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PROJECT GEMINI

A Technical Summary

by P. W. Malik and G. A. Souris

Prepared by
MCDONNELL DOUGLAS CORPORATION
St. Louis, Mo.
for Manned Spacecraft Center

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PROJECT GEMINI

A TECHNICAL SUMMARY

INTRODUCTION

Project Gemini was begun in November 1961 by the National Aeronautics and Space Administration as a follow-on program to Project Mercury, NASA's first manned space flight program. The Gemini program was concluded ahead of schedule and below anticipated costs in November 1966 after the successful flights of two unmanned and ten manned spacecraft. The McDonnell Company was the prime contractor for both Mercury and Gemini.

The Mercury program, in which the last of six manned space flights was made in May 1963 demonstrated that spacecraft could be launched on precise schedules, and could safely orbit the earth, re-enter, and land. Gemini showed that man could survive long periods in space and that spacecraft could rendezvous and dock with a target vehicle in space and could use the target vehicle's propulsion system to achieve a new orbit.

Thus, Gemini achieved all its goals (Table 1) to pave the way for the Apollo flights and for other space programs, such as the Air Force Manned Orbiting Laboratory.

MODULAR DESIGN CONCEPT

Gemini's modular system design, which replaced Mercury's stacked system concept, simplified construction, testing, and operation of the spacecraft. Modular design permitted virtually independent design, qualification, and system checkout. Reliability analysis was possible without the complications of interacting influences of associated systems.

Spacecraft 7/6 mission, hailed as one of the high points of the program, was made possible because launch crews were able, despite a tight time schedule, to remove the rendezvous and recovery section (R & R) of Spacecraft 7 and modify it for tracking by Spacecraft 6. Another example of the effectiveness of the modular design was the Gemini XII mission which was totally changed and replanned within two weeks.

Gemini system design was simplified by extensive use of manual sequencing and systems management, utilizing the astronaut's ability to diagnose failures and to take corrective action.

TABLE 1 GEMINI SPACECRAFT FLIGHT HISTORY

MISSION	TYPE	LAUNCHED	MAJOR ACCOMPLISHMENTS
GEMINI I	UNMANNED 64 ORBITS	8 APR. 1964	DEMONSTRATED STRUCTURAL INTEGRITY.
GEMINI II	UNMANNED SUBORBITAL	19 JAN. 1965	DEMONSTRATED HEAT PROTECTION AND SYSTEMS PERFORMANCE.
GEMINI III	MANNED 3 ORBITS	23 MAR. 1965	DEMONSTRATED MANNED QUALIFICATIONS OF THE GEMINI SPACECRAFT.
GEMINI IV	MANNED 4 DAYS	3 JUNE 1965	DEMONSTRATED EVA AND SYSTEMS PERFORMANCE FOR 4 DAYS IN SPACE.
GEMINI V	MANNED 8 DAYS	21 AUG. 1965	DEMONSTRATED LONG-DURATION FLIGHT, RENDEZVOUS RADAR CAPABILITY, AND RENDEZVOUS MANEUVERS.
GEMINI VI	MANNED 2 DAYS RENDEZVOUS (CANCELLED AFTER GATV FAILURE)	25 OCT. 1965	DEMONSTRATED DUAL COUNTDOWN PROCEDURES (GATV AND GLV-SPACECRAFT), FLIGHT PERFORMANCE OF TLV AND FLIGHT READINESS OF GATV SECONDARY PROPULSION SYSTEM. MISSION CANCELED AFTER GATV FAILED TO ACHIEVE ORBIT.
GEMINI VII	MANNED 14 DAYS RENDEZVOUS	4 DEC. 1965	DEMONSTRATED 2-WEEK FLIGHT AND STATION KEEPING WITH GLV STAGE II; EVALUATED "SHIRT SLEEVE" ENVIRONMENT; ACTED AS RENDEZVOUS TARGET FOR SPACECRAFT 6; AND DEMONSTRATED CONTROLLED RE-ENTRY WITHIN 7 MILES OF PLANNED LANDING POINT.
GEMINI VI-A	MANNED 1 DAY	15 DEC. 1965	DEMONSTRATED ON-TIME LAUNCH PROCEDURES, CLOSED-LOOP RENDEZVOUS CAPABILITY, AND STATION KEEPING TECHNIQUES WITH SPACECRAFT 7.
GEMINI VIII	MANNED 3 DAYS RENDEZVOUS AND DOCK (TERMINATED IN REV. 7)	16 MAR. 1966	DEMONSTRATED RENDEZVOUS AND DOCKING WITH GATV, CONTROLLED LANDING, EMERGENCY RECOVERY, MULTIPLE RESTART OF GATV IN ORBIT. SPACECRAFT MISSION TERMINATED EARLY DUE TO ELECTRICAL SHORT IN CONTROL SYSTEM.
GEMINI IX	MANNED 3 DAYS RENDEZVOUS AND DOCK, AND EVA	3 JUNE 1966	DEMONSTRATED THREE RENDEZVOUS TECHNIQUES. EVALUATED EVA WITH DETAILED WORK TASKS. DEMONSTRATED PRECISION LANDING CAPABILITY.
GEMINI X	MANNED 3 DAYS RENDEZVOUS AND DOCK, AND EVA	18 JULY 1966	DEMONSTRATED DUAL RENDEZVOUS USING GATV PROPULSION FOR DOCKED MANEUVERS. DEMONSTRATED REMOVAL OF EXPERIMENT PACKAGE FROM PASSIVE TARGET VEHICLE DURING EVA. EVALUATED FEASIBILITY OF USING ONBOARD NAVIGATIONAL TECHNIQUES FOR RENDEZVOUS.
GEMINI XI	MANNED 3 DAYS RENDEZVOUS AND DOCK, AND EVA	12 SEP. 1966	DEMONSTRATED FIRST-ORBIT RENDEZVOUS AND DOCKING. EVALUATED EVA. DEMONSTRATED FEASIBILITY OF TETHERED STATION KEEPING. DEMONSTRATED AUTOMATIC RE-ENTRY CAPABILITY.
GEMINI XII	MANNED 4 DAYS RENDEZVOUS AND DOCK, AND EVA	11 NOV. 1966	DEMONSTRATED OPERATIONAL CAPABILITY TO PERFORM COMPLEX AND LONG-DURATION EVA WITH NO NOTICEABLE ASTRONAUT FATIGUE. (THREE SEPARATE EVA OPERATIONS TOTALLED ABOUT 5.5 HOURS.)

EVA – EXTRAVEHICULAR ACTIVITY
GATV – GEMINI-AGENA TARGET VEHICLE
GLV – GEMINI LAUNCH VEHICLE
TLV – TARGET LAUNCH VEHICLE

SAFETY FIRST

The Gemini program stressed safety. As a result the ten manned Gemini Spacecraft flew a total of 969 hr and 56 min without an injury to any of the 16 crewman. All crewmen were recovered in excellent physical condition.

Major Spacecraft Safety Features

Inertial Guidance System. - The spacecraft inertial guidance system (IGS) serves as a back-up to the launch-vehicle guidance system during the launch phase. (See GUIDANCE AND CONTROL SYSTEM, page 110.)

Ejection Seats and Retro-rockets. - Ejection seats and retro-rockets provide escape modes from the launch vehicle during the prelaunch and the launch phases. (See ESCAPE, LANDING AND RECOVERY SYSTEMS, page 208 and Retrograde Rocket System, page 156.)

Secondary Oxygen Bottles. - Two secondary oxygen bottles are provided, either of which will support the crew for one orbit and re-entry, if the primary oxygen supply is lost. All other flight safety components in the environmental control system (ECS) are redundant. (See ENVIRONMENTAL CONTROL SYSTEM, page 157.)

Visual Aids. - In the event that a loss of reference of the guidance platform should occur, the crew can control re-entry using out-the-window visual aids.

Re-entry Control System. - The re-entry control system (RCS) is completely redundant. It is composed of two identical but independent systems, either of which can be used to control the vehicle through re-entry. These systems are sealed with zero-leakage valves until they are activated shortly before retrograde. (See Re-entry Control System, page 145.)

Drogue Parachute. - A drogue parachute, which is normally deployed at 50,000 ft altitude after re-entry, backs up the RCS for stability until the main parachute is deployed. (See ESCAPE, LANDING AND RECOVERY SYSTEMS, page 208.)

Ejection Seats. - Ejection seats provide an escape mode if the recovery parachute fails to deploy or is damaged.

BASIC OBJECTIVES

Basic Objectives Of Gemini And How They Were Met

Continuous Program at Minimum Cost. - To provide a continuation program of manned space flight objectives at minimum cost with major milestones to be complete as soon as practical. Gemini was completed months ahead of the schedule that was estimated in early-1963. Spacecraft 2 through 12 each was delivered at least a month ahead of schedule.

Rendezvous, Docking and Maneuvering. - To rendezvous and dock with a second orbiting vehicle and then perform combined maneuvering. Rendezvous was first achieved by Spacecraft 6; Spacecraft 8 was the first to dock. Maneuvering in orbit using the Agena Target Vehicle was first achieved by Spacecraft 10.

Long Duration Missions. - To expose two astronauts and their life support systems to long-duration missions to prepare for future earth orbit and lunar flights. Spacecraft 5 remained in orbit for 8 days and Spacecraft 7 remained in orbit for 14 days, demonstrating man's capability in a space environment. The Apollo lunar trip is expected to take eight days.

Precision Re-entry, Landing, and Recovery. - To develop and exercise precision re-entry, landing, and recovery of manned spacecraft. From Gemini VI on, all spacecraft landed within seven miles of the aiming point. The last five Gemini Spacecraft came down within three miles of the target. All landings were made in the ocean. However, early designs had provided for land landings, but the rate of technological development for such landings did not keep pace with the remainder of the program so that the land landing capability for the spacecraft was subsequently abandoned.

Extravehicular Activity. - To undertake extravehicular activity to evaluate man's ability to perform tasks in a zero g environment. Although EVA was not an original objective of the Gemini program, it was made possible by the design of personnel hatches with mechanical latches which enabled the astronauts to open and close the hatches manually. Astronauts on Spacecraft 4, 9, 10, 11, and 12 performed EVA.

Scientific Investigations. - To utilize the Gemini Spacecraft as an experimental test platform for scientific investigations, including photography, biomedical experiments, communications, navigation, meteorology, etc. Experiments were carried on all manned Gemini Spacecraft.

STRUCTURES

SPACECRAFT DESCRIPTION

The Gemini Spacecraft, designed to provide life support for two orbiting astronauts, is a 7000 lb conical structure 18.82 ft long and 10 ft in diameter at its base. It is composed of two major assemblies, a re-entry module and an adapter module. Both structural bodies are all metal, of stressed skin and semimonocoque construction. In addition, the re-entry module is designed to withstand the extreme heat of re-entry. The general arrangement of the Gemini Spacecraft is shown in Fig. 1.

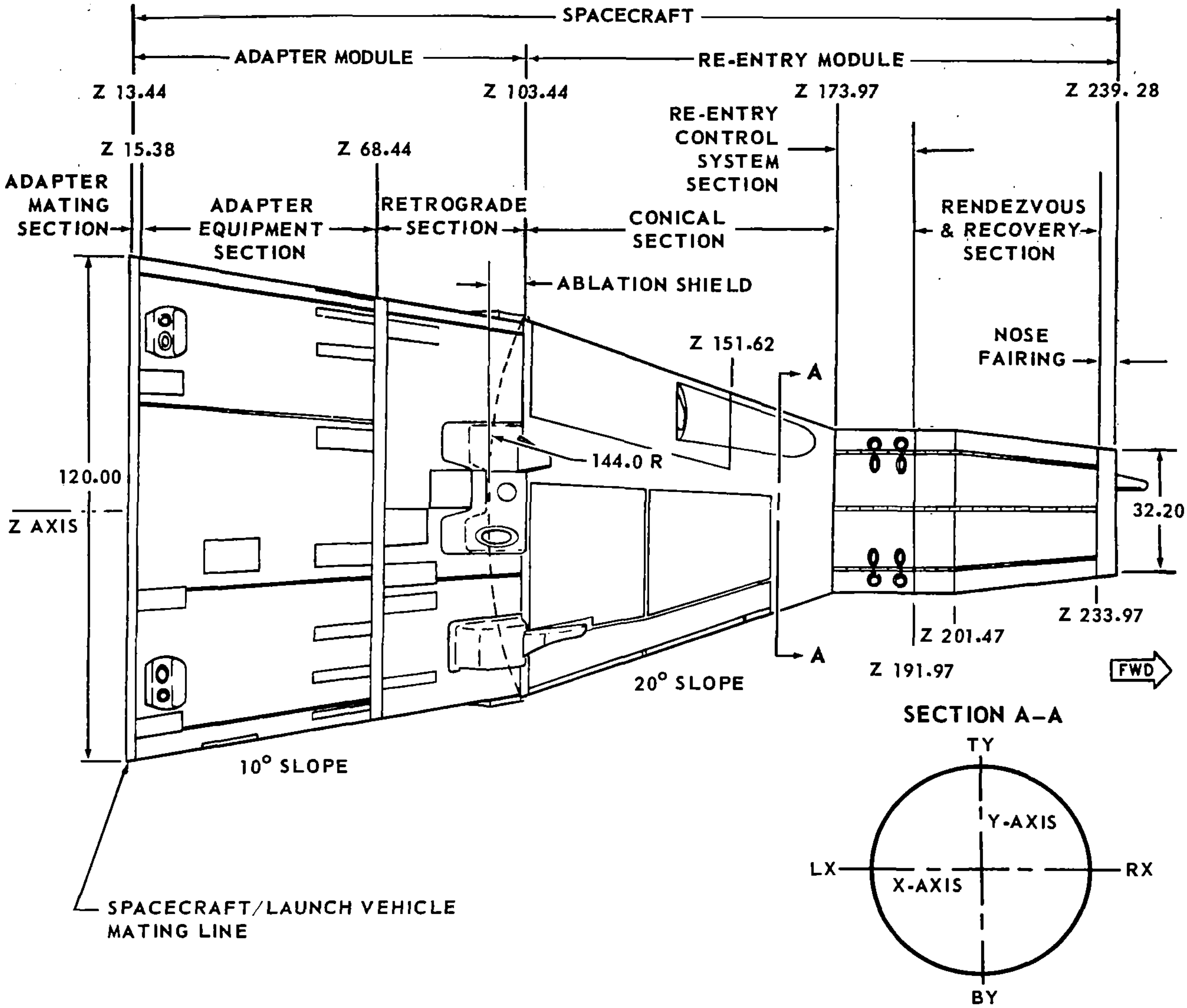


FIGURE 1 SPACECRAFT GENERAL ARRANGEMENT

The methods of construction and the materials used exemplify the care shown by McDonnell engineers to guarantee high strength-to-weight ratios. The requirement of heat resistance led to the choice of titanium and magnesium as the principal metals used in spacecraft fabrication. High-strength titanium bolts (Ti-6Al-4V) were used extensively; titanium was also used for the rings, stringers, interior skin, and bulkheads of the re-entry module. Aluminum was used inside the cabin where temperatures are not structurally critical.

Stringers and longerons were spaced around the circumference of the shell to carry nearly all axial and bending loads and to stiffen the framework. To protect against distortion to large, thin gauge panels, chemical milling rather than mechanical milling was employed.

For ease in identifying particular areas, the spacecraft is cut by two reference planes, one running longitudinally from adapter to nose, the other at right angles to this one.

Looking forward from the end of the adapter, one may divide a section of the spacecraft into four quadrants, thus creating four cardinal points - TY (Top Y) and BY (bottom Y) for the Y axis, and RX (right X) and LX (left X) for the X axis. A particular location on the circle is measured in degrees from any one of these points. (Ref Fig. 3)

The Z station locations are measured longitudinally from the rear of the spacecraft (adapter) and increase in magnitude as one approaches the nose. For example, the separation point between the re-entry module and the adapter is station Z 102.00 (in.); the nose of the spacecraft is station Z 239.53 (Z 239.28 plus 0.25 in. of ablative material).

Re-entry Module

The re-entry module, shown in Fig. 2, is composed of three primary sections, the cabin, the re-entry control system (RCS), and the rendezvous and recovery (R & R) sections. In addition, a heat shield is attached to the aft end of the cabin section, a nose fairing is fitted to the forward end of the rendezvous and recovery section, and a horizon sensor fairing is attached on the left side at the mating point of the cabin and the re-entry control system.

The R & R section is 47.31 in. long including the nose fairing. The re-entry control system section is 18.00 in. and the cabin section is 70.53 in. long, at the outer edge of the heat shield.

Rendezvous and Recovery Section. - This section houses the rendezvous radar equipment and the drogue, pilot, and main parachutes. The forward portion of the R & R section is a truncated cone, while the back portion is cylindrical. When the spacecraft was to dock with the Agena Target Vehicle, the R & R section also comprised three docking latch receptacles, a Fiberglas bumper, and a docking bar. A nose fairing made of Fiberglas-reinforced plastic laminate provides thermal protection for the sensitive radar equipment during the initial portions of powered flight. This fairing is jettisoned approximately 45 sec after ignition of the second stage engine of the launch vehicle by means of a short-time impulse pyrotechnic actuator. The R & R section is attached to the re-entry control system section by 24 frangible bolts. These bolts are tension separated during re-entry by an explosive strip of mild detonating fuse (MDF) after deployment of the pilot chute.

Re-entry Control System Section. - The RCS section is contained between the R & R section and the cabin section. This cylindrical section houses re-entry control system fuel and oxidizer tanks, and thrust chamber assemblies. In addition to accommodating the re-entry control system, which controls the spacecraft after the orbit attitude and maneuver system (OAMS) has been jettisoned, the RCS section also absorbs the loads induced by the deployment

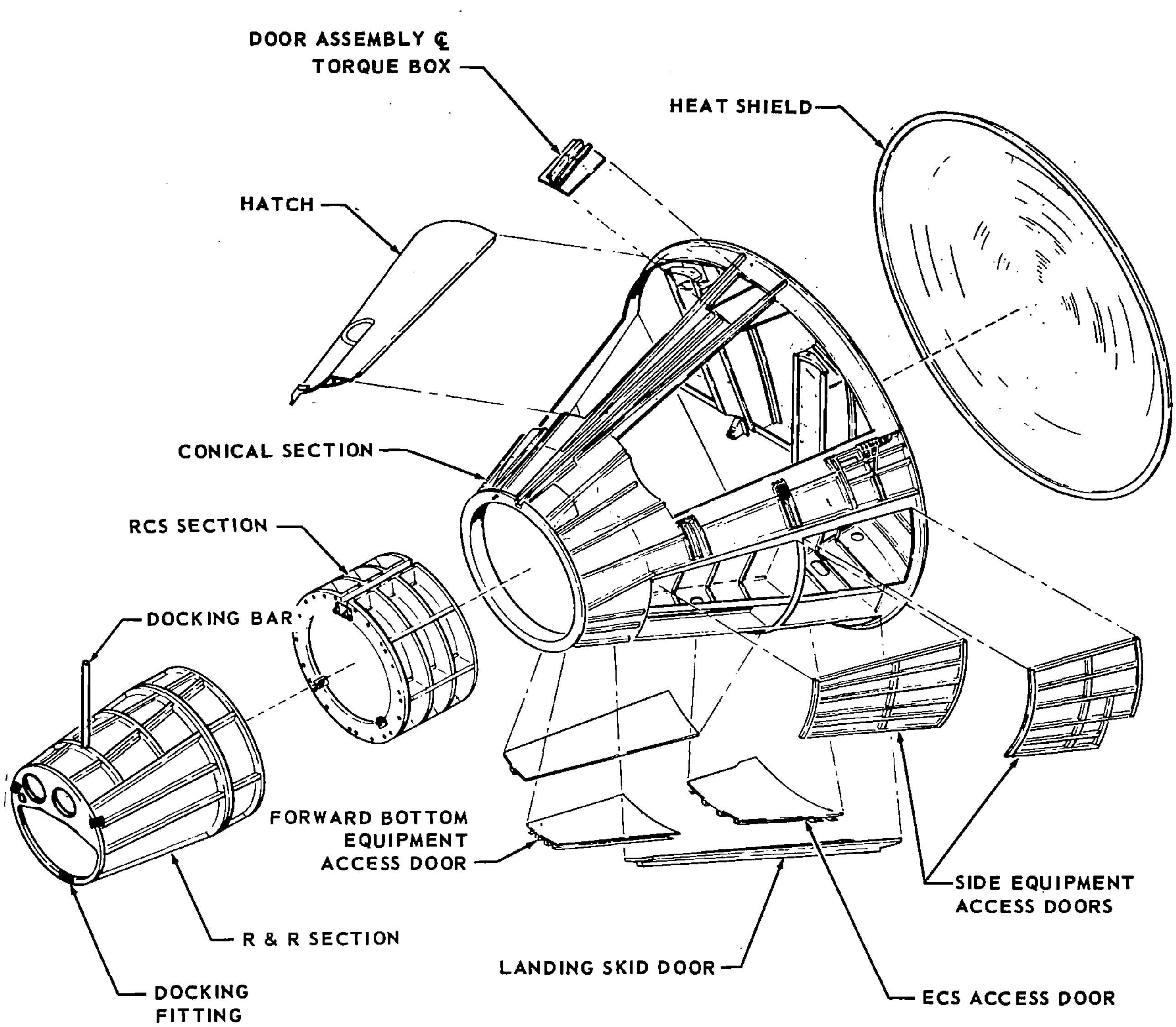


FIGURE 2 RE-ENTRY MODULE STRUCTURE

of the main parachute. The latter is attached to the parachute adapter assembly, which is installed on the forward face of the RCS section.

Cabin Section. - The cabin section lies between the RCS section and the adapter assembly. It is a truncated cone, 38.66 in. in diameter at the forward end and 90.00 in. in diameter at the aft end (with the heat shield attached). It consists of an internal pressure vessel, which is the crew station for the two astronauts and equipment bays located outside the pressure vessel. The heat shield, a major structural component of the cabin, is discussed in Heat Protection, page 13.

A. Internal Pressure Vessel. - The pressure vessel, in addition to housing the Gemini crew, contains electrical and life support equipment and various experimental devices. Accessible volume in the crew compartment (with the crew aboard) is approximately 55 cu ft.

The pressure vessel consists of a fusion-welded titanium frame to which are attached side panels, a hatch sill, and a fore and aft bulkhead. The side panels and pressure bulkheads are double thickness, thin-sheet titanium (0.01 in.), with the outer sheet beaded for stiffness.

In addition to fusion welding, resistance spot and seam welding are employed extensively throughout the pressure vessel to reduce the possibility of leakage. Individual titanium fusion weldments are made under a controlled inert gas atmosphere, then strain relieved to obtain a fully structural weld. Two hatches are hinged to the hatch sill for pilot ingress and egress.

The design of the pressure vessel not only provides an adequate crew station but also gives the pressure vessel a proper water flotation attitude. Structural design criteria for the pressure vessel require it to withstand an ultimate burst pressure of 12 psi and an ultimate collapsing pressure of three psi.

Two hatches, contoured to the shape of the cabin exterior, are symmetrically spaced on the top side of the pressure vessel. Each hatch, hinged on the outboard side, is manually operated by means of a handle and a mechanical latching mechanism. In an emergency, the hatches can be opened in a three-sequence operation employing pyrotechnic actuators. The actuators simultaneously unlock and open the latches, open the hatches, and supply hot gases to ignite the ejection seat rocket catapults.

An external hatch linkage fitting allows a recovery hatch handle to be inserted to open the hatches from the outside. The recovery hatch handle is stowed on the main parachute adapter assembly, located on the forward face of the RCS section. A hatch curtain is stowed along the hinge of each hatch. When the hatches are opened after a water landing, the curtains help keep water out of the cabin.

Each hatch incorporates a window, which contains three panes of glass, with an air space between each pane. The command astronaut's window has two outer panes of 96% silica glass and an inner pane of tempered aluminosilicate glass. For improved clarity while carrying out optical experiments, the inner pane of the copilot's window is a 96% silica panel with an optical transmission capability of more than 99%. The thickness of this pane has been increased from 0.22 to 0.38 in. to make it as strong as the aluminosilicate glass. Each pane, with the exception of the outer pane, is coated to reduce reflection, glare, and ultraviolet radiation.

The personnel access hatches are of skin and beam construction. Silicon rubber seals around each hatch sill and around the two inner panes of glass prevent the leakage of cabin pressure.

B. Equipment Bays - Three major equipment bays, designed to contain a variety of electrical and electronic equipment, are outside the cabin pressure vessel. Two of these bays are outboard of the side panels and one bay is beneath the pressure vessel floor. Unlike the Mercury Spacecraft, which had nearly all its systems inside the pressure shell, the Gemini Spacecraft has most of its system components in these unpressurized equipment bays. These components either require no pressurization or are internally pressurized. Since equipment is normally only one layer deep within the compartments, launch crews can remove an access door, quickly pull out a malfunctioning unit, insert a new one, reinstall the access door, and proceed with the launch.

Two main landing gear wells are located below the side equipment bays. Originally these wells were intended to house equipment for ground landings; however, this requirement was never put into practice. Consequently, the wells are used to house additional experiment equipment on some spacecraft; on other missions the wells carry ballast or remain empty.

Two structural doors are provided on each side of the cabin to allow access to the side equipment bays. The two main landing gear wells also have doors. On the bottom of the cabin, between the landing gear doors, two additional access doors are installed. The forward door allows access to the lower equipment compartment and the aft door provides access to the environmental control system (ECS) compartment.

C. Hoist Loop - A spring-loaded hoist loop, located near the heat shield between the hatch openings, is erected after landing to engage a hoisting hook for spacecraft retrieval.

Adapter Module

The adapter module extends from the end of the re-entry module heat shield to the spacecraft launch vehicle mating line. A truncated cone, the adapter assembly consists of three sections: the retrograde section, the equipment section, and the launch-vehicle mating section. The entire assembly is 90.00 in. long with a forward diameter of 88.30 in. and an aft diameter of 120.00 in. It contains the systems and equipment needed on long-duration orbital flights and provides the interface between the spacecraft and the launch vehicle. The basic adapter structure is illustrated in Fig. 3 and 4.

The adapter structure consists of circumferential aluminum rings, extruded magnesium alloy stringers, and a magnesium skin. The free end of the T-shaped stringers is a tube. Liquid coolant flows through this tube and transfers heat to the adapter skin for radiation into space.

The adapter is joined to the re-entry module by three titanium retaining straps external to the structure of both the re-entry module and the adapter section. Pyrotechnic separation rings are provided between the retrograde and the equipment sections, and between the equipment and the launch-vehicle mating sections.

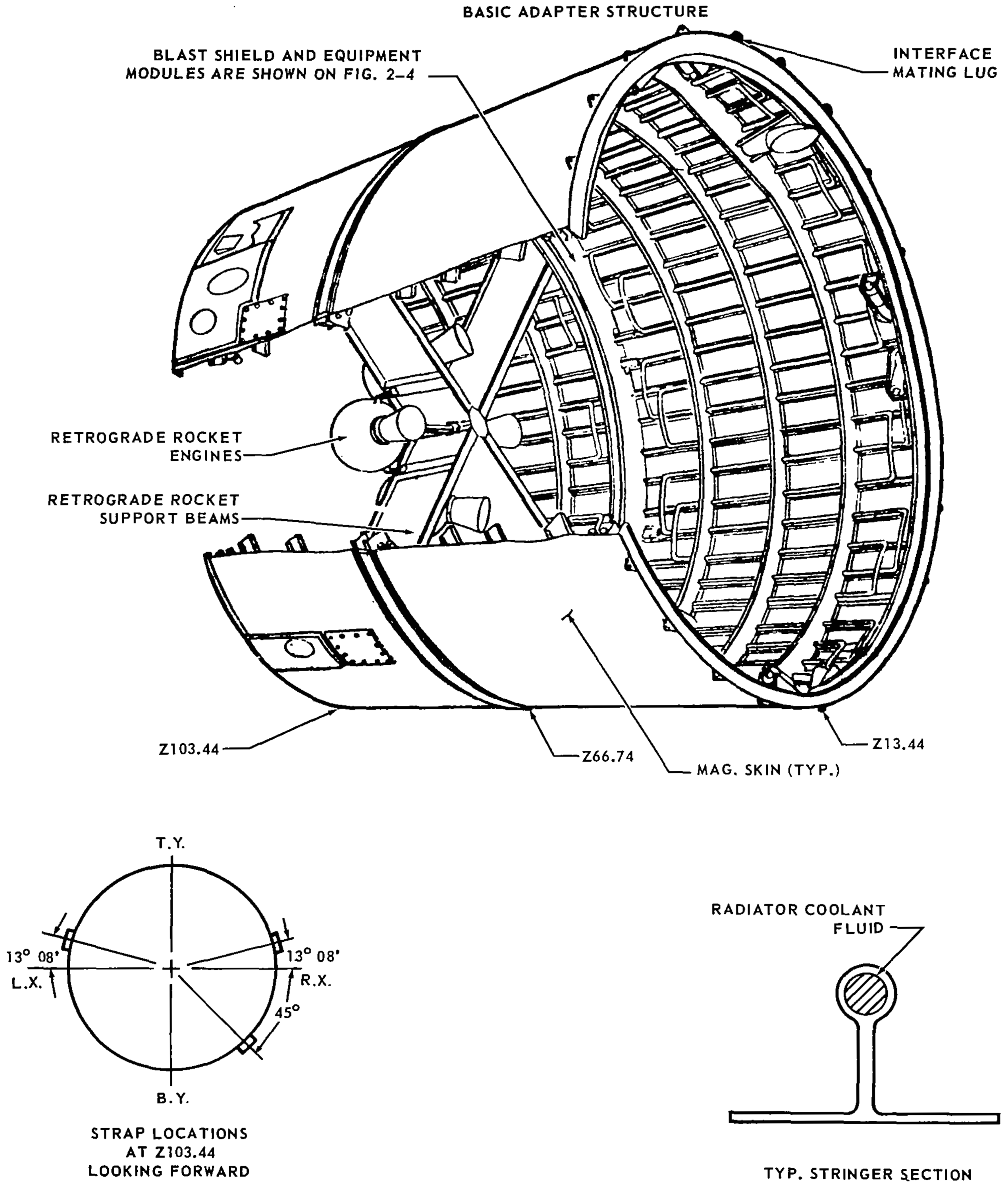


FIGURE 3 ADAPTER STRUCTURE

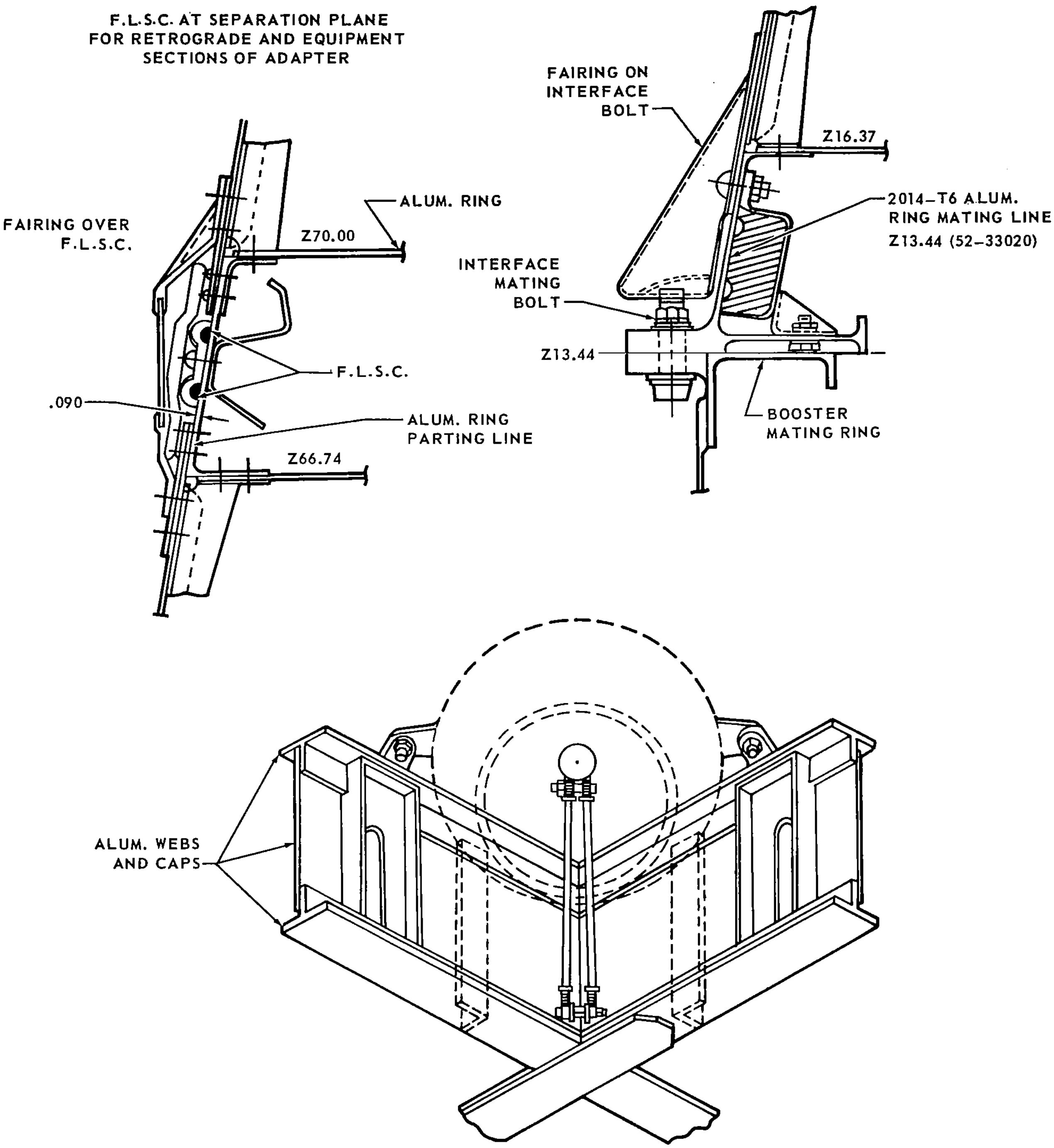


FIGURE 3 ADAPTER STRUCTURE (Continued)

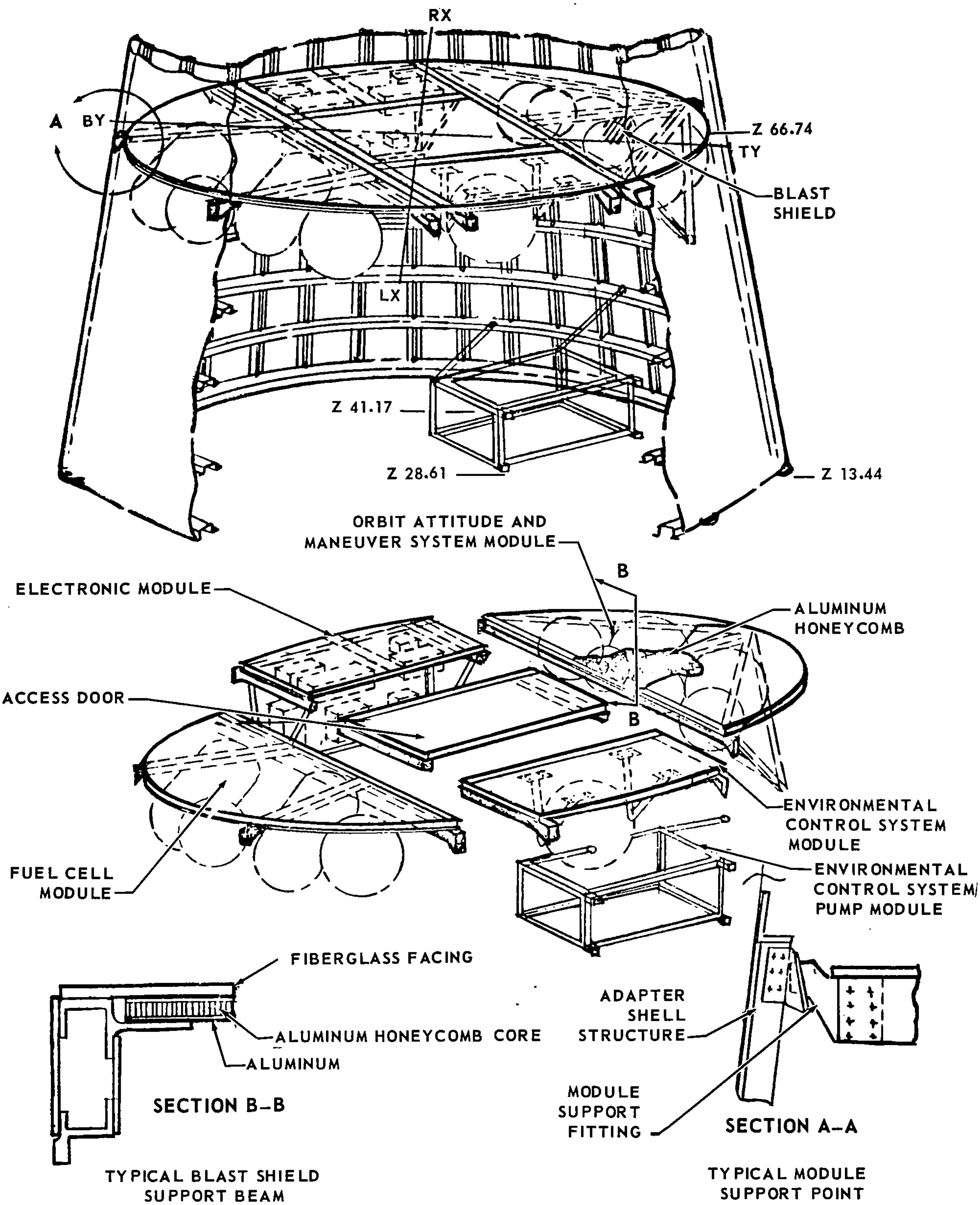


FIGURE 4 ADAPTER EQUIPMENT MODULE STRUCTURE

Retrograde Section. - At the small end of the adapter module is the retrograde section, 30 in. in length. The primary function of the retrograde section is to support the four retrograde rockets and six of the OAMS thrust chamber assemblies. To support the retrograde rockets, aluminum I-beams are assembled as a cruciform, with one retrograde rocket mounted in each quadrant.

Prior to retrofire a flexible linear-shaped charge cuts the adapter, separating the equipment from the retrograde section. After retrofire the three titanium retaining straps are cut by three flexible linear-shaped charges, severing the re-entry module from the retrograde section. These retaining straps are located to coincide with wire bundles and fluid lines, which are also cut by the shaped charges, thus minimizing the number of charges required.

Equipment Section. - The equipment section, which comprises the larger end of the adapter module, provides the space and the attach points for four major system modules, plus individual components. The four principal modules mounted in the equipment section are the orbit attitude and maneuvering system (OAMS) propellant tanks, the fuel cell (or battery) module, the environmental control system (ECS) primary oxygen supply, and the electronics module. These four modules are independent of one another, but their support panels and an access door combine to form a blast shield. This shield protects the equipment section and the dome of the Titan launch vehicle from excessive (explosion-causing) heat if it is necessary to fire the retro-rockets in an abort.

In addition to the four principal equipment modules, this section also houses the coolant supply, the water storage tanks, and ten OAMS thrust chamber assemblies. A gold deposited Fiberglas temperature control cover over the open end of the adapter protects the equipment from solar radiation after separation from the launch vehicle.

Mating Section. - The spacecraft is mated to the Titan II launch vehicle by a forged and machined aluminum alloy ring, 120 in. in diameter. This ring, approximately three in. wide, is joined to the launch vehicle mating ring by 20 bolts. The launch vehicle mating ring has four index marks, spaced at 90 degree intervals (at TY, BY, RX, and LX of the adapter), to insure proper alignment between the spacecraft and the launch vehicle. A flexible linear-shaped charge is fired to sever the adapter section approximately 1-1/2 in. above the spacecraft/launch vehicle mating point.

Heat Protection

During re-entry the spacecraft flies with the heat shield forward. This protects the forebody of the re-entry module from excessive heat flux during this critical mode. The rest of the spacecraft body is protected by two kinds of heat resistant shingles, Rene 41 and beryllium.

Heat Shield. - The heat shield is a dish-shaped structure that forms the large end of the re-entry module. The heat shield is an ablative device which protects the re-entry assembly from the extreme heat of re-entry into the earth's atmosphere. It is attached to the aft end of the cabin section by eighteen 0.25-in.-diameter bolts. Ninety inches in diameter, the shield has a spherical radius of 144 inches.

The ablative substance of the Gemini heat shield is a paste-like material which hardens in standard atmosphere after being poured into a honeycomb form.

Starting with a load-carrying Fiberglas sandwich structure consisting of two 5-ply faceplates of resin-impregnated glass cloth separated by an 0.65 in. thick Fiberglas honeycomb core, an additional Fiberglas honeycomb is bonded to the convex side of the sandwich and filled with Dow-Corning DC-325 silicone elastomer. The extreme edge of the heat shield is a circular Fiberite ring. It is interesting to note that the basic ablative substance of the heat shield, developed by the McDonnell Company, is now being produced for commercial applications.

Heat Resistant Shingles. - These overlapping Rene 41 and beryllium shingles, which provide both aerodynamic and heat protection, also hold shaped pads of flexible insulation in place. The beaded (corrugated) Rene 41 shingles (0.016 in. thickness) on the sides of the cabin are composed of 53% nickel, 19% chromium, 11% cobalt, 9.75% molybdenum, 3.15% titanium, 1.6% aluminum, 0.09% carbon, 0.005% boron, and less than 2.75% iron. The shingles are identical in composition and manufacturing technique to those used on Mercury. Extra large holes at the attachment bolts allow the shingles to expand during aerodynamic and solar heating. Oversized washers cover these holes to minimize heat and air penetration.

The R & R and RCS section surfaces are unbeaded shingles of cross-rolled beryllium. The plate is supplied to McDonnell in several sheet sizes ranging in thickness from 0.300 in. to 0.555 in. Shingles are finished by McDonnell to thicknesses, depending on spacecraft location, of 0.090 in. to 0.280 in. The shingles are attached to the spacecraft by beryllium retainers fabricated from similar plates.

Beryllium shingles used on Project Mercury were fabricated from hot-pressed beryllium blocks. The requirements for Gemini rendezvous flights were almost twice the strength and impact resistance available with hot-pressed beryllium blocks and this was provided by the cross-rolling technique.

Under the beryllium shingles are Thermoflex RF blankets held in place by a titanium mesh attached to the stringers. The outer surfaces of the rings and stringers are insulated with 0.0015 in. Inconel-foil-encased Min-K in Fiberglas channels.

Both Rene 41 and beryllium shingles are coated on the outer surface with a blue-black ceramic paint, to permit high thermal radiation from the spacecraft. The inner surface of the beryllium shingles has a very thin gold coating to attenuate thermal radiation into the spacecraft. The outer surface of the adapter module is coated with white ceramic paint and the inner surface is covered with aluminum foil to reduce emissivity. The heat protection devices are pictured in Fig. 5.

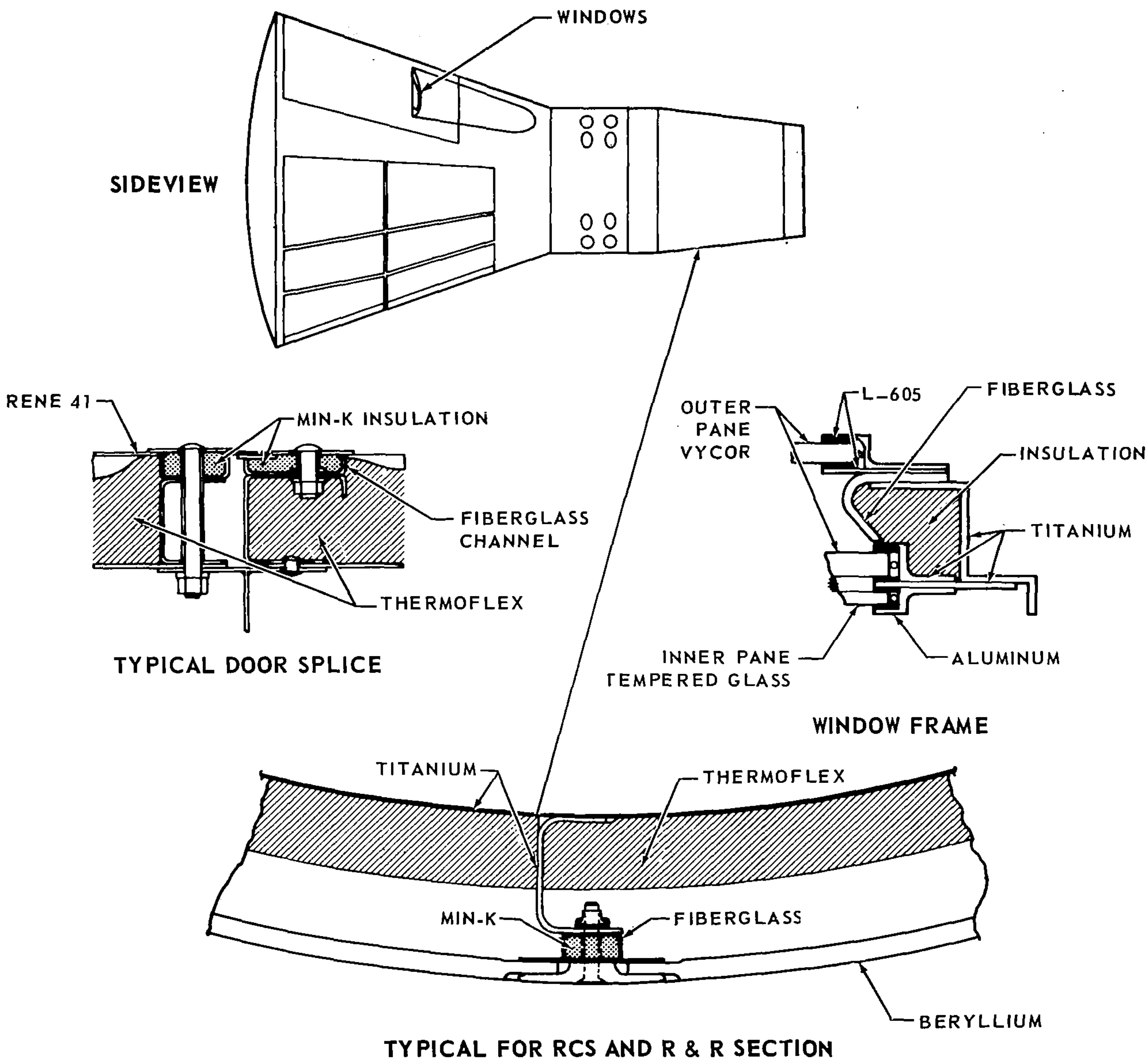


FIGURE 5 HEAT PROTECTION

Failure Summary And Analysis

No structural failures occurred during development or qualification testing. However, a change was incorporated in Spacecraft 6 and up as the result of an anomaly on Gemini IV in the hatch manual control mechanism. This mechanism change is illustrated in Fig. 6. The pinion drive shaft, which works the hatch latch linkages when the manual handle is operated, is driven by engagement of drive and gain pawls in a ratchet configuration. The automatic return of drive and gain pawls failed to operate positively due to frictional effects. This necessitated manual operation of the selector in Gemini IV. The efficiency of this mechanism has been greatly increased by reducing frictional effects and by increasing the return mechanical advantage by a factor of 10.

Furthermore, a sawtooth "gain hold" device is now installed on the hatch sill for use with the hatch closing device, to assist in holding the hatch closed against seal pressure just prior to the latching operation.

The redesigned manual control mechanism was satisfactorily endurance tested in temperature and pressure environments for over 1000 cycles.

Also, after re-entry of Spacecraft No. 2 localized heating was apparent in one area. Two small holes were burned in the shingles due to air flow around one of the umbilical fairings. To reduce localized heating the fairing was reconfigured, the affected shingles were increased in thickness to .025 in. and the angle of attack was lowered.

STRUCTURAL QUALIFICATION TEST PROGRAM

Most of the major Gemini structural and structural dynamic tests were performed between July 1963 and April 1965 on static No. 3 and static No. 4 test articles. Other representative static test articles were used to test the rendezvous and recovery section (R & R) and re-entry control system (RCS) section.

Structural Test Vehicles And Testing

Test Vehicles. - In addition to building twelve flight articles and seven target docking adapters, McDonnell was responsible for the manufacture of six boilerplate spacecraft and four static articles, plus static adapters and miscellaneous test vehicles. A brief description of these major test vehicles and their history is included here.

A. Boilerplate Re-entry Vehicles:

1. Boilerplate No. 1 - Boilerplate No. 1 was a steel mock-up of the re-entry module which was used primarily in parachute development testing. Ballast was installed to simulate spacecraft weight and cg. After fabrication

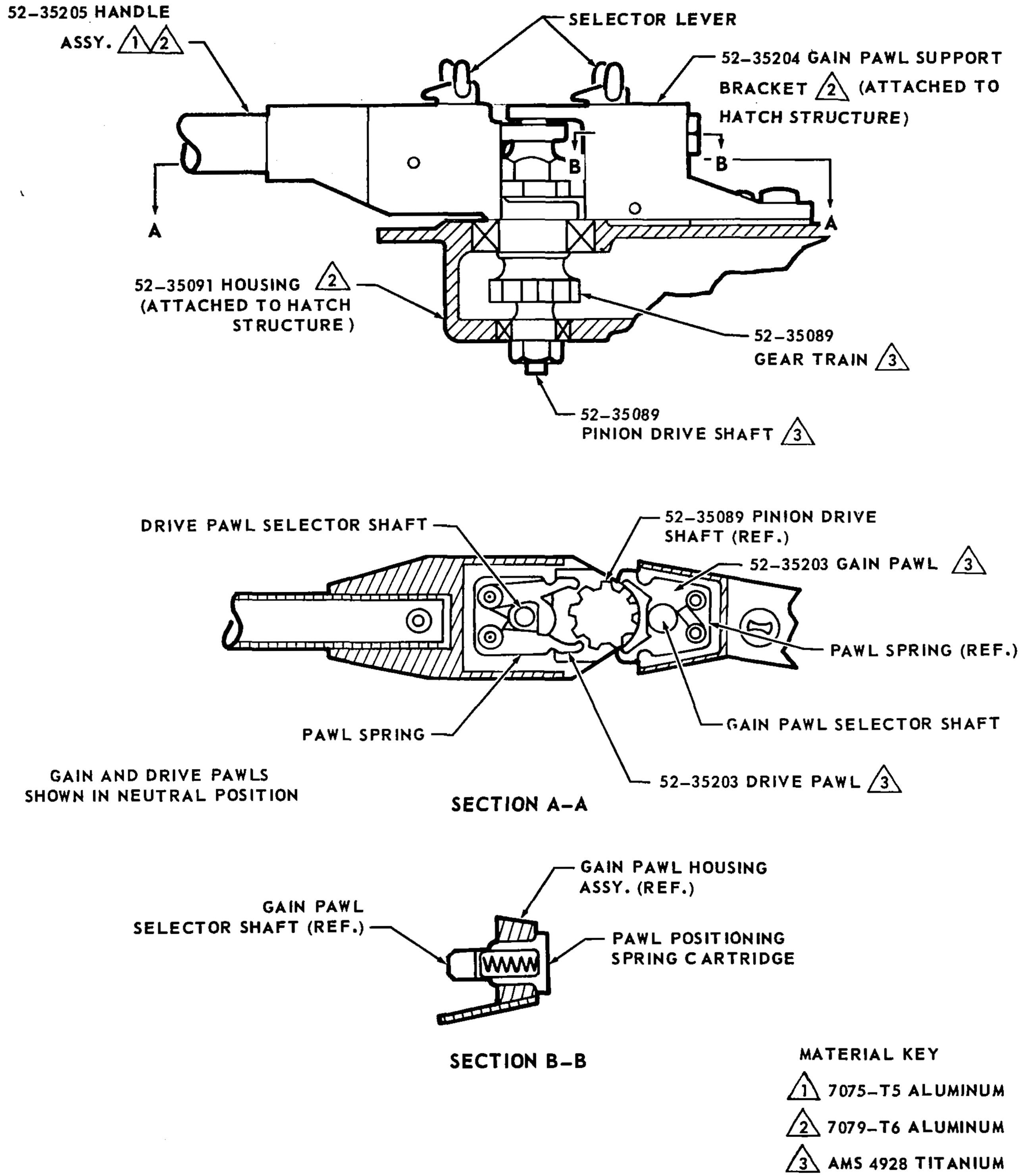


FIGURE 6 HATCH MANUAL CONTROL MECHANISM

at the McDonnell plant, the unit was shipped to Northrop-Ventura on 31 July 1962 and was thereafter assigned to North American Aviation for testing. The unit was destroyed on 30 July 1963 while undergoing drop tests at El Centro, California.

2. Boilerplate No. 2 - Boilerplate No. 2 was a welded steel mock-up of the pressurized cabin corresponding closely to the flight article in shape and volume. It was used for functional evaluations of the gaseous oxygen components of the environmental control system (ECS) under simulated mission environments; the effects of solar radiation and equipment heat exchange were especially significant. A complete ECS and related crew station controls were installed. The ECS was instrumented to record and evaluate system performance during normal and secondary modes of operation. Test conditions simulated regular mission phases as well as failures and other abnormal operations including crewman "ejection" during the prelaunch and the re-entry modes.

Boilerplate No. 2, with its complete ECS installed, was first used in evaluation testing in the McDonnell laboratory. On 9 April 1963, it was shipped to MSC, Houston, for further tests at that site.

3. Boilerplate No. 3 - Boilerplate No. 3 was a welded steel mock-up of the re-entry module, aerodynamically similar to the flight article. The boilerplate was utilized for ejection seat development sled runs. It contained the two seats, the seat rails, and the seat actuating mechanisms. A removable fairing simulated the adapter retrograde section. Boilerplate hatches were of the correct shape, but were fixed in the open position because no hatch sequencing tests were intended.

Boilerplate No. 3 was shipped to Weber Aircraft in July 1962. The first sled drag run took place on 9 November 1962, when extensive damage to the vehicle occurred due to the failure of one of the pusher sled rockets. The unit was subsequently repaired and utilized in further sled testing.

4. Boilerplate No. 3A - Boilerplate No. 3A had essentially the same structure as No. 3, modified by the installation of a production large pressure bulkhead, a seat rail torque box, two hatch sills, two side panels, two "light" hatches, and two flipper doors.

The unit was subjected to hatch firing functional tests at the McDonnell laboratory prior to delivery to Weber Aircraft. At Weber it underwent escape system qualification tests, comprising both SOPE (simulated off-the-pad ejection) tests and sled runs. Complete pyrotechnic system tests and sequencing were included in the program. These tests were performed at Weber during the greater part of 1964 and continued into the first months of 1965.

5. Boilerplate No. 4 - Boilerplate No. 4 was built by Weber Aircraft and delivered to McDonnell on 21 October 1963. Of aluminum skin and stringer construction, it was designed to carry ballast adjusted to the weight, cg and moment of inertia of a production spacecraft. The original intent was to utilize this boilerplate in evaluating the skid landing gear. These tests were deleted, however, when the ground landing mode was scrubbed for the

entire Gemini program. Instead, boilerplate No. 4 was used in a series of drop tests at the McDonnell facility.

6. Boilerplate No. 5 - A welded steel mock-up of the re-entry module, boilerplate No. 5 was designed for use in the Gemini parachute development program. It contained provisions for ballast to simulate spacecraft weight and cg. The unit was shipped to Northrop-Ventura on 31 August 1962, and employed in parachute testing. Subsequent to these tests, boilerplate No. 5 was refurbished and converted into static article No. 4A (18 September 1964), in which capacity it was utilized in high-altitude-drogue qualification tests along with its counterpart, static article No. 7.

B. Static Re-entry Vehicles:

1. Static No. 1 - Static article No. 1 was cancelled by agreement between McDonnell and the NASA; however, its re-entry module and adapter were reassigned to Spacecraft 3A.

2. Static No. 2 - Static No. 2 was intended to be a manned re-entry module designed specifically for qualifying the NAA Paraglider. The unit was cancelled when the paraglider was deleted from the program.

3. Static No. 3 - Static article No. 3 consisted of a complete re-entry module of the early paraglider configuration and an adapter module. The principal difference between it and the standard re-entry vehicle was the addition of the paraglider torque box structure, located between the hatches. (This torque box could accommodate parachute fittings, enabling the vehicle to be employed in either the paraglider or the parachute configuration.) The unit was delivered to the NASA on 15 May 1963. Static tests for this vehicle included landing conditions, parachute support structure tests, launch, abort, re-entry and heat shield back-up structure tests. For the launch and abort tests the Martin No. 2 adapter was mated to the re-entry module. Following the completion of its test program, static article No. 3 was reassigned to the Manned Orbital Laboratory (MOL) program for testing.

4. Static No. 4 - Static article No. 4 has the same structural configuration as No. 3 except that in addition it contained dummy equipment to simulate the mass and cg of the flight article. This vehicle was designed to undergo dynamic response tests, tests of the seat and hatch back-up structure, and ultimate pressurization tests. In addition, hoist loop and support tests and water drop tests were performed. The unit was delivered on 18 April 1963. After completion of the test program, this vehicle was reassigned to the MOL program.

5. Static No. 5 - Static article No. 5 had a complete re-entry vehicle structure (i.e., no adapter) and was designed for flotation stability tests and as an egress trainer. All equipment exterior to the pressure vessel and critical to flotation was carefully simulated to assure the proper water flotation attitude. In addition, ballast was installed to simulate the correct weight and cg as incorporated into the Spacecraft 3 configuration, the program's first manned vehicle.

The water flotation tests were successfully completed at the McDonnell facility and the unit was subsequently modified to the egress trainer configuration. This modification included the installation of those systems normally operative at splashdown - i.e., a partial ECS and communications system. All inoperative equipment exterior to the pressure vessel was simulated in this configuration; ballast to simulate correct weight and cg was also provided. The pressure vessel contained dummy ejection seats, a partially operative instrument panel and operating recovery equipment. Provisions for the installation of landing gear were included, although the gear was never installed. At the completion of the modification, the unit was delivered to the NASA for egress training.

6. Static No. 6 - Static No. 6 was to have been a back-up vehicle to static No. 2, but was cancelled with deletion of the paraglider.

7. Static No. 7 - Static article No. 7 consisted of a boilerplate pressure vessel and heat shield and a production RCS section and an R & R section. Since the function of this unit was to qualify the parachute recovery system, no adapter module was incorporated. However, all systems required to completely qualify the drogue, pilot and main parachute assemblies were installed. The unit was delivered to the NASA on 2 January 1964.

C. Static Adapters:

1. Static Adapter No. 1 - Static adapter No. 1 was designed for structural dynamics and related structural tests. The unit was completed at the McDonnell plant and shipped to the Martin Company for the test program early in 1963. In December 1963 it was returned to McDonnell for a series of dynamic response tests. Dummy equipment of proper weight and cg was then mounted in the adapter to check the response of the adapter shell.

2. Static Adapter No. 2 - In October 1962 construction on static adapter No. 2 was stopped due to budget considerations. With the initiation of the Popgun program (see 4, below), however, adapter No. 2 was reinstated to replace static adapter No. 4 in the structural test program.

After its fabrication, the retrograde rocket portion of the adapter was attached to static article No. 4 for dynamic response testing. The equipment portion of the adapter was later added and further dynamic testing was accomplished. At the conclusion of these tests, the adapter was returned to manufacturing for modification. Joined to re-entry test unit No. 3, the adapter was used for static tests of the "cold" launch condition.

3. Static Adapter No. 3 - This unit was used to structurally qualify the equipment mounts, the retro-rocket support structure, the blast shield access door, and the adapter itself in the "hot" launch condition.

4. Static Adapter No. 4 - After undergoing one pyrotechnic separation test at station Z 13, the adapter was assigned to Popgun testing. This test, which consisted of pyrotechnic separation at station Z 69 and the firing of the retro-rockets, had inconclusive results since considerable damage was

sustained in the retrograde section due to rocket assembly failure. The undamaged 52-33002-3 ring from the retrograde section was removed and utilized in the construction of a boilerplate adapter for further Popgun testing. This testing showed no Popgun effect. The equipment section of the static No. 4 adapter was used in another pyrotechnic separation test at Z 13.

D. Miscellaneous Test Vehicles:

1. Thermal Qualification Test Vehicle - This vehicle was a complete production spacecraft utilizing the No. 3A re-entry module (one of the 13 production units) and a test adapter. All systems and subsystems were flight worthy, qualified production items except for certain easily replaceable pieces of equipment such as the heat shield and the ejection seats. With NASA approval, nonflight articles were substituted for the latter.

Spacecraft No. 3A was delivered to the McDonnell laboratory on 15 October 1964. The thermal qualification test program ran until February 1965.

The qualification testing comprised mission simulation runs during which all systems were operated to their duty cycles. However, safety requirements for the vacuum chamber dictated the avoidance of hypergolics and cryogenic hydrogen; therefore inert fuels and bridgewire-type pyrotechnics were employed during these tests. In addition to these orbit simulation tests, special vibration and spacecraft system test (SST) tests were performed on Spacecraft 3A.

2. Electronics Systems Test Unit (ESTU) - The ESTU was a simplified re-entry module mock-up with provisions for mounting all electronic components in their flight locations. Prototype and early production units were installed and interconnected to simulate the spacecraft wiring conditions. Components, subsystems, and systems were at first operated component-by-component and then system-by-system to provide an initial evaluation of each componet when integrated with other units.

A configuration representative of that used in Spacecraft No. 2 was mounted and thoroughly investigated. The ESTU was also used to examine early problems and evaluate the corrective action. The ESTU was first put into operation on 19 November 1962. (Ref Development Program, page 76.)

3. Compatibility Test Unit (CTU) - The CTU was a spacecraft mock-up employing standard spacecraft wire bundles and having a structure very similar to the flight article. Prototype spacecraft systems were installed, creating a test vehicle with operational systems representative of Spacecraft 1, 2, 3, and 3A. The objectives of the test program involving the CTU were:

a. Provide compatibility tests (including radio noise) of the spacecraft systems to assure interference-free combined operation.

b. Establish spacecraft and ground support equipment (GSE) compatibility.

c. Furnish SST procedure evaluation and personnel training prior to production spacecraft tests.

d. Provide a test bed to evaluate spacecraft configuration changes and investigate problem areas.

The compatibility test unit was delivered from manufacturing in February 1963. The initial CTU tests were performed in the SST area utilizing SST personnel, procedures and test equipment. (Ref Development Program, page 76.)

4. Specimen Hatch - This test unit comprised a production hatch sill, side panels, hatches and latch mechanism mounted on a boilerplate box assembly. Latch rigging, functional, and leakage tests were performed on the unit, as well as static structural tests of the aft hoist loop fitting. Hatch testing was accomplished per TR 052-045.02. To perform the structural tests, proper structural representation required the inclusion of a portion of the large pressure bulkhead. This was subsequently installed. The unit was delivered from manufacturing in July 1963.

5. R & R and RCS Pyro Test Unit - This unit was composed of a full production R & R/RCS section equipped for the paraglider configuration. It was designed for pyrotechnic demonstration of the following operations:

a. Drogue mortar
b. Nose fairing separation
c. MDF ring separation at station Z 191.97
d. Nose landing gear deployment
e. Emergency docking release deployment, and
f. Docking bar assembly deployment.

The pyro test unit was originally scheduled for delivery from manufacturing in mid-December 1963, but the decision to abandon the paraglider concept involved major modifications to the test unit, resulting in a delivery date in the first quarter of 1964. Tests involving the landing gear (d, above) were deleted.

6. R & R and RCS Structural Test Unit - This unit, originally designed to contain a paraglider type R & R section and a parachute RCS, was used to structurally qualify the radar support structure, the RCS parachute support structure, and the nose fairing and support structure. Further testing involved qualifying the drogue parachute structure under re-entry temperatures, and the performance of a pyrotechnic separation of the MDF ring at Z 191.97. At the conclusion of these tests, the unit was installed on static article No. 3 (Ref B. Static Re-entry Vehicles, page 19) for parachute deployment tests at high temperatures.

Structural Testing.

A. Rendezvous and Recovery (R & R) Section Tests - The drogue chute mortar support structure was tested to determine its static strength, axial spring rate, and strain gauge calibration. The test article consisted of a tandem drogue rendezvous and recovery section attached to a rigid support at station Z 191.97. A uniformly distributed axial load was applied to the test article through the base of one of the mortar assemblies. When failure did not occur after the applied load had substantially exceeded the design ultimate load, testing was considered successfully completed.

B. Combined R & R/RCS Tests - The re-entry heating test of R & R - RCS structure with chute pull off loads was conducted to demonstrate the structural integrity of the R & R and of the attachment joint for the RCS section at station Z 191.97.

The structural integrity of the drogue parachute support structure during simulated re-entry heating and drogue parachute deployment loads also was demonstrated. The desired maximum temperature of 1600°F was achieved on the beryllium shingles at a heating rate of 6°F per sec. A limit load of 3550 lb and an ultimate load of 4850 lb were applied to the RX drogue cable at a loading rate of 3210 and 1735 lb per sec, respectively. Testing at sea level atmospheric conditions instead of in a near-vacuum caused fires which resulted in local structural damage such as broken thruster nozzles, dislodged shingle retainers and a bolt failure. However, test results indicated that the R & R attachment joint at Z 191.97, and the drogue parachute support structure were structurally adequate to withstand the re-entry temperatures and loads simulating deployment of the single drogue parachute.

C. Re-entry Module Tests

1. Structural Demonstration of the Re-entry Module for Parachute Deployment Loads - Two load conditions, representing different parachute pull-off angles, were tested consecutively. For each condition, the RCS section and a small portion of the adjoining conical section were heated by a quartz infrared lampbank prior to load application. Loading was initiated when a temperature of 260°F was recorded on the web of a stringer located on BY at Z 181.5. The structure sustained design ultimate load for both test conditions without failure. Several local fires during the heat test were attributed to laboratory atmospheric conditions.

2. Heating Test of Gemini Re-entry Module for Critical Re-entry Temperatures - Two test conditions were conducted to demonstrate the structural integrity of the re-entry module for the critical re-entry temperatures. The two conditions consisted of heating (1) the upper centerline in the vicinity of the hatches, and (2) the lower centerline in the vicinity of the ECS, equipment and landing gear doors. Temperatures and structural deflections at several locations were recorded. Test results indicated that the conical section was adequate structurally for the re-entry heating conditions tested.

3. Spacecraft Structural Evaluation for Re-entry Loads - This test subjected a spacecraft re-entry module to loads simulating critical re-entry conditions. Simulated aerodynamic pressure loads were applied to the heat shield in 5% increments to 136% design limit load. Loads were reacted on the aft section of the re-entry module. No damage was sustained during the test.

4. Static Test of Ejection Seat and Back-up Structure - This test determined the adequacy of the ejection seat support structure for the following conditions:

a. Condition IX $\beta = 0^{\circ}$, $\alpha = -15^{\circ}$ Ejection
b. Condition XI $\beta = 15^{\circ}$, $\alpha = 0^{\circ}$ Ejection
c. Condition V_b Landing.

The specimen sustained ultimate load (135% design limit load) for conditions IX and XI without failure.

For condition V_b, a torque box fitting failed at approximately 90% design limit load. The fitting was redesigned. Since condition V_b was a paraglider requirement, no further testing was required.

5. Static Test for Re-entry Module Pressurization - The requirements of this test were as follows:

a. To determine if the conical section of the spacecraft was structurally adequate for 200 cycles of internal burst pressurization from zero to six psig.

b. To determine what caused the left hatch on the spacecraft to unlatch and open at approximately 5.5 psig during cycle 4 of the test described in a above, and to repeat the malfunction of the left hatch opening under pressure with no changes to hatch rigging.

c. To determine if the conical section of the spacecraft was structurally adequate for ultimate internal burst pressurization (12 psig).

d. To determine if the conical section of the spacecraft was structurally adequate for ultimate external collapsing pressure (3 psig).

The specimen sustained 200 cycles of internal burst pressurization, ultimate internal burst pressure, and ultimate external collapsing pressure without failure or significant change in the leakage rate.

During the fourth cycle of the 200-cycle pressurization testing, the hatch mechanism rotated to the unlatched position, and the hatch opened from internal pressure. Examination of the hatch mechanism showed that a bolt in the hatch torque box cover was interfering with the hatch mechanism bellcrank assembly. The malfunction, which prevented the hatch mechanism from rotating full over center, was duplicated. The condition was eliminated by design change and the 200-cycle test was resumed without further malfunction. The specimen was determined to be structurally adequate for all conditions tested.

6. Spacecraft Water Impact Drop Tests - Simulated parachute landings on water were conducted to demonstrate the ability of the spacecraft structure to withstand impact loadings and to maintain a watertight crew compartment. The specimen was catapulted from a track into a pond to simulate worst conditions involving local water surface angle due to wave action and impact velocity resulting from wind and descent speed. Two impact attitudes were tested; RCS section forward, then heat shield forward. The weight and balance

of the unballasted test vehicle were determined experimentally. The mass moments of inertia of the basic vehicle were determined analytically. Impact accelerations along the Z (longitudinal) and Y (vertical) axes were measured. The cabin was pressurized to 5.0 psig before and after each drop test, and the pressure decay during a 30 min period was recorded in each case. The cabin was not pressurized when dropped.

On the heat shield forward drop, no structural damage occurred. On the test with the RCS section forward, shingles were deformed on the conical section adjacent to the main landing gear doors. A subsequent pressurization check detected a small leak at the forward edge of the ECS door. However, during a 39-hr flotation period following the drop test, only 20 oz of water was taken aboard. The spacecraft was considered structurally satisfactory to sustain water impact.

7. Static Test of Crew Hatch - These tests helped to evaluate the structural integrity of the crew hatch and hatch support for the following conditions:

a. Condition I - Ultimate external airload against the flipper door for astronaut egress during an abort.
b. Condition II - Ultimate air and inertial loads tending to rotate the hatch past 88 degrees full open, restrained by an extended simulated hatch actuator. This condition was critical for the hatch and hatch actuator support.
c. Condition III - Ultimate air, inertial and ejection seat catapult loads applied to hatches and support in the following sequence:

- Airloads were applied to both hatches when full open 88 degrees and supported by simulated hatch actuators.
- With ultimate airload applied to hatches, the ultimate ejection seat catapult load was applied to the right ejection seat catapult fitting.
- With ultimate load applied to hatches and right ejection seat catapult fitting, ultimate ejection seat catapult load was applied to the left ejection seat catapult fitting.
- With ultimate load applied to the hatches and both ejection seat catapult fittings, the applied catapult load on the right catapult fitting was reduced to zero.

Results of the crew hatch test were:

a. Condition I - The flipper door and adjacent support sustained 136% design limit load (6.2 psi) without failure.
b. Condition II - The crew hatch and support structure reached 120% design limit load before the hatch torque box skin failed. The crew hatch was strengthened by incorporating a machined stiffener and a doubler on the hatch torque box skin. When the condition was retested, the redesigned crew hatch failed at 155% design limit load.
c. Condition III - The hatches and support withstood the ultimate test load (136% design limit load) for all four phases of the test. The

flipper doors, strengthened crew hatches, and the hatch support structure were adequate for all conditions tested.

D. Adapter Section Tests

1. Static Test of the Retrograde Rocket Support Structure - This test helped to determine the adequacy of the retrograde rocket support structure for the critical abort condition. The structure sustained design ultimate load without failure. Load was applied simultaneously along the thrust axis of all four rockets, and no permanent deformation was observed.

2. Static Test of Gemini Equipment Modules and Module Support Structure - This test evaluated the adequacy of the following adapter equipment modules and their support structure for critical launch and abort conditions:

a. Orbital attitude maneuvering system module.
b. Fuel cell module, long mission.
c. Environmental control system oxygen module, long mission.
d. Environmental control system coolant module.

All four modules and their support structures sustained ultimate load (136% design limit load) for the critical launch condition without primary structural failure.

When the simulated hydrogen bottle was removed from the fuel cell module after the test, the press-fit sleeve and plug assembly, which restrains side motion of the bottle, had separated from the fuel cell blast-shield. The oxygen bottle sleeve assembly also was loosened. The press-fit sleeve and plug assemblies were replaced by threaded units prior to the critical abort tests.

The fuel cell module and orbital attitude maneuvering system module sustained design ultimate load for the critical abort test without failure. At 130% design limit load, the simulated pressurant bottle nearest LX deflected enough to contact the module support structure. To correct this the pressurant bottle support was stiffened. The adapter equipment modules and adjacent support structure were determined to be structurally adequate for the launch and abort conditions tested.

3. Static Test of the Spacecraft Adapter for Critical Launch Condition with Elevated Temperature - This test helped determine the structural adequacy of the adapter for launch condition 2h (Ref: structural design loads, McDonnell Report 9030), launch trajectory 333, which has a critical combination of load and elevated temperature.

The test adapter was mounted horizontally on the forward oxidizer skirt of a Titan launch vehicle so that the stringer No. 40 of the equipment section received maximum compression. The specimen was loaded axially (aft) in the Z direction. Shear was introduced perpendicularly to the Z axis and vertically downward. Adapter radiator leakage was checked, and gaps between the oxidizer skirt and the adapter were measured before and during the test. The temperature cycle was run under three conditions of load: limit (100% DLL), ultimate (136% DLL) and stretch (156% DLL).

The test structure satisfactorily sustained the stretch load at 156% DLL. The adapter radiator did not leak, and gap deflections between the oxidizer skirt and adapter were negligible.

The adapter was considered structurally capable of sustaining the stretch loads.

E. Complete Spacecraft Test - The static test of Gemini Spacecraft for a critical abort condition verified the structural and functional integrity of the spacecraft for the 5.5 degree angle of attack. Two tests were conducted. During test condition A, the specimen was subjected to both axial compression and bending moment about the Y-Y axis with the right (RX) side of the spacecraft in compression. The specimen was loaded in increments to 100% design limit load and the hatches were opened by hatch actuators. Inspection revealed no damage.

For test condition B, the specimen was subjected to bending moment about the X-X axis with the bottom (BY) of the spacecraft in compression, and a torque applied through the open hatches, as well as an axial load and bending moment about the Y-Y axis. The specimen, instrumented to record strains, was loaded in increments to 136% DLL without deleterious effect.

The spacecraft was considered functionally and structurally adequate for the abort conditions tested.

F. Gemini-Agena Target Docking Adapter (TDA) Test - The ultimate strength static test qualified the critical structure of the Gemini and Agena target docking adapter for orbital maneuvering ultimate loads. The test specimen consisted of the following:

1. A static target docking adapter.
2. A rendezvous and recovery module.
3. A static re-entry control system section.
4. An Agena forward auxiliary rack.

Two sets of loading conditions were tested. One consisted of bending with axial compression which simulated maneuvering in orbit with full thrust; the other consisted of bending without axial compression which simulated maneuvering in orbit with zero thrust.

Three tests were conducted with TDA andthe R & R latched together and rigidized to their maneuvering configuration. Loading conditions were as follows:

1. Bending With Axial Compression - The bending was applied about the X 0.00 axis placing the LX latch in tension.

2. Bending Without Axial Compression - The bending was applied about Y 0.00 axis placing the BY latch in tension.

3. Bending With Axial Compression - The bending was applied about the Y 0.00 axis placing the BY latch in compression. Bending was increased to failure.

The Gemini and Agena TDA withstood ultimate loads for orbital maneuvering imposed by the three tests.

A primary and secondary failure of the R & R occurred during test 3 at 147% axial design limit loads and 227% bending design limit load. The primary failure consisted of buckling the two stringers adjacent to the BY docking fitting. Shingle retainer failure in the same area constituted the secondary failure.

It was concluded that the Gemini and Agena TDA were structurally capable of withstanding ultimate loads for orbital maneuvering, and that the TDA was structurally stronger than the R & R when subjected to the loading conditions defined by this test request.

Structural Dynamics Tests

Dynamic Response Test of Spacecraft and Equipment Under a Vibration Environment. - This test helped to determine the following:

A. Spacecraft beam frequency response, beam resonant mode, and resonant mode damping decay characteristics.

B. Equipment frequency response and equipment resonant acceleration response mode characteristics.

C. Adapter shell resonant response mode characteristics.

The specimen was tested in the following configurations:

A. Abort Configuration - The abort module was suspended free-free at stations Z 109 and Z 192.

B. Re-entry Configuration - The re-entry module was suspended free-free at stations Z 109 and Z 192.

C. Landing Configuration - The landing module was suspended free-free at stations Z 104 and Z 192, with the landing gear under test in the extended position. The nose landing gear was tested in both the compressed and extended positions.

Dynamic Response of the Spacecraft in the Moored Configuration. - This test was conducted in the following three phases:

A. Phase 1: Dynamic Response of the Spacecraft in the Moored Configuration - The spacecraft was subjected to a sinusoidal vibration environment in its X, Y and Z axes. The spacecraft was moored to a target docking adapter (TDA) which was bolted to an Agena forward auxiliary rack cantilevered from the laboratory floor. The first three elastic body modes of the system for various static load conditions were determined for all axes of excitation. These static load conditions simulated the compression and moment loads generated by the thrust of the Agena engine. Phase 1 data include frequency response plots, modal response data, and damping data.

B. Phase 2: Determination of Moment of Inertia - The inertia of the spacecraft was determined about its X and Y axes. The spacecraft was supported on knife edges at Z station 192 and was suspended with soft springs of a known spring constant at Z station 13. The inertia was calculated by knowing the weight, center of gravity, spring location, knife edge location, and frequency of oscillation of the system.

C. Phase 3: Dynamic Response of TDA Equipment - The TDA was subjected to a sinusoidal vibration environment in its X, Y and Z axes. The TDA was attached to a fixture which was attached to an electromagnetic vibration exciter for excitation in each of three mutually perpendicular axes. Data presented for phase 3 included transmissibility plots and modal response data.

Full-Scale Structural and Functional Tests of an Agena TDA and a Gemini R & R Section Subjected to Mooring Shock. - Tests simulating orbital moorings between full-scale Gemini and Agena vehicles were conducted to demonstrate the structural and functional capabilities of a production Gemini R & R section and an Agena TDA when subjected to the mooring shock environment. Both the R & R section and TDA assembly were mounted on fabricated structural steel vehicles. Composite vehicle assemblies simulated the mass, cg location, and mass moments of inertia of their respective production vehicles for an orbital configuration.

Test vehicles were suspended as simple pendulums, 56.57 ft in length with a gimbal system at each cg, giving the vehicles freedom in pitch, yaw, roll and translation. The Gemini test vehicle was pulled back and then allowed to swing forward through a predetermined distance to attain various vehicle limit and ultimate closing velocities. Vehicle attitudes and locations of impacts also were controlled. TDA damper loads and strokes and the R & R section indexing bar bending moments were recorded.

The Agena TDA and the Gemini R & R section sustained all shock loads imposed with no structural or functional failures. All mooring systems that were operable during the test performed satisfactorily.

RELIABILITY AND QUALITY ASSURANCE PROGRAM

The objective of the Gemini reliability and quality assurance program was to attain the level of reliability required for all aspects of manned orbital flight. The emphasis was placed on achieving high mission dependability with maximum crew safety. A mission reliability goal of .95 and a crew safety goal of .995 were therefore specified, in which mission reliability is defined as the probability of accomplishing the objectives of the mission, and crew safety is defined as the probability of the crew surviving the mission. The methodology for accomplishing these two goals was:

A. Review systems' design to insure that the designs were inherently satisfactory.
B. Suggest changes to design engineering to maximize system reliability.
C. Conduct tests throughout the Gemini program to demonstrate systems' reliability and to verify satisfactory operating characteristics.
D. Provide reliability estimates for various missions to quantitatively express spacecraft and mission reliability and to identify equipment or systems which required reliability improvement.
E. Establish a control system which required the reporting, investigation, and correction of equipment malfunctions.
F. Develop a rigid quality control program to maintain the reliability inherent in the spacecraft design.

Evaluation Of Spacecraft Design

The design of the spacecraft was evaluated to insure that the reliability goals could be met. Wherever a single failure could be catastrophic to the crew or to the spacecraft or could jeopardize mission success, redundant systems or back-up procedures were deemed necessary and were instituted. Some examples of these redundant features are:

A. Every pyrotechnic function was supplied with redundant initiation circuitry and redundant cartridges.
B. Two independent re-entry control propulsion systems were supplied for re-entry safety.
C. Redundant horizon sensors and rate gyros were provided for the guidance and control system.
D. Six fuel cell stacks were installed to supply electrical power; only three to five stacks were required for nominal mission performance.

Reliability Reviews. - Reliability engineers reviewed designs independently of the initiators in order to insure an objective evaluation for reliability and crew safety. These reviews were begun as soon as the initial design was established and were continued throughout the design phase. Surveillance of the prime contractor's suppliers was also maintained in design, parts selection and application, reliability activities, and in proposed design problem solutions. In addition, specification and procurement documents, schematic and installation drawings, stress analyses, and design data sheets were evaluated to determine their effects on reliability. Detailed studies were performed to evaluate system configurations for possible design improvement. Some examples of these studies and their results are:

A. Reliability analysis of the environmental control system resulted in the installation of a redundant heater in the primary oxygen tank. (On Spacecraft 10, 11, and 12 these redundant heaters allowed both the ECS and the RSS to share a single, enlarged oxygen tank.)

B. A study was performed to determine the effect of the instrumentation signal conditioners upon maneuvering and attitude control reliability. The study indicated that spacecraft control would not be significantly degraded by the failure of the instrumentation equipment, since electrical isolation between the systems was part of the design concept.

C. Analysis of the flight director controller, located on the spacecraft right-hand instrument panel, resulted in including ON-OFF switch for the thrusters and related units. This conserved and improved reliability by not requiring the units to be continuously powered.

Failure Mode and Failure Effect Analysis. - A detailed failure mode and failure effect analysis was conducted on all functional spacecraft equipment. The analysis included the following steps:

A. A description of the failure mode.
B. The identification of the mission phases in which the failure was likely to occur.
C. An estimate of how the failure would affect the system and the mission.
D. A review of the failure indications available to the crew and to the ground.
E. An estimate of the maximum time the mission could be continued after a failure. This estimate measured the seriousness of the failure.

The analysis provided a means to estimate the impact of a failure upon mission success and crew safety. In addition, it helped determine the need for design changes to eliminate those failures which significantly jeopardized mission success or crew safety. Finally, the analysis led to a reexamination of the adequacy of failure indications. Single point failure modes and failure effects were analyzed for every manned mission. Action was taken (1) to eliminate the failure mode(s), or (2) to justify the design and recommend precautionary procedures or crew actions to minimize the likelihood of failure.

An abort time study was made to help determine whether recovery forces were adequately distributed. The probability of an abort, and the probability of permitting the flight to continue for a specified time after an abort-causing failure were estimated. If a failure occurred which required an abort, the study indicated that there was a 90% probability that at least 1.5 orbits could be completed prior to spacecraft re-entry.

Test Program For Reliability And Quality Assurance

The Gemini test program consisted of development, qualification, reliability, and equipment flight simulation tests (the last utilizing both estimated and real mission environmental data). Development tests used engineering models to establish the feasibility of the design concepts. Prior to qualification testing, the functional performance and structural integrity of production hardware were demonstrated. Qualification requirements were established for all spacecraft equipment, and sufficient testing was then performed to prove that a production unit met the design requirements. Qualification tests included simulated mission conditions, during which the equipment was operated to the duty cycles and in the modes expected during flight.

The environmental levels to which the equipment was subjected were based on the anticipated preflight, flight, and post-landing conditions. However, these environmental levels were revised whenever flight experience or other data revealed that the original environmental requirements were too stringent. Production equipment was also subjected to overstress testing, in which the equipment was operated under conditions more severe than the design requirements.

Tests were conducted on all flight articles to assure that the design reliability had not been degraded in the fabrication, handling and installation of the hardware. These tests comprised:

A. Receiving Tests - Parts were inspected by x-ray, spectrograph, and other techniques where appropriate, and functional tests were performed.

B. Production Tests - Inspection and tests were performed at various stages of equipment assembly to detect deficiencies early in the manufacturing process.

C. Predelivery Acceptance Tests (PDA) - Equipment performance was verified at the vendor's plant prior to delivery to the prime contractor or to the customer.

D. Preinstallation Acceptance Tests (PIA) - Equipment performance was reverified prior to installing the equipment in the spacecraft.

E. Spacecraft Systems Tests (SST) - Individual and integrated systems tests, simulated flight tests, and altitude chamber tests were performed after systems' installation in the spacecraft.

F. Spacecraft/Launch Vehicle Joint System Tests - Systems tests were performed at the launch site prior to mating the spacecraft to the launch vehicle. After spacecraft mate, integrated systems tests, simulated flights, and abort mode tests were conducted, utilizing the Mission Control Center, the Manned Space Flight Network, and the flight crew.

G. Countdown Tests - A final series of functional tests was performed on systems to verify their flight readiness.

Limited design assurance tests (DAT) and reliability assurance tests (RAT) were conducted; however, the quality and conclusiveness of the overall test program were sufficient to demonstrate equipment performance and flight worthiness.

Estimates Of Reliability Requirements

The reliability requirements were established with regard for each system's mission function and in keeping with the overall spacecraft reliability goal of .95. These requirements were consistent with the relative reliabilities of similar Project Mercury equipment. Each system's inherent reliability was then estimated, based primarily on component failure rate data. These estimates were applied to reliability models which called for a two-day rendezvous and a two-week orbital flight; the results indicated that the design would meet the reliability goals for mission success and crew safety.

As part of the continuing evaluation, a reliability estimate of mission success and crew safety was made prior to the flight of each manned mission. These estimates were based on the latest test and flight data. The close agreement of these estimates with the actual mission performance gave additional assurances that succeeding missions would be accomplished successfully and safely.

Monitoring And Analysis Of Equipment Malfunctions

A "closed loop" program of failure reporting, failure analysis, and subsequent corrective action was established to identify all equipment malfunctions and anomalies and to insure that corrective measures were taken. All malfunctions were recorded, from the first testing of engineering models up through the operation of all flight items. A material review board determined the disposition of the equipment for failure analysis, corrective action, and subsequent use.

The failures were analyzed at the vendor's plant, the McDonnell plant, or at Kennedy Space Center, depending upon the nature of the failure and the availability of facilities. Complex equipment was returned to the vendor when the analysis required special engineering and technical skills or special test equipment.

Failure analysis laboratories were established at the Kennedy Space Center and at the McDonnell plant to provide rapid detailed analysis of the cause of the failure. Laboratories were equipped with precision measuring devices, environmental chambers, and sensitive detectors. By the program's end, the McDonnell laboratory had performed 882 of these analyses.

All recommended corrective actions were evaluated and those deemed appropriate were implemented. The required corrective actions ranged from changes in design and manufacturing to revised quality control techniques and testing criteria. A "current status" summary of all trouble reports was maintained, in which each anomaly or failure was described, and the status of the analysis and corrective action was indicated. This list was continually reviewed by the customer and the contractor to insure acceptable and timely corrective measures; action was accelerated in those areas in which failure would have a significant effect on pending flights.

Development Of Quality Control

A rigid quality control system was developed to maintain the reliability inherent in the spacecraft design. This system required that flight equipment be produced, handled, and installed in a manner calculated to maintain a qualified configuration. A configuration control program was established which required that all changes to the spacecraft be documented, approved, implemented, and verified by quality control. This method permitted rapid changes accompanied by rigorous inspection. No change was made to the total configuration until it had been submitted to a management change board which evaluated the impact of the change upon the program cost and the schedule.

Each flight article was identified by part number, and all assemblies and components (such as pressure regulators and electronic boxes) were serialized, recorded, and accountable. Certain materials, such as pyrotechnic explosives, were kept track of by lot to permit their identification if it was determined that the materials used were substandard.

