPERATED DOOR THERMAL BARRIERS

The Orbiter is the first manned spacecraft to use aerodynamic control surfaces that are movable during the high heating pulse of entry. Aerothermal seals are required to restrict gas flow into the control surface hinge-cavity areas so as to maintain structural temperatures within acceptable limits. Such seals are used on the elevon, rudder/speed-brake, and body flap control surfaces. Each of the three control surface seals is unique in design and materials because of varying thermal requirements and varying hinge-cavity geometry. The elevon lower cove is closed out with a polyimide seal rubbing against aluminum and Inconel tubes and plates. The elevon upper surface is sealed with formed Inconel springs rubbing against Inconel panels. The body flap seal is composed of Inconel hinged panels. The rudder/speed-brake seal is made of graphite seal blocks rubbing on conic Inconel panels. Only the basic design/materials concept has been mentioned. Each of the aerothermal seals is quite complex in design detail as each includes several unique solutions to specific geometric and thermal problems. For example, the elevons are composed of two independently movable surfaces per side; thus, there are four ends of the elevon hinge cove per side which must be closed out. The hinge locations constitute another sealing challenge. The entire seal system must accommodate the surface distortion associated with high aerodynamic and thermal loads. It was even more difficult to achieve a design that could be assembled and verified by inspection after assembly.

Thermal barriers are provided to close out doors, hatches, and other penetrations through the vehicle thermal protection system. Because these hatches and doors are generally not cycled during the entry heating pulse, relatively simple, passive thermal barriers can provide aequate thermal sealing. Three basic types of thermal barriers are used in door edge gaps. In the nose gear door, a pad-type barrier is used to fill the gap out to the outer mold line and thereby to ensure a very smooth surface in this very high heating region. A fiber brush barrier is used to accommodate relative deflections at the edges of the extremely large payload bay doors. Most other doors and openings incorporate unique variations of metal springs filled with insulating material and wrapped with ceramic cloth. Figure 2 shows representative aerothermal seal and static thermal barriers. Figure 3 shows the body flap aerothermal seal.

The Orbiter aerothermal seals and thermal barriers will be flown for the first time in the upcoming Orbital Flight Test program. The approach and landing flight tests of 1977 did not include the extreme thermal environment; therefore, the sealing requirements were much less severe. Certification of the Orbiter aerothermal seals is to be accomplished through a program which combines the results of tests and analysis activities to ensure that the subsystem provides adequate protection to the structure when exposed to the mission environment.

Basic materials testing was performed to ensure thermostructural properties of all materials used. Samples of critical seals combined with adjacent reusable surface insulation (RSI) to make a test article of approximately 0.46 meter (1.5 feet) square were tested in arc-jet facilities at three NASA centers. These tests simulated approximate thermal entry environments. One full-size outboard elevon and portion of the trailing edge wing will be tested under entry thermal and structural loads. Various large panels and portions of the aerodynamic surfaces will be acoustic life tested with seals installed.

CREW ESCAPE SYSTEM

The general concept of the Orbiter is to provide a transport- or airliner-type vehicle to routinely shuttle multiple passengers and cargo to and from orbit. Provisions for handling emergency situations are therefore based on maintaining the integrity of the Orbiter to provide safe return of crew and passengers. This "intact abort" concept is the same as that used by commercial airliners. However, for the first few missions, the Orbiter is very much an experimental aircraft. During these missions, the crew will be limited to two; and, early in the program, it was recognized that a reasonable option existed to provide individual emergency escape capability for the critical launch and landing phases. Individual ejection seats have been incorporated for the pilot and the copilot. These crew escape provisions will be removed following the Orbital Flight Test series, which will provide added confidence in the reliability of Shuttle systems and procedures.

The general philosophy of using developed, tested, and qualified systems resulted in selection of the U.S. Air Force SR-71/F-12 ejection seat. A careful evaluation of the performance of all available qualified seats showed the SR-71 seat to have the greatest range of high- and low-altitude capability. Figure 4 shows the design altitude/velocity capability of the SR-71 seat system, as well as a typical Shuttle trajectory. The seat provides reasonable escape capability from sea level to an altitude of 21 336 meters (70 000 feet) during launch and provides complete coverage during final descent to landing and for the complete Approach and Landing Test (ALT) envelope.

A major effort was made to use the seat as designed; however, the unique Orbiter crew cabin and structure arrangement, plus the vertical launch attitude, indicated the necessity for seat modifications. One change resulted when early reach and vision studies showed the crew would have problems reaching and seeing the necessary controls to fly the Orbiter during ascent. These findings were verified in a centrifuge test. As a result, a two-position back angle was incorporated into the seat. Should ejection be required during ascent, the back would automatically reposition the crew aft before exit. Following launch, the crewmember manually repositions the seat back for entry and landing. Pyrotechnic devices already used in the seat are used also for operating this device. See figure 5 for details of the seat-back modification and the salient features of the system.

The sled test program disclosed two other problems. First, the side-by-side arrangement of the seats in the Orbiter led to very high rotation rates (in yaw direction) as the seats entered the airstream. This problem was alleviated by a slight change in timing of drogue parachute deployment so that the stabilizing forces of the drogue would be effective earlier. Second, during the second dynamic sled test, one of the seats recontacted and damaged the inflated personnel parachute. Another slight timing change, this time to increase the interval between separation of the crewman from the seat and deployment of the personnel parachute, significantly reduced the possibility of recontact. Both of these sequence timing changes were verified in subsequent sled testing.

The Orbiter inner and outer crew cabin and structural hatch required a unique mechanical system design to enable jettisoning of the two hatches before the seats are ejected. The method of accomplishing this is shown in figure 6. This requirement differs from normal aircraft design, in which only a single structural surface must be cut or jettisoned. The U.S. Air Force standard SR-71 ejection seat, with the modifications described previously, was flown on all ALT flights. The same system is being installed on the first orbital flight spacecraft.