All manufacturing and inspection personnel who required a specified skill level were especially trained and the quality of their work was periodically examined by means of proficiency tests. Parts and fabricated assemblies were inspected to maintain the spacecraft quality. All discrepancies found by this inspection and testing were recorded and corrected, despite their apparent insignificance.

All equipment installation and removal required inspection approval prior to changing the system configuration. Formal acceptance reviews were conducted by the customer and the contractor at critical stages of the spacecraft assembly and testing to isolate deficiencies that might have reduced spacecraft performance.

SPACECRAFT FLIGHT PERFORMANCE

Six major objectives were defined for the Gemini manned spacecraft program. Stated simply, these objectives were:

A. Expose two astronauts and their life support systems to long-duration missions in preparation for future earth orbit and lunar flights.

B. Develop and exercise precision re-entry, landing, and recovery of manned spacecraft.

C. Rendezvous and dock with a second orbiting vehicle and then perform combined maneuvering.

D. Undertake extravehicular activity to evaluate man's ability to perform tasks in a weightless environment.

E. Utilize the Gemini Spacecraft as an experimental test platform for scientific investigations.

F. Provide a continuation of manned spaceflight operation at minimum cost with major milestones to be complete as soon as practical.

The following synopses present the flight performance of each spacecraft and describe how the program achieved all of the major objectives, except land landing. All Gemini Spacecraft were launched from Complex 19, Cape Kennedy, Florida. The Gemini launch vehicle was a modified Titan II ICBM, "man-rated" for Gemini usage.

Gemini I Mission

The first Gemini mission was an unmanned orbital flight, launched successfully from the Cape on 8 April 1964. The objectives for this first mission were:

Gemini I - Primary Objectives.

A. To demonstrate the Gemini launch vehicle performance and to flight qualify the vehicle subsystems for future Gemini missions.

B. To determine the exit heating conditions on the spacecraft and launch vehicle.

C. To determine the structural integrity and compatibility of the combined spacecraft and launch vehicle through orbital insertion.

D. To demonstrate the ability of the Gemini launch vehicle and ground guidance systems to achieve the required orbital insertion conditions.

E. To monitor the switchover circuits on the Gemini launch vehicle and to evaluate their sufficiency for mission requirements.

F. To demonstrate the switchover function if anomalies occur within the primary autopilot or hydraulic systems that would require the use of the secondary autopilot or hydraulic systems.

G. To demonstrate the malfunction detection system.

H. To verify the structural integrity of the Gemini Spacecraft.

Gemini I - Secondary Objectives.

A. To evaluate the operational procedures used in establishing the Gemini launch vehicle trajectory and cutoff conditions.

B. To demonstrate the performance of the launch and tracking networks.

C. To verify orbital insertion conditions by tracking the C-band transponder system in the spacecraft.

D. To provide training for the flight dynamics, guidance switchover, and malfunction detection systems flight controllers.

E. To demonstrate the operational capability of the prelaunch and launch facilities.

The first production Gemini Spacecraft was utilized for this flight but it did not carry complete Gemini flight systems because the mission was primarily a test of structural integrity. Spacecraft 1 was launched at 11:00 a.m., EST. The mission was declared successfully concluded four hours and 50 min after lift-off. Tracking, however, was continued by the Goddard Space Flight Center until the spacecraft re-entered on the 64th orbital pass over the southern Atlantic Ocean.

Two subsystems employed on the Gemini I mission were the C-band radar transponder and the three telemetry transmitters. The transponder aided in accurate ground tracking of the spacecraft during the mission; the transmitters provided data on spacecraft heating, structural loading, vibration, sound pressure levels, and the temperatures and pressures encountered during the launch phase.

Miscellaneous equipment installed on the spacecraft consisted of a 24-volt DC silver-zinc battery, a cabin pressure relief valve, a prelaunch cooling system, and three spacecraft-to-ground umbilicals. Dummy equipment, having a mass and moment of inertia equal to the missing flight systems, was installed in the spacecraft.

The spacecraft/launch vehicle second stage combination was inserted into an orbit having a perigee of 86.6 nautical miles and an apogee of 173 nautical miles. These figures were within the design tolerance; the perigee was actually only 0.4 nautical miles short of the desired altitude. A 20 fps overspeed condition at orbital insertion produced an increase of 11 nautical miles in the apogee.

Although the trajectory was designed for an orbital lifetime of several days, the Gemini I mission was considered complete after three orbital passes over Cape Kennedy. All primary and secondary mission objectives were accomplished.

Gemini II Mission

The second Gemini mission was an unmanned suborbital flight launched at 9:04 a.m., EST, on 19 January 1965. The spacecraft was recovered by the primary recovery ship, the aircraft carrier Lake Champlain, at 10:52 a.m., EST. Splashdown was within three miles of the target. The major objectives of the Gemini II mission were:

A. To demonstrate the basic structural integrity of the unit throughout the flight environment.

B. To verify the adequacy of the re-entry heat protection under the most severe conditions.

C. To achieve satisfactory performance of vital flight control systems, life support systems, retrograde rocket system, recovery and landing systems, and other systems critical to flight safety and mission success.

D. Verify systems checkout and launch procedures.

E. Evaluate back-up guidance steering signals throughout launch.

Spacecraft 2 contained production units of all equipment used on the later manned missions with the exception of the rendezvous radar, the drogue parachute system, and the auxiliary tape memory. A fuel cell was also installed but the hope for engineering evaluation of flight performance was not accomplished because a prelaunch facility malfunction made timely activation impossible. An automatic sequencing device was installed in the spacecraft to control the operation and the sequencing of the Gemini systems throughout the flight. The major spacecraft functions performed during this mission were spacecraft/launch vehicle separation, controlled 180 degree turnaround, adapter equipment section jettison, retrofire, retrograde section jettison, controlled zero lift re-entry (10 degrees roll rate for 150 sec) and parachute landing. The spacecraft was recovered 1848 nautical miles down range from the launch site.

All major mission objectives were accomplished except for the performance of tests on the fuel cell. Due to the unpredictability experienced with fuel cell performance up to the time of Spacecraft 2 flight the decision was made to use the fuel cell as a separately loaded power source for engineering evaluation purposes rather than a source of spacecraft power on the Gemini II flight.

Two minor anomalies occurred on the mission. During the re-entry mode, temperatures near the adapter interconnect fairing on the cabin section were higher than the planned values. This resulted in damage to two of the Rene 41 shingles, one washer, and a circumferential strap. No damage was experienced by the underlying insulation, however. The corrective action taken for Spacecraft 3 and up was to increase the thickness of the shingles in this area, employ smaller diameter washers as fasteners, and thicken the circumferential strap. Also, temperature reduction was accomplished during the re-entry mode by reducing the design angle of attack which resulted in the center of gravity being off-set from e = .196 in. to e = 1.35 in. on Spacecraft 3, plus similar reductions on subsequent spacecraft, with an accompanying loss of footprint capability. No further difficulty was experienced in this area.

An indicated error in the inertial measuring unit (IMU) accelerometer output (inertial guidance system) of approximately 66.5 fps in the X axis was received at secondary engine cutoff. After failure analysis, corrective action was taken to redesign the rate network to accept saturated inputs. This change was made in the X loop only for the Gemini III configuration; however, on Gemini IV and all subsequent flights all accelerometer loops incorporated the redesign.

Gemini III Mission

The third Gemini flight and the program's first manned mission was launched at 9:24 a.m., EST, on 23 March 1965. The planned three-orbit mission had the following principal objectives:

A. Demonstrate manned orbital flight in the Gemini Spacecraft and further qualify both spacecraft and launch vehicle systems for future long-duration missions.

B. Evaluate the Gemini design and its effects upon crew performance.

C. Exhibit and evaluate operation of the worldwide tracking network with the spacecraft.

D. Demonstrate precise orbital maneuvering using the orbit attitude and maneuver system (OAMS).

E. Verify OAMS capability to perform retro back-up.

F. Evaluate the performance of major spacecraft systems.

G. Demonstrate the ability to control the re-entry flight path and to arrive at a predetermined landing point.

H. Verify systems checkout, prelaunch, and launch procedures for a manned spacecraft.

I. Recover the spacecraft and appraise the recovery system.

The flight crew successfully completed the mission, during which they employed several thruster firings to alter the spacecraft orbit and to perform small out-of-plane maneuvers. Both the OAMS and the spacecraft guidance and control system performed satisfactorily during the flight, with no significant anomalies.

The only primary objective unachieved during the Gemini III mission was a landing close to the recovery force. The actual landing point was about 58 nautical miles short of the planned retrieval point. A study indicated that the angle of attack had been slightly lower than predicted; however, the main reason for the short trajectory appeared to be a considerably lower lift coefficient and a corresponding reduction in the touchdown footprint. The flight data appeared to indicate a difference between the actual and the wind-tunnel-derived aerodynamics of the re-entry configuration. The experience acquired from this mission and the Gemini II flight were correlated with wind tunnel data to arrive at a more accurate prediction of the lift-to-drag ratio and corresponding footprints for later flights.

The mission was successfully concluded with recovery of the spacecraft by the prime recovery ship, the aircraft carrier U.S.S. Intrepid, at 5:03 p.m., EST. Two of the principal benefits accruing from the Gemini III mission were the qualification it gave the worldwide tracking network and the experience it provided to operations personnel for longer missions.

Gemini IV Mission

The Gemini IV flight, scheduled for a four-day mission, was launched from Cape Kennedy at 10:16 a.m., EST, on 3 June 1965. The primary and secondary objectives of the mission were:

Primary Objectives.

A. Evaluate the effects of prolonged exposure to the space environment of the two-man flight crew in preparation for longer missions.

B. Demonstrate and evaluate the performance of the Gemini Spacecraft systems for four days in space.

C. Evaluate previously developed procedures for crew rest and work cycles, eating schedules, and real-time flight planning for long flights.

Secondary Objectives.

A. Demonstrate extravehicular activity in space and evaluate attitude and position control using the hand-held propulsion unit or the tether line.

B. Conduct station keeping and rendezvous maneuvers with the expended second stage of the Gemini launch vehicle.

C. Conduct further evaluation of spacecraft systems as outlined in the in-flight systems test objectives.

D. Demonstrate the capability of the spacecraft and flight crew to make significant in-plane and out-of-plane maneuvers.

E. Demonstrate orbital attitude and maneuver system (OAMS) capability to operate as a back-up for the retrograde rocket system.

F. Execute 13 in-flight experiments.

The flight demonstrated effectually the astronauts' ability to adjust perfectly to a weightless environment and to perform all mission tasks with efficiency; both astronauts were in excellent physical condition at the conclusion of the flight. Of 13 scheduled in-flight experiments, the Gemini crew successfully conducted 11.

All primary mission objectives were met; however, a problem was encountered in the 48th revolution wherein the computer would not sequence off when the computer on/off switch was operated. Because of this condition a computer controlled (closed loop) re-entry could not be attempted and a zero lift re-entry was flown instead. Due to dispersions in retrofire attitude, retrofire time, and OAMS thrust, the landing point was 50 nautical miles short of the predicted retrieval point. Failure analysis revealed that the computer turnoff anomaly was intermittent and seven postulated failure mechanisms were identified. The exact cause of failure was never isolated however available evidence suggested that contamination of the on/off switch was the most probable cause. The corrective action was to install a manual shutdown sequence switch which would circumvent all the postulated failure modes should the problem reoccur.

An anomaly was also found to have occurred during re-entry when a brief period of excessive current drain was observed from the re-entry batteries. Failure analysis revealed that this anomaly was caused by electrical arcing between the computer +25 volt regulator transistor terminal and the computer case. To remedy this, an epoxy glass insulator strip was attached to the inside of the computer cover to provide an electrical and mechanical barrier between the transistors and computer cover on all subsequent flight units.

Late in the first revolution, the decision was made not to attempt the rendezvous with the Gemini launch vehicle second stage because the allocated propellant for the OAMS had been consumed during the station keeping exercise with the second stage.

Two thruster anomalies were experienced on the Gemini IV mission; however, neither was considered detrimental to mission success or to crew safety. No. 5 thruster on the re-entry control system (RCS) B-ring failed to operate during re-entry. Post-flight inspection revealed a broken wire to an electrical connector between the attitude control electronics (ACE) package and the thruster solenoid valve. In addition, a loss of thrust was experienced early in the mission from one aft-firing OAMS thruster. The most probable cause of this apparent loss of thrust was that the crew had inadvertently thrown the No. 9 circuit breaker to OFF during the final 30 min of the first orbit. During the 61st orbit, when the No. 9 thruster was again required, its operation appeared satisfactory.

The flight crew reported difficulty in closing and latching the hatch after EVA. This anomaly, and the corrective action employed, are fully discussed in Failure Summary And Analysis, page 16 of this report.

The mission was successfully concluded on 7 June 1965. Recovery of the spacecraft was made by the prime recovery ship, the aircraft carrier U.S.S. Wasp, at 2:28 p.m., EST. With minor changes, the Gemini Spacecraft was considered flight-qualified for longer missions.

Gemini V Mission

The fifth Gemini mission, launched at 9:00 a.m., EST, 21 August 1965 was the first long-duration flight to use fuel cells as the principal source of spacecraft power. The primary mission objectives were:

A. To demonstrate an eight-day flight capability.
B. To evaluate the performance of the rendezvous guidance and navigation system in conjunction with the rendezvous evaluation pod (REP).
C. To determine the effects of prolonged weightlessness upon the flight crew, in preparation for even longer missions.

A secondary objective was to demonstrate a controlled re-entry to a predetermined landing point.

During the first two orbits, all spacecraft systems were checked, a nominal perigee adjust maneuver was conducted, and the rendezvous evaluation pod was ejected on schedule.

Two of the three principal mission objectives were achieved; however, it was not possible to perform the radar evaluation with the REP because of the necessity to power down the spacecraft early in the mission. Power conservation was deemed necessary because of a loss of pressure in the fuel cell oxygen supply tank. The cause of the pressure decrease was believed to be in the reactant supply system (RSS) oxygen tank heater wiring. Since the cryogenic tanks, heaters, and associated wiring could not be recovered for analysis, it was impossible to determine the actual failure mode. However, all available development and qualification test data show the cryogenic tanks and heaters to be rigidly constructed and of sound design. Nevertheless, corrective action was implemented for Spacecraft 7 and all other flights utilizing fuel cells. Redundant heater wiring was provided to the RSS hydrogen tank, and a crossfeed was installed between the RSS and environmental control system (ECS) oxygen tanks to enable oxygen to be supplied to the fuel cells or the ECS loops in the event of heater malfunction. (See RSS Flight Performance, page 63.) Despite the loss of oxygen pressure, the fuel cells received adequate operating pressure to supply necessary electrical power for the duration of the mission.

At the end of revolution 17 the spacecraft was powered up to a high load condition, and a successful rendezvous radar test was conducted by tracking a transponder on the ground at Cape Kennedy. Further radar tests were conducted throughout the mission to evaluate the rendezvous system in lieu of the REP exercises. A simulated Agena rendezvous was conducted on the third day which indicated that the spacecraft could have been placed within 0.3 nautical miles of an Agena Target Vehicle (ATV).

During the mission, five OAMS attitude thrusters functioned abnormally, providing low or zero thrust. Though the astronauts made two in-flight attempts to restore operation, they were not successful. A post-flight study indicated that the probable cause of the thruster failures was propellant freezing, due to the long periods of valve heater and line heater inaction. To preclude future thruster failures, redundant heaters and circuits were installed on the next long-duration configuration (Spacecraft 7). In addition, thermocouples were provided to monitor the temperatures of the thrust chambers to assure satisfactory heater operation.

Spacecraft systems functioned normally during the re-entry mode, but ground transmission of incorrect navigational coordinates resulted in a landing 89 nautical miles short of the planned retrieval point. The spacecraft was recovered on 29 August 1965 by the aircraft carrier U.S.S. Lake Champlain. The experiment program for the mission was very successful; 16 of the 17 planned experiments were conducted, and a high percentage of desired data was accumulated.

Gemini VI Mission

The flight of Gemini VI constituted the first rendezvous mission of the program. The mission's primary objective was to achieve an orbital rendezvous with Spacecraft 7, which became the target vehicle due to the Agena's failure to achieve orbit on 25 October 1965.

Spacecraft 6 was successfully launched at 8:37 a.m., EST, on 15 December 1965, 11 days after the launch of Spacecraft 7. A "closed loop" rendezvous was achieved approximately six hr after launch. Nine maneuvers were performed by Spacecraft 6 to effect rendezvous. Initial radar lock-on with Spacecraft 7 occurred at a range of 248 nautical miles, with continuous lock-on beginning at 235 nautical miles. After rendezvous, station keeping was performed for about 3-1/2 orbits, with the spacecraft as close as one ft apart. The command pilot of Spacecraft 6 performed an in-plane fly-around maneuver, maintaining a distance of 150 to 250 ft from Spacecraft 7. Separation maneuvers were performed and the visibility of Spacecraft 7 as a target vehicle was evaluated. The flight progressed normally and was ended by a nominal re-entry and landing on 16 December within seven nautical miles of the planned retrieval point. The recovery ship was the aircraft carrier, the U.S.S. Wasp. All primary mission objectives were accomplished.

Gemini VII Mission

The Gemini VII mission, a maximum duration flight, was launched at 2:30 p.m., EST, on 4 December 1965. The primary mission objectives were to demonstrate a manned orbital flight of 14 days, and to evaluate the effects of the prolonged mission upon the crew. Secondary objectives included a rendezvous with Spacecraft 6, station keeping with that spacecraft and with the second stage of the launch vehicle, and the carrying out of 20 in-flight experiments.

Additional equipment on this flight included an L-band transponder which was used with the Spacecraft 6 radar to provide target position data for rendezvous.

After insertion, the spacecraft performed station keeping with the launch vehicle, maintaining distances of between 60 and 150 ft for 15 min. A closer approach was not attempted because of the high tumbling rate of the launch vehicle. On the fifth day of the mission, the spacecraft was maneuvered into a favorable orbit for the rendezvous with Spacecraft 6. No further adjustments to this orbit were required.

An additional accomplishment of this flight was the crew's demonstrated ability to perform many of the mission requirements while not wearing their pressure suits. Comfort and mobility were greatly increased with the suits removed; no detriment to either crew health or mission success resulted. The ECS provided a nominal atmosphere throughout the flight.

Fuel cells provided the principal power source for Gemini VII. These performed nominally throughout the greater part of the mission; however, at 286:57 hr ground elapsed time, two of the stacks (2A and 2C) were shutdown because they were producing less than the specified current. The remaining stacks provided sufficient electrical power until re-entry. No corrective action was taken because the remaining Gemini flights were scheduled for a maximum of four days or less than 100 hr, about one-third the time to failure on the Gemini VII mission.

Post-flight investigation attributed the fuel cell problem to a blockage in the water management system late in the mission. Since the rest of the fuel cell system continued to operate with nominal performance, the anomaly was not deemed a system failure. An analog pressure transducer was nevertheless installed on Spacecraft 8 to provide a better monitoring capability in the event of a recurrence of this failure.

Another anomaly occurred when the operation of the No. 3 and No. 4 yaw right OAMS thrusters was degraded during the last 47 hr of the flight. However, performance was not seriously affected because the crew was able to retain yaw right control through the use of a maneuver thruster. Post-flight investigation indicated that the degraded thruster performance was probably due to an obstruction in the propellant flow or to propellant contamination. Since the anomaly was believed to have occurred as a result of long-duration operation, no change was made to the system configuration.

The 14-day mission was successfully completed and controlled re-entry was demonstrated by landing the spacecraft within 6.4 nautical miles of the planned retrieval point on 18 December 1965. Recovery was made by the carrier U.S.S. Wasp. All primary and secondary mission objectives were accomplished.

Gemini VIII Mission

The eighth Gemini mission was the first rendezvous and docking mission with an Agena Target Vehicle. Spacecraft 8 was launched successfully at 9:00 a.m., EST on 16 March 1966, following the launch of the Atlas-Agena Target Vehicle one hr and 40 min earlier.

The primary objectives for the Gemini VIII mission were:

A. Perform rendezvous and docking with the Gemini Agena Target Vehicle (GATV).
B. Conduct extravehicular activities.

The secondary objectives for the mission were:

A. Rendezvous and dock with the GATV during the fourth revolution.
B. Perform docked-vehicle maneuvers using the GATV secondary propulsion system.
C. Perform systems' evaluation.
D. Conduct ten experiments.
E. Carry out docking practice.
F. Perform a rerendezvous maneuver.
G. Evaluate the auxiliary tape memory unit.
H. Evince a controlled re-entry.
I. Park the Gemini Agena Target Vehicle in a 220-nautical-mile circular orbit.

The primary objectives of rendezvous and docking were accomplished during the fourth spacecraft revolution. The secondary objectives of evaluating the auxiliary tape memory unit and demonstrating a controlled re-entry were also accomplished. Because the mission was terminated early, extravehicular activity was not performed and only two of ten scheduled in-flight experiments could be conducted.

The Agena Target Vehicle was inserted into a 161.3 nautical mile circular orbit by its primary propulsion system. Spacecraft 8 performed nine maneuvers to rendezvous with the target five hr and 58 min after spacecraft lift-off. The spacecraft docked with the target vehicle after approximately 36 min of station keeping. Once docked, a 90 degree yaw maneuver was performed using the Agena attitude control system.

At 7:00:30 hr ground elapsed time, unexpected yaw and roll rates developed while the two vehicles were docked, but the command pilot was able to reduce these rates to essentially zero. However, after he had released the hand controller, the rates began to increase again and the crew found it difficult to control the spacecraft without using excessive amounts of propellant. The spacecraft was undocked and the yaw and roll rates then increased to approximately 300 degrees per sec, causing the crew to deactivate the orbit attitude and maneuver system (OAMS) and to use both rings of the re-entry control system to reduce the rates. The problem was isolated to No. 8 OAMS thruster which fired continuously because its circuitry failed in an ON condition. Post-flight investigation subsequently concluded that the anomaly was caused by a short circuit in the wiring to the thruster. To provide for immediate shutdown of all thrusters in case of malfunction on subsequent spacecraft, a single switch was installed which interrupted power to all thruster electrical systems. In this way, a systematic check of each system could be made per established procedure after individual systems were shutdown and the single switch closed. Inspection procedures were modified to reflect the engineering change.

Because the re-entry control system had been activated, the mission was terminated during the seventh revolution in the secondary recovery area in the western Pacific Ocean. Retrofire was on time at 10:04:47 hr GET, and the re-entry was nominal, resulting in a landing within seven nautical miles of the planned retrieval point. The crew and spacecraft were recovered by the U.S.S. Leonard Mason approximately three hr and 11 min after landing.

Gemini IX-A Mission

The ninth Gemini flight was a rendezvous and docking mission with the augmented target docking adapter (ATDA), used as the target vehicle after the Atlas failed to insert the Agena into orbit on 17 May 1966. The ATDA consisted of a target docking adapter (TDA), a cylindrical equipment section, a re-entry control system for attitude stabilization, a battery module, and an ascent shroud.

The ATDA was successfully launched on 1 June 1966, into a nearly circular orbit of 161 nautical miles. The Gemini Spacecraft was launched successfully at 8:39 a.m., EST, on 3 June 1966.

The primary objectives of the Gemini IX-A mission were:

A. Perform rendezvous and docking maneuvers with the target vehicle.
B. Conduct extravehicular activities.

The secondary objectives of the mission were:

A. Rendezvous and dock during the third revolution.
B. Accomplish systems evaluation.
C. Perform an equiperiod rerendezvous.
D. Achieve rerendezvous from above.
E. Demonstrate a controlled re-entry.
F. Perform docking practice.
G. Conduct seven in-flight experiments.

Rendezvous with the ATDA was accomplished by performing seven maneuvers during the spacecraft's third revolution. The first three maneuvers were conducted using ground-computed data. The terminal phase initiate maneuver was conducted using information from the onboard computer, ground computer, and onboard charts. The final three maneuvers were conducted visually with the assistance of the onboard computer and displays.

It was impossible to dock with the ATDA because the ascent shroud on the ATDA had not separated as planned. Subsequent inspection revealed that the quick-disconnect lanyards had not been properly attached. Two additional rendezvous were therefore performed in accordance with the alternate plan. The first was an equiperiod rendezvous (in which the spacecraft has the same orbital period as the target) and the second was a rendezvous from above, which was to simulate conditions which could result if the Apollo command module were required to rendezvous with a disabled lunar module. A two-hr EVA was accomplished, but evaluation of the astronaut maneuvering unit was not performed due to fogging of the pilot's visor.

On the third day several of the uncompleted in-flight experiments were performed. A nominal re-entry in the primary recovery area resulted in a landing one-third mile from the planned retrieval point on 6 June 1966. Recovery was made by the aircraft carrier U.S.S. Wasp.

Gemini X Mission

The primary objective of the Gemini X mission was to rendezvous and dock with the target vehicle. Secondary objectives comprised:

A. Perform the rendezvous and docking operation during the fourth revolution.

B. Utilize large propulsion systems in space (by using GATV primary and secondary propulsion systems in an attempt at dual rendezvous).

C. Accomplish extravehicular operations.

D. Perform docking practice.

E. Conduct 14 in-flight experiments.

F. Perform systems evaluations.

The tenth Gemini flight marked the second successful rendezvous and docking mission with an Agena Target Vehicle. The Agena was successfully launched on 18 July 1966 at 3:39 p.m., EST; Spacecraft 10 was successfully launched approximately one hr and 40 min later at the beginning of a 35 sec launch window. The Agena was placed in a nearly circular orbit with an apogee of 162.0 nautical miles and a perigee of 156.6 nautical miles. After spacecraft insertion, the insertion velocity adjust routine (IVAR) of the onboard computer was used to calculate the necessary velocity increment to achieve the planned orbit. Only a single velocity increment of 26 fps at insertion was required; this was applied by the crew and resulted in a spacecraft orbit of 145.1 nautical miles apogee and 86.3 miles perigee. These altitudes were only 0.1 mile low at apogee and 0.4 mile low at perigee, compared with planned altitudes.

The crew completed the rendezvous during the fourth revolution as planned, at 5:23 hr ground elapsed time. Approximately 30 min later the spacecraft docked with the Agena Target Vehicle. Since more propellant was used during the terminal phase of the rendezvous than had been predicted, docking practice was not performed in order to conserve the remaining propellant.

The spacecraft remained docked with the target vehicle for approximately 39 hr, during which a bending mode test was conducted to determine the dynamics of the docked configuration. In addition, a 49 min standup EVA was accomplished, which included several photographic experiments. The Agena primary and secondary propulsion systems were used to successfully accomplish six maneuvers in the docked configuration in preparation for a passive rendezvous with the Gemini VIII Agena Target Vehicle.

Approximately three hr after separating from the Agena (at 48 hr ground elapsed time), the Gemini Spacecraft achieved its second rendezvous. The Agena for Spacecraft 8 was found to be in a stable attitude; this allowed the flight crew to bring the spacecraft very close to the passive ATV. A 38 min EVA was performed during this station keeping period. As part of this EVA, the pilot retrieved the micrometerorite packaged which had been stowed on the ATV.

The planned three-day mission was accomplished successfully and was followed by a nominal re-entry on 21 July 1966. Touchdown was within three nautical miles of the planned retrieval point in the primary landing area.

Gemini XI Mission

Gemini XI was launched from Cape Kennedy on 12 September 1966 at 9:42 a.m., EST. The Agena Target Vehicle, with which it was to rendezvous and dock, had been launched one hr and 37 min earlier. The primary objective of this mission was to rendezvous and dock with the Agena Target Vehicle during the first revolution. The secondary mission objectives were:

A. Conduct docking practice.
B. Perform extravehicular operations.
C. Accomplish 12 in-flight experiments.
D. Conduct maneuvers with the Agena Target Vehicle while in the docking configuration.
E. Perform a tethered vehicle test.
F. Demonstrate an automatic re-entry.
G. Place the Agena Target Vehicle in a parking orbit.

All primary and secondary mission objectives were achieved; however, because of astronaut fatigue after installing the tether line, the D016 minimum reaction power tool experiment was not attempted.

Following spacecraft insertion, five maneuvers were performed by the crew to achieve the first-orbit rendezvous with the target vehicle. Docking with the Agena occurred at approximately 1:34 GET. At 40:30 GET, using the

Agena's primary propulsion system, the flight crew increased the apogee of the docked vehicles to 741.5 nautical miles. While at this altitude, sequences of photographic and scientific experiments were performed.

The spacecraft was undocked at 49:55 GET to begin the tether evaluation. The 100 ft tether line, which the pilot had attached to the docking bar on the previous day's EVA, was unreeled. A light tension was maintained on the tether and a slight spinning motion was imparted to create a small gravity field. Initial attitude oscillations began to damp slowly, and after 20 min the rotating combination had become stable. Performance demonstrated that the rotation of two tethered vehicles was an economical and feasible method of achieving long term, unattended station keeping. Approximately three hr after initiation of the maneuver, the crew fired the aft thrusters to remove the tension on the tether line. The docking bar was then pyrotechnically jettisoned, releasing the tether.

At 65:27 GET, maneuvers were begun to perform the coincident-orbit rendezvous with the target vehicle. Station keeping was accomplished at 66:40.

The re-entry operation was conducted very precisely using the automatic mode. Splashdown was 2.5 miles from the prime recovery ship, the U.S.S. Guam.

There were two significant anomalies that occurred during the Gemini XI flight. These were a failure in the radar transponder and a failure of stack 2C of the fuel cell.

Radar Transponder - Information taken from the technical debriefing, the air to ground voice transmissions, and spacecraft and target vehicle telemetry showed that initially the transponder was operating properly. This data did show that the transponder transmitter was gradually deteriorating. The first rendezvous was completed satisfactorily; however, initial signs of failure were evident. Eventually complete transmitter failure occurred coincident with a failure in the antenna switching circuit. An analysis of the circuit indicated that a failure of diodes CR610 and CR611 in circuitry common to the high voltage enable and spiral disable functions was the cause of the antenna switching failure. The nature of this failure coupled with transponder flight performance history indicated that arcing occurred within the sealed transmitter assembly. The most probable cause of failure was concluded to be a leak in the sealed transmitter assembly allowing the assembly to reach critical pressure and, thereby, sustain arcing.

Fuel Cell - Fuel cell stack 2C failed 54.5 hr into the mission. The five remaining stacks operated satisfactorily in sharing the total load; however, during two periods it was necessary to activate one of the spacecraft main batteries to insure that proper voltage levels were maintained. Postflight analysis led to the conclusion that a fire had developed in the stack, due to spontaneous combustion between the oxygen and hydrogen. However, a check valve upstream of the fuel cell prevented additional hydrogen from entering the stack, thus isolating the failure. A reevaluation of the stack's fabrication and installation did not reveal any anomalous conditions; neither had continual monitoring of stack pressure given any indication of impending failure. This malfunction was therefore deemed a random failure.

Gemini XII Mission

Gemini XII was launched at 3:46 p.m., EST, on 11 November 1966. The mission's primary objectives were:

A. Rendezvous and dock with the Agena Target Vehicle.
B. Perform extensive extravehicular activity.

The secondary objectives were:

A. Achieve rendezvous with the ATV during the third revolution.
B. Conduct docking practice.
C. Accomplish 13 in-flight experiments.
D. Conduct a tethered gravity - gradient test with the Agena target vehicle.
E. Perform a successful automatic re-entry.
F. Perform systems' evaluation.
G. Utilize large propulsion systems for space maneuvers (by employing GATV primary and secondary propulsion systems).

The spacecraft was inserted into an orbit with a 151.9 nautical mile apogee and a perigee of 86.9 nautical miles. Rendezvous and docking were accomplished during the third revolution as planned, over the tracking ship U.S.S. Coastal Sentry, south of Japan.

By applying a retrograde burn of 43 fps using the Agena's secondary propulsion system, the configuration was placed in a 154 nautical mile orbit, so that it could phase with the 12 November total solar eclipse over South America. A second eclipse - phasing maneuver was subsequently performed, enabling the crew to obtain the first solar eclipse photographs taken from space.

During the course of the mission, the pilot performed a total of five hr, 37 min of extravehicular activity, including the longest duration single EVA (two hr, nine min) to date. Included in this record was the performance by the pilot of measured work tasks at the ATV and at a work station setup in the Gemini adapter section.

The gravity gradient mode of the tethered vehicle exercise (Ref Gemini XI Mission, page 46 and Stable Orbit Rendezvous, page 293) was successfully completed; the entire tethered exercise lasted four hr and 17 min.

The spacecraft splashed down at 2:21 p.m., EST, on 15 November 1966, within 2.7 miles of the planned retrieval point, providing a further demonstration of the accuracy of the automatic re-entry mode.

All primary mission objectives were successfully accomplished, and almost all secondary objectives were achieved as well. The requirement for evaluation of the astronaut maneuvering unit (AMU) was deleted before the commencement of the mission; in addition, the plan to utilize the Agena propulsion system to reach a 460 mile altitude while over the U.S. was deleted due to a suspected problem with the ATV.

At approximately 60:00 GET, during the rendezvous maneuver, the spacecraft rendezvous radar/transponder ceased to function. Since the R & R was not recovered subsequent to flight, a positive identification of the failure mode could not be made. At the time of the preparation of this report, a review of Spacecraft 8, 11, and 12 launch and qualification test vibration data are being conducted. The vendor is also performing transponder tests to further investigate failure modes.

Performance of the fuel cells was normal during the launch count. At 5:45 GET, a water management system problem was revealed by the illumination of both section oxygen-water delta pressure lights. All available data indicate that the storage volume for water had been depleted by an oxygen leak into the water system. This depletion resulted in period of fuel cell flooding.

The fuel cell electrical performance was affected to the extent that two stacks had to be shutdown (stacks 2B and 1C) and two others experienced a significant loss of power. Both failed stacks exhibited a rapid drop of open circuit voltage potential, an indication of a burnout mode. Nevertheless, the remaining stacks and batteries provided sufficient electrical power to achieve all mission objectives. Development and ground test data on fuel cell anomalies are given in McDonnell Report F-205, Gemini Fuel Cell Performance Analysis.

At approximately 40 hr through the mission, the crew reported degraded performance of OAMS thrust chambers Nos. 2 and 4. Tests performed later in the mission with the thrust chambers, and post-flight analysis of data revealed degraded performance from all OAMS attitude thrusters. The best appraisal of the available facts points to propellant flow restriction as the cause of the performance loss. This restriction could have resulted from propellant contamination or the precipitation of iron nitrate from the oxidizer. McDonnell Report F-206, Gemini TCA Anomaly Investigation is being prepared for submittal to NASA.

Immediately after arming of the RCS A-ring, regulated pressure rose to a high of 415 psia (nominal is 295 psia). Following thruster initiation, however, the regulated pressure decreased to normal levels and standard regulator performance was indicated. Post-flight inspection and PIA tests of the regulator revealed no anomalies. Subsequent disassembling of the regulator at the failure analysis laboratory provided no positive explanation of the malfunction; however, scratches were found on the bellows, a metal flake was found on the valve seat, and contaminant material was discovered on the regulator spring.

The contaminant was sent to KSC for analysis. Since normal purging of the ring seemed to remove the contaminant from the regulator seat and normal system operation was restored, no further investigation of this anomaly was deemed necessary.

At 78:30 GET the pressure in the hydrogen vessel of the RSS rose to a high of 350 psi. After the flight crew had been directed to position the heater switch to OFF and the heater circuit breaker was opened, the pressure declined and the mission was able to proceed with no further hydrogen malfunction. Post-flight investigation of the switch revealed no anomaly. (Ref RSS Flight Performance, page 63.) The cause of the H_2 pressure rise was assumed to be intermittent short pulses of power, less than 2.4 sec in duration. This power application may have been due to an unstable hydrogen tank pressure switch. Reliability data on the pressure switch indicate that the unstable mode of this switch is the most likely failure mode. (Ref Electrical Memo No. PG-5236, dated 9 January 1967.) Since the H_2 heater has a redundant manual back-up, this anomaly is not considered critical.

MAJOR SYSTEMS

ELECTRICAL SYSTEM

Power Sources

The basic electrical power in the Gemini Spacecraft is provided by batteries, supplemented by fuel cells on most missions. Power is supplied through a multiple bus DC system in the range of 22 to 30 VDC. Subsystems which require closely regulated DC or AC voltage contain their own power conversion components, thus permitting optimum compatibility during flight operations as well as more intensive development and qualification programs.

The spacecraft power systems consist of a main bus, two squib buses, and one control bus. The flight crew can control interconnection of these sources, thus providing additional redundancy and maximum power utilization from all onboard supplies.

Prior to launch, the spacecraft gets external electrical power through the umbilical to prevent undue depletion of the spacecraft power supply. For Spacecraft 1, 2, 3, 4 and 6, batteries provided all the power required for electrically operated equipment. Fuel cells and battery power were used on all other flights.

In addition, ample battery power was available for operations two hr before launching, and for 36 hr after landing to operate recovery equipment. The batteries can also provide emergency power to operate the suit compressor for 12 hr after landing.

Fuel Cell System. - The General Electric fuel cell provided the main supply of electrical power for seven Gemini Spacecraft (Spacecraft 5 and 7 through 12). The fuel cell subsystem is made up of two GE fuel cell sections and an AiResearch reactant supply system (RSS). (See Fig. 7.) Both fuel cell sections and the RSS are located in the adapter equipment section of the spacecraft. The RSS is discussed in Reactant Supply System, page 59 of this report.

The fuel cell, supplied with necessary reactants and coolant, provides the main bus power (ranging from 22 to 30 VDC) for all mission requirements from insertion until switchover. During the prelaunch and the launch phases, the fuel cell operates in parallel with the main silver-zinc batteries, insuring power continuity in the event of the latter malfunction.

A. Fuel Cell Technology - A fuel cell is an electrochemical device that converts the energy of a fuel and an oxidant into electricity by sustaining a continuous chemical reaction. Reversing the process of electrolysis, the fuel cell creates water as a by-product in a hydrogen-oxygen ion exchange that liberates electrical energy.

The electrolyte of the GE fuel cell is a thin, treated sheet of polymer plastic. A catalytic electrode structure is bonded to each side of this sheet to stimulate ion-electron flow at relatively low temperatures. The fuel cell contains an anode and a cathode. When hydrogen fuel is brought into contact with the anode and a catalyst, the hydrogen atom releases an electron to the load drawn by the anode, and also releases ions. A solid polymer electrolyte keeps the hydrogen and oxygen gases separate, but allows the hydrogen ions to migrate through to the cathode, where they combine with oxygen and with electrons that have passed through the external circuit, thus producing water.

Balance is maintained by the migration of electrons through the electrical loads and the migration of ions through the electrolyte. Interruption of the electron flow in the external electrical circuit stops the reaction.

The cell converts chemical energy to electrical energy with about 50% efficiency. The remainder of the energy is largely rejected as heat. For dissipation of this waste heat, the fuel cells are directly connected to the environmental control system coolant loops. The temperature of the coolant entering the fuel cell sections is controlled by an ECS vernatherm valve, which mixes coolant returning from the spacecraft radiator with coolant coming directly from the pump packages, thus maintaining a temperature balance.

The water created by the fuel cell must be removed as the reaction continues. The pressure in the oxygen cavity in the fuel cell (approximately 22.2 psi) is used to drive the water through the porous sintered glass separator plates into the water collection basins beneath the cell. The pressure in these basins is maintained at about 20 psi and regulated by an ECS pressure regulator.

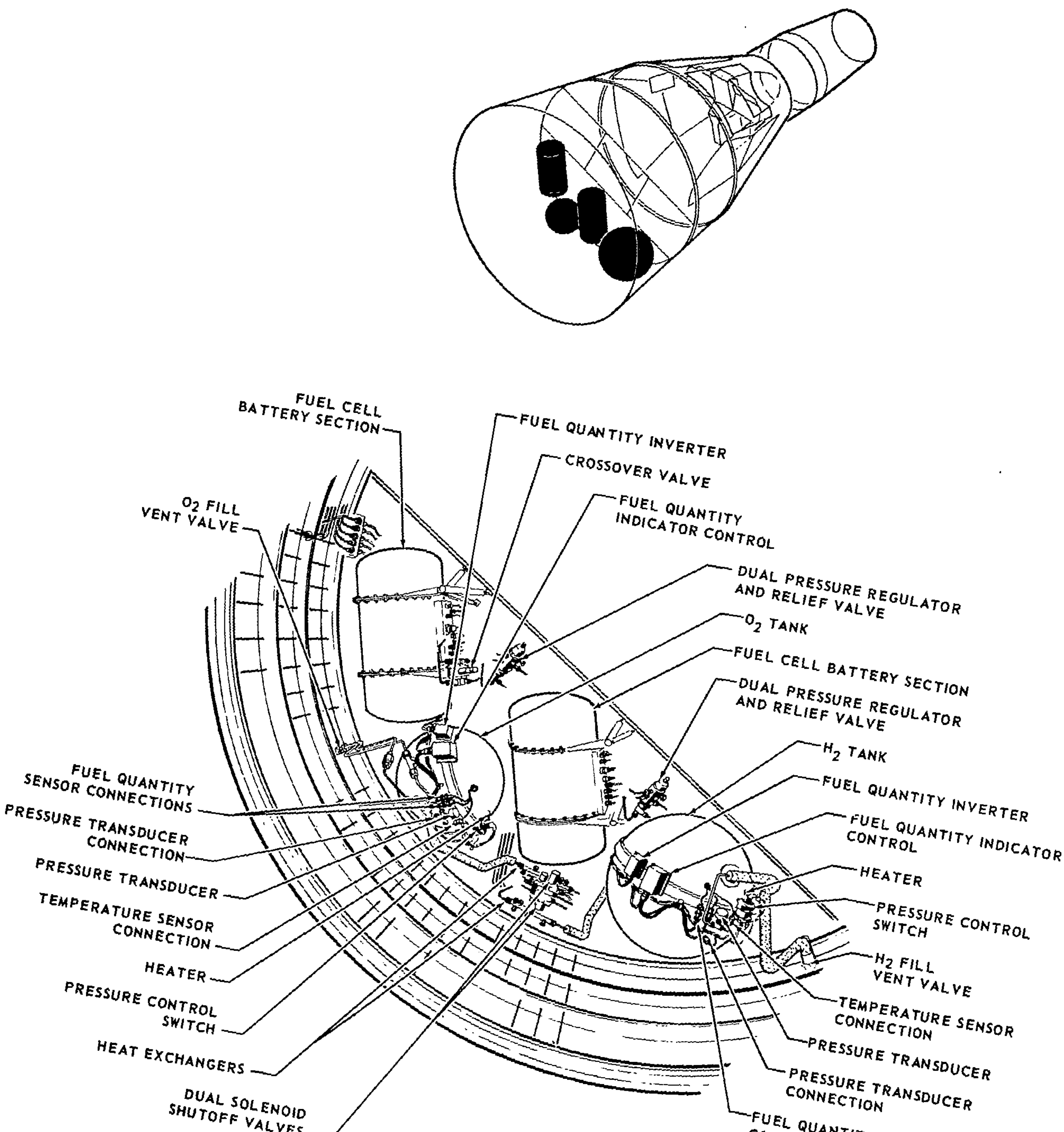


FIGURE 7 RSS/FUEL CELL MODULE

The Gemini fuel cell system comprises six electrically independent power-producing units (stacks). (See Fig. 8.) Each stack consists of 32 cells, connected electrically in series. The stacks are housed, in groups of three, in two cylindrical containers 0.610 meter long and 0.305 meter in diameter. Operating together, the two sections produce up to 2 kw at peak power. The crew can shut down individual stacks at will, or select any combination of stacks within the system. Each fuel cell section weighs 68 lb and produces up to 1 kw of DC power at 26.5 V, minimum load, or at 23.3 V, peak load. The latter values are exhibited at the end of rated life. Initial voltage is approximately 2 V higher in each case.

The fuel cell section is the smallest field-replaceable power-producing unit of the Gemini fuel cell system. An interface plate, which contains the hydrogen, oxygen, and water valves, the coolant ports, and the electrical connectors is mounted on each cylindrical container.

To prevent a malfunction from impairing the entire system operation, stack is isolated and independently supplied with regulated gaseous hydrogen and oxygen. In addition, each stack within a section had its own hydrogen inlet and purge valves. One oxygen purge valve is also provided for each section. Although purging is normally performed by section, individual stacks can be hydrogen-isolated or purged on the command of cabin control equipment.

The water collection basins, one beneath each stack, are connected in series; their output can be isolated from the spacecraft water system by means of a fuel cell latch solenoid valve.

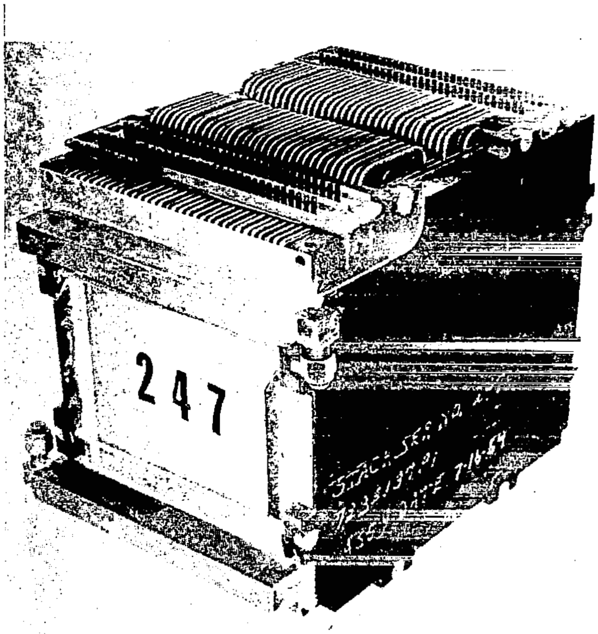
B. Development and Qualification Tests for the Fuel Cell - A significant advance in the state-of-the-art involved in the fuel cell was accomplished with the aid of an extensive development and qualification test program.

Early development tests concentrated principally upon various cell designs, arranged in configurations of fewer than 32 cells each. These tests served the dual purpose of developing efficient assembly techniques and verifying, on a small scale, cell performance for various current densities, coolant temperatures, and other operational parameters.

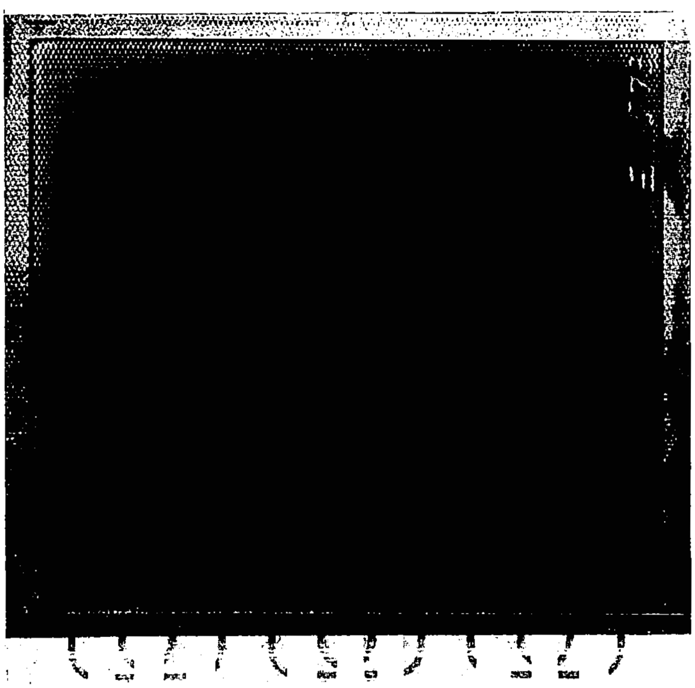
Performance investigations were conducted on this configuration, at the section level, to evaluate operational procedures, cell and stack temp profiles, purge flow rates, total heat rejection, vibration and life limiting factors.

These tests culminated in the first production-type sections of the P2B design.

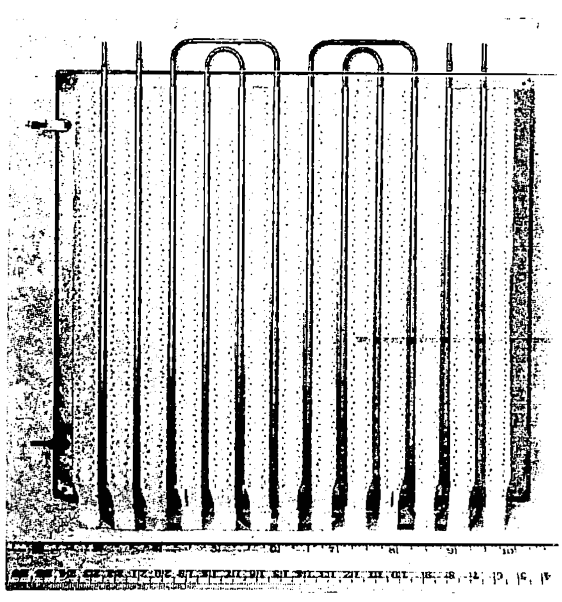
Qualification testing of the P2B cell design included vibration, acceleration and shock tests. The P2B test sections were successfully tested under the conditions producing an electrical load consistent with Gemini mission requirements.



FUEL CELL STACK CONTAINING INDIVIDUAL
FUEL CELL ASSEMBLIES



HYDROGEN SIDE OF INDIVIDUAL CELL WITH
MEMBRANE AND ANODE INSTALLED



OXYGEN SIDE OF INDIVIDUAL CELL ASSEMBLY
PRIOR TO MATING WITH THE MEMBRANE

FIGURE 8 FUEL CELL STACK AND CELLS

Based on the service life expectance of the P2B design, a number of design improvements were developed and were incorporated into a new configuration, known as the P3 design. Verification tests of the new design were conducted to determine such items as coolant and pressure level requirements, and operating life. The qualification test program imposed conventional limits of acceleration, low temperature, temperature-pressure, and vibration. Three simulated mission performance evaluation tests were conducted which demonstrated that the P3 design would meet operational requirements.

Separate qualification test programs were conducted on critical components such as valves, membrane and electrode assemblies, and manifolds. Deficiencies encountered were resolved on an individual basis and components were retested as necessary to prove the designs. The fuel cells were integrated with an RSS and a simulated spacecraft distribution system and were operated in several ambient tests and under temperature-pressure and vibration conditions. The fuel cell sections performed adequately in these tests, although a design problem involving water valve corrosion was encountered.

A stack storage program was conducted to determine any performance degradation or leakage. The results indicated that the storage time after initial activation of the fuel cell stacks must be limited, since performance decreases in direct proportion to storage time. It was also determined, however, that the voltage at second activation (at launch) is greater than the voltage level at first activation.

Therefore, the following procedure was employed. At the time of premate, the system was initially activated, and then drained of product water and flushed with distilled water to reduce corrosion hazards. The old water shutoff valve was replaced with a new valve prior to flight. This initial activation conditioned the stack electrodes, thus providing higher voltage levels during second activation. Experience gained from laboratory tests (and Spacecraft 5 flight) demonstrated no loss to system performance for at least 30 days after the first exposure to product water. The techniques evolved held fuel cell degradation to limits of the original acceptance criteria.

C. Reliability and Quality Assurance Program Results - The reliability and quality assurance program on the fuel cells comprised essentially four series of tests: (1) a nine-stack forced-failure test program, (2) a 5% sampling of the manufactured stacks, (3) a valve endurance test program, and (4) a number of stress-margin tests.

The nine-stack forced-failure test program was run at loading and environmental conditions that were well beyond specification limits. These tests showed that the most significant restraint on fuel cell operation was temperature. Longer life resulted as the coolant inlet temperature was lowered. The tests also showed that performance was lowered when the operating pressure of the hydrogen was reduced even though stack life was not significantly affected.

The manufacturing consistency and quality of production cells was monitored by sampling electrical performance capability of 5% of the stacks during life endurance tests. These stacks had an average life of 1086 hr, which well exceeded the requirements for a 14-day Gemini mission.

Endurance (cycling) tests were performed on the oxygen, hydrogen, and water valves and the pressure bellows to determine their endurance and to ascertain wear out pattern and failure rate. All three bellows were tested to 129×10^3 cycles and the three O_2 purge valves were tested to 194×10^3 before failure occurred. Each inlet and purge valve was tested to 21×10^3 cycles. Fifty percent of these failed, but results compared favorably with the 15×10^3 cycles quoted by the switch manufacturer. The H_2O valve was subjected to 8 x 103 cycles and no discrepancies were encountered.

Stress margin tests were conducted on the water separator assembly to determine the extent to which pressure, temperature, flow, and vibration would affect its operation. These tests demonstrated that the assembly could withstand stresses in excess of the SCD 52-79700 specification for normal operation. Of the eight units tested, five were tested for leakage under normal condition and exceeded specification limits of 50 cc/hr. The remaining units were subjected to three tests: high temperature random vibration, pressure cycling, from both the water and the oxygen sides, and imposition of a water pressure 20.5 psi greater than the oxygen pressure (reverse pressure).

These tests indicated that the assemblies could withstand conditions in excess of the required limits before leakage above the established specification was evident.

D. Significant Anomalies and Corrective Action for Fuel Cells - The major anomalies encountered during several Gemini missions and the corrective action taken are presented in the following paragraphs.

Spacecraft 2 - Electrical power for Spacecraft 2 was supplied by four silver-zinc main batteries, three silver-zinc squib batteries, and four silver-zinc special pallet batteries. All fuel cell stack hydrogen inlet valves were closed prior to launch because a prelaunch facility malfunction made timely activation impossible, the hoped for engineering evaluation of flight performance was not accomplished. However, in-flight information was obtained on launch effects on the pressurized static reactant supply system. The fuel cell sections of the first (P2B) design, as used in this mission, are mentioned in Development and Qualification Tests for the Fuel Cell, page 53.

Spacecraft 5 - In preparation for Spacecraft 5 delivery and launch, tests indicated an activation and storage limitation. Corrosion of the fuel cell shutoff valve and the spacecraft plumbing was eliminated by changing the material of the valve and water connections and by flushing and drying the system afer the first activation. (Ref Development and Qualification Tests for the Fuel Cell, page 53.) In addition, a valve for product water was installed. This valve was improved for later missions.

Contamination of the by-product water, primarily by organic acids, resulted in the withdrawal of the plan to use by-product water for drinking. (Ref Product-Water Potability, page 59.) Therefore, a storage tank was installed between the by-product water outlet and the ECS pressure regulator. A three-ply Teflon bladder in this tank isolated the by-product water from the cabin water supply.

During the Spacecraft 5 flight, the fuel cell performance was nominal, although system problems led to unusual modes of fuel cell operation. The following changes were therefore made on later spacecraft: (1) coolant pump inverters were redesigned to give a high or low flow capability for each loop, thus conserving power, and (2) coolant loops were reconnected to establish an independent coolant flow to each section.

Attention during the mission was focused on the water management system when calculated water production rates created concern over adequate storage volume. Actions resulting from this on later flights were to improve measurement of astronaut water consumption thus improving water production rate determination and the storage capacity required for maintaining the water pressure at acceptable levels was greatly increased.

Spacecraft 7 - The fuel cell configuration on Spacecraft 7 was essentially the same as that for Spacecraft 5. During the mission, an apparent blockage or restriction developed in the water management system, which periodically affected the performance of the three stacks of one fuel cell section. The evidence did not indicate a system design problem, however, and all phases of the mission were accomplished satisfactorily.

Pictures taken of Spacecraft 7 in orbit disclosed an ice formation at the hydrogen vent port. Modifications were made to the vent on later spacecraft to prevent this icing.

Spacecraft 8 - The Spacecraft 8 fuel cell system was the same as the Spacecraft 7 system, except for the hydrogen vent modification. Flight performance of the fuel cell during this mission was nominal.

Spacecraft 9 - The fuel cell configuration installed on Spacecraft 9 was the same as that used on Spacecraft 8. The stacks were activated about 15 hr before the scheduled launch on 17 May 1966. However, failure of the Agena Target Vehicle to achieve orbit resulted in a two-week postponement of the mission.

Because of apprehension that the twice-activated stacks would produce a reduced voltage (Development and Qualification Tests for the Fuel Cell, page 53), a new fuel cell system was installed and was launched on 3 June 1966, as part of the Spacecraft 9 mission. This system was nominally activated and performed without malfunction throughout the mission

Spacecraft 10 - To make room in the adapter section of Spacecraft 10 for two additional orbit attitude and maneuver system (OAMS) bottles, modifications were made to the fuel cell system. The life support oxygen supply, which had provided oxygen only to the cabin, was redesigned to supply the fuel cell, also. The installation and activation of the flight system was accomplished without problems, and system performance was nominal throughout the mission.

Spacecraft 11 - The Spacecraft 11 fuel cell system was the same as those on Spacecraft 7 through 10. In preparation for the first stack activation at premate, the system originally installed in Spacecraft 11 received an inadvertent overpressurization of oxygen. This imbalance was found to be caused by a malfunctioning aerospace ground equipment (AGE) regulator. The flight fuel cell sections were therefore replaced and the AGE regulator repaired; the old fuel cell sections were set aside for bench testing in order to determine their condition.

During the first activation load check of the replacement sections, an AGE valve failure caused helium to be introduced on the hydrogen side of the cells of both stack sections. The problem area was isolated from the system and hydrogen was reintroduced, whereupon a normal deactivation was begun.

It was subsequently discovered that, due to a procedural error, approximately 20 hr of deactivation were accomplished without coolant flow being circulated through the sections. An investigation was undertaken with the following results: (1) the sections were adjudged acceptable for flight; (2) the AGE components were given a special inspection and preventive maintenance measures were taken; (3) particular attention was given to fuel cell protective procedures.

The launch of Spacecraft 11 was twice delayed due to launch vehicle problems. During the delay, the fuel cells remained activated and were operated at three amp per stack. Once into the mission, the C stack of Sect. 2 failed at 54 hr and 31 min elapsed time. Review of the flight data revealed that this was a rapid failure, most probably attributable to burnout; no indication of impending failure had been received. It was impossible from the mission data to determine a single cause for this failure. Even with the failed stack, the electrical power generating capability of the fuel cell system was nominal, and all mission requirements for power were met.

Spacecraft 12 - Fuel cell installation and activation (Ref Development and Qualification Tests for the Fuel Cell, page 53) proceeded normally. All data indicated normal performance during the launch count. At 5:45 into the mission, both section oxygen-water delta pressure lights displayed the presence of an anomaly in the water management system. While no final determination of the failure mode can be made, indications are that a depletion of the water storage volume, used to maintain pressure control, had occurred. The cause of the depletion was most probably the leakage of oxygen into the water system. As a result of the loss, fuel cell flooding occurred.

The electrical performance of the fuel cell was affected to the extent that during the mission two stacks had to be shut down (stacks 2B and 1C) and two others experienced a significant loss of power. Both failed stacks exhibited a rapid drop of open circuit voltage potential - an indication of a burnout mode. However, the remaining stacks and batteries provided sufficient electrical power to achieve all mission objectives. At the time of this writing, McDonnell Report F-205, Gemini Fuel Cell Performance Analysis, is being prepared for submittal to NASA. It provides a detailed study of fuel cell performance and corrective action.

E. Product-Water Potability - Because of the original intent to utilize fuel cell product water for astronaut consumption, a tank for storage of the product water was mounted on the ECS oxygen module on Spacecraft 2. However, prior to the Gemini V flight (the next mission to utilize the fuel cells), tests indicated contamination of the by-product water by organic acids.

Neutralization of the corrosive and acidic qualities of the product water was attempted by using ion exchange columns. These columns were to be mounted on the fuel cell module, one column for each section. However, significant diversity appeared among the product water samples obtained in testing; experimentation with different resins failed to produce an ion column which would be suitable for all applications. Therefore the decision was made to store the fuel cell product water and provide additional containers for potable water.

The following storage provisions were therefore made:

1. On Spacecraft 5, two 20-in. diameter tanks were installed. The A tank contained drinking water inside a bladder, pressurized by gas. The B tank contained drinking water inside a bladder, and <u>both</u> product water and gas outside the bladder.
2. The same tanks were utilized on Spacecraft 7, except that the A tank, containing the drinking water, was pressurized by product water and the B tank contained gas in the bladder and fuel cell water outside. Because of the long duration of this mission, a third tank was installed, in which drinking water was pressurized by oxygen.
3. On Spacecraft 8 and up, two tanks were provided. In one, fuel cell product water was pressurized by nitrogen, in the other drinking water surrounded a bladder containing product water.

A more detailed analysis of this problem is presented in McDonnell's design note History of the Gemini Drinking Water System, dated 6 July 1966.

<u>Reactant Supply System.</u>

A. Design Concept - Reactant supplies for the fuel cell are located in the adapter section of the spacecraft in two double walled, vacuum insulated, spherical containers. The reactants, oxygen and hydrogen, are stored in these containers at supercritical pressures and at cryogenic temperatures. They provide a "single phase" fluid (i.e., neither a gas nor a liquid) to the heat exchangers and to the dual O_2-H_2 pressure regulators. Upon reaching the heat exchangers, each fluid is converted to a gas, and supplied to the fuel cells at operating temperatures.

Several advantages accrue to system design through the use of this cryogenic storage. Because this storage can be accomplished at lower pressures than would be feasible with a similar quantity of gas stored in the same amount of space, the danger of structural failure is greatly reduced. For the same reason, associated components can be fabricated from lighter materials. Furthermore, waste heat from the ECS coolant circuits is utilized in the heat exchangers, instead of being vented overboard.

If, on the other hand, the reactants were stored as low-pressured liquids, an additional system to create the necessary delivery pressure at high flow rates would be required; another disadvantage of this method would be that weightlessness and certain acceleration forces might at times provide a two phase (gas and liquid) or unstable, supply of reactants to the heat exchangers.

RSS oxygen is stored in the spacecraft at approximately -297°F at 14.7 psia; RSS hydrogen is stored at approximately -423°F at the same pressure. The tanks are initially filled with the hydrogen and oxygen gas which is replaced with cryogenic reactant 12 to 24 hr before launch. Because the tanks are filled from the bottom, it is impossible to drive out all the gas. This results in what is known as a two phase condition (i.e., gas and liquid). In order to restore the single phase condition, the heaters within the tanks are energized, causing the cryogenic liquid to expand. Since this expansion takes place in a closed tank, the internal pressure rises. The gas molecules are pressure driven into a homogeneous association with the liquid molecules. This compressed fluid state is the single phase condition which is necessary for optimum system operation.

The heaters remain ON until normal operating pressures are reached (approximately 910 psi for oxygen, 240 psi for hydrogen). Maximum bottle pressure is carefully maintained by the relief valves on the heat exchangers. At both high and low reactant density (i.e., during the initial and final phases of the mission), minimum heater activation is required. When the reactants are at approximately a middle density, at the midpoint of the flight, natural heat leak from the storage containers supplies minimum cryogen flow.

The flow of reactants to the fuel cell system is reduced to delivery pressures by the dual O_2-H_2 pressure regulators and dual relief valves. The pressure regulators use the regulated gas pressure in the by-product water storage tank as a reference to regulate the hydrogen pressure, which in turn is used as the reference to regulate the oxygen pressure. (A schematic of the RSS system is illustrated in Fig. 9.)

The system senses the generation of product water, and permits sufficient reactant flow to the fuel cell cavity to maintain a constant pressure on the porous glass separator plates above the water basins. (Ref Fuel Cell Technology, page 51.) Product water pressure is maintained at approximately 20 psi. Hydrogen pressure is reduced to approximately 21.7 psi for delivery to the fuel cell; oxygen pressure is reduced to approximately 22.2 psi. Thus the regulators are set to maintain a hydrogen pressure of approximately 1.70 psi greater than the product water pressure, and an oxygen pressure

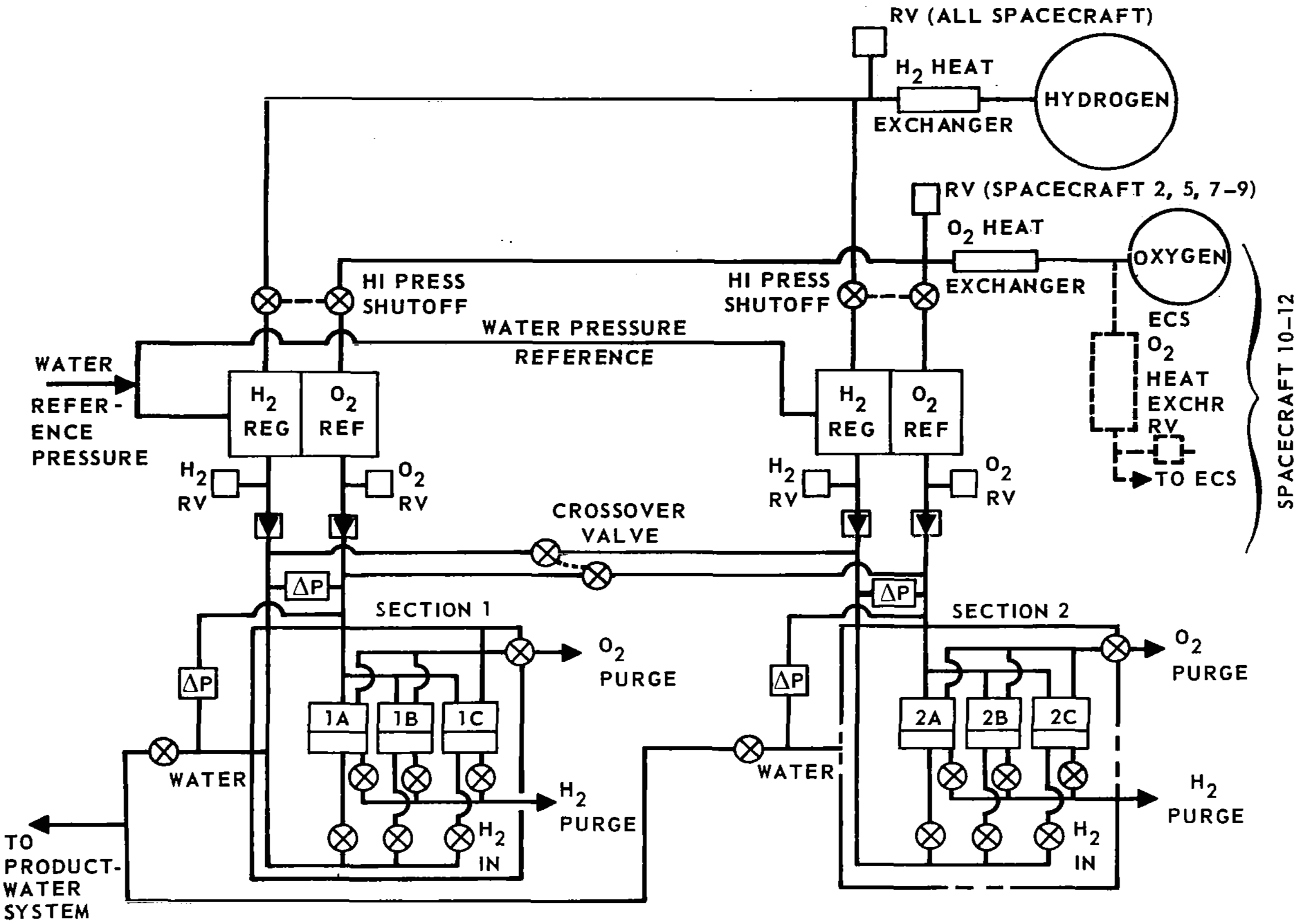


FIGURE 9 FUEL CELL RSS SCHEMATIC

approximately 0.5 psi greater than the hydrogen pressure. These pressures are read out on the differential pressure meter in the cabin and on ground telemetry.

The temperature of the reactants provided to the fuel cell is controlled by the reactant supply heat exchangers, located immediately downstream of the storage tanks. These heat exchangers automatically control the reactant temperature by absorbing and transmitting heat from the recirculating coolant fluid. When leaving the storage tanks, the temperature of the oxygen is in the range of -297°F to -160°F; the temperature of the hydrogen ranges between -423°F and -360°F. Before the reactants are supplied to the fuel cells, the heat exchangers raise their temperatures to a minimum of 50°F and a maximum of 140°F.

Normal operation of the fuel cell requires that the system be purged several times during a mission. Purging removes inert gases that accumulate within the cells and restores fuel cell operation to near its original level. The frequency of the purges is in proportion to the section current being drawn. For section currents of 18 to 30 amp, a purge is required every four hr; for currents of 30 to 45 amp, a purge is necessary every two hr. The normal section purge takes approximately 11 sec for hydrogen and two min for oxygen.

In addition to the components mentioned, provisions for servicing the system require shutoff valves which isolate the O_2 and H_2 storage tanks from the rest of the RSS, and check valves which permit gaseous reactants to be introduced from AGE sources.

B. Development Test for the Reactant Supply System - The components of the RSS were developed to meet the specific applications of a cryogenic, supercritical environment. Subjected to extensive development testing were the cryogenic storage tanks, the water and gas pressure regulators, the heat exchangers, the control, check, and relief valves, and the gauging systems.

This development testing evolved the ability to use the same type of heat exchanger in both hydrogen and oxygen systems. In addition, since the oxygen portion of the RSS was similar to that of the environmental control system (ECS), an interchange of information and hardware was possible in certain areas.

Development testing of both the hydrogen and oxygen high pressure relief valves revealed internal leakage and an out-of-tolerance condition caused by wear on internal operating parts. New valves which held tolerances to required specifications were therefore tested; in addition, revised inspection and cleaning procedures were adopted.

Tests conducted on the hydrogen cryogenic containers revealed the need for a redesign because the internal retaining nuts were backing off under vibration. Additional testing verified that new locking techniques would withstand vibration.

C. Qualification Tests for the RSS - Wherever possible, qualification of the reactant supply system was demonstrated at the system as well as at the component level, due to the close interrelationship of all components in supplying satisfactory performance.

The RSS qualification test status is summarized in Table 2. All subassemblies and components were successfully subjected to the complete test program, which includes environmental, dynamic and temperature-altitude testing.

In addition to this program, both the oxygen and hydrogen subsystems were successfully subjected to six 14-day simulated missions. Test conditions covered the full range of fluid quantities, system pressures and internal-heater duty-cycles expected in Gemini flights. In addition, fuel-cell purge cycles were simulated to include the effect of sudden, brief flow rate excursions.

TABLE 2 QUALIFICATION STATUS – REACTANT SUPPLY SYSTEM

SPACECRAFT 5 & UP DESIGN APPROVAL

COMPONENTS	MAJOR ENVIRONMENTS										
	DASH NO.	HIGH TEMP	LOW TEMP	VIBRATION	ACCELERATION	SHOCK	RFI	HUMIDITY	REQ'D 5 & 7 (LM)	REQ'D 8 & UP (SM)	COMPLETE QUAL
SUBSYS (TANK) SM O_2	–203	Δ	Δ	C	C	Δ	Δ	Δ	Δ		C
SM H_2	–205	C	C	C	C	Δ	Δ	Δ	Δ		C
GAGE CONT SM O_2	–83	Δ	Δ	C	C	Δ	C	Δ	Δ		C
SM H_2	–85	Δ	Δ	C	C	Δ	C	Δ	Δ		C
CHECK VALVE – H_2 & O_2	–95	C	C	C	C	Δ	Δ	C			C
LO PRESS VALVE & H_2–O_2	–181	C	C	C	C	Δ	Δ	C			C
REGULATOR H_2–O_2	–185	C	C	C	C	Δ	Δ	C			C
HI PRESS VALVE H_2–O_2	–195	C	C	C	C	Δ	C	C			C
CONTAINER SM O_2	–167	C	C	C	C	Δ	Δ	C	Δ		C
SM H_2	–165	C	Δ	C	C	C	Δ	C	Δ		C
PRESS SW – O_2	–183	C	C	C	C	Δ		Δ			C
H_2	–179	C	C	C	C	Δ	Δ	C			C
TEMP SENSOR	–37	C	C	C	C	Δ	Δ	C			C
PRESS TRANSD H_2	–197	C	C	C	C	Δ	Δ	C			C
O_2	–199	C	C	C	C	Δ	Δ	C			C
INVERTER – GAGE	–45	C	C	C	C	Δ	C	C			C
FILL & VENT VALVE H_2	–57	C	C	C	C	Δ	Δ	C			C
O_2	–59	C	C	C	C	Δ	Δ	C			C

C = TESTING COMPLETE Δ = TESTING NOT REQUIRED

D. RSS Flight Performance - The RSS performed with only minor anomalies on Spacecraft 2, 7 and 8. Shortly after the launch of Spacecraft 5, the oxygen pressure control heater failed. As a result, oxygen pressure decreased below the normal control range; however, the system continued to perform satisfactorily at the lower pressure level. It was not possible to determine whether this failure occurred within the heater or in the spacecraft wiring. The following corrective action was therefore taken:

1. Redundant wiring was provided to the oxygen heater.

2. A high-pressure crossfeed was installed linking the ECS oxygen supply tank and the fuel cell oxygen reactant tank. This crossfeed connection, made downstream of the heat exchangers, allowed warm oxygen from the ECS tank to be supplied to the RSS tank in the event of an RSS heater malfunction. Pressurization from one tank would be sufficient to pressurize both systems with the crossfeed on. The possibility of another failure was minimized, since the ECS oxygen tank has redundant heaters.

The long-mission hydrogen tank in Spacecraft 7 was modified to improve its thermal performance by adding a regenerative cooling line, insulated container mounts, and an external wrapping of several layers of Mylar. A pyrotechnic pinch-off tube cutter was added to blow open a valve in the outer hydrogen vessel in the event of heat leak during the mission. Blowing this valve allowed the pressure of the space vacuum to restore the vacuum between the inner and outer bottles of the hydrogen vessel, thus holding the heat leakage to a minimum.

Subsequent to Gemini VII, all spacecraft utilized the short-mission hydrogen vessel and either a short-mission RSS oxygen vessel cross-connected to a long-mission ECS oxygen vessel (Gemini VIII and IX) or a shared long-mission ECS oxygen vessel alone (Gemini X through XII). Pyrotechnic pinch-off tube cutters were added to each of the hydrogen vessels for all remaining spacecraft.

At 26:58 into the Gemini IX mission, the hydrogen vessel indicated zero fluid quantity, both on the cabin readouts and through telemetry. Postflight investigation revealed that the panel gauge, its internal telemetry potentiometer, and the gauge-control output circuitry were operating properly. It was therefore concluded that the failure had occurred in the quantity sensor or in the sensor-to-controller wiring. Since both of these potential failure points are in the equipment adapter, further failure analysis was impossible. However, since all other data indicated that the RSS continued to function properly, loss of hydrogen quantity indication was felt to be an insignificant anomaly, and the mission continued with no need to utilize either back-up systems or alternate procedures.

Special operating procedures were written for experimental in-flight use of the hydrogen pinch-off-tube cutters on Spacecraft 9 through 12. Loss of the hydrogen quantity readout on Spacecraft 9 prevented precise assessment of the thermal performance improvement. The cutter was again successfully blown on Spacecraft 10 but the lack of long-duration constant-extraction-rate periods prevented accurate performance determinations. On Spacecraft 11 the hydrogen vessel exhibited a 9.5 improvement (heat-leak decrease) 24 hr after blowing the valve; on Spacecraft 12 flight data confirmed a 22% improvement. It is apparent that valve blowing is effective in improving vessel performance to the original as-built level, thereby nullifying the degradation which occurs in the months prior to flight. There is no indication, however, that this action improves the vessel beyond the original performance limits.

At 78:30 hr into the Gemini XII mission, the pressure in the hydrogen vessel of the RSS rose to 350 psi (240 psi is the normal operating pressure).

This situation was corrected by directing the crew to reposition the heater switch from AUTO to OFF. Subsequently the H_2O_2 heater circuit breaker was opened. Analysis demonstrated that the heater had been intermittently activated over a 103-min period during the H_2 pressure increment. However, flight data revealed that after switch operation the tank pressure gave indications of declining prior to the opening of the circuit breaker. Post-flight investigation included an inspection of the connector potting, of the wire bundles, and of the switch mounting. No shorting conditions were revealed. The heater switch was then removed for failure analysis, during which it was subjected to x-ray inspection, contact IR drop measurements, cycling, dielectric testing, and dissection. No discrepancies were found as a result of this investigation. Because of adapter loss, it was not possible to determine if failure in the pressure-switch or in the adapter loss, it was not possible to determine if failure in the pressure-switch or in the adapter wiring had caused the anomaly. Once the heater had been deactivated, the mission was able to proceed with no further RSS incident.

Batteries.

A. System Concept - Three types of silver-oxide zinc Eagle Picher batteries were employed on Gemini Spacecraft. The standard complement was four 45 amp/hr main batteries, located in the re-entry module, three 15 amp/hr squib batteries, stowed in the spacecraft right hand equipment bay, and, for spacecraft which did not employ fuel cells, three 400 amp/hr batteries installed in the adapter section. (See Fig. 10.) All the batteries are activated and sealed at sea level pressure; all are capable of operating in any attitude in a weightless state.

Each battery case has a pressure relief valve to permit the escape of gases. Battery temperatures are controlled by mounting the battery cases in direct contact with spacecraft coldplates.

In every case, prime consideration was given to the maximum power-to-weight and power-to-volume ratios, and to high reliability, storage, and activated life. Simplified activation and test procedures, plus minimized handling requirements, were important factors in the design and development of these batteries.

A significant advancement in the state-of-the-art was the use of titanium instead of stainless steel as the case material in both the main battery and the squib battery. This substitution resulted in a weight decrease of 16% or more with no loss of battery strength.

B. Main Batteries - The four batteries installed in the re-entry module supplied a portion of the main bus power during launch and all of the main bus power during the re-entry, landing, and post-landing phases of the mission. Each of these batteries has a rated capacity of 45 amp/hr at a discharge rate of 20 amp and an activated stand life of 30 days. An analysis of individual battery performance based on telemetered data and post-mission testing indicated that each battery possessed an average capacity of 49.32 amp/hr at potentials up to 20 volts.

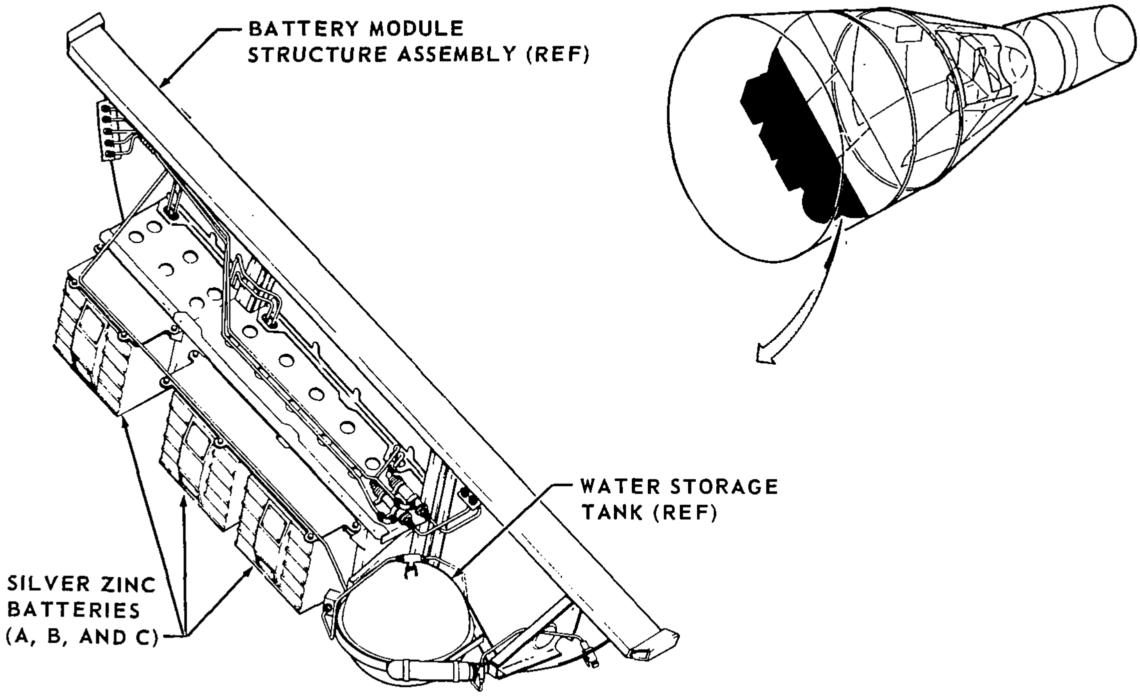


FIGURE 10 ADAPTER BATTERY MODULE

The change to titanium cases increased the performance at 24 volts from 45 hr per pound of battery weight to 54 hr per pound. Estimated mission loads on the four batteries varied from 32.0 amp/hr on Gemini V to 118.01* amp/hr on Gemini XII, with an average mission load of 55.12 amp/hr per spacecraft through the Gemini XII flight. (*This figure includes fuel cell back-up usage prior to retrofire.)

C. Squib Batteries - The three squib batteries supply power for squib-activated pyrotechnic devices throughout the mission; in addition, they provide power to the common control bus for the relays and solenoids to the environmental, communication, instrumentation, and propulsion systems. These batteries are mechanically and electrically isolated from all other batteries in the system.

To provide a redundant power supply in the event of the malfunction of the squib battery section, the astronauts can connect squib circuitry to the main batteries.

Each squib battery has a rated capacity of 15 amp/hr, an activated stand life of 14 days, and a power-to-weight ratio of 42 watt-hr per pound of battery weight. Post-flight discharge testing on each battery indicated an average capacity of 14.73 amp/hr. An additional property of this battery is its ability to respond rapidly to high loading conditions, making it especially effective in firing high-load, short-duration squibs.

Power delivered to the spacecraft by the three squib batteries was estimated to range from a low of 6.0 amp/hr on the Gemini VI mission to a high Of 27.4 amp/hr on the Gemini V mission.

In addition to the installation on the Gemini Spacecraft, two 15 amp/hr squib batteries were mounted in the augmented target docking adapter (ATDA) for the Gemini IX A mission. Their purpose was to supply power to the squib bus for ATDA/launch vehicle separation, for shroud separation, and for RCS pyrotechnic devices.

D. Adapter Batteries - Adapter batteries were used in Spacecraft 3, 4 and 6. Spacecraft 3 and 6 each had three 400 amp/hr batteries installed in the adapter section; these batteries provided the primary source of electrical power for the two relatively short missions. Spacecraft 4, the only spacecraft to fly an extended mission (four days) without fuel cells, required six 400 amp/hr batteries.

On the three missions in which these batteries were utilized, they provided power to the main bus for the communications, environmental, guidance, and instrumentation systems, as well as cabin and EVA lighting and power for customer experiments.

The discharge rate of each 400 amp/hr battery is between 20 and 25 amp; each has an activated stand life of 30 days. By using magnesium for the battery case, a power-to-weight ratio of 83 watt-hr per pound of battery weight was achieved. Power delivered to the spacecraft by the adapter batteries was estimated to range between 265 amp/hr for the Gemini III mission and 2032 amp/hr for the Gemini IV mission. Neither the ATDA nor the adapter sections were recovered; therefore no post-flight measurement of the capacity of the flight batteries was possible.

Three 400 amp/hr batteries were installed on the ATDA for the Gemini IX A mission. These batteries provided primary power to the main bus for the following systems and functions: the digital command system (DCS), the attitude thrusters, two C-band radar transponders, the TM subsystem, the rendezvous radar transponder, and the docking, latching and unrigidizing operations.

E. Qualification and Reliability Test Program for Batteries - Eleven 45 amp/hr batteries, eleven 15 amp/hr batteries, and nine 400 amp/hr batteries were used in the qualification and reliability test program.

The philosophy of the qualification test program was to duplicate, as closely as possible, maximum-duration mission conditions, from battery activation through spacecraft recovery. For this purpose, batteries were subjected to a sequence of environmental tests, including vibration, high temperature, landing loads, and shock tests. In addition, the stand life and mission duration capabilities of the batteries were rated.

The reliability tests demonstrated that the three types of batteries could withstand extremes of environment and activated stand life and still produce the rated capacity. Additional tests on the 400 amp/hr battery indicated that the battery's ability to deliver the rated capacity is impaired after a stand life in excess of 30 days.

Extra tests demonstrated the load sharing characteristics of the 45 amp/hr and the 400 amp/hr batteries when connected in parallel. Results indicated a combined capacity of the two batteries which ranged from 500 amp/hr to 509 amp/hr. Until it was practically expended, the 45 amp/hr battery contributed almost the same portion of the total load current as the larger battery. This test effectively demonstrated the compatibility of the two batteries.

Power Distribution and Management

Power is fed from the primary power sources to the main, squib, and control bus systems for Spacecraft 3 through 12. The systems used on Spacecraft 1 and 2 were unique, and therefore are discussed in separate paragraphs.

Main Bus System. - Power from the main bus is fed through circuit breakers or fuses to the using equipment. Astronauts control onboard equipment powered from the main bus based on an evaluation of instrument panel display or at the direction of mission control.

Squib/Control Bus System. - The squib/control bus system consists of three diode-isolated 15 amp/hr silver zinc batteries, which provide power to two isolated and redundant squib buses and to the control bus through the series diodes. As an emergency measure, "bus tie" switches on the instrument panel permit application of power to the squib and control buses from the main bus. Circuit protection for squib/control bus loads is provided by fusistors for pyrotechnic firing circuits and by circuit breakers for control circuits. Power management for this system is similar to that for the main bus system.

Spacecraft 1 Power Distribution and Management. - All power for Spacecraft 1 systems was supplied by one 45 amp/hr silver zinc battery through circuit breakers. Since all equipment was powered continuously from lift-off, no in-flight switching was required. However, power control was provided via umbilical to the blockhouse for prelaunch activities, and by an ON-OFF switch for earlier ground operations.

Spacecraft 2 Power Distribution and Management. - In addition to the main and squib/control bus systems noted above, two sequencer buses with circuit breaker overload protection were provided on Spacecraft 2 to power the flight sequencers. During flight, equipment was powered either by the preset sequencers or by ground command via the digital command system.

Problems Encountered and Resolution.

A. Ammeter and Voltmeter Fluctuations - A review of films from cameras on Spacecraft 2 showed fluctuations of the ammeter and voltmeter needles after landing. The films also showed fluctuations of the attitude ball corresponding in frequency to those of the ammeter and voltmeter. It was concluded that the oscillations were induced by wave motion after splashdown. The magnitude of the motion overrode the mechanical damping capability of the measuring instruments. To verify the conclusion, a stock ammeter was tested and it was determined that movement parallel to the meter-sensitive axis resulted in fluctuations similar to those shown in the films. Since any recurrence would be recognized and would have no deleterious effect on mission accomplishment, no further action was taken.

B. Fuse Block Moisture Penetration - Several fuse blocks showed evidence of moisture penetration when opened for post-recovery inspection. An improved water proofing technique was used on all fuse blocks, starting with Spacecraft 4. In addition, the inertial guidance system fuses and primary O_2 heater fuses were moved downstream of the control switches for Spacecraft 7 and subsequent spacecraft. With this circuitry, employment of a power-down procedure shortly after impact would remove power from all fuse blocks and eliminate any adverse effects due to salt water penetration.

C. Circuit Breaker Opening - Post-flight inspection of Spacecraft 2 indicated that a number of circuit breakers were in the tripped position even though they had functioned properly during the flight. It was found that the circuit breakers in the re-entry control system had been tripped by salt water shorting at the thruster solenoid disconnects due to an undersize O-ring seal. All other tripped circuit breakers were inadvertently actuated by the recovery team during power-down. To eliminate salt water shorting after landing, astronauts were instructed to turn off all switches and circuit breakers associated with hot circuits. O-rings of the proper size were installed on the re-entry control system thruster solenoid valves, starting with Spacecraft 3.

D. Post-landing Indications of Main Bus Power Fluctuation - For Spacecraft 9 and 10, telemetered data indicated excessive variations in main bus power. Considerable post-mission test and effort were expended but the cause could not be determined. On Spacecraft 11, a determined effort was made to monitor main bus voltages and currents after landing, and to review the telemetered data. Here, there was no indication of abnormal operation.

A rereview of available information included static fire sea tests and instrumentation system immersion qualification tests. It was concluded that the improper indications were the result of salt water immersion of the instrumentation system. Control of the environment of each spacecraft prior to flight assured that missions would not be affected. Further effort became the concern of the instrumentation system rather than the power and the power distribution system.

E. Inadvertent Circuit Breaker Openings - On Spacecraft 9A, 10, 11 and 12, various circuit breakers were found open at times of greatest astronaut activity, such as EVA. In all cases, these circuit breakers were reset and remained closed during the rest of the mission.

Sequential Systems

There were no sequential system requirements for Spacecraft 1. For Spacecraft 2 through 12, the sequential systems were utilized to separate spacecraft modules, jettison aerodynamic fairings, ignite the retro-rockets, deploy parachutes, and perform numerous other functions. The sequential systems are discussed in the following paragraphs.

Boost, Insertion, and Abort Sequential System. - This system enables the astronauts to separate the spacecraft from the launch vehicle and to jettison the nose and horizon scanner fairings. In case of an abort during launch the system provides for shutdown of the booster engines and separation from the launch vehicle.

Retrograde Sequential System. - This system ignites the retro-rockets, separates the equipment adapter and retroadapter modules, and jettisons the horizon scanner heads. The astronauts initiate all retrograde sequential events except for the ignition of retro-rockets, which are ignited automatically by the TR signal from the time reference system with manual back-up by the astronauts at TR + 1 sec.

Landing and Post-landing Sequential System. - Astronauts initiate landing and post-landing sequential events based on their evaluation of instrument panel displays. The system deploys the drogue, pilot, and main parachutes, actuates the cabin air inlet door, separates the rendezvous and recovery section, jettisons the main parachute, and extends the recovery hoist loop and flashing light. On Spacecraft 2, the drogue stabilization chute was not installed.

Flight Sequencers. - Spacecraft 2, was adapted for unmanned flight by installing two redundant flight sequencers to initiate functions which would normally be performed by the astronauts. The sequential systems described above were not altered to accommodate the flight sequencers.

Docking Sequential System. - A docking sequential system was installed on Spacecraft 6 and 8 through 12 to incorporate rendezvous and docking functions. The docking system controls enable the astronauts to extend the index bar, to separate from the Agena vehicle in an emergency, to initiate the rigidizing and unrigidizing sequences, and to arm the Agena vehicle engine. The index bar is jettisoned and the doors are closed over the latch receptacle when the pilot jettisons the retroadapter. If the docking latches have not been jettisoned or the index bar has not been extended, these functions also are performed with retroadapter jettison. To accommodate the Gemini-Agena tether experiment, the docking system on Spacecraft 11 and 12 also jettisons the index bar and latches, and closes the doors over the latch receptacle independently of jettison of the retroadapter.

Problems and Resolutions in Sequential Systems. - Listed below are major problems encountered in sequential systems operation during flight or post-flight inspection. Action taken to preclude reoccurrence is also presented.

A. Retropackage Separation Sensor Failure - Instrumentation parameter AD04, retropackage separation, was not received on Spacecraft 2. Post-flight inspection revealed that retroseparation limit switches 2 and 3 had not been actuated. A failure analysis revealed excessive corrosion on one of the failed switch assemblies. As a result of this corrosion excessive external force was needed to operate the switch. On Spacecraft 3 the switches were deactivated and on Spacecraft 4 and up the switches were removed.

B. Loss of Pilot Chute Deploy Instrumentation Parameter - Telemetry indication parameter AE02, pilot chute deploy, was not received. An inspection of the toggle switch which senses the pilot chute deploy revealed that the lanyard had become untied. The length of lanyard recovered was sufficient to allow it to be tied to the switch bat handle with adequate slack remaining. Closer attention was paid to the tie procedure for subsequent spacecraft.

C. Thruster Activity After Impact - Spacecraft 2 recovery forces reported RCS thruster activity in the water. Post-flight inspection of the attitude control electronics package indicated that an aerospace ground equipment disconnect was not adequately waterproofed, thereby causing thruster firing due to salt water shorts. The following action was taken effective Spacecraft 3 and up to prevent reoccurrence of the problem:

1. The AGE disconnect was waterproofed.
2. Astronauts were instructed to open all RCS thruster circuit breakers after impact.
3. Motor operated shutoff valves were incorporated in the fuel and oxidizer lines to be closed prior to impact.

D. Loss of Manual Fire Retroinstrumentation Parameter - Telemetry indication of parameter AD06, manual fire retro, was not received on Spacecraft 4. Investigation revealed that relay K4-37, manual retro latch relay, was not latched in. The manual retrofire circuit and associated items were thoroughly checked and found satisfactory. The anomaly could be duplicated only by not pushing the switch through the operate point. No further action was taken.

E. Loss of Equipment Adapter Separate Instrumentation Parameter - Telemetry indication of parameter AD02, equipment adapter separate, was not received on Spacecraft 4. Investigation revealed that relays K4-66, and K4-67, abort discrete latch relays, which are energized by the equipment adapter separation sensor switches, were not latched in. These relays, plus all wiring associated with the equipment adapter separate sensing circuitry that was recovered, were thoroughly checked and no discrepancies were found. Since all recovered circuitry checked out, the most likely cause of failure was the separation sensor switches. On Spacecraft 5 and up, the only function of the equipment adapter separate sensing switches is to illuminate the SEP ADAPT telelight green and to provide telemetry indication of separation. The telelight switch is used in place of the separation sensor switches to provide the abort discrete to the computer.

F. Optical Sight Malfunction - On Spacecraft 5, the reticle light on the optical sight did not illuminate. The failure was traced to an open wire in the utility cord that provided power to the optical sight reticle light. The cord failed because of excessive strain that was applied when the cord was extended. A design change added a strain relief clamp at the connector back shell, insuring sufficient wire slack.

G. Docking Pyro Igniter Failure - Post-flight analysis of Spacecraft 6 revealed that the latch release, latch door release, and index bar jettison igniters powered from squib bus No. 1 had not fired. All pyro devices functioned normally due to ignition of the redundant squib bus No. 2 igniters. All unfired igniters are normally fired at retropackage jettison if the nose fairing has been previously jettisoned. The firing signal is interlocked by a latch relay which is latched in when the astronaut depresses the jettison fairing switch. It was determined that relay K3-86, nose fairing jettison latch relay, had not been latched in, thereby locking out the squib bus No. 1 firing signal. The failure was isolated to the nose fairing jettison switch, part No. 52-79719-7, during component tests on the switch. An operational test at reduced pressures revealed that one pole of the switch was inoperative due to canting of the switch actuator plate. For Spacecraft 8, all switches were replaced with units which had been subjected to a 48 hr soak at 10^{-5} psi and a 48 hr soak at five psi with a functional test conducted each hr. On Spacecraft 9 and up, all switches were replaced with units which had been modified to permit an additional 0.015 in. of plunger travel. In addition, all switches are subjected to a 48 hr soak at 10^{-5} psi, with a functional test conducted each hour during exposure.

H. Amber IND RETRO ATT Telelight Anomaly - On Spacecraft 8, the astronauts reported that the IND RETRO ATT telelight failed to illuminate amber at TR-256 sec. When the telelight switch was depressed, the green light was illuminated as expected. The circuitry is such that if the TR-256 sec signal from the time reference system is not received, neither the green nor the amber light will illuminate. Therefore, the failure was restricted to the light assembly or to the wire connected to the amber telelight. All circuitry and the telelight switch were examined after flight and no discrepancies were found. No further action was taken.

I. OAMS Yaw Thruster No. 8 Anomaly - This thruster intermittently fired without command for 17.5 min while Spacecraft 8 and the Agena were docked and undocked. The exact cause of failure was never determined and is considered to be complex. To prevent a recurrence, a switch was installed on the astronaut's control panel in Spacecraft 9 through 12 to cut off electrical power to all 16 attitude and maneuver thrusters.

J. Loss of Automatic Control of Hydrogen Tank Heater - At approximately 75 hr of Spacecraft 12's mission, the H_2 tank pressure could not be controlled with the H_2 heater switch in AUTO position. The H_2 heater switch was placed to OFF and Cryo O_2 and H_2 HTR circuit breaker OPEN. Post-flight analysis revealed no discrepancies in the recovered portion of the circuit and indicated that the most likely cause of the malfunction was a failed H_2 tank pressure switch.

K. Summary - There were no major problems associated with the overall program in the power distribution and management or sequential systems. Hardware, design, manufacturing and operational failure were minimal and random in nature. Cause of several of the failures could not be determined, either because the circuits functioned properly on post-flight inspection or because the failed component was not recovered.

COMMUNICATION AND TRACKING SYSTEM

The communication and tracking system provides two-way voice communication, ground-to-spacecraft command link, spacecraft-to-ground telemetry transmission, radar tracking signals and recovery aids. Subsystems consist of telemetry, tracking, voice communications, digital command, antennas, and recovery aids. (See Fig. 11 and 12.) These subsystems are discussed in Voice Communications, page 73 through Major Problems Encountered with Delivered DTS Tape Recorders, page 109 of this report.

Voice Communications

The voice communications subsystems include the voice control center, two UHF voice transmitters-receivers (transceivers) and one HF voice transceiver. The UHF transceivers have a transmitter output of three watts each. The HF transmitter output is five watts.

The voice communication system provides communication between the astronauts, between the blockhouse and the spacecraft during launch, between ground stations and the spacecraft from launch through re-entry, and between the astronauts and the frogmen during the water recovery. The voice subsystem also supplies communication between the spacecraft and recovery forces during landing and post-landing.

The voice control center (VCC), located on the spacecraft center instrument panel, is the central control and distribution point for the voice communication system. The VCC provides for intercommunication between the astronauts, for control and distribution of audio to and from the transceivers, intercommunication with the ground complex prior to launch, and a tone for direction finding.

The voice communication system is designed to furnish independent or "time-shared" capability in either UHF or HF voice links to the astronauts. Each astronaut has redundant microphone and headset amplifiers, mode switches and volume controls. Controls common to both astronauts are the HF and UHF select switches, voice tape recorder control switch, squelch controls, keying mode switch, and a silence switch for use during sleep periods.

Electro-Voice noise cancelling microphones are supplied to NASA for use in the helments. Lightweight Electro-Voice headset assemblies are also provided for use when the helmets are off.

The redundant UHF voice radios provide increased reliability. Only one HF radio is employed, however, because of its limited use. The UHF and HF

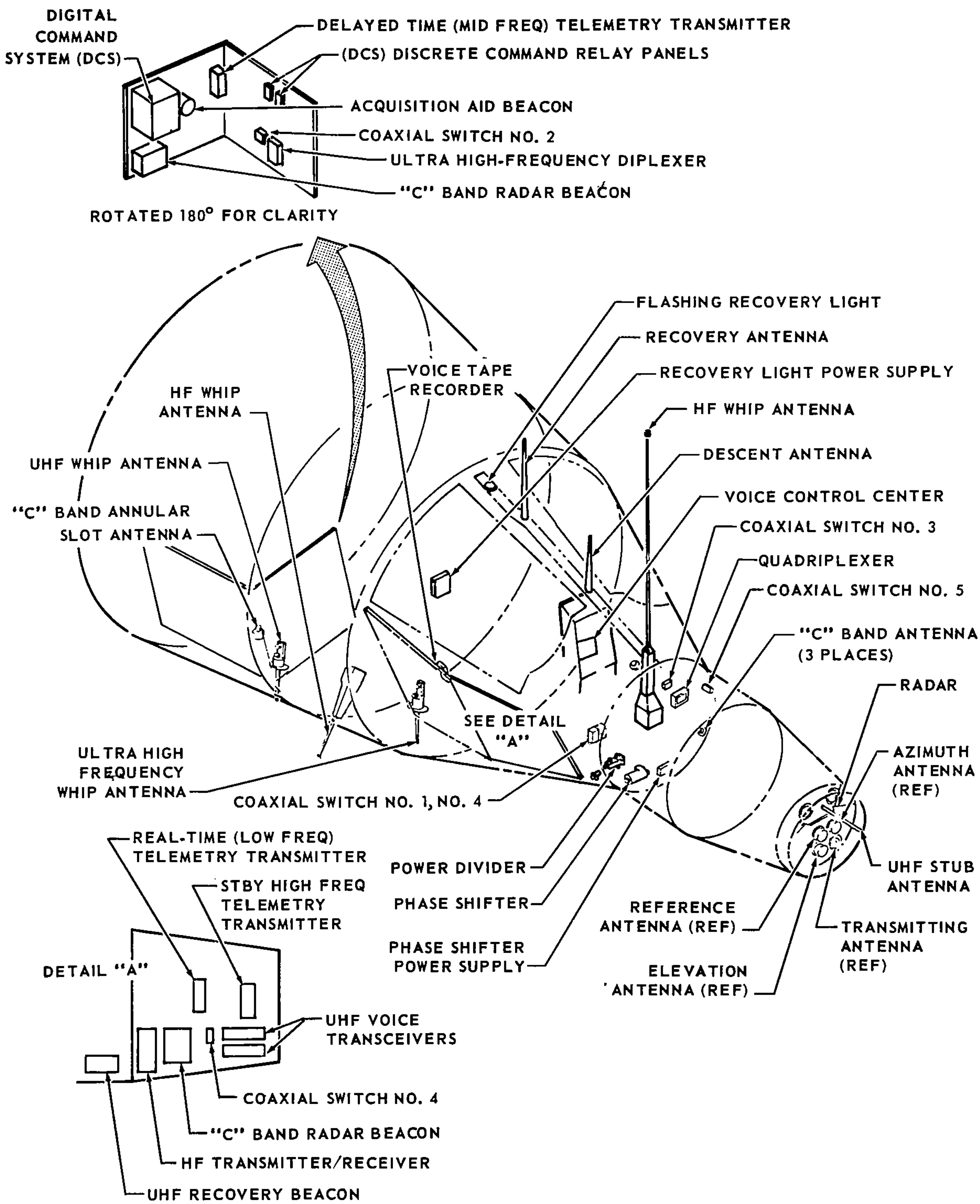


FIGURE 11 COMMUNICATION SYSTEM

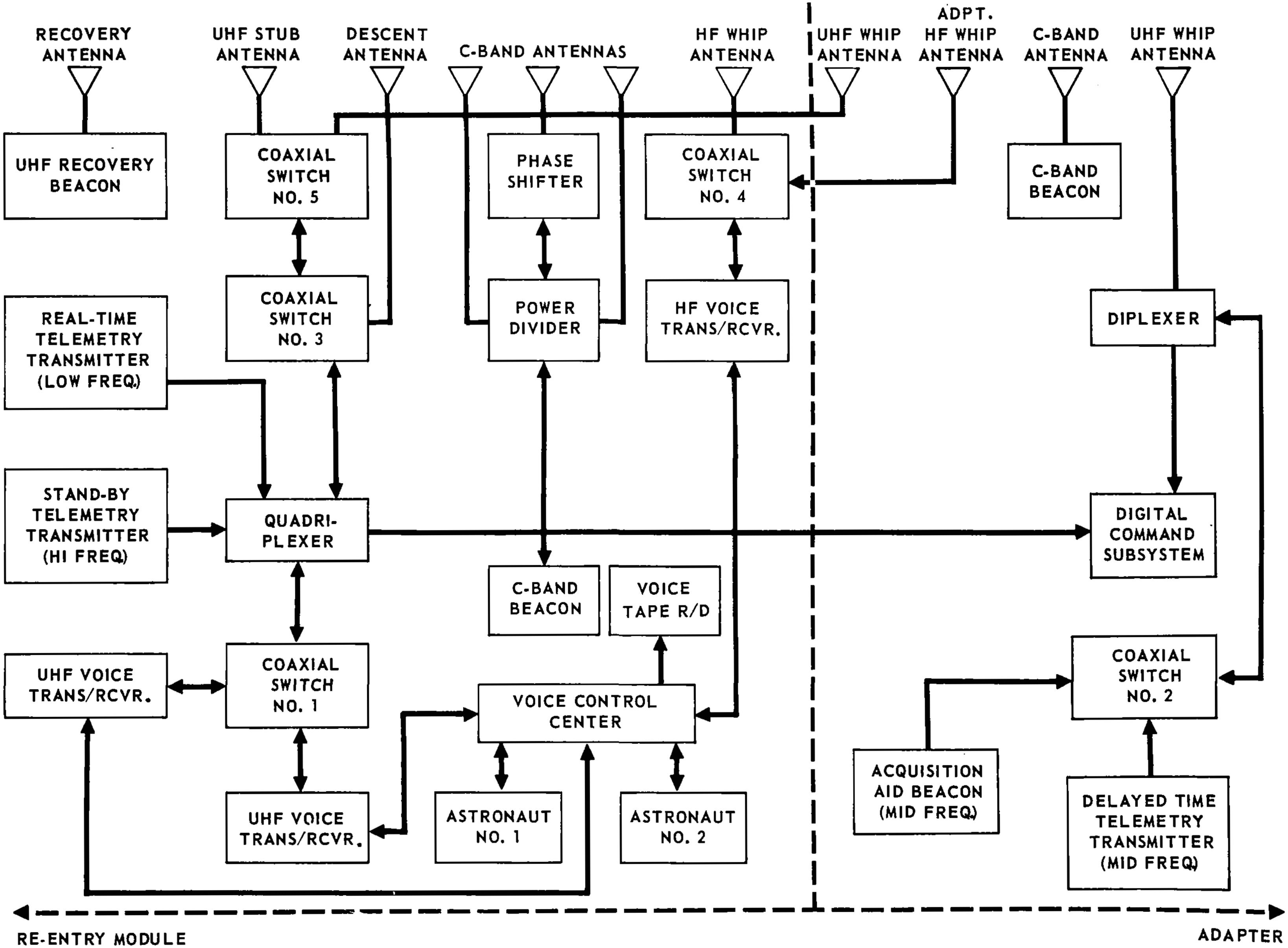


FIGURE 12 GEMINI COMMUNICATIONS SYSTEM BLOCK DIAGRAM

systems are complementary in that line-of-sight communications will utilize UHF and HF will be used for over-the-horizon spacecraft-to-ground communication. These units are isolated electrically to minimize failure modes.

Subsystems and Spacecraft Evaluation.

A. Subsystem Evaluation - All the components used in the voice system were subjected to extensive testing prior to installation in the spacecraft. The vendor performed a predelivery acceptance test on each unit. This consisted of a low-temperature/high-temperature test, a vibration test and a complete electrical performance test. After delivery to McDonnell, the unit was again subjected to a complete electrical performance test before it was installed in the Gemini Spacecraft.

B. Spacecraft Evaluation - After installation in the spacecraft, voice communications operation was evaluated at the system level. Time critical parameters such as RF power, center frequency, audio distortion and receiver sensitivity were monitored throughout the spacecraft testing at both St. Louis and at Cape Kennedy for possible equipment replacement.

Design, Development and Qualification Program.

A. Design Program - The voice subsystem components (HF and UHF transceivers and the VCC) were designed and built by Collins Radio Company to McDonnell's specifications. System design was based on the design utilized for Project Mercury. Problem areas encountered during the Mercury program were reviewed and corrective action was taken to avoid a recurrence on Gemini. An extensive development test program was initiated during the early design-phase to uncover possible problem areas requiring redesign.

B. Development Program - During the design phase, the individual voice system components were subjected to development tests in those environments which were considered critical. A summary of the environments used to qualify the units is attached as Table 3. In addition to the component tests, extensive development tests have been conducted at the system level. These programs are summarized in Table 4. A brief description of the tests performed follows:

1. Electronics Systems Test Unit (ESTU) - The ESTU was a laboratory-fabricated spacecraft mock-up. The objective of this test was to assemble all electronics systems and determine if any interface problems existed. Various systems were operated simultaneously to determine the amount of interference among them.

This testing revealed several problems. Among these were inverter noise coupling into the audio lines, feedback between the cabin speaker and the astronaut's helmet, and incorrect audio levels in the VCC. Corrective action consisted of installing double-shielded wiring for the audio; ultimate elimination of the cabin speaker, and vendor testing and modification of the VCC.

2. Compatibility Test Unit (CTU) - The CTU was a more sophisticated spacecraft mock-up used to gain further information on systems performance. This mock-up employed standard spacecraft wire bundles and had a structure similar to a production spacecraft.

The problem areas discovered during ESTU testing were further investigated during the CTU tests. Test methods and procedures to be used during production spacecraft system testing (SST) were finalized.

3. RF Compatibility Test - The RF compatibility test was performed on a screen wire mock-up equipped with all the spacecraft antennas and the spacecraft and Agena RF components. The RF sources were operated to determine if any RF interference existed between the systems.

The test results indicated that RF interference existed on the UHF receiver when the spacecraft and the Agena telemetry transmitters were all operating. This condition would never exist during a normal mission, since only two of the three telemetry transmitters are operable at the same time during flight. For this reason, the RF interference was not considered a system incompatibility.

4. Tracking Station Fly-Over Tests - The fly-over program, in which the voice communication system was installed in an aircraft, was conducted to verify station operation and check the compatibility of the spacecraft equipment with the ground stations' equipment. No problems were encountered with voice system operation during these tests.

C. Qualification Program - All of the voice system components successfully completed the qualification program. A detailed summary of the qualification program is contained in Table 3.

Reliability and Quality Assurance Program. - In order to maximize the reliability of the voice system components, the vendor incorporated all of the standard reliability procedures into the program. These consisted of vendor surveillance, qualification testing, parts screening, reliability estimates, design review, and failure analysis and reporting.

The reliability assurance test consisted of two parts: (1) voltage and temperature overstress, and (2) vibration overstress. Performance degradation caused by the overstress conditions was noted on several parameters, but it was determined that the degradation was not significant enough to jeopardize successful voice communications.

Flight Results. - The only flight failure of a voice system component was an HF voice transceiver anomaly on Spacecraft 2. The HF unit failed to operate properly after splashdown. It was removed from the spacecraft and returned to the vendor for failure analysis; subsequent investigation revealed that the failure was caused by a shorted diode. Tests indicated that at turn on, the peak-inverse voltage (PIV) of the diode was exceeded, causing the failure. This problem was corrected on all remaining HF radios by substituting a diode with a higher PIV. No further anomalies were encountered.

TABLE 3 QUALIFICATIONS TESTS
COMMUNICATION AND TRACKING

ENVIRONMENT	ACQUISITION AID BEACON	C-BAND BEACON	S-BAND BEACON	C-BAND ANNULAR SLOT	S-BAND ANNULAR SLOT	RF COAX SWITCH	C-BAND RE-ENTRY ANTENNA SYS.			QUADRIPLEXER	DIPLEXER	HF WHIP ANTENNA	UHF WHIP ANTENNA	UHF DESCENT ANTENNA
							ANTENNA	POWER DIVIDER	PHASE SHIFTER					
HIGH TEMP	A	A	A	C & DOC-1	C	C	C	A	C	A	A	A	A	C
LOW TEMP	A	A	A	A	DOC-2	A	A	A	A	A	A	A	C	
VIBRATION	A	C	C	A & DOC-1	A	A	A	A	A	A	A	A	A	A
SHOCK	A	C	C			A	A	A	A	A	DOC-5	A		A
TEMP ALT.	A	A	A				DOC-3	DOC-3	DOC-3			C		
ACCEL.							DOC-3	DOC-3	DOC-3				A	
RFI	C	B	B									A		
HUMIDITY	A	A	A				C	A	A	DOC	DOC	A	A	
ACOUSTIC NOISE		A	A							DOC-4	A			
EXPLOSION	A	A	A							DOC	DOC	A		
HIGH PRESS.						A								
RAIN				DOC	DOC		DOC-3	DOC-3	DOC-3	DOC	DOC	A	A	
0_2 ATMOS.														
SAND & DUST				DOC	DOC		DOC-3	DOC-3	DOC-3	DOC	DOC	A	A	
FUNGUS				DOC	DOC	DOC	DOC-3	DOC-3	DOC-3	DOC	DOC	DOC	DOC	
SALT SPRAY		A	A	A	DOC-2		A	C	C	DOC	DOC	A	A	A
STRUCTURAL												A		
IMMERSION						A				A	A			
DECOMPRESSION														
POWER HANDLING VS. ALTITUDE				C	C									A
ENDURANCE														
SALT FOG														

A – SUCCESSFULLY COMPLETED WITHOUT RERUN
B – FAILED INITIAL TEST BUT PASSED RERUN WITHOUT EQUIP. MODIFICATION
C – EQUIP. HAD TO BE MODIFIED TO PASS TEST
D – TEST REQ. HAD TO BE WAIVED OR LOWERED
F – TEST REQ. BUT HAS NOT BEEN PERFORMED
G – RETEST REQ. DUE TO UNRELATED EQUIP. MODIFICATIONS

Major Problem Associated with Program. - During the development program on the UHF radios, a problem was encountered with the 8185 power amplifier tubes. The tubes exhibited "cathode slump" and fatigue of the mica spacers. The result was a drop in RF power. The corrective action employed was to incorporate an improved type tube with a modified spacer and different (Class C) "burn-in" procedures.

During installation of the evacuation tube on the HF and UHF radios, solder balls were inadvertently introduced into the case. This problem was corrected by employing a new case design and sealing process.

Antennas

Spacecraft and Subsystems Evaluation. - The antenna subsystem provides transmission/reception capability for the communication system. Radiation

TABLE 3 QUALIFICATIONS TESTS (Continued)
COMMUNICATION AND TRACKING

UHF RECOVERY ANT.	UHF NOSE STUB ANT.	DIGITAL CMD SYS.	EVENT TIMER	ELECTRONIC TIMER	GMT CLOCK	UHF TR	HF T/R	VOICE CONTROL CENTER	RECOVERY BEACON	FLASHING LIGHT	MICROPHONE	HEAD SET ASSY.	C-BAND BCN (ADPT.)
DOC-6	A	A	A	A		A	A	A	A	A	A	A	QUALIFIED FOR ANOTHER PROGRAM
	A	A	A	A		A	C	A	A	A	A	A	
DOC-6	B	C	C	C	D	C	B	A	C	B	A	A	
DOC-6		A	A	A	A	A	A	A	A	A	A	A	
		A	A	A	F	A	B		A	C			
		A	A	A	A								
		DESIGN GOAL	INFO ONLY	DESIGN GOAL		A	A	A	D				
		A	C	A	C			A			A	A	
		A				A	A		A		A		
												A	
								A		A			
								A					
			A	C	A			A		A	A	A	
		DOC	DOC	DOC				A					
		DOC	DOC	DOC		DOC	DOC	DOC					
DOC-6	A							A				A	
						A	A		C	C			
								C			A		
DOC-6	A												
		A	A	A	F								
		A	A	A	C			C					

DOC – DOCUMENTED
DOC–1 – SIMILARITY TO S-BAND ANTI. (REQUAL)
DOC–2 – SIMILARITY TO C-BAND ANT.
DOC–3 – SIMILARITY TO MERCURY
DOC–4 – SIMILARITY TO DIPLEXER
DOC–5 – SIMILARITY TO QUADRIPLEXER
DOC–6 – SIMILARITY TO DESCENT ANTENNA

coverage requirements vary with the mission phase depending upon spacecraft stabilization mode and ground coverage requirements. The antenna system provides roll symmetrical radiation coverage for C-band tracking and UHF voice, telemetry and command during the launch phase. During stabilized orbit attitude, the antenna system provides yaw symmetrical horizon-oriented radiation coverage. HF voice and DF capability is provided by an HF whip antenna on the adapter for orbit use and on the re-entry module for post-landing use. For drifting flight with uncontrolled spacecraft attitude, the antenna system provides complementary yaw symmetrical and roll symmetrical coverage. The astronauts select the antenna system usage to obtain the optimum coverage for voice, telemetry and tracking. During the re-entry phase, C-band and UHF coverage is utilized and is similar to the launch phase coverage. The recovery phase requires antenna capability for HF and UHF frequencies. The antenna systems are comprised of high efficiency, high reliability antennas and RF components which are of minimum weight and volume. The C-band tracking antenna system consists of three cavity helix antennas, a power divider, a

TABLE 4 DEVELOPMENT TESTS
COMMUNICATIONS - RADAR - TIME REFERENCE SYSTEM

COMMUNICATIONS
1. INTEGRATED SYSTEM TESTS - ELECTRONIC SYSTEM TEST UNIT (ESTU), COMPATIBILITY TEST UNIT (CTU).
2. RFI COMPATIBILITY TEST - LOCKHEED TEST, TOWER TEST.
3. RECOVERY TEST - BOILER PLATE NO. 3.
4. FLY-OVER RANGE TESTS -
5. RELIABILITY ASSURANCE TESTS (OVER-STRESS).
6. GEMINI/AGENA PLAN X TEST
RENDEZVOUS RADAR
1. INTEGRATED SYSTEM TESTS - ELECTRONIC SYSTEM TEST UNIT.
2. RFI COMPATIBILITY TESTS - LOCKHEED TEST (IN PROGRESS), TOWER TEST.
3. WHITE SANDS FLIGHT TEST -
4. NEW MEXICO STATE ACCURACY AND ANTENNA VERIFICATION.
5. GEMINI/AGENA PLAN X TEST.
TIME REFERENCE SYSTEM
1. INTEGRATED SYSTEM TESTS - ESTU, CTU.
2. FLY-OVER RANGE TESTS -
3. INTERFACE TESTS - COMPUTER, DCS AND DTS.
4. GEMINI/AGENA PLAN X TEST.

phase shifter and associated power supply, and an annular slot type antenna. The UHF antenna system consists of the nose stub, forward adapter whip, aft adapter whip, descent and the recovery antennas and four RF coaxial switches, a quadriplexer and a diplexer. The HF antenna system consists of two HF whip antennas and an RF coaxial switch.

State-of-the-Art Advances. - The UHF nose stub antenna is located on the radar ground plane on the nose of the spacecraft and may be exposed to impact during spacecraft rendezvous and to impact by a parachute cable during single point release. The antenna is, therefore, designed to be self-erecting after having been impacted and flexed in any direction and in addition to survive re-entry heating and air loads. The upper section of the antenna mast pivots about a mating socket when impacted. A compression spring is preloaded so that when the antenna is flexed the action of the spring and a cable assembly will exert a force to restore the mast to its normal in-line position.

The UHF adapter whip antenna is a quarter-wave stub, which is unique in that it is self-extendible. It consists of a thin strip of specially heat treated beryllium copper which is furled into a cylindrical housing under tension. There is sufficient stored energy to eject a retaining cap and latching post assembly and to automatically extend itself to form a tubular quarter-wave stub antenna matched to a 50 ohm feed system. The cap and latching post assembly is electrically released when a solenoid actuated latching mechanism is energized. The element requires manual retraction by proper furling into the housing.

Design, Development Ground and Qualification Program Results. - The criteria for the design and development of the antennas and RF components was based on operational requirements in predicted critical environments. Modifications were made as the test program revealed design deficiencies and resulted in pertinent components qualified to the environments. Table 3 shows the qualification test program for each antenna and RF component. The comprehensive test program and quality assurance surveillance resulted in high reliability components.

Results of Flight Mission. - There have been no problems relative to the UHF and C-band antenna systems during the launch, orbit, re-entry and recovery phases of the flight missions. The use of HF antennas for experiments during the orbital phase of the missions has been successful. The re-entry HF whip antenna encountered problems during the post-landing phase of the early missions; however, modifications sealed the unit and corrected the problem for later missions.

Discussion of Major Problems Associated with Overall Program Problem. - Potential damage to the UHF nose stub antenna if impacted during either rendezvous or by parachute cable at single point release during re-entry due to its location on the radar ground plane.

Corrective Action - Redesign to an antenna with a five in. long section on the end that can be flexed in any direction if impacted. A socket assembly with a cable and spring under tension inside the antenna body returns the antenna section to its normal inline position after having been flexed.

Problem - The HF whip antenna (located on the re-entry module) failed to radiate the RF signal during the post-landing phase because sea water leaked in shorting out the radiating element and antenna RF feed system.

Corrective Action - The antenna assembly case was sealed and a special pressure equalizing breather cartridge was attached. The tip plug which contains the ablative section for closing the antenna element egress hole in the spacecraft skin was modified so that it could seal against the muzzle end of the antenna assembly in its fully retracted position. In addition, a lexan polycarbonate (plastic) guide block replaced the metal one at the muzzle end to prevent grounding and cellulose sponge sections were installed in the muzzle and the silicone rubber splash boot to absorb sea water resulting from spray.

Problem - The UHF adapter whip antenna element failed to extend due to improper furling when manually retracted after ground tests.

Corrective Action - The metal cap which retains the furled antenna element was replaced with a clear lexan polycarbonate (high temperature plastic) cap so that proper furling of the antenna element into the housing can be observed through the cap prior to latching. Proper furling allows the antenna element to automatically extend at spacecraft separation when the cap latching mechanism is electrically released.

Electronic Recovery Aids

The electronic recovery aids consisted of one UHF recovery beacon, 52-85719 and one flashing recovery light, 52-85720. The UHF recovery beacon provided a homing signal for the recovery forces to locate the spacecraft after re-entry and the flashing recovery light provided a visual aid to pin-point the location of the spacecraft in the event of recovery after dark. Both the RF beacon and the light (if required) are turned on at approximately 10,000 ft.

Subsystem and Spacecraft Evaluation. - Prior to installation in the spacecraft each subsystem was subjected to temperature, vibration and a complete electrical test at the vendor's facilities and to a complete electrical test at McDonnell.

After installation in the spacecraft, the critical parameters were monitored throughout spacecraft testing to assure acceptable operation.

Design, Development and Qualification Program.

A. Design Program - ACR Electronics Corporation was the vendor for both the UHF recovery beacon and the flashing recovery light.

The recovery light was almost identical to the one used on Project Mercury and the design changes were minimal.

The UHF recovery beacon was a hybrid utilizing transistor regulator, DC-DC converter, and pulser with a vacuum tube transmitter.

B. Development Program - During the design phase each subsystem was evaluated and tested in all critical environment which included temperature, altitude-immersion, and vibration. These tests uncovered problem areas which are described in Problems Associated with Program, page 83. In the integrated system tests summarized in Table 4, both the recovery beacon and the flashing light caused excessive interference to both radio and audio frequencies. As a result, an RF bandpass filter was added to the output of the recovery beacon and shielded wires were used in the spacecraft to carry the flashing light signals.

A fly-over test was run using the UHF recovery beacon to verify its range capability using both beacon type receiver and the standard ARA-25 direction finders.

C. Qualification Program - Each subsystem was subjected to qualification tests as noted in Table 3. All tests were successfully completed with the exception of RFI on the recovery beacon. This requirement was waived and as noted in B. Development Program, page 82, an external bandpass filter was added to the beacon output.

Flight Results. - There have been no flight failures of any electronic recovery aid subsystem.

Problems Associated with Program.

A. Recovery Beacon - In development testing, the 6939 output tube failed under vibration. The corrective action was to strengthen the internal structure of the tube and the mounting structure supporting the tube.

During qualification testing, the beacon failed to pass immersion tests after being subjected to vibration. The corrective action was a complete redesign of the case incorporating an O-ring seal instead of epoxy.

After qualification testing, a catastrophic failure mode of the regulator was discovered. The failure caused the regulator to go out of regulation when the input power was turned off and back on very quickly. The failure was traced to an RC time constant in the regulator which became marginal under complex variations in temperature, warm-up time and the elapsed time between turnoff and turnon.

For Spacecraft 2, a temporary fix was made by adding a capacitor to the RC circuit to increase the time constant. This fix provided reliable operation in all except extreme combinations of the critical criteria. For Spacecraft 3 and up, the regulator was redesigned to be completely fail-safe.

B. Flashing Recovery Light - During the altitude immersion part of qualification testing, the lamp failed to operate during the immersion environment. Failure analysis revealed an air bubble seepage path through the epoxy inside the lamp which allowed the water to form an electrical path between the case and a high voltage terminal. The corrective action was to modify the potting procedure which included performing the potting in steps and subjecting the epoxy to a vacuum prior to curing.

Tracking

The tracking system consisted of one C-band beacon, 52-85707, one S-band beacon and one acquisition aid beacon, 52-85706.

Effective Spacecraft 4 and up the S-band beacon was replaced by a second C-band beacon. (A Motorola DPN/66.) The acquisition aid beacon provided an RF link from the spacecraft to the ground tracking station to facilitate spacecraft acquisition.

The C and S-band radar beacons provided very accurate tracking data.

Subsystem and Spacecraft Evaluation. - Prior to installation in the spacecraft, each subsystem was subjected to temperature, vibration and complete electrical test at the vendor and a complete electrical test at McDonnell. Additionally, the C and S radar beacons were leak tested at the vendor.

After installation in the spacecraft, the critical parameters were monitored throughout spacecraft testing to assure acceptable operation.

Design, Development and Qualification Program.

A. Design Program - In the development of the acquisition aid beacon it was decided to utilize a standard TM transmitter to provide the low level CW output required. In this application the modulation capability was not used.

The re-entry C-band beacon is basically an improved and repackaged version of the radar beacon used on the Mercury flights.

The S-band beacon is very nearly identical to the C-band beacon with the exception of necessary frequency dependent differences. The operation capabilities, construction, development and test of the S-band beacon are identical to the C-band beacon. In fact, all the frequency independent subassemblies were interchangeable between the two units. Therefore, the subsequent discussion of the C and S-band beacons will be handled as a single item except where specified differences are noted.

The adapter C-band beacon (which replaced the S-band beacon Spacecraft 4 and up) was procured as an "off the shelf" item. It was a slightly modified version of the AN/DPN-66 manufactured by Motorola.

B. Development Program - The acquisition aid beacon, the re-entry C-band beacon, and the S-band beacon were a part of the system tests shown in Table 4. There were no changes required as a result of the system integration tests.

The adapter C-band beacon (AN/DPN-66) was not being utilized at the time the system tests of Table 4 were being made. Of course, it was given a system integration test as a part of an actual spacecraft. There were no system compatibility of interference problems with the AN/DPN-66.

C. Qualification Program - Each unit was subjected to the qualification tests as noted in Table 3. All tests were successfully completed although some equipment modifications were required. A discussion of problems encountered in qualification testing is included in Problems Associated with Program, on the following page.

Flight Results. - There have been no flight failures of any radar beacon or acquisition aid beacon.

Problems Associated with Program.

A. Acquisition Aid Beacon - The acquisition aid beacon failed to meet the RFI requirements during qualification testing. There were two problem areas; the more serious was excessive spurious output at VSWR and the other was the radiation and conduction of interference from the power lines. The spurious output was corrected by a redesign of the RF filter, and a line filter was added to the power leads to eliminate the radiated and conducted interference.

B. Re-entry C and S Radar Beacons - The MECA module packaging technique proved to be quite fragile under random vibration. As a result it was necessary to:

1. Rigidize the mother board by applying epoxy to the bottom and each of the side rails.
2. Install neoprene foam pads above and below the MECA modules.
3. Rigidize the beacon cover over the MECA modules by adding ribs across the cover.

The local oscillator tube in the re-entry C-band beacon developed leaks after unpredictable lengths of time. The result was catastrophic failure of the beacons. Corrective action was to change from the 6299 vacuum tube to a 7486 vacuum tube. Effectivity was Spacecraft 6 and up.

C. Adapter C-Band Beacon - Insofar as use of the AN/DPN-66 beacon on the Gemini program, there have been no serious problems. The DPN-66 was qualified and used on many other vehicles and has been proven to be a reliable beacon.

RENDEZVOUS RADAR SYSTEM (RRS)

System Description

The rendezvous radar system consists of the rendezvous radar range and range rate indicator, and command link encoder all installed on the Gemini Spacecraft. The units of the system installed in the target docking adapter of the Agena Vehicle are the transponder, boost regulator, dipole antenna and spiral antennas. Fig. 13 depicts the system block diagram. Fig. 14 shows the equipment installation on the Agena Vehicle.

The rendezvous radar system is a pulsed system that utilizes a transponder for return responses. The radar operates on a frequency of 1528 mc/sec with a pulse width of 1.0 μsec and a pulse repetition frequency (PRF) of 250 PPS. Upon interrogation by the radar, the transponder delays the return pulse for 2.0 μsec and transmits a 6.0 μsec reply pulse on a frequency of 1428 mc/sec. The internal delay in the transponder allows for operation to essentially zero range.

The radar utilizes the interferometer principle to determine target angle. This is accomplished by measuring the difference of phase of the return beacon

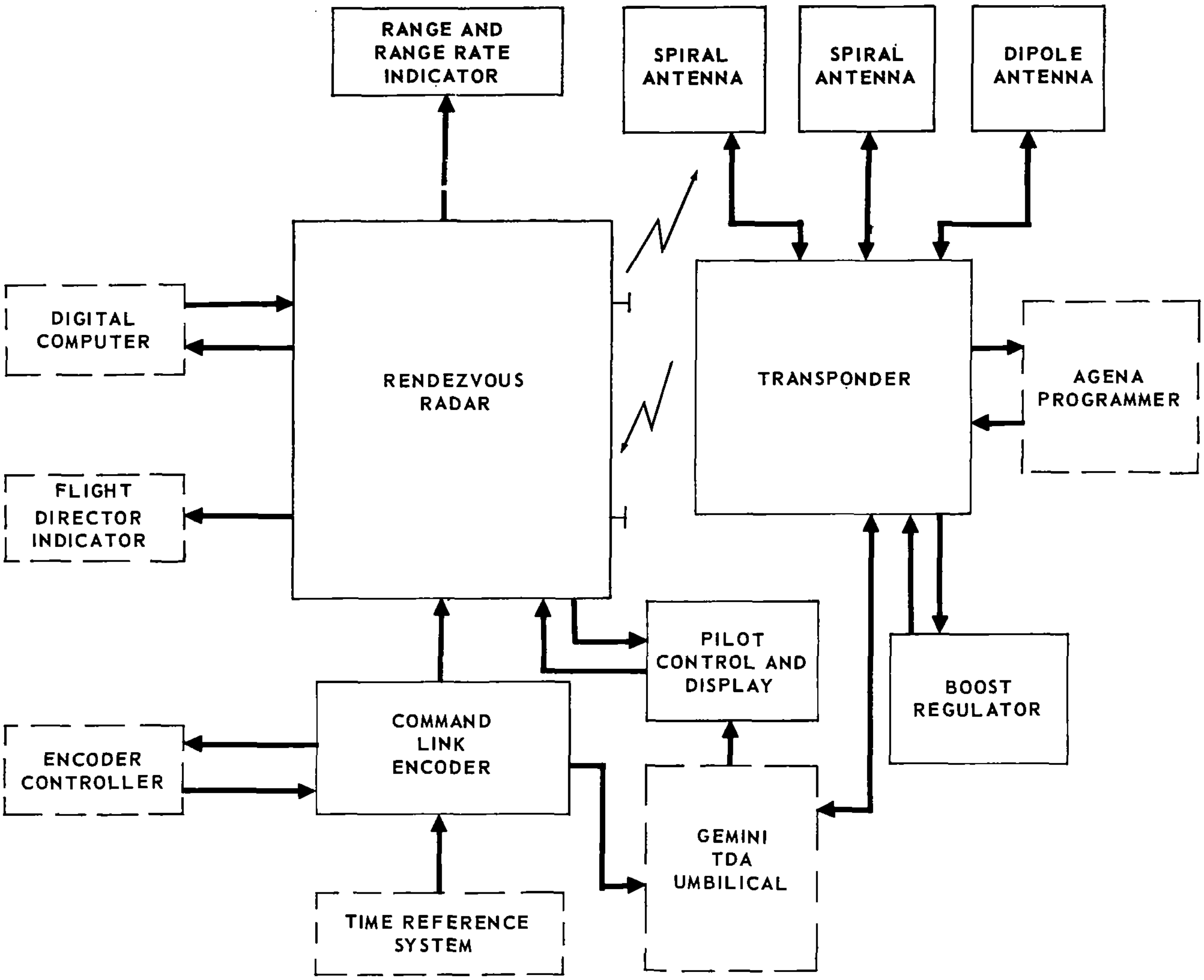


FIGURE 13 RENDEZVOUS RADAR SYSTEM

signal at two separate radar antennas. Three broad beam, circularly polarized spiral antennas are used for two interferometer pairs. (One antenna is common for both interferometers.) The radar and antennas are oriented in the spacecraft so that the antenna boresight axis is parallel to the spacecraft roll axis. One interferometer lies in the pitch plane and determines target elevation; the other interferometer is in the yaw plane and determines target azimuth. A fourth spiral antenna is utilized for transmitting.

The transponder receives and transmits as a result of the radar interrogation either through the two spiral antennas or the dipole antenna. The radar, upon receipt of the 6.0 μsec reply pulse from the transponder, extracts range and angle information from each pulse.

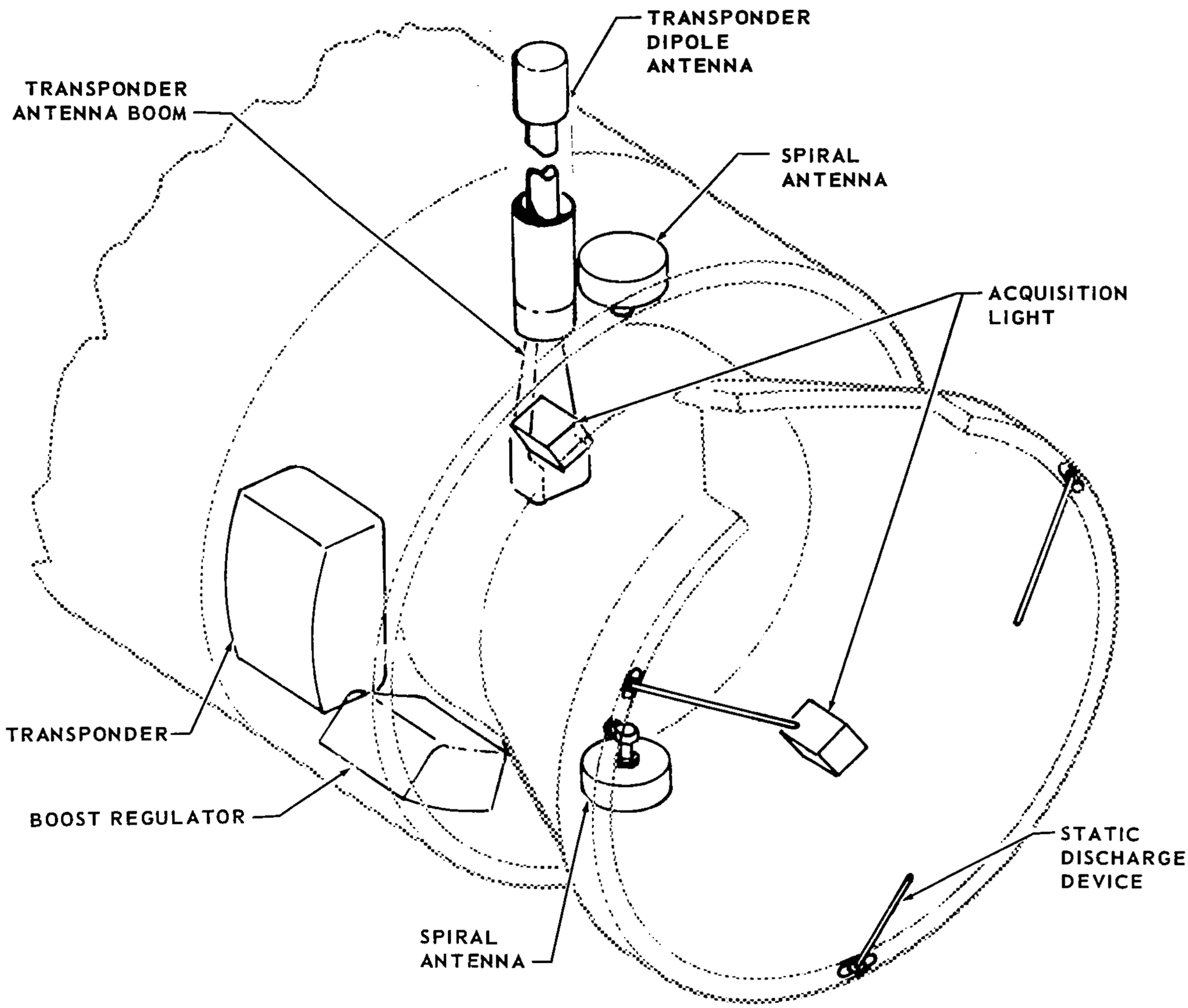


FIGURE 14 TDA ELECTRONICS SYSTEM

The range and range rate indicator displays range from zero to 50 nautical miles, and range rates from -100 to +500 fps. Digital range and angle information is supplied to the digital computer in serial binary form. The radar provides target azimuth and elevation information on the flight director indicator (FDI).

The rendezvous radar system includes the following command link capability. Prior to docking, the radar system provides for pilot command control of the Agena Vehicle and Agena systems by pulse position modulation of the radar transmitter. The receipt of a valid message is acknowledged by pulse width modulation of the transponder transmitter. The Gemini-TDA umbilical provides a means for command control over the Agena systems after docking.

In the radio frequency (RF) command mode of operation, the rendezvous radar is modulated at a PRF of 256 cps by the command link encoder through timing signals generated by the time reference system. To transmit a command, the crew selects one of 128 possible commands by positioning three switches on the encoder controller to correspond to the three digits of the command. The switch positions are read out to the encoder and the encoder codes the message into a digital format.

The digital message, with a 0 representing one time slot and a 1 another time slot, is transmitted to the transponder by a pulse train of 60 pulses.

The message is received by the transponder where an automatic phase lock loop is used to demodulate the command. The command message is then transmitted to the Agena programmer for decoding. When the Agena programmer recognizes a command, a signal is sent to the transponder that results in the pulse width modulation of the transponder reply and a receipt of message indication is provided the crew.

In the hardline mode of operation, after docking, the astronaut's commands are transferred to the Agena programmer via the Gemini-TDA umbilical. The receipt of message indication is also provided through the umbilical.

This Agena command capability was not used on Spacecraft 5, 6 and 9.

Systems Which Differed From Standard RRS. - The following systems differed from the standard rendezvous radar system described above.

A. Spacecraft 5 Radar System - The standard system was utilized but the Agena components were packaged in a rendezvous evaluation pod (REP). The REP was stowed in the adapter section of Spacecraft 5 and was ejected for practice rendezvous exercises.

B. Project 7/6 Radar System - The standard system was installed on Spacecraft 6 except that the transponder, boost regulator and one spiral antenna were installed in the rendezvous and recovery section of Spacecraft 7.

C. Spacecraft 9 Radar System - The standard system was utilized, but the Agena components were packaged in the augmented target docking adapter (ATDA). A transponder antenna switch control was added to the ATDA to facilitate radar tracking of the tumbling ATDA. The switch control anticipates loss of lock and switches the transponder antenna for maximum received signal. In addition, a simplified command link was added which provided the crew with one real time command.

Development Tests For RRS

Development tests for the RRS were performed by the vendor, by McDonnell, and by NASA.

RRS Vendor Tests.

A. Prequalification Test - The radar, transponder, range and range rate indicator, boost regulator, command link encoder and antennas were subjected to launch vibration and high and low temperature environments prior to formal qualification testing. Integration tests were made between the radar and range and range rate indicator, the radar and the command link encoder, the radar and transponder, and the transponder and boost regulator.

B. Modification of Antennas - A series of omnipatterns, mutual coupling, cross coupling, and phase linearity tests on the radar antenna resulted in the addition of Eccosorb (an RF absorbing material) to the radar ground plane to reduce the cross coupling between the azimuth and elevation angle channels. An extensive test program was required on the dipole antenna array to develop a design that could be manufactured with repeatability.

RRS McDonnell Tests.

A. Electronic System Test Unit (ESTU) - An engineering prototype radar was tested as part of the ESTU. During this test the engineering prototypes of the guidance and control system were interfaced in a simulated spacecraft. Interconnecting cables of the proper length and type were used to insure that the radar interfaces with the guidance and control system were compatible.

The radar command link interface was demonstrated by interconnecting the time reference system (TRS), command link encoder, encoder controller, and rendezvous radar in the ESTU mock-up with a transponder for both RF and hardline operation. The test included the transmission of RF commands, receipt of RF message acceptance pulses (MAP), transmission of time critical commands, docked mode operation (hardline), and evaluation of the TRS interface pulse shapes.

B. Radar Boresight and Angle Linearity - The boresight of the radar was measured on an outdoor antenna range at New Mexico State University to get a basis of comparison of radar boresight and angle linearity. This data was compared to data from the same radar in the Westinghouse anechoic chamber and the NASA radar antenna range on Merritt Island (KSC). The test also evaluated the effect of spacecraft configuration on the radar boresight, the reference radar radiation pattern, and the Agena vehicle antenna pattern and ellipticity.

NASA Tests.

A. Command Link Test at Lockheed Missile and Space Center (LMSC) - A TDA-Agena interface test at LMSC demonstrated the compatibility of the rendezvous radar transponder and the Agena programmer. The capability of the transponder to operate in the Agena RFI environment was insured. Also, the operation of the transponder was tested during the firing of the Agena main engine.

B. Gemini/Agena RF and Functional Compatibility and Mission Simulation (Plan-X) - Functional compatibility of the Gemini and Agena vehicles in the docked and near-docked positions was demonstrated during "Plan-X." This was the first test in which the complete radar command link (i.e., TRS, encoder controller, encoder, radar transponder, and Agena programmer) was used to command the Agena. All commands were transmitted successfully.

C. Rendezvous Radar Flight Test Program - The dynamic performance of the radar was demonstrated in a flight test at White Sands Missile Range, New Mexico. The transponder and spiral antenna were mounted on a T-33 aircraft and flown over the radar which was mounted in a ground test stand. By variation of aircraft altitude and speed, the range rate and angular tracking rate of the rendezvous radar was exercised.

Qualification Tests for RRS

Table 5 lists the various environments to which the units of the radar system were subjected.

Radar. - The radar boresight shifted during the humidity portion of the qualification tests. The shift was attributed to moisture being absorbed in the Eccosorb and entering the reference antenna assembly. Measures were taken to keep the Eccosorb dry prior to flight, and the reference radome was sealed.

Range and Range Rate Indicator. - All tests were passed successfully without retest or redesign.

Command Link Encoder. - All tests were passed successfully without retest or redesign.

Transponder. - Problems encountered during humidity testing were resolved by sealing the transponder to prevent moisture from getting into the casing. Extensive difficulty then was experienced in the low pressure environment. The transmitter cavity was pressurized to prevent electrical breakdown. In addition, all microwave components and connectors in the RF transmitting path were vented to prevent corona discharge. Also added was a plug which was removed prior to launch, allowing the transponder to outgas at an accelerated rate after launch.

Spiral Antenna. - Exposure of the spiral antenna to the low pressure environment required venting of the balun and connector to prevent corona discharge. All other tests were successfully completed without retest or redesign.

Dipole Antenna. - All tests were passed successfully without retest or redesign.

TABLE 5 RENDEZVOUS RADAR QUALIFICATION TEST

SYSTEM UNIT	ENVIRONMENTS																	
	HIGH TEMP	LOW TEMP	VIBRATION	SHOCK	TEMP/ALTITUDE	LOW PRESSURE	ACCELERATION	RFI	HUMIDITY	ACOUSTIC NOISE	EXPLOSION	RAIN	OXYGEN ATMOSPHERE	HIGH PRESSURE	DECOMPRESSION	SAND & DUST	FUNGUS	SALT SPRAY
RADAR	X	X	X	N/A	X	X	X	X	X	X	D	N/A	N/A	N/A	N/A	D	D	D
R & R INDICATOR	X	X	X	X	N/A	X	X	X	X	D	D	N/A	X	X	X	D	D	X
COMMAND LINK ENCODER	X	X	X	X	N/A	X	X	X	X	D	D	N/A	X	X	X	D	D	X
TRANSPONDER	X	X	X	N/A	N/A	X	X	X	X	X	X	D	N/A	N/A	N/A	D	D	X
SPIRAL ANTENNA	X	X	X	N/A	X	X	X	N/A	X	X	D	D	N/A	N/A	N/A	D	D	X
DIPOLE ANTENNA	X	X	X	D	D	X	X	N/A	X	X	D	D	N/A	N/A	N/A	D	D	X
BOOST REGULATOR	X	X	X	N/A	N/A	X	X	X	X	X	X	D	N/A	N/A	N/A	D	D	X

X – TEST PERFORMED
D – TEST DOCUMENTED
N/A – TEST NOT APPLICABLE

Boost Regulator. - Exposure to humidity required the addition of a conformal coating to the internal chassis and components. All other tests were completed successfully without retest or redesign.

Flight Mission Results Of The RRS

The rendezvous radar system has been implemented in the following configurations.

Spacecraft 5. - The system flown is described in Spacecraft 5 Radar System, page 88. Ejection of the rendezvous evaluation pod from the spacecraft was nominal and radar system operation and performance was nominal. The practice rendezvous exercise was terminated a short time after REP ejection due to a spacecraft fuel cell difficulty. On the second day of the mission, the radar system was exercised in a Cape Kennedy fly-over test with a ground-based transponder. Radar performance was nominal for this test. On the fourth and subsequent days of the mission, fly-over tests were conducted and a digital range problem developed. The maximum digital range was 24,800 ft. The problem was attributed to low temperature operation. Additional temperature tests were added to the test cycle for Spacecraft 6 and up.

Project 7/6. - The system flown is described in Project 7/6 Radar System, page 88. The radar system performed the closed-loop rendezvous maneuver. Radar performance was nominal for all phases of the mission.

Spacecraft 8. - The system flown is described in System Description, page 85. Radar system performance was nominal for the majority of the first rendezvous maneuver. A problem developed from 46 nautical miles to 22 nautical miles in the approach to the Agena. The crew reported fluctuations of the target angles on the flight director indicator (FDI). The fluctuating display did not prevent the crew from maintaining boresight on the Agena vehicle and did not hamper the rendezvous exercise. A subsequent analysis of all pertinent data revealed that a high frequency electrical breakdown caused the difficulty. As a result, all of the TDA coaxial connectors associated with the transponder were packed with a low vapor pressure grease.

Spacecraft 9. - The system flown is described in Spacecraft 9 Radar System, page 88. The radar used for the first and third rendezvous and radar system performance was nominal for all phases of the mission for the tumbling ATDA target vehicle. The high lobe structure of the tumbling ATDA antenna patterns caused variation in received signal strength and variation in target ellipticity with a corresponding fluctuation in radar AGC voltage and approximately one degree peak to peak excursions of the FDI needles. This was as anticipated.

Spacecraft 10. - The system flown is described in System Description, page 85. The radar system performed the rendezvous maneuver. Radar performance was nominal for all phases of the mission.

Spacecraft 11. - The system flown is described in System Description, page 85. Radar system performance was nominal for the first orbit rendezvous except for the transponder transmitter output which began to degrade in the latter stages of the rendezvous. The transmitter failed later in the flight. The most probable cause of failure was attributed to a leak in the transmitter oscillator assembly which allowed the pressure to decrease to a point where RF arcing occurred. Additional procedures were added to Spacecraft 12 checkout to insure the integrity of the oscillator assembly pressure seal.

Spacecraft 12. - The system flown is described in System Description, page 85. Initial lock-on was obtained at a range to Agena of approximately 236 nautical miles. Spacecraft angle and range information was sporadic throughout the flight and consequently the radar was not used as the primary source of rendezvous range and angle data. The sporadic range and angle information was due to an erratic transponder transmitter. The most probable cause of the transmitter failure was concluded to be arcing within the sealed transmitter oscillator assembly. The evidence indicates the pressure seal did not survive the stresses of Agena launch.

The arc-over could occur within the transmitter cavity pressure vessel in the following areas:

A. From the high voltage feed-through to the pressure vessel case.
B. From the high voltage terminal on the transmitter cavity to case.
C. From the transmitter tube plate ring of the oscillator cavity to case.
D. From the cavity plate (high voltage) to the tube grid ring of the cavity.
E. A DC arc-over within the tube.

The most probable areas of occurrence are A, B, and C.

DIGITAL COMMAND SYSTEM

System Description

The digital command system (DCS) 52-85714 consists of one receiver/decoder package and three relay packages. The DCS receives, decodes, and transfers to using systems digital commands transmitted from ground stations. Commands are categorized as either real-time commands (RTC) for spacecraft equipment selection or stored program commands (SPC) that consist of data for the time reference system or the digital computer. The system contains two FM receivers, either of which is capable of supplying sufficient output for proper decoder operations. The decoder verification circuitry provides for subsystem reset when invalid messages are detected or when data transfer is not accomplished within a fixed time duration. The probability of the decoder rejecting a valid message is less than 1×10^{-3} while the probability of accepting an invalid message is less than 1×10^{-6}.

DCS Subsystem And Spacecraft Evaluation

The receiver/decoder and relay units were subjected to extensive tests prior to installation in the spacecraft. A predelivery acceptance test was performed by the vendor on each unit. These tests included:

A. Visual and mechanical inspection.
B. Operation during and after random vibration.
C. Low temperature operation.
D. High temperature operation.
E. Room temperature operation.

Typical tests while at temperature extremes include message rejection rates and signal to noise ratio measurements.

After receipt at McDonnell the unit was further tested for:

A. Bandwidth and center frequency of receivers.
B. Quieting.
C. Improper vehicle address rejection.

D. Stored program command processing.
E. Real-time command processing.
F. Telemetry voltages.

After installation on the spacecraft, the operation of the DCS was again verified. The DCS was used to update the time reference system and computer with stored program information. In addition, the real-time command capability was used to control various spacecraft functions.

DCS Design, Development And Qualification Program

Design Program. - The digital command system was designed and built by Motorola's Western Military Division, Scottsdale, Arizona.

The need for a command system was realized early in the Mercury program. A tone command was used for real-time commands throughout Project Mercury.

The stored program capability to update the computer and time reference systems was unique to the Gemini mission. The real-time capability was a carry-over from the Mercury system. The command system was designed especially for the Gemini requirements.

Development Program. - All component parts used in the DCS were subjected to extensive qualification and reliability assurance testing. Standard logic design practices of logic diagrams, timing diagrams and circuit analysis were employed to locate marginal design areas.

The DCS was interfaced with the electronic timer and computer early in the program to verify the system integrity. The DCS was further subjected to ESTU, CTU and fly-over range tests with no problems.

Qualification Program. - The 52-85714-15 receiver/decoder and the 52-85714-21 and -23 relay packages were qualified as a system.

The command systems' only problems were encountered during random vibration testing of the decoder package. The problems were numerous with approximately 350 min of vibration time accumulated before successful completion of testing. Problems such as contaminated solder joints and wire breakage were uncovered. As a result of this test the wiring harness, deck plate mounting technique was redesigned.

DCS Reliability And Quality Assurance Program

Motorola utilized the standard reliability techniques of reliability estimates, failure mode and effect analysis and an internal failure reporting and corrective action system in order to achieve a reliable product. In addition, Motorola centered much of their program around their piece parts program. All parts were procured by Motorola part numbers to Motorola "high-reliability" specifications. These specifications required in addition to extensive screening that all parts be "burned-in."

Overstress Tests. - An overstress test was performed on the digital command system to further add confidence to the system integrity. Environments investigated were acceleration, vibration, low temperature and low voltage, and high temperature and high voltage while overstress testing disclosed a relay contamination problem and led to change in manufacturing process by relay vendor.

Flight Results

Analysis of flight data verified that the digital command system performed its function with no malfunctions.

TIME REFERENCE SYSTEM

System Description

The time reference system (TRS) is the central timing system of the Gemini Spacecraft. The system consists of an electronic timer, an event timer, a GMT clock, an Accutron clock, a mission elapsed time digital clock, and a time correlation buffer. (See Fig. 15.) The system is defined by McDonnell Report 8664 and A641.

Electronic Timer. - The electronic timer is a crystal controlled digital counter with a specification accuracy of 25 parts per million per day (roughly ±3 sec/day). Through a series of countdown chains and storage registers, it keeps track of elapsed time (time since lift-off), time to go to retrofire (TR) and time to go to equipment reset (TX). The basic timing unit of the timer is 1/8 sec intervals and all counting functions are kept to within this 1/8 sec interval.

The electronic timer is approximately 5-1/2 in. high, 5-1/2 in. wide and 8 in. long. It weighs about 9.3 lb and consumes seven watts of power.

The electronic time TR and TX values can be updated by the ground complex (by use of the digital command system) or by the crew (by use of the manual data insertion unit). The value of elapsed time cannot be altered. To prevent inadvertent or premature countdown to retrofire as a result of equipment failure or personnel error during updating, the timer will not accept any new time value less than a preprogrammed number. (512 sec or 128 sec dependent upon timer configuration.)

The timer provides timing information for the computer, mission elapsed time digital clock and PC telemetry system. It also provides timing pulses for the radar.

Event Timer. - The event timer provides a digital stopwatch for the crew. The display capacity of the timer is 59 min and 59 sec with a resolution of 0.2 sec. The timer will count up or down. It may be preset to a value and

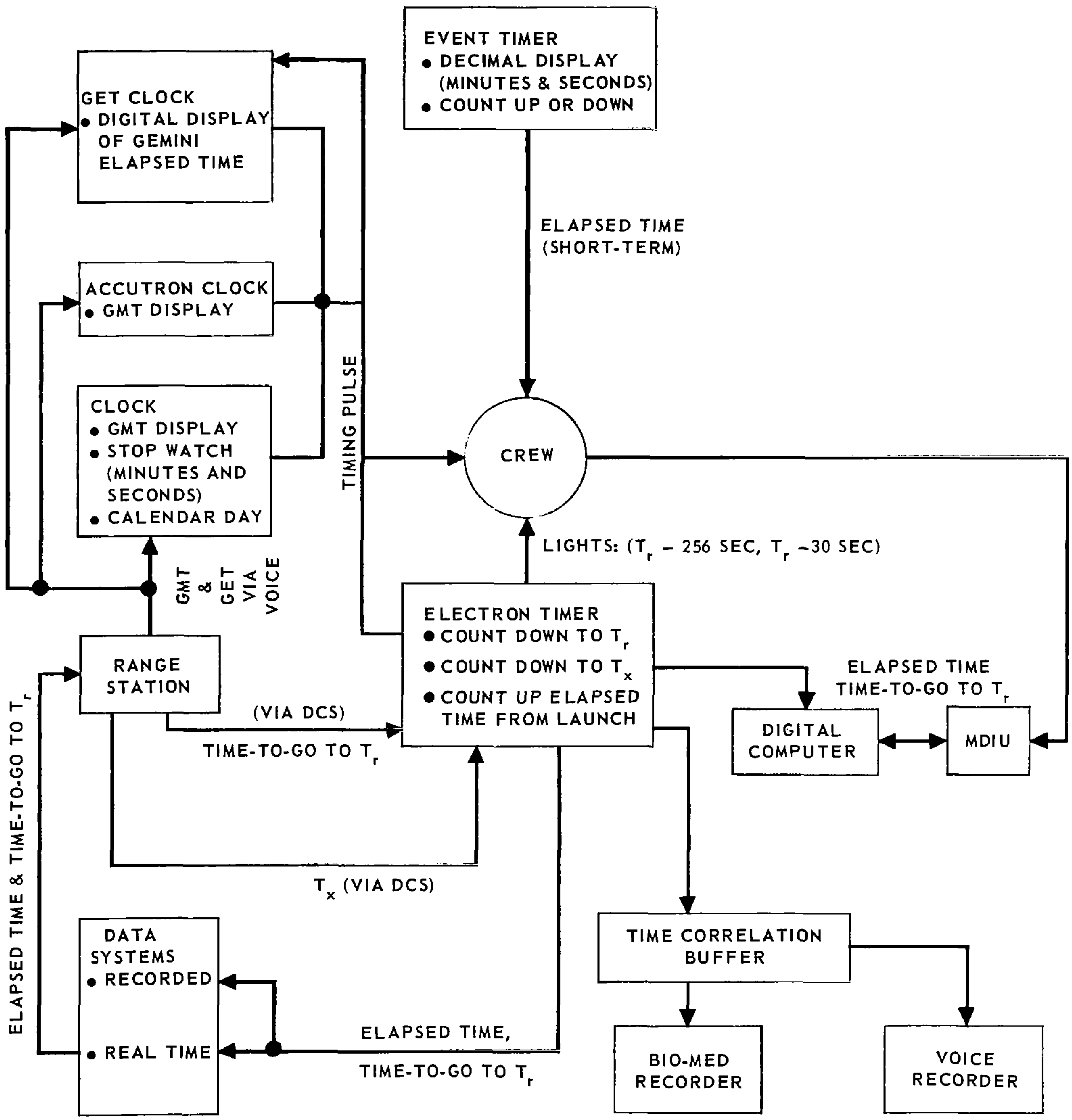


FIGURE 15 TIME REFERENCE SYSTEM FUNCTIONAL DIAGRAM

started or stopped by use of front mounted switches. It normally starts counting automatically up from zero at lift-off by receiving the lift-off signal from the electronic timer.

The timer is approximately two in. high, four in. long and four and one-half in. deep. It consumes two watts of power.

Greenwich Mean Time (GMT) Clock. - The clock displays Greenwich mean time in hr and min. It is a modified aircraft clock with a standard 24-hr face. As additional features a stopwatch and calendar day are provided. The clock also contains two markers by which the crew can mark significant events. Clock accuracy is ±15 sec/day.

Accutron Clock. - This clock displays GMT to the command pilot. The display is a 24-hr type with a sweep second hand. The accuracy is ±3 sec/day at 5.1 psia. It was used in Spacecraft 4 through 12.

Time Correlation Buffer. - The buffer provides time correlation to the biomedical and voice tape recorders. It accepts elapsed time and TR from the electronic timer and after format changes sends them to the applicable recorder. It was used in Spacecraft 4 through 12.

Mission Elapsed Time Digital Clock. - This clock provides a continuous display of mission elapsed time from lift-off. The display capability of the clock is 999 hr, 59 min and 59 sec. The clock accepts timing pulses from the electronic timer, making its accuracy the same as the electronic timer. The clock is started automatically at lift-off. It has the capability of being stopped, set to a time value, and restarted by the use of front mounted switches.

The digital clock was requested by NASA (per RFECP 231) because of the difficulty experienced by the crew in converting GMT to mission elapsed time. Since all spacecraft maneuvers and work tasks are defined in mission elapsed time, it was felt that an immediately available readout of this time was essential. Therefore, effective Spacecraft 6, the digital clock was installed in all spacecraft, although the GMT clock was not deleted. Identical electronic modules, mechanical design and construction techniques were used in the digital clock and the event timer. Because of the sameness of the components, only the event timer was subjected to actual qualification testing. The digital clock experienced no problems in SST or in flight.

Time Reference Subsystem And Spacecraft Evaluation

All the individual equipments used in the time reference system were subjected to extensive tests prior to installation in the spacecraft. The vendor performed a predelivery acceptance test on each unit. This consisted of a low-temperature test, a high temperature test, a vibration test and a complete electrical performance test including 24-hr accuracy. After installation on the spacecraft, the operation of the time reference system was again verified. The operation of the electronic timer was verified through its interface circuitry while an operational and accuracy test was performed on the other components.

Design, Development And Qualification Program

Design Program. - The time reference system excluding the GMT and Accutron clocks was designed and built by McDonnell's electronic equipment division. The Accutron was supplied by Bulova Watch and the GMT was furnished by Aerosonic Corporation.

The need for a central timing system was dictated by the use of controlled re-entry.

Development Program. - During the design phase the components of the TRS were subjected to extensive qualification and reliability assurance testing. A breadboard electronic timer was fabricated and interfaces were simulated in order to achieve confidence in the design integrity of the system. Numerous refinements and eliminations of possible problem areas were made during this phase of the program.

The ESTU was a laboratory fabricated spacecraft mock-up. The objective of this test was to assemble all electronic systems and determine if any interface problem existed. The electronic timer was subjected to this test and satisfactory results were obtained. No problems were encountered with CTU tests or fly-over range tests.

Qualification Program.

A. Electronic Timer - The A05A0017 electronic timer configuration was qualified. Summary of significant events of this testing is as follows:

1. Oxygen Atmosphere - Oscillator failure which was due to temperature sensitivity of oscillator circuit. The internal packaging of the oscillator was redesigned from printed circuit boards to terminal board construction. The resistors used in the oscillator were changed from Daven 1/10 watt metal film type to I.R.C. 1/10 watt metal film type.

2. Vibration - During the vibration test two separate problems were uncovered. The crystal oscillator broke loose from its mounting and sheet metal heat fins were fractured. The crystal oscillator broke loose from its cement bond which held it to module No. 1 causing a broken wire. It was found that the tinned surface of the oscillator (area cemented to module No. 1) should be etched prior to the bonding operation. Sheet metal heat fins which form a part of the base plate assembly were fractured during the vibration environment. It was found that the heat fins had become brittle as a result of a brazing operation used during their fabrication. The base plate was redesigned to utilize riveted rather than brazed heat fins.

B. Event Timer - The A05A0018 event timer was the qualification test unit. A summary of significant events of this testing is as follows:

1. Vibration - Failure in vibration was due to a broken wire lead from the electronics to the stepper motor. Corrective action consisted of adding additional wire ties to the wire bundle. After incorporating the additional ties the timer successfully completed the vibration test.

2. Humidity - Unit passed the humidity test after revising the method of applying the humiseal coating to the electronic modules.

C. GMT Clock - The J05A0002 clock was the qualification test unit. A summary of significant events of this testing is as follows:

1. Salt-Fog - This test uncovered a problem with the glass to bezel seal. The bezel gasket had to be modified and the face glass had to be sealed with epoxy to the bezel.

2. Vibration - Throughout the vibration tests numerous failures were encountered due to slippage of the elapsed time minute hand during the vibration environment. After much testing and corrective action, it was decided that the elapsed time minute had would always be susceptible to the vibrational environment. The spcecification was changed allowing slippage of the hand during vibration as long as the clock function operated properly after removal of the environment.

3. Humidity - Corrective action resulting from this test was plating the knob shafts to preclude their rusting.

D. Accutron Clock - No qualification testing was required on this item.

E. Mission Elapsed Time Digital Clock - The clock was qualified by similarity to the A05A0018 Event Timer.

F. Timer Correlation Buffer - The TCB was subjected to a qualification test program limited to the environments of vibration, temperature altitude and RFI. No problem areas were uncovered.

Time Reference Reliability And Quality Assurance Program

In order to maximize the reliability of the unit the vendor incorporated all of the standard reliability techniques into his program. Such items as reliability estimates, failure mode and effect analysis, worst case analysis and an internal failure reporting and corrective action system were implemented. In addition, the piece parts used within the timer were carefully screened. All piece parts were environmentally qualified at the part level and lot acceptance as well as 100% screening inspection was performed.

Overstress Tests.- Overstress testing was performed on the electronic timer and event timer.

A. Electronic Timer - Acceleration, vibration and temperature overstress environments were investigated. Test results verified the existence of a failure mode at temperatures above 250°F due to solderable $NYLEZ_e$ (Phelps Dodge Copper Products) 44 gauge core windings and the material used for encapsulation of the magnetic shift registers. At such temperatures, these materials became pliable, resulting in the possibility of shorts between the core windings in the magnetic shift registers. No other failure modes were revealed as a result of environmental stress levels above the design requirements.

B. Event Timer - Acceleration, vibration and temperature overstress environments were investigated. The actual vibration test was not run as the unit was qualified at a vibration level of 12.6 g RMS in lieu of the 8.8 g RMS level, which demonstrated an adequate safety factor. The timer operated satisfactorily in the overstress environments of acceleration and temperature. Both environments were approximately 125% of the qualification level.

Time Reference Flight Results

The flight of Spacecraft 4 disclosed that the Accutron clock lost approximately five sec per day. The specification limit for this clock was ± 3 sec per day. Investigation disclosed that the clock was not calibrated for the zero g environment. After a change in the calibration procedure flight results were well within the required accuracy. No other problems in the TRS were uncovered by flight data.

INSTRUMENTATION AND RECORDING SYSTEM

The Gemini instrumentation system may be divided into two categories, (1) signal sources and (2) the data transmission system, both of which have been fully qualified for manned flight.

Signal Sources

DC-DC Converter A05A0048. - The DC-DC converter regulator supplies prime regulated power to the instrumentation pulse code modulation (PCM) system tape recorder and associated signal sources such as pressure, temperature and synchro repeaters.

The design offered no major problems nor was there any significant advancement of the state-of-the-art.

During qualification testing with salt spray and humidity, bubbling and flaking of the protective coating on the magnesium case were detected. This problem was traced to improper application of the protective coating. Additional inspection steps were added to the painting process to correct this. Several such units were considered for reflight, but the case of each was found to be severely damaged by salt water. The successful solution was to replace the surface paint by an epoxy base paint.

The converters were manufactured by Conductron-Missouri in accordance with a rigid reliability and quality assurance program initiated by McDonnell/NASA. All performed satisfactorily during missions with one exception. During the flight of Spacecraft 3, S/N 119 DC-DC converter malfunctioned due to a loose nut causing an internal electrical short circuit. The loose nut was attributed to the failure of a star lockwasher to lock the nut. This anomaly revealed that component locking devices were a major problem area. The contractor reviewed all Gemini spacecraft equipment for star lockwasher use, and took the following corrective action on the DC-DC converter:

A. All future applications involving screws, studs or equivalent employed only split type lockwashers, unless the application was for other than vibration resistance.

B. All DC-DC converters were reworked using epoxy to bond the locking devices.

A similar Conductron-Missouri converter was used to supply power to the Gemini biomedical instrumentation system.

Temperature Sensors. - Platinum-resistive element sensors and thermocouples are the two types of temperature sensors. Thermocouples, manufactured by McDonnell, were used with the McDonnell reference junction box to monitor structural and skin temperatures on Spacecraft 1 through 4. Platinum element sensors, supplied by Rosemount Engineering Company, were used on all spacecraft and were defined in McDonnell specification control drawing (SCD) 52-88721.

The units were installed by McDonnell on spacecraft structure skin, in fluid lines, on fluid lines, and other subcontractor-supplied equipment. The platinum element sensors represent the recognized state-of-the-art methods of manufacturing reliable, strainfree, and accurate sensors.

Several problems were encountered in the qualification program. Per specifications, the desired repeatability is 0.2%. Two units experienced values of 0.28%. These readings were considered tolerable since the overall accuracy remained within the required 1.0%, and the later is the critical criterion.

The inline sensor failed under vibration during reliability overstress testing. Failure analysis disclosed the platinum sensor had ruptured due to cantilever mounting. The problem was resolved by eliminating the cantilever mount.

Credit is given to excellent vendor workmanship for temperature sensors which performed so well in flight. Over 500 devices were used to sense the various physical phenomena on the Gemini program, with only one significant flight failure. In this instance, suit temperature data for Spacecraft 7 were erroneous after 11 days in orbit. This failure was caused by water which accumulated in the suit circuit and penetrated the ceramic coating of the element. Per NASA direction, no corrective action was taken because no subsequent long-duration Gemini missions were planned.

Some problems were encountered with small platinum sensors which have delicate lead wires between bridge and element. These sensors occasionally were damaged in the course of additional work in the area. These sensors were kept small for the following reasons:

A. To preclude error from their effect as a heat sink.
B. To guarantee rapid response to temperature changes.
C. To minimize weight.

Pressure Transducers. - Two types of transducers, potentiometric and variable reluctance, are used to monitor the various parameters. Many of the potentiometer types are supplied with the spacecraft subsystem, such as the environmental control system (ECS) and the orbit attitude and maneuver system (OAMS). In addition, potentiometer units for suit and cabin pressure and static pressure, were supplied by Fairchild Controls Corp. in accordance with McDonnell SCD 52-88705.

The vendor adapted existing designs to the requirements with a minimum of development, and there were essentially no advances in the state-of-the art.

Qualification testing showed no significant problems. One unit did experience slight out-of-tolerance readings during acoustic noise testing. Analysis revealed the wiper-to-potentiometer pressure was insufficient. Manufacturing and quality assurance standards were revised to insure proper workmanship. The transducers performed well in flight with no significant loss of data.

Subsequently, two problems were encountered with the 52-88705 transducers. Solder contamination within the pressure bellows resulted in out-of-tolerance calibrations. This was overcome by revising the assembly of the bellows, changing the brazing material, and providing a final x-ray examination.

The second problem consisted of calibration shift after extensive shelf life. Potentiometer-type transducers tend to acquire a small amount of set in the bellows flexure through lack of exercise. Therefore, these units were retested every six months.

Variable reluctance-type transducers, supplied by Consolidated Controls Corp., are defined in McDonnell SCD 52-88722 and monitor fuel cell pressures. The transducers performed without failure in qualification tests and during flights. Their special use on Spacecraft 2 to monitor local static pressures during re-entry constituted an advancement of the state-of-the-art. One range was zero to ten mm of mercury, and accurate performance was required during elevated temperatures and re-entry vibrations. These units performed remarkably well with no loss of data.

One significant problem developed late in the program. All units exhibited susceptibility to external magnetic fields. This susceptibility was traced to the transformer in the oscillator circuit. Predelivery acceptance and preinspection acceptance testing insured that the susceptibility would be within allowable limits for accuracy.

Accelerometers. - The accelerometers used on the Gemini vehicle were the force balance or servo type supplied by Gulton Industries, Inc. in accordance with McDonnell specification control drawing 52-88712. This SCD also covers linear accelerometers for spacecraft static accelerations and systems for low frequency vibration measurements. This accelerometer did not represent an advancement in the state-of-the-art but the use of completely electrical damping was a significant development. The basic design has been in use for several years and proved to be highly accurate.

One failure occurred during qualification tests. After vibration, the unit was not accurate during a static acceleration test. Analysis showed a lead wire to the seismic system sensing vane had parted. The wire was repaired and the unit operated satisfactorily.

During vibration testing of Spacecraft 3, the static accelerometers revealed vibration noise. The seismic system which is the sensor portion of the accelerometer is supported by a jewel pivot. Investigation traced the vibration noise to excessive pivot to jewel clearance in the sensor. To correct this, all units were vibration-tested for proper clearance during the vendor and McDonnell tests. No accelerometer failures were noted during spacecraft flight. Data were used by aerodynamics and structural dynamics personnel to evaluate the spacecraft performance.

Synchro Repeater (52-88723). - Three synchro repeater assemblies in each spacecraft monitor the synchros on the IGS platform gimbals. Each synchro repeater output is a DC signal proportional to the spacecraft roll, yaw, and pitch attitude in terms of platform coordinates.

Design and development of the synchro repeater offered no major problems nor any significant advancement in the state-of-the-art.

During qualification tests a mechanical sealing problem occurred on the (-5) repeater. The external configuration was modified, using electron beam welding to seal the repeater hermetically.

Another qualification unit, serial No. 26, failed vibration due to a broken motor shaft. The failure was traced to faulty machining in the motor shaft. Manufacturing and inspection procedures were revised to insure proper workmanship. Serial No. 22 replaced serial No. 26 and the vibration test was passed successfully, along with other phases of the qualification test.

A major problem late in the program was damage of the synchro gears by electron beam welding. Although visual inspection of the sealed-in gears is not possible, most of those delivered suffered some gear damage during welding. Corrective action included requalifying the electron beam process and burning-in the synchro gears for several hours. Synchro repeaters provided satisfactory data on all Gemini missions.

Signal Conditioners. - The signal conditioners on the Gemini program consist of DC voltage monitors, AC voltage monitor, AC frequency sensor, phase sensitive demodulator, attenuators, DC millivolt monitor, and other specialized cards. These cards were designed and manufactured by McDonnell with the exception of the one type of phase sensitive demodulator and DC millivolt monitor which was designed and manufactured by Eclipse-Pioneer Division of Bendix Corp. Problems encountered during the qualification program concerned the drift of several percent of a number of modules during vibration testing. To correct this, the internal package connectors were modified to assure more positive contact.

No signal conditioner problems were encountered in flight. Most of the instrumentation packages recovered for post-flight analysis had remained sealed and none contained salt water. All the packages and signal conditioners passed a thorough post-flight checkout, and several conditioners were allocated for reflight in subsequent spacecraft.

CO_2 Partial Pressure System (52-88715). - The detector, mounted on the ECS package, beneath the spacecraft crew seats, monitors the condition of the carbon dioxide absorber in the ECS. An eight cc/min sample of the heat exchanger effluent is supplied to the CO_2 sensor. After measurement the gas passes into the compressor inlet. The concentration of CO_2 is displayed on an indicator in the cabin and is transmitted to the ground via the Gemini PCM telemetry system. Within the detector the stream is divided into two substreams. One, the reference stream, passes through a filter which removes CO_2. The other, the measurement stream, feeds through a passive filter which does not remove CO_2. Both streams then pass through identical ion chambers that contain a small amount of radioactivity which ionizes the gas. Since CO_2 ionizes more readily than O_2, the ion current from the measurement stream will be higher than that from the reference stream. The ion currents are subtracted by the bridge circuit formed by the ion chambers, polarization voltage sources and a high megohm resistor. The difference current flowing through the high megohm resistor is proportional to the amount of CO_2 in the measurement stream.

This measurement technique and its associated equipment advanced the state-of-the-art, and proved successful in the Gemini program. During the mission simulator phase of the qualification program, the sensor did not respond properly to CO_2 in the presence of a high percent of water vapor. The problem was eliminated by reducing the gas flow, thus reducing the water vapor to a level that could be absorbed by the filter.

This device passed all other phases of the qualification test, and provided satisfactory data on all Gemini missions.

PDA and PIA tests on the PCO_2 detectors revealed out-of-tolerance zero shift and pressure sensitivity. This problem was eliminated by changing the radioactive sources to an americium isotope. Subsequent testing showed in-tolerance zero drift and pressure sensitivity.

Pulse Code Modulation (PCM) Telemetry

This technique of data transmission utilizes digital (binary) coded pulses to transmit the sampled data. After measurements are encoded, they are transmitted over a radio link to ground receiving stations. The PCM data system is supplied by Electro-Mechanical Research.

Real Time Data Transmission System. - The Gemini data transmission system consists of:

A. A pulse code modulation subsystem.
B. A FM transmitter.
C. A spare FM transmitter.

The transmitters are arranged so that the spare unit may be used to transmit real time data (in the event of real time transmitter failure), on command from the astronaut or on command from the ground station digital command system (DCS).

This system sampled a maximum of 368 parameters. The system will receive and transmit:

A.	106	High level analog parameters	0 to +5 V
B.	117	Low level analog parameters	0 to +0.020 V
C.	1	8 bit digital time word	
D.	24	24 bit digital words	
E.	88	Bilevel - one bit indications (on - off)	+24 V or 0 V
F.	32	Bilevel pulse - one bit (on - off)	+24 V or 0 V

The PCM system consists of a programmer, two high level multiplexers, and three low level multiplexers. If the maximum system capacity is not required for a specific mission, any or all of the remote multiplexers may be removed without affecting the rest of the system. The parameter monitoring capacity of individual system components is shown on page 11.1 of specification control drawing 52-85713.

State-of-the-art construction and packaging techniques were used in the construction of the first production models. The state-of-the-art of the multilayer printed circuit board may have been advanced by solutions to the problems encountered in the construction of programmer mother cards. The state-of-the-art for low signal level, solid state DC amplifiers was advanced by the development of the amplifier-clamp and sample-and-hold circuitry.

The transmitter design advanced the state-of-the-art by development of fully transistorized 2.5-watt transmitters, with adequate heat sinking of components and without temperature-controlled components.

A typical production unit of each data transmission system (DTS) item was subjected to the qualification environment specified for Gemini.

A. Problem Areas Detected During Qualification Testing

UNIT UNDER TEST	TEST WHERE PROBLEM APPEARED	FAULT	CORRECTIVE ACTION
1. PCM programmer L.L. multiplexer H.L. multiplexer	Low Temperature	Data error as a function of temperature.	Specification was widened to accept this error.
2. Low level multiplexer	Low Temperature and High Temperature	Cold solder joints in solder pins of wiring harness.	Wires were crimped in pins.
3. Low level multiplexers High level multiplexers	Vibration	Lockwashers did not hold.	Lock nuts were used and epoxy was applied to bolt and nut assemblies.
4. Low level multiplexers High level multiplexers	Humidity	Water collected inside case.	Silicone rubber compression gasket was added to O-ring seal.
5. PCM programmer	Salt water immersion	Water seeped into the case.	Internal printed circuit boards were impregnated with a resilient varnish, connectors filled with silicone grease.
6. Transmitter	High Temperature	Transistor failure.	Redesign of output transistor heat sink.