LANDING DECELERATION SYSTEM

The Orbiter landing deceleration system consists of a conventional tricycle landing gear including struts, wheels, tires, brakes, brake skid control, and nosewheel steering (fig. 7). The general design philosophy is to use conventional components and materials, yet to provide the lightest system capable of meeting predicted requirements. The Orbiter system is unconventional in three major areas, all based on achieving a lightweight system: (1) the landing gear is deployed by gravity rather than using hydraulic actuators - possible because there is no in-flight gear retraction requirement; (2) the brakes are composed of beryllium heat sinks with carbon linings - a combination with very high energy capacity; and (3) brakes and tires are designed for 5 uses rather than the usual 100 or more uses. This limited use requirement narrows the gap between the planned operating condition and the emergency ultimate operating capability of the hardware. As a result, a much more detailed and accurate analysis of Orbiter landing performance than that typically given commercial and military aircraft systems was required. This analysis is based on a limited number of very carefully planned tests.

Except for the brakes, the Orbiter landing deceleration system is designed of materials similar to those on other aircraft. The struts are machined from 300M alloy forgings - the same material used on 727, 737, 747, DC-10, etc., aircraft (fig. 8). The tires are manufactured of nylon cord and natural rubber - the same as commercial and military aircraft. The wheels are conventional aluminum forgings. The brake/skid-control system is the same brand used on nearly all large commercial and military transport aircraft, and the steering package is the same brand used on the B-1 bomber.

The brakes are of a multidisk design similar to that of other current large aircraft. One brake is included in each of the four main wheels. The brakes are commanded electrically through foot-pedal-operated transducers, which control hydraulic actuators. The unique feature of the brakes is the use of carbon linings attached to beryllium heat sinks. Other aircraft use brakes with beryllium heat sinks - the C-5A and the F-14, for example - and several use all-carbon brakes. The unique Orbiter carbon/beryllium combination was developed from that technology. The Orbiter brakes weigh about half as much as conventional sintered-iron-lined brakes for the same energy absorption capacity. (See fig. 9.)

The Orbiter size and landing weight are comparable to a medium size commercial airliner, but the landing speed is almost twice that of the typical airliner or military aircraft. Landings at a speed of 102.9 m/sec (200 knots) will be routine. Emergency landings with the full 29 484-kilogram (65 000 pound) payload can result in speeds as high as 115.7 m/sec (225 knots). Fortunately, for this case, all landings will occur at the launch site, where a 4572-meter (15 000 foot) runway is available. The brakes are capable of bringing the Orbiter to a safe stop under these conditions but are expended in the process. The four brakes must absorb as much as 301 megajoules (222 million foot-pounds) of energy at a peak rate of nearly 22 371 kilowatts (30 000 horsepower). Some brake elements reach temperatures of 1255 K (1800° F). Several tests of the brake at these conditions have been executed on a dynamometer. The carbon/beryllium combination has proved to be an excellent emergency brake. Of course, the normal operating conditions are much less severe; and, whereas the design requirement is for five uses, many uses per brake are expected to be obtained with routine landings.

An interesting characteristic of the Orbiter brake is that it does not "fade" as it heats during a hard stop. Rather, it produces more and more torque for a given hydraulic pressure as the Orbiter velocity decreases, with low-speed torque about 50 percent higher than that available at braking initiation. This characteristic contributes significantly to the brakes' good emergency performance, but it requires that the pilot "ease off" the brakes at low speed during a normal stop to reduce the possibility of brake damage.

SEPARATION SYSTEM

The major components for separation of the Orbiter from the Shuttle external tank (ET) are illustrated in figure 10. At the forward attachment, release is accomplished by a pyrotechnic-actuated shear-type separation bolt. This type device is used at the forward attachment to satisfy a stringent requirement for a smooth outer mold line after separation and to withstand high structural loading before separation. The purpose of the smoothness criteria shown in figure 10 is to control the aerothermal heating on the spacecraft in order to prevent excessive structural temperatures during entry. The unique feature of this device is the internal fracture surface. After a piston shears the fracture plane, it closes out the bolt cavity to provide the required smooth surface.

At the two aft structural attachments, a 6.35-centimeter (2.5 inch) frangible nut is used as the release device. These frangible nuts are also fractured by pyrotechnic charges. They have recently been increased in diameter from 5.08 centimeters (2 inches) to 6.35 centimeters (2.5 inches) to provide additional structural margin at these attachments. Once the Orbiter is separated from the external tank, doors close out the cavities at the aft structural and umbilical attachments to protect the Orbiter structure during entry.

These structural attachments were used to release the Orbiter from a Boeing 747 aircraft during a series of Orbiter atmospheric test flights completed in 1977. The release devices for these tests were pyrotechnic-actuated separation bolts which fractured in tension, rather than shear as in the present bolt configuration. The shear bolts and frangible nuts are qualified to the flight environments as components, then the forward and aft structural assemblies are flight certified in a series of functional separation and structural loads tests. These tests are scheduled for completion by late August 1979.

ORBITER/ET UMBILICAL SEPARATION SYSTEM

The Orbiter/ET umbilical separation system provides the separation interface between the Orbiter vehicle and the external tank for fluid lines and electrical connections. There are two separate umbilical disconnect clusters between the Orbiter and the ET; both are located in the bottom aft fuselage of the Orbiter. The left-hand umbilical contains an electrical disconnect assembly and the liquid hydrogen (LH_2) system fluid disconnects, which include a 43.18-centimeter (17 inch) diameter propellant feed line, a 5.08-centimeter (2 inch) diameter tank repressurization line, and a 10.16-centimeter (4 inch) diameter prestart and tank replenishing line. The right-hand umbilical is similar, with an electrical disconnect assembly and two liquid oxygen (LO_2) fluid disconnects: one 43.18-centimeter (17 inch) diameter propellant feed line and a 5.08-centimeter (2 inch) diameter tank pressurization line. The umbilical separation system is designed to provide complete separation of the fluid and electrical interfaces between the Orbiter and the ET before initiation of structural separation. Figure 11 contains details of the system.