B. Major Problems Encountered with Delivered Data Transmission Systems

PART NUMBER	FAULT	CORRECTIVE ACTION
1. 52-85713-475	High offset on several channels when multiplexer installed in adapter location.	Low level amplifier frequency response adjusted to a control bandwidth for noise elimination.

	PART NUMBER	FAULT	CORRECTIVE ACTION
	High Level Multiplexer		
2.	52-85713-387	Erroneous resets encountered from noise on control lines.	Diode quad inserted in reset base circuit.
3.	52-85713-77	Inadequate counterdrive and leakage encountered in landing phase of flight and difficulty encountered in handling multiplexers in spacecraft assembly area.	Decrease series resistance in counterdrive circuitry. Redesign case to accommodate cover holddown screws.
4.	52-85713-377	Water leakage encountered in post-flight phase of mission.	Gasket formed in place with RTV 891 silastic.
	Data Transmission System		
5.	52-85713-63, -65, -67 and -69	Air leakage occurring during exposure to orbital environment caused corona to disable transmitter.	All seal screws and connectors were coated with epoxy sealing materials.
	PCM Programer		
6.	52-85713-283	Noise on ground lines caused high data error.	Changes grounding philosophy of shielded wiring carrying control signals to remote units.
7.	52-85713-383	Leakage encountered in programmers used in Spacecraft 3 flight.	Conformal coating seal applied to the mother boards and DC-4 grease applied to connectors. Leakage test of all programmers initiated.
8.	52-85713-483	Multiplexers were erroneously reset by noise on the control lines.	Increased the power supply drive capability and inserted new N gate in reset drive circuitry.
	Low Level Multiplexer		
9.	52-85713-85	Leakage occured during salt water immersion.	Seal around lid was improved.

PART NUMBER	FAULT	CORRECTIVE ACTION
	Vibration caused discontinuity in internal connectors.	Electronic circuitry was encapsulated in resilient potting.
10. 52-85713-385, -485	Erroneous resets encountered from noise on control lines.	Diode-quad inserted in reset emitter circuit and in reset base circuit.
11. 52-85713-75	Difficulty encountered in handling multiplexers in spacecraft assembly area.	Redesigned case to accommodate cover holddown screws.
12. 52-85713-375	Inadequate counterdrive and leakage encountered in flight.	Decreased series resistance in counterdrive circuitry and improved silastic gasket.

Delayed Time Data Transmission System. - The Gemini delayed time data system consists of a tape recorder, a FM transmitter, and the spare FM transmitter. The recorder accepts all low sample rate data from the PCM in the RZ data form, converts it to a diphase signal, and stores it on magnetic tape. When the spacecraft comes within RF range of a data handling ground station, the recorder may be commanded to play back the recorded data, which is transmitted to the ground station in the NRZ's data form. The transmitters are arranged so that the spare unit may be used to transmit delayed time data on command from the astronaut or on command from the ground station digital command system (OCS).

The state-of-the-art was advanced in obtaining a bit-packing density of 2600 to 3000 bits/in. The art was advanced also by the mechanical handling of the tape using negator springs to maintain tension between the reels and using Mylar belts to prevent backlash of the tape drive mechanism.

A. Major Problems Encountered in the Tape Recorder - The tape recorder qualification test revealed a severe problem with high frequency bit rate stability when data was recorded while the unit was being vibrated.

Any vibration which is transmitted to the tape transport causes flutter of the tape as it passes over the record head. Flutter causes the data bits to be deposited unevenly along the tape. Since this recorder did not use an output storage register, but played back directly from the tape to the NRZ's converter, the output data string presented a bit jitter situation which was not acceptable to the ground station decommutation system.

Spacecraft 6 and 7 experienced a stopped condition of the tape recorders while operating in the normal orbital phase of the mission. Post-flight investigation of both recorders revealed a frozen bearing in identical locations in each of the recorders. The frozen bearing was failure analyzed and was determined to be an application and design fault, rather than a bearing fault. The problem was solved by redesigning the shaft collar located next to all bearings in the clutch mechanism.

In Spacecraft 5 systems testing, drive belts in the tape recorders broke, causing recorders to stop running after approximately 100 hr in operation. The problem was attributed to the manufacturing process involved with forming the belts from flat Mylar stock. This problem was corrected by tighter quality control on the manufacturing process and by careful handling.

B. Tape Recorder Problems Detected During Qualification Testing

1. Tape recorder	Vibration	Redesign of tape drive system to employ dual capstan drive, plus addition of internal isolation mounts.
	Thermal shock	Reflective tape was applied to cover.

C. Major Problems Encountered with Delivered DTS Tape Recorders

RECORDERS	FAULT	CORRECTIVE ACTION
1. 52-85713-31	Excessive jitter of data recorded during vibration environment (2 g random).	Interim fix was to operate recorder in high speed record mode.
2. 52-85713-35	Internal connection of signal ground to power ground fed noise back into spacecraft grounding system.	Isolated grounds within the recorder and incorporate a floating power supply.
	Vibration environment (6 g) produced excessive jitter on playback of data.	Vibration isolation mounts were installed within the recorder case to separate the tape handling transport from the vibrating case.
3. 52-85713-37	Data format prevalence of zeros caused loss of sync of ground station caused by absence of transitions used for sync lock.	Recorder playback circuitry was modified to provide NRZ's data format.
4. 52-85713-41 -39	Spacecraft 6 and 7 experienced a failure of bearings in the tape drive clutch mechanism.	Redesigned the clutch shaft and bearing retainer.

RECORDERS	FAULT	CORRECTIVE ACTION
	Mylar drive belt broke after relatively few operating hours.	Improved the belt manufacturing procedure and strengthened the quality control of the belt handling practices.
5. 52-85713-339 -341	Negator-spring not adequately wound during manufacture or during one of the service periods.	Modified the test procedures to include a preflight checkout per RCA Doc. No. 564-125, in which the winding of the negator-springs is checked.
6. 52-85713-441	Clutch bearing froze during normal operating conditions in Spacecraft 11 SST.	All recorders were returned to RCA for new bearings, and lubricated with J-type oil prior to prelaunch tests.

GUIDANCE AND CONTROL SYSTEM

The Gemini guidance and control (G & C) system consists of subsystems which perform onboard navigation and attitude control of the spacecraft. Fig. 16 presents a block diagram of the G & C subsystems.

Design And Development

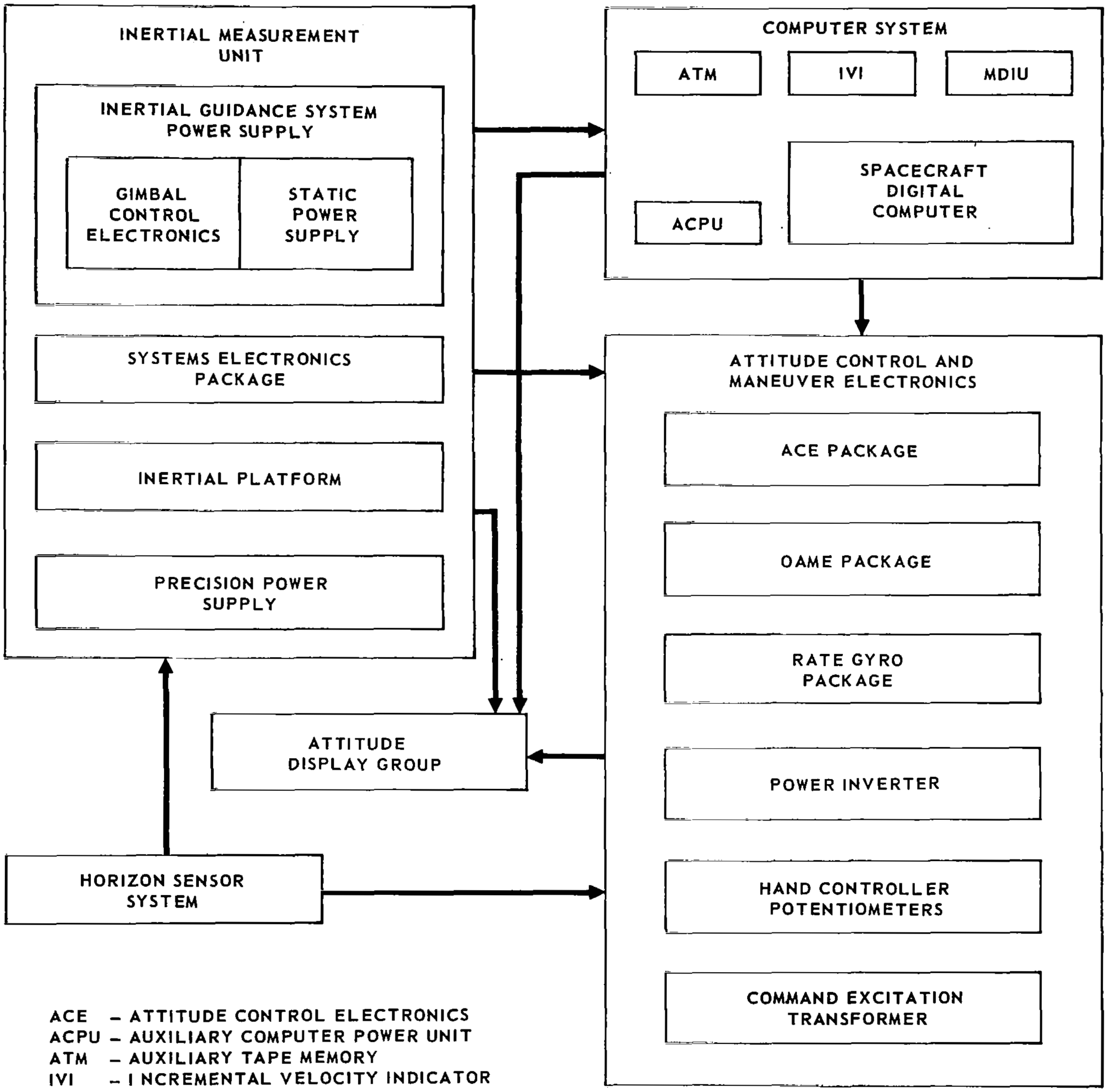
The program objectives of long duration, rendezvous, and controlled re-entry missions have placed special requirements on the spacecraft guidance and control systems. These objectives required maximum reliability and flexibility in the equipment. This was accomplished by utilization of simple design concepts, and by careful selection and multiple application of the subsystems to be developed. The guidance and control subsystems and their development problems are briefly described hereunder.

Inertial Measurement Unit (IMU).

A. IMU System Description - During initial evaluation of the Gemini mission the inertial measurement unit evolved into equipment consisting of an inertial platform, platform mount (including coldplate), system electronics, and inertial guidance system power supply. Major requirements for individual elements were:

1. Inertial Platform

   a. All attitude freedom.



ACE – ATTITUDE CONTROL ELECTRONICS
ACPU – AUXILIARY COMPUTER POWER UNIT
ATM – AUXILIARY TAPE MEMORY
IVI – INCREMENTAL VELOCITY INDICATOR
MDIU – MANUAL DATA INSERTION UNIT
OAME – ORBIT ATTITUDE AND MANEUVER ELECTRONICS

FIGURE 16 GUIDANCE AND CONTROL BLOCK DIAGRAM

b. Dual analog gimbal angle readout for pitch, yaw, and roll; one signal to be a synchro output and one signal to be convertible to digital representation.

c. Accelerometers to be compatible with digital rebalancing.

d. Capable of operation in space vacuum, with in-orbit turn on/off and orbital alignment capability.

2. Platform Mount - Provide temperature control for the inertial platform.

3. System Electronics - Provide accelerometer rebalance loops to receive accelerometer analog signals, compensate as required, and convert rebalance current to digital representation for the computer.

4. Static Power Supply - Convert spacecraft bus power, as required, to supply all inertial guidance system components.

5. Gimbal Control Electronics - Contain all circuitry associated with the gimbal servo loops.

B. IMU Development Tests - Aside from breadboard testing, developmental testing was conducted on four engineering model IMU's, which were fabricated and allocated as ESTU, CTU, CCTU, and HI test bed units.

1. Electronic System Test Unit (ESTU) - This IMU was used at McDonnell in a spacecraft mock-up to evaluate system and electrical interface capability and to verify spacecraft wiring and test procedures. This system was also used for dynamic analysis and astronaut familiarization.

2. Compatibility Test Unit (CTU) - This IMU was used at McDonnell in a production prototype spacecraft to assess complete spacecraft compatibility. The CTU tests included evaluation of coldplate performance, system/telemetry compatibility, and system/aerospace ground equipment compatibility.

3. Configuration Control Test Unit (CCTU) - IBM used this IMU for IGS integration and for developing equipment and procedures employed in production testing of IMU-computer integration.

4. HI Test Bed - This engineering model, allocated for use at Honeywell, was used initially in functional and output signal verification tests and for preliminary interface testing. It was used also in exploratory environmental tests to evaluate thermal and vibration design capabilities.

C. IMU Problems Located During Developmental Tests

1. Static Power Supply (SPS) Problems - The problems which were encountered with the static power supply are:

a. Computer DC Voltages - Computer loading introduced noise and transients on the output voltages which caused malfunctions. This was corrected by raising computer DC voltages 3% which compensated for the poor static set point on this section. Additionally, the Engineered Magnetics 1262 SPS was modified for Spacecraft 2 to accommodate the auxiliary computer power unit (ACPU).

b. Computer Section Leakage Voltages - Computer malfunctions were caused by leakage voltages through semiconductor switches on the computer DC lines during IMU turnon. The SPS leakage voltage problem was corrected by the incorporation of a relay into the computer 20.7 VDC line.

c. 26 Volt - 400 cps Section - Because of a power factor mismatch, the 400-cycle section performed out of specification when maximum 400-cycle IGS loads were applied. The 400-cycle section of the EM 1262 supply was improved by incorporating a parallel inductor to tune the SPS to the load.

d. Cross-Coupling Between SPS Sections - Computer dynamic loading on the power supply caused cross-coupling effects. Pulsating computer loads caused a low frequency noise which was reflected back through the SPS, onto the spacecraft bus, and back into the IMU and AC sections.

e. Power Supply Damage - Several power supplies were damaged by low resistance loads on the output lines. The Engineered Magnetics 1262 SPS design afforded overload protection for the application of hard shorts, but for low resistance loads the SPS would overheat and damage power transistors or cause them to fail.

For problems IV and V, EM 1262 SPS was modified to the EM 1262A configuration which incorporated the following changes:

- The PWR frequency response was converted from 2400 cps to 15 KC and magnetic amplifiers were replaced with transistorized switching circuits. These changes reduced dynamic loading and cross-coupling problems.
- The 400-cycle section components were selected to match the load.
- Overload protection was added only for ground operations to minimize the effects of inadvertent overloads and short circuits during ground handling. The individual SPS sections were fused so that an in-flight short would disable only the affected section. The 400-cycle section was modified to incorporate an overload protection circuit with an automatic reset capability which remained in effect during flight.

2. Inertial Platform Problems - The problems which were encountered with the inertial platform were:

a. Vibration Problems

- During low level (± 2 g peak) sinusoidal vibration and random (to 6.2 g RMS) vibration, excessive electrical lead breakage was experienced. This was corrected by improving wire routing and by securely attaching circuit elements.
- During subsequent low level sinusoidal vibration and random vibration up to 12.6 g RMS, additional mechanical problems were encountered. Also, excessive gyro drift rates and losses of gimbal reference were experienced. Tests revealed that rotary components introduced weak spring constants

into the gimbal support system, causing unequal spring rates across a gimbal. These conditions resulted in inner gimbal motion with crosstalk and rocking. Consequently, the rotary components were stiffened and the bearings were changed to utilize higher contact angles and preloads.

- Under additional low-level sinusoidal vibration and random vibration up to 12.6 g RMS the platform still indicated gimbal No. 3 loss of reference, high gyro drift rates, and nonrepeatability of results. However, structural integrity and performance through 6.2 g RMS were considered acceptable.

b. Synchro Phase Shift Problem - Phase shift on the platform synchro outputs introduced excessive deadband in the ACME control. The platform was modified to include an inductance-capacitance phase shift network, which corrected the phase shift from 15 degrees to 9 degrees.

c. Slip Ring Problem - Problems were caused by slip rings which intermittently opened. The platform slip rings had been noted earlier as a possible problem area and funds were allocated at contract initiation for a slip ring improvement program. This program was expedited and the improved (Gemini-NASA) slip ring was incorporated in all platforms.

3. Gimbal Control Electronics Problems - GCE Problem with -30 VDC Regulator - The GCE was exposed to a temperature of 50°F. At fast heat dropout all gimbals would break into ± 1 degree oscillations because of low current gain in the -30 VDC regulator, which would not turn on at low temperatures. Since this output served as a supply voltage for the GCE rate and summing amplifiers, the loop stabilization compensation was degraded and oscillation resulted. The corrective action consisted of modifying the -30 volt regulator circuit.

4. System Electronics - Horizon Sensor Ignore Delay Problem - The platform was driven out of alignment whenever the horizon sensor reacquired the horizon after a loss-of-track condition. The System Electonics was modified to generate a time delay upon a signal that the horizon sensor had regained track. The ignore relay was delayed in dropout for seven to ten sec.

D. IMU Design Problem

1. Gimbal Control Electronics - During predelivery acceptance tests on the first production IMU, a gimbal oscillation problem occurred. The problem originated from 400-cycle pickup from the spin motor appearing on the gyro signal generator. Coupling resulted because of perpendicularity between separate gyro input and output axes. Corrective action consisted of adding a notch filter circuit in parallel with the summing amplifiers.

<u>Attitude Control and Maneuver Electronics.</u>

A. ACME System Description - The basic attitude control and maneuver electronics (ACME) system consists of an attitude control electronics (ACE) package, an orbit attitude and maneuver electronics (OAME) package, a power inverter, and two rate gyro packages. In this discussion the command excitation transformer and three hand controller potentiometers are included in the

ACME system. The redundant rate gyro packages, along with the internal redundancy of the ACE and OAME packages, provide a high system reliability for prolonged missions.

1. Attitude Control Electronics (ACE) - The ACE package contains the circuitry which translates input signals into thruster command signals. The command signals, at the discretion of the astronauts, may be used to activate solenoid valve drivers (SVD) in either the re-entry control system (RCS) or the orbit attitude and maneuver system (OAMS).

The various modes of operation provide the astronauts with specialized control for all phases of the mission. Modes of operation are:

a. Direct - A manual mode wherein switches in the attitude hand controller activate the appropriate SVD's.

b. Pulse - Same as direct, except that ACE limits the thrust to 20 ms for each activation.

c. Rate Command - Combines astronaut commands from the hand controller potentiometers with the rate gyro signals to provide position and/or rate control.

d. Hor Scan - Utilizes position signals from the horizon sensor to maintain the spacecraft pitch and roll axes relative to the horizon. Yaw control is the same as in the pulse mode.

e. Platform - (Not used in Spacecraft 3 and 4.) An automatic, closed-loop control mode that establishes thruster firing for all axes as a function of both attitude signals from the platform and rate sign signals from the rate gyros.

f. Re-entry Rate Command - Same as rate command mode, but control is coarser and includes a roll-to-yaw rate cross coupling feature.

g. Re-entry - A closed-loop mode translates rate signals and the computer roll signal into thruster commands to achieve the desired downrange and crossrange impact point.

2. Orbit Attitude and Maneuver Electronics (OAME) - The OAME package provides solenoid valve drivers for the OAMS attitude thrusters and for the OAMS maneuver thrusters for Spacecraft 2, 3, and 4. ACE attitude thruster commands and bias power activate the attitude SVD's; the maneuver SVD's are controlled by the maneuver hand controller. For Spacecraft 5 and up, the maneuver thrusters are directly controlled by the maneuver hand controller.

3. Power Inverter (PI) - The PI package is used to provide 26 VAC when the IGS inverter is not energized.

4. Rate Gyro Package (RGP) - The RGP (two per spacecraft) contains three orthogonally-mounted rate gyros and associated circuitry. Each gyro within the package requires independent 26 VAC, 400 cps excitation and has both a self-test torquer and a spin motor rotation detector (SMRD) for ground checkout.

5. Hand Controller Potentiometer - The hand controller potentiometer consists of three ganged sections. One section provides rate command signals to ACME, another section provides an output signal to telemetry, and the third

is not used. The attitude hand controller contains three potentiometers, one for each axis.

6. Command Excitation Transformer - The command excitation transformer steps down the 26 V, 400 cps to 15 V for excitation of the rate command sections of the hand controller potentiometers.

B. ACME Development Tests - McDonnell performed the following tests on ACME hardware:

1. Interface Breadboard (IBB) - The ACME IBB consisted of the circuitry necessary to provide:

a. Proper loading for the attitude hand controller, horizon sensor, platform, computer, and radar.
b. Generation of simulated rate gyro signals.
c. Complete selection of all ACME modes of operation. The ACME IBB was interconnected with similar IBB's of the other guidance and control equipment. No problems affecting the ACME were uncovered.

2. Electronic Systems Test Unit (ESTU) - The ESTU tests, while not revealing any new ACME problems, did provide an excellent test bed to investigate anomalies discovered in earlier phases of the Gemini program.

3. Compatibility Test Unit (CTU) - No ACME problems were uncovered during the CTU testing.

4. Thermal Qualification - The CTU was subsequently redefined as Spacecraft 3A and subjected to a thermal qualification test. The ACME performance was satisfactory.

5. Dynamic Testing - ACME hardware from the ESTU and CTU programs was used in three dynamic test programs:

a. Fixed base simulator.
b. Three-axis motion.
c. Air bearing table. ACME performance was satisfactory.

C. ACME Design Problems

1. ACE Package

a. Early testing revealed that the sensitivity of the half-wave demodulator to harmonics and quadrature was such that the switching deadbands converged when high attitude and rate error signals were summed. The problem was resolved by incorporating a full-wave demodulator.
b. Due to the "break before make" mode switch in the spacecraft, the selection of any mode except hor scan caused momentary spurious jet firing pulses. A capacitor was added to the mode ACE logic circuit to prevent the spurious pulses.
c. Low temperature tests revealed that the minimum pulse generator would not always turn off. A coupling capacitor was found to cause excessive

attenuation of the turn-off signal. The problem was eliminated by increasing the value of the coupling capacitor.

d. An analysis of the pitch axis hor scan mode operation revealed that false thruster firings could be generated. This defect was eliminated by using the horizon sensor "loss of track" signal to activate a mode logic relay.

e. Solder particles were found inside several Babcock power relays. The problem initially was resolved by utilizing reworked relays. Later package configuration used welded can relays.

f. It was determined that excessive bounce in the hand controller switch could temporarily disable the single pulse generator input circuit. The deficiency was eliminated by modifying the generator input circuit.

2. OAME

a. The Babcock latching relays failed to withstand prolonged dry circuit operation. The replacement relay, manufactured by C. P. Clare, was expressly designed for low current circuits.

b. Vibration tests revealed that additional support was needed for the major trunks of wire harness. This was accomplished by the application of RTV 7228.

c. Solder particles were found inside several Babcock power relays. The problem initially was resolved by utilizing reworked relays. Later package configuration used welded can relays.

3. Power Inverter - The susceptibility of other systems to transients dictated the need for suppression of the output turnon transient. The suppression was accomplished by installing back-to-back zener diodes.

4. Rate Gyro - Life tests uncovered a bearing problem that caused an unpredictable reduction in the gyro life span. The problem was solved by switching to bearings with impregnated phenolic retainer rings.

5. Hand Controller Potentiometer

a. The potentiometer failed completely during the salt-fog exposure of the qualification testing. This problem was eliminated by sealing the potentiometer.

b. Slight shifts in the deadband tap locations occurred during qualification vibration testing. The tap deadband was widened to circumvent potential problems due to shifting. This relaxation has no measurable effect on spacecraft operation.

Computer System.

A. Computer System Description - The computer system is comprised of the spacecraft digital computer and supplementary equipment which consists of the manual data insertion unit, the incremental velocity indicator, the auxiliary computer power unit, and the auxiliary tape memory.

1. Computer - The spacecraft digital computer is a binary, fixed point, stored program, general purpose computer that operates at a 500 KC arithmetic

bit rate. It interfaces with the inertial platform, platform electronics, IGS power supply, auxiliary computer power unit, manual data insertion unit, incremental velocity indicator, rendezvous radar, time reference system, digital command system, data acquisition system, attitude control and maneuver electronics, GLV autopilot, console controls and displays, auxiliary tape memory, and aerospace ground equipment.

The computer provides the guidance and control outputs required for the spacecraft during various mission phases.

The computer memory is a random access, coincident current, non-destructive readout ferrite array, which has a 250 KC cycle rate. The memory array provides for 4096 words of storage or 159,744 bits. All memory words of 39 bits are divided into three syllables of 13 bits. Instruction words (13 bits) are intermixed in all three syllables. Data words (25 bits plus sign) are located in syllables zero and one.

The operational computer program contains interleaved diagnostic subroutines that permit malfunctions to be detected during operation. When a fault is detected, a computer malfunction lamp is lit on the astronaut's control panel.

2. Incremental Velocity Indicator (IVI) - The IVI was designed to provide a visual indication of the incremental longitudinal, lateral, and vertical velocity components and the associated sign for each to be imparted to the spacecraft by the maneuver thrusters.

The IVI has three digital displays with a range of 0-999. Each display consists of three counters, which can be set automatically by knobs on the front of the computer.

The IVI accepts two types of inputs from the computer:

a. Information Pulses - Each pulse corresponds to a one fps increment of spacecraft velocity.

b. Zero Set Pulses - A "zero set" pulse triggers 50 PPS fixed oscillator in the IVI which drives the indicators to zero at the rate of 50 fps.

3. Manual Data Insertion Unit - The manual data insertion unit (MDIU) consists of a manual data keyboard (MDK) and a manual data readout (MDR) unit. The MDK inserts decimal numbers into the spacecraft digital computer. The manual data readout displays address and data as a readout from the computer. The MDIU can insert up to 99 words into the computer and display the same number from it. All interface of the manual data keyboard is made with the manual data readout, which in turn interfaces with the manual data readout, which in turn interfaces with the digital computer.

4. Auxiliary Computer Power Unit (ACPU) - The ACPU operates in conjunction with the IGS power supply. It maintains stable input voltages at the digital computer during spacecraft bus transients and depressions to prevent alteration of the computer memory. The ACPU contains a battery which provides

regulated voltage to the computer converter within the IGS power supply during spacecraft bus transients of 100 ms or less. The ACPU also sends a signal to shutdown the IBM computer when the transient exceeds 100 ms.

5. Auxiliary Tape Memory (ATM) - The ATM operates in conjunction with the spacecraft digital computer. It provides additional program storage for the computer and can load the computer memory with any operational program which is stored within it. The programs are stored on magnetic tape. Data are transferred from the ATM to the computer under computer control with sum checks and parity checks eliminating any erroneous data.

The ATM was first installed on Spacecraft 8.

B. Computer System Development Tests - During the development stage, electronic systems test unit (ESTU) and compatibility test unit (CTU) tests were performed on the computer system (excluding the ATM). The ESTU tests included a mission simulation test to determine the operational compatibility of the guidance and control, communications, telemetry, and sequenctial systems by approximating the system combinations and sequences of events during a rendezvous mission. No changes in the design of the spacecraft digital computer were required as a direct result of the mission simulation test.

The CTU test program operationally verified each of the computer's interfaces. During the computer/ACPU tests, a computer turnon - turnoff oscillation was caused by normal ACPU operation in the presence of a critical bus voltage which was affected by computer loading.

The corrective action for Spacecraft 2 was a wiring change in the 52-89723-7 ACPU, which disconnected the low voltage sensor from the spacecraft bus and connected it to the "low voltage sense" output going to the IBM computer.

C. Computer System Design Problems

1. Computer - No major problems were encountered during the qualification testing.

2. IVI - Hardware changes resulting from design problems included changing the switch stepping rates from 1 through 10 and 50 fps to 1, 2, 5, 10, and 50 fps; increasing the rigidity of the case to improve sealing; and incorporating new transistors to eliminate short circuiting.

3. MDIU - No significant design problems were encountered.

4. ACPU - Design problems included failure of the trickle charger circuit and the high frequency oscillation under load. Both problems were corrected by circuit changes. A modification was made to increase the battery capacity.

5. ATM - An excessive error rate was caused by insufficient write current. The error rate was decreased to one-tenth of its former rate by

decreasing the pole tip width and by modifying the write amplifiers to increase the write current by 50%.

During prequalification vibration testing, the flutter tolerance specification was not met. Subsequently, the tolerance specification was increased. The vibration levels were reduced to meet the new requirements by using brass flywheels and weights on the tape reels.

The first attempt to operate the ATM with the unit sealed and pressurized with nitrogen failed due to electrostatic discharges which altered the data read from the tape. The problem was eliminated by using a gas mixture of 70% nitrogen, 20% freon and 10% helium.

Horizon Sensor System.

A. System Description - Each horizon sensor system consists of a sensor head and an electronics package. The system establishes a spacecraft attitude reference to earth local vertical, and generates error signals proportional to the difference between spacecraft attitude and the local earth vertical.

Attitude error signals are used to align either the spacecraft or the inertial platform to earth local vertical. Two sensor systems are installed on each spacecraft. The second system is redundant.

The horizon sensor system operates by receiving, detecting, and tracking are infrared radiation gradient between space and earth. The system employs infrared optics, infrared detection, and three closely related servo loops. The sensor head contains equipment that scans, detects, and tracks the infrared gradient. The electronics package contains the circuitry that provides azimuth, and search or track logic signals to the sensor head, and circuitry that provides attitude error signals to ACME and platform systems.

B. Horizon Sensor Development Tests - The McDonnell conducted system interface testing (ESTU and CTU) provided the opportunity to modify the spacecraft wiring and the vendor supplied subsystems before a major spacecraft problem developed. Problems which were uncovered in development testing included radio frequency interference (RFI) susceptibility, susceptibility to power transients, and interface incompatibility. RFI susceptibility required modifications to the spacecraft wiring (shielding) and to the horizon sensor. A suppressor network was incorporated into the horizon sensor to eliminate the effect of starting transients on the 26 VAC. The horizon sensor's output impedance was reduced to make it compatible with the loads.

C. Horizon Sensor Design Problems - Problems which were encountered in the qualification program and the development program were:

1. The horizon sensor output impedance in the ESTU was too high for total guidance and control system requirements. To correct this, the output circuits were redesigned to provide an output impedance of less than 100 ohms.

2. Because of the distributed line capacitance, the low output impedance units shorted out when connected to spacecraft cabling. Circuit

changes were incorporated into the horizon sensor to enable it to operate into actual spacecraft loads.

3. Starting transients caused by the spacecraft 26 VAC inverters overstressed the regulated power supply of the horizon sensor. A transient suppressor circuit was designed and incorporated into production hardware.

4. The pivots of the flexure-pivoted positor fractured during vibration in the predelivery acceptance test. Studies traced the failure to high magnification ratios in the head-to-positor interface. After the casting material and the pivot configuration had been modified, the vibration requirement was met.

5. Redesigned preamplifier and signal amplifier modules were incorporated into the infrared detection circuits.

6. A design analysis indicated that the extreme operating temperatures of the sensor head would cause interference between the bearings and shafts or seat. Sensor head bearings were changed to increase radial play, increase seat clearance, and serialize the bearings.

Attitude Display Group.

A. System Description - The attitude display group provides an all-attitude unambiguous display of spacecraft attitude using attitude information from the Gemini inertial platform and command information from the inertial platform, the digital computer, the rendezvous radar and the rate gyros. The attitude display group consists of the attitude director indicator and flight director controller.

The three-axis sphere of the attitude director indicator contains 360 degrees of freedom about three axes. It is driven by signals from synchro control transmitters in the inertial measurement unit.

The flight director needles of the attitude director indicator display attitude errors and rate commands. The information displayed by the needles is selected by the flight director controller and consists of the following:

1. Computer Data

a. Ascent Mode - Roll, pitch, and yaw attitude error signals indicative of Gemini launch vehicle attitude errors.

b. Catchup and Rendezvous Mode - Pitch and yaw attitude error signals. Roll signal indicates spacecraft deviation from platform zero roll reference.

c. Re-Entry Mode - Downrange and crossrange errors for pitch and yaw, and bank command or bank rate command for roll.

2. Radar Data - Radar-generated pitch and yaw error signals of the spacecraft with respect to the target.

3. Platform Data - Platform-generated roll, pitch, and yaw synchro attitude error signals.

4. Rate Gyro Data - Roll, pitch, and yaw spacecraft body rates. The Gemini attitude indicator was adapted with a minimum of modification from the Lear Siegler, Inc. Model 4060C Attitude Indicator. Over 1000 similar instruments were produced for the USAF, USN, and RCAF for use in X-15, CF104, and F-4 aircraft, thus demonstrating the indicator capability.

B. Attitude Display Group Development Tests - The attitude display was tested in ESTU and CTU; its interface compatibility with other systems was satisfactorily demonstrated. No changes were required as a result of development tests.

C. Attitude Display Group Design Problems - Because of random vibration test problems, stems and needles were changed to beryllium and supporting struts were added on meters. A new meter movement (giving five degree full-scale deflection) was added.

A scale factor or range switch which was added to the bezel of the indicator increased pitch and yaw range by a factor of two and roll range by a factor of three to give a wider display range capability.

During acceptance vibration tests, the needle structure of the attitude indicator was not sufficient to maintain needle displacement at the specified values. A modification reduced needle movement.

During qualification tests, oil leaked from the flight director controller modules. The problem was corrected by filling the modules with a silicone resinous material and by a new module sealing method.

Precision Power Supply.

A. Description - A special, precision frequency, low power inverter was designed and fabricated for Spacecraft 2 only. The special unit was required because instability in the 400 cps voltage source caused unstable phase shift resolver gimbal angle readouts. The problem was later corrected by a modification to the EM 1262 static power supply, but the interim type correction was immediately required. The interim unit was installed in the pressurized area of the spacecraft and tee-type cabling was employed to accommodate the installation. The unit consisted of audio filtering, countdown, regulation-inversion, and output filtering elements.

B. Precision Power Supply Development Tests and Design Problems - The precision power supply was not utilized in any developmental tests and no design problems were encountered.

GCS State-Of-Art Advances

In general, the guidance and control system required no breakthroughs in the state-of-art design concepts or manufacturing processes. However, the auxiliary tape memory concept can be considered an advancement in the field of highly reliable onboard bulk storage media for spacecraft computer operational programs. The ATM achieves data accuracies of 3×10^{14} bits per error by using triple redundant recorded data and a slow data transfer rate of 200 bits per sec.

GCS Qualification Program Summary

Qualification Status. - The guidance and control system was considered functionally qualified at the time of the Spacecraft 2 flight. In the brief summary of the qualification status for each system that follows, the term "resolved" indicates that tests were no longer outstanding, as a result of completion, deletion, or waiver.

A. IMU - All environmental tests except vibration were resolved prior to the Spacecraft 2 flight. Based on vibration test results on design approval test units, the IMU was considered acceptable for flight. The formal vibration requirement was completed in February 1965.

B. ACME - All environmental tests were resolved prior to the Spacecraft 2 flight. Final approval of the test results (documentation) was not given until after the flight but this was due to omission of data and did not reflect upon the qualification status.

C. Computer System - All tests for the manual data insertion unit and the computer were resolved prior to the Spacecraft 2 flight. The incremental velocity indicator lacked tests for decompression, temperature-altitude, fungus and high test pressure at the time of the Spacecraft 2 flight; all tests were resolved prior to the Spacecraft 3 flight. At the time of the Spacecraft 2 flight, the auxiliary computer power unit lacked salt fog testing, which was resolved prior to the Spacecraft 4 flight. All tests for the auxiliary tape memory were resolved prior to the Spacecraft 8 flight, the first mission in which the ATM was used.

D. Horizon Sensor - All environmental tests except high temperature, low temperature, humidity and salt fog were resolved. It was decided that these tests were not necessary for the Spacecraft 2 because of its short flight. All tests were completed by July 1965.

E. Attitude Display Group - All environmental tests were resolved prior to the Spacecraft 2 flight.

F. Precision Power Supply - All environmental tests were resolved prior to the Spacecraft 2 flight.

GCS Qualification Test Problems.

A. Inertial Measurement Unit

1. Humidity - After exposure to humidity, the IMU showed considerable

corrosion at all magnesium-to-aluminum interfaces. It was apparent that the IMU units would not pass salt sea atmosphere tests, either. The modifications recommended by the vendor to provide humidity and salt sea atmosphere integrity would have delayed the Gemini IMU schedule and would have increased costs. Therefore, McDonnell advised the NASA (TWX 306-16-8317 dated 7 December 1964) that no additional humidity or salt spray efforts would be expended on the IMU. This agreed with the NASA's request of 10 August 1964 via GP-54886.

2. Vibration - During design tests at a 6 g level, gimbal No. 3 changed reference (approximately 700 arc sec) two times. The IMU was then exposed to a lateral axis, ascent orientation, random vibration of 6.2 g RMS and gimbal No. 3 changed reference approximately 3.3 degrees during one min. The platform was then exposed to a special, unit level, acceptance type 6.2 g RMS vibration test; gimbal No. 3 changed reference approximately 600 arc sec during one min, confirming that the platform had degraded since formal acceptance testing. Use of the platform was discontinued when partial disassembly showed extensive damage to the rotary component stiffener and slip ring damage. A second platform was substituted and the IMU was subjected to DAT vibration in the lateral axis with the gimbals positioned to the ascent orientation. Inertial reference was lost during five min of 6.0 g's random vibration, resulting in an X-gyro drift rate which was equivalent to 180 degrees/hr. In order to evaluate the IMU, special tests were performed. Conclusions reached from these tests were:

a. The maximum random vibration level that the S/N 105/H-11 platform could take without loss of inertial reference was 5.0 g's.
b. Loss of reference was relatively independent of the platform gimbal orientation.
c. IMU performance was satisfactory (no loss of reference) when vibrated at levels up to 8.8 g's with the power spectral density minimized in the platform resonant frequency region.

McDonnell investigated the feasibility of lowering the spacecraft structural resonance in the platform mounting area to permit use of a notched spectrum as a realistic vibration environment. Dead weight was added to the platform to reduce the spacecraft structural resonant frequency in the platform mounting area.

Tantalum platform end covers were mounted on the test unit. Subsequent testing showed no mechanical problems or gimbal loss of reference, but the gyros showed excessive drift rates.

The platform was vibrated along each of three axes with and without the tantalum end plates at both ascent and acceptance test procedure gimbal angles. The results indicated mechanical integrity, no loss of reference, and no significant difference in performance between the weighted and unweighted platform. Subsequent tests revealed that structural resonance could be reduced by a three-point mount. A three-point configuration, without weights, was used on Spacecraft 5 and up.

B. ACME Qualification Test Problems

1. Vibration - The initial vibration of the ACE and OAME resulted in multiple failures which required structural redesign. Redesigned units were successfully subjected to vibration.

2. Hand Controller Potentiometer

a. Deadband Shifts - During the qualification vibration test, slight shifts occurred in the deadband tap locations. The deadband was increased from 6.0 degrees to 6.6 degrees.

b. Salt Fog Test Failure - Insulation resistance fell below the specified valve after salt fog exposure. The salt fog test was then conducted on two additional potentiometers with the terminal areas sealed. Both units corroded badly.

Changes were incorporated which enabled the potentiometer to pass the salt fog requirements.

C. Computer System Qualification Test Problems

1. Incremental Velocity Indicator

a. Random Vibration - After two vibration runs in each of the unit axes, the unit had an excessive leak rate. The unit was modified and was then successfully subjected to random vibration.

b. Temperature-Altitude - The unit failed to meet the leak rate requirements. The unit was twice repaired, retested and failed to meet leak rate requirements after decompression and temperature-altitude testing. The modified unit with a redesigned case-to-bezel seal was then successfully subjected to the test.

2. Manual Data Insertion Unit

a. Humidity Test - The manual data keyboard malfunctioned when using the keyboard to insert digits into the unit tester. The unit was returned to manufacturing and reworked. The unit was then retested in the humidity environment and passed all functional requirements.

3. Auxiliary Computer Power Unit

a. Operational Life Test - During the temperature-altitude operation life portion of the design approval test (DAT) the main power transistor in the regulator circuit avalanched, causing the battery to discharge. The circuit was redesigned for better operation at elevated temperatures and potting was added to aid heat conduction. After the modification, the ACPU successfully completed the test.

4. Auxiliary Tape Memory - During prequalification vibration testing of the engineering model No. 1 tape transport, the original flutter tolerance specification of 3% peak-to-peak at the orbit vibration level was not met. A study resulted in an increased flutter tolerance specification of 5%

peak-to-peak at a reduced orbit vibration level of 0.003 g^2/cps from 20 to 200 cps and 0.0015 g^2/cps from 200 to 500 cps with an overall acceleration level of one g RMS. Several tape transport configurations were used in the effort to reduce flutter to this acceptable limit. Brass flywheels and weights on the tape reels provided the best configuration and were installed in all flight units.

D. Horizon Sensor Qualification Test Problems

1. Vibration - Horizon sensor system (-307) consisting of a -17 sensor head and a -15 electronics package failed the vibration requirement. The electronic package and the sensor head were modified. The reworked system passed random vibration.

E. Attitude Display Group Qualification Test Problems

1. Attitude Display Indicator - The attitude display indicator was not subjected to a formal qualification test program because it was similar to the Lear Siegler, Inc. 4060C attitude indicator, which had been subjected to a comprehensive qualification test program for the USAF.

However, a random vibration test (8.6 g RMS), performed to verify that the indicator could meet Gemini standards, revealed a structural design deficiency in the roll gimbal support. Structural modifications were made to the unit. Additional vibration tests (6.5 g RMS, each axis for 15 min) then were performed successfully.

2. Flight Director Controller

a. Salt Fog Test - Salt fog caused out-of-specification outputs from the FDC. Investigation revealed salt residue on the terminal board and around module header holes. A silicone rubber gasket was added on top of the module mounting plate, thus sealing leaks under the module headers. The unit then passed the salt fog test.

b. Humidity - After the FDC failed humidity testing, the gasket material and fabrication procedures were changed. The retest was successful.

c. High Temperature Test - Investigation after testing revealed a leak in the power supply module caused by a break in the solder seal. A change in the module filling and sealing procedure enabled the unit to pass the test.

F. Precision Power Supply Qualification Testing - No problems were encountered with qualification tests. As an interim unit used only on Spacecraft 2, the PPS was not processed to all tests normally specified for cabin-mounted equipment. Upon satisfactory completion of high temperature testing, the PPS was accepted as qualified for a suborbital, nonmanned, flight.

Reliability Program Summary for GCS

The guidance and control reliability programs had as their objectives:

A. To achieve system designs capable of meeting or exceeding the system reliability requirement.

B. To produce system hardware capable of meeting these requirements under field operation.

Inertial Measurement Unit Reliability Program. - Overstress testing for the IMU was planned only for the inertial guidance system power supply and the system electronics. The inertial platform was omitted because the platform is temperature-controlled and not amenable to temperature overstress testing. In addition, no data was available to correlate vibration overstress with extended inertial component life data.

A. System Electronics Vibration - A system electronics package was submitted initially to an exploratory/quality verification test and then was subjected to overstress vibration. After completion of 15.1 g RMS exposure, two switching relays were inoperative.

A new relay module was installed and the system electronics package was vibrated at a 12.6 g RMS random level for 15 min in each of its three mutually perpendicular axes; the same relays failed again. The relays required a modification to the coil attachment. With modified relays, the systems electronics performed satisfactorily during overstress vibration.

B. System Electronics Temperature/Altitude - A system electronics package was submitted to overstress temperature/altitude testing. This test had no measurable effect on the operation of the system electronics.

C. IGS Power Supply Vibration - An EM 1262 static power supply was mated with a gimbal control electronic and subjected to overstress vibration. After 1.5 min of vibration at 9.7 g RMS, a noise with a 60-80 cycle frequency appeared on the static inverter DC and AC output voltages; it continued throughout the 10.2 g vibration period. Before starting the 10.6 g vibration, the input voltage was raised to 30 VDC. The 60-80 cycle noise was not present when the input voltage was maintained at the 30 VDC level. No other discrepancies occurred during this test. Inherent regulation instability, which was corrected in the EM 1262A SPS, caused the noise.

D. IGS Power Supply Temperature/Altitude - During the first turnon, with the altitude chamber at a pressure of 1.47×10^{-5} psia, the 400 and 1200 cycle outputs were out of regulation on both amplitude and frequency. Problem isolation revealed that a capacitor (large can) had faulted with an internal open lead. This capacitor presented a high impedance to the sync oscillator which distorted the switching output. This distortion resulted in overstress and damage to all pulse width regulator circuits. The capacitor failure was attributed to the extensive vibration that the power supply had previously undergone.

ACME System Reliability Program.

A. ACME Operating Life Tests - A combined mission simulation and operating life test was conducted on the ACME system for 1200 hr, of which more than 1000 hr were at a pressure of 1.0×10^{-8} psia. The performance of the system was satisfactory throughout the test.

The ACME hand controller potentiometer was subjected to a 500-hr operating life test at $80^{\circ}F$ and 6 psia in a 100% O_2 environment. A cycling rate of five cycles/min was maintained throughout the test. The results were satisfactory. Two ACME command excitation transformers were subject to a 500-hr operating life test at $160^{\circ}F$ wall temperature and 1.0×10^{-8} psia. The results were satisfactory.

B. ACME Overstress Tests - The ACME system passed the following overstress tests:

1. Combined Temperature - Altitude/Hi-Low Voltage

   a. Chamber pressure: 1.47×10^{-5} psia, maximum.
   b. Chamber shroud temperature: $165 \pm 5^{\circ}F$.
   c. Coldplate inlet temperature: $165 \pm 5^{\circ}F$ to $205 \pm 5^{\circ}F$ (ACE, OAME, PI), and $125 \pm 5^{\circ}F$ to $165 \pm 5^{\circ}F$ (RGP).
   d. Coolant flow rate: 50 lb/hr maximum.
   e. Supply voltage: 18.0 VDC to 30.5 VDC.

2. Voltage and Frequency Stability

   a. AC voltage: 21 VAC to 31 VAC
   b. AC frequency: 388 cps to 412 cps

3. Vibration - Although no formal overstress vibration tests were conducted, tests conducted early in the qualification program demonstrated that the system could withstand overstress vibration.

Two command excitation transformers were subjected to overstress tests which consisted of 12-1/2 cycles, each cycle lasting 20 hr, alternately at $-55^{\circ}C$ and $125^{\circ}C$ with units operating at 125% of rated load. During the 20-hr periods, the samples were cycled with power "on" for 40 min and power "off" for 80 min. At the conclusion of each 20-hr cycle, the transformers were removed from the test chamber and left deenergized at room ambient for not less than four hr prior to starting the next cycle. The results were satisfactory.

Computer System Reliability Program.

A. Computer Reliability Program - The spacecraft digital computer was subjected to a low temperature threshold test, a high temperature threshold test and a $200^{\circ}F$ re-entry profile evaluation.

1. Low Temperature Threshold Test - This test was performed at the following test conditions:

a. Internal chamber pressure: 1.47×10^{-5} psia.

b. Thermal shroud mean temperature: 0°F.

c. Computer coldplate inlet temperature: Varied from 60°F to 35°F in 5°F increments.

No computer malfunction or performance degradation was experienced during this test.

2. High Temperature Threshold Test - This test was conducted at the following test conditions:

a. Internal chamber pressure: 1.47×10^{-5} psia.

b. Thermal shroud mean temperature: 160°F.

c. Computer coldplate inlet temperature: Varied from 95°F to 120°F in 5°F increments.

A malfunction occurred during testing at a temperature of 110°F. The failure was attributed to power line fluctuations in test equipment, which resulted in the imposition of transients on the computer chassis. No computer malfunction was noted during subsequent testing above 110°F.

3. 200°F Re-Entry Profile Evaluation - This test was conducted at a chamber pressure of 1.47×10^{-5} psia and the shroud and coldplate inlet temperatures initially stabilized at 160°F and 88°F, respectively. The coolant flow through the coldplate was then halted and the mean temperature of the shroud was increased to 200°F. Simultaneously with the shroud temperature increase, the internal chamber pressure was increased to the micron range and maintained for the duration of the test.

A malfunction which occurred prior to the attainment of a mean shroud temperature of 200°F was attributed to a test equipment power line fluctuation. The test was repeated without computer malfunction.

B. Incremental Velocity Indicator Reliability Program

1. High Temperature Test - The powered-up unit was subjected to 120% of the design approval high temperature test level (192°F) for four hr. At the end of this period the unit was tested functionally. The test caused no damage to the unit.

2. Vibration Test - The IVI (power-on) was subjected successfully to 120% of the design approval vibration test level for 15 min in each axis. This was repeated in each axis.

3. Power Test - The unit was subjected successfully to 120% overvoltage (nominal operating) and 120% undervoltage for ten min. Thereafter a performance test was conducted with voltages maintained.

4. Pressure Test - The IVI was placed in a pressure chamber at a 2.0 atmosphere level. Then the chamber was depressurized suddenly to a one-atmosphere level, thus leaving a pressure differential of one atmosphere applied to the IVI case. The IVI was monitored for leakage and passed the test with no damage or deterioration of performance.

C. Manual Data Insertion Unit Reliability Program - Overstress testing on the manual data insertion unit was deleted on a unit basis. Reliability analysis stress-data was submitted on the MDIU on a component part basis. This data summarized and recommended stress levels and application information for the electronic components used in the MDIU and the IBM component part derating practice employed on the unit.

Horizon Sensor Reliability Program. - For contractual reasons, overstress testing was not conducted on a complete horizon sensor system nor was it performed on either of the two units of the system. However, all major parts and assemblies were individually subjected to various overstress tests. The tests were of two general categories: Module overstress tests and electrical-mechanical overstress tests.

Module tests consisted of the following, although all modules were not necessarily subjected to all tests:

A. High temperature storage.
B. High or low temperature and voltage overstress test and/or thermal shock.
C. Vibration (sinusoidal).
D. Transient noise or transient voltage.

The error amplifiers and the drive amplifier failed test B above and had to be redesigned. Retest results were satisfactory. Other discrepancies were minor and were either the result of parts which had not been burned in or were readily corrected by the addition of trim resistors, diode suppressors, etc.

Electrical-mechanical tests consisted of the following and all modules were not necessarily subject to the same test:

A. 1000-hr life test.
B. Temperature cycling.
C. Vibration (sinusoidal).

The only failure consisted of a torsion rod which failed the 1000-hr life test. A modified torsion rod passed the test.

Attitude Display Group Reliability Program. - Overstress testing on the attitude display group was confined to the flight director controller. Testing of this type had been previously accomplished on the attitude display indicator on other programs.

The following tests were performed to determine the effect of extreme environmental conditions on the performance of the controller:

A. High Temperature - The unit passed a test in which it was energized and subjected to a temperature of 210°F for a soak period of four hr.
B. Overpower - The controller was energized with nominal input voltage (26 VAC, 400 cps) and outputs were monitored. Input voltage was then increased to 30 VAC, 400 cps and maintained for 15 min, after which the controller

outputs were again monitored and the unit was inspected for damage. No damage was incurred by the unit.

The unit was subjected to a vibration test, which consisted of 110% design approval test level for 15 min in each of the three mutually perpendicular axes. The controller passed the vibration test.

GCS Quality Assurance

Inertial Measurement Unit. - The type of quality assurance test performed on elements of the IMU varied with items. The testing was as follows:

A. System Electronics and Gimbal Control Electronics - An engineering specification (ES) test was performed to set all adjustments and gains and to verify that overall unit-level performance was within specification. The ES test tolerances were generally less than those permitted in higher level tests.

A unit level acceptance test including vibration testing was performed with Government and McDonnell witnessing. Upon satisfactory completion of this test and accumulation of 100 hr of operation, a unit was considered flight worthy. If the units were integrated in a system and submitted to an IMU acceptance test, the 100-hr operate time applied upon completion of system level acceptance testing.

B. Static Power Supply Quality Assurance - The SPS was fabricated and tested by Engineered Magnetics Division of Gulton Industries. Prior to shipment from EM to Honeywell, each static power supply was tested, and the test was witnessed by Government and Honeywell representatives. At Honeywell, the SPS was processed to a receiving inspection test. This test was a duplicate of the predelivery test performed at EM. Each SPS was submitted to an individual vendor acceptance test which included vibration; Government and McDonnell witnessing was required. Upon satisfactory completion of this test the SPS was processed through a system level acceptance test as part of an IMU, or shipped for field use. In either case, 100 hr of operate time was required at shipment.

C. Inertial Platform Quality Assurance - Each platform was processed to a Honeywell ES test, including adjustment of gains, optical and mechanical alignment, and gimbal friction and balance checks. The platform level acceptance test depended upon whether the unit was to be integrated and tested at the IMU level or accepted and shipped as a platform. In either case, Government and McDonnell witnessing was required.

Pre-IMU acceptance testing consisted of unit level vibration, accelerometer loop gain checks, synchro error and static accuracy checks, cube to case alignment checks, synchro transmitter output checks, and gas leak and pressurization checks.

If a platform was to be accepted for shipment on the basis of IMU level tests, it received all testing performed in a pre-IMU test, and additionally was processed to testing designed to assure satisfactory system level performance. This testing included loop phasing checks, gyro open loop gain checks, and pre and post cool down inertial parameter checks. Essentially, the platform was evaluated for system performance, utilizing test station power sources and control loops. The gyro control loops do not duplicate system performance, and analog (in lieu of digital) accelerometer rebalance loops are employed.

The platform level acceptance test was intended primarily as a procedure for evaluating and accepting individual platforms returned from the field for rework where system performance has been verified previously.

D. IMU System Level Quality Assurance - Each platform accepted for Gemini was processed through an IMU acceptance test, in which all equipment of an IMU system was integrated and tested to verify flight worthiness. Except in rare instances (i.e., when a spare SPS, static power supply; SE, system electronics; or GCE, gimbal is required on short notice), items of an IMU were accepted as part of some IMU acceptance test procedures.

The system level test includes verification of acceptable inertial component parameters, gyro and accelerometer loop performance, caging and gyrocompassing loop performance, mode relay verification, and acceptability of all system interfaces.

Upon satisfactory completion of this test the IMU was ready for shipment to McDonnell for spacecraft installation or to IBM for IGS integration.

ACME System Quality Assurance. - The listing below applies to the attitude control electronics, orbit attitude and maneuvering electronics, rate gyro and power inverter packages and does not include any testing prior to the initial assembly of the individual package. The tests listed in A through L below were the vendor's engineering specification (ES) test profile and were performed in the order listed except the high temperature and low temperature tests were sometimes interchanged. Unless otherwise stated, the following tests were applicable to the four listed ACME packages.

A. Initial Room Temperature - Complete functional check of unit.

B. High Temperature (ACE, OAME, PI) - Four-hr soak at 160°F, nonoperating, followed by specific operational test while still at 160°F.

C. High Temperature (RGP) - Same as B above except the gyros were energized during the soak and the temperature was 120°F.

D. Operational Check - Perform specific operational test at room temperature.

E. Low Temperature (ACE, OAME, PI) - Four-hr soak at 0°F, nonoperating, followed by specific operational tests while still at 0°F.

F. Low Temperature (RGP) - Same as E above except the gyros were energized during the soak and the temperature is -10°F.

G. Operational Check - Same as D above.

H. Vibration - Three-axis random vibration test, one min/axis duration. The vibration spectrum was in keeping with Gemini requirements with an overall level of 6.2 g RMS. Limited functional tests were performed during the vibration.

I. Vibration - The RGP for Spacecraft 2 and the PI's for Spacecraft 2 and 3 were subjected to three-axis sinusoidal vibration instead of random vibration. The vibration consisted of a logarithmic scan of 15-2000-15 cps for 15 min/axis.

J. Operation Check - Same as D(Operational Check) on preceding page.

K. Remaining Burn-In - If the total operating hours to this point, plus the anticipated operational time for final room and predelivery acceptance tests did not equal or exceed 100 hr, the additional operation time was accrued at this point.

L. Final Room Temperature - A preselloff functional check of the unit.

M. Predelivery Acceptance (PDA) - This was the formal selloff tests, run to the appropriate McDonnell approved acceptance test procedure (ATP) at the vendor's facilities.

N. Preinstallation Acceptance (PIA) - Similar to M above but run at McDonnell to a McDonnell/NASA-approved service engineering department report (SEDR). Following the PIA test the equipment was installed in the spacecraft.

The following tests were performed on the two remaining items in the ACME, the hand controller potentiometer and the command excitation transformer:

A. Predelivery acceptance testing at the respective vendor's facilities.

B. Preinstallation acceptance testing per McDonnell SEDR 360 at McDonnell, St. Louis.

Computer System Quality Assurance.

A. Computer - The following test programs were conducted at IBM, Owego, New York, on the spacecraft digital computer prior to its installation into the spacecraft. These test programs are performed at IBM, Owego, New York.

1. Preacceptance Test - The preacceptance test consisted of a continuity test, an interface test, a diagnostic test, a memory evaluation test, a pretemperature test and a previbration test.
2. Acceptance Test - The acceptance test consisted of a vibration test, a temperature test, the interface measurements test and a diagnostic test.
3. IGS Integrated Acceptance Test - The integrated acceptance test consisted of an IGS interface test, IGS computer mode testing, system power, voltage, and frequency testing, an IMU/computer interface voltage transient test, and an auxiliary computer power unit test.

B. Incremental Velocity Indicator Quality Assurance - The following tests were performed on the incremental velocity indicator prior to its installation in the spacecraft:

1. Preacceptance test at Lear Siegler, Inc.
2. Predelivery acceptance test at Lear Siegler, Inc.
3. Acceptance test at IBM (for IBM acceptance).
4. Predelivery acceptance test at IBM (for McDonnell acceptance).
5. IGS integration test at IBM.

C. Manual Data Insertion Unit Quality Assurance - The following tests were performed on the manual data readout and manual data keyboard prior to their installation in the spacecraft:

1. Preacceptance test at IBM.
2. Acceptance test at IBM.
3. IGS integration test at IBM.

D. Auxiliary Computer Power Unit Quality Assurance - The ACPU was tested by the vendor per the McDonnell approved acceptance test procedure. If this testing was witnessed by McDonnell Engineering, no PIA testing was performed on the unit before its installation into the spacecraft. When the ATP was not witnessed by McDonnell Engineering, a PIA test was performed at McDonnell.

E. Auxiliary Tape Memory Quality Assurance - Raymond Engineering Laboratories conducted an IBM acceptance test on each unit. IBM then conducted a McDonnell acceptance test on each unit. The McDonnell acceptance test was the same as the IBM acceptance test but includes temperature/altitude and vibration tests.

Compatibility of the ATM with the spacecraft digital computer was verified by testing an engineering model and the first production model in an IGS system at IBM.

Horizon Sensor System Quality Assurance. - The vendor predelivery acceptance (PDA) test was performed to assure that the materials, workmanship, and performance of units programmed for delivery to McDonnell were not faulty and that the units had been manufactured to approved drawings and specifications.

The horizon sensor systems were tested for compliance with the detailed functional requirements of the McDonnell approved PDA test procedure. These tests included: physical inspection, weight, squib continuity check, power consumption, pitch and roll output null and scale factor, pitch and roll output cross coupling, loss-of-track, output time lag, temperature environment, and random vibration environment.

Attitude Display Group Quality Assurance. - The following tests were performed on the attitude display indicator and flight director controller prior to their installation in the spacecraft:

A. Preacceptance test at Lear Siegler, Inc.
B. Predelivery acceptance test at Lear Siegler, Inc.
C. Preinstallation acceptance test at McDonnell.

Precision Power Supply Quality Assurance.

A. Unit Level Tests - The precision power supply was processed to acceptance tests at both Engineered Magnetics and at Honeywell, Florida.

B. System Level Tests - The S/N 100 PPS was processed through the IBM acceptance test. During this phase the PPS was tested explicitly for voltage amplitude and frequency. In addition, it was verified as a result of gimbal angle accuracy tests.

The PPS went through the IGS integration tests at IBM prior to delivery to McDonnell for spacecraft installation.

GCS Equipment Evaluation

Inertial Measurement Unit Evaluation. - The inertial components constitute the time critical items that affect IMU performance. The gyros have an allowable life of 3000 hr and the accelerometers have an allowable life of 5000 hr.

For the gyros, the gravity sensitive and gravity insensitive drift parameters were measured and evaluated throughout the life of the platform. For the accelerometers the scale factor, bias, and input axis alignment were measured and evaluated.

ACME System Evaluation. - The following parameters were evaluated, by means of time history plots, as an indication of overall ACME performance:

A. ACE - Rate command mode switching levels (RCS and OAMS): re-entry and re-entry rate command mode switching levels; orbit mode pulse widths and frequencies; bias power supply voltages; RCS spike suppression.
B. OAMS - OAMS spike suppression.
C. PI - Output voltage and frequency.
D. RGP - Run up times; run down times; in phase nulls.

Computer System Evaluation.

A. Computer - The spacecraft digital computer contains no time critical parameters. The computer memory, however, was verified a minimum of four times prior to launch.

Analog voltage output deviation curves were plotted for each unit. A list of these curves is presented below.

1. Deviation of GLV signal versus theoretical value-computer pitch channel.
2. Deviation of GLV signal versus theoretical value-computer roll channel.
3. Deviation of GLV signal versus theoretical value-computer yaw channel.
4. Deviation of fine AC signal versus theoretical value-computer pitch channel.
5. Deviation of fine AC signal versus theoretical value-computer roll channel.
6. Deviation of fine AC signal versus theoretical value-computer yaw channel.
7. Deviation of coarse AC signal versus theoretical value-computer pitch channel.
8. Deviation of coarse AC signal versus theoretical value-computer roll channel.

9. Deviation of coarse AC signal versus theoretical value-computer yaw channel.

B. Auxiliary Computer Power Unit Evaluation - The battery in the ACPU should be charged after 14 days of nonoperation. The ATM, the IVI and the MDIU have no time critical parameters which are monitored prior to launch.

Horizon Sensor System Evaluation. - Selected horizon sensor system parameters were monitored after installation into the spacecraft, to insure unchanged operational performance. These parameters included pitch output, roll output, loss-of-track output, position detector output (primary system only), and azimuth overshoot output (primary system only). The attitude display group and precision power supply had no time critical parameters.

GCS Flight Performance

Gemini II. - During the ascent phase, the inertial guidance system indicated errors when compared to tracking data. The anomaly was evidenced as an out-of-tolerance downrange error and it appeared as erroneous X-accelerometer velocity signals near BECO and SECO. Tests confirmed that the errors were caused by saturation effects within the rebalance loop.

Gemini III Performance. - During the ascent phase of Gemini III the inertial guidance system indicated errors when compared with tracking data. The anomaly was evidenced as out-of-tolerance errors in the downrange and vertical velocities as indicated in TRW-STL report 4160-6047-TC000(NAS9-2938) - "Gemini GT-3 Inertial Guidance System Evaluation Trajectory Reconstruction."

Post-flight failure analysis revealed that the ascent velocity error was due to scale factor and bias instability in the X-accelerometer loop. Further examination related this instability to an interim modification which was employed on the X-accelerometer loop. For corrective action to the Gemini III and IV configurations, (See Gemini II Mission, page 36).

Post-flight analysis also revealed a gyro problem which caused pitch axis drift. This problem was caused by a defective header seal which allowed fluorolube to escape and subsequently allowed air to be drawn into the gyro. When the acceleration vector changed direction relative to the gyro, such as at 180 sec in the Gemini III mission, the air bubbles caused torques which resulted in transient drift. The defective seal was considered to be an isolated case and no design changes were implemented. However, a tilt test was incorporated at the vendor's facilities and at Cape Kennedy to insure detection of this problem should it occur on subsequent units.

For additional information, refer to McDonnell report B984, "GT-2 and GT-3 IGS Ascent Anomaly Investigation (U)," and McDonnell guidance and control mechanics Gemini design note 298.

The horizon sensor system experienced many losses of track during the Gemini III mission. However, it was able to align the platform and to provide hor scan mode control without difficulty. Post-flight analysis of telemetry records revealed that the Gemini horizon sensors lost track 382 times during the mission. The total accumulated loss of track time was 366.1 sec out of the 4 hr, 28 min and 3 sec between initial lock-on and deenergization of the horizon sensors. Of that total, 291 losses of track (293.7 sec) were explained through the correlation of excessive spacecraft attitude, switching between sensors, sunset, and sunrise. Correlation studies were undertaken in an attempt to relate clouds and special thruster firings to losses-of-track, but the results were inconclusive. Testing which was later conducted on a horizon sensor system proved that the sun will cause losses of track when it is located between the earth's horizon and approximately ten degrees above the horizon, if the sun is within the azimuth coverage of the sensor. To isolate additional causes of horizon sensor losses-of-track, plans were made to add special TM parameters to the horizon sensor system on Spacecraft 4.

Gemini IV Performance. - During the flight the astronauts experienced difficulty in turning the computer off using the computer ON/OFF switch. Removal of the IGS power supply 28 V input failed to shutdown the computer. Special tests confirmed that the computer was not operating properly and the flight continued without the use of the computer.

Post-flight analyses to determine the cause of the anomaly included analyses of the telemetry data and of the voice transmissions. Post-flight testing was of considerable magnitude and at considerable expense, and comprised unit tests on the computer, a system test which employed an ACPU and a power supply, and a disassembly and analysis of each of the components which might have been related to the failure. All tests and analyses failed to reveal the cause of the failure.

Post-flight tests of the computer indicated that an electrical arc between the computer 25 volt regulator transistor terminal and the computer case caused an overload condition in the IGS power supply and failed a transistor component in the IGS supply. This problem was unrelated to the TURN-OFF failure.

Detailed information regarding this failure may be found in the following reports:

A. McDonnell report E193, "Final Report - Gemini Spacecraft 4 Post-Flight Failure Investigation IGS Static Power Supply Re-Entry Failure."

B. IBM No. 65-928-93, "Final Report on the Post-Flight GT-4 Failure Analysis."

C. McDonnell report E106, "Final Report - Gemini Spacecraft 4 Post-Flight Failure Investigation Compute Power Sequencing Anomaly."

The special horizon sensor TM post-flight data analysis of Gemini IV confirmed that certain cloud conditions can cause horizon sensor output variations. These data confirmed the Gemini III post-flight analysis.

Gemini V Performance. - The performance accuracy of the ACME platform mode was questioned by the crew. A comprehensive post-flight test program indicated that the platform mode operated satisfactorily during the mission.

The crew also reported an anomaly associated with the flight director indicators of the attitude display group. With the range switch in the LO position the downrange (pitch) needle was displaced full-scale, but with the range switch in HI the needle was displaced less than half-scale. Post-flight testing revealed that a difference existed between the electrical full-scale limits and the meter mechanical limits which accounted for the apparent discrepancy.

The primary horizon sensor became inoperative during the Gemini V flight. The anomaly was isolated to the sensor head, which is jettisoned prior to re-entry. Post-flight analysis of the telemetry data revealed that the yoke azimuth motion became uncontrolled.

Gemini VI Performance. - The astronauts reported that the downrange (pitch) needles on both flight director indicators deflect 2-1/2 needle widths below the miniature airplane dot after retrofire (the needles should have been centered on the dot). Post-flight testing was performed on both attitude display groups, the digital computer, and the No. 2 instrumentation package. The reported anomaly was never duplicated.

Gemini VII Performance. - It was noted that the removal of the AC power from the ACME caused all eight OAMS thrusters to fire for a short (less than 300 ms) period. Post-flight analysis proved that this was characteristic of the system and that it does not affect system performance.

Gemini VIII Performance. - Shortly after docking with the Agena Target Vehicle the spacecraft experienced an uncommanded firing of OAMS thruster No. 8. This might have been caused by a failure in the OAME package. However, post-flight analysis of the telemetry data indicated that a failure of any of the critical OAME components did not fit all of the failure modes. Therefore, it was concluded that the OAME package did not fail.

The astronauts reported that their first attempt to use the ATM on Gemini VIII was unsuccessful because the ATM run light and the IVI's did not respond as had been expected. The astronauts had observed this same anomaly in the mission trainer. Post-flight analysis has not positively identified the cause of the anomaly but the most probable cause is that the MDIU cancel button was inadvertently depressed while the insert button was depressed.

Gemini IX Performance. - Early in the mission the crew experienced an anomaly which caused the same effects as a continuous START COMP discrete. Subsequently the problem cleared itself.

Post-flight tests were conducted on the computer, the COMP-START switch and the Manual RETRO switch (connected in parallel with the COMP-START switch). No indications were found as to the possible cause of the anomaly.

Gemini X, XI, and XII. - There were no significant equipment changes or anomalies.

PROPULSION SYSTEMS

There are three propulsion systems on Gemini. Two of them, the orbit attitude and maneuver system (OAMS) and the re-entry control system (RCS), are liquid propellant systems. The retrograde rocket system is a solid propellant system.

Orbit Attitude and Maneuver System

Design Concepts. - During the orbit phase of the mission the OAMS provides (1) attitude control torques about the spacecraft pitch, yaw, and roll axes, and (2) translational thrust along these same axes. The system is located in the adapter section of the spacecraft, as shown in Fig. 17.

A. Thrust Engine Design - Attitude control is obtained by firing selected pairs of the eight thrust engines arranged around the spacecraft periphery. Each chamber fires in a fixed direction with a constant thrust level, nominally 23.5 lb. The spacecraft is maneuvered by firing thrust chambers of higher output (two of 79 lb output and six of 94.5 lb) which are oriented so that the thrust acts through the vehicle center of gravity. Two of the 94.5 lb chambers fire together to provide forward thrust, while the other four fire radially to yield up and down or side-to-side translation. Aft thrust is obtained by firing the pair of 79 lb thrust chambers, which are pointed forward but are canted obliquely to prevent the exhaust plume from damaging the spacecraft skin.

The attitude control chambers are fired in short, variable length pulses. In order to achieve precision in attitude control, very short pulse length capability is required. High response propellant valves and small injector volumes meet this requirement. The maneuver chambers are operated for comparatively longer periods and therefore the high response is not required.

B. Thrust Chamber Temperature Control - The thrust chambers are installed below the spacecraft skin, with the exhaust nozzles terminating flush with the vehicle exterior. The submerged installation, adopted because of aerodynamic considerations required that the temperature of the exterior walls of the thrust chambers be limited to avoid damage to neighboring systems and structures. This was achieved by using ablative material in the construction of chamber walls. Heat is dissipated by being absorbed into the chamber wall where the ablative material chars and releases gaseous products back into the chamber and out the exhaust nozzle.

C. Propellant Design Concepts - Tank propellant weight is kept at a minimum by the use of a high energy bipropellant, monomethyl hydrazine, as fuel and nitrogen tetroxide as oxidizer. These propellants are hypergolic with each other; therefore no ignition provision is required. The propellants have additional advantages: they do not require special conditioning, such as refrigeration, for storage, and they do not degrade with time. Disadvantages

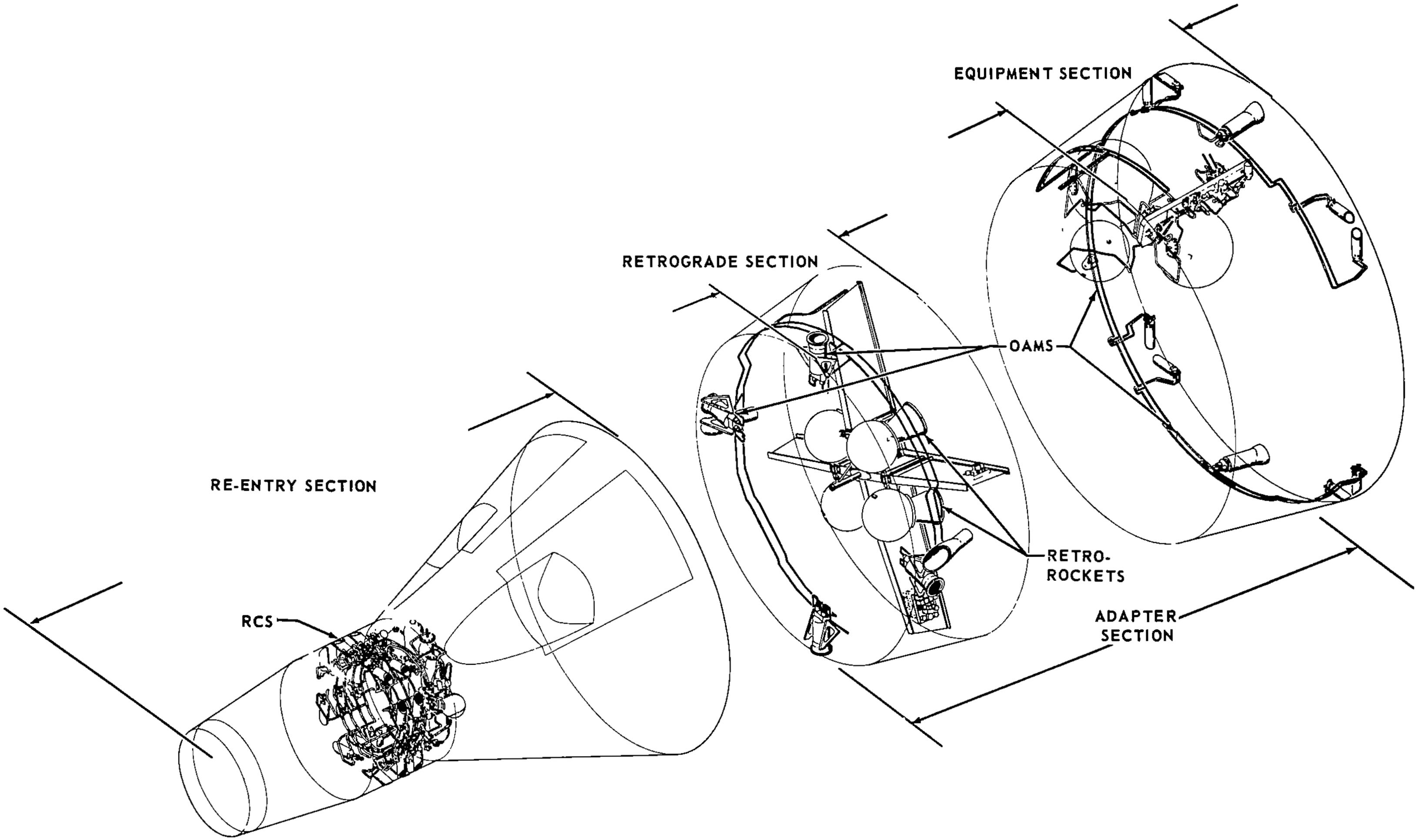


FIGURE 17 PROPULSION SYSTEMS-OAMS, RCS, RETRO
LOCATION IN SPACECRAFT

also exist: the propellants are highly toxic and, particularly in the case of the oxidizer, corrosive.

In order to combat corrosion, all parts exposed to propellants are made from corrosion resistant steel, except for the elastomeric materials, in which Teflon emerged as the only suitable material. Toxicity is dealt with by strict control of leakage so that the propellants are securely contained within the system. This containment is insured by welding all closures in the various components (with a very few exceptions) and by using brazed joints (as opposed to threaded connections) between components. Access ports needed for servicing and ground testing cannot be welded, but are double or triple sealed against leakage. In addition to control of external leakage, the propellants are isolated in their storage tanks until the system is activated. In this way, potential leakage locations are held to a minimum prior to system use. Such techniques have resulted in a system with extraordinarily low leakage characteristics.

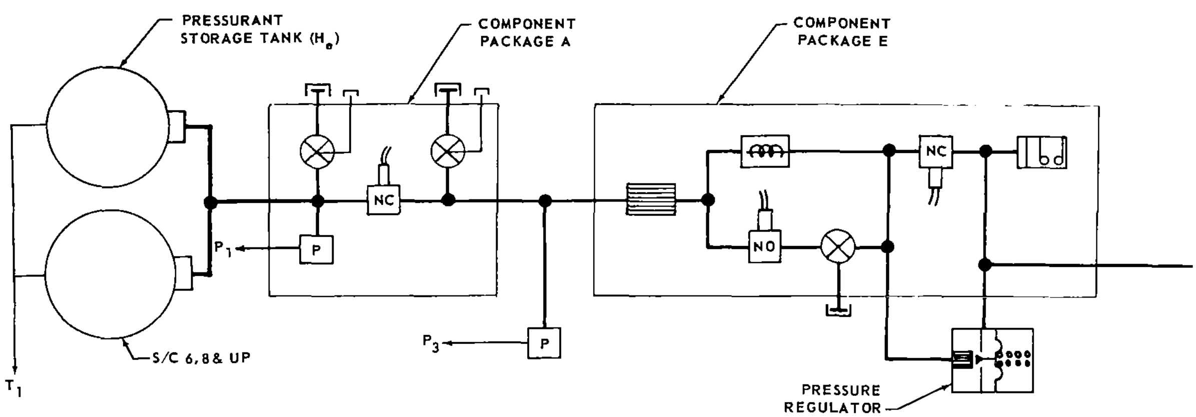
The propellants are stored in spherical tanks of all-welded titanium. In the absence of gravity the propellant is expelled by pressurant gas acting upon the outside surface of bladders which contain the propellant. This pressurant gas (helium stored at high pressure) flows through a regulator and a series of check valves before it is introduced into the propellant tanks. As in the case of the propellant, the pressurant is stored behind isolation valves to reduce potential leakage locations. The pressure regulation is redundant; if the primary regulator fails, the crew can regulate the system feed pressure manually by switches and valves.

OAMS Function. - The orbit attitude and maneuver system is arranged schematically as shown in Fig. 18.

A. Pressurant Storage - Pressurant helium is stored at a nominal 3000 psi in spherical titanium tanks, isolated before use by a cartridge-actuated valve in the component package A, immediately downstream of the pressurant tanks. A manual access valve in this same package provides a means of pressurant loading. The gas pressure, called "source pressure," is monitored by a pressure transducer which provides inputs to a cabin display and to ground telemetry. This transducer is upstream of the cartridge valve.

B. Pressure Regulator - Upon activation of the system by firing cartridge valves, gas flows unimpeded through component package E to the pressure regulator, which reduces the pressure to 295 psig, the system feed pressure. Malfunction of the regulator, so that it is unable to limit the downstream pressure, is detected by a pressure switch in package E. When this occurs, a normally-open cartridge valve, also in package E, is fired to interrupt the supply of high pressure gas to the regulator inlet. The crew may repressurize the regulator inlet by opening a solenoid valve which bypasses the now closed cartridge valve. If the regulator failure blocks the pressurant gas flow, a regulator bypass valve (cartridge-actuated) can be fired by the crew and pressure can be regulated by crew operation of the same solenoid valve. The regulator override equipment has all been collected into package E.

PRESSURIZING GROUP



NOTE: 1 ALL COMPONENTS EFFECTIVE ON S/C 2–12 UNLESS OTHERWISE NOTED.

2 NOT ON S/C 3

LEGEND

TCA	THRUST CHAMBER ASSEMBLY
	CONNECTED
	NOT CONNECTED
T_1, T_2 & T_3	TEMPERATURE (TRANSMITTED TO PROPELLANT TEMPERATURE AND PRESSURE INDICATOR OR INSTRUMENT PANEL) (T_1 ALSO TRANSMITTED TO OAMS PROPELLANT QUANTITY INDICATOR OR INSTRUMENT PANEL)
P_1 & P_2	PRESSURE (TRANSMITTED TO PROP. TEMP. AND PRESSURE INDICATOR ON INSTRUMENT PANEL)
P_3	PRESSURE (TRANSMITTED TO OAMS PROPELLANT QUANTITY INDICATOR ON INSTRUMENT PANEL)
	MANUAL VALVE
NC	CARTRIDGE-ACTUATED VALVE (NORMALLY CLOSED)
NO	CARTRIDGE -ACTUATED VALVE (NORMALLY OPEN)
P	PRESSURE TRANSDUCER
R	RELIEF VALVE
	SOLENOID VALVE (NORMALLY CLOSED)
	PRESSURE SWITCH
	BURST DIAPHRAGM
	CHECK VALVE
	FILTER
	GROUND TEST CONNECTION
	PROPELLANT SHUTOFF VALVE (MOTOR OPERATED)
	23 LB. THRUST TCA (YAW, PITCH & ROLL)
	95 LB. THRUST TCA (TRANSLATE FORWARD)
	PROPELLANT LINE CUTTER-SEALER ASSEMBLY
	95 LB. THRUST TCA (TRANSLATE RIGHT, LEFT, UP, OR DOWN)
	79 LB. THRUST TCA (TRANSLATE AFT)

FIGURE 18 ORBIT ATTITUDE AND MANEUVER SYSTEM SCHEMATIC

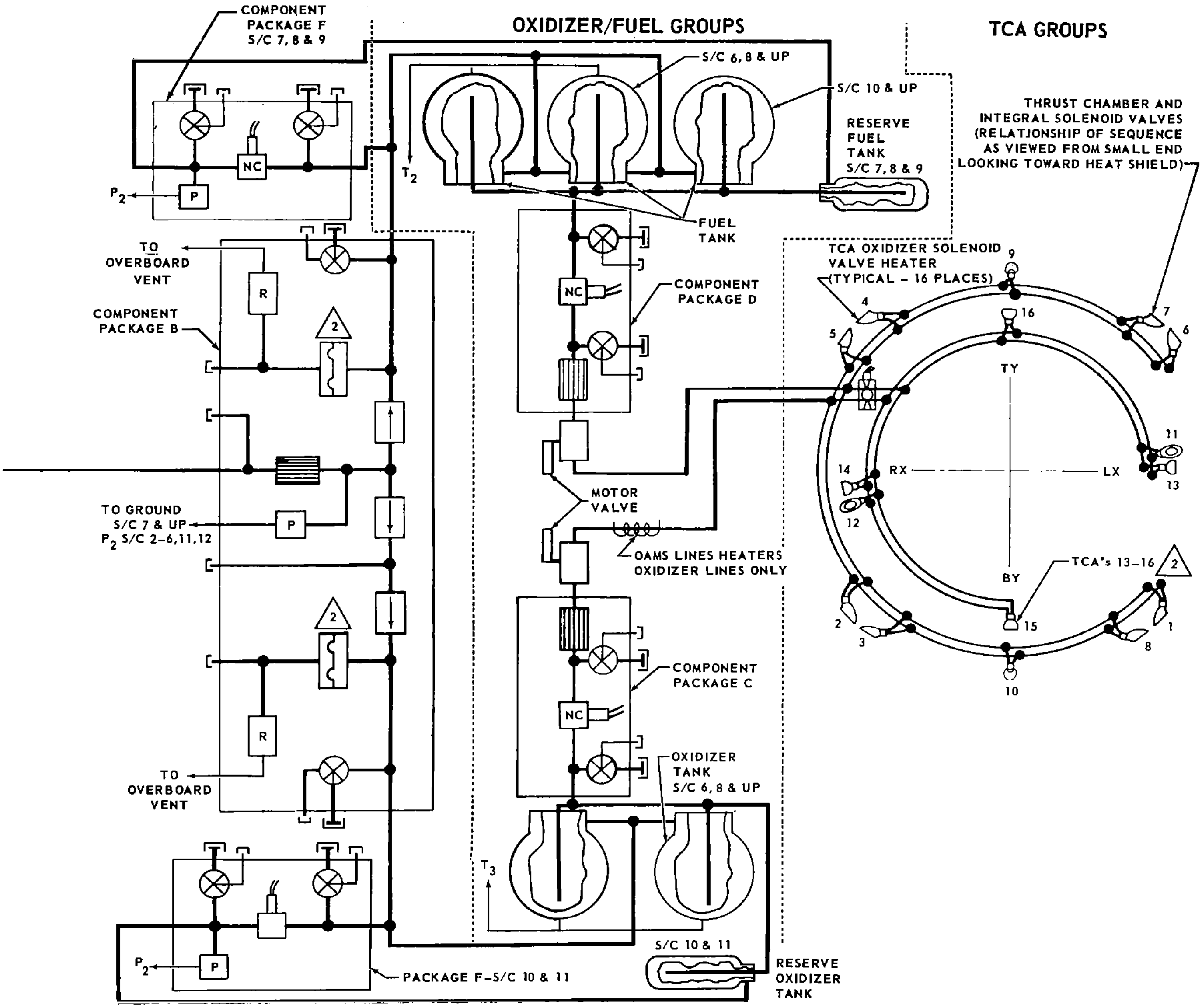


FIGURE 18 ORBIT ATTITUDE AND MANEUVER SYSTEM SCHEMATIC (Continued)

C. Pressure Transducer and Relief Valves - Pressurant emerging from the pressure regulator flows through component package B to the propellant tanks. In package B the flow streams divide and pass through check valves which isolate the fuel and oxidizer gas systems from each other. A pressure transducer in this package senses regulated pressure for cabin display and telemetry. Relief valves on both the oxidizer and fuel sides of the package prevent propellant overpressurization. These relief valves are isolated from the helium system by burst diaphragms which rupture at a pressure approximately equal to the relief valve cracking pressure. Thus the system is hermetically sealed until a diaphragm ruptures and no relief valve leakage is possible until an overpressure condition actually exists.

D. Propulsion Tank Operation - The pressure from the gas system is applied to the liquids by means of an intervening expulsion bladder, made of two plies of Teflon for the oxidizer tanks and three plies of Teflon for the fuel tanks. The liquids are isolated within these tanks by a cartridge valve located in component package C on the oxidizer side and in component package D on the fuel side. These packages also contain the manual valves needed for servicing and testing of the tanks and the downstream system components. Motor operated valves are located in the main feedline and are used, if needed, to shut off flow to the thrust chamber assemblies (TCA's). The TCA's have fast-acting solenoid valves in the feedline for each propellant. These valves can be opened to permit the propellants to flow to the thrust chamber injector; at this point they are fed into a combustion chamber where they ignite hypergolically. The combustion products are then expanded through a converging-diverging nozzle to provide reaction thrust. Flow rates are controlled by fixed orifices in the propellant valve inlets.