Two seconds after main engine cutoff (MECO), the LH_2 and LO_2 43.18-centimeter (17 inch) disconnect valves are closed pneumatically. Two seconds later, the LH_2 and LO_2 umbilical carrier plate separation command is issued, detonating the pyrotechnics which fracture three frangible nuts in each umbilical plate. A simultaneous command is issued to activate three hydraulic retractors attached to each of the umbilical plates. Both umbilicals then retract 6.35 centimeters (2.5 inches) into the Orbiter in approximately 4.5 seconds. The ET structural separation is commanded approximately 11 seconds after MECO.

Verification and qualification testing is being conducted on flight configuration LH_2 and LO_2 umbilical assemblies mounted in specially designed support fixtures. All LH_2 separation tests are conducted with liquid hydrogen in the system. During LO_2 separation tests, liquid nitrogen is substituted for LO_2 for safety considerations. During this series of tests, each hydraulic retractor will be individually disabled to simulate failed retractors to demonstrate a design requirement that any two of the three retractors shall provide adequate separation of the umbilical assemblies. Backup propellant valve closure modes are also being tested whereby valve closure is actuated mechanically during separation of the umbilical plates.

The umbilical separation system will be flown for the first time on spacecraft STS-1 since it was not required during the ALT program.

PAYLOAD BAY DOORS

The purpose of the Shuttle spacecraft, as the name implies, is to deliver a cargo to and/or return a cargo from Earth orbit. Since the deliverable item can be as large as 4.57 meters (15 feet) in diameter and

18.29 meters (60 feet) long, the Shuttle Orbiter is a unique spacecraft that requires two payload bay doors that are more than 18.29 meters (60 feet) in length - the largest to ever be operated in space. The graphite epoxy doors, which serve as the base for radiator retention and deployment, must react vehicle loads and protect the payloads during boost and entry and, yet, open and close as required once in orbit. The specific details of the payload bay doors are as presented in reference 1. The general configuration of the payload bay door and radiator system is shown in figure 12.

PAYLOAD RETENTION SYSTEM

The payload retention concept was selected shortly after the Space Shuttle Program was initiated, primarily because the selected approach would have a significant effect on the design of the Orbiter structure and, hence, on vehicle weight. The basic approach is a statically determinate load reaction system as shown in figure 13. This approach would use two retention fittings on the longeron to react X- and Z-axis loads, one additional longeron to react Z-axis loads, and a single reel fitting for Y-axis loads. Each fitting is designed to slide with minimum friction in those axes not designated as primary load carrying. Based on data available, the coefficient of friction achievable was selected as a maximum of 0.1 for the expected temperature range. To date, this value has not been achievable for temperatures to 200 K (-100° F), apparently because sliding surfaces are also geometrically sensitive.

To further complicate the problem, many of the payload suppliers desire to use a nondeterminate, four-point, longeron attach scheme. This design requires that the retention latch reach and "latching" pulldown force be sufficient to accommodate the out-of-plane condition that will exist during payload retrieval. Retrieval also required that some sort of integral guides be added to the retention fitting to aid in the installation of payloads in orbit.

In summary, what appeared at the outset to be a simple actuator design has evolved into a complex system when considering present Orbiter/payload restrictions. The design and function of the payload retention system is being evaluated in various ground test efforts involving a mockup of the payload bay, dummy payloads, and the remote manipulator system.

CONCLUDING REMARKS

In this paper, only a brief introduction to some of the more important Orbiter mechanical systems has been presented. Each of these systems involves interesting design features and unique performance requirements. Several papers could be written on special testing efforts, such as tire and wheel-bearing tests to actual load, yaw angle, and velocity histories, and arc-jet testing of aerothermal seal components and sections. Obviously, in a general paper of reasonable length, only the highlights of each system can be presented. Those having special interest in a particular system or technology field covered herein are encouraged to contact the authors for more detailed discussion or information exchange.

REFERENCE

1. McAnally, Bill M.: Space Shuttle Orbiter Payload Bay Door Mechanisms. Paper prepared for presentation at the 13th Aerospace Mechanisms Symposium (Johnson Space Center, Tex.), Apr. 26-27, 1979. Rockwell International rep. SOD 79-0072.

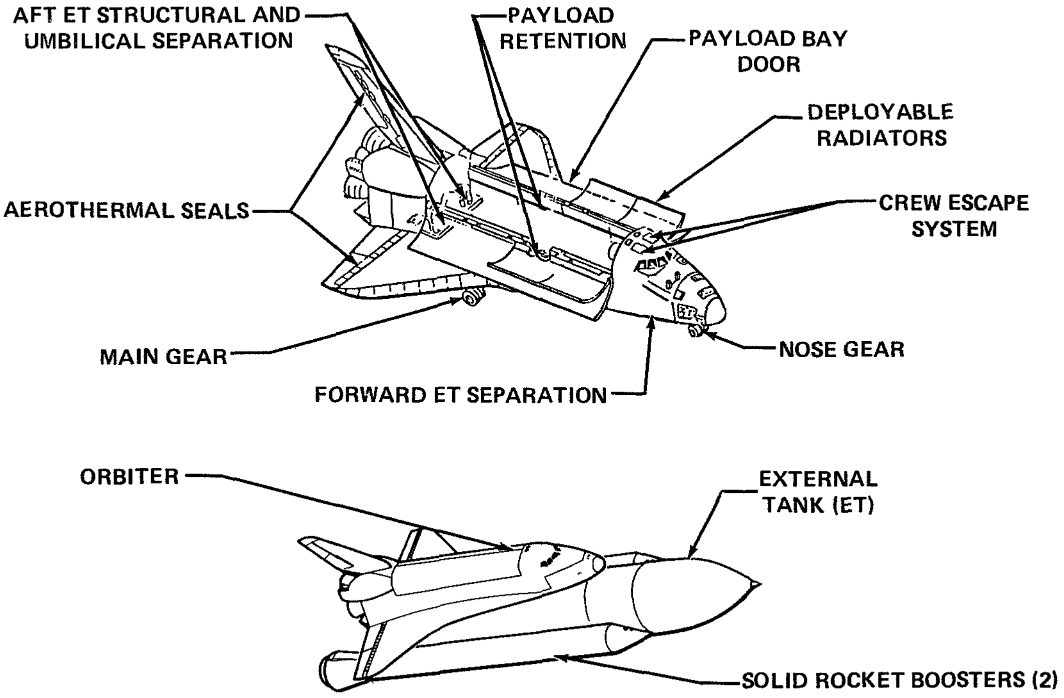


Figure 1.- Orbiter mechanical systems.

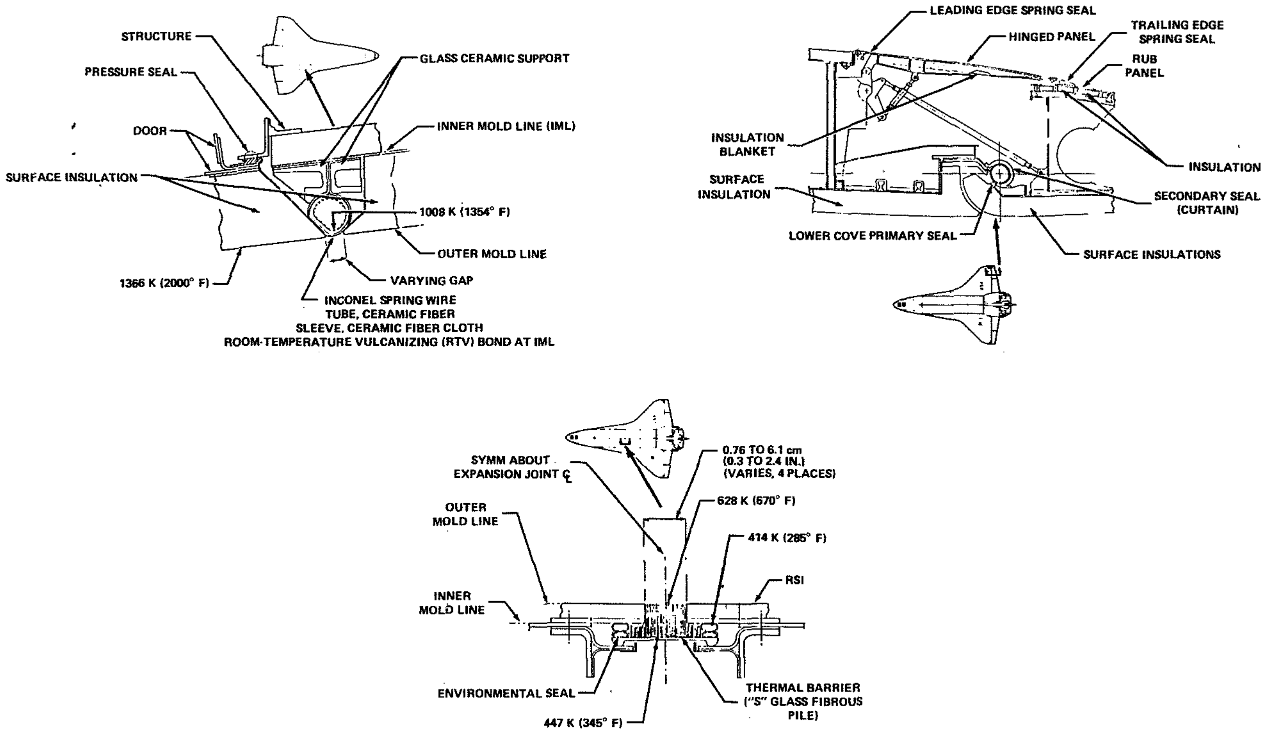


Figure 2.- Composite aerothermal seals and static thermal barriers.

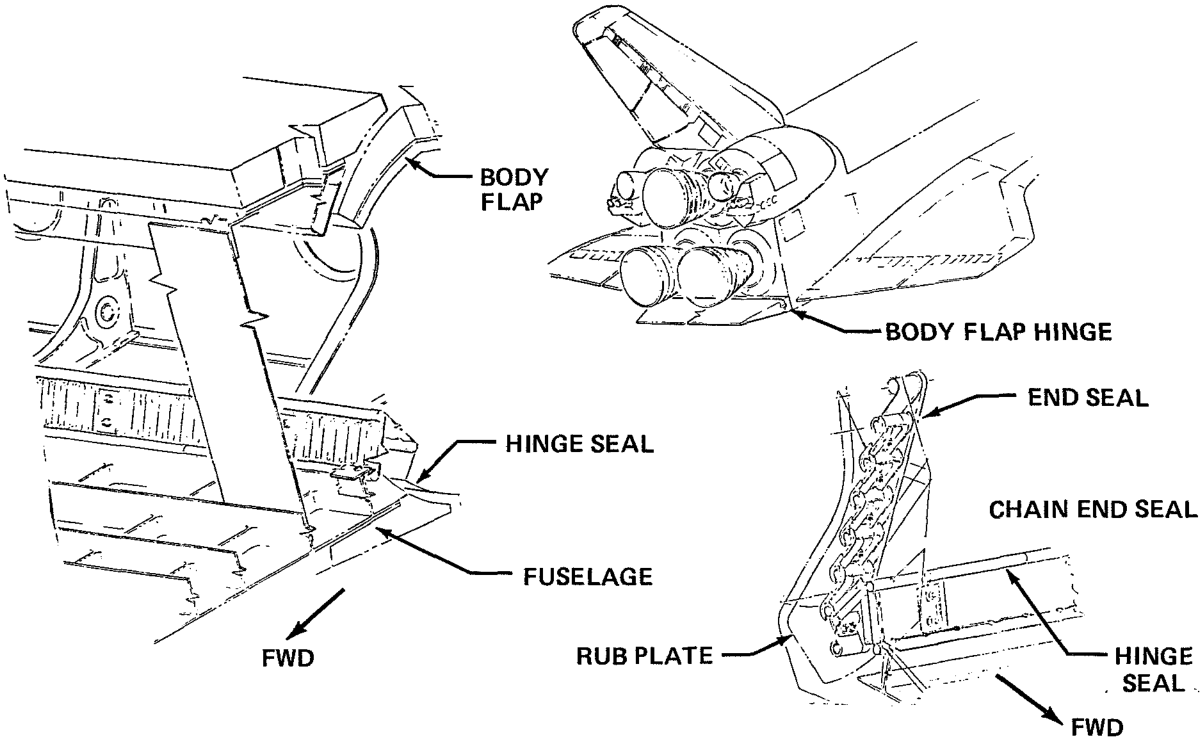


Figure 3.- Body flap aerothermal seal.