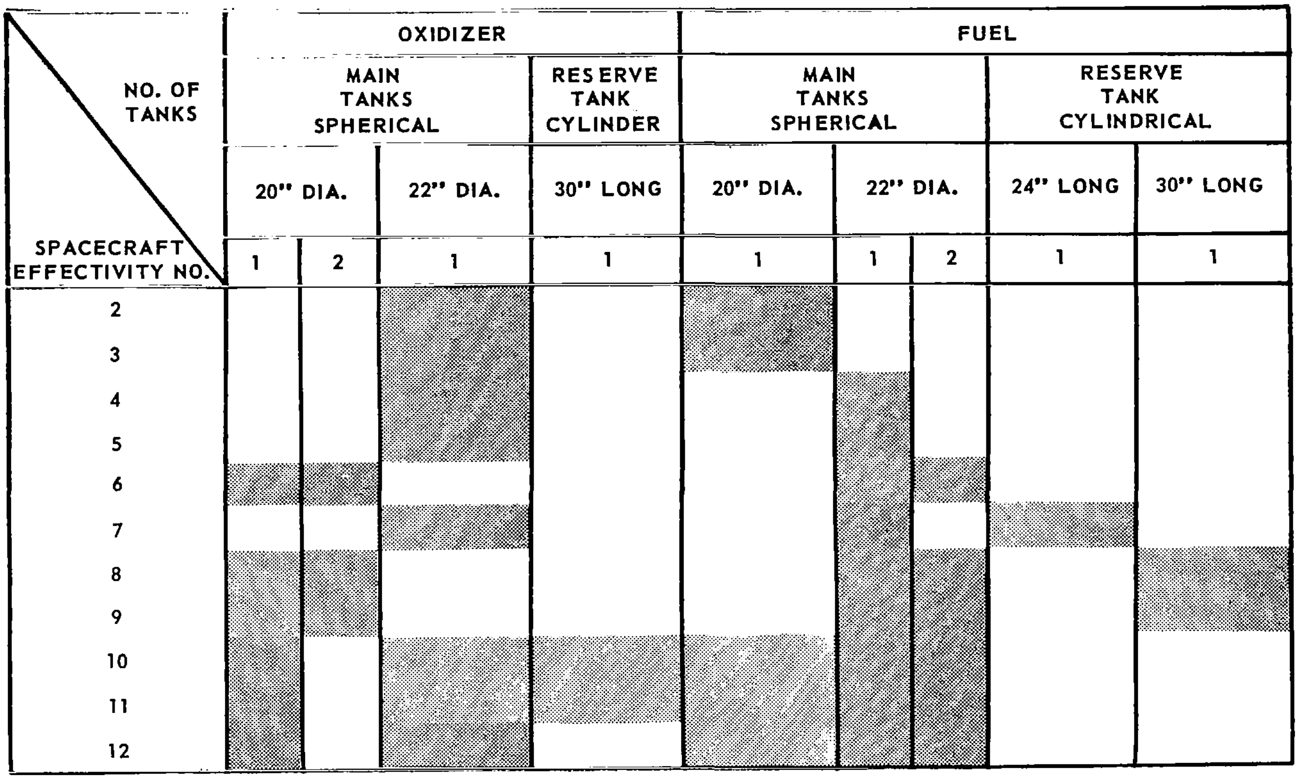
Filters are distributed throughout the gas and liquid systems to protect sensitive elements from particulate contamination. The oxidizer, nitrogen tetroxide, has a comparatively high freezing point, 12°F, and heaters are installed on the TCA oxidizer valves and the oxidizer feedlines to prevent freezing.

E. Measuring Remaining Propellant - The amount of propellant remaining in the OAMS is estimated by measuring the pressures and temperatures of the pressurant gas in the high pressure and regulated pressure regions of the gas system. By knowing the initial quantity of pressurant and by measuring subsequent temperatures and pressures, the principle of conservation of mass in the gas system can be applied to compute the gas volume increase in the propellant tanks. From this result the propellant volume and mass change can be derived. Source temperature and pressure signals are combined to provide a propellant quantity cabin readout. This indication can be corrected, using regulated pressure and temperature values, to refine the quantity estimate.

Throughout the Gemini program the trend has been to increase the available propellant in the OAMS. The number of tanks has ranged from two to six. The tankage combinations and their effectivities are indicated in Fig. 18 and summarized in Table 6.

F. Reserve Tank - On later spacecraft (7 through 9), a small, cylindrical reserve tank and an F package were added to the system. The tank provides a

TABLE 6 GEMINI OAMS PROPELLANT TANKAGE



known, small amount of fuel which remains available after the other fuel tanks have been emptied. The F package indicates to the crew when the main propellant supply has been depleted and withholds the pressurant from the reserve tank until the crew fires the cartridge valve in the package. Prior to the depletion of the main fuel tanks, the F package pressure transducer reads the system regulated pressure, which is imposed indirectly by the liquid fuel system upon the pressurized gas volume in the reserve tank ullage. When depletion of the main tank occurs, fuel is then expelled from the reserve tank under the trapped ullage pressure, which decreases rapidly as the reserve fuel is used. The crew can identify this rapid drop in the regulated pressure and can fire the cartridge valve to provide the normal system feed pressure of 295 psig. On Spacecraft 10 and 11, the reserve tank and F package were moved to the oxidizer system when a third fuel tank was added. The reserve tank and F package were deleted on Spacecraft 12 to reduce spacecraft weight.

Re-entry Control System

Design Concept. - The re-entry control system (RCS) provides for attitude control torques about the spacecraft pitch, yaw and roll axes after separation of the equipment section of the adapter, during the firing of the retrograde rockets, and during re-entry. Eight thrust chambers each rated at 23.5 lb thrust supply the output of each of two independent and completely redundant rocket engine systems. The two systems are in the re-entry section of the vehicle, as illustrated in Fig. 17. The eight TCA's of each system are arranged, as in the OAMS, about the periphery of the vehicle.

The RCS is similar in design concept to the OAMS in that the TCA's are fired in pairs and are submerged beneath the spacecraft skin. They are ablatively cooled, pulse-operated, and are fixed in thrust level and in thrust vector location. The same propellants used in the OAMS, monomethyl hydrazine and nitrogen tetroxide, are used in the RCS, but nitrogen is substituted for helium as the pressurant to minimize leakage. No regulator redundancy is employed beyond that afforded by the duplication of systems. System reliability is maintained by the isolation of propellants and pressurant behind cartridge-actuated valves until shortly before re-entry, thereby minimizing the effects of leakage and propellant exposure. The re-entry phase of the mission is relatively short and the quantity of propellant is less than in the OAMS. However, either of the two RCS's is fully capable of meeting re-entry requirements.

RCS System Function. - The system is arranged schematically as shown in Fig. 19. The function of the RCS is identical to the function of the OAMS, (OAMS Function, page 141) with the following exceptions.

A. Since redundancy in pressure regulation is not provided, no component package E is found in the RCS.

B. Reserve propellant provisions are not provided, hence there is no component package F.

C. There are no maneuver TCA's in the RCS.

D. Each propellant tank is cylindrical and has approximately one-tenth the capacity of the spherical OAMS propellant tank. Both the fuel and the oxidizer bladder in the RCS are made of two plies of Teflon. Only one fuel, oxidizer, and pressurant tank is provided in each RCS.

E. Electrical heaters are provided on the TCA oxidizer valves and on the component packages C and D, but not on the propellant feedlines.

RCS and OAMS Development Program. - The development programs on both the OAMS and RCS (See Table 7 and Table 8) were run to permit evolution of the various designs to the point at which they could survive the DAT (design approval tests) planned. Numerous problems were encountered and solved in the course of this evolution. In general, development problems were concerned with leakage, life capability and vibration resistance. These problems usually yielded to straightforward solutions. In addition to these, however, there were other more sophisticated solutions, some of which are cited below.

A. TCA Development - Early TCA development was aimed at achieving high performance (I_{sp}) and long life. Problems were encountered with delamination of the ablative chamber, throat erosion, and ablative material glass migration. These problems were not fully solved on the early spacecraft TCA, although the TCA's were fully adequate to meet the requirements of the early missions. A reconfiguration was introduced in which the mixture ratio and performance were reduced and the life span was greatly increased. At this time other improvements were introduced, such as ablative material with differently oriented wrapping and an attachment of the TCA valves to the injector, which

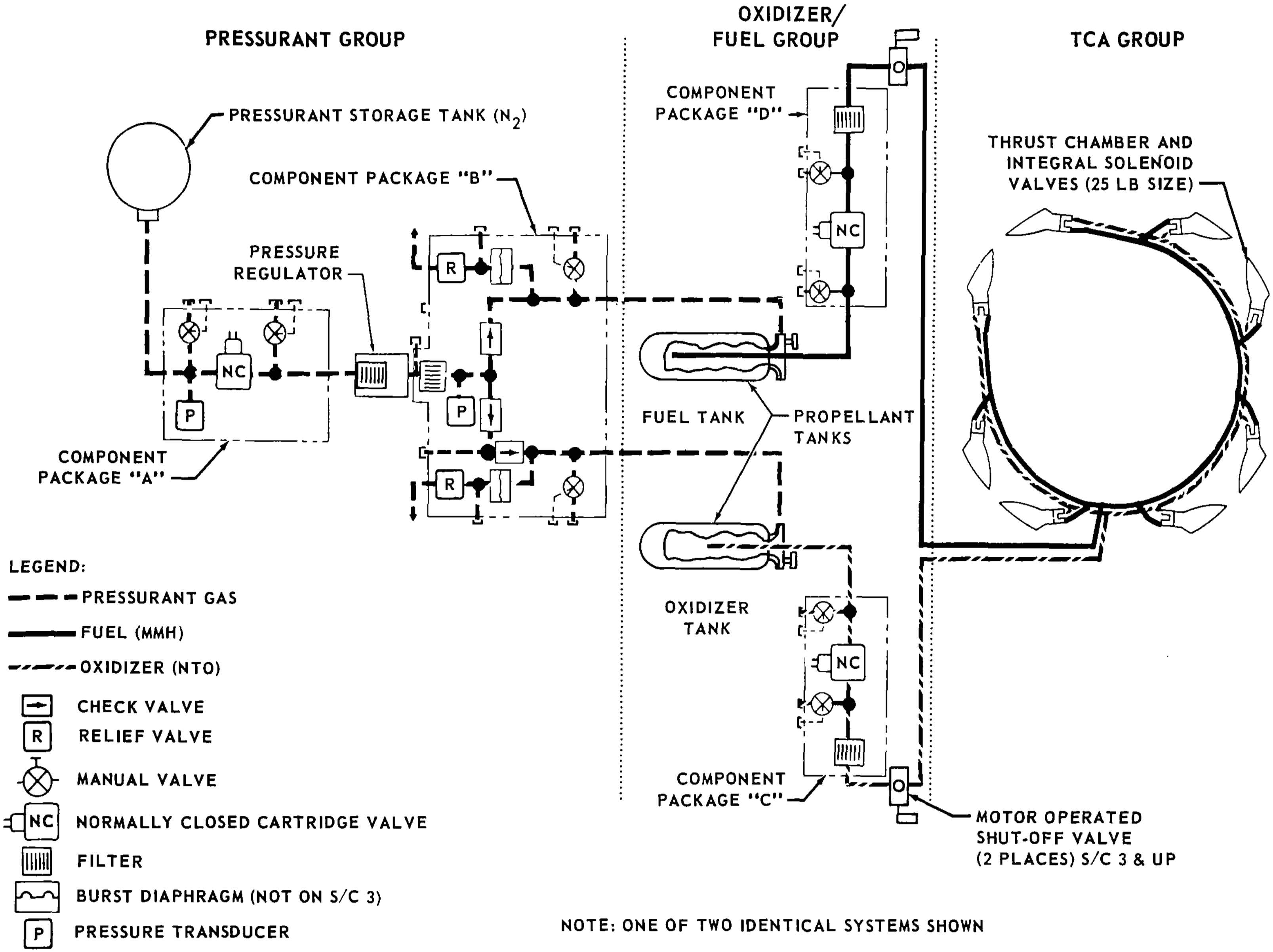


FIGURE 19 REENTRY CONTROL SYSTEM (RCS) SCHEMATIC

TABLE 7 DEVELOPMENT PROGRAM – PROPULSION SYSTEMS – OAMS, RCS

ITEM	TESTS PERFORMED	SYSTEM
A PACKAGE	1, 2, 3, 4, 5, 10, 11	OAMS–RCS
B PACKAGE	1, 2, 3, 4, 5, 6, 7, 11	OAMS–RCS
C PACKAGE	1, 2, 5, 6, 7, 8, 9, 10, 11	OAMS–RCS
D PACKAGE	1, 2, 5, 6, 7, 8, 9, 10, 11	OAMS–RCS
E PACKAGE	1, 2, 3, 4, 5, 6, 9, 10, 11, 12	OAMS
REGULATOR	1, 2, 3, 4, 5, 6, 7, 8, 9, 11, 13, 22, 31, 32	OAMS–RCS
MOTOR SHUT-OFF VALVE	1, 2, 3, 4, 5, 6, 8, 9, 11, 12, 13	OAMS–RCS
PRESSURANT TANKS	1, 5, 7, 9, 11	OAMS–RCS
PROPELLANT TANKS	1, 2, 3, 4, 6, 7, 9, 11, 15, 16, 17	OAMS–RCS
CARTRIDGES	1, 3, 4, 5, 8, 11, 12, 13, 22, 25	OAMS–RCS
CARTRIDGE VALVES	1, 3, 4, 5, 8, 9, 11, 12	OAMS–RCS
PRESS-TEMP INDICATOR	1, 3, 4, 12	OAMS–RCS
THRUST CHAMBER ASSY – 25 LB	1, 4, 11, 19, 20, 27, 28, 35, 36, 37, 38	RCS
THRUST CHAMBER ASSY – 25 LB	3, 4, 11, 20, 26, 28, 29, 35, 36, 38, 39	OAMS
THRUST CHAMBER ASSY – 85 LB	1, 3, 4, 11, 20, 28, 33, 36, 37	OAMS
THRUST CHAMBER ASSY – 100 LB	1, 4, 26, 27, 28, 29, 37	OAMS
OAMS SYSTEM	11, 14, 21, 23, 24, 30	OAMS
RCS SYSTEM	11, 14, 18, 19, 20, 21, 23, 24	RCS

LEGEND:
1. VIBRATION
2. SERVICE LIFE
3. HIGH TEMP
4. LOW TEMP
5. MECH SHOCK
6. COMPATIBILITY
7. ACCELERATION
8. SALT SPRAY
9. BURST
10. CARTRIDGE FIRING
11. AMB FUNCTIONAL
12. ELECTRICAL
13. HUMIDITY
14. FILL, DRAIN, CLEAN
15. SLOSH
16. BLADDER LIFE
17. EXPULSION EFFICIENCY
18. MDC-LOW TEMP
19. MDC-LOW TEMP/ALT
20. MDC-HIGH TEMP/ALT
21. OVERTEST ELEC
22. ALTITUDE
23. HOT FIRE
24. SKIN HEATING
25. AUTO IGNITION
26. INVERSION
27. OFF-MIXTURE RATIO
28. PULSING
29. STUCK VALVE
30. MAN REGULATION
31. SAND, DUST
32. CONTAMINATION
33. TEMP SHOCK
34. BASE HEATING
35. MDC-ALTITUDE
36. THRUST VECTOR
37. LOW SUPPLY PRESS
38. PULSE FATIGUE
39. WALL TEMP

TABLE 8 DEVELOPMENT TEST SUMMARY
PROPULSION SYSTEMS – OAMS, RCS, RETRO

T – TEST PERFORMED
Δ – TEST NOT REQUIRED

COMPONENTS	MAJOR ENVIRONMENTS								
	HIGH-LOW/TEMPERATURE	VIBRATION	SHOCK	ACCELERATION	FULL FUNCTION	LIFE	COMPATIBILITY	SLOSH	TEMPERATURE-ALTITUDE
A, B, C, D, AND E PACKAGES	T	T	T	T	T	T	T	Δ	Δ
REGULATOR; MOSOV	T	T	T	T	T	T	T	Δ	T
PROPELLANT TANKS	T	T	Δ	T	T	T	T	T	Δ
THRUST CHAMBERS	T	T	T	Δ	T	T	T	Δ	T
CARTRIDGE; RETRO ROCKET ASSEMBLY	T	T	T	T	T	Δ	Δ	Δ	T
PRESSURANT-TEMPERATURE INDICATOR	T	T	Δ	Δ	T	Δ	Δ	Δ	Δ
OAMS	T	Δ	Δ	Δ	T	T	Δ	Δ	Δ
RCS	T	Δ	Δ	Δ	T	T	Δ	Δ	T

was reconfigured to improve producibility. Also the valve mounts were strengthened and the valve poppet clearance was increased to improve contamination resistance. The life of the maneuver TCA's, (the 79 and 94.5 lb TCA's) was increased by boundary layer cooling, in which fuel, in addition to that injected for primary combustion, was sprayed along the chamber walls for cooling and subsequent secondary combustion. The effectivities of the TCA performance changes are shown in Table 9.

TABLE 9 CHANGE HISTORY OF TCA PERFORMANCE

EFFECTIVITY	THRUST - LB				MIXTURE RATIO - O/F					LIFE (IN-SPEC/FAILURE) - SEC					I_{SP} (NOM/MIN) - SEC				
	RCS	OAMS			RCS	OAMS				RCS	OAMS				RCS				
	25	25	85	100	25	25	85	100 RAD.	100 AFT	25	25	85	100 RADIAL	100 AFT	25	25	85	100 RADIAL	100 AFT
R&D	25.0	25.0	85.0	100	2.05	2.05	2.05	2.05	2.05	180/	270/	270/	540/	540/	300/	300/260	306/	306/	306
R&D	23.5	23.5	79	94.5	1.3	1.3	1.6	1.6	1.6	100/136	118/160	147/270	220/300	220/300	275/266	283/272	299/280	295/280	295/280
SPACECRAFT 2, 3																			
SPACECRAFT 4		23.0				.7					425/578					258/250			
SPACECRAFT 5									1.2					557/757					275/273
SPACECRAFT 6 & UP							1.2	1.2				557/757	557/757				267/255	267/255	

Other early R & D experience resulted in a change from a hard, metal-to-metal valve seat to a soft seat design for improved contamination resistance. In addition, the injector feedlines and flow passages were strengthened to improve their explosion resistance, and a variable flow control orifice was replaced with a fixed orifice to insure repeatable flow characteristics.

Vibration failures in the valve mount area of the 79 and 94.5 lb TCA's necessitated the installation of an external valve support bracket for early spacecraft and a strengthened version for later spacecraft.

B. Propellant Tanks Development - During the development program, the endurance of the propellant tank bladders was found to be limited. The problem was greater for the spherical OAMS tanks than for the cylindrical RCS tanks, and more characteristics of the fuel tanks than of the oxidizer tanks. All bladders were initially of two-ply construction; however, the OAMS fuel tank bladder proved marginal and was replaced by a three-ply bladder, which was more satisfactory.

During development, the tanks were tested extensively to establish durability limits. Vibration tests, expulsion cycling tests, slosh tests, and acceleration tests were run and the overall results proved that the finally selected bladders had a greater life capability than had been expected. The RCS tanks were significantly more durable than the OAMS tanks. In fact, three RCS tanks from Spacecraft 6 were reused, without refurbishing, in the attitude control system of the augmented target docking adapter (ATDA).

C. B Package Development - The B package burst diaphragm originally selected was a metallic gold shear disc, which proved to be erratic relative to rupture pressure. Spacecraft 2 was flown with the gold disc; Spacecraft 3 did not have burst diaphragms. By Spacecraft 4 a substitute burst diaphragm had been evolved which had a tighter and more predictable rupture pressure range. This new diaphragm was composed of a hemisphere, pressurized on its convex side, which collapsed onto a set of knives at the rupture pressure.

D. Regulator Development - Due to a slight leakage from the fuel/oxidizer group, propellant compatibility was considered necessary in the pressure regulator. This goal was easily achieved except at the metal-to-metal valve seat, where very hard materials were needed for durability, because of the high unit loads required for sealing.

The first attempt was to make the poppet ball of tungsten carbide with a cobalt binder. This solution proved unfeasible because the tungsten was attacked by the oxidizer. The Spacecraft 2 OAMS regulator, with a ball made of this material, required replacement at Cape Kennedy because of corrosion. This problem was finally resolved by using Kennamental K601 as the poppet ball material.

E. State-of-Art Advances in Development of RCS and OAMS - During the development of the RCS and OAMS, certain state-of-the-art advances were made. In summary form they are:

1. Application of ablative cooling philosophies to actual practice in the case of small pulsing rocket engines.
2. Closed welding of complex component packages to prevent leakage.
3. Manufacture of filters from 304L CRES, a weldable, propellant compatible material.
4. Creation of a composite TCA valve seat, combining the desirable features of soft seats to overcome contamination sensitivity and hard seats for endurance.
5. Development of high temperature bonding adhesives for TCA assemblies.
6. Cleanliness control at component and system levels to a degree not previously achieved.

Design Approval Test Program for RCS and OAMS. - Components tested in the design approval test (DAT) program were subjected to varying sequences of environments to demonstrate either of two conditions: (1) component survival after testing, or (2) proper operation of the component during the test. (See Tables 10 and 11.) In some instances, parallel test specimens were run, but to different sequences of tests. This was done, when circumstances permitted, to conserve time or to permit running of several types of destructive tests. When the component was considered sufficiently durable, only one unit was tested (e.g., the C and D packages). If the component was life-limited, more test specimens were employed. In the case of the propellant tanks, it was discovered that all dynamic and functional tests contributed to the ultimate failure of the bladders; hence all test series designed to show bladder integrity included each of the predicted spacecraft dynamic environments. Several thrust chambers, whose life was limited by the amount of ablative

TABLE 10 DESIGN APPROVAL TEST PROGRAM
PROPULSION SYSTEMS – OAMS, RCS

ITEM	TESTS PERFORMED	SYSTEM
A PACKAGE	1, 2, 3, 4, 6, 7, 8, 9, 31	OAMS–RCS
B PACKAGE	1, 2, 6, 7, 8, 10, 18	OAMS–RCS
C PACKAGE	1, 2, 3, 4, 6, 7, 8, 9, 10	OAMS–RCS
D PACKAGE	1, 2, 6, 7, 8, 10, 31	OAMS–RCS
E PACKAGE	1, 2, 3, 6, 7, 8, 10, 11, 31	OAMS
REGULATOR	1, 2, 7, 8, 10, 17, 18, 20, 31	OAMS–RCS
MOTOR SHUTOFF VALVE	1, 2, 7, 8, 10, 11, 17, 30	OAMS–RCS
PRESSURANT TANK	1, 2, 3, 4, 5, 6, 7, 8, 31	OAMS–RCS
PROPELLANT TANK	1, 2, 3, 6, 7, 8, 10, 31, 32	OAMS–RCS
CARTRIDGES	1, 5, 7, 8, 12, 13, 14, 31, 34	OAMS–RCS
CARTRIDGE VALVES	1, 2, 5, 7, 8, 15, 16, 17, 18, 19, 31, 34	OAMS–RCS
PRESSURE TEMP INDICATOR	1, 2, 3, 6, 7, 8, 20, 31	OAMS–RCS
THRUST CHAMBER ASSY – 25 LB*	1, 2, 3, 4, 5, 6, 15, 17, 18, 21, 23, 24, 25, 26, 27, 28, 31	OAMS–RCS
THRUST CHAMBER ASSY – 25 LB*	1, 2, 3, 4, 5, 6, 10, 14, 21, 24, 25, 26, 27, 29, 31	RCS
THRUST CHAMBER ASSY – 85 LB*	1, 2, 3, 4, 5, 6, 15, 21, 24, 25, 26, 28, 30, 31	OAMS
THRUST CHAMBER ASSY – 100 LB*	1, 2, 3, 4, 5, 6, 10, 15, 21, 23, 24, 25, 26, 27, 31	OAMS
OAMS SYSTEM	1, 2, 6, 7, 8, 10, 22, 25, 31, 35	OAMS
RCS SYSTEM	1, 2, 3, 7, 8, 15, 22, 31, 35	RCS

LEGEND:

1. AMB FUNCTIONAL
2. VIBRATION
3. ACCELERATION
4. BURST
5. MECH SHOCK
6. SERVICE LIFE
7. HIGH TEMP
8. LOW TEMP
9. CARTRIDGE VALVE ACTUATION
10. COMPATIBILITY
11. CONTAMINATION
12. NO-FIRE CURRENT
13. AUTO IGNITION
14. MULTIPLE FIRING
15. HI-LO-TEMP ALT
16. SAND-DUST
17. SALT SPRAY
18. HUMIDITY
19. ACOUSTIC
20. RESPONSE
21. THERMAL SHOCK
22. SEA LEVEL FIRING
23. PROP VALVE – CONTINUOUS DUTY
24. MDC TO FAILURE
25. MDC TO GUARANTEED LIFE
26. MDC TO "IN SPEC"
27. EXPLOSIVE ATM
28. VALVE SERVICE LIFE
29. PULSING
30. THRUST VECTOR
31. PROOF PRESSURE
32. SLOSH
33. DROP
34. BRUCETON
35. PROPELLANT UTILIZATION

TABLE 11 DESIGN APPROVAL TEST SUMMARY
PROPULSION SYSTEMS – OAMS, RCS

COMPONENT	MAJOR ENVIRONMENTS								
	HIGH-LOW/TEMP	VIBRATION	SHOCK	ACCELERATION	HUMIDITY	LIFE	COMPATIBILITY	SLOSH	TEMP/ALTITUDE
A, C, D, E PACKAGES; PRESSURANT TANK	T	T	Δ	T	Δ	T	T	Δ	Δ
B PACKAGE	T	T	Δ	T	T	T	T	Δ	Δ
REGULATOR	T	T	Δ	Δ	T	T	T	Δ	Δ
MOTOR OPERATED VALVE	T	T	Δ	Δ	Δ	T	T	Δ	Δ
PROPELLANT TANKS	T	T	Δ	T	Δ	T	T	T	Δ
THRUST CHAMBERS	T	T	T	Δ	T	T	T	Δ	T
CARTRIDGE; RETRO ROCKET ASSEMBLY	T	T	T	T	T	Δ	Δ	Δ	T
PRESSURANT-TEMPERATURE INDICATOR	T	T	Δ	T	Δ	Δ	Δ	Δ	Δ
OAMS	T	T	Δ	Δ	Δ	T	T	Δ	Δ
RCS	T	T	Δ	T	Δ	T	T	Δ	T

LEGEND:
T = TEST PERFORMED
Δ = TEST NOT REQUIRED

material present, were used in the testing so that each unit might keep within its rated life for all test environments. "One-shot" items of equipment, such as the cartridges and cartridge valves, were tested in large quantities to acquire statistical performance data.

Complete systems were also tested to demonstrate interaction between components and system function at the temperature extremes. Each system was also vibration tested, the OAMS module being separated from its TCA manifolds for this test. System flushing and drying procedures were demonstrated during system DAT. In the case of the OAMS both two-tank and four-tank modules were tested.

DAT problems generally led to corrective action involving improved quality control, both in test performance and in fabrication. They also brought about a reduction of test severity so that updated, more reasonable test conditions came to be employed.

In the propellant tanks, for example, a redesign resulted because initial qualification testing showed that some tanks suffered degradation of the flange assembly during test. The specified flange leakage rate was exceeded when a lower pressure situation was encountered after the assembly has been exposed to the stringent environment. The original specification required that the tanks maintain an acceptable leak rate at both proof pressure and nominal system pressure. However, it was apparent that such a requirement could not be repeatedly maintained with the existing flange design. The principal cause of flange leakage was a propellant compatibility test environment of 160°F for two weeks; this is now known to be an overly severe condition and unrepresentative of mission environment.

The flange seal was modified by increasing Teflon thickness in the flange area. Modified tanks were then tested, employing a test time-temperature profile more representative of mission environment. The test requirements were successfully passed by tanks incorporating the revised flange design. All tanks used on Spacecraft 6 and later spacecraft incorporate the revised flange design. This type of leakage has been limited to test programs. Service experience with tanks has been free of flange leakage.

Reliability Assurance Test Program for OAMS and RCS. - In this program components were subjected to environments which were more severe than those required for qualification. (See Table 12.) The components were subjected to overstress environments of vibration, temperature extremes, endurance, and contamination to demonstrate safe margins existed in performance, durability and strength. The components tested survived these overstresses satisfactorily, although certain characteristics were revealed:

A. The A package pressure transducer shifted out of specification tolerance under extreme temperature conditions.

B. The filters in the TCA's became plugged when large amounts of contamination were introduced. One filter ruptured under the pressure differential.

C. The TCA skin temperature went over 700°F at 80% of its rated life when inlet pressures equal to proof pressure were imposed.

TABLE 12 RELIABILITY ASSURANCE TEST PROGRAM ENVIRONMENTS

	VIBRATION		LOW TEMPERATURE		HIGH TEMPERATURE		CONTAMINATION SUSCEPTIBILITY	VACUUM AGING	PROOF PRESSURE FIRING
	OPERATING	NON-OPERATING	FUNCTIONAL	ENDURANCE CYCLING	FUNCTIONAL	ENDURANCE CYCLING			
PACKAGE A		T		T		T			
PACKAGE E		T		T		T	T		
REGULATOR	T			T		T	T		
MOTOR VALVE		T	T		T				
TCA PROPELLANT VALVES									
25 LB				T		T	T		
100 LB				T		T	T		
TCA – RCS 25 LB	T	T						T	
TCA – OAMS 25 LB		T							T

T – TEST PERFORMED

D. The E package and pressure regulator leakages went up to five times specification values when large amounts of contamination were introduced into the packages.

E. The E package pressure switch failed to actuate at temperatures below -24°F.

<u>OAMS and RCS Flight Performance.</u> - The RCS performed without malfunction on all flights except the Gemini XII mission, on which the A-ring regulated pressure was excessive for a brief period following system arming.

The OAMS experienced thrust reduction in attitude control thrust chambers on the long duration missions Gemini V and Gemini VII, during the extensive duty of Gemini VIII, which was terminated early, and during the Gemini IX, Gemini XI and Gemini XII missions. The cause of the thrust reduction is being investigated and the results of the study will be submitted in McDonnell report F-206, Gemini TCA Anomaly Investigation. Table 13 of this report presents a resume of flight experience relative to propellant used and system performance.

TABLE 13 PROPULSION SYSTEM FLIGHT RESULTS

SPACECRAFT	OAMS TOTAL PROPELLANT TANKED #	OAMS PROPELLANT EXPENDED #	RCS TOTAL PROPELLANT TANKED #	RCS PROPELLANT EXPENDED #	SYSTEM PERFORMANCE	PROPULSION EVENTS DURING MISSION
2	49	12.2	70.4	66.4	OAMS, RCS NOMINAL	SEPARATION, RETRO, RE-ENTRY (BALLISTIC)
3	366	186	73.3	54.2	OAMS, RCS NOMINAL	SEPARATION, OUT-OF-PLANE MANEUVERING, RETRO, RE-ENTRY (3 ORBITS)
4	411	327	72.3	68.1	OAMS, RCS NOMINAL	SEPARATION, 4 DAYS ATTITUDE CONTROL, RETRO, RE-ENTRY (62 ORBITS)
5	385	280	72.2	58.7	RCS NOMINAL ; OAMS NORMAL EXCEPT REDUCED THRUST FROM ATTITUDE TCA's AFTER THIRD DAY – ATTRIBUTED TO OXIDIZER FREEZING	SEPARATION, 8 DAYS ATTITUDE CONTROL, RETRO, RE-ENTRY (120 ORBITS)
6	713	424	72.0	68.1	OAMS, RCS NOMINAL	SEPARATION, CIRCULARIZE ORBIT, RENDEZVOUS WITH SPACECRAFT 7, RETRO, RE-ENTRY (17 ORBITS)
7	428	320	71.9	68.0	RCS NOMINAL; OAMS NOMINAL EXCEPT REDUCED THRUST FROM ATTITUDE TCA's #3 & 4 AFTER THE 12TH DAY – ATTRIBUTED TO PLUGGING OF FEED LINE AT TCA	SEPARATION, CIRCULARIZE ORBIT, RENDEZVOUS TARGET FOR SPACECRAFT 6, 14 DAYS ATTITUDE CONTROL, RETRO, RE-ENTRY (206 ORBITS)
8	767	564	72.0	68.1	RCS NOMINAL; OAMS NOMINAL UNTIL RATED LIFE OF ATTITUDE TCA's WAS EXCEEDED. REDUCED THRUST FROM ATTITUDE TCA's 3, 7, & 8 EXPERIENCED IN OVERLIFE PERIOD	SEPARATION, CIRCULARIZE ORBIT, RENDEZVOUS AND DOCK WITH AGENA. RETRO, RE-ENTRY (7 ORBITS) EARLY TERMINATION OF MISSION
9	762	707	72.1	46.0	RCS NOMINAL; OAMS NOMINAL EXCEPT REDUCED THRUST FROM ATTITUDE TCA's #1 AND 3. RESTRICTION OF PROPELLANT FLOW APPEARED TO BE THE MOST PROBABLE CAUSE	SEPARATION, CIRCULARIZE ORBIT, RENDEZVOUS WITH AUGMENTED TARGET DOCKING ADAPTER (ATDA), RETRO, RE-ENTRY (46 ORBITS)
10	967	917	72.0	58.3	OAMS, RCS NOMINAL	SEPARATION, CIRCULARIZE ORBIT, RENDEZVOUS AND DOCK WITH AGENA, DUAL RENDEZVOUS, RETRO, RE-ENTRY, (44 ORBITS)
11	967	855	72.0	56.7	RCS NOMINAL, OAMS NOMINAL EXCEPT FOR THRUST DEGRADATION EXPERIENCED FROM SEVERAL ATTITUDE TCA's. RESTRICTION OF PROPELLANT FLOW APPEARED TO BE THE MOST PROBABLE CAUSE.	SEPARATION, CIRCULARIZE ORBIT, RENDEZVOUS AND DOCK WITH AGENA, MULTIPLE DOCKING, RERENDEZVOUS, RETRO, RE-ENTRY (44 ORBITS)
12	931	712	71.9	65.1	REGULATED PRESSURE IN RING OF THE RCS INCREASED TO 415 PSIA FOLLOWING SYSTEM ARMING. PRESSURE DECREASED TO NOMINAL WITH TCA USAGE AND REGULATOR EVENTUALLY LOCKED-UP. THE ANOMALY WAS ATTRIBUTED TO PARTICLE CONTAMINATION AT THE SEAT OF THE REGULATOR. RCS PERFORMANCE EXCEPT FOR THE ABOVE. OAMS NOMINAL EXCEPT FOR THRUST DEGRADATION EXPERIENCED FROM SEVERAL TCA's. CAUSE OF DEGRADED PERFORMANCE APPEARED TO BE PROPELLANT FLOW RESTRICTION	SEPARATION, CIRCULARIZE ORBIT, RENDEZVOUS AND DOCK WITH AGENA, MULTIPLE DOCKING, RETRO, RE-ENTRY (62 ORBITS)

There were no thruster anomalies on Spacecraft 3, 4, 6 and 10. Therefore these missions are not discussed in the following paragraphs.

A. Gemini V Experience - Gemini V data indicated that during the third day of the mission, OAMS attitude TCA No. 7 ceased to provide normal thrust. On the fourth day, thrust from TCA No. 6 became erratic. The astronauts also reported that all attitude TCA's seemed sluggish. The malfunction followed a two day period during which the heaters, used to control the temperature of the OAMS TCA oxidizer valves, were switched off to conserve electrical power. Thermal analysis indicates that without heating, and under adverse flight conditions, the OAMS oxidizer system can drop below the freezing point of nitrogen tetroxide. Performance of all but one of the malfunctioning TCA's was restored by reactivating the valve heaters. It has been concluded that the oxidizer lines froze and that the heaters were not adequate to thaw the last TCA, which was located in the coldest position in the spacecraft.

B. Gemini VII Experience - On the twelfth day of the Gemini VII mission, OAMS attitude control TCA's No. 3 and No. 4 yielded reduced thrust (estimated at 25% of rated thrust) during pulse mode operation. These same TCA's lost virtually all thrust during the direct mode of operation. Examination of all available data has resulted in the following conclusions:

1. Freezing was not a factor in the erratic performance of the TCA's. The propellant supply did not freeze nor did power ever fail to the redundant valve heaters (one of which was under thermostatic control and the other continuously active).

2. The two TCA's which failed had had more than 80 lb of propellant expended through them. This was considerably more than their rated life.

3. After thorough investigation, it was inferred that the TCA failures were probably caused by:

a. blockage of the propellant valve orifices or inlet filter by contaminants, or

b. deformation of the valve seat due to heat soakback resulting from the heavy duty cycle.

C. Gemini VIII Experience - After seven hr of the Gemini VIII mission, OAMS TCA No. 8 commenced firing without pilot actuation, was briefly interrupted, and then recommenced firing for two min and then for 11 min until the circuit breaker was opened. Examination revealed that this TCA's thrust performance was nominal until its rated life was exceeded, after which the thrust degraded. It is therefore concluded that the TCA was fired as a result of a short somewhere in the TCA valve electrical circuit.

D. Gemini IX Experience - At various intervals during the Gemini IX mission, OAMS attitude control TCA's 1 and 3 yielded reduced thrust, causing pitch/roll coupling and yaw/roll coupling, respectively. These anomalies occurred while the spacecraft attitude control was in pulse, direct, horizon scan, and rate command modes of operation. Restriction of propellant flow appeared to be the most probable cause of failure.

E. Gemini XI Experience - During the Gemini XI mission, the crew reported degraded performance from TCA's No. 6 and No. 8. In addition, the performance from TCA No. 15 was believed to be intermittent during the first and last rendezvous. The post-flight data review indicated several TCA's in addition to TCA's No. 6 and No. 8 produced degraded thrust at various times during the mission. The data review did not confirm the reported TCA No. 15 anomaly. The data indicates a problem similar to the Gemini IX experience. The degraded TCA performance was attributed to flow restriction, possibly resulting from particulate matter and/or precipitation of iron nitrate from the oxidizer.

F. Gemini XII Experience - At approximately 40 hr through the mission, the crew reported degraded performance of OAMS thrust chambers Nos. 2 and 4. Tests performed later in the mission with the thrust chambers, and post-flight analysis of data revealed degraded performance from all OAMS attitude thrusters. The best appraisal of the available facts points to propellant flow restriction as the cause of the performance loss. This restriction could have resulted from propellant contamination or the precipitation of iron nitrate from the oxidizer. All data are currently being reevaluated to establish a test program to investigate the problem. McDonnell report F-206, Gemini TCA Anomaly Investigation is being prepared for submittal to NASA.

Immediately after arming of the TCA A-ring, regulated pressure rose to a high of 415 psia (nominal is 295 psia). Following thruster initiation, however, the regulated pressure decreased to normal levels and standard regulator performance was indicated. During post-flight testing, the regulator functioned properly. The anomaly was attributed to particulate contamination at the regulator seat which was subsequently purged from the regulator as pressurant was used.

Retrograde Rocket System

Location. - The four retrograde rockets are located in the retrograde section of the adapter, as shown in Fig. 17. Mounted on crossed aluminum I-beams, the retrograde rockets are positioned symmetrically about the longitudinal axis of the spacecraft. They are exposed when the adapter equipment section is jettisoned shortly before the retrograde maneuver. The retrograde section itself is jettisoned approximately 45 sec after firing of the rocket motors.

Purpose. - The retrograde rockets are used to decelerate the spacecraft at the start of the re-entry maneuver. As an alternate task, these motors increase spacecraft velocity to aid in separation from the launch vehicle during a high altitude, but yet suborbital, abort. During retrograde, the motors are fired sequentially at five and one-half sec intervals. In the abort situation the rockets are fired in salvo.

Retrograde Rocket Description. - The four retrograde rocket motors are solid propellant devices, each generating a nominal thrust of 2500 lb. Each motor assembly consists of a solid propellant grain (polysulfide/ammonium perchlorate), a motor case, an exhaust nozzle and two pyrogen-type igniter assemblies, as illustrated in Fig. 20. All four motors are identical, with the single exception that the igniter electrical connectors are broached differently to prevent incorrect electrical hookup and therefore insure proper firing sequence. The ignition of these motors follows this order. An electrical signal, communicated through bridgewire, ignites a prime mix and booster charge which provide the heat and pressure required to ignite boron-potassium nitrate pellets. These, in turn, burn to ignite the pyrogen sustainer grain. The pyrogen exhausts into the motor cavity, igniting the main propellant charge, which provides thrust and impulse through the exhaust nozzle.

Retrograde Rocket Development Program. - Thirty-three motors and 50 pyrogen igniters were tested during the development program. The units were subjected to structural, temperature, functional, and aging tests. (See Table 14.) Eight motors, or two sets, were used in two altitude rocket abort (popgun) tests, performed to verify the structural integrity of the spacecraft blast shield and structure under altitude abort conditions. During the first test, three of the four rockets ejected their exhaust nozzles. Since loss of the exhaust nozzle endangers the nozzle throat, without which the rocket motor cannot fire, a redesign which would retain the nozzle more securely was considered necessary. Following this redesign, 18 motors were successfully tested, four of these in a repeat of the popgun test. An igniter reliability test series was performed, in which the fourth unit to be tested misfired. The remaining igniters were redesigned to provide a higher initiator output and then successfully tested.

Retrograde Rocket Design Approval Test Program. - The design approval test program consisted of environmental, structural, and functional tests. (See Table 15.) The program was carried out with no failures.

Flight Experience. - The retrograde rockets have functioned without anomaly on all Gemini flights. The motors have not been called upon to perform their secondary or abort task.

ENVIRONMENTAL CONTROL SYSTEM

Major functions of the environmental control system (ECS) are to:

A. Control the suit and cabin atmosphere.

B. Control the temperatures of the astronaut suits and the spacecraft equipment.

C. Provide drinking water for the crew and a means for storing or disposing of waste water.

Suit And Cabin

Atmosphere Control. - A simple schematic of the suit and cabin atmospheric system is shown in Fig. 21. The primary oxygen system provides oxygen during launch and orbital flight, and the secondary oxygen system provides oxygen during retrograde and re-entry.

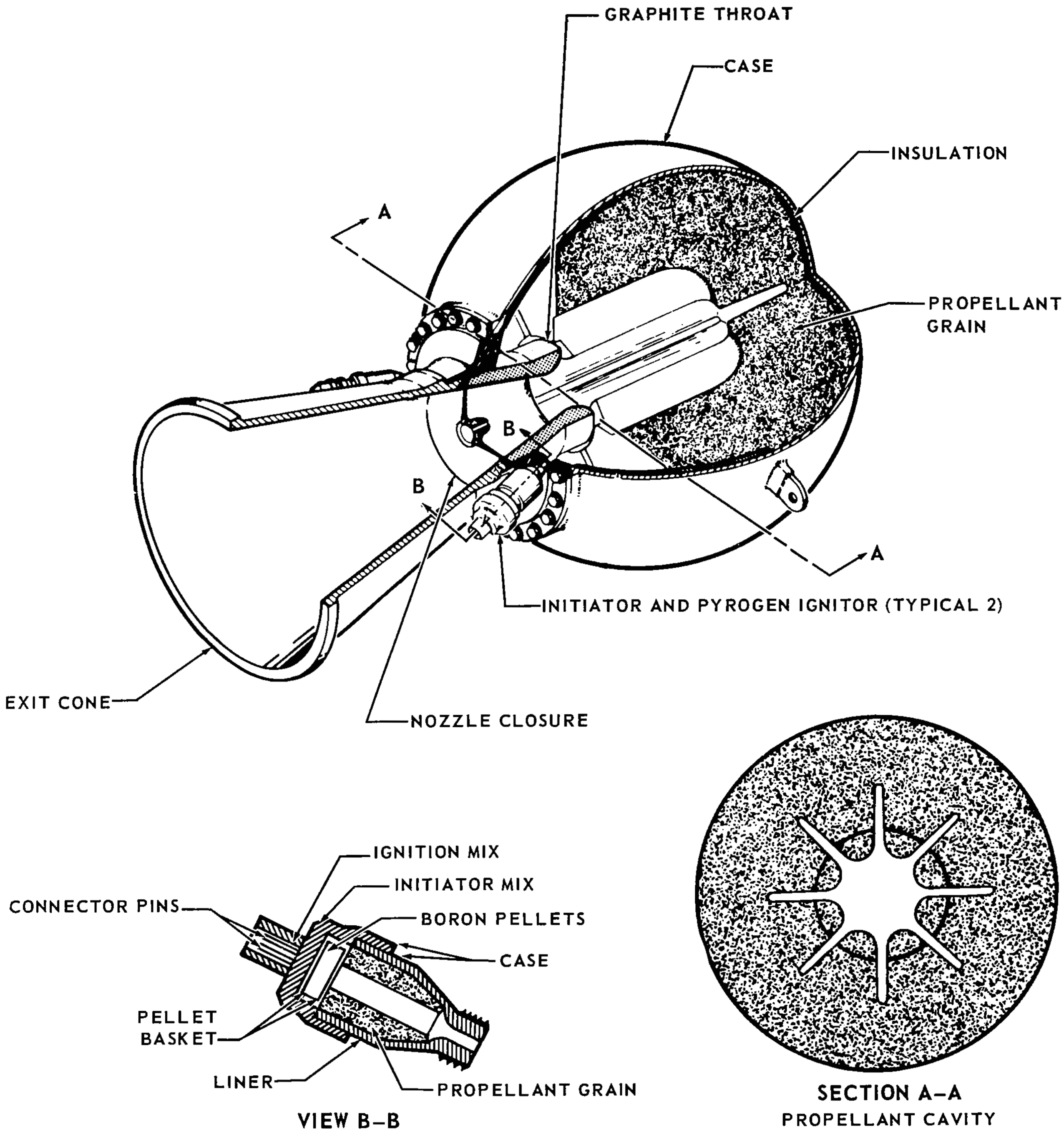


FIGURE 20 RETROGRADE ROCKET

TABLE 14 DEVELOPMENT PROGRAM – RETROGRADE ROCKETS

ROCKET MOTOR TESTS

TEST	NO. OF UNITS TESTED	
	ORIGINAL CONFIGURATION	FINAL CONFIGURATION
TEMPERATURE CYCLING	4	4
VIBRATION – HIGH TEMPERATURE	–	2
– NORMAL TEMPERATURE	2	–
– LOW TEMPERATURE	–	2
DROP	2	4
ACCELERATION	1	2
HUMIDITY	2	4
TEMPERATURE GRADIENT	–	3
STATIC FIRE – SEA LEVEL – HIGH TEMP	1	6
– NORMAL TEMP	4	1
– LOW TEMP	1	–
STATIC FIRE – VACUUM – HIGH TEMP	1	2
– LOW TEMP	2	2
STATIC FIRE – SEA LEVEL (ALTITUDE IGNITION)	3	–
SPACE AGING	–	2
ACCELERATED AGING	–	1
POP-GUN	4	4

PYROGEN TESTS

TEST	NO. OF UNITS TESTED
VIBRATION – LOW TEMP	17
– NORMAL TEMP	16
– HIGH TEMP	17
ACOUSTIC NOISE	6
STATIC FIRING – SEA LEVEL – LOW TEMP	8
– NORMAL TEMP	6
– HIGH TEMP	8
STATIC FIRING – VACUUM – LOW TEMP	9
– NORMAL TEMP	10
– HIGH TEMP	9

TABLE 15 DESIGN APPROVAL TEST PROGRAM
RETROGRADE ROCKET

TEST	NO. OF UNITS TESTED
TEMPERATURE CYCLING	18
VIBRATION – HIGH TEMPERATURE	6
– NORMAL TEMPERATURE	6
– LOW TEMPERATURE	6
DROP	10
ACCELERATION	9
HUMIDITY	9
TEMPERATURE GRADIENT	6
VACUUM SOAK – HIGH TEMPERATURE	8
– LOW TEMPERATURE	8
STATIC FIRE – SEA LEVEL – HIGH TEMP	5
– NORMAL TEMP	5
– LOW TEMP	8
STATIC FIRE – SEA LEVEL (VACUUM IGNITION)	
– HIGH TEMP	8
– LOW TEMP	8
STATIC FIRE – ALTITUDE	6

TOTAL OF 40 UNITS TESTED

Operational functions of the suit and cabin atmospheric control system are as follows:

A. The oxygen supply system consists of primary O_2 (stored cryogenically at supercritical pressure), secondary oxygen, and egress oxygen. (The egress oxygen supply has been deleted for Spacecraft 8 through 12. Ref Environmental Control System Design Changes, D. page 168).

B. Makeup oxygen is provided to the suit loop, and to the cabin by the primary O_2 system at a regulated pressure.

C. Cabin pressure is regulated via the suit loop.

D. The cabin is repressurized by opening the repressurization valve.

E. Suit pressure is maintained by the suit demand regulators.

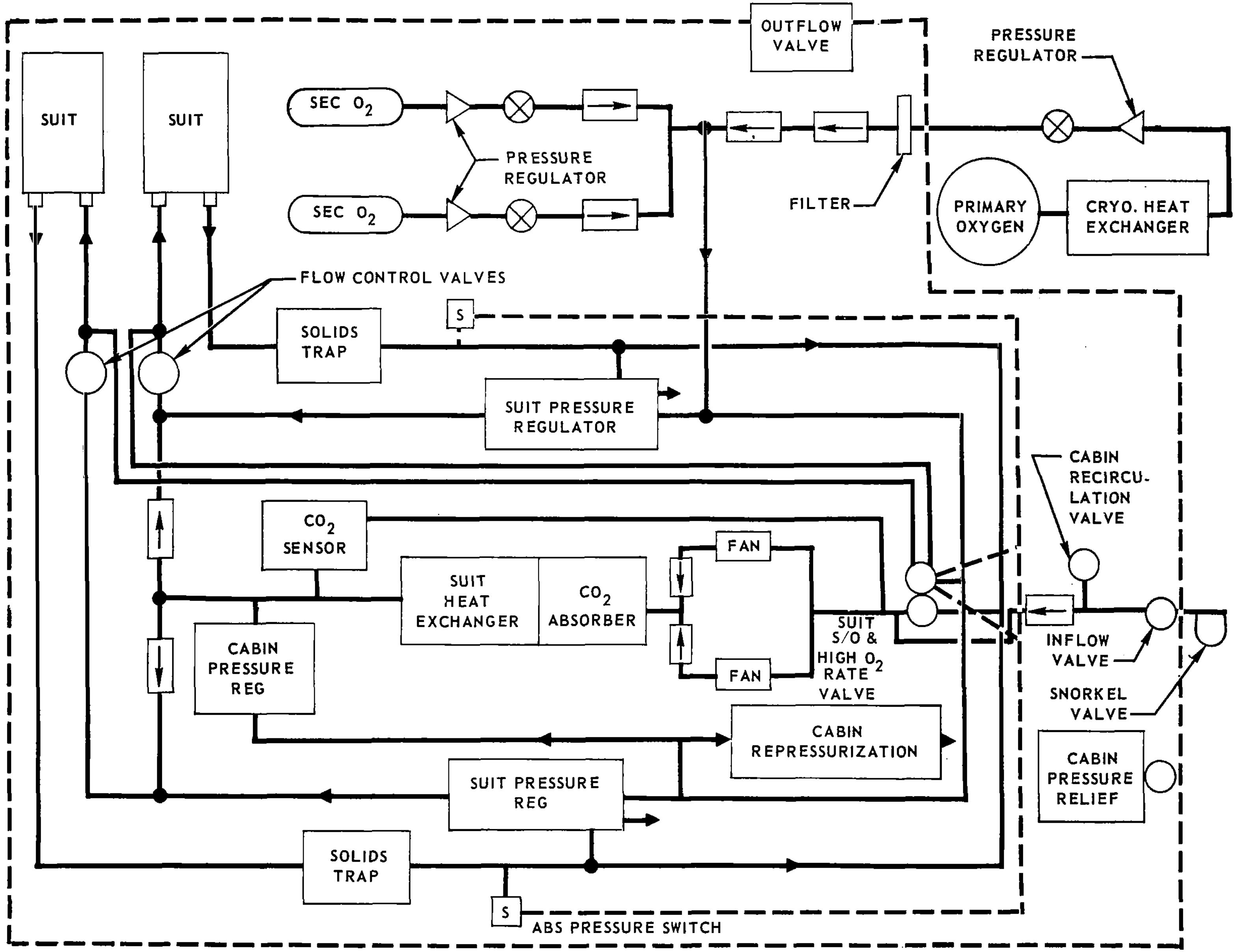


FIGURE 21 SUIT AND CABIN ATMOSPHERE CONTROL

F. Suit circulation and atmospheric control are provided by fans, suit heat exchanger and condensate removal, CO_2 and odor removal, and solid traps.

G. Cabin moisture is partially removed by flow of water vapor through the cabin recirculation valve into the suit loop, where the water vapor is condensed and separated by the suit heat exchanger.

H. Condensate collected by the suit heat exchanger flows to the launch heat exchanger, where it is dumped overboard as liquid or vapor.

I. For parachute descent, the cabin pressure relief valve is opened, the inlet snorkel and outflow vent valves are closed, and the O_2 high-rate is selected.

J. After landing, the inlet snorkel, outflow vent, and recirculation valves are opened so that the suit compressor can provide atmospheric circulation.

Suit and Cabin Safety and Redundancy. - Safety and redundancy features of the suit and cabin atmosphere control system are as follows:

A. Suit plus cabin pressurization.

B. Sufficient secondary oxygen (in case of failure of primary O_2 supply) in each of two bottles for one orbit at normal consumption rate plus re-entry at high O_2 consumption rate.

C. Automatic high rate O_2 flow in case of decompressed cabin and excessive suit circuit leakage.

D. Cabin pressure regulator preserves O_2 supply in case of excessive cabin leakage.

E. Combined dual cabin pressure relief valve and water seal valve.

F. Redundant suit fans and power supplies.

G. Dual cabin pressure regulator.

H. CO_2 sensor to check CO_2 concentration in suit loop.

I. Redundant primary O_2 check valves to prevent loss of secondary O_2 at adapter separation.

J. Redundant means of initiating high O_2 rate automatically.

K. Redundant suit pressure regulators.

Cooling System

Heat Rejection Devices. - A schematic of the cooling system is shown in Fig. 22. By active and passive cooling, the system provides temperature control for the crew and spacecraft equipment. Heat sinks, utilized for temperature control during the different phases of the mission, are as follows:

A. Prelaunch - Heat is rejected to the ground cooling system via the coolant loop and the ground cooling heat exchanger.

B. Launch and Insertion - Heat is rejected to space via the coolant loop by boiling water in the launch heat exchanger.

C. Orbit - Heat is transferred by the coolant loop to the radiator where it is radiated to space.

D. Re-entry - Heat is radiated to space by the re-entry module surfaces and is absorbed by the heat sink capacity of the re-entry module.

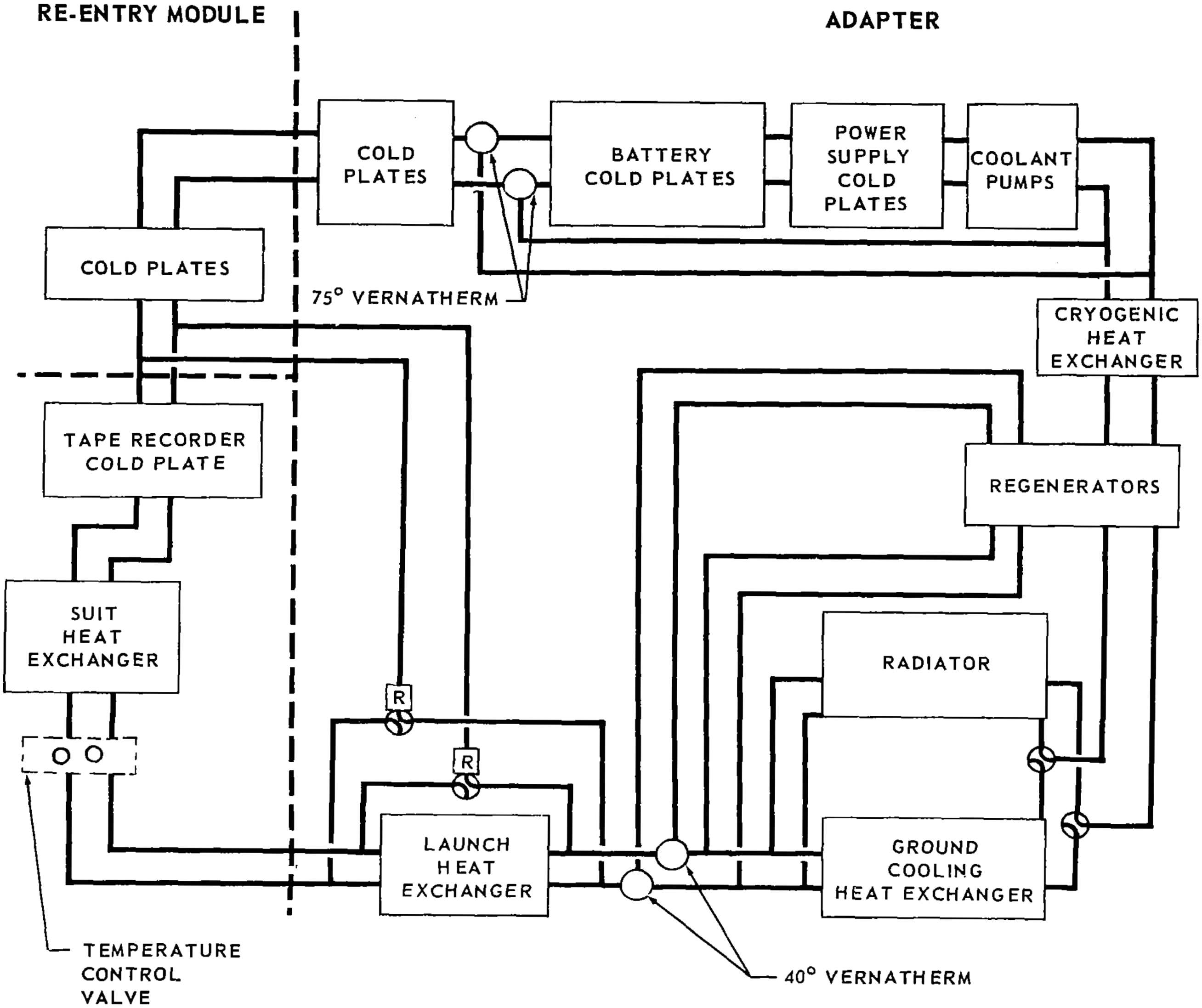


FIGURE 22 TEMPERATURE CONTROL

Cooling System Operational Functions. - Operational functions of the cooling system are as follows:

A. Coolant is forced through the system by pumps.
B. Heat is transferred from the coolant to the primary oxygen via a cryogenic heat exchanger.
C. Heat is rejected from the coolant loop by either the ground cooling heat exchanger, the launch heat exchanger, or the radiator. Outlet temperature of the ground cooling heat exchanger and the radiator is controlled to approximately 40°F by means of a regenerative loop and a vernatherm temperature control valve.

D. Heat is transferred from the suit and cabin atmosphere to the coolant via the suit heat exchanger.

E. Heat is transferred from the equipment to the coolant via coldplates. Coolant inlet temperatures to the batteries and fuel cells are controlled to a range of 50°F to 90°F by means of a bypass coolant loop and a vernatherm valve.

F. Water lines in the adapter, RCS propellant valves, and portions of the OAMS oxidizer system are electrically heated to prevent freezing.

G. Passive temperature control is maintained by special surface coatings and thermal insulation.

Cooling System Safety and Redundancy Features. - Safety and redundancy features of the cooling system are as follows:

A. Two independent cooling loops.

B. Two parallel coolant pumps, with independent power supplies in each coolant loop (except in Spacecraft 6). Because it is a battery-powered version, Spacecraft 6 required only one coolant pump in each loop for both high and low electrical loads.

C. Launch heat exchanger, capable of performing the radiator heat rejection function for a limited time.

D. Coolant bypass loop to cool re-entry module equipment when cabin coolant flow is manually reduced.

E. Warning lights to indicate lack of coolant fluid in each loop and loss of pump flow.

Water Management System

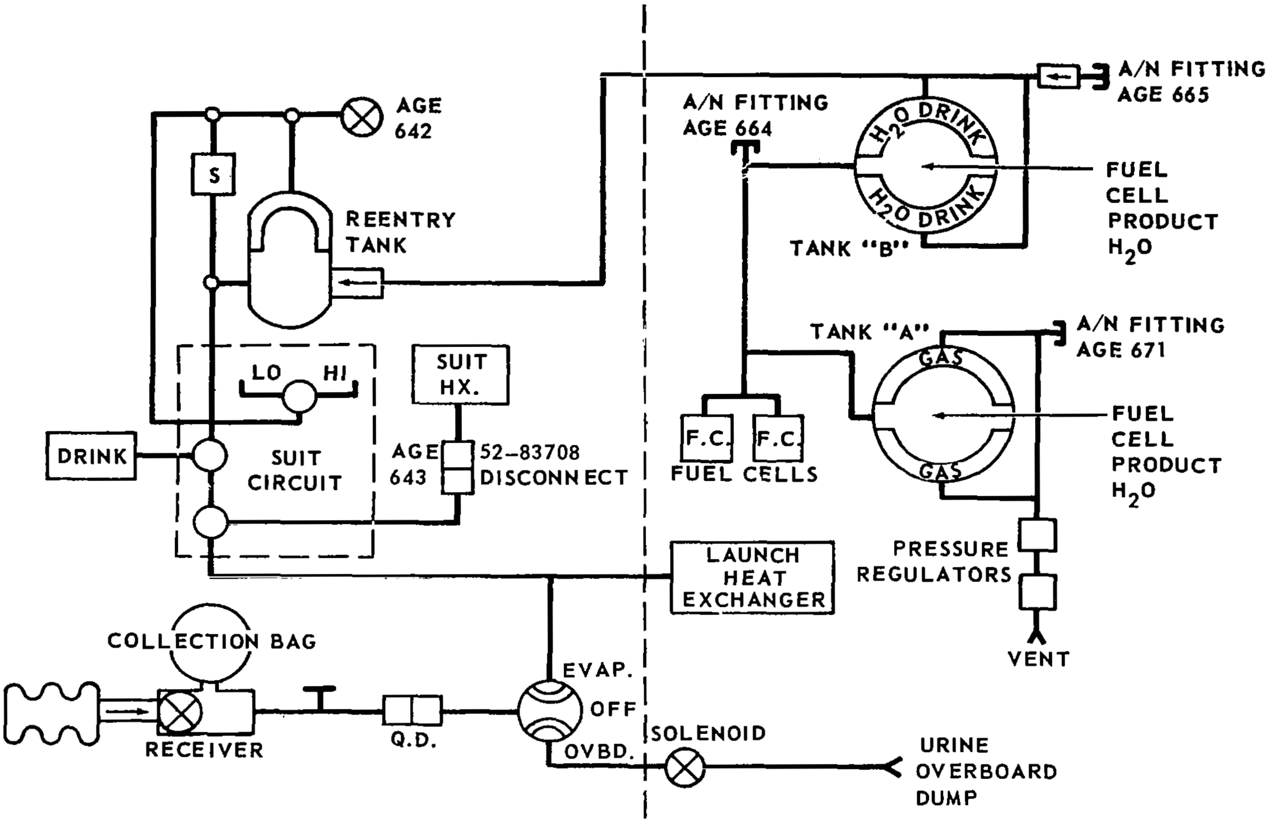
Function. - The water management system provides drinking water for the crew during flight and after landing, and provides a means for disposing of or storing waste water. The Spacecraft 10 water system is shown schematically in Fig. 23 and its operational functions are described below:

A. Tank A has a 42 lb capacity, is located in the adapter module, and is initially serviced with nitrogen at 19.0 psia. Fuel cell product water is transferred into tank A by means of a bladder, the expansion of which replaces the nitrogen and forces it overboard through two regulators (the second gives redundancy). These pressure regulators are set at 20 $\pm$ 0.5 psia, in order to prevent feedback into the tank and fuel cells.

B. Tank B, also 42 lb capacity tank, located in the adapter module, is initially serviced with potable water and is pressurized with fuel cell water at 19.0 psia. As fuel cell product water flows into tank B, potable water is transferred into the re-entry tank.

C. The re-entry tank has a 16 lb capacity and is located in the re-entry module. Since the re-entry tank is referenced to cabin pressure (nominally 5.0 psia), the adapter tanks will keep the re-entry tank filled.

D. Water is supplied to the drink nozzle from the cabin tank via the drinking water tube. Water transfer pressure is provided by: (1) the adapter water tanks, (2) suit compressor outlet pressure, or (3) the biomed bulb. Water can be drunk with or without cabin pressurization.



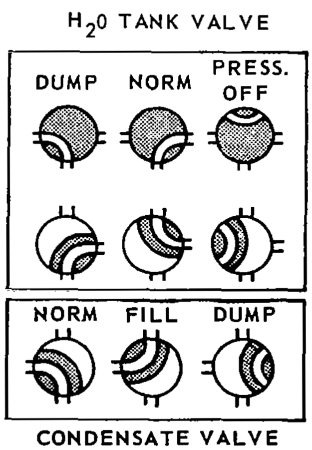


FIGURE 23 WATER MANAGEMENT SYSTEM

E. The launch heat exchanger is preloaded with seven lb of water prior to launch and is utilized during the launch phase, in lieu of the space radiator, for coolant system heat dissipation. During orbit, condensate water from the suit heat exchanger water separator is directed overboard through the launch heat exchanger.

Water Management Safety and Redundancy Features. - Safety and redundancy features of the water management system are as follows:

A. Three water tanks.

B. Pressurization of cabin water tank by either suit oxygen (suit compressors) or adapter water tanks.

C. Gas relief valve prevents overpressurization of water tanks.

D. Warning lights indicate failed closed poppet valve on the launch cooling heat exchanger.

E. Redundant means of urine dump: directly overboard or through the launch heat exchanger.

Environmental Control System Development Test Status

During the development of the environmental control system components, designs were verified by production prototype tests rather than by engineering model tests. For example, if a pressure regulator was to be produced as a casting, the engineering test model was also produced as a casting rather than by a more convenient machining process. Thus, additional testing for producibility was avoided, and confidence in flight worthiness was accumulated from developmental tests as well as from later qualification and system reliability demonstration tests.

As shown in Table 16, major system developmental tests of the environmental control system that have been completed include: (1) full-scale radiator tests under simulated orbital conditions to determine pressure drop and radiator heat transfer characteristics, and (2) 129 hr of manned runs using the boilerplate No. 2 test article to determine the performance of the suit and cabin atmosphere control system in all modes of operation.

Environmental Control System Qualification Test Status

Wherever possible, qualification of the environmental control system has been demonstrated at the system level rather than at the component level because of the close interrelationship of its components, especially with respect to thermal performance. System qualification tests are followed by mission reliability demonstration tests, and, in the case of Spacecraft 3A, a flight-type spacecraft with operating systems has been subjected to simulated orbital environments to qualify the temperature control system for the extremes of flight conditions.

TABLE 16 ECS DEVELOPMENT STATUS
MAJOR DEVELOPMENT TESTS COMPLETED

TEST	SUMMARY OF TEST RESULTS
B/P 2 SUIT AND CABIN ATMOSPHERE SYSTEM TESTS	COMPLETED 20 HOURS OF PRE-LAUNCH TESTING, 96 HOURS OF ORBITAL TESTING, AND 13 HOURS OF POST LANDING TESTING. VERIFIED PERFORMANCE OF CABIN AND SUIT ATMOSPHERE SYSTEM. INSULATED COOLANT LINES AS RESULT OF B/P 2 TESTS.
FULL SCALE RADIATOR TESTS	SIMULATED ORBITAL OPERATION VERIFIED RADIATOR PRESSURE DROP AND HEAT TRANSFER PERFORMANCE.
URINE DUMP	DEVELOP DIRECT OVERBOARD DUMP AS PRIMARY MEANS OF URINE DISPOSAL DUMP THROUGH LAUNCH HEAT EXCHANGER ALSO PROVIDED FOR REDUNDANCY.

The environmental control system qualification test status is summarized in Table 17. All subassemblies and components were successfully subjected to the complete qualification test program, which includes environmental, dynamic, and temperature-altitude testing.

In addition to the qualification test program summarized in Table 17, six two-day mission reliability demonstration tests of the suit and cabin atmosphere system were completed with no failures. In these tests, all cabin and adapter mounted suit and cabin atmosphere components were exposed to simulated altitude, temperature cycling, and temperature extremes in an altitude chamber. Moisture and CO_2 atmospheric conditions were provided by crewman simulators. After each of these tests, oxygen containers were serviced and LiOH canisters were replaced, but otherwise the same environmental control system components were used for all six tests.

Three seven-day and six 14-day mission reliability demonstration tests have also been completed. The first test also serves as a temperature/altitude long-mission qualification test of the suit and cabin atmosphere and oxygen supply systems.

Environmental Control System Failure Summary

A summary of the major problems that were encountered and solved in the development of the environmental control system is as follows:

A. Coolant Pump Performance - Some pump units were unable to meet specification performance. The problem was solved by application of more stringent quality control procedures in manufacturing.

TABLE 17 ECS QUALIFICATION STATUS

SYSTEM	MAJOR ENVIRONMENTAL											REMARKS
	HI-LOW TEMP	VIBRATION	SHOCK	TEMP-ALT	ACCELERATION	HUMIDITY	O_2 ATMOS	PRESSURE	IMMERSION	LEAKAGE	OTHERS*	
SUIT & CABIN ATMOSPHERE	34	14	15	12	13	17	10	15	8	8	2	COMPLETE
PRIMARY & SEC O_2 SUPPLY	22	10	9	9	9	10	6	9	3	5	9	COMPLETE
EGRESS O_2 SUPPLY	5	4	3	3	4	4	3	4	3	-	2	COMPLETE
COOLING SYSTEM	23	12	4	12	12	13	7	12	1	5	4	COMPLETE
WATER MANAGEMENT	18	10	8	7	6	6	3	8	3	1	2	COMPLETE

*RFI, STRUCTURAL, EXPLOSION, HEAT TRANSFER, & OTHER QUALIFICATION TESTS.

X — COMPLETELY QUALIFIED. NO. OF REQUIRED TESTS INDICATED.

X — NO. OF TESTS SUCCESSFULLY COMPLETED.

X — NO. OF TESTS REQUIRED.

TOTAL TESTS REQUIRED = 461
TOTAL TESTS SUCCESSFULLY COMPLETED = 461

B. Coldplate Manufacturing - Difficulty was encountered in assuring that the coldplates were completely flushed after salt brazing. The problem was solved by changing the process to a nonflux, inert gas braze.

C. Cryogenic Vessel Forming - Wrinkles and cracks occurred in early attempts to form titanium half-shells for cryogenic vessels. The problem was solved by variations in the deep draw forming process.

Environmental Control System Design Changes

As a result of redefined mission requirements, several changes to the environmental control system have been required.

A. To provide for mission durations in excess of two days, the two-day LiOH cartridge has been replaced by a seven-day LiOH cartridge, for Spacecraft 6 and Spacecraft 8 through 12. This caused a nine lb weight increase.

B. Due to an increase in OAMS requirements for Spacecraft 10 through 12, the RSS oxygen tank has been deleted. The ECS oxygen tank now has the dual requirement of supplying the environmental oxygen as well as the reactants (fuel cell) oxygen supply.

C. For Spacecraft 10 and 11, two additional secondary oxygen tanks, plus various valves, regulators, and associated connections, have been added in the adapter module to supply gaseous nitrogen for EVA astronaut maneuvering.

D. The egress oxygen supply has been deleted, for Spacecraft 8 through 12. The composite disconnect and tank block remains to permit separation from the suit loop in the event of abort via the ejection seat.

E. Due to the significant quantity of water recovered from the Spacecraft 3A cabin at the conclusion of the thermal qualification test, some method of preventing or controlling condensation within the pressurized area was deemed necessary. Several approaches to this problem were available. One potential solution was to insulate all cold areas of the cabin to prevent the exposed surfaces from falling below the dew point temperature of the gas in the cabin. Another solution was to cover the cabin walls with a blotting material which would absorb the moisture which condenses on the walls.

Experimentation with wallpaper (an absorbent material fabricated from urethane foam or regenerated cellulose) disclosed that it could feasibly provide moisture absorption but would not effectually control temperatures. For the long duration mission of Gemini VII, it was therefore decided to modify the ECS by installing egress-kit bypass hoses to the suit outlet ducts to eliminate heat transfer between the suit outlet and inlet gas. This modification reduced water condensation in the lithium hydroxide charge by delivering higher temperatures, thus extending the life of the LiOH charge and providing additional cooling for the crew. In addition, the carbon dioxide and odor absorber canister in the re-entry assembly ECS package also contained an insulated LiOH cartridge having greater CO_2 absorption capacity than the Spacecraft 5 cartridge, thereby avoiding the possibility of lithium hydroxide hydration.

These changes were instituted in addition to the wallpaper, for the Spacecraft 7 mission only. The wallpaper was installed in Spacecraft 4 and up. Sheets of Amsco sponge cloth were stitched to cotton broadcloth and treated to enhance their fire retardant capability. Approximately 7,000 square in. were installed wherever practical on cabin interior surfaces. Areas of installation included the bulkheads, the sidewalls, the floor, the hatch sills, the hatches, the consoles, debris guards and the stowage boxes.

Before flight qualification, some concern was felt over the possible boiling of the absorbed water during the re-entry mode. The astronauts reported that if any steam generation did occur (especially at the ECS doors), the levels were not significant enough to cause the removal or redesign of the wallpaper. On Spacecraft 4, 5, 7, 8 and 9 attempts were made to analyze wallpaper samples to determine their absorption rate. The results of these studies were inconclusive, because a reliable quantitative indication of absorption was not established. However, since the additional weight of the installed wallpaper is negligible, and since its absorbent qualities are good, a sound basis exists for its use.

Problem Areas and Corrective Action

Urine Filters. - Effective for the 14-day mission of Gemini VII, two new urine filters were installed in an effort to reduce the amount of contamination at the system solenoid. The urine system functioned properly during flight; a vigorous inspection of the filters and the overboard dump solenoid valve was undertaken at the conclusion of the mission. Examination of the filters (one used for ten and the other for four days) revealed a certain amount of coating of the outer surfaces, naturally more pronounced on the ten day filter. Microscopic analysis of the foreign material disclosed normal urine sediment, some amorphous material (probably calcium phosphate), some crystals, some cells and cellular debris. Also present were a number of fibers. The latter agreed in width and appearance with samples of glass wool. In addition, a few flakes of green plastic material were found. No metal fragments were discovered.

In conclusion, it was determined that the filters had held back the precipitated material which normally forms in a urine sample as it becomes alkaline; certain foreign substances had also accumulated, but their amounts were negligible and their composition was not such as to cause alarm.

Disassembly of the urine solenoid valve revealed small amounts of contaminant in the area of the inlet port. No contaminant was visible at the outlet port. The foreign substances fell into four distinct categories:

A. White crystalline powder coating the entire inner surfaces of the valve

B. Yellow crystalline powder coating the entrance port

C. Black particles loosely bound to the inner surface of the valve, and

D. One large yellow particle found inside the valve.

Samples of this foreign matter were removed and subjected to chemical analysis. The analysis demonstrated that the white and yellow crystalline powders (A and B above) were products of dried urine. The black particles were not positively identified, but they are a typical aliphatic hydrocarbon, such as wax. The large yellow particle was identified as an epoxy.

The deposition of the crystalline powders was considered to be normal for a mission. The origin of sample C was not determined. Sample D was felt to be a piece of epoxy tube coating which had become dislodged. The amounts of contaminant material discovered by this investigation prove that the new urine filter was effective in reducing contamination and preventing a stoppage in the solenoid valve. A review of manufacturing techniques demonstrated that normal handling or flexing of the epoxy-coated aluminum lines between the urine filter and the solenoid valve would not be detrimental to the internal coating.

Solids Traps.

A. A routine investigation was conducted after the completion of the 14-day mission of Gemini VII to determine if the solids traps were still functioning and to record the amount of filter saturation that had occurred. Both traps were removed from the spacecraft and sent to the laboratory for analysis.

Inspection revealed that the residue from the left-hand duct and trap weighed 0.787 grams; that from the right-hand duct weighed 0.914 grams. The composition of the collected material was dust, lint, skin flakes and hair. The contents of the traps differed slightly, in that there was more lint and hair in the left-hand duct and trap than in the right-hand unit. The right-hand unit appeared to contain more dust or sand particles than the left. Both solids traps were found to be operative and not saturated. No abnormal conditions were observable in either traps or filters.

B. The flight crew reported considerable eye irritation during the standup EVA on the Gemini X mission. A thorough post-flight investigation of the problem was made; this included a careful examination and chemical analysis of the contents of the solids traps. The constitution of the material in the traps was as follows:

LH SOLIDS TRAP	RH SOLIDS TRAP
Gold Flakes	Gold Flakes
Unidentified Plastic	Hair
Li: 0.13 ppm (parts per million)	Li: None
Ca: 0.148 ppm	Ca: 0.23 ppm
Na: 7 ppm	Na: 4.3 ppm
K: 2.4 ppm	K: 1.0 ppm

As a result of this investigation, it was demonstrable that the quantities of material found in the solids traps were not significant enough to be harmful to mission success; there was also no evidence that the substances found in the traps had produced the reported eye irritation.*

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*Traces of trichloroethane were found in the charcoal charge of the LiOH canister on Spacecraft 10. During the Spacecraft 12 altitude runs, gas samples also revealed the presence of trichloroethane. McDonnell was therefore directed by the NASA to discontinue the use of all chlorinated solvents for cleaning spacecraft and AGE components, and to substitute an alkaline compound with detergent characteristics slightly stronger than hand soap. In addition, the decision was made to employ only one suit fan during both the standup and the umbilical EVA, in order to decrease the air flow across the astronauts' faces during these periods.

C. The relative cleanliness of the suit circuit for the entire Gemini program was demonstrated by the negligible amounts of foreign material found collected in the solids traps during post-flight investigations.

Drinking Water Contamination. - A post-flight investigation was made of the Spacecraft 9 drinking water system to determine the water's conformity with specifications MMS606 and PS20531. Analysis revealed that both the particle count and the impurities present were higher than specification. Further examination disclosed that those lines utilizing aluminum Voi-shan flare savers (seals) exhibited corrosion damage, resulting in the deposition of particles of beta-alumina trihydrate. Lines without the Voi-shans, or having nickel flare savers, gave no evidence of corrosion.

As a result of this investigation, aluminum Voi-shans were deemed unacceptable for use in the water system; replacement with nickel Voi-shans was recommended.

It is noteworthy that the chemical analysis of drinking water remaining in the cabin water tank detected no platinum or sulfonates; therefore the possibility of fuel cell water seepage was excluded.

Cabin Pressurization. - At approximately 66:30 GET in the Gemini IXA mission, the cabin pressure declined from a normal 5.1 psia to 4.68 psia at 67:37 GET. The pressure then recovered and rose to 5.19 psia. An intensive post-flight investigation was made of the pressure regulator, pressure transducer, and the cabin pressure relief valve. No anomalies were discovered in either the transducer or the regulator. The pressure relief valve was subjected to cycling above the crack pressure of 5.5 to 6.0 psia and allowed to reseat ten times. No abnormal leakage was encountered in this testing.

The valve was disassembled in the McDonnell failure analysis laboratory and examined for possible valve seat damage and contamination. This investigation revealed no damage to the seats; however, a white and brownish-pink residue was discovered in the following areas:

A. On the inner face of both main poppets.
B. On the springs and diaphragms of both pilot valves.
C. On the housing of both pilot valves.

Chemical analysis of the residue revealed that it was composed of sodium chloride with traces of calcium sulphate. A study of the flight plan indicated that the valve had been open during the landing mode; the deposits were therefore most probably due to exposure to sea water. After this analysis was completed, the valves were reassembled and retested. No anomalies were discovered.

Despite the most thorough investigation of all hardware involved in the cabin pressurization loss, no final determination of the anomaly could be made.

CREW STATION

The focal point in the operation of a spacecraft is the crew station. The primary goal of the design phase was to produce a simple, reliable and effective display and control system which would complement an intelligent, well-trained astronaut in the performance of the mission.

To achieve this objective, simple and available components such as meters were used for display purposes in lieu of cathode ray tubes, etc. Reliance on crew judgment was reflected in a design which provided the astronauts with suitable controls and displays, and eliminated ground-based or automatic onboard systems from important control functions. Independent operation of subsystems was provided to the crew through mode switching of both controls and displays. Critical sequences, such as retrofire, were implemented with dual mechanical and electrical interlocks to preclude malfunctions or inadvertent actuations.

Gemini originally was designed to accommodate a 75 percentile man in the sitting position. It was then learned that some second generation astronauts, although six ft or under, were greater than 75 percentile in sitting height. In addition, some of these individuals grew up to two in. when torso length was measured lying on their backs, simulating a weightless condition. For this reason it was determined that more height in the crew area was required. However, since external geometry as well as seat configuration was fixed, the obvious solution was outruled. The egress kit containing oxygen was cut 1.75 in. by making the part a machining rather than containing bottles. In addition the hatch was internally "bumped" in the region of the helmet area to give the astronauts additional room above their heads. An additional .75 in. was gained in this manner and proved to be a great aid for ingress from EVA.

Controls And Displays

Spacecraft 12 control and display panel is shown in Fig. 24. The basic arrangement places piloting functions at the left hand station, engineer and navigation duties at the right and common functions at the center.

Differences in panel arrangement resulted from system changes and flight crew experience during the program. The change from batteries to fuel cells and the variation in experiment controls account for right hand panel differences, whereas only minor switches were added on the left. On the center panel a ground elapsed time clock replaced the flight plan roller, and minor changes were made to the attitude control and computer mode switches.

Several lighting trade-off studies were made for the Gemini crew station. Because of weight and power considerations, flood lighting was selected for Gemini. Since red and white lights having a dimming capacity were desired, incandescent bulbs with rheostats were selected. Additional warning and telepanel lights were used to denote critical functions and conditions. In several cases these warning lights remained illuminated for prolonged periods, and were

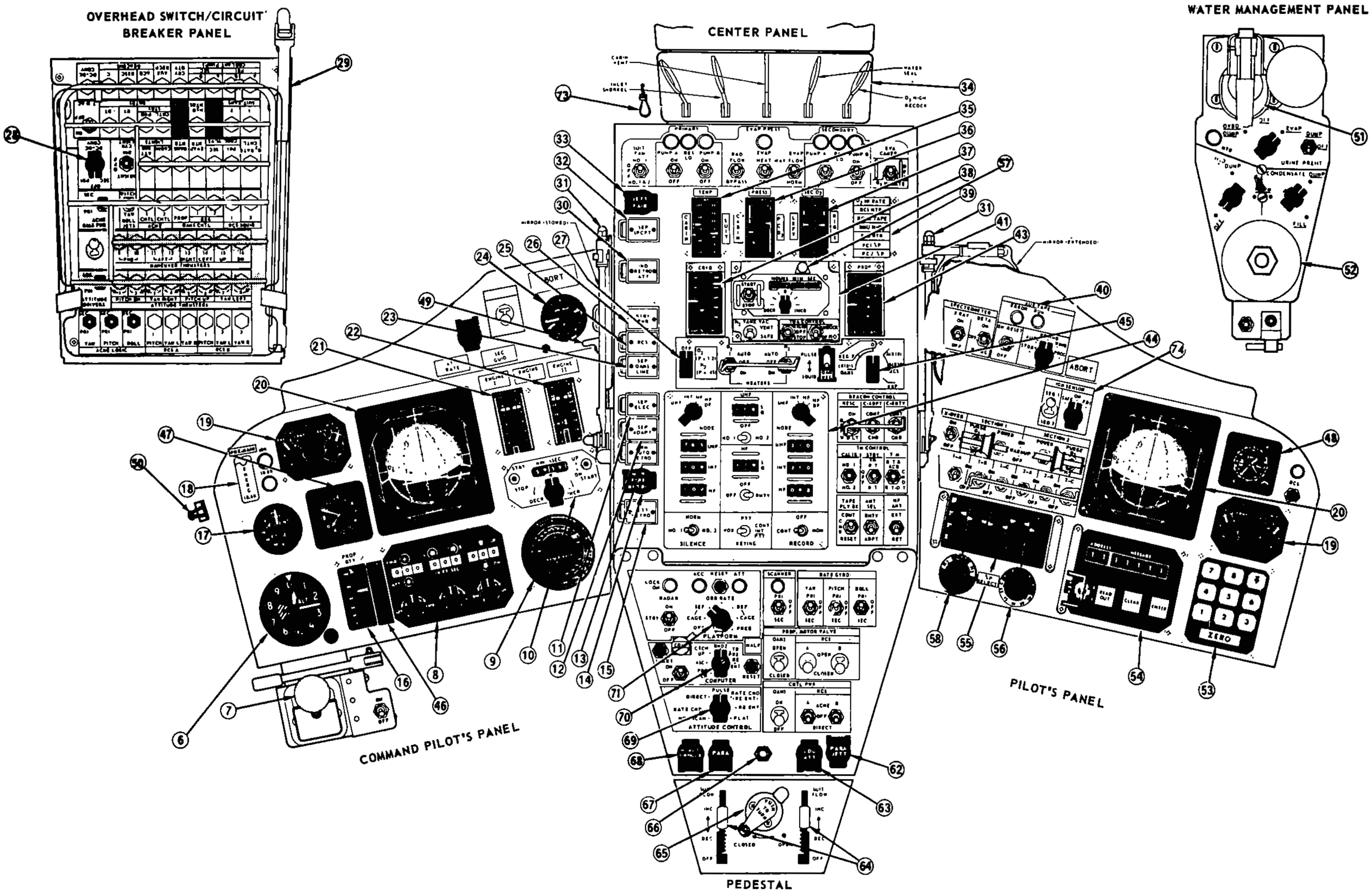
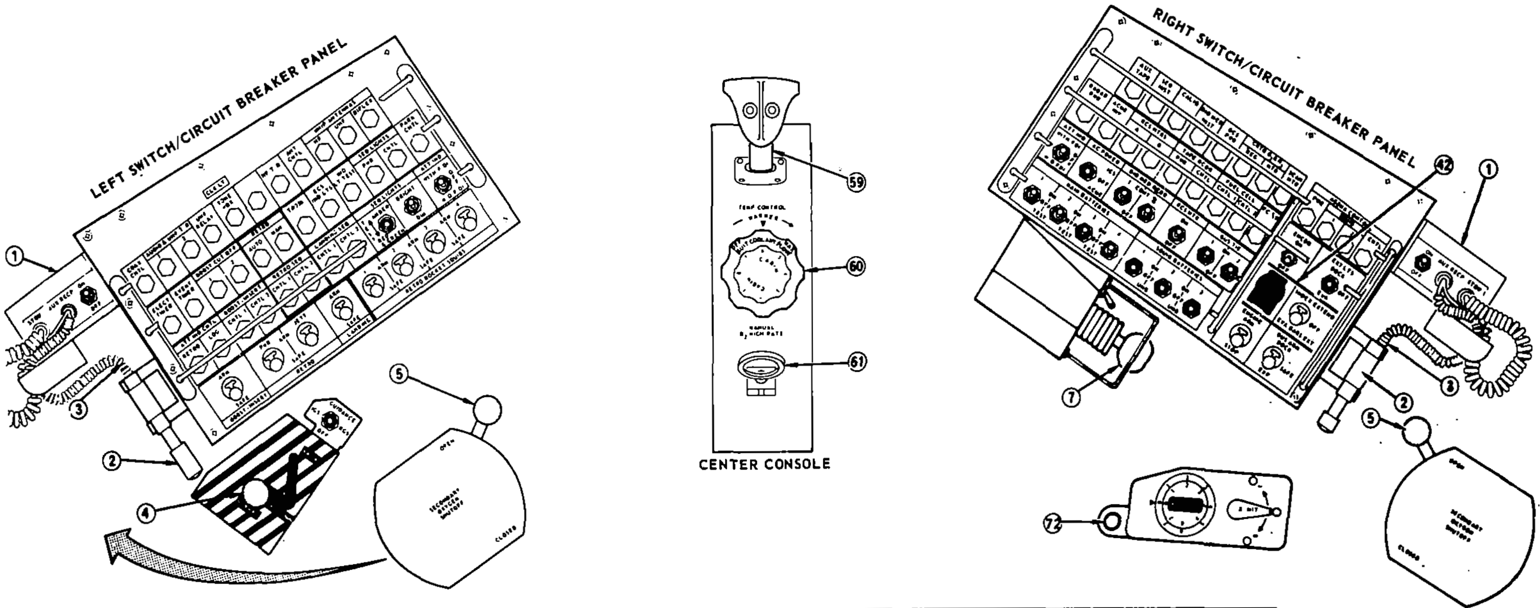


FIGURE 24 SPACECRAFT 12 CREW STATION DISPLAYS, CONTROLS AND FURNISHINGS



		CODE
1	UTILITY BRACKET	C
2	UTILITY LIGHT	C
3	*UTILITY CORD*	C
4	ABORT HANDLE INSTALLATION	C
5	SECONDARY OXYGEN SHUTOFF INSTALLATION	D
6	ALTIMETER	K
7A	MANEUVER CONTROLLER INSTALLATION	H
7B	MANEUVER CONTROLLER INSTALLATION (AUXILIARY)	H
8	INCREMENTAL VELOCITY INDICATOR	H
9	RANGE AND RANGE RATE INDICATOR	H
10	*EVENT TIMER*	P
11	SEPARATE ELECTRICAL TELELIGHT/ SWITCH	C
12	SEPARATE ADAPTER TELELIGHT/ SWITCH	C
13	ARM AUTO RETROGRADE TELELIGHT/ SWITCH	C
14	MANUAL FIRE RETROGRADE SWITCH	C
15	JETTISON RETROGRADE TELELIGHT/ SWITCH	C
16	PROPELLANT QUANTITY INDICATOR	L
17	RATE-OF-DESCENT INDICATOR	K
18	EMERG 10. 6K SWITCH	K
19	FLIGHT DIRECTOR CONTROLLER	H
20	ATTITUDE DIRECTOR/INDICATOR	C
21	GLV FUEL/OXIDIZER INDICATOR *STAGE I*	C
22	GLV FUEL/OXIDIZER INDICATOR STAGE II	C
23	SEPARATE OAMS LINES TELELIGHT/ SWITCH	L
24	LONGITUDINAL ACCELEROMETER AND RESET BUTTON	C
25	RCS TELELIGHT/SWITCH	L
26	BATTERY POWER TELELIGHT	F
27	CRYOGENIC QUANTITY AND PRESSURE SWITCH	D, F
28	DC-DC CONVERTER SELECT SWITCH	H
29	SWIZZLE STICK INSTALLATION	C
30	INDICATE RETRO ATTITUDE TELE-LIGHT/SWITCH	C
31	MIRROR	C
32	SEPARATE SPACECRAFT TELELIGHT/ SWITCH	C
33	JETTISON FAIRING SWITCH	C
34	ECS QUADRANT	D
35	CABIN/SUIT TEMPERATURE INDICATOR	D
36	CABIN PRESSURE/SUIT CO_2 PARTIAL PRESSURE INDICATOR	D
37	SECONDARY O_2 PRESSURE INDICATOR	D
38	CRYOGENIC QUANTITY INDICATOR	D
39	ANNUNCIATOR PANEL	C
40	*AUXILIARY TAPE MEMORY UNIT*	*H*
41	DIGITAL ELAPSED TIME CLOCK	P
42	AGENA EMERGENCY RELEASE PUSH-BUTTON	H
43	PROPELLANT TEMPERATURE/PRESSURE INDICATOR	L
44	VOICE CONTROL CENTER	B
45	PROPELLANT MONITOR SELECT SWITCH	L
46	PROPELLANT QUANTITY INDICATOR CONVERSION TABLE	L
47	ACCUTRON CLOCK	P
48	24-HOUR GMT CLOCK	P
49	SPLASH CURTAIN CLIP	C
50	SPRING CLIP	C
51	WATER DISPENSER	D
52	URINE DISCONNECT ADAPTER	D
53	MANUAL DATA KEYBOARD	H
54	MANUAL DATA READOUT	H
55	POWER SYSTEM MONITOR	F
56	VOLTMETER SELECT SWITCH	F
57	DIGITAL ELAPSED TIME CLOCK POST LAMP	C
58	FUEL CELL PRESSURE DIFFERENTIAL SELECT SWITCH	F
59	ATTITUDE CONTROLLER	H
60	SUIT/CABIN TEMPERATURE CONTROL	D
61	MANUAL O_2 HIGH RATE CONTROL	D
62	PARACHUTE JETTISON SWITCH	K
63	LANDING ATTITUDE SWITCH	K
64	SUIT FLOW CONTROL INSTALLATION	D
65	REPRESSURIZATION CONTROL	D
66	EVA UMBILICAL DISCONNECT	D
67	PARACHUTE SWITCH	K
68	HIGH-ALTITUDE DROGUE SWITCH	K
69	ATTITUDE CONTROL SWITCH	H
70	COMPUTER SWITCH	H
71	PLATFORM SWITCH	H
72	MESSAGE ACCEPTANCE LIGHT	H
73	CABIN VENT VALVE PULL RING	D
74	ION SENSOR MODE SELECTOR SWITCH	G

CODE

- B COMMUNICATION AND TRACKING
- C CREW STATION DISPLAYS, CONTROLS, AND FURNISHINGS
- D ENVIRONMENTAL CONTROL SYSTEM
- F ELECTRICAL SYSTEM
- G EXPERIMENTS
- H GUIDANCE AND CONTROL SYSTEM
- K LANDING SYSTEM
- L PROPULSION SYSTEMS
- P TIME REFERENCE SYSTEM

= CIRCUIT BREAKER (52-79721)
= TOGGLE SWITCH (52-79705)
= LEVER-LOCK SWITCH (52-79705)
= INDICATOR LIGHT (52-79718)
= PUSHBUTTON SWITCH (52-79719)
TELELIGHT (52-79710)

FIGURE 24 SPACECRAFT 12 CREW STATION DISPLAYS, CONTROLS AND FURNISHINGS (Continued) excessively bright and annoying, i.e., fuel cell ΔP lights and computer start light. These lights were equipped with covers to reduce the light emitted. The start comp light was fitted with an adjustable polaroid cover.