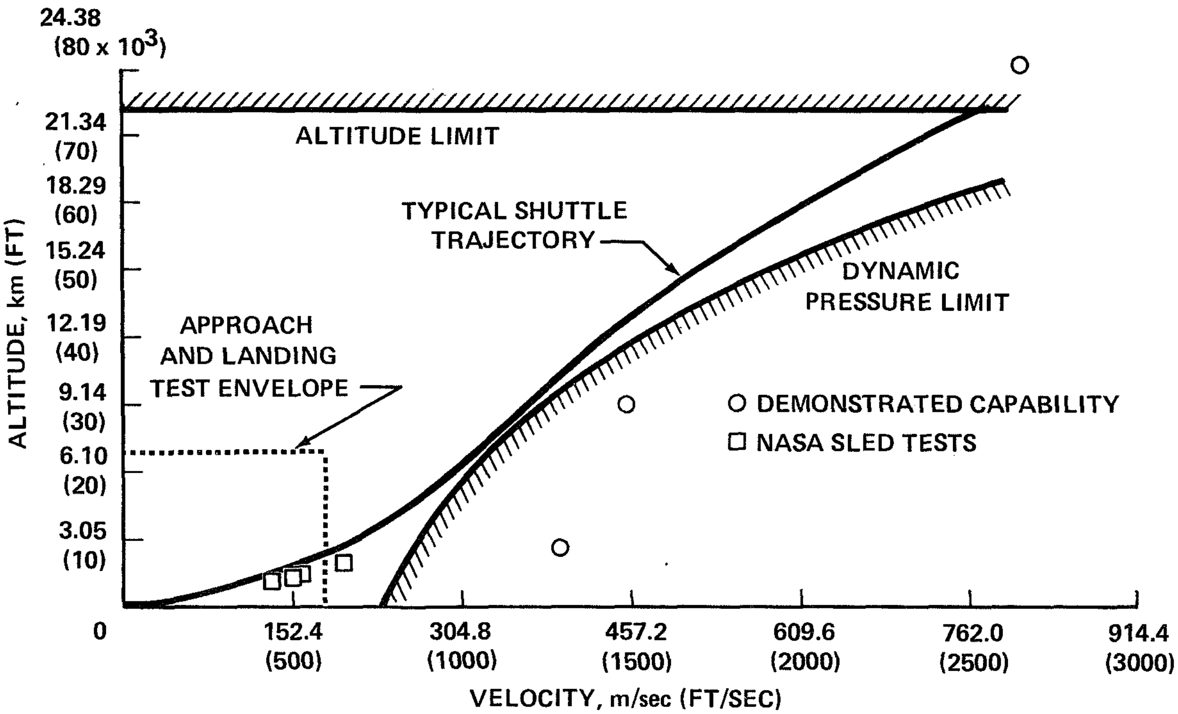


Figure 4.- Ejection seat capability.

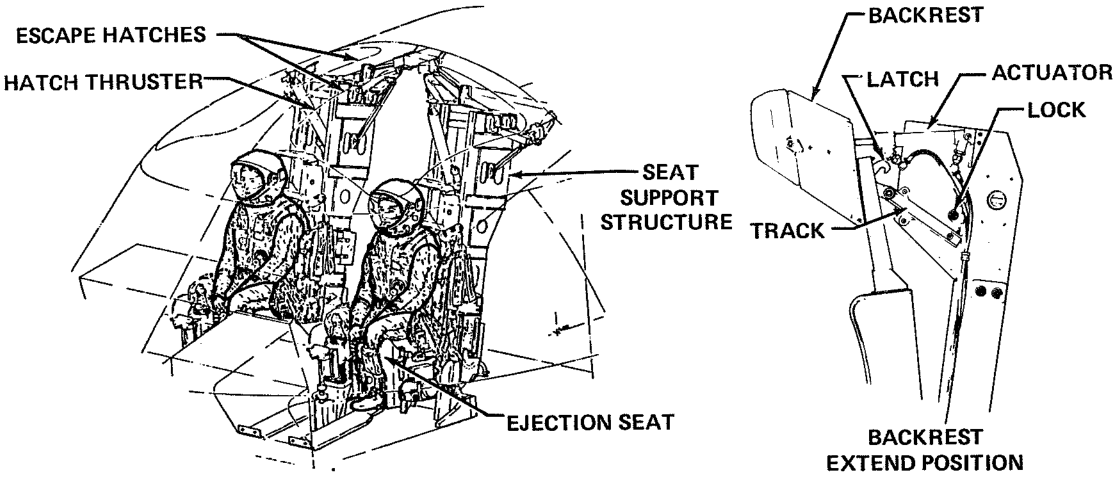


Figure 5.- Crew escape system installation.