Because of long-duration Gemini missions and increased intravehicular activity as well as anticipated extravehicular activity, switch and circuit breaker panel protectors were provided. All bus switches and critical function switches such as pyrotechnics employed lever lock switches, recessed and covered push buttons, or slide lock push buttons. Of the remaining switches and circuit breakers, a first-design attempt of solid plastic covers which sprang out of the way of the panel when a push button was actuated was abandoned due to excessive inadvertent actuation and breakage. The configuration that did fly on all Gemini Spacecraft was a guard arrangement with a bar pass over each row of switches and circuit breakers. Despite all the activity on Gemini flights, very few switches or circuit breakers were inadvertently operated.

Malfunction Detection System (MDS). - The MDS design (See Fig. 25) was

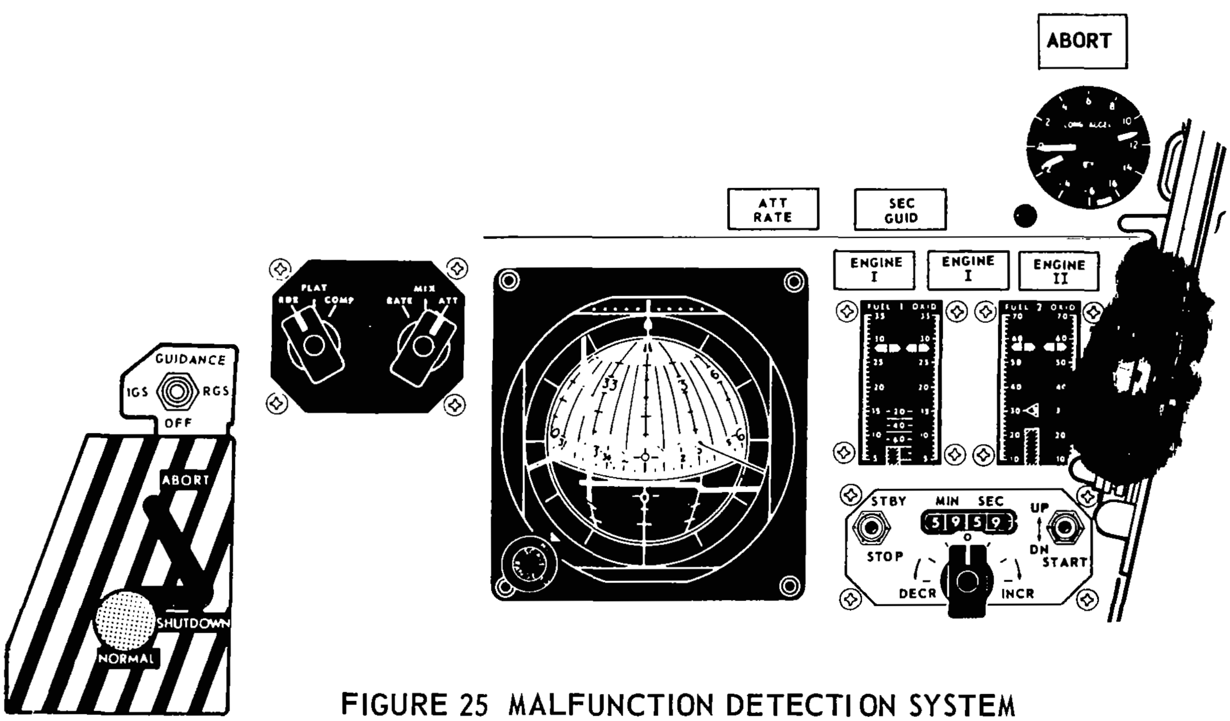


FIGURE 25 MALFUNCTION DETECTION SYSTEM

based on a detailed study of booster systems, Titan II flight histories and actual and probable malfunctions. It was decided that the crew would determine shutdown, ejection or abort in Modes I, II, or III, thus sufficient displays were provided to keep the crew aware of abort or impending abort.

Placement of the displays on the command pilot panel was such that at lift-off the command pilot could monitor all critical parameters - the two ENGINE I under pressure lights, ATTITUDE overrate indicator, ABORT light (ground command), fuel and oxidizer pressure needles, accelerometer and the Flight Director Attitude Indicator (FDAI). A minimum of two cues are required before an abort is initiated.

The first and second stage fuel and oxidizer pressure indicators contain redundant pointers powered from separate booster electrical sources. Power failure causes needles to fail up. A cross-hatched area on the center of the scales indicates minimum allowable tank pressures. The first stage indicator contains a time scale to show the minimum required pressure at 20, 40, and 60 sec after lift-off. The astronaut then is able to use both needle displacement and movement rates to predict an impending failure which may result in an abort or which may require crew action to save the mission. The S index on the right hand gauge indicates minimum allowable second stage fuel pressure at staging.

Engines 1 and 2 lights illuminate when booster thrust chamber pressure drops below 68%, and 57%, respectively. The lights thus indicate engine start, shutdown, and malfunction conditions. Normally the Engine 2 light will be on during first stage burning and go out after second stage ignition and the Engine 1 light will be out during first stage burning and on briefly at staging.

Attitude rate and guidance lights indicate whenever these errors exceed allowable values. Rates in excess of the following will illuminate the ATT RATE light:

	Stage I	Stage II
Pitch	$+2.0$ to -3.0°/sec	$\pm 10^{\circ}$/sec
Roll	$\pm 20^{\circ}$/sec	$\pm 20^{\circ}$/sec
Yaw	$\pm 2.5^{\circ}$/sec	$\pm 10^{\circ}$/sec

A display of digital time from launch provided the astronaut with a time reference for correlation of events during boost. An accelerometer was used with the timer for a gross indication of booster performance.

Both FDAI's effectively furnish the crew with "how-goes-it" information during launch and insertion. The command pilot monitors attitude rates for any deviation from normal rates while the pilot selects computer reference to ascertain differences between the launch vehicle and onboard guidance signals.

Landing Systems. - The landing system controls and associated instruments (See Fig. 26) have not changed throughout the manned portion of the Gemini program. Initiation of all sequences is entirely manual.

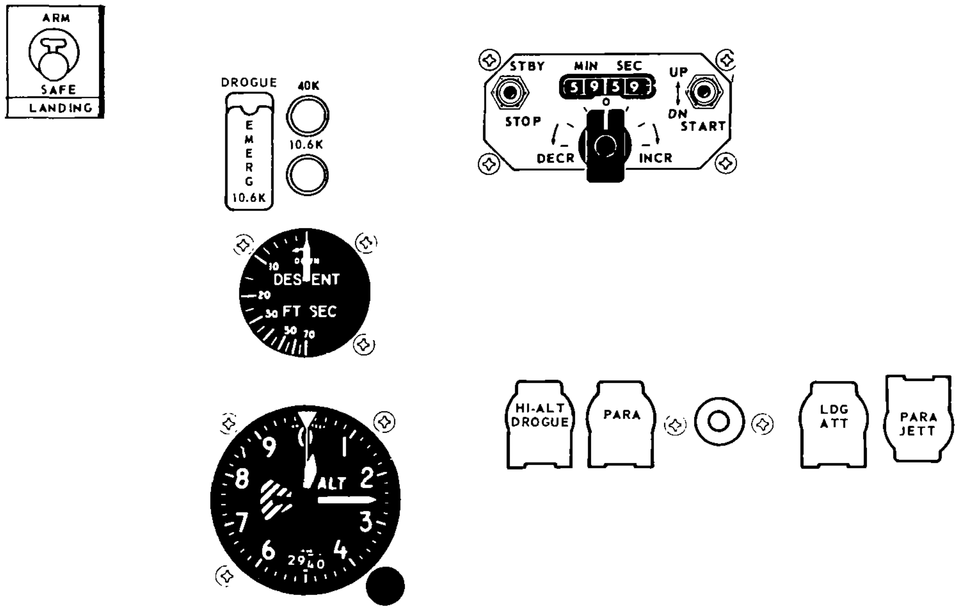


FIGURE 26 LANDING INSTRUMENT SYSTEM

Redundant altitude indications are provided in the form of an aircraft type barometric altimeter and lights operated by pressure sensing switches. The altimeter is a three-pointer type with a zero to 100,000-ft range. Although some astronauts would have preferred a counter-pointer type, this was rejected due to larger case size requirements and decreased high altitude sensitivity. Amber warning lights provided indicator back-up at 40,000 and 10,000 ft.

After re-entry, drogue and main chute deployments are initiated at 50,000 and 10,000 ft, respectively. In addition to using altimeter and lights, the astronauts could estimate altitude by aerodynamic stability characteristics, the computer program, and elapsed time from retrofire.

A rate of descent indicator, utilizing standard aircraft mechanical barometric indicator principles was included also. Although the condition of the main and drogue chutes was ascertained visually, this instrument allowed greater confidence in chute performance and provided better anticipation of events such as water impact.

A lever lock switch, used to arm the landing sequence functions, was initiated by push-button switches protected with covers to prevent inadvertent operation. The first button deployed the drogue chute. This pulled out and the pilot chute which, in turn, pulled the R & R can from the spacecraft. The second button deployed the main chute. Once the pilot established that the main chute had deployed, he pressed the third button to invert the spacecraft into the landing attitude. After touchdown the main chute was cut free by the last button on the landing sequence panel. If the drogue chute failed to deploy, an emergency switch to deploy it was provided on the command pilot's panel. The emergency switch cut the drogue chute free; the pilot chute was fired by its own mortar.

Guidance System. - Rendezvous and docking missions along with experiment pointing requirements demanded a more sophisticated guidance system on the Gemini than on the Mercury program. Controls and displays were arranged so that the command pilot primarily would control the spacecraft, while the pilot would function as systems monitor. The computer was designed for operation by the pilot with an incremental velocity indicator (IVI) on the command pilot's panel to show computer commanded translations in three axes.

The guidance system controls (platform, computer, attitude control mode selector) were located on the center panel accessible to either pilot though they ordinarily were used by the command pilot. With the central attitude control stick and dual maneuver controllers, guidance could be controlled by either pilot. Each pilot had selectable modes of display and reference inputs for his flight director indicator (FDI) as shown in Fig. 27.

A. Launch - Redundancy was the primed consideration in the design philosophy of launch displays. At lift-off the command pilot's FDI needles indicated booster guidance information while as a back-up the pilot monitored the Gemini computer guidance signals. In the event of excessive errors in the booster guidance system the command pilot selected the spacecraft platform/computer guidance system to control booster guidance. The mode selector on the computer panel was placed in ASCENT and the platform in FREE mode. All launch oriented controls and displays were evaluated for position and operability by the crew under the maximum g loads and vibration which might be encountered on launch.

B. Rendezvous - Development of guidance controls and displays was based on a requirement to rendezvous, independent of ground inputs utilizing radar, computer, and - for verification and back-up - manual computation. Computer controls were located in front of the pilot enabling him to make inputs into the computer and extract readouts from it while the commanded ΔV for orbital or planar corrections were displayed on the IVI for the command pilot. The displays allowed the pilot the option of monitoring FDI outputs that differed from those viewed by the command pilot; e.g., while the command pilot monitored computed rendezvous steering information, the pilot could monitor actual target aximuth and elevation angles from the radar.

The radar system, acting in conjunction with a cooperative transponder in the target, furnished range, range rate, azimuth and elevation to the computer in digital form and to panel displays in analog form. Target azimuth and elevation signals were displayed on the flight director indicator needles provided that the pilot selected RDR (Radar) on his flight director reference switch. Range and range rate were displayed on a dial on the command pilot's panel (See Fig. 28). Since range and range rate were critical as closer ranges, a special range sweep circuit was used in the radar to expand the sweep at closer ranges. The near range (zero to 3000 ft) was nine times more

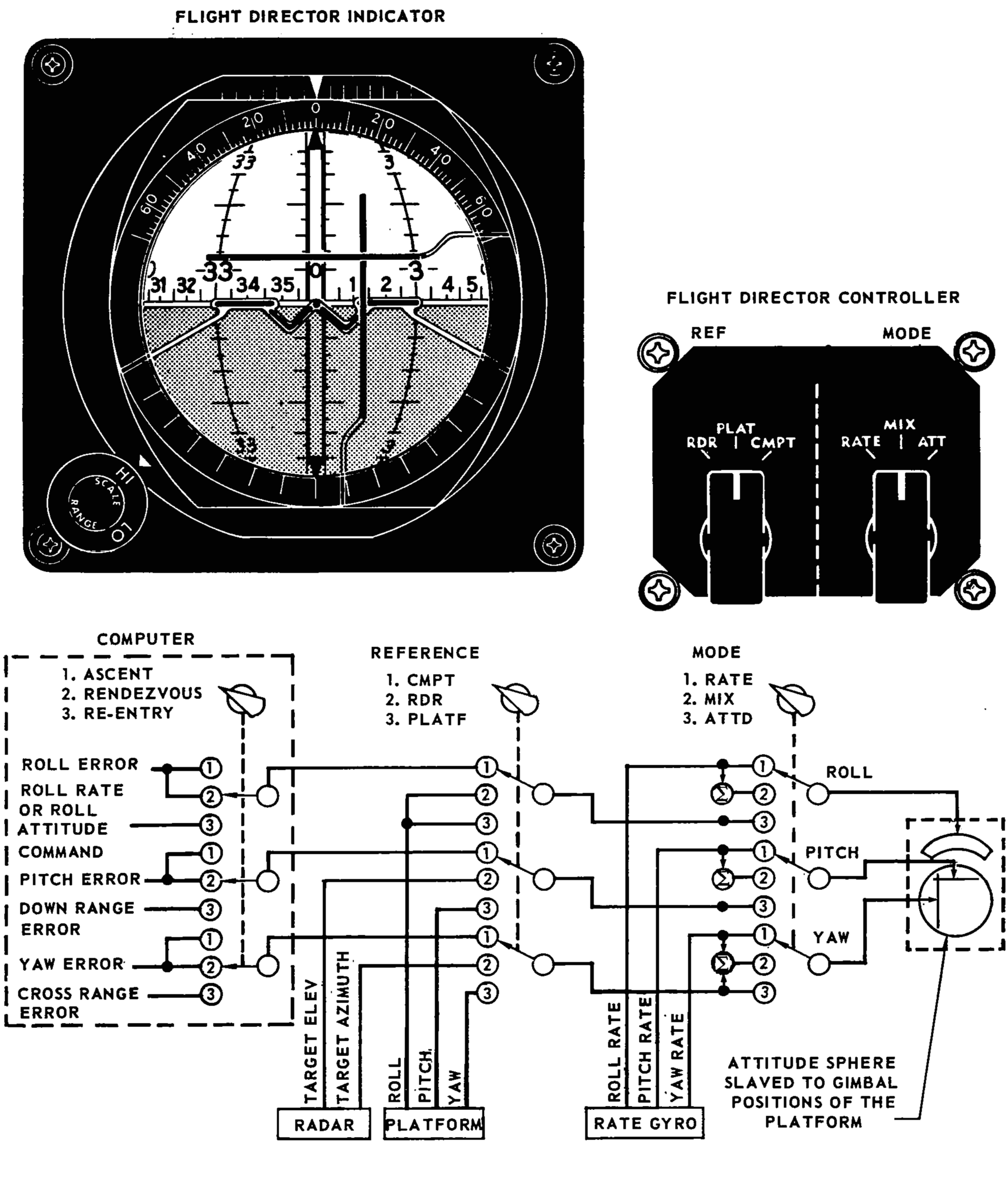


FIGURE 27 ATTITUDE DISPLAY GROUP

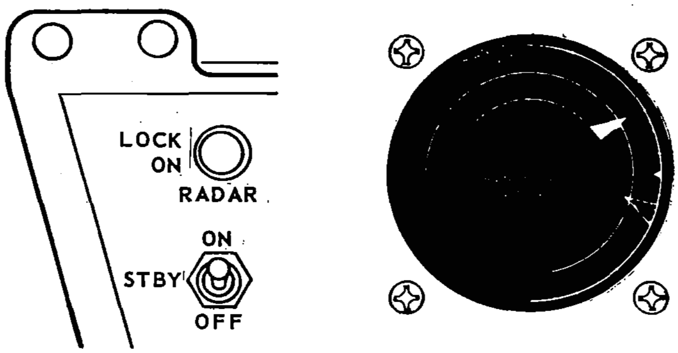


FIGURE 28 RADAR AND RATE INDICATOR

sensitive than the middle range (3000 ft to 30,000 ft) and 90 times more sensitive than the far range (30,000 ft to 300,000 ft). The needle jumped as it passed from one range to the other. With the exception of the vernier zero ±5 fps scale in the center which was beneficial in docking, the meter face was inadequate for the following reasons:

1. The range rate meter index point appeared to point to the range scale.
2. The point of the range index pointed to the range rate scale.
3. The broad portion of the pointers covered the numerals to be read, making extrapolation necessary on a nonlinear scale.
4. 250,000 ft was an impractical figure for pilots accustomed to miles at far ranges. Ranges in excess of 6000 ft should be calibrated in miles.

Three and four above introduced a time-consuming factor for extrapolation and interpretation while a simpler dial could be "scan read" by the command pilot during rendezvous when time is critical. Operational controls of the radar have been simplified to one OFF-STBY-ON toggle switch and a light to indicate lock-on.

Optical rendezvous could be accomplished, even with a radar failure, as on Gemini XII, by using the optical sight and back-up charts. The sight reticle mounting lugs were placed in the left hand window and aligned and inspected to within ±0.1 degree of the spacecraft longitudinal axis. The sight itself was guaranteed to within ±0.1 degree of its mounting points. All experiments, including radar, were aligned to within ±0.1 degree. In other words boresighting the spacecraft guaranteed ±0.3 degree accuracy with any experiment. Flight experience showed that these accuracies were well within the guaranteed ±0.3 degree and in most cases within ±0.1 degree.

C. Re-entry - Control provisions were provided for a man-in-the-loop re-entry control system as well as an automatic mode. Spacecraft 3 through 10 re-entries were controlled manually by the command pilot centering the FDI needles to maintain the proper attitude while the pilot extracted the predicted touchdown earth coordinates from the computer. Spacecraft 11, and 12 utilized the RE-ENT mode of computer.

Power Management Section. - The electrical system power management controls and displays were subjected to more changes and modifications than any other system on the spacecraft. Continuity was not achieved until Spacecraft 10, 11, and 12, for which power management controls and displays, were the same. Changes were dictated by introduction of fuel cells on Spacecraft 5 and by development work involving the fuel cell and its associated reactant supply system (RSS).

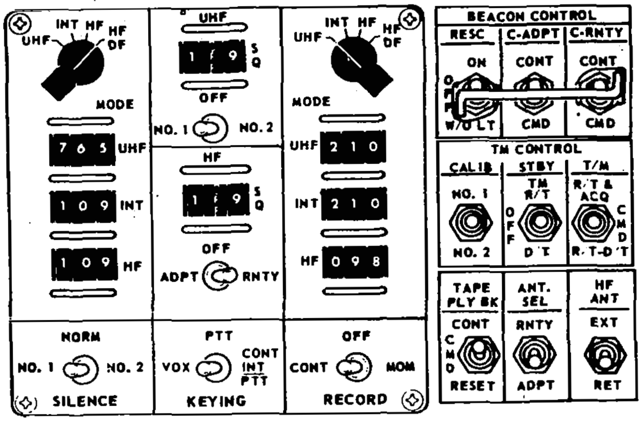
Table 18 lists every switch associated with the electrical power system monitoring and control. This figure shows the increasing need for monitoring more parameters and controlling more functions by the crew to preclude loss of a mission due to electrical power failure. The number of controls and displays of necessity were increased as the fuel cells were introduced, since some batteries were retained for redundancy.

The crew of Spacecraft 5 demonstrated what could be accomplished when provided with adequate controls to correct fuel cell/RSS malfunctions. The six vertical scale ammeter readouts for the six fuel cell stacks provided the best method for constantly monitoring the overall conditions of the cells and electrical loads. This enables the crew to observe deviations more rapidly rather than wait for a preset time to rotate a knob and check for deviations.

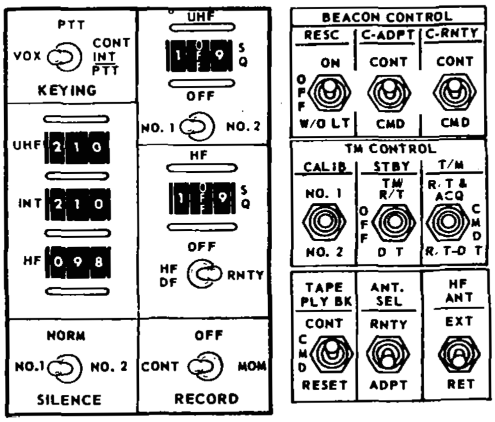
Communications Section. - The voice control center (VCC), located on the upper center console, contained the select and volume switches for voice communication on UHF, HF, and intercom. Beacon, telemetering and antenna control switches were located immediately to the right of the VCC. Switch type circuit breakers for the communication system were located in the top row of the left hand circuit breaker panel and in the left aft corner of the overhead circuit breaker panel.

Additional console space could be made available by reducing VCC size and by incorporating the functions of the audio mode switch into the UHF and HF select switches, and dual volume control switches into a single set. (See Fig. 29.)

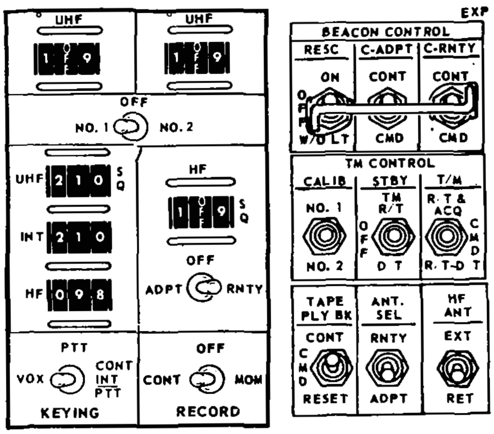
The audio mode switches on the VCC were positioned by the crew to select UHF, INT (intercom), HF, and HF-DF. The switches were independent so that while one crew member was transmitting and receiving on UHF, the other crew member could transmit and receive on HF and also receive on UHF. Selection of the HF-DF position on either switch initiated a 1000 cps tone which was audible to both crew members regardless of the position of the other switch. The INT position was selected to reduce the noise level in the respective headset. The RCD (record) position on this switch was used for the voice tape recorder on Spacecraft 3 and 4. Beginning with Spacecraft 5 it was removed and the function was incorporated into a toggle switch located in the lower right hand corner of the VCC.



PRESENT CONFIGURATION



PROPOSED CONFIGURATION



ALTERNATE PROPOSED CONFIGURATION

FIGURE 29 VOICE CONTROL CENTER AND BEACON, TM AND ANTENNA CONTROL SWITCHES

The volume control switches were located directly below the audio mode switches. Each crew member had individual wheel controls for UHF, INT, and HF. These switches operated quite satisfactorily and they required less panel space than rotary switches, which would have had to be large enough to be manipulated with a pressurized glove. However, due to limited panel space, one of the two sets of controls could be eliminated. On Spacecraft 7, removable plastic covers were provided to prevent moisture from getting into the controls. It is suggested that future applications of this type control be sealed against moisture.

The UHF and HF squelch and select switches located in the center of the VCC functioned satisfactorily throughout the program. Prior to Spacecraft 3,

TABLE 18 ELECTRICAL POWER SYSTEM MONITORING AND CONTROL SWITCHES

SPACECRAFT	BATTERY	FUEL CELL	AMP SELECTOR SWITCH	VOLT SELECTOR SWITCH	MAIN BATTERY SWITCH	ADAPTER BATTERY ISOLATION SWITCHES	CONTROL SWITCH	POWER SWITCH	ADAPTER BATTERY AMMETER	PRIMARY AMMETER
GEMINI III 3 ORBITS	(3) SQUIB (3) ADAPTER (4) MAIN 10 TOTAL		(7) POSITION ROTARY	(11) POSITION ROTARY	(4) ON OFF TEST	6 PUSH BUTTON TELELIGHTS 2 LITES CONNECTED IN PARALLEL TO (1) BATTERY	TWO	TWO	ALL READINGS MULTIPLIED BY 1.25	(2) SCALES MONITORS HALF THE FLOW IN ADAPTER BATTERIES & MAIN 1 & 2
GEMINI IV 4 DAYS	(3) SQUIB (6) ADAPTER (4) MAIN 13 TOTAL		NO CHANGE	NO CHANGE	NO CHANGE	(6) 1 LITE PER BATTERY	NO CHANGE	NO CHANGE	20 AMP SHUNT ALL READINGS 1 TO 1	(2) SCALES MONITORS FLOW IN ADAPTER BATTERIES 1A, 1B, 1C, AND MAIN BATTERIES 1 & 2
GEMINI V 8 DAYS	(3) SQUIB (4) MAIN 7 TOTAL	(2) SECTION 3 STACKS PER SECTION 32 CELLS PER STACK	NONE	NO CHANGE	NO CHANGE	(6) 1 LITE PER STACK	NO CHANGE	NO CHANGE	SIX AMMETERS TO MONITOR SIX FUEL CELL STACKS. MULTIPLY BY 1.25	(2) SCALES MONITORS FLOW IN FUEL SECTION 1 PLUS FLOW IN MAIN BATTERIES 1 & 2
GEMINI VI 1 DAY	(3) SQUIB (3) ADAPTER (4) MAIN 10 TOTAL		SAME AS 3 & 4	NO CHANGE	NO CHANGE	(6) SAME AS 3	NO CHANGE	NO CHANGE	SAME AS 4	SAME AS 3
GEMINI VII 14 DAYS	SAME AS 5	SAME AS 5	NONE	NO CHANGE	NO CHANGE	SAME AS 5	NO CHANGE	NO CHANGE	SAME AS 5	N/A
GEMINI VIII 6 ORBITS	SAME AS 5	SAME AS 5	NONE	12-POSITION AC VOLTAGE NOW INCLUDED ACME INVERTER AND IGS	NO CHANGE	NONE	STACK CONTROL SWITCHES-SIX STACK CONTROL RELAY	SECTION POWER SWITCHES-TWO ΔP TELELIGHT	SAME AS 5 EXCEPT READINGS AND PROCEDURES DIFFERENT	N/A
GEMINI IX 3 DAYS	SAME AS 5	SAME AS 5	NONE	SAME AS 8	NO CHANGE	NONE	SAME AS 8	SAME AS 8	SAME AS 8 EXCEPT READINGS ARE DIFFERENT	N/A
GEMINI X 3 DAYS	SAME AS 5	SAME AS 5	NONE	SAME AS 8	NO CHANGE	NONE	SAME AS 8	SAME AS 8	SAME AS 9	N/A
GEMINI XI 3 DAYS	SAME AS 5	SAME AS 5	NONE	SAME AS 8	NO CHANGE	NONE	SAME AS 8	SAME AS 8	SAME AS 9	N/A
GEMINI XII 4 DAYS	SAME AS 5	SAME AS 5	NONE	SAME AS 8	NO	NONE	SAME AS 8	SAME AS 8	SAME AS 9	N/A

TABLE 18 ELECTRICAL POWER SYSTEM MONITORING AND CONTROL SWITCHES (Continued)

SQUIB ARM SWITCH	VOLT METER	FUEL CELL DIFF. PRESS. WARNING LITE	FUEL CELL PURGE SWITCHES	FUEL CELL RSS X-OVER	BATTERY POWER TELELIGHT	O_2 CROSS FEED SWITCH	FUEL CELL PRESS. DIFF. SELECT SWITCH	DIFF. PRESS. METER	H_2 TANK VACUUM SWITCH
(3) SAFE ARM A. BOOST, INSERT ABORT B. RETRO POWER C. LANDING & POST LANDING	READS UP TO 50 VOLTS AT SPECIFIED POINTS				LITE ILLUMINATES AMBER AT 5 MIN. PRIOR TO RETRO BATTERY ON LITE GREEN				
(8) A. BOOST INSERT B. RETRO POWER C. RETRO ROCKET (4) D. RETRO JETT E. LANDING (INCLUDING RECOVERY	NO CHANGE				NAME CHANGE TO BATTERY POWER SEQUENCE LITE. COMES ON AT TR-256				
(9) A. BOOST INSERT B. AGENA BUS C. RETRO POWER D. RETRO ROCKET (4) E. RETRO JETT F. LANDING (INCLUDES RECOVERY)	TWO VOLT SCALES – AC & DC. AC PORTION INOPERATIVE. DC VOLTS USE A SELECTOR SWITCH SCALE READS FROM 18–33	(3) FCΔP ON ANNUNCIATOR FCΔP SECTION 1 FCΔP SECTION 2	(2) H_2 OFF O_2	H_2 OR O_2 TO EITHER OR BOTH FUEL CELL SECTIONS REGULATORS	SAME AS 4				
(9) SAME AS 5	SAME AS 3	N/A	N/A	N/A	SAME AS 4				
SAME AS 5	SAME AS 5	SAME AS 5	SAME AS 5	SAME AS 5	SAME AS 4	PRESSURIZE TSS O_2 FROM ECS O_2 OR VICE VERSA			
SAME AS 5	SAME AS 5 EXCEPT AC HOOKED UP	(2) ON ANNUNCIATOR PANEL	SAME SYSTEM AS 5 EXCEPT HAS LOCKS FOR HOLDING DOWN	SAME AS 5	SAME AS 4	SAME AS 7	5 POSITION ROTARY	USED IN CONJ. WITH AMMETER TO DISPLAY FUEL CELL ΔP	
SAME AS 5	SAME AS 8	SAME AS 8	SAME AS 8	SAME AS 5	SAME AS 4	SAME AS 7	SAME AS 8	SAME AS 8	
SAME AS 5	SAME AS 8	SAME AS 8	SAME AS 8	SAME AS 5	SAME AS 4	SINGLE O_2 BOTTLE	SAME AS 8	SAME AS 8	PINCH OFF TUBE RSS H_2 TANK
SAME AS 5	SAME AS 8	SAME AS 8	SAME AS 8	SAME AS 5	SAME AS 4	SAME AS 10	SAME AS 8	SAME AS 8	SAME AS 10
SAME AS 5	SAME AS 8	SAME AS 8	SAME AS 8	SAME AS 5	SAME AS 4	SAME AS 10	SAME AS 8	SAME AS 8	SAME AS 10

an HF transmitter-receiver (T/R) was planned for installation in the adapter section in addition to the one in the re-entry section. However, due to the poor HF transmission qualities, the T/R was replaced by another HF whip antenna in the adapter. The HF select switch remained as a three-position switch with two unused positions.

The audio mode switch functions could be relocated to the UHF and HF select switches as follows:

AUDIO MODE FUNCTION	UHF SELECT SWITCH	HF SELECT SWITCH
A. UHF	to No. 1 or No. 2	
B. INT (for CP)	to position now used for OFF	
C. INT (for P)		to center position now used for OFF
D. HF		to RNTY
E. HF-DF		to position formerly designated for ADPT

Incorporation of the above changes would make additional panel space available for other controls and/or displays.

The silence switch was incorporated on the lower left hand corner of the VCC beginning with Spacecraft 5. In NORMAL position there was no change to the communications capabilities. In either the No. 1 or No. 2 position all UHF, HF, and INT receiving capability was removed from the respective headset. This addition enabled one crew member to sleep or rest without any radio interference. However, beginning with Spacecraft 6 the flight plan called for simultaneous crew rest periods which limited the use of this switch, making its usefulness questionable.

The keying switch, located in the lower center part of the VCC, had three positions: VOX, PTT, and CONT INT-PTT. In the VOX position either crew member communicated with ground stations as well as each other without pressing the switches on the attitude hand controller. In the PTT position, the switches on the attitude hand controller or pressure suit electrical umbilical were actuated (1) for communication with ground stations when the respective audio mode switch was in UHF or HF position, and (2) for conversation between crew members when the respective audio mode switch was in INT position. In the CONT INT-PTT position the crew members conversed without actuating any switches. However the switches on the attitude hand controller or suit electrical umbilical switch were pressed for communication with ground stations. This last position was the only one connected to the voice tape recorder. The VOX position, though rarely used on flights, could be utilized during an emergency that would preclude the use of the switches on the attitude hand controller. The PTT position had little utilization and its importance in an emergency situation is nebulous. The CONT INT-PTT position was used most because it afforded the convenience desired most by the crew members. Depending on the degree of importance placed on the features of the VOX position, the keying switch could be eliminated from the VCC and features of the CONT INT-PTT position could be incorporated into the VCC without the switch.

The record switch located in the lower right hand corner of the VCC is for the voice tape recorder. As mentioned previously, this switch was added prior to Spacecraft 5, giving the crew greater control capability with its MOMENTARY and CONTINUOUS positions. To operate the recorder, the record switch was placed in either the CONT or MOM position, the keying switch in CONT INT-PTT position, and the audio mode switch in the UHF, HF, or HF-DF position.

Beacon Control Switches - These switches were located immediately to the right of the VCC. The RESC (rescue) switch was labeled ON-OFF W/O LT. In the ON position the UHF rescue beacon and the flashing recovery light were operative, but in the W/O LT position only the UHF rescue beacon was operative provided the Vm bus was operative. To avoid possible transmitter damage, the switch was not actuated from the OFF position until the rescue antenna had been extended at main chute single point release.

Spacecraft 3 had one S-band beacon located in the equipment adapter section and one C-band beacon located in the re-entry section. Beginning with Spacecraft 4 the S-band beacon was replaced with a C-band beacon. The C-ADPT and C-RNTY switches were two-position switches labeled CONT and CMD. In CONT position the beacon was powered continuously, but this position did not disable DCS command "on" capability. The CMD position placed the beacon in a standby mode until a ground DCS command was sent to activate the beacon. The C-band helix antenna (re-entry) phase shifter was powered by C-RNTY switch provided that the ANT SEL switch was in the RNTY position. Only one beacon was on at any given time, since both beacon characteristics were the same.

The configuration and operation of these switches were satisfactory throughout the program.

Telemetry Control Switches - These switches were located directly below the beacon control switches. The location and arrangement of these switches did not change throughout the program.

The CALIB switch has three positions: No. 1, OFF, and No. 2. Position No. 1 initiated calibration of low calibrate point (20% full scale signal) of ECS O_2, FC O_2, and FC H_2 quantity and pressure gauges, and a one-point calibration of 20 other parameters. Position No. 2 initiated calibration of high calibrate point (80% full scale signal) for the same items as No. 1.

The STBY switch had three positions: R/T, OFF, and D/T. In R/T (real time) position the standby T/M transmitter was powered for R/T transmission and R/T T/M data was transferred from the PCM Programmer to the standby

T/M transmitter. In the OFF position there was no power to the standby transmitter. In the D/T (delayed time) position the standby T/M transmitter was powered for D/T transmission. This position also enabled tape playback control through TAPE PLYBK switch in CONT and DCS when TAPE PLYBK switch was in CMD.

The T/M switch had three positions:

A. When in R/T and ACQ position, the switch powered R/T T/M transmitter continuously (the R/T T/M transmitter was always connected to the quadriplexer), activated the acquisition aid beacon and connected the beacon to the diplexer.

B. In the CMD position, the T/M switch enabled DCS to command R/T and D/T T/M transmitters on, disconnect the acquisition aid beacon from the diplexer and connect D/T T/M transmitter to the diplexer. It also activated the acquisition aid beacon when the D/T T/M transmitter was off and connected the acquisition aid beacon to the diplexer.

C. In the R/T D/T position, the switch powered the R/T T/M and D/T T/M transmitters continuously and connected the latter to the diplexer. The T/M switch also enabled tape playback control through the TAPE PLYBK switch in CONT and DCS when TAPE PLYBK switch was in CMD.

On Spacecraft 6 and up the TAPE PLYBK switch was located directly under the T/M CALIB switch. On Spacecraft 3 through 5 it was located in the same group with the ANT SEL and HF ANT, but the order of position varied with each spacecraft. In the CONT position the PCM T/M recorder played back continuously through the T/M control T/M switch. In the CMD position the recorder played back (dumped) when commanded on by DCS. On Spacecraft 6 and up the RESET position was used to reset DCS relays 1 through 8. Calibration (Channel 9) and abort (Channel 10) were not reset. Spacecraft 3, 4, and 5 did not have this reset capability.

The ANT SEL (antenna select) switch was a two-position switch located directly below the T/M control switches. In the RNTY position the re-entry C-band beacon phase shifter was powered (when C-band beacon is on), and the UHF stub antenna (re-entry R & R section) was connected to the quadriplexer through the descent antenna coax switch. In the ADPT position the UHF whip antenna (retroadapter) was connected to the quadriplexer.

The three-position HF ANT select switch was located on the upper center console with the TAPE PLYBK and ANT SEL switches. In the EXT position the re-entry HF whip antenna was extended through the LANDING switch in ARM position. The adapter HF whip antenna extended through the LANDING switch (SAFE), and the HF T/R connected to the extended antenna. In the OFF position power was removed from the antenna motors. In the RET position either extended HF whip antenna was retracted. During docking or during Agena burns while docked, the HF whip antenna should be retracted to prevent possible antenna breakage.

Attitude Display. - A simulation program was established to determine the best approach to an attitude reference display duplicated on each side of the cockpit. The display most acceptable to pilot/astronauts was essentially the same type attitude ball found on contemporary fighter aircraft with some modification. Each display incorporated a three-axis attitude reference ball having 360 degrees of rotation about each axis. The attitude reference balls were slaved to the position of the inertial platform gimbals to provide a continuous all-attitude reference of roll, pitch and yaw. A needle at the top of the indicator provides roll rate indication in one degree/sec increments to full scale needle deflection of five degrees. The pitch reference scale marking was changed from five degree increments to one degree increments effective on Spacecraft 6. Pitch readouts became critical as inputs were needed for pilot manual computation in certain rendezvous malfunction modes with one-half degree or better pitch reference.

The attitude ball was selected for Gemini because it provided a simplified method of using gimbal position information from the stable platform as an all-attitude display. The sphere through mutual interaction of the axes provides an unambiguous, all-attitude display which is a tremendous improvement over the separate axis Mercury-type display.

An example of the lack of mutual interaction in a Mercury-type display was shown by the following maneuver: Yaw to 180 degrees, then pitch to 180 degrees. This resulted in a roll of 180 degrees with a yaw and a pitch. Since the Mercury display elements were single plane the interaction did not occur. However, in rotating a sphere 180 degrees in yaw, and 180 degrees in pitch as in the Gemini display, it provided its own interaction and displayed the effect as 180 degrees roll.

In event of a partial or complete platform failure, considerable simulation was accomplished using a star background and reticle for attitude reference. Based on the simulations, it was considered feasible to rendezvous despite a platform failure, though it was more difficult and the probability of error was increased.

The inertial guidance system (IGS) selected for the Gemini was a four-gimbal system utilizing an inner roll gimbal with ± 15 degrees freedom, in addition to the 360 degrees outer roll gimbal to prevent gimbal lock at the zero degree yaw, zero degree pitch, 90 degree roll attitude. The stable element contained the gyros, accelerometer, servo loops and alignment prism. The system was designed so that the spacecraft could be aligned in flight by aligning the blunt end forward or small end forward with the horizon scanner and caging the platform. Guidance System, page 179 describes operational controls of the IGS system. No major problems were encountered with the IGS/attitude display combination which proved accurate and reliable in orbital flight.

<u>Flight Control Console.</u> - The flight control console included the manual attitude controller and the control mode select switch. Studies on the manual controller included location of the controllers, loads versus displacement of the controller, and determination of optimum movement about the axes.

Initially, rudder pedals were envisaged; however, weight and space limitations forced abandonment of pedals in favor of placing a third axis on the manual controller. Either crew member could operate the controller while in the restrained position through wrist articulation and palm pivot motion only, to preclude body movements from being transmitted to the controller. The handle was spring loaded to provide an increasing resistance as the handle was moved away from neutral. Controller force/displacement originally had a step function designed in all three axes, but was later revised to a smooth curve as shown in Fig. 30 for all three axes. Redundant switches were incorporated for selectively energizing solenoid valves in the attitude control system. Total travel of the hand controller was 10 $\pm$ 1 degrees from neutral in pitch and yaw axes and 9 $\pm$ 1 degrees in the roll axis. Rotary movement of the handle about a transverse axis located at the palm pivot point effected a corresponding spacecraft motion about the pitch axis. Rotary displacement in a clockwise or counterclockwise direction in a transverse plane with respect to an adjustable canted axis below the pilot's wrist effected a similar movement about the spacecraft roll axis. Clockwise or counterclockwise rotation of the controller about the longitudinal axis of the handle effected a corresponding movement about the yaw axis.

Due to extended operation in this mode, the resistant stick forces tended to cause wrist fatigue. Consequently, the control stick was modified to assimilate a T at the top. This enabled the pilot to grasp the top of the stick palm down if desired for more ease of yaw control. A guard was built up on the top to prevent depressing the communications transmit buttons while grasping the stick in this manner.

Evaluation of the many attitude controller designs included operation of the stick with a bare hand, a soft glove or a pressurized glove, as well as consideration of the man pressurized or unpressurized, in zero g or under heavy re-entry g loads.

The attitude controller (See Fig. 31) worked in conjunction with a rotary mode selector slightly forward and left of the stick. This was needed to allow the pilot minimal three-axis response for fine maneuvering such as docking (pulse) or larger orders of magnitude in response for gross corrections (rate command or direct).

The modes made available to the pilot were:

A. HOR SCAN - The horizon sensors provided a reference in pitch and roll to automatically control a limit cycle mode ± 5 degrees in these axes. The yaw axis was maintained by the pilot using the pulse mode which was maintained on all axes in this mode.

B. RATE CMD - Pitch, roll and yaw rate gyro outputs were compared with controller positions to produce attitude rates proportional to controller deflection. (Operationally, this mode was effective in correcting the fairly high cross-coupling rates developed when the maneuver controller was used to translate.)

C. DIRECT - Provided direct control to open thrust chamber solenoids when the attitude controller was deflected approximately 25% of full travel. (The

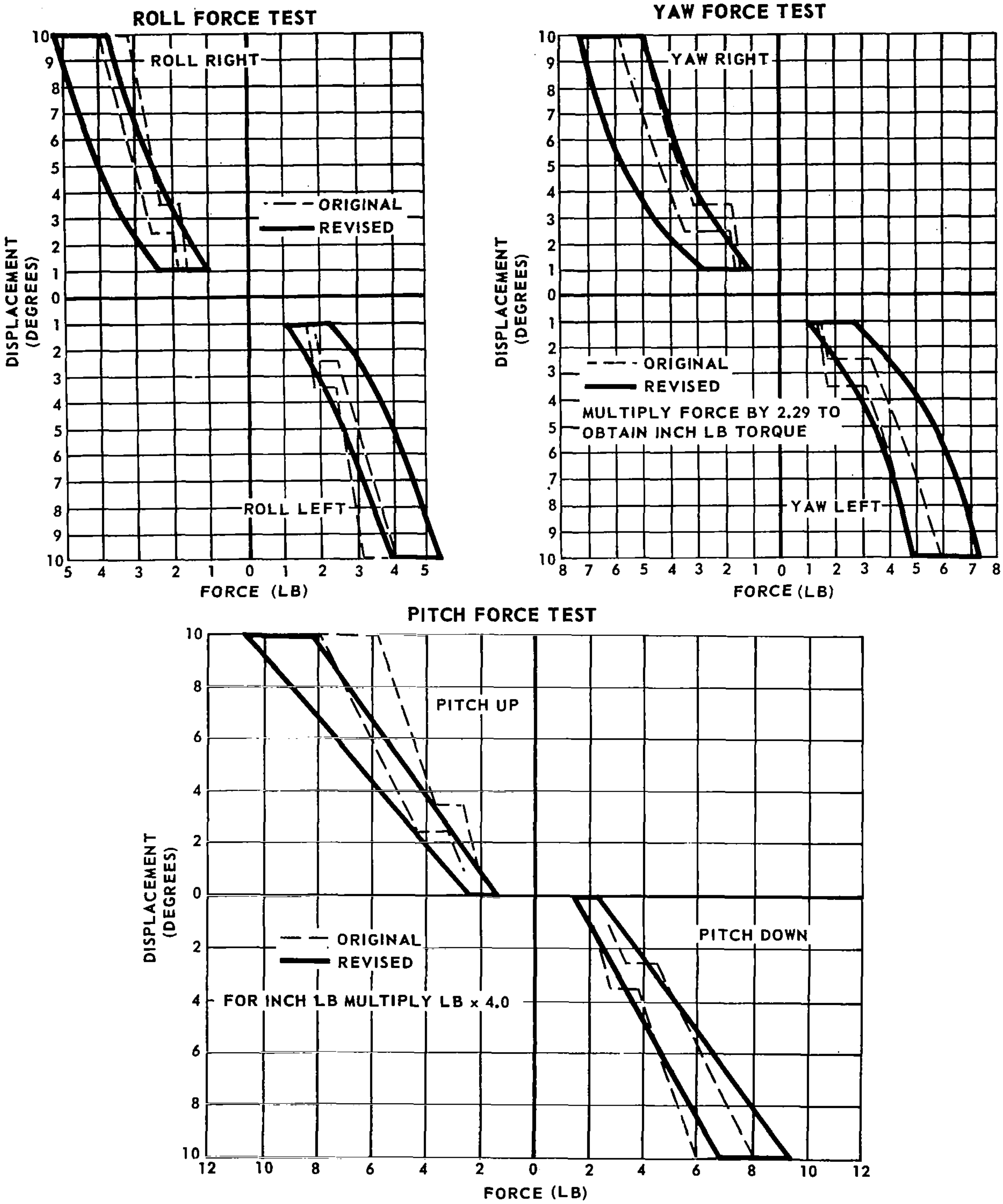


FIGURE 30 ATTITUDE HAND CONTROLLER FORCES
(ORIGINAL AND REVISED LIMITS)

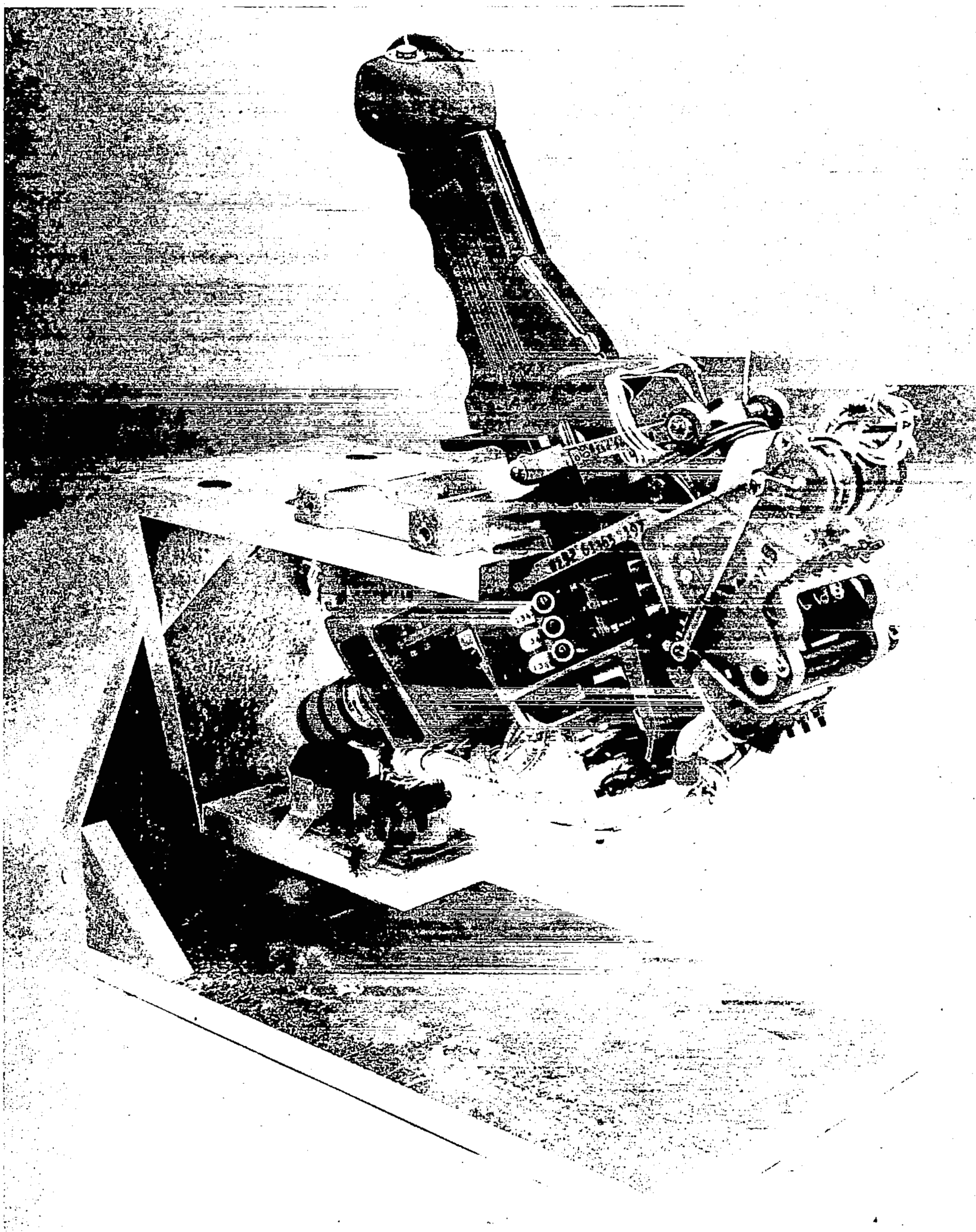


FIGURE 31 ATTITUDE HAND CONTROLLER

utmost discretion was used in this mode, as it tended to waste fuel.)

D. PULSE - For each deflection of the controller away from the center position, a single short duration (20 msec) pulse was applied to the appropriate axis.

E. RATE CMD, RE-ENT - Similar to rate command with a wider neutral band and gain crossfeed from roll to yaw. (Designed for use in manual re-entry.)

F. RE-ENT - Pitch and yaw axes in rate damping control mode, with roll axis slaved to bank-angle command from the computer.

G. PLAT - ACME accepted attitude information from the platform and provided outputs to the thrusters to maintain spacecraft attitude automatically within pitch, yaw and roll deadbands.

H. PARA - A mode designed for use with a paraglider which was eliminated before the first manned flight. (On Spacecraft 5 and up, this selector position was used for the PLATFORM mode.)

Environmental Control Section (ECS). - The control and displays for the environmental control system are shown in Fig. 32. Several significant changes in the ECS (See ENVIRONMENTAL CONTROL SYSTEM, page 157) are reflected on the panel. The overhead manually-operated ECS handles were the same on all spacecraft. A redundant valve was added to back up the vent valve on Spacecraft 7 and up. On Spacecraft 7 it was merely a cap that was pulled off the valve opening with a lanyard, but on Spacecraft 8 and up it was a spring-loaded permanent installation that could be recycled for EVA. The electrical switch to activate O_2 high rate was located on the sequence panel on Spacecraft 3 and 4, but was determined by test not to be a sequential requirement. When the function was removed from the sequential panel and placed on the lower position of the cabin fan switch on Spacecraft 5 and up, the associated light in the sequence panel was placed on the annunciator panel. The manual O_2 high rate pull ring located on the center pedestal between the seats did not change throughout the Gemini series. The suit fan switch positions changed from No. 1 and No. 2 to No. 1 and No. 1 and No. 2 (to run both fans simultaneously) on Spacecraft 4 and up in order to get greater circulation in the latter position. The cabin fan was removed in Spacecraft 8 and up and other functions placed on its switch position. The remainder of the ECS switches and warning lights on the upper center panel remained the same in all spacecraft although the functions varied slightly. Spacecraft 6 had one coolant pump in each loop in order to save weight since it was a one-day mission and of the battery configuration.

Pump B in each loop also was configured to a lower flow rate to conserve power on Spacecraft 5 and up for periods of low activity and power down. All the gauges associated with the ECS remained the same on all spacecraft except that a rotary selector was added to read the cryo gauge when fuel cells were added to Spacecraft 5 and up so that fuel cell O_2 and H_2 could be read in addition to ECS O_2. A crossfeed switch was added to Spacecraft 7, 8 and 9 in order to supply ECS O_2 to the fuel cells and vice versa. This action was taken as a result of the FC O_2 cryo heater failure on Spacecraft 5. Spacecraft 10 and up had only one O_2 supply for both ECS and fuel cells. A tube cutter was added to the H_2 cryo vacuum bottle on Spacecraft 7 and up to improve the thermal characteristics. This cutter was activated by the crossfeed switch with the experiments bus armed but was labeled separately on

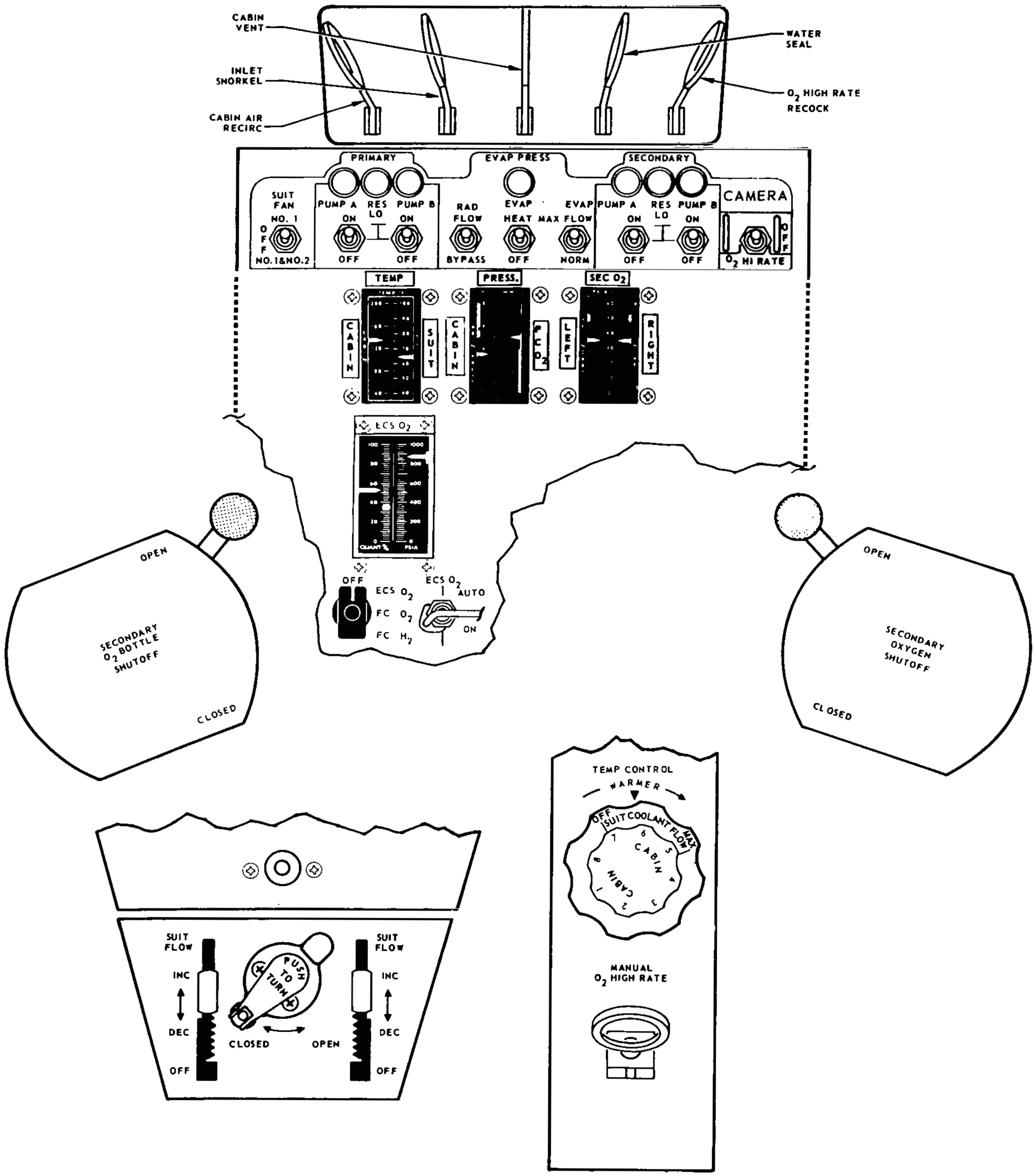


FIGURE 32 ECS CONTROLS AND DISPLAYS

Spacecraft 10 and up. The manual heater in the O_2 bottle was originally a momentary switch but was changed to a fixed position and resulted in adding a light ECS HTR to the annunciator panel. The temperature selector on the center console served the same function throughout the program but, beginning with Spacecraft 8 it was marked "minimum" and "maximum" because the flow was controlled over such a short valve travel. The center pedestal items did not change except for the addition of a positive lock to the repressurization valve. The repressurization oxygen was ducted through the console to a quick disconnect so it could be used with an EVA umbilical.

Propellant Quantity Indication. - As originally conceived, the measurement of OAMS propellant quantity was based on display of temperature and pressure of the helium OAMS fuel pressurant from which the crew would interpolate quantity by utilizing a graph curve. To avoid this inconvenience and provide a constant quantity readout, an instrument and associated circuitry were designed which would integrate the temperature and pressure parameters and present a zero to 100% OAMS quantity indication.

A pressure transducer furnished an input to one side of a bridge while a sensitive platinum element provided a temperature variable input to the other side, resulting in an integrated signal to an indicator (See Fig. 33) with

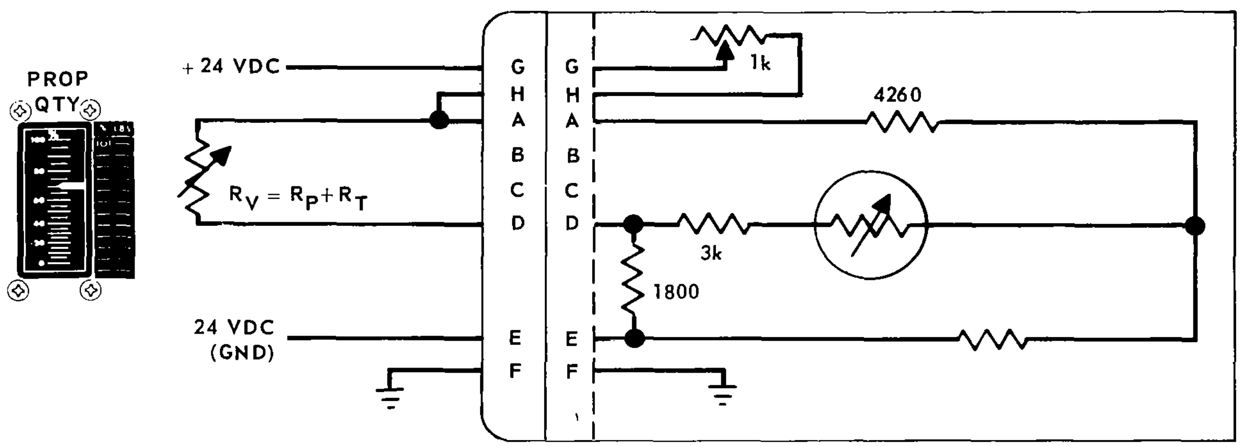


FIGURE 33 OAMS PROPELLANT QUANTITY INDICATOR

internal variable resistance to compensate for fuel loadings which vary between spacecraft. Uneven fuel/oxidizer burning ratios principally accounted for the ±10% accuracy of the present gauge.

Agena Status Display Panel. - When docking was established as a mission requirement, McDonnell assumed the design responsibility for a status display panel for the Agena target docking adapter (TDA).

Due to space limiations in the Gemini cockpit, the panel was designed for installation on the Agena so as to be visible and functional to astronauts in the Gemini cockpit.

No new design problems were involved because the instruments were to be subjected to almost the same environment when the hatches were opened and the cockpit depressurized for EVA.

This type of installation shown in Fig. 34 probably was the first

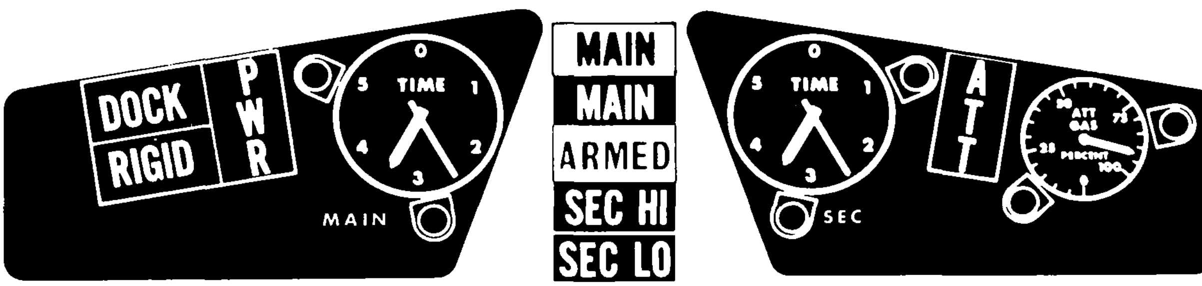


FIGURE 34 AGENA STATUS DISPLAY PANEL

utilized in this manner and proved highly beneficial because of savings in Gemini cockpit instrument panel space, spacecraft weight and related electrical wiring.

Qualification tests proved uneventful with all display indications and lights passing the predicted testing requirements.

The ASC panel was mounted atop the forward end of the TDA so that it was visible to the astronauts during the docking sequence and during subsequent maneuvers in the docked configuration. The display consisted of nine lights and three analog dials that provided the following data:

A. TDA docking system ready for docking.
B. TDA rigidized.
C. Agena Target Vehicle (ATV) electrical power system operating properly.
D. Amount of PPS burn time remaining.
E. PPS safety status (main red).
F. PPS pressure status.
G. Status of PPS and SPS thrust allow - thrust stop circuit.
H. SPS gas pressure status.
I. Amount of SPS burn time remaining.
J. Vehicle under control of Agena attitude system.
K. Amount of attitude control gas remaining.

With the exception of the main red, the attitude control system (ACS) gas pressure display (ATT gas dial) and the propulsion system clocks, the ASD panel was deenergized until activated by ground or spacecraft command. The ACS gas pressure display and the ΔP portion of the main red light were always energized and the propulsion system clocks were energized whenever the applicable engine fired.

Two levels of illumination, bright and dim, were available to all displays except the main red which was either off or on bright.

Dock (green light) - When illuminated, the docking cone was unrigidized, the latches were reset and the TDA was in the proper condition to receive the nose section of the spacecraft.

Rigid (amber light) - When illuminated, the docking cone was rigidized and ready to allow post-docking maneuvers.

Pwr (green light) - When illuminated, the 28 volts DC unregulated 28 volts DC regulated, -28 volts DC regulated, 115 volts, 400 cps, single-phase and 115 volts, 400 cps three-phase parameters were operating within tolerances.

Main (red light) - When illuminated, indicated the following:

A. When the ASD was activated:

1. When the main engine was firing, the turbine exceeded 27,000 RPM, the hydraulic pressure was below 1500 $\pm$ 20 psia or the differential pressure between the fuel and oxidizer tanks was below 3 $\pm$ 2 psid, fuel above oxidizer.

2. When the main engine was not firing, the differential pressure between the fuel and oxidizer tanks was below 3 $\pm$ 2 psid, fuel above oxidizer.

B. When the ASD was deactivated: the differential pressure between the fuel and oxidizer tanks was below 3 $\pm$ 2 psid, fuel above oxidizer.

Main (green light) - When illuminated, the main fuel tank was above 15 $\pm$ 2 psia, the main oxidizer tank above 15 $\pm$ 2 psia, the hydraulic system pressure above 50 $\pm$ 5 psia, and the PPS safely commanded to fire.

Armed (amber light) - When illuminated, the engine control circuits were closed and either the main or secondary engines were commanded to fire.

Sec Hi (green light) - When illuminated, more than 1110 $\pm$ 20 psia expulsion gas pressure existed in both nitrogen spheres for a 50 sec unit II (200 lb SPS thrust chamber) firing and more than 170 $\pm$ 5 psia regulated pressure existed in both propellant tank gas manifolds.

Sec Lo (green light) - When illuminated, more than 360 $\pm$ 20 psia expulsion gas pressure existed in both nitrogen spheres for a 150 sec unit I (16 lb SPS thrust chamber) firing and more than 170 $\pm$ 5 psia regulated pressure existed in both propellant tank gas manifolds.

ATT (green light) - When illuminated, the Agena attitude control system was active.

Main time (clock display) indicated by minute and second hands the time remaining for main engine burn. The regulated 28 VDC was applied to the main time display unit when the start signal was applied to the main engine. This caused the display unit to decrease the time remaining indication at a rate of one sec/sec burn time.

Sec Time (clock display) indicated by minute and second hands the time remaining for 200 lb SPS burn. The regulated 28 VDC was applied on separate wires for high and low thrusters of the SPS. This caused the display unit to decrease the time remaining indication at a rate of one sec/sec burn time for the high consumption and a rate of one-twelfth (1/12) sec/sec burn time for the low consumption rates.

ATT Gas (synchro display) indicated the percentage of total pressure remaining in the Agena attitude control system gas spheres.

Spacecraft 8 produced two minor problems, the first being the difficulty in distinguishing the docking light illumination from the rigid light because of their close proximity and similar color (green). The second became evident during the daylight pass. At 50 to 75 ft (the station keeping range) it became impossible to read either lights or gauges due to the sunlight reflecting off the lenses. The acquisition of a film, probably Agena propellant mist, on the entire Agena status display panel is believed to have contributed to this condition. The problem was overcome by sighting the lights through the telescopic portion of the sextant.

Stowage

Allotting space and providing the means to secure stowage items on each spacecraft presented problems on most missions.

Manned missions called for known quantities of food, and rendered known quantities of waste. Cameras, film, camera accessories, and various hard and software were required for experiments. Logbooks and data cards were used for recording events during the missions. "Zero g" conditions necessitated stowing each item. Other factors affecting stowage included the duration of each mission, the size and nature of the item to be stowed, and its immediacy and frequency of use.

Due to the relatively short duration of the Spacecraft 3 mission and the small amount of equipment necessary, stowage was not a problem. Spacecraft 3 did not have all the fixed stowage containers of later spacecraft. Consequently, the right hand aft box and the modified whirlpool center food box contained excess space. But on subsequent spacecraft, except Spacecraft 6, all available stowage space was utilized.

All items had to be secured to prevent them from floating. The method of securing "loose items," i.e., not bolted to the spacecraft or removed from a fixed container, was to apply velcro hook about the spacecraft walls, instrument panels, hatches and seats, and velcro pile to the item to be secured. After each flight, the next crew usually requested that additional velcro hook be applied to the interior of each spacecraft through Spacecraft 8 when nearly all the usable space was occupied.

Beginning with Spacecraft 4, the aft and center stowage area was redesigned to the following configuration:

A. A center stowage rack fitted with one to three Fiberglas containers.
B. Left and right aft food boxes.

On Spacecraft 4 and 5, the center stowage rack was configured with three Fiberglas containers which housed cameras and camera accessories. A mounting adapter for the 200 mm lens was stowed on the center stowage rack door. However, the center stowage containers on Spacecraft 4 did not have adequate space for stowing all of the camera accessories. Based on the probable low use rate, three lenses were stowed in the right hand aft box in pouches. Since the film magazines could withstand more shock and vibration than the accessories, they were stowed in the side food boxes due to the lack of space in the center containers and the need for accessibility.

Commencing on Gemini VI a special centerline rack was installed to carry the extravehicular life support system (ELSS) and a Fiberglas camera case. Although the ELSS was not carried on Spacecraft 6 and 7, this rack was carried to maintain a standardized stowage configuration throughout the program. The ELSS was placed into the rack on a guide rail and structurally supported by two pins on the rear of the rack and two pins in the door of the rack that mated with corresponding holes in the ELSS. On several flights considerable difficulty was encountered in closing the door on the stowage rack. This problem was resolved by removing part of the structure in the rack (making it more flexible) so it could more easily be placed in a mating position with its door. A theory that the structure shifted as the spacecraft was pressurized prompted this action.

The Fiberglas container arrangement on Spacecraft 6 and 7 was changed to two containers, upper and lower, which allowed more stowage space. Except for pen lights on Spacecraft 6, cameras and camera accessories were stowed in these containers. The primary stowage objective was to locate all equipment needed for a given photographic experiment or documentation in one container.

On Spacecraft 7, cameras and camera accessories were placed in the lower container and the extra film magazines were stowed in the right hand pedestal auxiliary food container. The upper container was used partially for items that were usually stowed in the sidewall containers on other spacecraft. These included tape cartridges, suit repair kit, urine collection system components, plastic zipper bags, tissue dispenser, vision testing and dewpoint hygrometer components, interference filters, and medical accessory kit.

The upper part of the center stowage area on Spacecraft 8 through 12 was used for the extravehicular life support system (ELSS) package. The lower portion contained a Fiberglas box with foam molded to accept cameras, film magazines, and camera accessories. On each of these spacecraft there was not sufficient room in this box to contain all the equipment, and overflow items such as lenses and film magazines sometimes were stowed in pouches in the aft and side food boxes.

In addition to center stowage overflow of cameras and accessories, the left and right aft boxes were utilized for stowing food, extravehicular activity (EVA) equipment, urine bags, defecation devices, and other items not requiring frequent use. Each item, with the exception of the food packs, was in an individual pouch. Some items such as cameras with lenses were in padded pouches. Typical stowage is illustrated in Fig. 35.

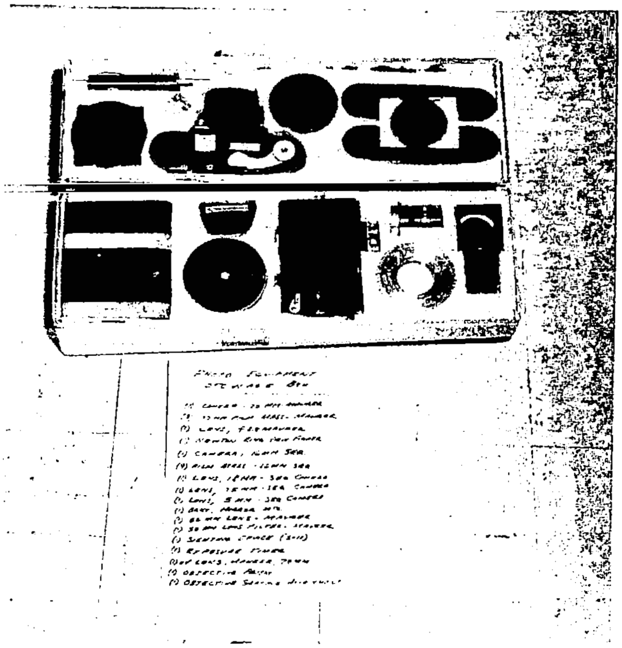
On Spacecraft 4 and 5, the left aft box was used for food, but the right aft box was used to stow "infrequent use" items such as camera lenses, umbilical hose, defecation bags, suit repair kit, and Zodiacal light camera. On Spacecraft 4, the size and shape of the in-flight exerciser, blood pressure adapter, urine receiver and hose, and umbilical guide was responsible for these items being stowed in the right aft box.

The left aft box on Spacecraft 6 was used for food, personal effects items such as tissue dispenser, bio-med fitting wrench, waste containers, roll-on cuff receiver, and defecation devices, and for items of a general nature such as lightweight headsets, humidity sensor, voice tape recorder cartridges, 16 mm film magazines, and a collection bag assembly. The right aft box contained auxiliary water bags for use in obtaining a drink during prelaunch count down.

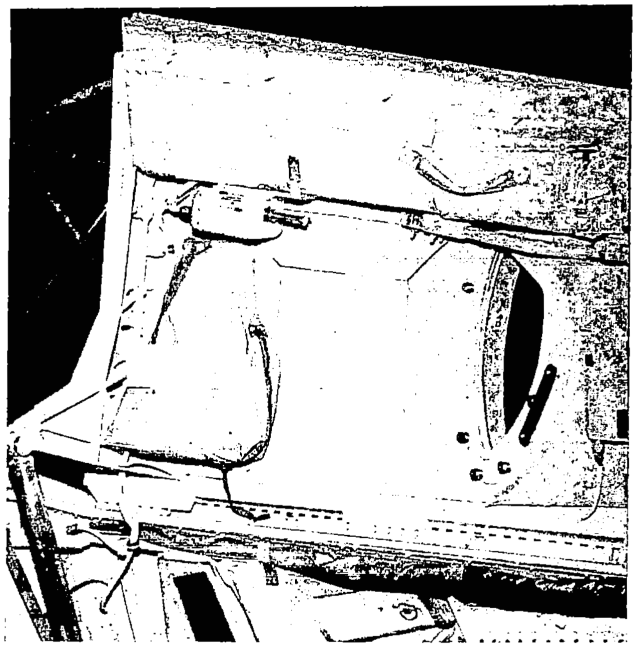
Both of the aft boxes on Spacecraft 7 were used mainly for food. Personal hygiene towels also were stowed in each of the aft boxes.

Beginning with Spacecraft 8, mission requirements became more uniform with respect to duration, EVA, and photography. On Spacecraft 9, 11 and 12, the left aft box was utilized primarily for EVA equipment. On Spacecraft 10, the EVA equipment was divided between the two aft boxes. The right aft box contained items which could not be stowed elsewhere or which had a low level of use. The right aft box on Spacecraft 11 and 12 was used principally for food, blood pressure bulb, waste container, defecation devices, and camera equipment that could not be stowed in the center stowage container due to lack of space.

The left and right sidewall boxes were utilized primarily for items intended to be easily accessible, or of a small volume. Typical of such items were personal hygiene towels, waste containers, pen lights, defecation devices, voice tape recorder cartridges, the spot meter exposure dial, and lightweight headsets. On most spacecraft two man-meals were stowed in each sidewall box to eliminate the need for unstowing of the aft boxes or the hatch food pouches until after the first sleep period. The "small volume" hard items such as hose nozzle interconnectors and 16 mm and 70 mm film

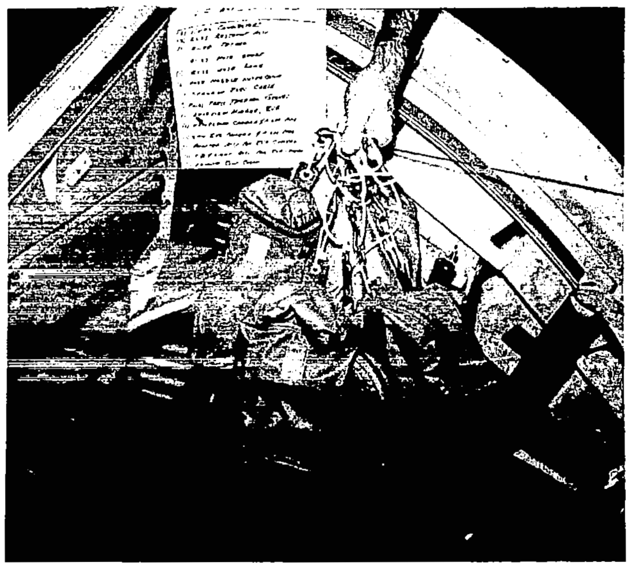


TYPICAL CAMERA STOWAGE



TYPICAL HATCH STOWAGE

FOOTWELL SHOWING TYPICAL STOWAGE POUCHES



TYPICAL AFT BOX STOWAGE POUCHES

FIGURE 35 CABIN STOWAGE

magazines were placed in the sidewall boxes rather than the pedestal, footwell, or circuit breaker panel fairing pouches, due to launch conditions. On some spacecraft the circuit breaker and light module assemblies, plastic zipper bags, utility electrical cords, and velcro hook strips and velcro pile strips were stowed only in this location because of crew preference that they be equally accessible in the pedestal pouches or the utility pouch under the right hand instrument panel.

The sidewall extension boxes were incorporated beginning with Spacecraft 4. They were located directly aft of the sidewall boxes. Accessibility to these boxes was limited due to proximity of the ejection seat. On Spacecraft 4, they were used for stowing a lightweight headset and defecation devices. On Spacecraft 6, the boxes contained one hose nozzle interconnector each and on Spacecraft 7, seven defecation devices each. On Spacecraft 5 through 12, the crew preference kits were stowed in these boxes.

Underneath each sidewall box a long, narrow, auxiliary stowage pouch was affixed to the sidewall beginning with Spacecraft 5. These pouches were used generally to stow the blood pressure bulb and one of the urine hose and filter assemblies. On some spacecraft they contained small items such as hose nozzle interconnectors. Beginning with Spacecraft 8, a pair of debris cutters were stowed in the right hand sidewall pouch.

There were six "hard-mounted" items about the spacecraft. The in-flight medical kit was mounted between the left hand seat and sidewall on Spacecraft 3 through 7 and between the right hand seat and the sidewall on Spacecraft 8 and up. The voice tape recorder was mounted permanently between the right hand seat and the sidewall on Spacecraft 3 through 7. Beginning with Spacecraft 8, the recorder was interchanged with the in-flight medical kit and at the same time the mounting was changed to allow the recorder to be removed from its bracket and be stowed on the left hand circuit breaker panel fairing pouch cover during orbit. It was made portable primarily for EVA purposes. The 16 mm camera bracket that mounts on the outboard corner of the window frame was stowed in a holster mounted on the left sidewall aft and slightly lower than the sidewall box. A similar holster for the right window bracket was mounted on the right sidewall directly opposite the mounting on the left side.

The swizzle stick was located on the right side of the overhead circuit breaker panel switch guard. Either crew member could use this device to actuate switches in the other cockpit. An optical sight was stowed under the left instrument panel on each spacecraft.

A pouch approximately 4 x 6 x 1.5 in. deep was located directly below the left and right circuit breaker panels. It was constructed so as to fair in with the panels. These pouches were used primarily for small, lightweight items such as the urine collection device (UCD) clamps velcro strips, small rolls of tape, and a urine system component.

Each spacecraft had dry stowage bags mounted on the footwell sidewalls with velcro. These bags were used for stowing charts, sun shades, reflective shades, and flight data books. On Spacecraft 3 through 9, a plotboard pouch was installed on the inboard sidewall of the left footwell. This pouch contained charts and flight data books in addition to the plotboard. On Spacecraft 4, the ventilation control module (VCM) was stowed in the right footwell. The VCM was the forerunner of the ELSS.

On Spacecraft 7, an auxiliary food pouch was stowed in the right footwell. On Spacecraft 8 and 11, a TV monitor was mounted on the inboard sidewall of the right footwell for launch and re-entry. For the rest of the flight the TV monitor was stowed in a specially cut mounting on top of the inboard side of the left ejection seat. On Spacecraft 10 and 11, a 50 ft umbilical was stowed in the forward portion of the left footwell. The debris guard under the left instrument panel was modified to make more room for the umbilical. This was done to provide more clearance from the ejection envelope.

An orbital utility pouch was installed under the right instrument panel on Spacecraft 4 through 12. On Spacecraft 4 through 6, this pouch was small and held only a few items. On Spacecraft 4, it was primarily for the hatch closing device. On Spacecraft 7, it was enlarged to provide additional stowage space. Spacecraft 8 through 12 retained the Spacecraft 7 configuration.

Due to the requirement for additional stowage space for experiment and photographic equipment on Spacecraft 9 through 12, food pouches were installed on each hatch. Similar pouches were installed on Spacecraft 7 due to the increased need for food on the 14-day mission. Initially the pouches were not rigid enough on Spacecraft 9 to hold the food in place before the hatch was closed. Before the launch of Spacecraft 9, elastic straps were sewed onto the pouches to provide the required rigidity. As a precaution the crew planned to eat at least two meals each from the right hatch pouch to avoid any difficulty in closing the hatch after the first EVA exercise.

Incorporated in the water management panel was a water drink gun on the early spacecraft. On Spacecraft 8 and up, a water metering device was used which would measure in half-ounce increments and provisions were made for stowing an additional urine hose and filter assembly.

The left and right hatch torque boxes were utilized to install various experiments in hard-shell containers. The design requirements for these containers included adequate clearance with the ejection seat, noninterference with the crew's helmets, and ability to withstand hatch opening shock loads of up to 40 g's. The list of experiments stowed on the hatch torque boxes is shown below:

S/C	LEFT HATCH TORQUE BOX	RIGHT HATCH TORQUE BOX
3	Sea Urchin Egg Unit	Blood Chromosome Unit
4	Dose Rate Indicator - Type 5 (Removable)	Dose Rate Indicator - Type 1 (Fixed)

S/C	LEFT HATCH TORQUE BOX	RIGHT HATCH TORQUE BOX
5	Vision Tester	Photometer
6	Dose Rate Indicator - Type 1 Dosimeter, Passive	Dose Rate Indicator
7	Vision Tester	Photometer
8	Frog Egg Container	Frog Egg Container
9	50 mm Lens and Filter (S-11)	Sextant
10	Star Occultation Photometer	Sextant
11	Blood Irradiation Device	Sextant
12	Spectrometer Sensor	Frog Eggs Experiment

Personal Equipment And Survival Equipment

All of the personal and survival equipment was government furnished. Personal equipment largely consisted of the space suit and associated equipment. Fig. 36 illustrates the Gemini pressure suit. The following suits were flown:

SPACECRAFT	SUIT
3	G3C
4	G4C
5	G3C
6	G3C
7	G5C
8 and up	G4C

Personal equipment consisted of:

QTY.	PART NUMBER	NOMENCLATURE
2 sets	ACS-1505	Wrist Dam
2	CF55019-16	Flight Booklet
2	CF55033	Chronograph Wrist Watch
2	CF55044-1	Leg Strap
2 sets	CF55047-5	Flight Data Cards
4	CF55050-1	Marking Pen
4	CF55051-2	Mechanical Pencil
2	CF55052-5	Watch Band
2 pr	CF55081-1	Sunglasses
2	CSD20537-1L	Life Vest Assembly
2	CSD20537-1R	Life Vest Assembly

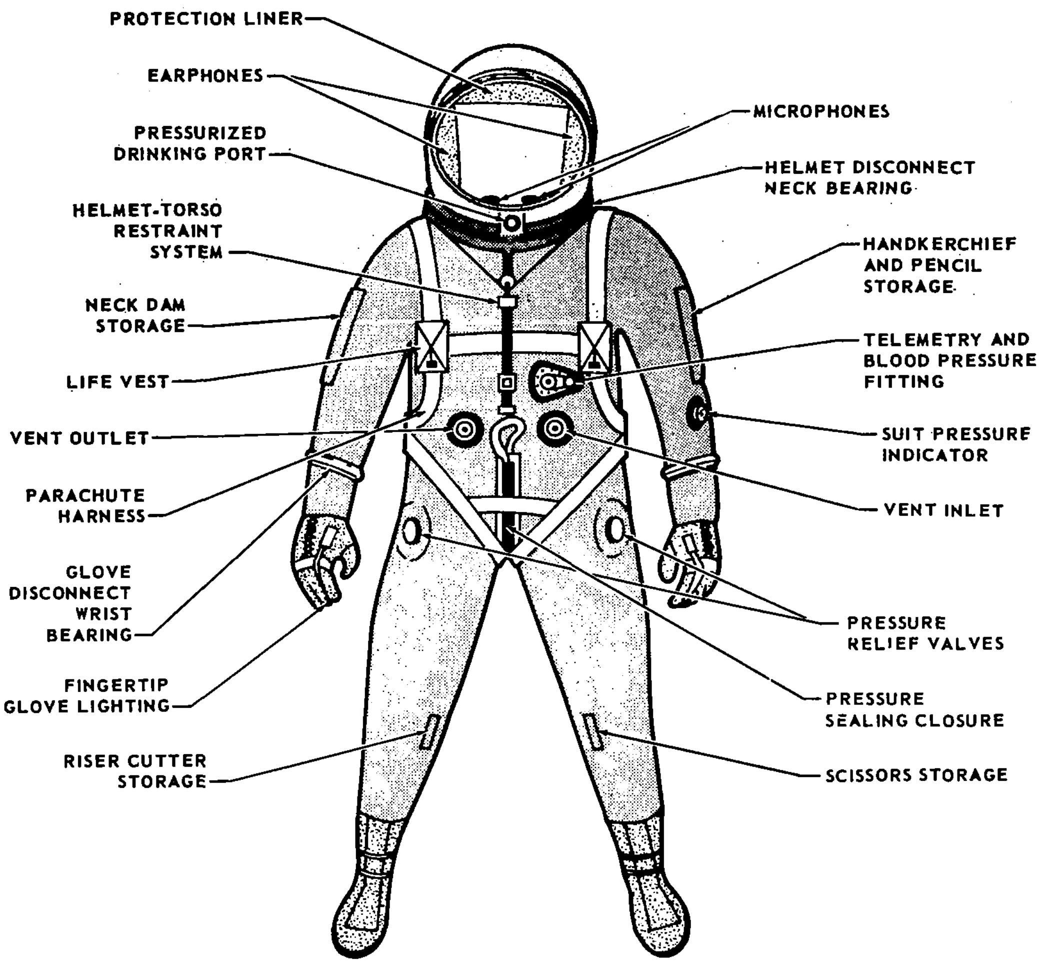


FIGURE 36 GEMINI PRESSURE SUIT

QTY.	PART NUMBER	NOMENCLATURE
2	CSD20542	Surgical Scissors
1	EA35998-1	Radiation Pocket Dosimeter
8	EA35999-1	Passive Radiation Packet
2	EC30045	Space Suit Knife
1	EC30047-1	Visor Cover
2	G4C	Space Suit Assembly
4	MSC-ECG-SIG-GF-S1	ECG Signal Conditioner
2	MSC-HAR-SYS-GF-G2-1	Bio-Med and Communication Harness
2	MSC-TEM-PRB-GF-S1	Oral Temperature Probe
2	MSC-TEM-SIG-GF-S1	Oral Temperature Signal Conditioner
2	MSC-ZPN-SIG-GF-S1	Impedance Pneumograph Signal
2	9557-3-112-3	Launch Day Urine Bag
2	IRI-GBE-A	Auxiliary, ELG Electrode Harness
2	IRI-GBE-S	Sternal, ECG Electrode Harness

The following equipment is in the survival kit:

QTY.	PART NUMBER	NOMENCLATURE
2	CSD20539	Survival Pack (Less Container)
2	CSD20503A	Life Raft
2	CSD20548	Sea Anchor
2	CSD20543	Sunbonnet
2	CSD20505	CO_2 Cyliner and Valve
2	CSD20536-1	H_2O Container
2	CSD20538	Combination Lite Assembly
2	CSD20511	Desalter Kit
2	CSD20534	Medical Kit
2	CSD20526	Machete and Sheath
2	CSD20513	Sunglasses
4	CSD20504	Sea Dye Marker
2	2588218	Radio-Beacon

The survival kit is pictured on Fig. 37 and 38.

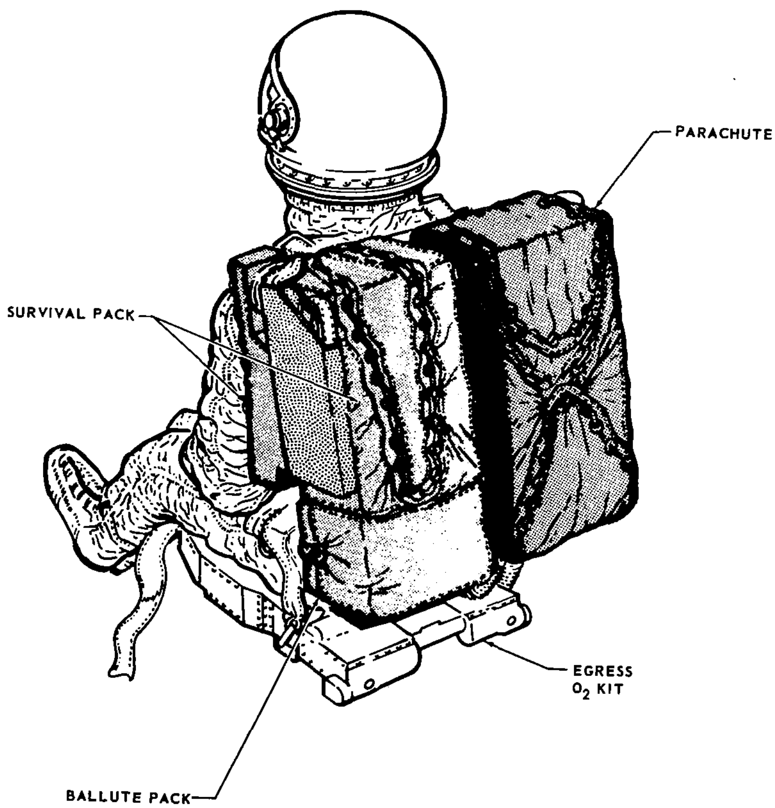


FIGURE 37 BACKBOARD/EGRESS KIT ARRANGEMENT

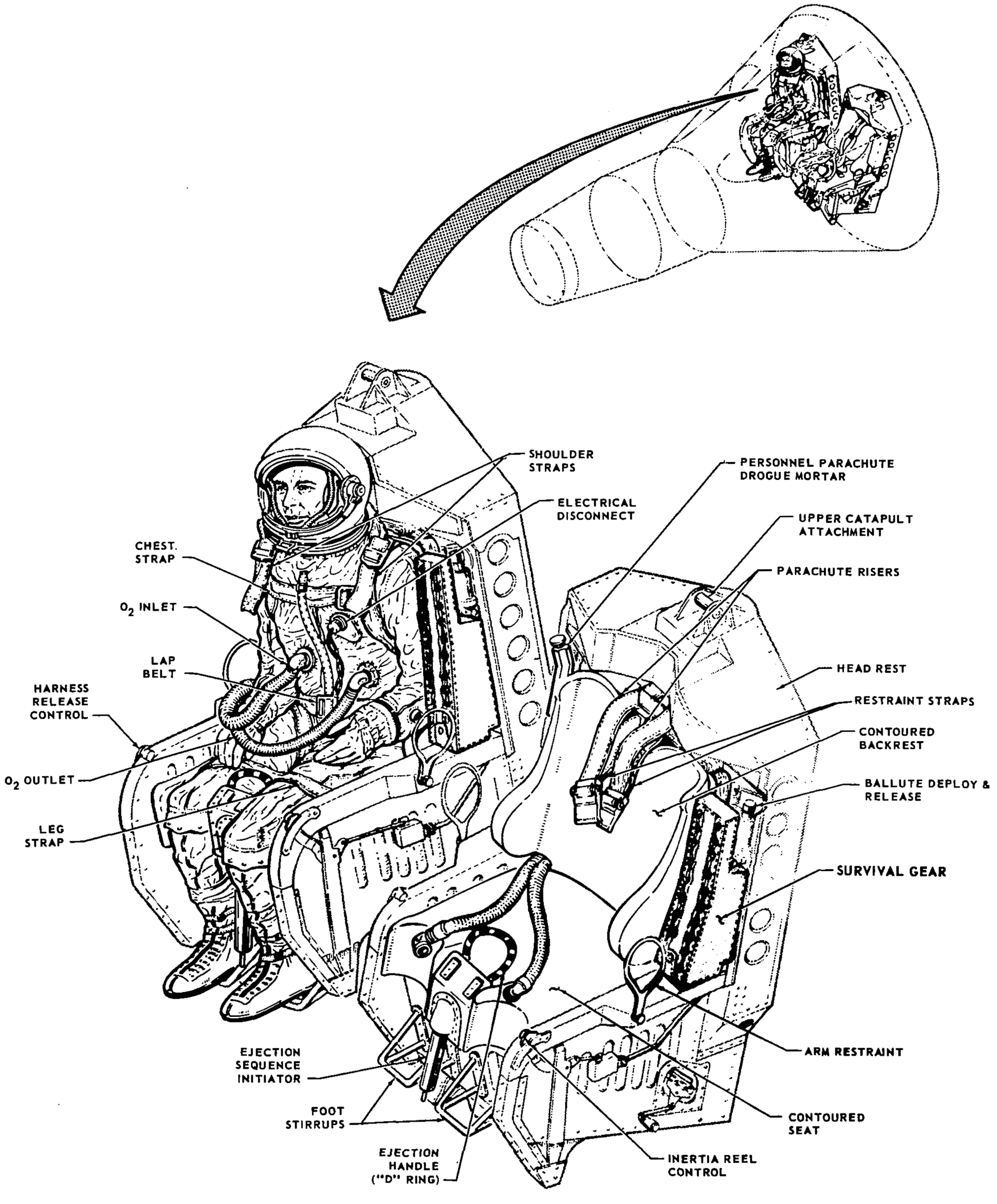


FIGURE 38 CREW SEATING AND RESTRAINT DETAILS

ESCAPE, LANDING AND RECOVERY SYSTEMS

Escape System

The ejection seat system, designed as the primary escape system for crew safety, consists of ejection seats, seat rocket/catapult, hatch actuating system, personnel parachute system, and survival equipment. Maximum design operating parameters are at altitudes up to 70,000 ft, speeds up to Mach 2.86 and dynamic pressures up to 820 psf. The ejection system also has the capability of being utilized at zero altitude - zero velocity thus, the system will allow the astronauts to escape from the spacecraft while on the launch pad, during launch or after re-entry. However, the operational use of the seat was restricted to altitudes below 15,000 ft by decree, although the design and qualification tests were as stated. The reasons for this restriction are as follows:

A. Titan II booster flights N-7 and N-20, where breakup occurred disclosed no hazards that would have been significant to the spacecraft during a "ride it out" abort. This, coupled with additional analysis indicated a high degree of confidence in crew safety could be obtained by remaining in the spacecraft during a booster malfunction and then using a Mode II (retro-rocket salvo) abort to escape from the booster.

B. Although high speed ejection seat sled tests were satisfactory, the random nature of possible ejection conditions such as attitudes, oxygen hose flapping, astronaut sizes and pressure suit modifications did not instill confidence in high altitude or high dynamic pressure ejections.

C. Water recovery after an ejection posed as additional safety hazard which would be compounded if an injury were sustained during the ejection.

D. Use of the ejection seat made an abort decision more time critical since a late ejection could result in recontact with the booster or fireball. Thus, ejection had to occur before significant booster attitude excursions occurred. This was nominally about 5 degrees while booster breakup angles were much greater and therefore the decision to eject was time critical based on ejection seat rather than booster restrictions.

Optimum trajectory and stability of the seat during its powered flight is maintained through position control of the seat-man combination center-of-gravity. After the powered phase of the flight the man is separated from the seat and descends under a stabilizing device called a ballute, until a desirable parachute opening altitude is reached. After parachute opening, the backboard assembly and egress kit are jettisoned, leaving the man to descend under his parachute with his survival gear secured to a lanyard below him.

The ejection seat system as it is currently designed has a .995 reliability factor of successful operation.

The basic location and arrangement/configuration of the ejection system are shown in Fig. 39. The crew members are seated side by side, faced toward the small end of the re-entry module. Each seat is canted 12 degrees outboard

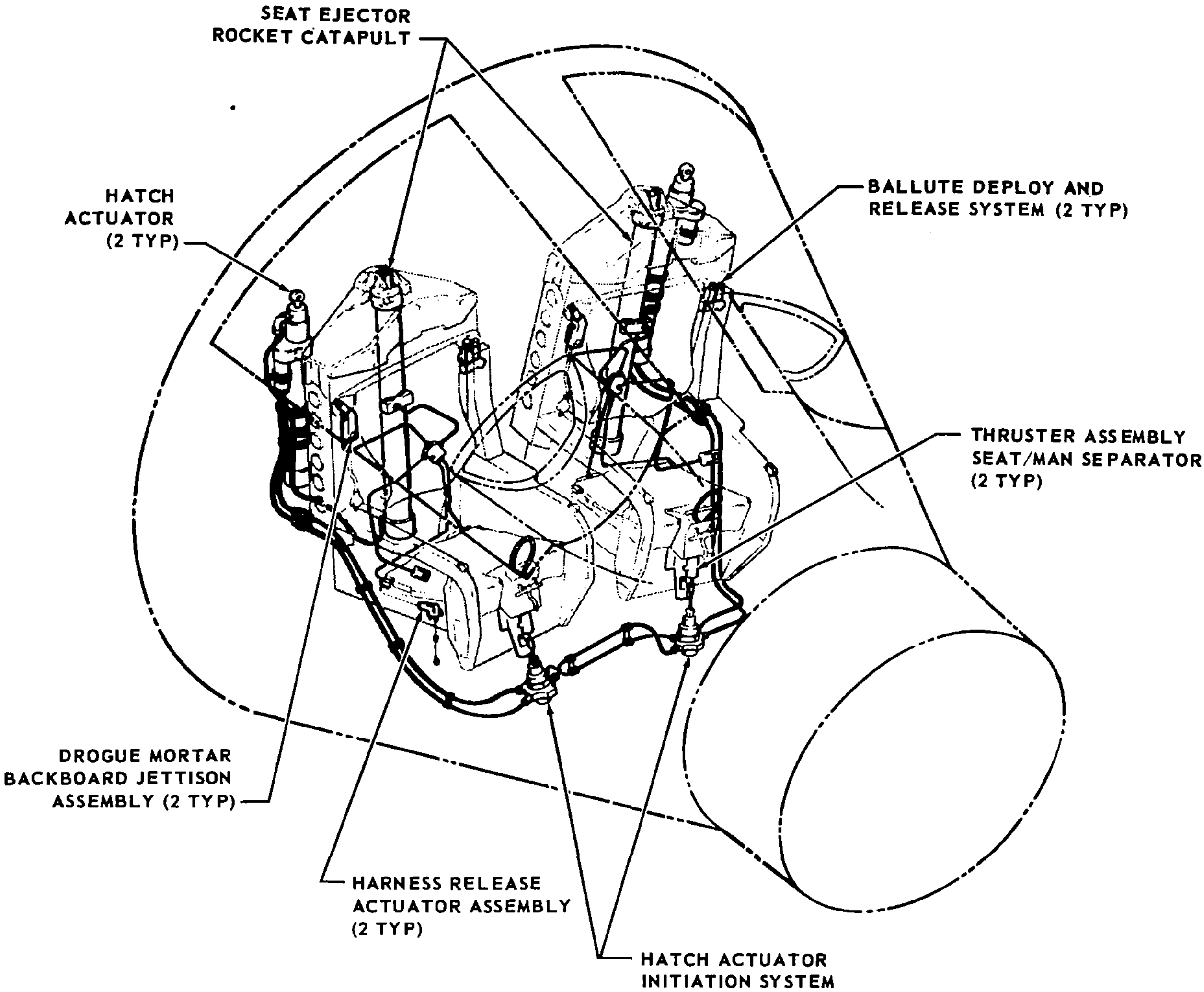


FIGURE 39 EJECTION SEAT SYSTEM AND DEVICES

and 8 degrees 20 min forward to assure separation and provide required elevation in the event ejection off the pad is necessitated.

Ejection Seat. - The ejection seat as shown in Fig. 40 is basically a sheet metal structure built on a torque box frame principle. The seat is mounted in the spacecraft on T-type rails attached to the large pressure bulkhead and held in position by a single bolt at the top of the seat back where it attaches to the rocket catapult. The seat structure includes a seat back, seat pan, side panel, arm rests, padded headrest, stirrups, stowable arm restraints and seat slider blocks. Attached to the seat structure is the equipment required to separate the man from his seat. This includes the ejection control mechanism, seat-man separator thruster, seat-man

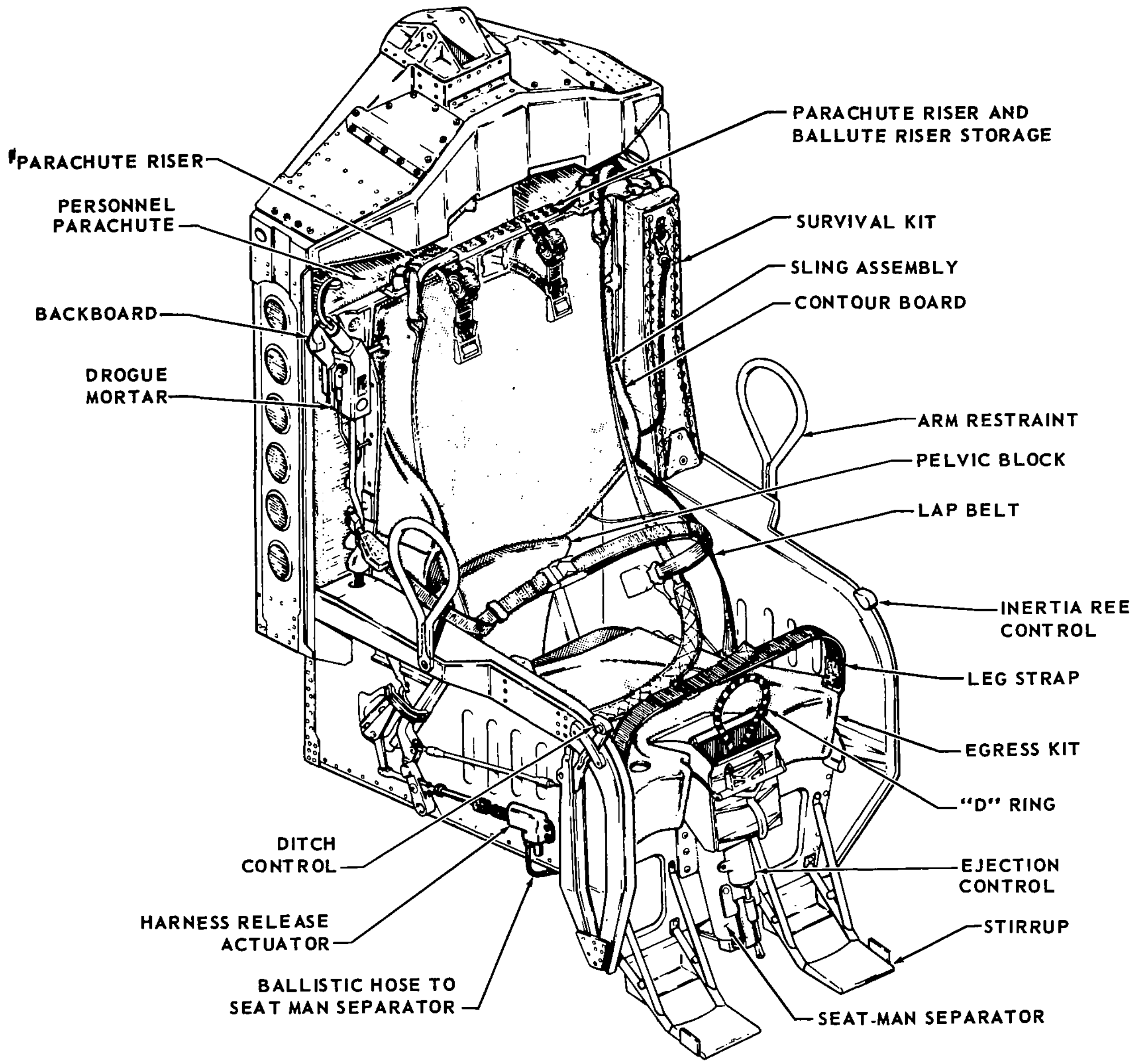


FIGURE 40 EJECTION SEAT ASSEMBLY

separator strap with personalized contoured pelvic block attached, harness release actuator, lap belt releases, and upper and lower backboard releases and their associated linkage. Attached to the left seat side panel is the inertia reel control handle, which allows the man to select either a manual lock or automatic lock position. The manual position prevents the astronaut's shoulders from moving forward by mechanically preventing extension of the shoulder straps. The automatic lock position allows the astronaut, to move forward slowly but will lock with a fast movement of 2 g's. If locking occurs in the automatic position, the astronaut releases himself by cycling the control handle to manual and back to automatic lock. On the right hand seat side panel, a ditching handle releases the man with his backboard assembly and the egress kit from the seat structure. This provision permits an over-the-side bailout.

Backboard Assembly. - The backboard assembly mounts on the ejection seat structure through the upper and lower backboard releases. Mounted to the backboard assembly are survival items such as the survival kits, personnel parachute system, and the ballute system (a high altitude, high velocity stabilization device).

Mounted to the backboard is an inertia reel with restraint straps that pass over the top of the backboard and are attached to the parachute risers at the parachute quick disconnect fittings. A personalized cushion contoured for safety and comfort, is mounted to the front of the backboard. Also mounted on the backboard are the pyrotechnic devices that initiate system operation. These devices include the drogue mortar and its associated equipment jettison system and the ballute deploy and release system.

Egress Kit. - The egress kit assembly rests on the seat structure and is restrained to the seat structure and backboard assembly by the lap belt system. The egress kit contains the emergency bailout oxygen system. For Spacecraft 8 and up, the egress kit height is lowered by removing the bailout oxygen system to allow more room in the seats. Removal of the bailout oxygen system lowers the rated ejection altitude to 15,000 ft. For pilot comfort, a contoured cushion is mounted on top of the egress kit. Located at the front of the egress kit is a single point release handle which, when activated, releases the egress kit from the seat structure and backboard assembly. This manner is convenient for a low altitude over-the-side bailout. The ejection control handle (D ring) is normally stowed under a sliding door at the front of the egress kit to prevent inadvertent ejection. However, during launch and re-entry, it is in operational position. At the rear of the egress kit is a pytotechnic-activated quick disconnect system. The supply and return oxygen hoses from the suit as well as the communications wire bundle are attached to this disconnect. Because the egress kit was lowered for Spacecraft 8 and up (see below), the single point release handle was removed and the D ring was stored on the seat-man separator thruster.

After the flight of Spacecraft 3, the astronauts reported that the egress kit knee supports provided were uncomfortable and actually hampered their leg movements when they tried to shift their position. In addition, it was felt that the excessive height of these supports might render egress during EVA more difficult. As a result, the thickness of the egress kit cushion was trimmed (deleting the knee supports) for Spacecraft 4 through 7; this served to lower the astronauts in their seats and provided a more comfortable attitude. However, the old (52-82911) structure was still employed and the single-point release system was still incorporated.

For Spacecraft 8 and up, the trimming was carried further, resulting in a thinner egress cushion in the forward position. A new egress kit structure was employed, and a new ejection control mechanism was incorporated. The single-point release system was eliminated. In addition, the ejection control handle storage container which went between the legs was deleted.

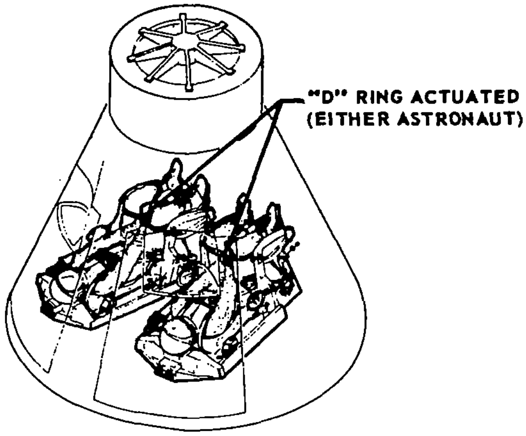
Lap Belt Assembly. - The lap belt assembly is a webbing arrangement designed to hold the egress kit into the seat and the astronaut into the ejection seat system. After ejection and subsequent seat-man separation it secures the bottom of the backboard and the egress kit to the astronaut. The lap belt assembly includes a manual disconnect and a pyrotechnic disconnect. The lap belt disconnect was modified for Spacecraft 11 and 12 by adding belt guides to the disconnect. The guides prevent belt jamming in the disconnect during adjustment by the astronaut.

Hatch Actuating System. - The hatch actuating system, when initiated by either pilot, activates the firing mechanisms of both hatch actuators. This system consists of two manual firing mechanisms cross-connected by two rigid mild detonating fuse (MDF) crossovers and two pyrotechnically actuated hatch actuators redundantly interconnected by four rigid and four flexible MDF lines.

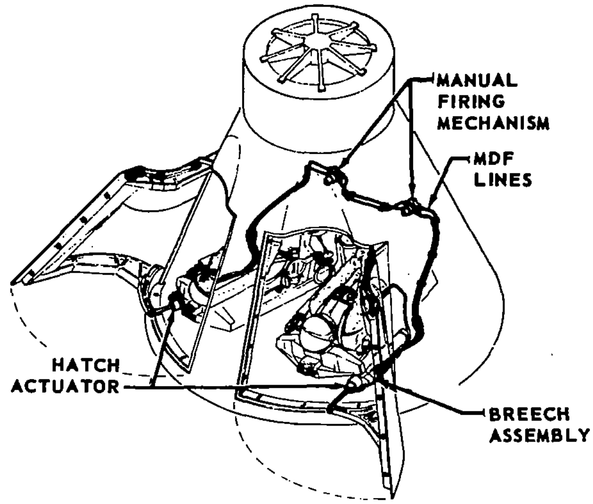
An MDF interconnect is a single strand of high explosive (2 gr/ft of RDX) encased in a lead sheathing and installed in flexible or rigid steel tubing. A small booster charge is incorporated at each end of the explosive strand. Attachment fittings are incorporated on each end of the tubing.

The pyrotechnic hatch actuator unlocks, opens, and mechanically retains the egress hatch in the open position. This assembly also furnishes gas pressure to initiate the firing mechanisms of the seat ejector-rocket catapult after opening the hatch. A lanyard, attached to the locking mechanism of the actuator, permits the hatch to be unlocked manually.

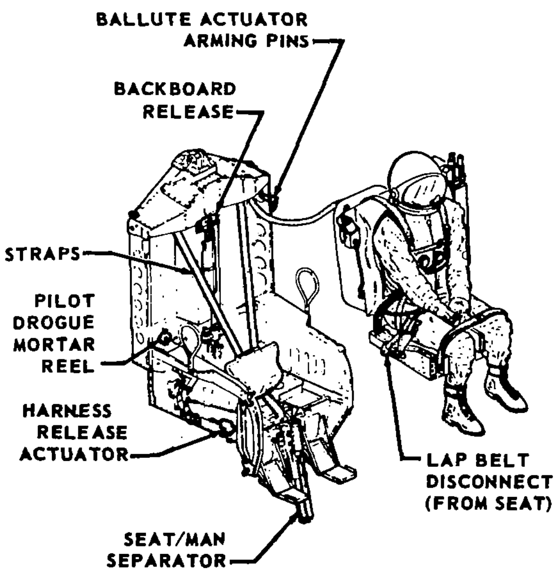
Escape System Operation (Fig. 41). - The ejection sequence is initiated when either crew member pulls the D ring. The D ring then activates a detonator in the manual firing mechanism which induces propagation in the MDF interconnects. These MDF lines redundantly provide the initiation pulse to the dual cartridges in each hatch actuator breech. The large volume of high pressure gas developed in the breech is ported into the actuator. The gas pressure extends the latch piston, which unlocks the egress hatch through a bellcrank/push rod mechanism. The gas pressure then acts on the piston/stretcher assembly, moving it through the length of the cylinder. At approximately 0.6 in. prior to the stretcher assembly reaching full extension, gas pressure is exhausted into a ballistic hose. The ballistic hose then delivers the gas pressure to the firing mechanism of the seat ejector-rocket catapult. As the stretcher assembly reaches the fully extended position, the lock pin of the locking mechanism engages the piston of the stretcher assembly, and holds the hatch open. The catapult firing mechanism ignites a relay and main charge in the catapult. The resulting gas pressure propels the rocket motor through the length of the catapult housing. After approximately 31 in. of stroke of the catapult, the gas initiates dual igniter charges in the sustainer motor which provides additional thrust through the cg of the ejection seat. The ejection seat moves up the rails actuating four lanyards



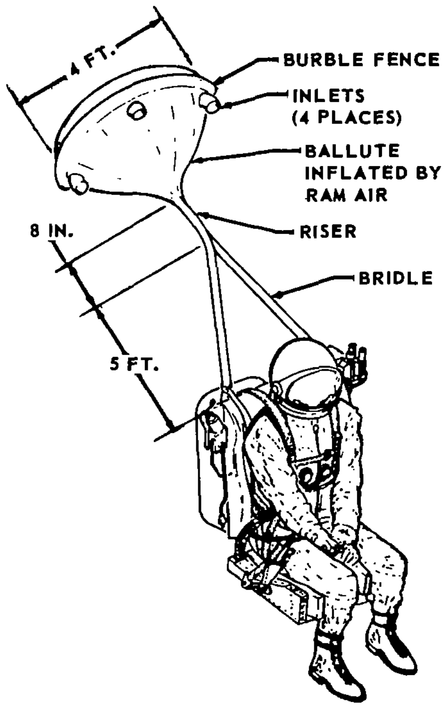
SYSTEM ACTIVATION – 0.0 SECONDS
SEQUENCE –1–



HATCHES OPEN – 0.250 SECONDS
SEQUENCE –2–

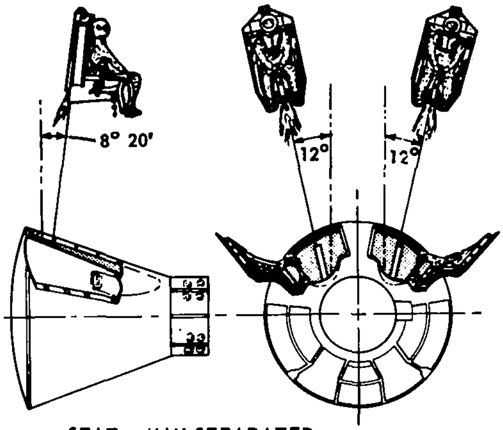


HARNESS RELEASE FIRES – 1.474 SEC.
SEAT/MAN THRUSTER FIRES – 1.524 SEC.
SEAT/MAN SEPARATED –
LANYARDS PULLED –

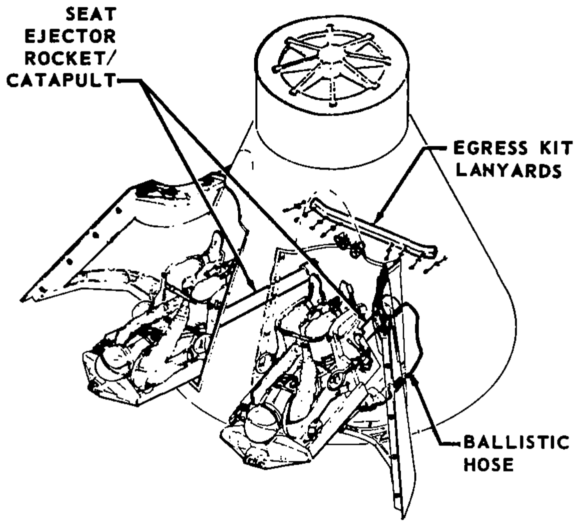


BALLUTE IS DEPLOYED 5.00 SEC.
AFTER SEPARATION
BALLUTE IS RELEASED AT 7500 FT.

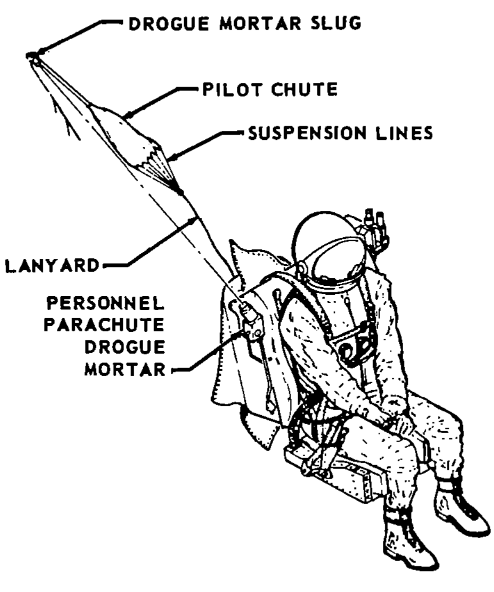
FIGURE 41 EJECTION SEAT SEQUENCE OF OPERATION



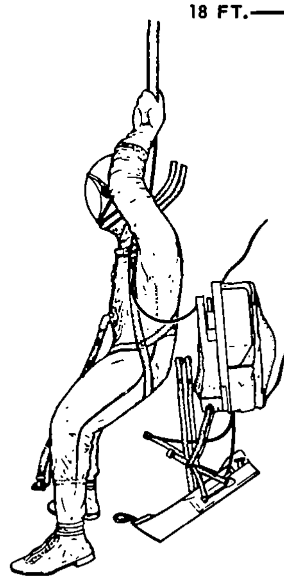
SEAT - MAN SEPARATED FROM SPACECRAFT



ROCKET CATAPULT FIRES - 0.333 SEC.
LANYARDS PULLED - 0.394 SEC.
SEQUENCE -3-



DROGUE MORTAR IS ACTUATED AT ALTITUDE OF 5700 FT. OR LESS. 2.30 SEC. TIME DELAY AFTER ACTIVATION FOR FIRING.



BACKBOARD & EGRESS KIT JETTISON 3.00 SEC. AFTER DROGUE MORTAR FIRES.

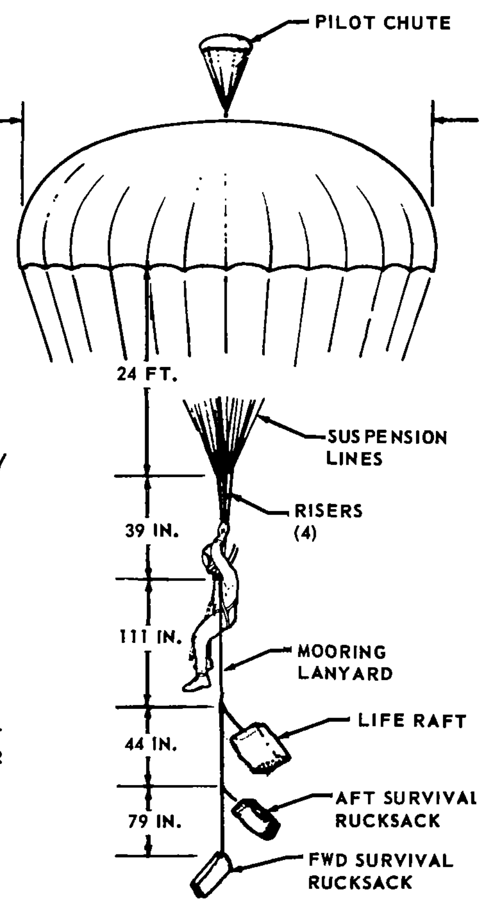


FIGURE 41 EJECTION SEAT SEQUENCE OF OPERATION (Continued)

attached to the spacecraft structure. Three of the lanyards also are attached to the egress kit, and accomplish the following functions:

A. Remove lock pin from composite disconnect allowing separation of spacecraft oxygen hoses and wire bundle from the seat.

B. Trip pressure regulator valve, initiating oxygen flow from egress kit to astronaut.

C. Initiate control check valve to provide pressure in suit.

Effective Spacecraft 8 and up, the lanyards to the oxygen regulator and the control check valve are removed as the bailout oxygen is removed.

The fourth lanyard is attached to the firing mechanism of the harness release actuator, which fires a percussion cartridge to release the backboard and egress kit from the seat structure after a 1.08 sec pyro time delay. The gas pressure of the harness release actuator cartridge is then vented through a flexible hose to initiate the seat-man separator thruster cartridge. The seat-man separator thruster pushes against a Y strap, fixed at all three points, and forcibly separates the astronaut with his backboard and egress kit from the ejection seat. During seat-man separation, lanyards for the ballute deploy and release mechanism initiate a time delay deploy cartridge and arm a barostat-controlled firing mechanism for the release function. Below 7500 ft the barostat fires the ballute release cartridge. For off-the-pad ejection, the release function occurs prior to the ballute deploy, thereby freeing the ballute if the deployment function should inadvertently occur. Also, at seat-man separation, a lanyard attached to the seat structure fires the parachute drogue mortar, or if separation occurs above 5700 ft, unlocks the firing pin to be released by a barostat when the 5700 ft altitude is reached. The drogue mortar deploys the personnel parachute and also supplies pressure to initiate a time delay cartridge propagating MDF which accomplishes the following:

A. Disconnects the suit hoses and wiring (Jetelox) from the egress kit.

B. Releases the lap belt which allows the egress kit to fail away.

C. Cuts the inertia reel straps which releases the backboard to fall away.

As the backboard and egress kit fall away, a mooring lanyard attached to the astronaut becomes taut and extracts the life raft and rucksacks (containing survival items) from their packs on the backboard. The astronaut descends with the survival kit items attached to him.

Escape System Development and Qualifications Tests. - Ejection seat system development tests and qualification tests are shown on Tables 19, 20 and 21, respectively. Firings of ejection seat of pyrotechnic components and devices are tabulated in Table 22. Before each flight, verification firings were made of units from the same lots as those installed in the spacecraft.

Failure Summary. - The failure summary is shown in Table 23.

TABLE 19 EJECTION SEAT DEVELOPMENT TEST

TEST COMPLETED	TYPE OF TEST	REMARKS
	10% SEAT-MAN COMBINATION 20% SEAT-MAN COMBINATION	POLYSONIC WIND TUNNEL TESTS TO OBTAIN THE AERODYNAMIC CHARACTERISTICS USING SCALE MODELS
4	SIMULATED EJECTION (SLED TEST)	SEAT EJECTION SYSTEM FROM BOILERPLATE UNDER LAUNCH AND RE-ENTRY AIR LOAD CONDITIONS
11	SIMULATED OFF-THE-PAD EJECTION (SOPE)	SEAT EJECTION SYSTEM FROM TOWER FOR EJECTION FROM LAUNCHING PAD
14	LOW ALT. DUMMY DROP	PERSONNEL PARACHUTE
3	ROCKET/CATAPULT (RPI)	SEAT-MAN TRAJECTORY
2	THROUGH CANOPY (NAA)	SEAT EJECTION THRU THE CANOPY OF NAA PARAGLIDER CAPSULE
3	ROCKET/CATAPULT (AEDC)	ALTITUDE FIRING CHARACTERISTICS AT 70K
1	ROCKET/CATAPULT – SEAT	VERIFY POCKET BUMPER FITTING OF EJECTION SEAT
1	SEAT DROP TEST	TO PROVE OUT 3-POINT RELEASE SYSTEM
2	ROCKET/CATAPULT	VERIFY EJECTION SEAT FLAME BUCKET
5	SEAT – EGRESS KIT SEPARATION	DYNAMIC SEPARATIONS OK EGRESS O_2 DISCONNECTS AND VALVE INITIATION
5	ROCKET CATAPULT	THRUST ANGLE VERIFICATION
2	LOW ALT DUMMY DROP	VERIFY PERSONNEL PARACHUTE
14	SEAT – MAN SEPARATION	VERIFY HARDWARE BEFORE SLED AND SOPE TESTS

PYROTECHNIC COMPONENTS AND DEVICES – NUMEROUS DEVELOPMENT TESTS WERE CONDUCTED. THESE TESTS WERE AT THE COMPONENT LEVEL, SYSTEM LEVEL AND BREADBOARD TYPE TESTS TO OBTAIN THE CORRECT OPERATING TIME, PRESSURE AND MECHANICAL FUNCTION OF EACH DEVICE.

Mission Anomalies. - No mission anomalies occurred with the escape system.

Mission Evaluation. - Pyrotechnic components of Spacecraft 2 were removed and post-flight fired. Pyrotechnic components of Spacecraft 10 (except hatch actuator and rocket catapult) were post-flight fired in the seat. No anomalies were encountered.

In-flight safety pins for the drogue mortar were repositioned for easier insertion.

TABLE 20 WIND TUNNEL TESTS

BALLUTE		
FULL SCALE SERIES I	MCDONNELL LOW SPEED WIND TUNNEL	16 RUNS
FULL SCALE SERIES I	AEDC 16' x 16' TUNNEL	11 RUNS
FULL SCALE SERIES II	AEDC 16' x 16' TUNNEL	8 RUNS
FULL SCALE SERIES III	AEDC 16' x 16' TUNNEL	8 RUNS
DROGUE		
5% MODEL	LANGLEY SPIN TUNNEL	48 RUNS
BAROSWITCH		
FULL SCALE	MCDONNELL LOW SPEED WIND TUNNEL	7 RUNS
EJECTION SEAT		
20% MODEL SERIES I	MCDONNELL POLYSONIC WIND TUNNEL	115 RUNS
20% MODEL SERIES II	MCDONNELL POLYSONIC WIND TUNNEL	51 RUNS
20% MODEL SERIES III	MCDONNELL POLYSONIC WIND TUNNEL	65 RUNS
10% MODEL SERIES I	MCDONNELL POLYSONIC WIND TUNNEL	149 RUNS

Landing And Recovery System

A three-parachute system is series stabilizes and retards the re-entry vehicle velocity. During the final stage of descent, the main parachute suspension is shifted from a single-point to a two-point system in order to achieve a more favorable attitude for a water landing. The landing system consists of three parachutes (drogue, pilot, and main parachute), two motars, reefing cutters, pyrotechnic cutters and separation assembly, disconnects, risers, and attaching hardware. The system is located primarily in the rendezvous and recovery (R & R) module as shown on Fig. 42. Only Spacecraft 2 utilized a two-parachute system (pilot and main). The recovery system consists of a hoist loop, flashing recovery light, pyrotechnic cutter, dye marker, and flotation material.

Parachutes. - A high altitude drogue parachute stabilizes the re-entry vehicle. The drogue parachute assembly consists of an 8.3 ft D_O, 20 degrees conical ribbon chute with twelve 750-lb tensile strength suspension lines. A pyrotechnic-actuated mortar deploys the chute. Two 16-sec pyrotechnic time delay reefing line cutters disreef the chute after deployment. A three-legged riser assembly attaches the parachute assembly to the R & R module. Three electrically initiated pyrotechnic cable cutter guillotines, each utilizing dual cartridges for redundancy, sever the chute risers from the R & R module. Attached to one of the drogue riser legs is an apex line which extracts the pilot parachute from its own mortar. A pilot parachute, used in tandem with the drogue chute, decelerates the re-entry vehicle, removes the R & R module, and deploys the main parachute. The pilot parachute assembly consists of an 18.3 ft D_O ringsail chute with sixteen 500-lb tensile strength suspension

TABLE 21 EJECTION SEAT QUALIFICATION TEST

TEST COMPLETED	TYPE OF TEST	REMARKS
3	SIMULATED EJECTION (SLED TEST)	SEAT EJECTION SYSTEM FROM BOILERPLATE TO QUALIFY IT UNDER LAUNCH AND RE-ENTRY CONDITIONS
3	SIMULATED OFF-THE-PAD EJECTION (SOPE)	SEAT EJECTION SYSTEM FROM TOWER TO QUALIFY IT FOR EJECTION FROM THE LAUNCHING PAD
1 1 1	STATIC EJECTION 15k EJECTION 40k EJECTION	SEAT EJECTION FROM F-106 AIRCRAFT TO DEMONSTRATE EJECTION CAPABILITY AT ZERO VELOCITY AND ZERO ALTITUDE AND MACH 1.72 AND 40k ALTITUDE
20 (4 LOW ALT, 16 HIGH ALT)	DUMMY DROP	TO QUALIFY ALL ESCAPE SYSTEM COMPONENTS ATTACHED TO THE ASTRONAUT AFTER SEAT SEPARATION
18 (12 LOW ALT, 6 HIGH ALT)	LIVE DROP	
12	STATIC TEST	TO QUALIFY EJECTION SEAT FOR STRUCTURAL INTEGRITY
6	ENVIRONMENTAL	TO QUALIFY SEAT ASSEMBLY FOR HIGH AND LOW TEMP, TEMP/ALT, HUMIDITY, O_2 ATMOSPHERE AND PRESSURE
15	ENVIRONMENTAL	TO QUALIFY PERSONNEL PARACHUTE AFTER EXPOSURE TO VARIOUS ENVIRONMENTAL CONDITIONS
10	ENVIRONMENTAL	TO QUALIFY BALLUTE AFTER EXPOSURE TO VARIOUS ENVIRONMENTAL CONDITIONS
9	ENVIRONMENTAL	TO QUALIFY SURVIVAL KITS AFTER EXPOSURE TO VARIOUS ENVIRONMENTAL CONDITIONS
1	SEAT PYRO FIRING	TO DEMONSTRATE FUNCTIONING OF PYRO SECOND SOURCE VENDOR
6	DUMMY DROP FROM TOWER	TO QUALIFY PERSONNEL HARNESS INTEGRITY

lines. A two-legged riser assembly attaches the parachute to the R & R module. Two pyrotechnic reefing cutters disreef the pilot chute six sec after deployment. The pilot chute can be pyrotechnically deployed by its mortar in the event the drogue parachute does not deploy or deploys improperly. An electrically initiated pyrotechnic apex line guillotine, with dual cartridges for redundancy, is provided to sever the apex line to free the pilot chute from the drogue chute in the event of a drogue chute malfunction. The design of the guillotine allows the apex line to pull free when the pilot chute is deployed by the drogue chute.

The main parachute assembly consists of an 84.2 ft D_O ringsail chute with seventy-two 550-lb tensile strength suspension lines. Three pyrotechnic reefing cutters disreef the main chute 10 sec after deployment. The main

TABLE 22 ESCAPE SYSTEM PYROTECHNIC COMPONENT FIRINGS

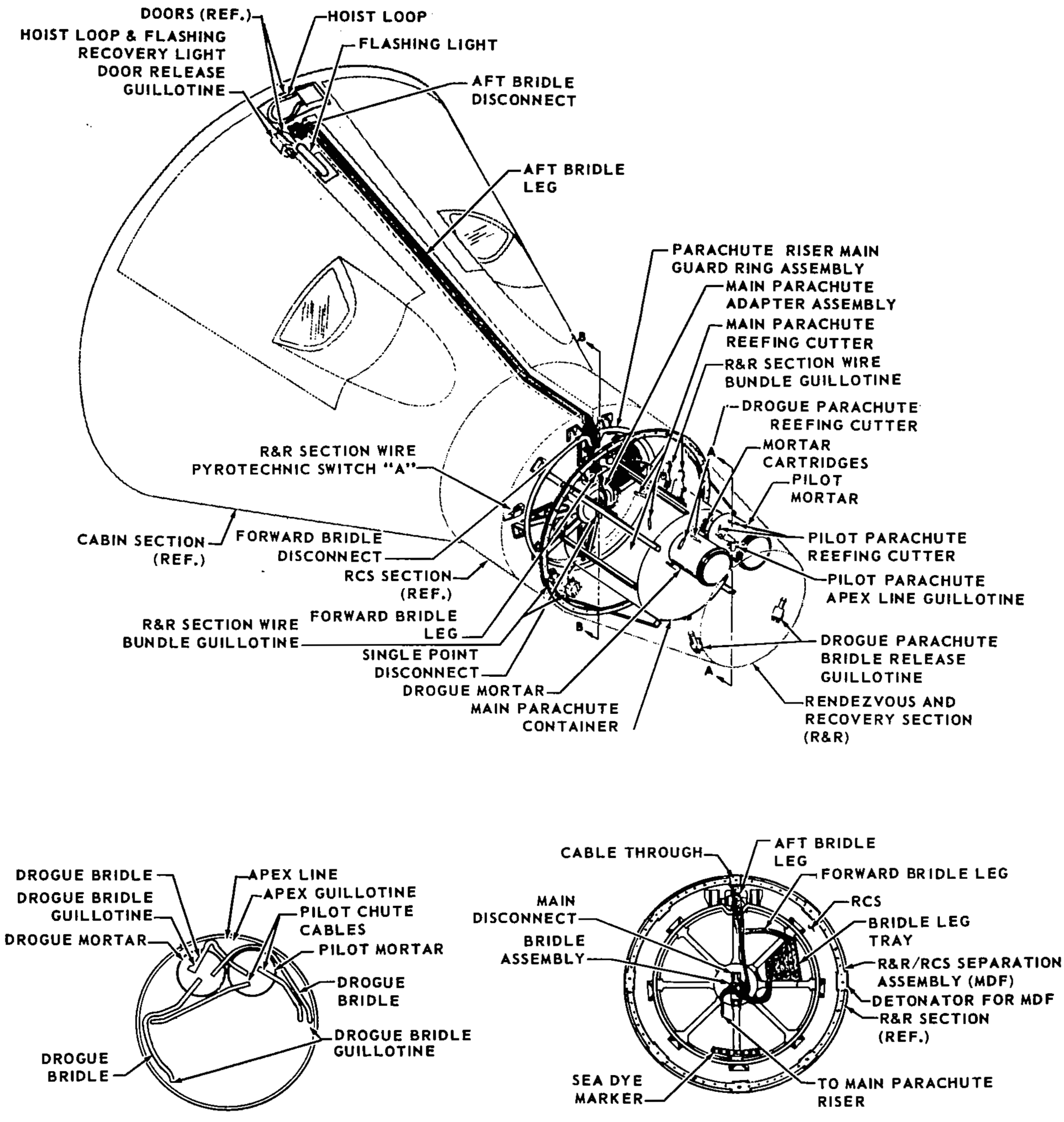
COMPONENT	VENDOR	QUAL TEST	RELIABILITY ASSURANCE TEST	SLED	SOPE	F-106	DUMMY DROP	LIVE DROP	SPECIAL TEST @ VENDOR
HARNESS RELEASE ACTUATOR (HRA)	OA/WAC	29	10	6	5	3	-	-	-
	CTI	21	8	-	-	-	-	-	16
FIRING MECH (HRA)	OA/WAC	33	10	6	5	3	-	-	-
	CTI	21	8	-	-	-	-	-	16
CARTRIDGE (HRA) (1.08 SEC TIME DELAY)	CTI	21	8	-	-	-	-	-	16
SEAT-MAN SEPARATOR THRUSTER (SMS)	OA/WAC	26	19	6	5	3	-	-	-
	CTI	21	8	-	-	3	-	-	12
FIRING MECH (SMS)	OA/WAC	37	25	6	5	3	-	-	-
	CTI	21	8	-	-	-	-	-	12
CARTRIDGE (SMS)	CTI	21	8	-	-	-	-	-	12
ROCKET CATAPULT	RPI	89	-	6	5	-	-	-	-
HATCH ACTUATOR	NATIONAL WATERLIFT	26	-	6	5	-	-	-	-
BREECH-HATCH ACT	MCDONNELL	31	-	6	5	-	-	-	-
BALLUTE ANEROID	MCDONNELL/ OA	15	-	6	5	3	18	20	-
BALLUTE REL CARTRIDGE	CTI	24	8	-	-	-	-	-	8
BALLUTE DEPLOY CARTRIDGE	CTI	24	8	-	-	-	-	-	8
DEPLOY CUTTER	OA/WAC	17	2	6	5	3	18	18	-
	CTI	24	8	-	-	-	-	-	8
BALLUTE GUILLOTINE	OA/WAC	17	3	6	5	3	18	18	-
	CTI	24	8	-	-	-	-	-	8
DROGUE MORTAR BACKBOARD JETTISON SYSTEM									
MAIN CARTRIDGE	CTI	21	14	-	-	-	-	-	15
INITIATOR CARTRIDGE	CTI	21	19	-	-	-	-	-	15
TIME DELAY CARTRIDGE	CTI	21	11	-	-	-	-	-	15
FLEX MDF INTER	CTI	10	-	-	-	-	-	-	-
RIGID MDF INTER	OA	24	12	12	10	6	36	36	-
RESTRAINT STRAP CUTTER	OA	12	-	6	5	3	18	18	-
LAP BELT DISC	OA	7	-	6	5	3	18	18	-
JETELOX PIN DISC	OA	7	-	6	5	3	18	18	-
MDF MANIFOLD	OA	6	-	6	5	3	18	18	-
MANUAL FIRING MECH	OA	15	-	-	-	-	-	-	-
MDF FIRING MECH	OA	19	34	6	5	3	18	18	-
DROGUE MORTAR	MCDONNELL/ OA	7	-	6	5	3	18	18	-
HATCH MDF INITIATION SYSTEM									
CROSSOVER	CTI	-	-	-	-	-	-	-	-
RIGID MDF INTER	CTI	-	-	-	-	-	-	-	-
FLEX MDF INTER	OA	50	-	6	6	-	-	-	-
MANUAL INITIATOR	OA	88	10	6	6	-	-	-	-
CARTRIDGE (HRA)	*OA	40	10	6	5	3	-	-	-
CARTRIDGE (SMS)	*OA	34	19	6	5	3	-	-	-
MAIN CARTRIDGE (BACKBOARD JETT)	*OA	18	-	6	5	3	18	18	-
INITIATOR CARTRIDGE (BACKBOARD JETT)	*OA	18	45	6	5	3	18	18	-
TIME DELAY CARTRIDGE (BACKBOARD JETT)	*OA	18	23	6	5	3	18	18	-
FLEX MDF INTER (BACKBOARD JETT)	*OA	30	24	12	10	6	18	18	-
CROSSOVER (HATCH MDF)	*OA	50	-	12	10	-	-	-	-
RIGID INTER (HATCH MDF)	*OA	100	20	12	12	-	-	-	-
RELEASE CART ((BALLUTE)	*OA	27	3	6	5	3	20	16	-
DEPLOY CART (BALLUTE)	*OA	27	2	6	5	3	6	16	-

*SIMILAR UNITS THAT WERE UTILIZED IN TEST AND ON EARLY SPACECRAFT, BUT WERE NOT AVAILABLE FOR SPACECRAFT 5 AND UP.

NOTE: OA/WAC – TESTS AT WEBER AIRCRAFT CORP.
CTI – TESTS AT CENTRAL TECHNOLOGY INC.

TABLE 23 ESCAPE SYSTEM FAILURE SUMMARY

TEST	FAILURE	CORRECTIVE ACTION
SLED #8	LH SIDE PANEL OF EJECTION SEAT HAD STRUCTURAL FAILURE DUE TO LATERAL LOADING CONDITION.	STRUCTURAL ATTACHMENT OF SIDE PANELS TO SEAT PANEL MODIFIED. FOOT STIRRUPS MODIFIED TO RE-STRAIN LATERAL FOOT MOVEMENT.
SOPE #12	HATCH ACTUATOR SUPPLIED PRESSURE TO ROCKET CATAPULT PREMATURELY CAUSING EJECTION OF SEAT PRIOR TO COMPLETE OPENING OF HATCH. SEAT AND DUMMY IMPACTED AGAINST HATCH DURING ATTEMPTED EJECTION.	HATCH ACTUATOR BREECH END CAP MODIFIED TO ELIMINATE PREMATURE PRESSURE DUMP TO ROCKET CATA-PULT BY INCORPORATING REDUNDANT SEALS AND INCLUDING A VENT AREA TO INSURE NO PREMATURE PRESSURE BUILD-UP.
DUMMY DROP TEST #1 & 2	SUIT OXYGEN HOSES AND ELECTRICAL WIRE BUNDLE (JETELOX) FAILED TO DISCONNECT, WHICH PREVENTED JETTISON OF EGRESS KIT.	HOSE FITTINGS IN DISCONNECT MODI-FIED TO PREVENT BINDING. SEPARA-TION COMPRESSION SPRINGS STRENGTH-ENED.
LIVE DROP TEST #2	BACKBOARD AND EGRESS KIT FAILED TO JETTISON AFTER PERSONNEL PAR-ACHUTE DEPLOYED, AS RESTRAINT STRAP FLEXIBLE LINEAR-SHAPED CHARGED (FLSC) CUTTER DID NOT CUT STRAP.	QUALITY CONTROL PROBLEM. DOWEL PINS MISSING FROM FLSC ASSEMBLY. INSPECTION CHECK POINT WAS ADDED DURING INSTALLATION OF RESTRAINT STRAP IN CUTTER.
DUMMY DROP	ANEROID MECHANISM JAMMED OR "HANG FIRED" AND FAILED TO FIRE OR FUNC-TION AT THE PROPER PRECALIBRATED ALTITUDE.	MCDONNELL REDESIGNED AND RE-WORKED THE ANEROID MECHANISM TO ELIMINATE THE HANG UP.
QUAL TEST	ANEROID MECHANISM FAILED AFTER EXPOSURE TO A HUMIDITY ENVIRON-MENT PER MIL-E-5272 PARA 4.4.1.	EXAMINATION, ANALYSIS, AND RE-PEATED TEST OF THESE ASSEMBLIES. THE ORIGINAL TESTS WERE NOT REA-LISTIC. SPACECRAFT 3 & 4 UNITS WERE POST FLIGHT FIRED AND WERE WITH-IN TOLERANCE. A 14-DAY HUMIDITY TEST WAS CONDUCTED ON THE UNITS FROM SPACECRAFT 4 AND THE UNITS WERE IN TOLERANCE.
QUAL TEST	JETELOX LINES FAILED TO FIRE DUR-ING QUAL TESTING.	HYDROSTATIC PRESSURE CHECK OF LINES WAS DELETED SO THAT NO WATER WOULD BE LEFT IN THE LINES. LINES ARE NOW CHECKED WITH DRY NITROGEN.
QUAL & FIELD TEST	PERCUSSION CARTRIDGES FAILED DUE TO THE LACK OF SOLID SUPPORT FOR THE PRIMER.	BOTH DESIGN AND PRODUCTION DE-FICIENCIES EXISTED. THE PERCUSSION CARTRIDGES WERE REDESIGNED TO HAVE SOLID SUPPORT FOR THE PRI-MERS AT SECOND SOURCE VENDOR AND THE NEW CARTRIDGES WERE QUAL TESTED AND UTILIZED SPACECRAFT 5 AND UP.



SECTION A–A

SECTION B–B

FIGURE 42 LANDING & RECOVERY SYSTEM

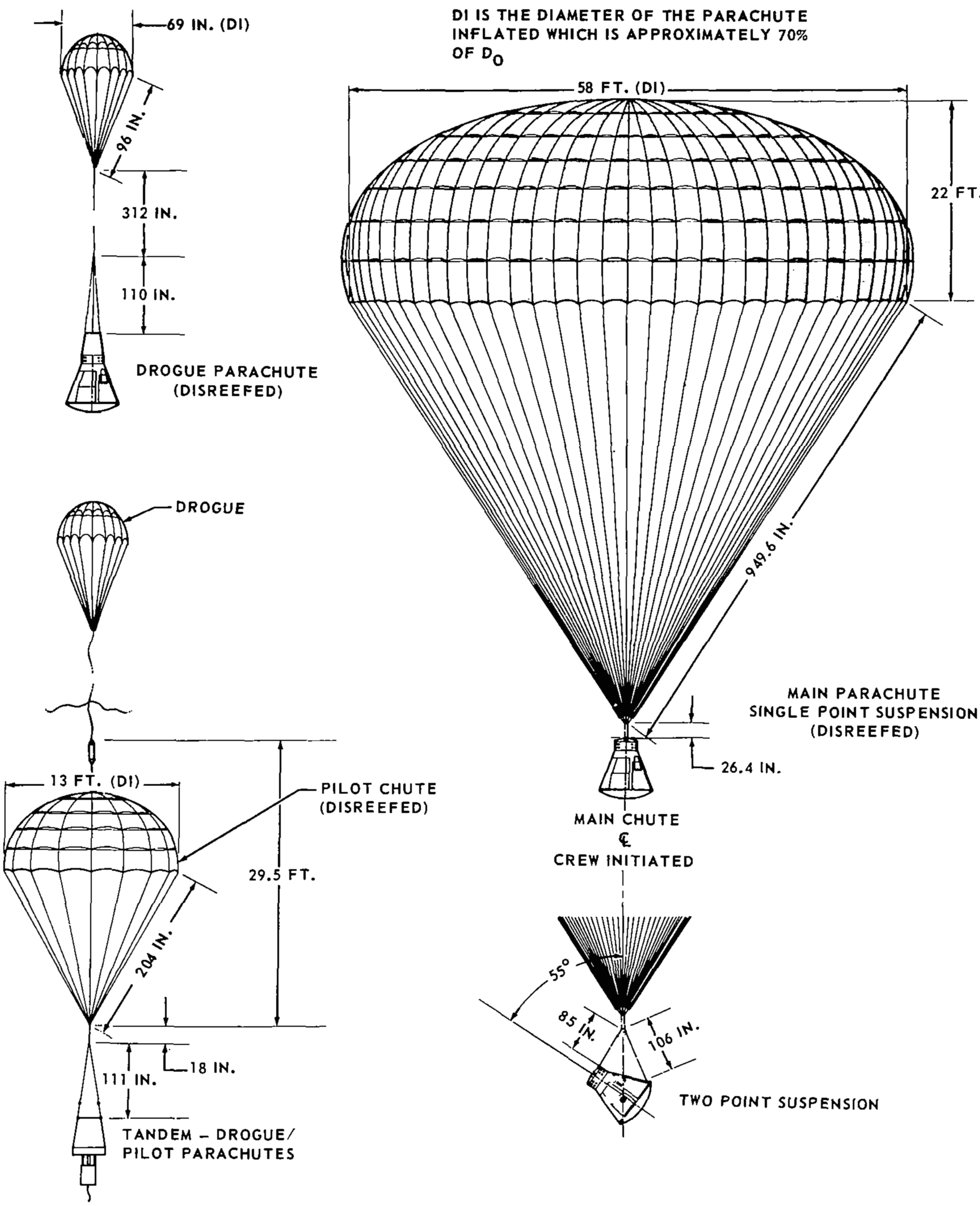


FIGURE 42 LANDING & RECOVERY SYSTEM (Continued)

chute canopy can withstand a dynamic pressure of 120 psf. However, by reefing the main chute, a maximum of 16,000 lb is experienced at deployment. The disreefed main parachute allows a maximum average rate of descent of 31.6 fps for a module weight of 4400 lb.

R & R Re-entry Separation Assembly. - A mild detonating fuse (MDF) ring assembly separates the rendezvous and recovery module from the re-entry vehicle. This separation assembly primarily consists of MDF, MDF housing/mating ring, two detonators, two detonator housings, two booster charges, and frangible bolts. Two strands of high explosive (5 gr/ft of RDX) encased in lead sheathing are installed in parallel grooves in the housing ring, which faces the re-entry vehicle. The grooves converge at the booster charges which are installed approximately 180 degrees. The MDF housing is fastened to the R & R module which is attached to the RCS of the re-entry vehicle by 24 frangible bolts. The detonator housings are installed in the RCS so that the detonators are indexed directly under the booster charges when the sections are mated. The detonators have redundant electrical circuitry to insure activation of the booster charges. Each booster charge strengthens the detonation wave and transmits it to the dual strands of MDF. When initiated, the MDF exerts a force against the RCS and the R & R section mating surface. This force breaks all the frangible bolts and allows the pilot chute to pull the R & R module free of the re-entry vehicle, thereby deploying the main parachute from its container in the R & R module.

Main Parachute Disconnects. - The main parachute disconnects include the single point disconnect assembly, and the forward and aft bridle disconnect assemblies. The disconnect assemblies are identical in design and function. The assembly consists of a breech assembly, arm, and two electrically fired gas pressure cartridges. The arm with the parachute strap attached is held in position against the breech by a piston. The cartridges produce gas pressure in the breech which exerts a force on the piston, propelling it all the way into the arm and into a lead slug. This action prevents the piston from hindering arm operation. The pull of the parachute causes the arm to cam open after release, thus releasing the attached riser or bridle.

Landing System Pyrotechnics. - Additional pyrotechnic devices associated with the landing system consist of:

A. Wire bundle guillotines utilized to sever the two electrical wire bundles routed from the re-entry vehicle to R & R module. For redundancy, two cutters are provided for each wire bundle, one on each side of the separation plane.

B. A pyroswitch to deadface the electrical circuitry before severance by guillotine action.

Static System. - The static pressure system operates the rate of descent indicator, altimeter, and the two barometric pressure switches which provide for illumination of the 10.6 K and 40 K warning indicators.

Recovery Aids. - Flotation of the re-entry vehicle is achieved by displacing water with the cabin vessel and the equipment in the floodable bays. The flotation characteristics are a function of the cg location. To improve the flotation attitude, additional flotation material (styrofoam) was installed under the equipment in the side bays and inside the RCS.

A sea dye marker is utilized as a visual rescue aid. The dye is packaged in a Fiberglas-reinforced plastic laminate container which releases the dye through openings covered with water soluble film. The container is mounted below the flotation line on the forward end of the RCS and is exposed when the R & R module is jettisoned. A cable cutter utilizing two electrically fired gas pressure cartridges, each with a separate circuit, severs a holding cable after main parachute jettison. This allows the spring-loaded hoist loop and flashing recovery light to jettison their protective doors. The hoist loop, which is located near the heat shield and between the hatches, provides a hoisting attachment point for lifting devices used for recovering the re-entry vehicle. The flashing recovery light is located forward of the hoist loop and provides a visual rescue aid. The flashing light, which utilizes its own battery power independent of the spacecraft power supply, is energized by the positioning of the RESC beacon control switch.

Landing System Operation (Fig. 43). - Operation of the landing system is as follows:

When the altimeter indicates 50,000 ft, the HI-ALT drogue switch is activated manually by the astronaut. The drogue switch energized two single-circuit pyrotechnic cartridges in the drogue mortar. The drogue mortar deploys the drogue chute reefed to 43% of the parachute diameter in order to reduce the opening shock load. Sixteen sec after deployment, two lanyard-initiated pyrotechnic reefing cutters disreef the chute. A barostatically operated instrument panel light is illuminated at 40,000 ft to signal that the drogue chute should have been deployed. At 10,600 ft another barostatically operated light signals the astronaut to actuate the PARA switch, which initiates the main parachute deployment sequence.

The PARA switch energizes six 80 msec pyrotechnic time delay cartridges, which are installed in the three high altitude drogue cable guillotines. After the drogue risers have been cut, the dorgue parachute pulls away from the re-entry vehicle extracting the pilot parachute from the pilot mortar tube with the apex line. When deployed, the pilot chute is reefed to 11.5% in order to limit the opening shock load to 3000 lb. The PARA switch also energizes a 2.5 sec time delay relay to the separation sequence to allow sufficient time for the pilot chute to be deployed. The 2.5 sec time delay relay then energizes the cartridge in a pyroswitch to deadface the electrical wire bundles to the R & R module. The four cartridges installed in the wire bundle guillotines also are energized. These guillotine cartridges have a 120 msec pyrotechnic time delay to assure deadfacing of the wires before severing. The two MDF ring detonators are energized at the same time to fire the MDF ring which separates the R & R module from the re-entry vehicle. The MDF detonators have a 160 msec pyrotechnic time delay to assure that the wire bundles have been cut before the MDF ring fires. The wires to the re-entry

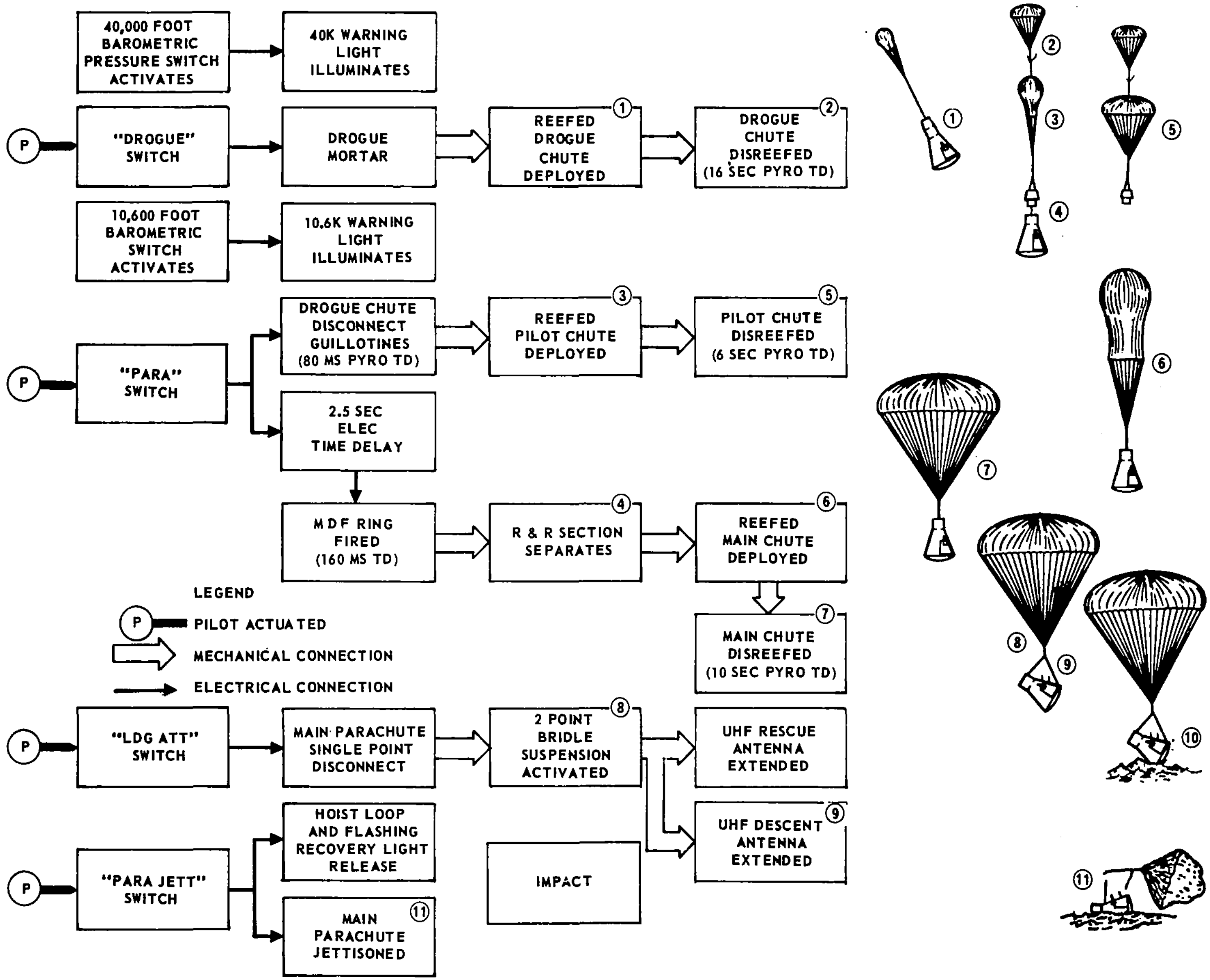


FIGURE 43 LANDING SYSTEM SEQUENTIAL BLOCK DIAGRAM

umbilical also are deadfaced at this time by energizing another pyroswitch cartridge.

As the landing vehicle falls away from the R & R module, the main parachute is deployed in a reefed condition (10.5%). Six sec after deployment of the pilot parachute, two lanyard-initiated pyrotechnic reefing cutters disreef the pilot chute. The main parachute is disreefed by three lanyard-initiated pyrotechnic reefing cutters ten sec after deployment. The two decelerations provided by the main parachute divide the retarding shock load. The astronaut visually monitors all parachute deployments and after the main chute disreefs and is fully inflated, he actuates the LDG ATT switch. The LDG ATT switch energizes two pyrotechnic cartridges in the single point release (main disconnect) to change the single point suspension system to a two-point suspension system. Upon landing, the main parachute is jettisoned by activating the PARA JETT switch. The PARA JETT switch energizes the forward and aft bridle disconnects releasing the main parachute. The PARA JETT switch also energizes the two single pyrotechnic cartridges in the cable cutter allowing the hoist loop and flashing light to be erected.

Emergency Operation of Landing System (Fig. 44). - In the event the drogue parachute does not deploy or deploys improperly, the drogue emerg. 10.6 K switch is actuated. This switch fires the three drogue cable guillotines, the apex line guillotine, and the pilot parachute mortar, and initiates the 2.5 sec time delay to the separation sequence. After the pilot mortar deploys the pilot parachute in a reefed condition, the remaining emergency sequence is the same as that for a normal landing.

Landing System Development and Qualification Tests. - The landing system development and qualification tests are shown on Table 24. The number of actual firings of the pyrotechnic components, and devices utilized in the landing system is shown on Table 25. In addition, verification firings of units from the same lots as those installed in the spacecraft were made before each flight.

Failure Summary. - The failure summary is shown on Table 26.

Mission Anomalies. - The landing system had the following anomalies on Spacecraft 2:

A. Problem - One wire bundle guillotine on the RCS failed to completely cut a wire bundle at the R & R separation plane. The redundant guillotine installed on the R & R section did sever the wire bundle which allowed the separation of the R & R re-entry sections.

Investigation/Conclusion - The wire bundle in question was installed in an abnormal position. It was pulled up tightly against the cutter blade which requires additional energy of the cartridge. The cartridge was developed to cut the wire bundle when it was held against the anvil and did not have sufficient margin of energy to handle the abnormal, installation. The failure was duplicated in the failure lab.

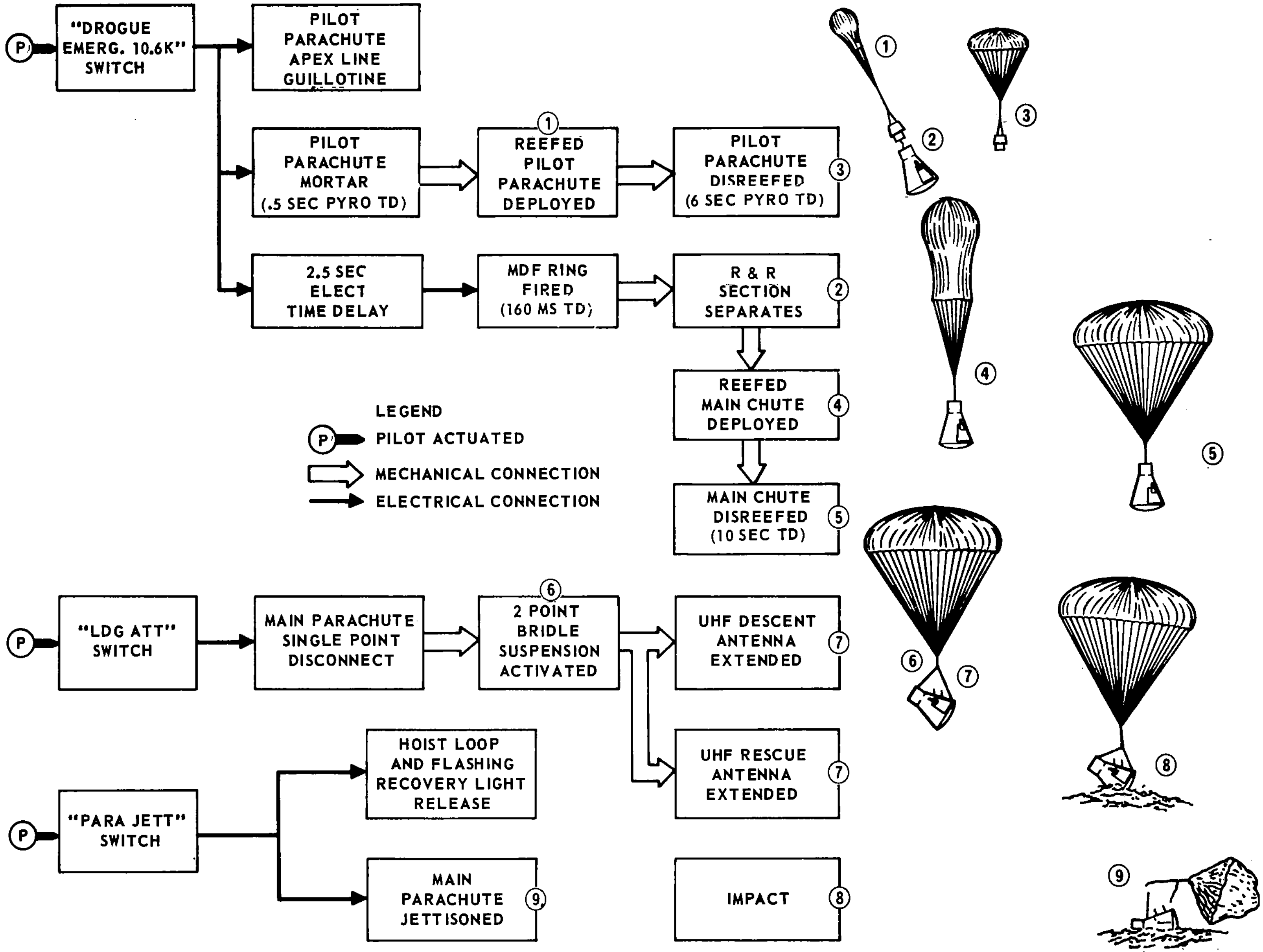


FIGURE 44 EMERGENCY LANDING SEQUANTIAL BLOCK DIAGRAM.

TABLE 24 DEVELOPMENT AND QUALIFICATION TESTS

LANDING SYSTEM DROP TEST

TEST	DROGUE PARACHUTE	PILOT PARACHUTE	MAIN PARACHUTE	LANDING SYSTEM
DEVELOPMENT	14	18	30	14
TYPE OF TEST	PTV (BOMB) DROP TEST FROM AIRCRAFT 42k	PTV (BOMB) DROP TEST FROM AIRCRAFT 10k	PTV (BOMB) DROP TEST FROM AIRCRAFT 10k	BOILERPLATE SPACECRAFT DROP TEST FROM AIRCRAFT 20k
QUALIFICATION TEST	8	13	13	13 3 - WITHOUT DROGUE SPACECRAFT 2 CONFIGURATION 2 - EMERGENCY MODE
TYPE OF TEST	BOILERPLATE SPACECRAFT WITH PRODUCTION RCS & R&R SECTION. DROP TEST FROM AIRCRAFT 3 AT 20k - SPACECRAFT 2 CONFIGURATION 2 AT 17k - EMERGENCY MODE 8 AT 33k - NORMAL LANDING MODE			

TEST COMPLETED	TYPE OF TEST	REMARKS
20	ENVIRONMENTAL	TO QUALIFY BAROSWITCHES FOR ADDITIONAL ENVIRONMENTS NOT QUALIFIED BY PROJECT MERCURY. BAROSWITCHES IN GENERAL WERE QUALIFIED BY SIMILARITY TO PROJECT MERCURY HARDWARE.
2	STATIC TEST	TO QUALIFY THE INVERSION BRIDLE FOR STRUCTURAL INTEGRITY AFTER 24 DAYS OF EXPOSURE TO 160°F.

Action Taken - Effective Spacecraft 3 and up, the length of the wire bundle was adjusted to allow the blueprint installation of the wire bundle solid against the anvil. A CTI cartridge with higher energy output was used in lieu of the OA cartridge.

B. Problem - The flashing recovery light did not deploy.

Investigation/Conclusion - A review of engineering drawings and spacecraft revealed that a tolerance buildup could exist and result in binding the door on structure thus preventing the door from opening.

Action - Effective Spacecraft 3 and up, the door fitting was shortened to preclude any tolerance buildup interference and inspection requirements were added to assure satisfactory operation of the door after installation and prior to final rigging of door restraining cable.

TABLE 25 LANDING SYSTEM COMPONENT FIRINGS

COMPONENT	VENDOR	DROP TEST	SPECIAL TEST @ VENDOR	MCDONNELL TEST	QUAL TEST	RAT
CARTRIDGE APEX CUTTER	CTI	4	20	18	61	32
HIGH ALTITUDE DROGUE CABLE GUILLOTINE (HAD)	MCDONNELL/ OA	30	7	22	23	36
CARTRIDGE (HAD)	CTI	60	45	38	61	36
CARTRIDGE* DROGUE MORTAR	ORDCO	18	–	–	368	28
CARTRIDGE* PILOT MORTAR	CTI	–	5	–	–	–
CARTRIDGE* MAIN DISCONNECT	SDI	–	–	–	353	19
CARTRIDGE* BRIDLE DISCONNECT	SDI	–	–	–	78	19
BRIDLE DISCONNECT ASSEMBLY	N–V	54	–	2	21	19
	@ CTI	–	5	–	21	19
CARTRIDGE BRIDLE DISCONNECT	CTI	–	21	–	42	19
DROGUE MORTAR ASSEMBLY	N–V	33	–	1	–	28
	@ CTI	–	21	21	21	25
CARTRIDGE DROGUE MORTAR	CTI	–	21	32	24	25
PILOT MORTAR ASSEMBLY	N–V	27	–	2	72	–
	@ CTI	–	4	–	–	–
CARTRIDGE PILOT MORTAR	ORDCO	8	-	–	368	–
R&R SEPARATION MDF RING ASSY	OA	27	–	19	40	18
APEX GUILLOTINE	MCDONNELL OA	2	5	10	23	32
REEFING CUTTER PILOT CHUTE (6 SEC TD)	OA	32	–	–	63	12
REEFING CUTTER MAIN CHUTE (10 SEC TD)	OA	106	–	–	53	12
REEFING CUTTER DROGUE CHUTE (16 SEC TD)	NOP	–	–	62	32	10
REEFING CUTTER* DROGUE CHUTE (16 SEC TD) (BACKUP)	SOS	16	–	–	103	19
MAIN DISCONNECT ASSEMBLY	N–V	27	–	1	71	19
	@ CTI	–	5	–	21	19
CARTRIDGE MAIN DISCONNECT	CTI	–	5	–	42	19

*SIMILAR TO UNITS UTILIZED ON SPACECRAFT 3 AND UP.

TABLE 26 FAILURE SUMMARY

PROBLEM	CORRECTIVE ACTION
PRIME VENDOR HAD DIFFICULTIES WITH MANUFACTURING AND QUAL TESTING OF DETONATORS, GAS GENERATING CARTRIDGES, SOME GUILLOTINES AND PYROSWITCHES.	SPACECRAFT 2 UTILIZED THE PRIME VENDORS CARTRIDGES AND DETONATORS WITH DUAL BRIDGE WIRE CIRCUITS. ABBREVIATED QUAL TESTS WERE CONDUCTED ON THESE UNITS WHICH CONSISTED OF HI-TEMP/ALT, HUMIDITY, VIBRATION, AUTOIGNITION, PROOF AND BURST PRESSURE, 40-FT DROP, 1 AMP NO-FIRE AND 1 WATT NO-FIRE. THE UNITS WERE ACCEPTANCE TESTED AS FOLLOWS: BRIDGE WIRE RESISTANCE, LEAKAGE, VIBRATION, 200°F TEMP, SOAK 1/2 HR, RESISTANCE & DIELECTRIC.
	SPACECRAFT 3 – MCDONNELL ASSUMED THE DESIGN AND MANUFACTURING RESPONSIBILITY OF THE GUILLOTINES AND PYROSWITCHES WHICH WERE NOT AVAILABLE. MCDONNELL PROCURED SECOND SOURCE VENDORS FOR DETONATORS AND CARTRIDGES. THE DETONATOR AND CARTRIDGES WITH THEIR ASSOCIATED DEVICES WERE QUALIFIED BY THE SECOND SOURCE VENDORS AS REQUIRED. DISCONNECT CARTRIDGES – SPECIAL DEVICES INC. MORTAR CARTRIDGES – ORDCO DETONATOR AND CARTRIDGES UTILIZED FOR GUILLOTINES, PYROSWITCHES AND TUBE CUTTER – CTI.
	SPACECRAFT 4 AND UP – CTI CARTRIDGES AND DETONATORS WERE UTILIZED ENTIRELY WITH THE LANDING SYSTEM EXCEPT FOR THE PILOT MORTAR (ORDCO).
MORTAR BREECH BLOCK FAILED DURING ENVIRONMENTAL QUAL TESTING, WITH ORDCO CARTRIDGES.	PILOT MORTAR BREECH MATERIAL WAS CHANGED FROM ALUMINUM TO STEEL. UNITS WERE RETESTED WITHOUT ANY FAILURES. DROGUE MORTAR BREECH WAS TESTED AT CTI WITH THEIR CARTRIDGES REPEATEDLY WITHOUT ANY ANOMALIES.
	SPACECRAFT 4 AND UP CONFIGURATION – CTI CARTRIDGES IN THE DROGUE MORTAR WITH AN ALUMINUM BREECH AND PILOT MORTAR WITH STEEL BREECH AND ORDCO CARTRIDGES.

Mission Evaluation. - The landing and recovery systems functioned successfully on all missions.

PYROTECHNICS

Pyrotechnic devices as shown on Fig. 45 perform many of the spacecraft sequential operations. They provide the modes to function or diable systems and separate or jettison various sections or assemblies. The pyrotechnic system consists of a wide range of explosive devices, their actuation system, and related equipment. The systems utilize both low explosives which burn and generate gas pressure, and high explosives which detonate and generate a shock front. Hot gas pressure from the slowly decomposing low explosives is used to actuate a device or propel a projectile. Pyrotechnic devices are described in subsequent paragraphs except for the pyrotechnics utilized for the escape system, recovery system, and the retro-rockets, which are discussed in their appropriate sections.

The pyrotechnic devices in the spacecraft are installed so that the explosions are confined within the device or are shielded to preclude damage to the spacecraft.

A reliability program was conducted throughout the design, development, fabrication, and qualification of the pyrotechnic system. Testing by McDonnell and its vendors insured a system reliability of .995. Before each flight a unit from the same manufacturing lot as the pyrotechnic device installed on the spacecraft was test fired for verification.

Pyrotechnic Components

Pyrotechnic items, such as cartridges, detonators, flexible linear-shaped charges (FLSC), guillotines, and pyroswitches were used extensively throughout the spacecraft. Their description and operation are outlined below. Later sections describe their use in various systems.

Cartridges and Detonators. - The electrically initiated cartridges and detonators have the same basic design. They consist of a body, bridgewire, circuit, ignition mix, and an output charge. In some instances, a pyrotechnic time delay column was used to provide a delay between ignition mix and output charge ignition. The cartridges utilize low explosives to produce a specific gas pressure output to operate the device in which they are installed. The detonators utilize high explosives which transmit a shock wave to the assembly in which they are attached. Cartridges and detonators are provided with one or two electrical bridgewire circuits. The second circuit is redundant. Each bridgewire circuit is independent of the cartridge or detonator body.

The body of the cartridge or detonator provides for the installation of the explosive train. The bodies are threaded on one end for installation into the device. An electrical receptacle is machined on the other end for connection to the spacecraft wire bundle.

Each firing circuit, consisting of two electrical pins with a bridgewire attached between them, is installed in the body. The electrical pins are insulated from the body by a ceramic material.