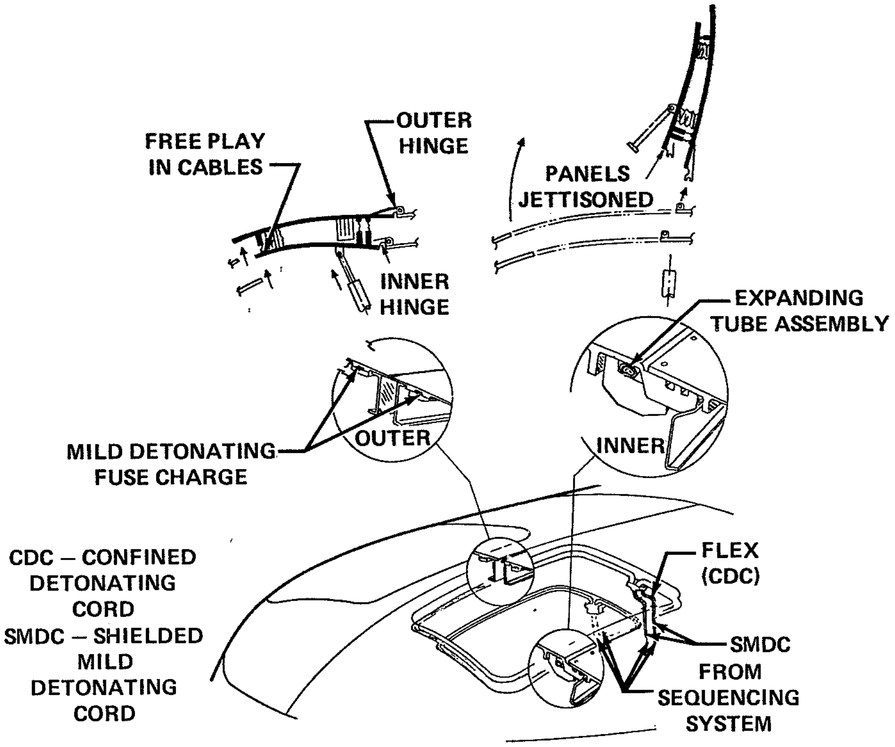
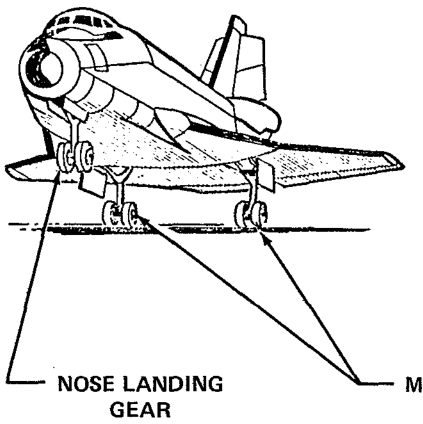
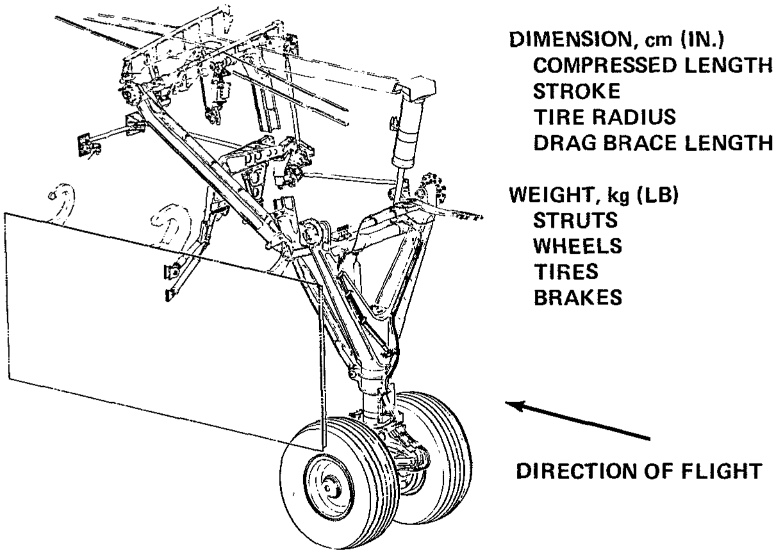


Figure 6.- Crew escape hatches.



- NOSE AND MAIN LANDING GEAR
- GEAR UPLOCK, EXTENSION, AND RETRACTION MECHANISMS
- MAIN WHEEL BRAKES
- SKID-CONTROL SYSTEM
- NOSEWHEEL STEERING

Figure 7.- Landing deceleration system.



	MAIN		NOSE	
DIMENSION, cm (IN.)				
COMPRESSED LENGTH		233.7 (92)		137.2 (54)
STROKE		40.6 (16)		55.9 (22)
TIRE RADIUS		55.9 (22)		40.6 (16)
DRAG BRACE LENGTH		307.3 (121)		182.9 (72)
WEIGHT, kg (LB)				
STRUTS	2 EA	1080.5 (2382)	1 EA	215.0 (474)
WHEELS	4 EA	213.2 (470)	2 EA	35.4 (78)
TIRES	4 EA	272.2 (600)	2 EA	43.5 (96)
BRAKES	4 EA	257.6 (568)		–

Figure 8.- Main landing gear.

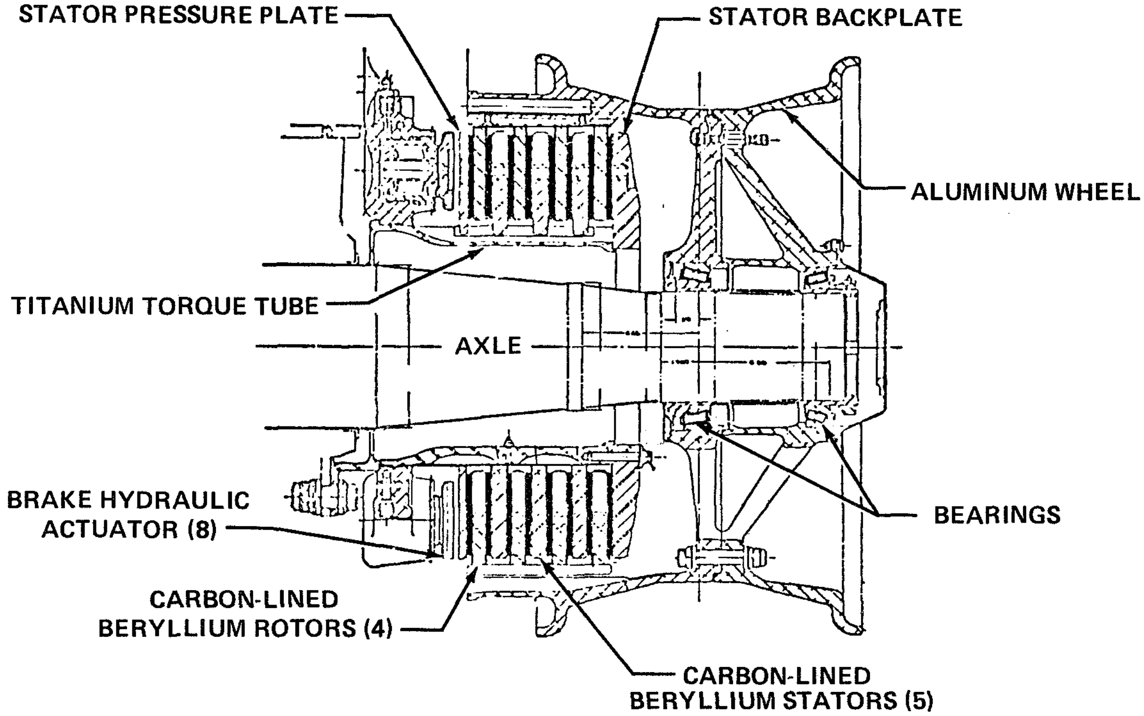


Figure 9.- Wheel brake assembly.

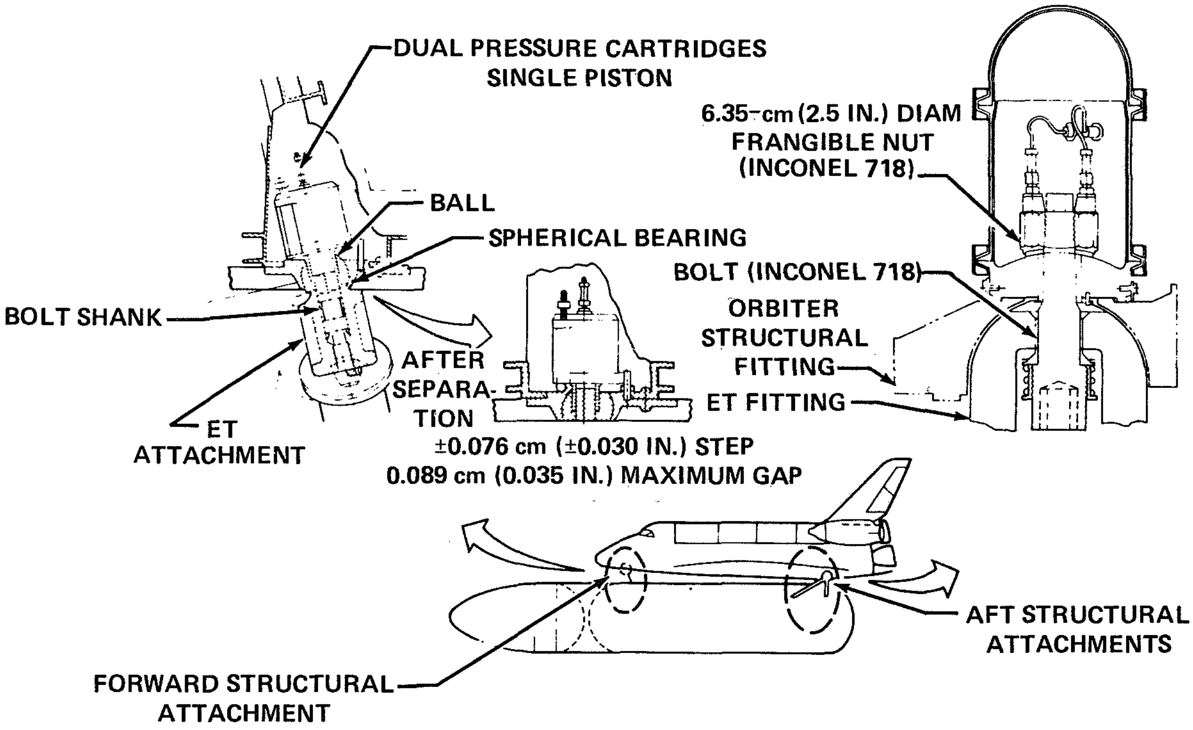


Figure 10.- Orbiter/external tank structural separation system.

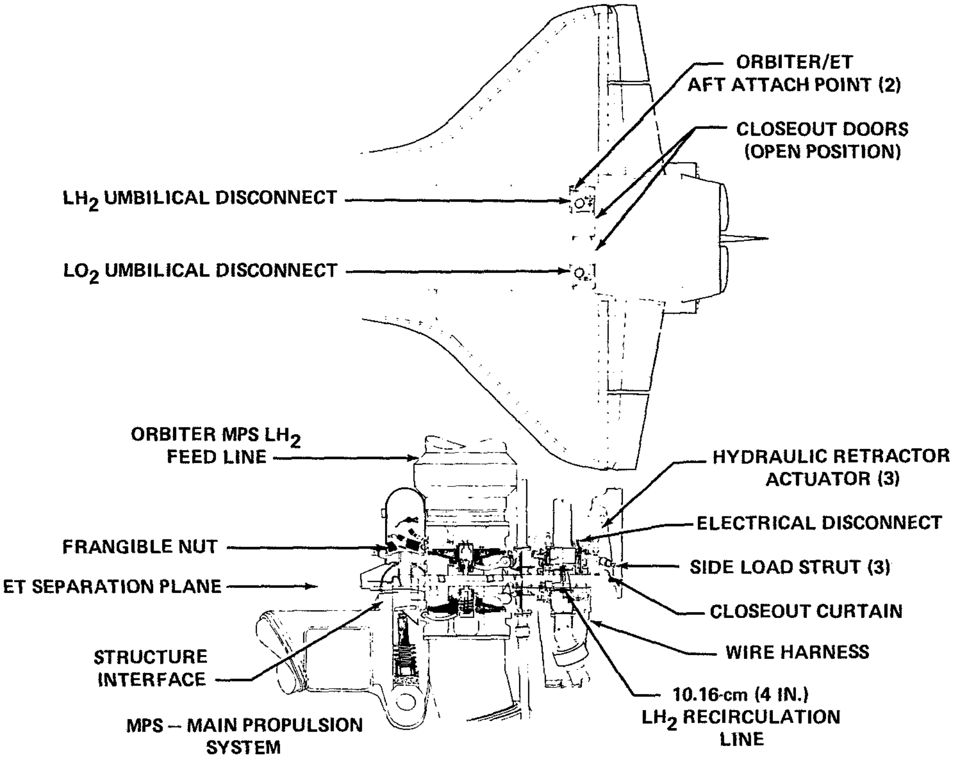


Figure 11.- Orbiter/external tank umbilical separation system.

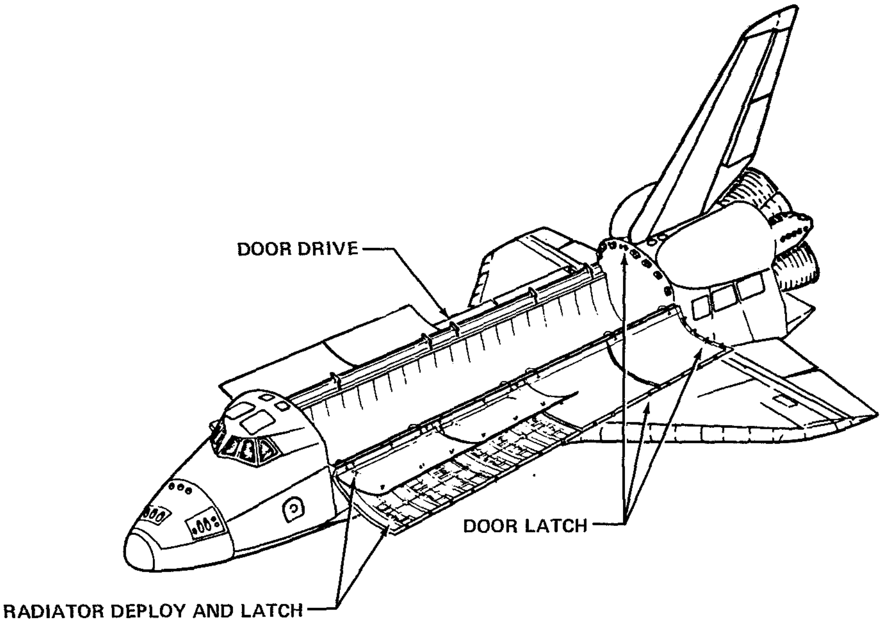


Figure 12.- Payload bay door/radiator system.

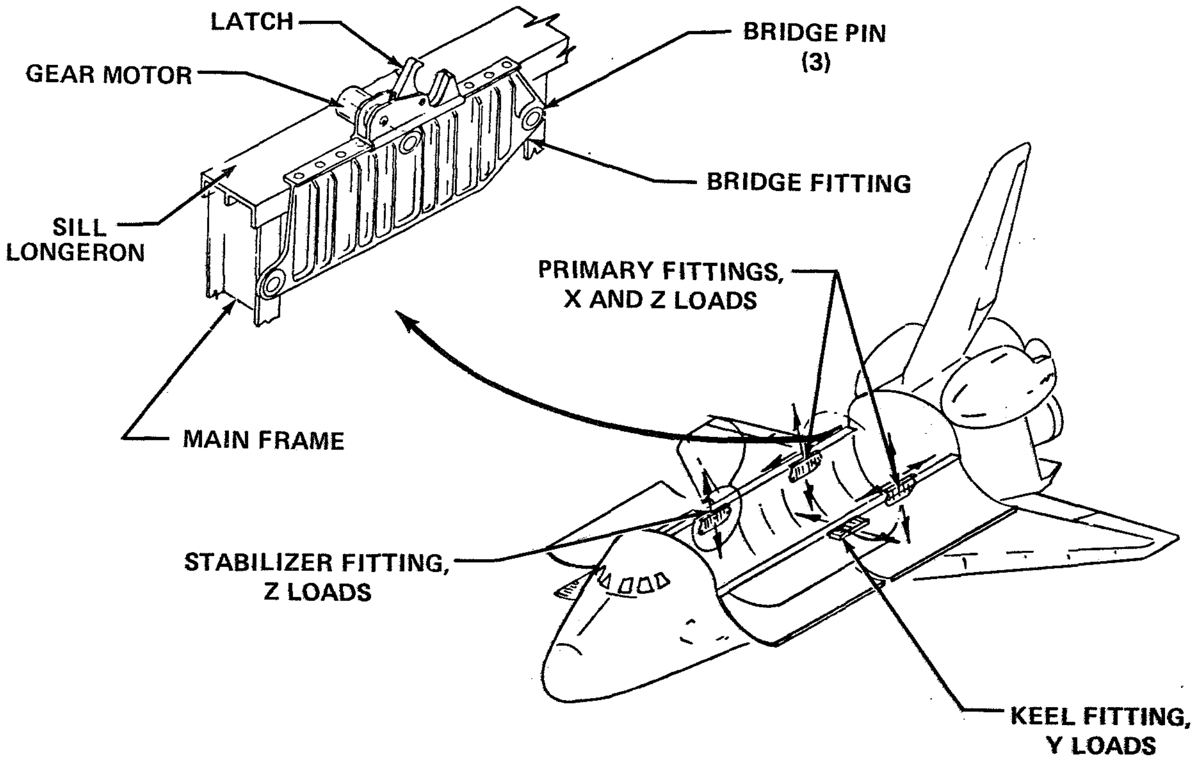


Figure 13.- Payload retention system.