The heat-sensitive mix, which encases the bridgewire, ignites when electricity passes through the bridgewire. This in turn initiates the next explosive element in the train of explosives. The number and type of explosive elements in the train from the ignition mix to the final output charge vary with requirements. Cartridges are designed with an energy output

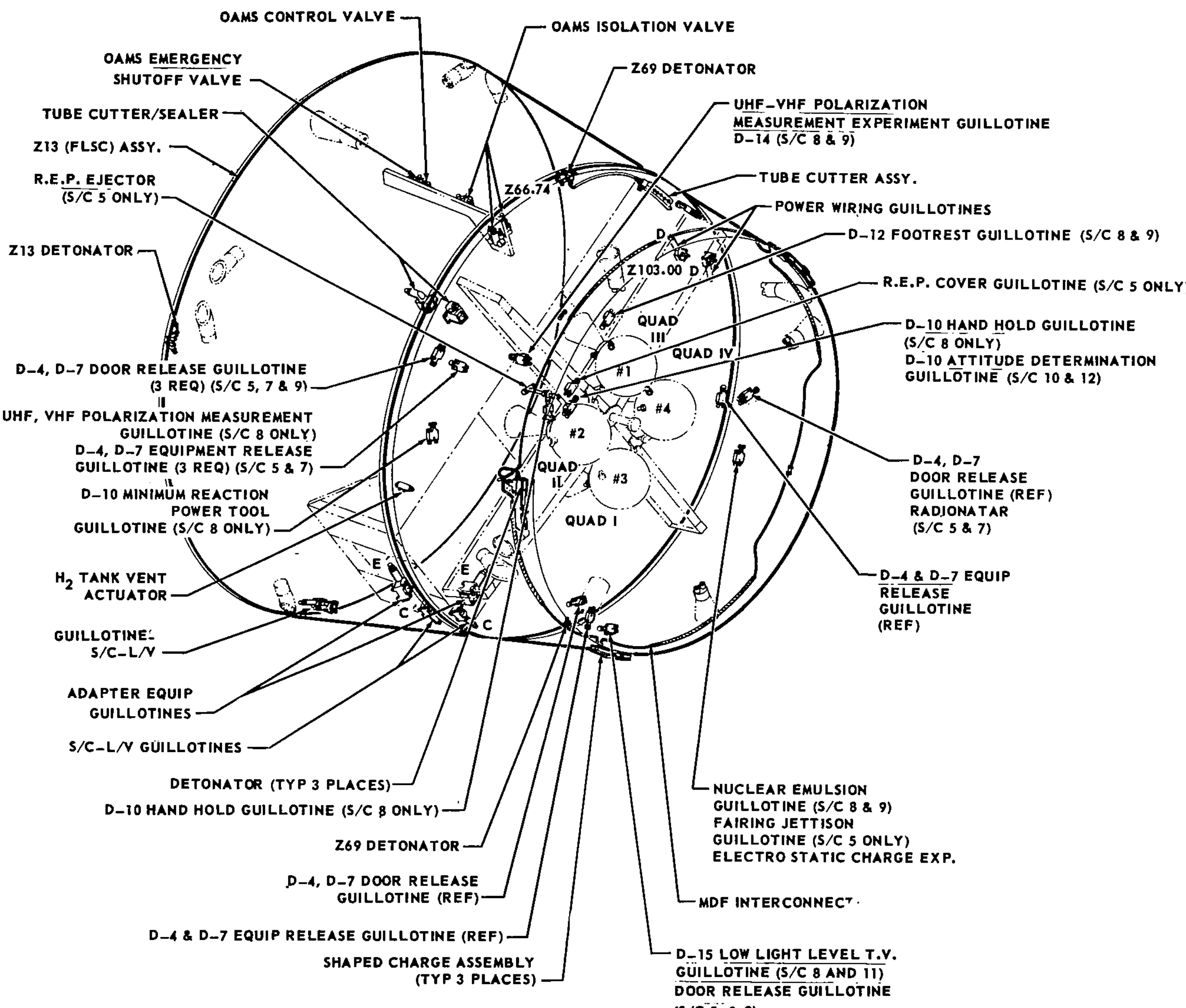


FIGURE 45 SPACECRAFT PYROTECHNIC DEVICES

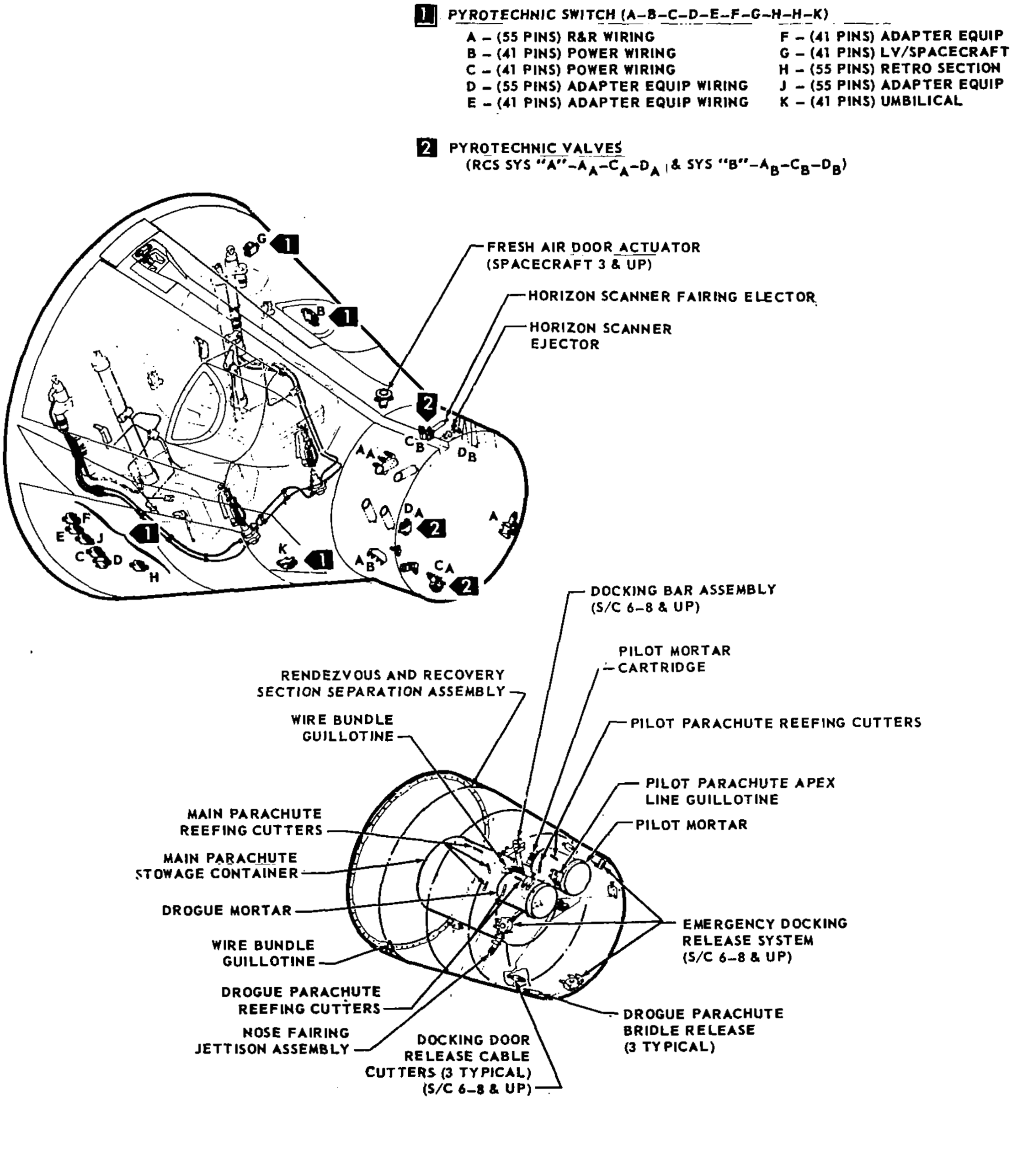


FIGURE 45 SPACECRAFT PYROTECHNIC DEVICES (Continued)

of 1.5 times the energy required to operate the device. Detonators have an energy output of 1.3 times that necessary to initiate the next inline explosive element and will produce sufficient energy to propagate across a gap 1.5 times the nominal design gap between the detonator and the next inline explosive element.

The design of the cartridges and detonators also includes the following:

A. The cartridges and detonators without pyrotechnic time delays begin to function within 0.004 sec after 15 VDC is applied to the bridgewire circuit.

B. Bridgewire circuit resistance is 1.0 $\pm$ 0.2 ohm.

C. The cartridges and detonators have a dielectric strength capable of withstanding:

1. Five hundred volts (RMS) for one min at sea level between all pins tied together and the case.
2. Two hundred fifty volts for 20 min at altitude conditions.

D. No-Fire Requirements.

1. The units will not fire inadvertently due to transient currents or static charges.
2. The units will not fire when one ampere or one watt is applied simultaneously to either or both circuits for five min.

E. All-Fire Requirements - The units will fire with 40 msec when four amperes is applied to the circuit.

Flexible Linear-Shaped Charge. - The flexible linear-shaped charge (FLSC) is a V-shaped, flexible lead sheathing containing a high explosive RDX core. FLSC is used in the separation assemblies to sever various materials. The FLSC is installed in a molded back blast shield with the open portion of the V-shaped sheathing placed toward the material to be severed. The FLSC explosive core, set off by the detonator, collapses the sheathing in the V-groove. This produces a cutting jet composed of explosive products and minute metal particles which cut the target material. The FLSC is designed with a minimum cutting ability of 20% greater than required.

Guillotines. - Guillotines are designed in two basic configurations, one to cut electrical wire bundles, the other to cut stainless steel cables.

A. Wire Bundle Guillotines - Wire bundle guillotines are provided in two sizes to cut bundles up to 1-1/4 and 2-1/2 in. in diameter. The guillotines consist of a body, end cap or anvil, piston/cutter blade, shear pin, and an electrical-fired gas-pressure cartridge. The body houses the piston/cutter blade held in a retracted position by a shear pin and provides for installation of the cartridge and attachment of the anvil. The anvil is removable to facilitate installation of either the guillotine or the wire bundle or cable. Lugs, for attaching the guillotine to the spacecraft structure, are an integral part of the guillotine body. The pressure produced by firing the cartridge exerts a force on the piston/cutter blade. When sufficient force is applied, the piston/cutter blade first severs the shear pin, and then the wire bundle with a high velocity stroke. The stroke of the piston/cutter blade is stopped by the anvil. The wire bundle is then free of the guillotine body. Two guillotines are used per wire bundle, one on either side of the separation plane.

B. Cable-Cutting Guillotines - Cable-cutting guillotines are provided in three sizes to sever cables up to 1/8, 1/4, and 9/32 in. diameters. The guillotine consists of a body, piston/cutter blade, shear pin, anvil end cap, and two electrically fired gas-pressure cartridges. The body provides for the installation of the two cartridges and incorporates a port between the cartridge areas to the piston/cutter area. The body has an integral barrel section which houses the piston/cutter blade retained in the retracted position by a shear pin and provides for the attachment of the end cap. The barrel section is slotted for installation and positioning of the cable normal to the position/cutter blade. The anvil is retained in the barrel under the cable by the end cap. The anvil and end cap are removable to permit guillotine cable installation. Lugs, for attaching the guillotine to the spacecraft structure, are an integral part of the body.

Pyroswitches. - Pyrotechnic switches are used to open electrical circuits in various wire bundles before the bundles are severed. They are provided in two sizes, one with 41 electrical contacts, and the other with 55 pyroswitches consist of a body, piston, shear pin, spring lock, and electrically-fired gas-pressure cartridge. A dual-bridgewire cartridge with separate electrical circuits provides redundancy. The body houses the cartridge and the piston. Incorporated in opposite ends of the switch body are two electrical receptacles which contain hollow spring leaf contacts. These contacts are connected axially in the switch-closed position by pins mounted in the piston. The piston is held in place by a shear pin. The pressure produced by firing the cartridge is ported through the switch body to a flange on one side of the piston. When sufficient gas pressure is generated, the piston severs the shear pin and moves axially in the body. As the piston moves, the connector pins mounted in the piston are disengaged from the hollow contacts at one end and driven farther into the hollow contacts at the other end. The spring lock moves into place behind the piston to lock it, and residual gas pressure provides a redundant means for preventing piston rebound. The piston thus is maintained in the SWITCH OPEN position.

Separation Assemblies

Z 13 Separation Assembly. - The launch vehicle is separated from the spacecraft/adapter equipment section by cutting the structure skin with redundant dual strands of FLSC. The separation assembly consists of two strands of FLSC, three detonator blocks each containing a crossover and two boosters, and three detonators (two single-bridgewire and one dual-bridgewire), each with a pyrotechnic time delay of 240 msec. The FLSC is initiated when the redundant detonators receive the proper electrical signal from the SEP SPCFT switch. The detonators transmit a shock wave to the crossover charge in the detonator block, which in turn initiates the boosters. The boosters propagate the wave to the FLSC in both directions simultaneously. The FLSC detonates and functions redundantly to sever the adapter skin at the parting line (located at station Z 13). The FLSC is installed around the inner periphery of the adapter skin with the detonator blocks as an integral part of the assembly. The integral blast shield protects the structure and equipment from shrapnel. Proper detonation of one strand of FLSC will achieve separation.

The electrical circuitry to the launch vehicle is deadfaced by the activation of pyroswitch G (located in the re-entry vehicle). The electrical wire bundle to the launch vehicle is severed by wire bundle guillotines located on either side of the separation plane. The pyroswitch, two guillotines, and FLSC assembly are energized by the activation of the SEP SPCFT switch.

The sequence is as follows:

A. Initiated simultaneously are three detonators with 240 msec pyrotechnic time delay, two wire bundle guillotine cartridges with 120 msec pyrotechnic time delay and a relay with 50-70 msec electrical time delay.

B. The 50-70 msec relay initiates the pyroswitch cartridge to deadface the circuits, removing the power from the cartridges and detonators.

C. The cartridges fire after the 120 msec delay, activating the guillotines to cut the deadfaced wire bundle.

D. The detonators fire after the 240 msec delay, initiating the FLSC, to cut the structure for complete separation.

Z 69 Separation Assembly. - The equipment adapter section is separated from the spacecraft retro section by an FLSC assembly. The FLSC assembly consists of two main units: A separation assembly of FLSC which is similar in design and operation to the Z 13 assembly and a FLSC tubing cutter assembly.

A. FLSC Separation Assembly - The FLSC of the separation assembly is installed around the outer periphery of the adapter skin and insulated to protect it from launch heating. The adapter skin is slotted in three places under the FLSC to provide for the installation of three detonator blocks on the inside of the skin. One of the detonator blocks provides for installation of both a detonator and an explosive interconnect (MDF). The other two contain detonators only. The three detonators utilized for this separation have no pyrotechnic time delay. A sheet metal blast deflector, which is part of the adapter structure, protects the structure and equipment from shrapnel. The initiation and function of the separation assembly is the same as Z 13.

B. FLSC Tube Cutter Assembly - The FLSC tube cutter is mounted to the inside of the retrograde section of the adapter forward of the parting line (Z 69). This tube cutter severs 12 aluminum tubes and one nylon tube from the equipment section. The cutter assembly consists of a housing, molded back-up retainers, two strands of FLSC with boosters attached, explosive interconnect, and a dual bridgewire detonator. The FLSC is installed on opposite sides of the tubes in the back-up retainers of the housing and staggered to provide redundant cuts. The detonator is installed in the housing with its output end adjacent to the booster charge of one strand of FLSC. The explosive interconnect is installed with one end in the tube cutter housing and the other end in a detonator block of the separation assembly. The explosive interconnect is a single strand of high explosive (RDX) encased in a lead sheathing and covered by a flexible vinyl tube. Attachment fittings are incorporated on each end of the interconnect for installation. A small booster charge is incorporated at each end of the explosive strand, which is installed adjacent to the booster charge of the strand of FLSC and the booster charge of one FLSC section in the tube cutter. The explosive interconnect transmits detonation from the separation assembly to one strand of FLSC in the tube cutter. The detonator in the tube cutter and the detonators in the separation assembly are initiated simultaneously. Proper detonation of one strand of FLSC in both the separation assembly and tube cutter assembly, will achieve separation.

C. Electrical Wire Bundle Guillotine - The electrical circuitry to the equipment section is deadfaced by the activation of pyroswitches B, C, C, E, F, and J. The three separate electrical wire bundles to the equipment section are severed by wire bundle guillotines C, D, and E located on both sides of the separation plane.

D. OAMS Line Cutter - Two stainless steel, Teflon-lined tubes containing the hypergolic propellants used in the OAMS are sealed and severed by the tubing cutter/sealer located on both sides of the Z 69 separation plane. The tubing cutter/sealer assembly consists of a body, anvil, electrically fired gas pressure cartridge, four shear pins, and cutter assembly. The cutter assembly consists of a piston, crimper, and blade. The crimper and blade are attached to the piston by two shear pins (sequencing pins). The body houses the cutter assembly retained in a retracted position by two shear pins (lock pins), and provided for installation of the cartridge and attachment of the anvil. The gas pressure produced by firing the cartridge exerts a force on the piston of the cutter assembly. When sufficient force is applied, the two lock pins are severed and the cutter assembly actuates with a high velocity. The blade and crimper, which extend past the end of the piston, contact the two tubes first. The crimper flattens the tubing against a raised portion of the anvil and comes to a halt, shearing the two sequencing pins that hold the crimper and blade. The piston and blade continue to travel, allowing the blade to sever the tubing. Proper functioning of one tubing cutter/sealer will achieve separation.

E. Sequencing - Three switches achieve separation of the adapter equipment section from the spacecraft retro section.

Sequencing is as follows:

1. SEP OAMS Line Switch

a. Initiates simultaneously two tubing cutter/sealer cartridges (instantaneous type), three wire bundle guillotine (C' -D' -E') cartridges (with 120 msec pyrotechnic time delay) installed below the separation plane, and energizes a 50-70 msec electrical time delay relay.

b. The 50-70 msec relay initiates the pyroswitch (B-C-D-E-F-J) cartridges to deadface the circuits and remove electrical power from the cartridges and detonators.

c. The guillotine cartridges fire after the 120 msec delay, actuating the guillotines to cut the deadface wire bundles.

2. SEP ELECT Switch - Energizes a 50-70 msec electrical time delay relay which initiates simultaneously three wire bundle guillotine (C-D-E) cartridges (instantaneous type) installed above the separation plane to actuate the guillotines and cut the wire bundles.

3. SEP ADAPT Switch - Energizes a 70-90 msec electrical time delay relay which initiates simultaneously three detonators of the FLSC separation assembly and the detonator of the FLSC tube cutter.

Z 100 Separation Assembly. - The retrograde section separates from the re-entry vehicle when three titanium straps and various tubes and wire bundles are cut by redundant dual strands of FLSC. The separation assembly consists of three cutter assemblies, three detonator housing, three dual bridgewire detonators (160 msec pyrotechnic time delay) six explosive interconnects, and three inert unions. Each cutter assembly severs one titanium strap, electrical wires varying in size from No. 20 to coaxial cable, .35 diameter, and up to six 1/4 in. diameter tubes. The cutter assembly consists of two parallel machined bars. Two RDX FLSC cords, with their apexes 1/2 in. apart, are embedded in each bar. One end of each FLSC strip is bent 45 degrees away from the bar. When joined together with an 0.4 in. gap between bars, the FLSC strips face each other and the bend ends converge at the booster. Proper gap is maintained by spacers and bolts. The detonator housing is installed on one end of the cutter assembly and contains a parallel booster assembly and an interconnect booster. The detonator housing also provides for the installation of a detonator and two explosive interconnects with their output ends adjacent to the interconnect booster. The interconnect booster is a column of RDX explosive in a holder with two output ports which mate with the converged strands of FLSC. The parallel booster assembly consists of a main column, crossover column, and two dual output columns installed in a charge holder. The main booster column is installed with one end adjacent to the interconnect booster and the other end sitting on the crossover column. The output boosters have chevron-shaped output cans on both ends of each booster. The two output columns are parallel planes, adjacent to each end of the cross-over column. Each end of the output column is mated with the open V-groove of the FLSC.

The cutters are located in three places around the parting line (Z 100) and are linked by explosive interconnects. Two explosive interconnects joined by a union are installed between each cutter. Two explosive interconnects are installed in each detonator housing with their booster charges adjacent to the booster column of the housing. A detonator is installed in each housing with its output end adjacent to the booster column of the housing and to the boosters of the interconnects. The titanium strap, electrical wires, and tubes to be severed are installed between the two bars of the cutter. The FLSC is initiated when the redundant detonators receive the proper electrical signal from the JETT RETRO switch. The detonator transmits a shock wave to the interconnect booster, which relays propagation to the FLSC in the cutter at the point where they converge and simultaneously propagates the main booster of the parallel booster assembly. The main column charge relays propagation redundantly to the four strands of FLSC through the crossover and two output charges. The detonator also simultaneously propagates the two explosive interconnects.

The interconnects transmit the detonation wave to all three cutter assemblies. This is to assure detonation of three cutters, in the event one or two detonators should not function. Proper detonation of one set of two opposing strips of FLSC in each cutter is sufficient to achieve separation. The electrical circuitry to the retrograde section is deadfaced by pyroswitch H. This pyroswitch utilizes an 80 msec pyrotechnic time delay cartridge. The pyroswitch and separation assembly are energized by the JETT RETRO switch.

The sequence is as follows:

A. Initiated simultaneously are three detonators with 160 msec pyrotechnic time delay and the pyroswitch cartridge with 80 msec pyrotechnic time delay.

B. The pyroswitch cartridge fires after the 80 msec delay to deadface the circuit and also removes the electrical power from the cartridges and detonators.

C. The detonators fire after the 160 msec delay, initiating the FLSC, to cut the straps, tubes and deadfaced wires.

Fairing Release

The horizon scanner is uncovered by a pyrotechnic ejector which jettisons the horizon scanner fairing. The radar, parachute canisters, and nose antennas are uncovered by a pyrotechnic ejector which jettisons the nose fairing. Spacecraft 2, 3, and 4 utilized a cable cutter to sever a holding cable that releases the nose fairing for spring-driven jettison. The nose fairing and horizon scanner fairing are jettisoned by the JETT FAIRING switch.

Horizon Scanner Fairing Release. - The horizon scanner fairing release assembly secures the fairing to the spacecraft and, when initiated, jettisons it. The assembly consists of an actuator housing, actuator rod, main piston, release piston, eight locking pins, and two electrically fired gas pressure cartridges. The actuator rod forms a positive tie between the actuator and the scanner fairing. The actuator rod is locked to the main piston by four locking pins. The main piston is locked in the base of the housing by four locking pins that are held in place by the release piston. The release piston is spring-loaded in the locked position. The housing provides for installation of the pistons, actuator rod, and two cartridges and for mounting of the assembly. The gas pressure produced by firing the cartridges is ported through the housing to the base of the piston. The gas pressure moves the release piston forward, enabling the four locking pins to cam inboard and thus release the main piston. The gas pressure causes the main piston, with attached actuator rod, to move through the length of the housing. A shoulder stops the piston at the end of the housing. The four locking pins, securing the actuator rod to the piston, cam outboard into a recess and release the actuator rod. The actuator rod is jettisoned with the scanner fairing attached. Proper function of one cartridge is sufficient to jettison the fairing.

Nose Fairing Release. - The nose fairing ejector, which secures the rendezvous and recovery section nose fairing to the spacecraft, is initiated to jettison the fairing. The assembly consists of a breech, ballistic hose, actuator assembly, bellcrank mechanism, and a dual-bridgewire electrically fired gas pressure cartridge. The nose fairing is attached to the bellcrank mechanism. The actuator rod forms a positive tie between the actuator body and the crank mechanism. It is locked to a piston in the actuator by two locking pins and held in place in the actuator body by a shear pin. The actuator assembly is connected to the breech by the ballistic hose. The breech provides for installation of the cartridge and is positioned approximately nine in. from the actuator. The actuator is installed on the antenna support/actuator fitting of the rendezvous and recovery (R & R) section and is located on the X axis, five in. up from Y zero. The gas pressure produced by firing the cartridge in the breech is transferred through the ballistic hose to exert a force on the piston in the actuator. The piston with rod attached severs the restraining shear pin. The gas pressure causes the piston, with attached rod, to move through the length of the actuator housing. As the piston reaches the end of the housing, the two locking pins securing the rod to the piston cam outboard into a recess and release the actuator rod. The actuator rod then is jettisoned with the nose fairing. A hinge on the nose fairing, located on the outer moldline, releases and directs the path of the fairing away from the spacecraft. The bellcrank mechanism provides for angular jettison of the fairing using the axial movement of the ejector shaft.

Horizon Scanner Release

The horizon scanners are jettisoned from the spacecraft by the JETT RETRO switch. The horizon scanner release assembly secures the scanners to the spacecraft and jettisons them when actuated. The assembly consists of an actuator housing, actuator rod, locking mechanism, cartridge housing, and two electrically fired gas pressure cartridges with an 80 msec pyrotechnic time delay. The actuator rod is secured in the actuator housing by the locking mechanism. This mechanism consists of a tang lock, tang lock retainer, and a shear pin. The tang lock is secured to the base of the actuator housing. The actuator housing is attached to the spacecraft structure. The cartridge housing with both cartridges installed is attached to the actuator rod. The gas pressure produced by firing the cartridges is ported through the hollow actuator rod to the base of the actuator housing. Slots in the tang lock allow the gas pressure to exert a force against the base of the tang lock retainer. The tang lock retainer then severs its shear pin and moves axially in the housing, exposing the tines of the tang lock. The tines cam open, releasing the actuator rod and allowing the gas pressure to jettison the actuator rod with horizon scanners attached. Proper functioning of one cartridge is sufficient to jettison the horizon scanners.

Fresh Air Door Actuator

A fresh air door actuator ejects the door when initiated by the HI ALT DROGUE switch. The fresh air door actuator is located in the unpressurized area forward of the egress hatches, to the left of the spacecraft centerline and below the outer moldline. The actuator consists of breech, plunger, screw (shear pin), and two electrically fired gas pressure cartridges. The plunger forms a positive tie between the fresh air door and the breech, and is retained in the breech by the screw which acts as a shear pin. The breech provides for installation of the two cartridges and attachment to the spacecraft structure. The gas pressure produced by firing the cartridges exerts a force on the plunger, which severs the screw. The plunger and fresh air door then are jettisoned free of the spacecraft.

Docking System Pyrotechnic Devices

The pyrotechnic devices utilized for the docking mission of the spacecraft are located in the R & R section of the spacecraft as shown on Fig. 45 and consist of the following:

A. A pyrotechnically-actuated docking bar assembly extends and locks the indexing bar prior to the docking maneuver and jettisons the indexing bar after rendezvous operations are completed. The indexing bar aids both visually and mechanically in the docking maneuver.

B. Three pyrotechnically released latch receptacles mate with the mooring latch hooks on the docking cone of the target docking adapter (TDA). The latch receptacles normally are jettisoned during the retrograde sequence or they can be released from the spacecraft to provide an emergency demate capability in the event of a malfunction of the normal TDA release system.

C. Three pyrotechnically actuated cable cutters release the three latch covers for re-entry heat protection.

Docking Bar. - The docking bar has been designed and developed with two cycles (extension and jettison). The indexing bar is retained in the retracted position until cycled. The extension cycle moves the indexing bar 13.82 in. from the retracted position to the fully extended position and locks the indexing bar until jettison is required. The indexing bar, when in the extended position, can withstand a 1500 lb ultimate load applied normally to the outer end of the bar. The jettison cycle jettisons the indexing bar without recontact with the spacecraft.

The docking bar assembly consists of cylinder/housing, pistons, indexing bar, manifold assembly, one extension cartridge, and two jettison cartridges. The cylinder/housing is mounted to the spacecraft structure along the X axis. The manifold assembly, which is attached to the top of the cylinder housing and attaches to the spacecraft structure, contains a spring-loaded locking pin mechanism and two breeches. The extension cartridge is installed in the breech on the left side of the manifold and the two jettison cartridges are installed in the breech on the right. The indexing bar, with a hollow base, is secured in the outer piston by a shear pin. A drilled orifice in the upper portion of the outer piston mates with an orifice in the hollow base of the indexing bar. An inner piston is installed under the indexing bar, inside the outer piston. A hollow extension on the inner piston protrudes through the outer piston base. The indexing bar assembly is retained in the retracted position inside the housing by a retaining pin (shear pin) through one side of the manifold.

The indexing bar is extended by the gas pressure generated when the dual-bridgewire extension cartridge is initiated by the INDEX EXTENT switch. The gas pressure is ported into the cylinder/housing and enters the orifice in the base of the indexing bar. The gas pressure then is ported through the hollow inner piston extension and exerts a force on the bottom of the outer piston. When sufficient force is applied to the base of the outer piston, causing it to move, the retaining pin in the manifold/indexing bar is severed. The gas pressure causes the piston, with attached indexing bar, to move through the length of the cylinder housing. The engagement of the locking pin in the manifold with a recess in the outer piston stops the travel and secures the outer piston with the attached indexing bar in the fully extended position.

The docking bar is jettisoned by the gas pressure generated when the two jettison cartridges are initiated by the JETT RETRO switch. The gas pressure generated by firing the jettison cartridges is ported into the cylinder/housing and enters an orifice in the lower portion of the outer piston into a cavity between the inner and outer pistons. The thrusting action of the inner piston causes the indexing bar to sever the shear pin (indexing bar/outer piston) and jettison both the inner piston and the indexing bar. Initiation of one jettison cartridge is sufficient to complete the jettison cycle. The jettison cartridges have a 2.0 sec pyro delay to assure that during an abort mode the extension cartridge will fire first to extend the bar before the jettison cycle is initiated.

<u>Emergency Docking Release System.</u> - The emergency docking release system consists of three pyrotechnically released docking latch receptacles, which release the spacecraft from the docking vehicle if the normal release system fails. Each release assembly consists of a body, latch receptacle, piston/stud assembly, shear pin, closure end cap, and two gas pressure cartridges. The body houses the piston/stud assembly and provides for the installation of the two cartridges and attachment of the end cap. Flanges for attaching the body to the spacecraft structure are an integral part of the body. The piston/stud assembly is installed with the stud protruding through the body. The end cap seals the area above the piston and provides a mechanical stop. The latch receptacle is secured to the structure by a holding shear pin through the

protruding piston stud. The latch is preloaded on assembly by torquing the stud in the piston. The design of the latch, stud, and shear pin arrangement allows the unit to withstand a 5770 lb ultimate docking load on the latch without failure. The gas pressure produced by firing the two cartridges is ported to the underside of the piston. The pressure displaces the piston, separating the shear pin from the latch receptacle and moving the piston stud into the body to clear the latch receptacle. The latch receptacle being held by the docking vehicle latch is then parted from the release body, allowing the spacecraft freedom to move out of the docked condition.

The EMERG REL switch simultaneously initiates the dual cartridges in each device. The JETT RETRO switch also will initiate the release system if not fired previously. Function of one cartridge in a device provides latch release. Function of two of the three releases will achieve separation.

Latch Cover Cable Cutter System. - One latch cover is located at each emergency docking release to cover the opening created when the emergency docking release latch is released and to prevent excess thermal energy from entering the spacecraft. Each system consists of a spring-loaded latch cover, a restraining cable and a dual cartridge guillotine. Firing either or both cartridges actuates the guillotine that severs the restraining cable. The spring-loaded cover then slides forward to cover the docking latch cavity. All cartridges are initiated simultaneously by the JETT RETRO switch. The cartridges contain a 1.0 sec pyrotechnic time delay to insure emergency release prior to latch cover release.

Pyrotechnic Valves

Pyrotechnic valves in the orbit attitude and maneuvering system (OAMS) and re-entry control system (RCS) are one-time actuating devices which control fluid flow. Types of valves utilized are either normally open or normally closed. A package valves are normally open, C and D package valves are normally closed, and E package valves include one of each type.

The pyrotechnic valve consists of a valve body, nipple, ram, and a dual-bridgewire electrically fired high explosive cartridge/detonator. When the proper electrical signal initiates the cartridge/detonator, the resulting gas pressure and shock wave actuates the ram and either opens or closes the fluid line depending on the type of valve.

Fuel Cell Hydrogen Tank Vent Actuator

The fuel cell hydrogen tank vent actuator opens the hydrogen tank pinch-off tube, allowing gas molecules in the void between the inner and outer tank walls to escape when the spacecraft is in orbit. This increases system efficiency by reducing heat transfer. The actuator assembly consists of a body, piston/cutter blade, shear pin, breech ballistic hose, and a dual-bridgewire electrically fired gas pressure cartridge. The body houses the piston/cutter blade retained by a shear pin and provides for attachment of a blade guard. The actuator body assembly is bonded and strapped in place on the fuel cell hydrogen tank and connected to the breech by a ballistic hose. The breech provides for installation of the cartridge and is positioned approximately nine in. from the actuator to protect the cartridge from the extreme cold of the H_2 tank. The gas pressure produced by firing the cartridge is transferred through the ballistic hose, causing the piston/cutter blade to sever the shear pin and to move forward with a high velocity to split open the vent tube. As the piston/cutter blade reaches the end of the body, a shoulder stops the piston travel. The blade guard attached to the actuator body provides a trap for fragments.

Pyrotechnic Devices For Experiments

The experiments utilizing pyrotechnic devices (cable-cutting guillotines) to sever restraining cables and allow the release of spring-loaded doors or equipment include D-4, -7, -10, -12, -15, -16, MSC-1, -2, -3, -6, and S-9. A large wire bundle guillotine with a dual-bridgewire cartridge was used to sever a holding bolt on the extravehicular support package and a holding bolt and two tubes on the astronaut maneuvering unit.

The rendezvous evaluation pod was jettisoned from the adapter of Spacecraft 5 by a pyrotechnic ejector employing two electrically fired gas pressure cartridges. This ejector was similar to the horizon scanner release assembly in design and operation, except for output charge of the cartridge.

Pyrotechnic Development And Qualification Tests

Numerous breadboard development tests were conducted on each pyrotechnic device at component and system levels to obtain the correct operating time, pressure, and mechanical function.

Qualification tests of devices shown on Table 27 were made while under the environmental condition, or after exposure to the condition.

Z 13 Separation Assembly Tests. - The Z 13 separation assembly was environmentally tested utilizing test sections or strips. The test units consisted of a section of simulated spacecraft structure approximately 24 in. long with a section of FLSC assembly attached as in the production assembly. One single and one dual-bridgewire detonator were installed on each section. The units were fired by applying four amps or 30 VDC to one or both detonators according to the test schedule. The nine vendor system or performance tests were full scale separation assemblies mounted on boiler plate structure. These units were fired with 60 linear in. of the shaped charge at 288°F. All tests were successful. The six McDonnell system tests were conducted on boiler plate structure utilizing production pyroswitches, guillotines and wire bundles. The system was sequenced the same as production systems. All tests were successful.

TABLE 27 PYROTECHNIC QUALIFICATION TEST SUMMARY

DEVICE	VENDOR	MCDONNELL SYSTEM TEST	VEND. SYS. TESTS	HI-TEMP/ALT.	LO-TEMP/ALT.	HI-TEMP STORAGE	SALT SPRAY	SHOCK (15G.)	HUMIDITY	VIBRATION	ACCELERATION	AUTOIGNITION	PROOF/BURST PRESS.	40-FT. DROP	NO FIRE (1) AMP	NO FIRE (1) WATT	BRUCETON NO FIRE (AMP)	BRUCETON NO FIRE (WATT)	BRUCETON ALL FIRE	RELIABILITY ASSURANCE TEST			
																				100%	80%	60%	40%
Z13 SEP. ASSY.	ET	6	9	20	10	10	5	10	15	15	5	5	–	6	25	25	50	50	50	SEE NOTES			
Z69 SEP. ASSY.	ET	6	9	15	5	5	3	3	10	10	2	2	–	–	25	25	25	25	25	SEE NOTES			
Z100 SEP. ASSY.	OA/CTI	6	–	3	3	3	3	3	3	*	3	1	–	–	–	–	–	–	–	20	3	3	3
DETONATOR	CTI	–	–	3	3	3	3	3	3	*	3	–	–	–	–	–	–	–	–	–	–	–	–
HORIZON SCANNER FAIRING REL. ASSY.	OEA/ HOLEX	6	–	20	10	10	5	20	20	*	5	5	2	6	25	25	50	50	50	10	3	3	3
HORIZON SCANNER REL. ASSY.	TALLEY	–	–	20	10	10	5	20	20	*	–	5	2	6	25	25	50	50	50	10	3	3	3
RENDEZVOUS EVAL. POD REL. ASSY	TALLEY	2	–	6	6	–	–	–	2	*	5	–	–	–	6	6	6	6	6	10	–	–	–
FRESH AIR DOOR	OEA/ HOLEX	–	–	3	3	3	3	3	3	*	3	3	2	6	6	6	6	6	6	10	3	3	3
NOSE FAIRING REL. ASSY.	CTI	–	26	–	–	3	–	3	3	*	3	–	2	–	–	–	–	–	–	10	3	–	–
PYROSWITCH (41 PINS)	OA/ MCDONNELL	42	–	3	3	3	3	3	3	*	3	–	2	–	–	–	–	–	–	–	–	–	–
PYROSWITCH (55 PINS)	OA/ MCDONNELL	4	7	3	3	3	3	3	3	*	3	–	2	–	–	–	–	–	–	–	–	–	–
PYROSWITCH CARTRIDGE INSTANTANEOUS	CTI	7	–	3	3	3	3	3	3	*	3	–	–	6	–	6	–	–	–	12	3	3	3
PYROSWITCH CARTRIDGE 80 MS TIME DELAY	OA/ MCDONNELL	–	–	3	3	3	3	3	3	*	3	–	–	6	–	6	–	–	–	12	3	3	3
LARGE WIRE BUNDLE (LWB) GUILLOTINE	OA/ MCDONNELL	12	–	3	3	3	3	3	3	*	3	–	2	–	–	–	–	–	–	–	–	–	–
(LWB) CARTRIDGE (INST.)	CTI	–	–	3	3	3	3	3	3	*	3	–	–	6	25	25	50	50	50	19	3	3	3
(LWB) CARTRIDGE (120 MS TIME DELAY)	CTI	–	–	3	3	3	3	3	3	*	3	–	–	6	25	25	50	50	50	19	3	3	3
SMALL WIRE BUNDLE (SWB) GUILLOTINE	OA/ MCDONNELL	36	3	3	3	3	3	3	3	*	3	–	2	–	–	–	–	–	–	–	–	–	–
(SWB) CARTRIDGE (INST.)	CTI	–	–	3	3	3	3	3	3	*	3	–	–	6	–	6	–	–	–	19	3	3	3
(SWB) CARTRIDGE (120 MS TIME DELAY)	CTI	–	–	3	3	3	3	3	3	*	3	–	–	6	–	6	–	–	–	19	3	3	3
1/8 DIA. CABLE CUTTER GUILL. (EXPERIMENTS) (WITH INST. CART.)	OA/CTI	–	–	3	3	3	3	3	3	*	3	–	2	6	–	6	–	–	–	23	3	3	3
1/16 DIA. CABLE CUTTER GUILL. (LATCH COVERS) (2.0 SEC. TIME DELAY CART.)	MCDONNELL/ CTI	18	–	3	3	3	3	3	3	*	3	3	2	6	–	6	–	–	–	23	3	3	3
EMER. DOCK. REL. (1/8 DIA. CABLE CUTTER GUILL. CART.)	MCDONNELL	18	–	3	3	3	3	3	3	*	3	–	2	–	–	–	–	–	–	10	3	3	3
DOCKING BAR	MCDONNELL	3	–	3	3	3	3	3	3	*	3	–	2	–	–	–	–	–	–	10	3	3	3
EXTEND CART.	CTI	3	–	3	3	3	3	3	3	*	3	3	–	6	–	6	–	–	–	10	3	3	3
JETTISON CART.	CTI	6	–	6	6	6	6	6	6	*	6	3	–	6	–	6	–	–	–	10	3	3	3
																				100%	95%	90%	85%
TUBING CUTTER/SEALER	MCDONNELL /CTI	12	7	3	3	3	3	3	3	*	3	3	2	6	–	6	–	–	–	22	3	3	3

*UNITS WERE FIRST HUMIDITY – TESTED, THEN VIBRATION – TESTED.

The following reliability assurance tests were conducted to demonstrate (1) a detonator output of 1.3 times that required, and (2) that detonators will initiate the boosters which in turn initiate the FLSC with a gap of 1.5 times maximum blueprint gap.

A. Fifty detonators were divided into five groups of ten each. Each group was loaded with a reduced output charge and fired into production FLSC at a maximum blueprint gap of 0.015 in.

GROUP	PERCENT NORMAL OUTPUT	FLSC INITIATED
1	90	10 out of 10
2	70	10 out of 10
3	50	7 out of 10
4	30	6 out of 10
5	10	None

B. Thirty production detonators were divided into six groups of five each. Each group was fired into a detonator block and FLSC assembly with identical gap adjustments between the detonator to crossover and crossover booster to FLSC as shown.

GROUP	TEST GAP (INCHES)	MAX. BLUEPRINT GAP (INCHES)	FACTOR	FLSC INITIATED
1	0.023	0.015	1.5	5 out of 5
2	0.030	0.015	2.0	5 out of 5
3	0.038	0.015	2.5	5 out of 5
4	0.045	0.015	3.0	5 out of 5
5	0.053	0.015	3.5	5 out of 5
6	0.060	0.015	4.0	5 out of 5

Thirty-five firings were made with shaped booster caps and FLSC. All the firings were successful. The tests were divided into two groups as follows: Fifteen were made with booster against FLSC with a lead sheath of 0.025 in. thick and ambient conditions. Twenty were made with an 0.050 in. straight-line gap between the booster and FLSC with a production lead sheath of 0.010 in. thick and fired while at -54°F.

Z 69 Separation Assembly Tests. - The Z 69 separation assembly was environmentally tested using test sections as was the Z 13 separation assembly. The Z 69 test unit also included the tube cutter, detonator, and interconnect. Simulated tubes filled with liquid at production pressures were installed in the tube cutter.

Nine vendor performance tests were similar to the Z 13 except with 60 linear inches at 270°F and 60 linear in. on the opposite side at -90°F. All tests were successful.

Six successful McDonnell system tests were conducted similar to the Z 13 separation assembly tests. In addition, one assembly was placed in an altitude chamber for 14 days at a pressure level of 800,000 ft before firing (static 3).

Z 100 Separation Assembly Tests. - The Z 100 separation assembly was environmentally tested at McDonnell, to accelerate the test program. Each test assembly consisted of three cutters, six mild detonating fuse (MDF) interconnects, three unions and three detonators. The detonators were manufactured by Central Technology Inc. and the remainder of the assembly was manufactured by Ordnance Associates, Inc. The assemblies were mounted on simulated back-up structure for firing and contained the tubes, straps, and wire bundles to be used as production assemblies. The tubes were filled with representative liquid and capped. All tubes, titanium strap, and wire bundles in each cutter were cut in two planes. The detonators, which are the same type as those used for the R & R separation assembly, also were environmentally tested at CTI. All tests were successful. The following reliability assurance tests were conducted to demonstrate a detonator output of 1.3 times that required by reducing the output load. Tests also were conducted to demonstrate propagation with gaps of 1.5 times maximum blueprint gaps for the components of the cutter. The gaps are as follows:

GAP	LOCATION	100%	150%
A	Detonator/Interconnect Booster	0.040	0.060
B	Interconnect Booster/ Interconnect	0.032	0.051
C	Interconnect Booster/ Parallel Booster	0.014	0.021
D	Interconnect Booster/ FLSC	0.020	0.030
E	Parallel Main Column/ Crossover	0.078	0.113
F	Crossover/Output Can	0.027	0.041
G	Output Can/FLSC	0.088	0.132

A. Detonator Output Tests (Five Groups)

1. Ten detonators with 100% output charge and 150% gaps A-B-C (all fired).
2. Ten detonators with 100% output charge and 100% gaps A-B-C (all fired).
3. Three detonators with 80% output charge and 100% gaps A-B-C (all fired).
4. Three detonators with 60% output charge and 100% gaps A-B-C (all fired).

5. Three detonators with 40% output charge and 100% gaps A-B-C (all fired).

B. Gap Tests (Three Groups)

1. Five detonators with 100% output and 150% gaps E-F-G (all fired).
2. Five interconnect/interconnect booster (B) 150% gap (all fired).
3. Five interconnect booster/FLSC (D) 150% gap (all fired).

C. Six McDonnell system tests were conducted on boiler plate structures as Z 69 assemblies and sequentially fired as a system. All tests were successful.

Emergency Docking Release Tests. - The emergency docking release (EDR) unit was manufactured by McDonnell and environmentally qualification-tested with its cartridges at CTI. The cartridges were manufactured by CTI and are the same as those utilized with the 1/8 in. diameter cable cutter guillotine. After subjecting the EDR assembly to various environments, the devices were installed on a load test fixture and fired to verify proper operation. Ten assemblies were fired after a limit load of 4250 lb was applied to the latch and then released, and five were fired as this limit load was being applied. Two units without cartridges were static-tested by applying a load in excess of the 5770 lb ultimate load until the shear pins failed. The latch receptacles from five assemblies without cartridges were torqued to 100 in. lb and held for 200 hr. The torque was verified once daily to insure that the receptacles had not loosened. Two of these assemblies then were mounted on the load test fixture and subjected to more than 1500 cycles of loads from zero to 4250 lb to zero on the latch. All tests were successful.

The normal reliability assurance tests were conducted utilizing a single cartridge with reduced output charge to actuate the device. All the cartridges with 67% or more output charge actuated the device. The 40% output charge cartridges did not, as expected, actuate the device.

Nine reaction load tests were conducted on assemblies utilizing two cartridges with the output charge uploaded as follows:

A. Three firings with 100% output charge.
B. Three firings with 115% output charge.
C. Three firings with 130% output charge.

The assembly was proof-tested by utilizing two cartridges with an output of 120%. The assembly was also burst-tested by utilizing two 150% normal output cartridges to insure that no yielding or deformation of the pressure chamber occurred. All tests were successful.

In addition to the qualification firings, McDonnell conducted six full-scale firings on an R & R section as a system test with three EDR's and three latch cover systems on each firing. All tests were successful.

Docking Bar Assembly Tests. - The docking bar manufactured by McDonnell was environmentally tested by CTI with its CTI-designed and manufactured cartridges. Four units were fired simultaneously so that the extension and jettison cartridges were pulsed simultaneously to insure proper sequence operation.

Twenty units were sequentially fired so that the extension cartridge was fired first to extend the indexing bar, then the jettison cartridges were fired. Twelve of the sequentially-fired units had a 1100 lb limit load applied at the extreme end of the extended indexing bar, normal to the centerline; this was then released and the jettison cartridge fired.

Two units without cartridges were static-tested by extending the bar, one unit was tested by applying a load 700 lb in excess of the 1500 lb ultimate load normal to the bar until failure occurred. The other unit was tested by applying a load 1070 lb in excess of the 1500 lb ultimate load at 15 degrees above normal until failure occurred. Two units without cartridges were static-tested to insure that the indexing bar would be retained during the ascent forces. A calculated force of 31.2 lb was applied without any failure. The load force then was increased until failure occurred verifying the 1.5 margin of safety.

Three system tests for the assembly included a dynamic force test conducted by the vendor. The docking bar assembly was mounted on a production support structure of the R & R section and test-fired as follows:

A. One assembly stabilized at 200°F and held for a period of four hr before firing.

B. One assembly at room ambient conditions.

C. One assembly stabilized at -65°F and held for a period of nine hr before firing.

Nineteen reliability assurance tests were conducted on the assembly with both the extension and jettison cartridge containing reduced output charges and sequentially fired.

A. Five assemblies with 100% output and 100% shear pin - all fired.

B. Three assemblies with 80% output and 100% shear pin - all fired.

C. Three assemblies with 60% output and 100% shear pin - all fired.

D. Three assemblies with 40% output and 100% shear pin - did not function.

E. Five assemblies with 100% output and 80% jettison shear pins - all fired (shear pin held until jettison cycle).

One assembly with the indexing bar in the retracted position was hydrostatically proof (120%) and burst (150%) tested.

McDonnell conducted full-scale shock tests per TR 052-044.09 in which the docking bar and emergency release assemblies (all without cartridges) were installed on a production R & R section. The indexing bar was manually extended, and the R & R section assembly then was impacted approximately 20 times with the docking cone simulating various docking conditions.

McDonnell conducted a docking bar, Spacecraft 11 configuration demonstration test per TR 052-069.56 in which the indexing bar was manually extended and a docking bar mirror, tether clamp and tether weighted to three lb were attached. Two jettison cartridges were installed and initiated electrically. The indexing bar was jettisoned and the weighted tether separated from the indexing bar.

McDonnell conducted development and demonstration tests per TR's 052-069.57, 052-069.57.01, 052-069.57.02 and 052-069.57.03 to determine under what applied side load the indexing bar could be jettisoned with single cartridge firing. A side load of 100 lb was determined to be acceptable for mission plans which was successfully demonstrated with single and dual cartridge firings.

Mission Anomalies

The anomalies associated with the pyrotechnic system were electrical circuitry problems, whereby electrical energy was not delivered to the redundant cartridge.

A. Spacecraft 3 - Fresh Air Door - The device functioned but one cartridge did not fire.

B. Spacecraft 6 - Docking System - The system functioned but one jettison cartridge (docking bar) and one cartridge in each of the emergency docking release assemblies and cover release guillotines did not fire.

Pyrotechnic Qualification Test Summary - (See Table 27)

Pytotechnic Failure Summary - (See Table 28)

Mission Evaluation

The pyrotechnic systems have functioned successfully on all missions.

MISSION PLANNING

INTRODUCTION

The major objective of the Gemini mission planning was to maximize, within reasonable limits, the probability of mission success. To accomplish this objective, two basic phases of planning were undertaken.

Early in the program, mission and design objectives were developed so that the systems configuration requirements could be determined for both the Gemini Spacecraft and the modifications to the target and launch vehicles. Following this, design reference missions were established which permitted the detailed specification of the hardware requirements.

The second phase was the development of mission plans for each flight including both onboard operations and the ground tracking and computation operations. This phase established a logical progression of missions and maximized the probability of success of each mission.

In addition to preflight planning, some mission planning was conducted during flights in response to nonnominal situations.

SOFTWARE

During the Gemini program, nine different configurations of the onboard computer program were defined and, with only three exceptions, were carried through acceptance test selloff. The large number of configurations was a result of additional mission requirements that were defined as the Gemini program progressed. Subsequent paragraphs will describe the evolution of the different versions of the operational programs, the major differences between them, the analysis effort involved in defining the various computer modes, and the various levels of testing which were used to verify the computer programs.

Description Of Math Flow

The Gemini Spacecraft digital computer (SDC) program originally was designed to accomplish the mission objectives defined early in the program. On later flights, the SDC program was revised to accommodate new mission objectives within the limited computer memory capacity of 12,288 13-bit instructions. The computer program also was changed to improve guidance technique and accuracy. Consequently, the SDC program evolved from the relatively simple first program, titled "Gemini Math Flow" which consisted of four operational modes, to the complex Math Flow Seven, which utilized the auxiliary tape memory (ATM) and consisted of six program modules containing nine operational modes.

The first math flow contained the ascent, catchup, rendezvous, and re-entry modes and was used in acceptance testing of engineering model computers. The second math flow evolved from the desire to provide orbital navigation and re-entry mode initialization capabilities. The computer memory capacity was insufficient for this math flow and, consequently, Math Flow Two was short-lived.

Math Flow Three, which was used in Spacecraft 2, was similar to the second math flow except for the deletion of orbital navigation and re-entry, mode initialization capabilities. Math Flow Four contained the touchdown

TABLE 28 PYROTECHNIC FAILURE SUMMARY

TEST	PROBLEM	CORRECTIVE ACTION
(1) GENERAL	PRIME VENDOR HAD DIFFICULTIES WITH THE MANUFACTURING AND QUAL TESTING OF DETONATORS, CARTRIDGES, GUILLOTINES, PYROSWITCHES, AND TUBING CUTTER/SEALER.	SPACECRAFT 2 UTILIZED THE PRIME VENDOR'S (OA) CARTRIDGES AND DETONATORS WITH DUAL BRIDGEWIRES CIRCUITS IN ALL UNITS. ABBREVIATED QUAL TESTS WERE CONDUCTED ON THESE UNITS WHICH CONSISTED OF HI-TEMP/ALT, HUMIDITY, VIBRATION, AUTO-IGNITION, PROOF AND BURST PRESSURE 40-FT DROP, ONE AMP NO-FIRE, ONE WATT NO-FIRE. THE UNITS WERE ACCEPTANCE TESTED AS FOLLOWS: BRIDGEWIRE RESISTANCE, LEAKAGE, VIBRATION, 200°F TEMP SOAK ½ HOUR, RESISTANCE AND DIELECTRIC. SPACECRAFT 3 AND UP – MCDONNELL ASSUMED THE DESIGN AND MANUFACTURING OF THE GUILLOTINES, TUBING CUTTER/SEALER AND PYROSWITCHES WHICH WERE NOT AVAILABLE. MCDONNELL ENGAGED SECOND VENDOR (DTI) FOR DETONATORS AND CARTRIDGES WITH THEIR ASSOCIATED DEVICED WERE QUALIFIED.
(2) QUAL (Z13)	DETONATOR INSTALLATION PUSHED FLSC AWAY FROM INITIATING BOOSTER. VARIATION OF STAND-OFF BETWEEN BOOSTER/FLSC TOO GREAT.	MADE DETONATOR BLOCK A PART OF THE SHAPED CHARGE ASSY AND SHAPED BOOSTER TO FIT FLSC SO THAT BOOSTERS WERE IN CONTACT WITH FLSC.
(3) (Z69)	DIPAM SHAPED CHARGE BLISTERED.	CHANGED FLSC FROM DIPAM TO RDX. SPACECRAFT 2 UTILIZED DIPAM FLSC.
(4) (Z69)	TUBE CUTTER FAILED TO INITIATE.	REDESIGNED TUBE CUTTER, ADDED A DETONATOR AND REMOVED ONE EXPLOSIVE INTERCONNECT. SPACECRAFT 2 UTILIZED ASSY WITH TWO INTERCONNECTS.
(5) QUAL (Z69)	TUBE CUTTER INTERCONNECT FAILED TO PROPAGATE.	INCREASED ACCEPTER CHARGE IN EXPLOSIVE INTERCONNECT (40 TO 55 mg).
(6) QUAL (Z69)	ONE DETONATOR BLOCK BLEW FLSC FROM UNDER OTHER DET BLOCK AT DOUBLE DET BLOCK AREA.	ADDED SOLID BLAST SHIELD BETWEEN DETONATOR BLOCKS AND ADDED TWO STIFFENERS AT END OF SHAPED CHARGE.
(7) SPACECRAFT 8 VERIFICATION	THE SINGLE BRIDGEWIRE DETONATOR FAILED TO INITIATE FLSC BOOSTER OF TEST STRIP.	A NEW LOT OF DETONATORS WERE FABRICATED INCORPORATING "TARE" WEIGHED CHARGE BUILD-UP.
(8) MCDONNELL TEST (VIB) RENDEZVOUS EVAL POD REL ASSY	THE ASSEMBLY FAILED TO WITHSTAND AN AXIAL LOAD OF 1500 LB FOR FIVE MINUTES DURING RANDOM SPECTRUM.	THE TANG LOCK RETAINER WAS REPOSITIONED TO COVER MORE OF THE TANGS, PREVENTING THEM FROM CAMMING OUT FROM UNDER THE RETAINER PREMATURELY.

TABLE 28 PYROTECHNIC FAILURE SUMMARY (Continued)

TEST	PROBLEM	CORRECTIVE ACTION
(9) CART QUAL HOR SCANNER REL ASSY	THE ORIGINAL CARTRIDGE FAILED TO MEET THE ONE AMP/WATT NO-FIRE REQUIREMENT.	VENDOR FABRICATED NEW CARTRIDGES (REMOVED EPOXY SEAL) AND REDESIGNED TIME DELAY COLUMN TO PREVENT OUTAGE.
(10) QUAL – (VIB) HORIZON SCANNER RELEASE ASSY	ASSY WITH ALUMINUM PISTON FAILED SINUSOIDAL VIBRATION.	REDESIGNED ASSEMBLY WHICH INCORPORATED A STAINLESS STEEL PISTON. THIS ASSEMBLY PASSED RANDOM VIBRATION TEST. STEEL PISTON ASSY INCORPORATED SPACECRAFT 4 AND UP.
(11) MCDONNELL TEST HORIZON SCANNER	THE CHAMBER PLUG BLEW OUT DURING AMBIENT TEST.	CHAMBER PLUG AND ADJOINING AREA WAS REDESIGNED TO INCREASE THREAD STRENGTH.
(12) SPACECRAFT 4 VERIFICATION (HSFR)	AHSFR FAILED TO FUNCTION. ANALYSIS INDICATED THAT THE LOCKING PINS HAD INTERFERED WITH THE ID OF CYLINDER ENERGY.	TO INSURE ZERO INTERFERENCE IN THIS AREA A 100% PIN TO PIN PRE-ASSEMBLY ACCEPTANCE INSPECTION WAS ADDED.
(13) CARTRIDGE	THE CART WAS SUSPECTED OF YIELDING LOW ENERGY.	A NEW LOT OF CARTRIDGES WAS FABRICATED INCORPORATING "TARE" WEIGHED OUTPUT CHARGE.
(14) QUAL EMERGENCY DOCKING RELEASE	PISTON FRACTURED ALLOWING IT TO DEFORM THE CLOSURE CAP. HOWEVER, AT ALL TIMES THE DEVICE FUNCTIONED.	THE PISTON WAS REDESIGNED TO INCORPORATE A LARGER FILLET RADIUS. THE NEW PISTON WAS TESTED 18 TIMES WITHOUT FAILURE.
(15) DEVELOPMENT DOCKING BAR EXTEN CYCLE	ONE CARTRIDGE HAD TO EXTEND THE BAR WHILE IN A TEMP RANGE OF –65°F TO +250°F. WHEN TWO CARTRIDGES WERE USED IN THE DEVICE FOR REDUNDANCY, THE PRESSURE GENERATED WOULD SHEAR THE PIN AND JETTISON THE BAR.	A SINGLE CARTRIDGE WITH DUAL BRIDGEWIRES AND A SLOW BURNING PROPELLANT WAS DEVELOPED AND QUAL TESTED WITH THE DEVICE FOR THE EXTENSION CYCLE.
(16) QUAL (HI-TEMP/ALT) DOCKING BAR	THE ELECTRICAL PORTION OF THE JETTISON CARTRIDGE WAS EXPELLED FROM THE CARTRIDGE. CARTRIDGE WALL RUPTURED AT THREAD RELIEF.	A NEW LOT OF CARTRIDGES WAS FABRICATED WITH HEAVIER/STRONGER BODIES. (INTERNAL DIA AT THREAD RELIEF WAS REDUCED FOR FOR THICKER WALL).
(17) QUAL (VIB) DOCKING	BREECH CRACKED ON SINUSOIDAL VIBRATION	BREECH REDESIGNED WITH LARGER RADIUS AND MATERIAL CHANGED TO STAINLESS STEEL. THIS ASSY PASSED RANDOM VIBRATION TEST PER THE SCD.
(18) SPACECRAFT 11 VERIFICATION Z13 DETONATOR	ONE DETONATOR FAILED TO INITIATE THE FLSC BOOSTER OR THE Z13 TEST STRIP.	A NEW LOT OF DETONATORS WAS FABRICATED, INCORPORATING "TARE" WEIGHED OUTPUT CHARGE.

predict mode which provided re-entry mode initialization nearly independent of the ground. This program utilized 12,150 of the 12,288 available instructions and was at that time planned to be flown on Spacecraft 3 and up. A February 1964 NASA directive changed the guidance logic of the re-entry mode to a constant bank angle logic rather than the proportional bank angle and constant roll rate logic common to all previous math flows. This, and other changes, became Math Flow Five, and Math Flow Four was terminated. Math Flow Five saturated the computer memory and was cancelled at a McDonnell/NASA management meeting approximately three days after being defined.

At the same meeting, it was decided to fly slightly modified versions of Math Flow Three on Spacecraft 3 and 4 and to define a new program, Math Flow Six, effective Spacecraft 5 and up. The modified versions of Math Flow Three which were flown on Spacecraft 3 and 4, were titled Math Flow Three Mod I and Mod II, respectively.

The addition of the ATM to the IGS, effective Spacecraft 8 and up, provided the means for increasing the SDC program's capabilities. The resulting modularized program was called Math Flow Seven.

Table 29 provides a brief general description of the Gemini SDC math flows. The descriptions are given in the form of major changes and/or additions to the previous edition.

Analysis

Simulations were utilized extensively in defining and evaluating the different computer modes of the Gemini operational programs. These simulations generally fall into the following categories:

A. Early Simulations - used to evaluate equations and served as a basis for configuring a given computer mode.
B. Man-in-the-loop Simulation - to define input/output requirements, procedures, and display format.
C. Refined Digital Simulations - to determine performance characteristics and to perform error analysis.

Fig. 46 portrays the recommended procedure for establishing a specification control document (SCD), which was the basis for the development of an operational computer mode. Of the ten computer modes ultimately developed, only four (ascent, catchup, rendezvous, and re-entry) went through this type of development. The analysis of the remainder of the computer modes was accomplished concurrently with their programming by IBM for the onboard computer. A summary of McDonnell simulation programs used in the development, definition, and evaluation of the various computer modes is presented in Table 30.

Testing

Verification of the equations and determination that accuracy requirements could be met was initiated at IBM Space Guidance Center, Owego, New York immediately upon receipt of the McDonnell SCD. The results of FORTRAN simulation of the operational mode were compared to a simulation of the exact equations to verify the validity and accuracy of the equations. The basic testing of the coded program was conducted at IBM in the configuration control test system (CCTS) laboratory. Flight hardware used in CCTS checkout consisted of the computer, manual data insertion unit (MDIU), and incremental velocity indicators (IVI). Inputs from other systems, e.g., radar and accelerometer, were provided by simulators.

The formal acceptance testing was performed when the operational program was considered to be ready for flight. This testing was conducted using hardware and simulators similar to those in the CCTS lab. Post-acceptance testing of the operational program was conducted at IBM, McDonnell, St. Louis and Cape Kennedy.

The purpose of the mission verification simulation (MVS) was to verify that the IGS, loaded with the operational program and operating with dynamic inputs, performed in an acceptable and predictable manner. Actual flight-type hardware used in MVS testing consisted of the computer, IVI, MDIU, time reference system, and attitude display group. Other inputs were supplied by a 7090 computer through hardware simulators.

The post-acceptance testing included spacecraft systems tests (SST) at St. Louis and launch pad tests at Cape Kennedy, and were essentially the same except that testing for booster guidance compatibility with that of the spacecraft was only conducted at the launch pad. The tests established the proper operation of the computer and the operational program in the total system configuration. These tests were similar to the acceptance tests at IBM during computer and operational program selloff.

The operational program errors which were detected during post-acceptance testing are summarized in Table 31.

DEVELOPMENT OF MISSION AND DESIGN OBJECTIVES

The rendezvous and re-entry requirements had a major effect on mission and design objectives. Early analyses established the need for the radar, inertial guidance system, orbital attitude and maneuvering system and digital command system. The inertial guidance system was required primarily for navigating through re-entry, and the radar and maneuvering system were needed for rendezvous. Additional studies evaluated the orbital requirements, particularly the desired target orbit altitude and inclination to provide satisfactory launch windows. A description of the launch window problem is given in the following paragraphs.

TABLE 29 DESCRIPTION OF MATH FLOW

MATH FLOW	S/C USE	START DATE	COMPLETE DATE	COMPUTER OPERATIONAL MODES			
				ASCENT	CATCH-UP	RENDEZVOUS	RE-ENTRY
1	NONE	3–62	4–63	BACK-UP GUIDANCE AND NAVIGATION DURING FIRST AND SECOND STAGE BOOSTER OPERATION; ORBIT INSERTION ADJUST ROUTINE (IVAR)	ACCEPTED GROUND GENERATED ΔV'S VIA DCS/MDIU; TRANSFORMED TO SPACECRAFT AXIS AND DISPLAYS ON IVI'S; MONITORED THRUSTING	ACCEPTED RADAR DATA TO SOLVE C–W EQUATIONS TO DETERMINE ΔV TO RENDEZVOUS IN 270° OF ORBIT TRAVEL; INTERMEDIATE CORRECTIONS AT 60° INTERVALS WITH THE LAST AT 30°	GUIDANCE AND NAVIGATION INITIATE AT 400k USING GROUND PREDICTED INITIAL CONDITIONS AT 400k WITH COMPENSATION FOR OFF NOMINAL RETRO ROCKET OPERATION; PROPORTIONAL BANK ANGLE WITH CONSTANT ROLL RATE LOGIC
2	NONE	2–63	6–63	ADDED ASCENT-ABORT RE-ENTRY	MINOR CHANGES	MINOR CHANGES	NAVIGATION INITIATED AT RETROFIRE USING GROUND OR ON-BOARD GENERATED INITIAL CONDITIONS; PROGRAMMED ATMOSPHERE APPROXIMATION TO 400k
3	2	6–63	5–64	MINOR CHANGES	MINOR CHANGES	MINOR CHANGES	MINOR CHANGES
4	NONE	8–63	6–64	MINOR CHANGES	IVI ZEROING LOOP BYPASSED AFTER 50 SEC	IVI ZEROING LOOP BYPASSED AFTER 50 SEC	MANUAL INITIATE VIA START COMP BUTTON; DCS LOCKOUT AT 128 SEC BEFORE RETROFIRE
5	NONE	6–64	6–64	IMPROVED IVAR	SAME AS MF4	SAME AS MF4	CONSTANT BANK ANGLE LOGIC WITH HALF-LIFT RANGE PREDICTOR
3 MOD I	3	6–64	11–64	SAME AS MF3 WITH IMPROVED SWITCHOVER FADE-IN EQUATION	SAME AS MF3	SAME AS MF3	SAME AS MF3
3 MOD II	4	8–64	12–64	IMPROVED IVAR	SAME AS MF3 MOD I	SAME AS MF3 MOD I	SAME AS MF4
6	5, 6, 7	6–64	6–65	FOUR-STEP PITCH PROGRAM; NON-ZERO FLIGHT PATH ANGLE AT INSERTION; IVAR CHANGES	SAME AS MF4 WITH ΔV'S ALSO ON MDIU; START COMP LIGHT ON CUE TO START THRUSTING; RADAR DATA VALIDITY VERIFIED; RADAR ANGLE DATA LOCKOUT	SAME AS MF4 WITH ΔV'S ALSO ON MDIU; 11-POINT POSITION TABLE; ωt VARIABLE VIA MDIU; DCS LOCKOUT OF $\Delta\dot{X}$ $\Delta\dot{Y}$, $\Delta\dot{Z}$, RADAR DATA VALIDITY VERIFIED RADAR ANGLE DATA LOCKOUT	SAME AS MF5; RE-ENTRY SELF-TEST (INITIAL CONDITIONS INSERTED); DCS LOCKOUT 120 SEC BEFORE RETROFIRE
7	8 → 12	7–65	5–66	GYRO DRIFT COMPENSATION; STATE VECTOR TRANSFER; ABORT RETROFIRE ATTITUDE COMMAND; ABORT – RE-ENTRY SELF-TEST; READOUT OF CENTRAL ANGLE TRAVEL AND OUT-OF-PLANE POSITION	RADAR DATA ON MDIU; RANGE RATE AND RANGE RATE ERROR ON MDIU; AFT OR FWD THRUSTER IVI OPTION; FDI INERTIAL, LOCAL VERTICAL, OR IVI REFERENCE; RADAR MISALIGNMENT COMPENSATION; ORBIT RATE TORQUING COMPENSATION; 10:1 IVI FWD/AFT CHANNEL ATTENUATION	ALL THE MF7 CATCH-UP CHANGES PLUS VARIABLE RADAR SAMPLING INTERVAL; RENDEZVOUS SELF-TEST	PROPORTIONAL BANK ANGLE, CONSTANT ROLL RATE LOGIC; ZERO LIFT RANGE PREDICTOR; RE-ENTRY SELF-TEST (INITIAL CONDITIONS PRE-PROGRAMMED AND INSERTABLE)
REFERENCES – GUIDANCE & CONTROL MECHANICS, GEMINI DESIGN NOTES:				41, 108, 200, 245, 325, 333, 350	30, 200, 325, 350, 377	2, 7, 25, 200, 325, 350, 377	32, 170, 193, 200, 325, 350

TABLE 29 DESCRIPTION OF MATH FLOW (Continued)

COMPUTER OPERATIONAL MODES					
ORBIT NAVIGATION	ORBIT PREDICTION	RETRO TIME PREDICT	RELATIVE MOTION	ORBIT DETERMINATION	TOUCHDOWN PREDICT
N/A	N/A	N/A	N/A	N/A	N/A
NAVIGATION THROUGH MANEUVERS UPDATING GROUND OR ON-BOARD GENERATED INITIAL CONDITIONS	PREDICTED SPACECRAFT EPHEMERIS USING GROUND OR ON-BOARD GENERATED INITIAL CONDITIONS AUTOMATIC DISPLAY OF PREDICTION TIME, LATITUDE, EARTH LONGITUDE AT FIXED INTERVALS OF PREDICTION TIME	INITIALIZED THE RE-ENTRY MODE BY DETERMINING TIME TO RETROFIRE TO LAND AT A SELECTED SITE; SITE SELECTION VIA MDIU; USED ON-BOARD OR GROUND-GENERATED INITIAL CONDITIONS	N/A	N/A	N/A
DELETED	DELETED	DELETED	N/A	N/A	N/A
N/A	N/A	N/A	N/A	N/A	INITIALIZED THE RE-ENTRY MODE BY DETERMINING TOUCHDOWN SITES FOR MDIU INSERTED TRIAL RETRO-FIRE TIMES; GROUND GENERATED INITIAL CONDITIONS
N/A	N/A	N/A	N/A	N/A	SAME AS MF4
N/A	N/A	N/A	N/A	N/A	DELETED
N/A	N/A	N/A	N/A	N/A	N/A
N/A	N/A	N/A	N/A	N/A	N/A
SAME AS MF2 WITH OPTIONS TO USE IVI WITH FWD/AFT THRUSTERS; 10:1 IVI FWD/AFT ATTENUATION	DETERMINED FUTURE AND PAST SPACECRAFT EPHEMERIS BASED UPON MDIU INSERTED SOLUTION TIME; OPTION TO DETERMINE TARGET VEHICLE OR SPACECRAFT RELATIVE TO TARGET VEHICLE; INITIAL CONDITIONS FROM GROUND OR ON-BOARD SOURCES; OPTION TO SIMULATE SPACECRAFT MANEUVERS IMPULSIVELY	N/A	DETERMINED THE TWO IMPULSIVE ΔV'S REQUIRED TO TRANSFER BASED UPON MDIU INSERTABLE INITIAL AND FINAL RELATIVE STATE VECTORS	IMPROVED THE ON-BOARD GENERATED STATE VECTOR BY PROCESSING STAR-HORIZON VERTICAL MEASUREMENTS	SAME AS MF4 WITH RETROGRADE ΔV'S AND L/D INSERTABLE VIA DCS; GROUND OR ON-BOARD GENERATED INITIAL CONDITIONS
130, 350	5, 40, 130, 350	56, 130, 350	350	4, 14, 292, 350, 363, 387	130, 148, 325, 350

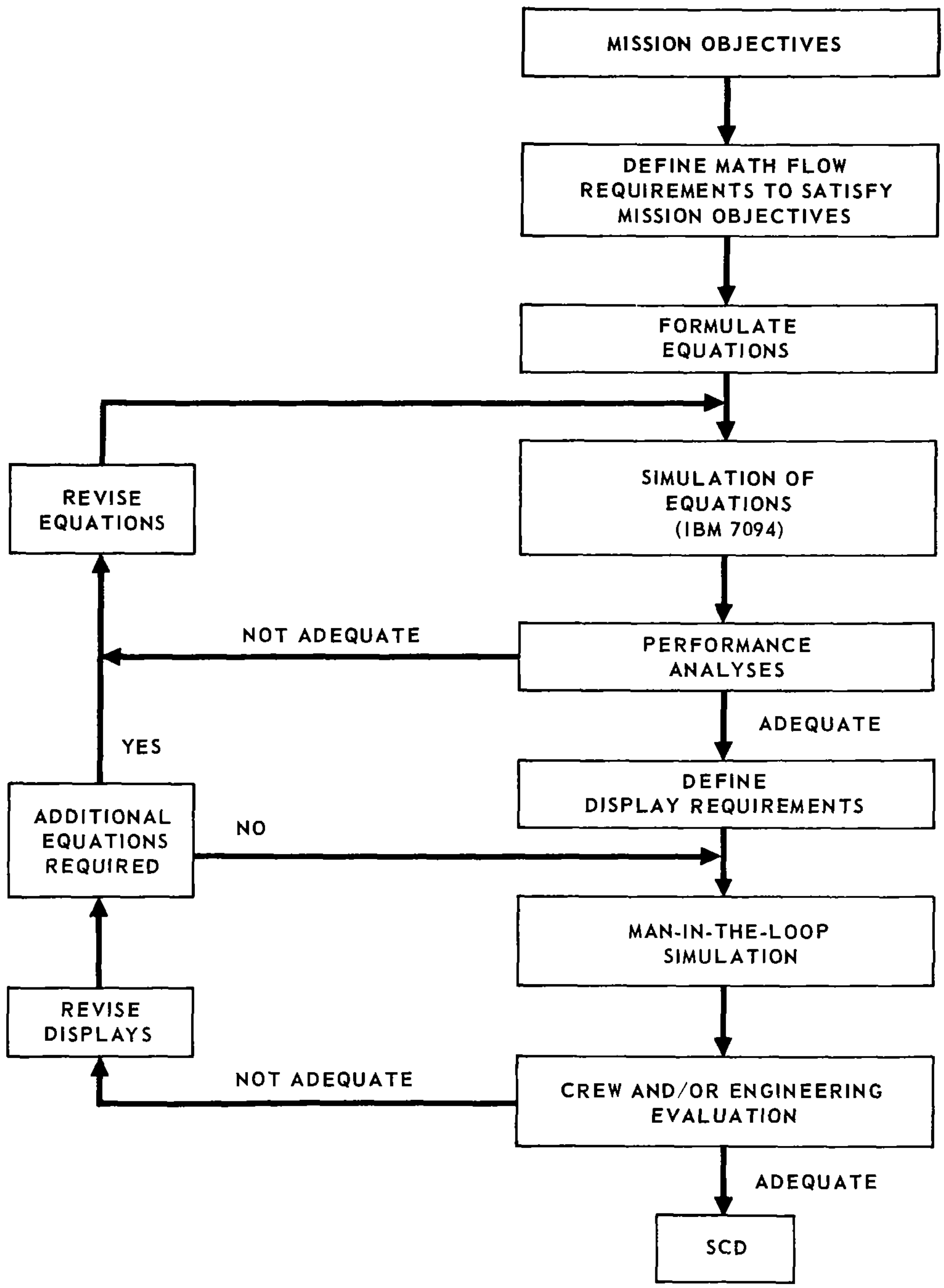


FIGURE 46 SOFTWARE SPECIFICATION CONTROL DRAWING (SCD) DEFINITION PROCESS

TABLE 30 MCDONNELL SIMULATIONS UTILIZED DURING COMPUTER MODE DEVELOPMENT

OPERATIONAL MODE	SIMULATIONS	A, ANALOG D, DIGITAL H, HYBRID	ERROR ANALYSIS	MAN-IN-THE-LOOP
ASCENT	1. RADIO GUIDANCE SYSTEM	D		
	2. COMBINED RADIO AND INERTIAL GUIDANCE SYSTEM	H		X
	3. COMBINED RADIO AND INERTIAL GUIDANCE SYSTEM	D	X	
	4. INERTIAL GUIDANCE SYSTEM	D	X	
	5. ORBITAL INSERTION VELOCITY ADJUST	D	X	
	6. ORBIT INSERTION VELOCITY ADJUST	A		X
	7. ASCENT POST FLIGHT	D		
CATCH-UP AND RENDEZVOUS	1. FORTRAN SIMULATION OF RENDEZVOUS MATH FLOW	D		
	2. CATCH-UP SIMULATION	D		
	3. CLOSED LOOP TERMINAL GUIDANCE	D	X	
	4. CLOSED LOOP RENDEZVOUS	H		X
	5. RENDEZVOUS GUIDANCE SIMULATION	D	X	
	6. CATCH-UP MODE	H		X
ORBIT NAVIGATION, ORBIT PREDICT, RETRO TIME AND TOUCHDOWN PREDICT	1. RETROGRADE TIME COMPUTATION	D		
	2. REAL TIME THRUSTING IN ORBIT	D		
	3. ORBIT EPHEMERIS COMPUTATION	D	X	
	4. NUMERICAL INTEGRATION ACCURACY	D	X	
	5. TOUCHDOWN PREDICT MODE	A		X
	6. TOUCHDOWN PREDICT MODE	H		X
	7. TOUCHDOWN PREDICT MATH FLOW SIMULATION	D	X	
	8. ORBIT PREDICT MODE	H		X
	9. ORBIT NAVIGATION MODE	H		X
RE-ENTRY	1. RE-ENTRY GUIDANCE AND NAVIGATION	D	X	
	2. SIX DEG. FREEDOM REAL TIME HYBRID	H		X
	3. PSUEDO SIX DOF MAN-IN-THE-LOOP	H		X
	4. RE-ENTRY HEATING STUDIES	D		
	5. ALTERNATE RE-ENTRY GUIDANCE	D		
	6. SIX DOF RE-ENTRY SIMULATION	D	X	
	7. RE-ENTRY MATH FLOW SIMULATION	D		
	8. RE-ENTRY WITH NAVIGATION FROM RETROFIRE	D	X	
ORBIT DETERMINATION	1. ORBIT DETERMINATION SIMULATION	D		
	2. AUTONOMOUS NAVIGATION	D		
	3. ORBIT DETERMINATION MATH FLOW	D	X	
	4. ORBIT DETERMINATION MODE	H		X

TABLE 31 PROGRAM ERRORS DETECTED DURING POST ACCEPTANCE TESTING

MATH FLOW	ERROR DETECTED	LEVEL OF TESTING
3	PLATFORM RELEASE TIMING ERROR ΔV_X OVERFLOW DURING SECO COUNTDOWN TELEMETRY TIME SCALING ERROR V_X OVERFLOW LIFTOFF TIME SYNCHRONIZATION ERROR ADDED Δt TO RE-ENTRY DATA ACQUISITION SYSTEM (DAS) AS A RESULT OF MVS TIMING DISCREPANCY IN GIMBAL ANGLE SUBROUTINE	MISSION VERIFICATION SIMULATION(MVS)(IBM) ↕ MVS SST (MCDONNELL-ST. LOUIS)
3 MODE II	ELAPSED TIME LAGS ACTUAL TIME DUE TO ERROR IN CONSTANT	MVS
6	PLATFORM MISALIGNMENT CAUSING INITIAL POSITION ERROR LIFTOFF TIME SYNCRONIZATION ERROR PROBABILITY OF MISSING DAS FRAME AT SECO ERROR IN SECO TIME DUE TO SEPARATION OF ACCELEROMETER AND CLOCK SUBROUTINES	MVS ↕ MVS
7	MISSING DAS FRAME AT SECO RADAR COMPUTER OPERATIONAL PROGRAM INCOMPATIBILITY	MVS SST

If the target orbit inclination is chosen to be slightly greater than the launch site latitude, the displacement of the launch site from the orbit plane will be small for a fairly long period of time as shown in Fig. 47. By using a variable launch azimuth, the spacecraft can be inserted parallel to the target orbit plane thus minimizing the out-of-plane displacement. In addition, yaw steering can be used during the second stage of powered flight to eliminate the out-of-plane displacement entirely if the displacement does not exceed a certain limit. This limit is dependent upon the payload penalty the planner is willing to pay and was chosen to be 0.53 degrees for Gemini. By selecting a target orbit inclination 0.53 degrees greater than the launch site latitude, a continuous launch capability is available for 135 min as shown in Fig. 48. This time is known as the launch plane window. Higher inclinations break this window into two parts and lower inclinations decrease the length of the window.

The target orbit altitude also affects the launch window. Since the altitude specifies the period of the orbit, it determines the points in the plane window where the in-plane phasing is equal to the desired value. The range of acceptable phasing values determines the phasing windows or "panes" within the launch window as shown in Fig. 48. A desirable situation would be to achieve a zero phasing error at the same time that the out-of-plane displacement is zero on several consecutive days. While this can be achieved, the required altitudes are either so low that lifetime constraints are violated or so high that the launch vehicle capabilities are exceeded. An altitude of 161 nautical miles was finally chosen since it provides a phasing

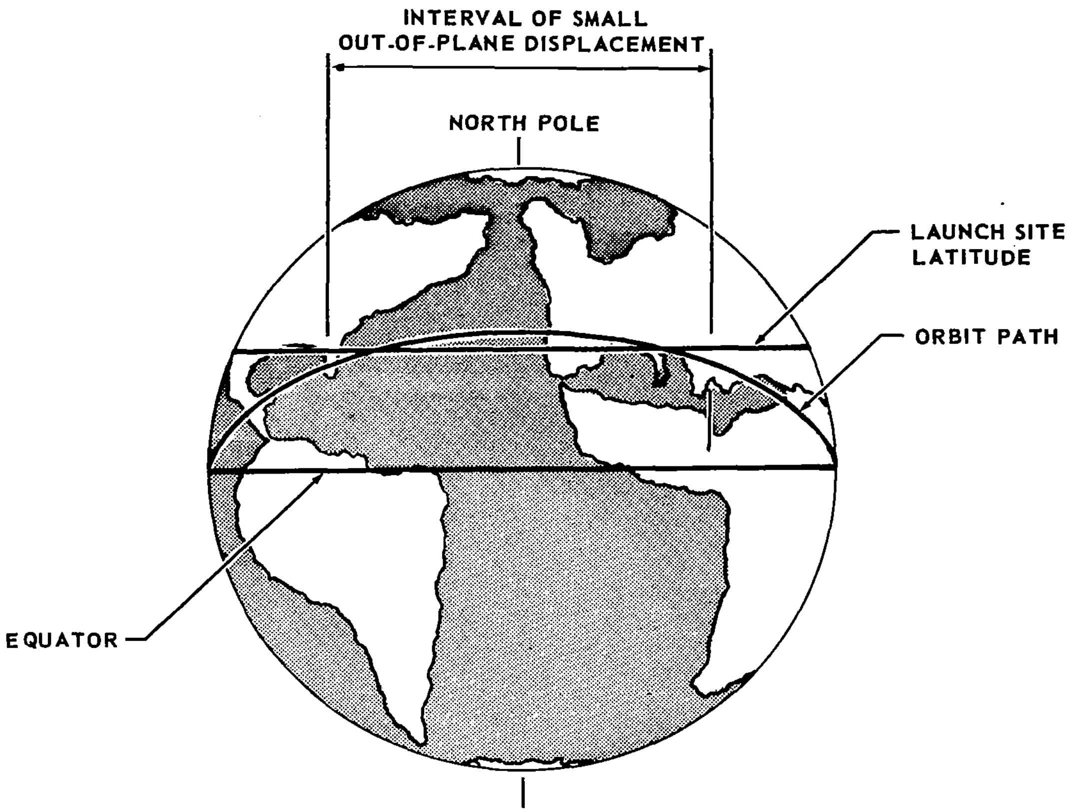


FIGURE 47 ORBIT PATH RELATIVE TO LAUNCH SITE LATITUDE

window within the plane window for several consecutive days and furnishes a near optimum launch condition on the second day after target insertion.

Another orbital selection of importance is the spacecraft insertion altitude. An evaluation was made of the launch vehicle radio guidance system accuracies (which are a function of the elevation angle at insertion), the exit heating requirements, and the launch vehicle performance capability. As a result, an insertion altitude of 87 nautical miles was chosen for the design requirement.

After having developed the basic systems configuration and orbital objectives, three basic design reference missions were established to permit detailed hardware specifications to be written. The first of these was an unmanned ballistic mission for systems and heat protection qualification. The second was a manned orbital 14-day mission with a closed-loop guidance re-entry, and the third was a manned orbital rendezvous and docking with closed-loop guidance re-entry. These missions were flexible enough to allow performance of many other exercises such as extravehicular activity.

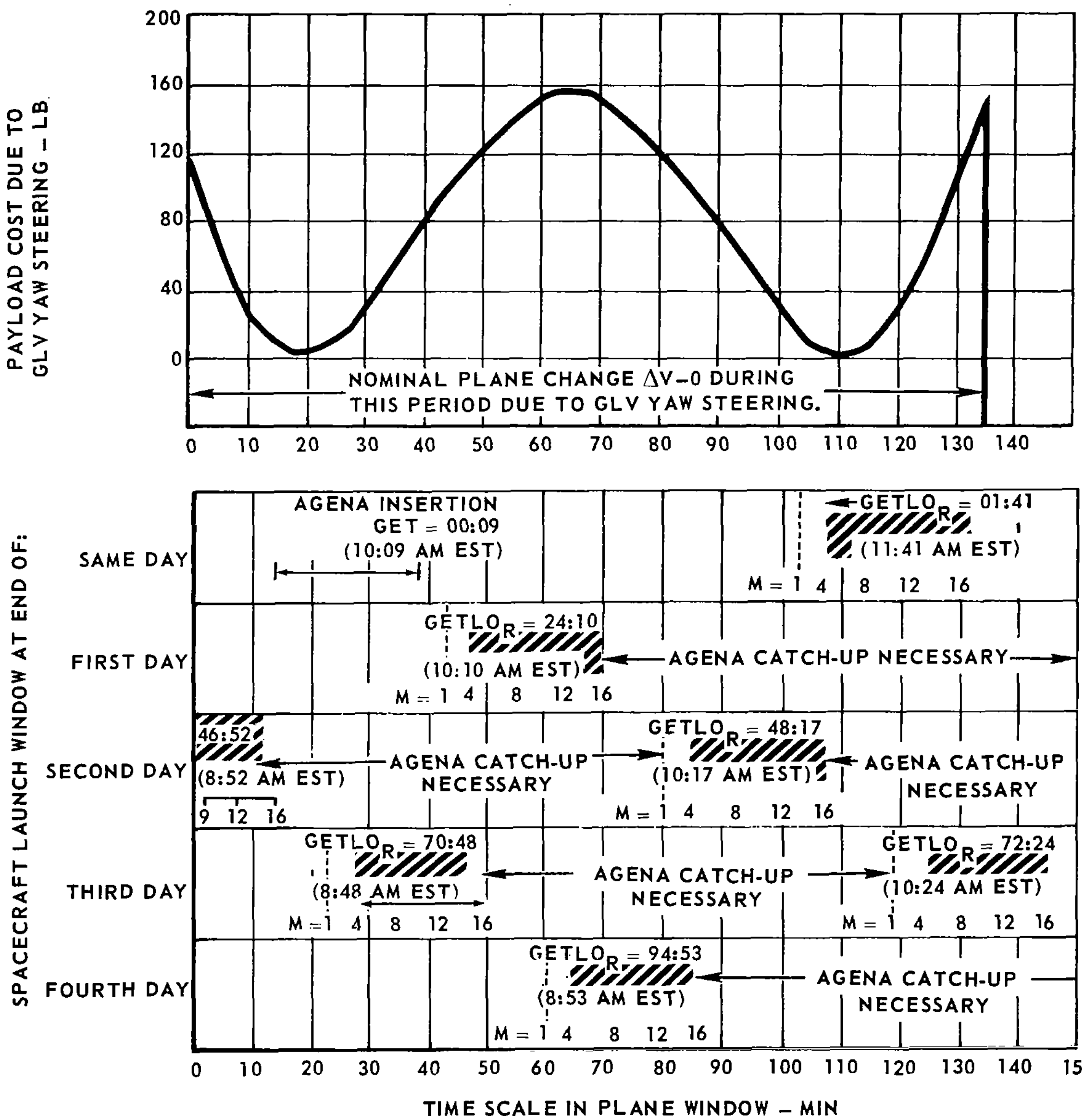


FIGURE 48 LAUNCH WINDOWS FOR TYPICAL GEMINI MISSION

DEVELOPMENT OF OPERATIONAL MISSION PLANS

A logical progression of missions was developed in each of several areas of the missions. As examples, the rendezvous and closed-loop re-entry guidance progressions will be described.

The rendezvous planning began with the Spacecraft 3 mission in which the propulsion and guidance and control systems were evaluated. It continued with the Spacecraft 4 mission where a plan was developed for rendezvous maneuvers and station keeping with the spent second stage of the launch vehicle. The plan for Spacecraft 5 was a rendezvous with a rendezvous evaluation pod (REP) which carried aloft with the spacecraft and ejected in orbit. This pod included a radar transponder and flashing lights to allow a complete checkout of the onboard rendezvous system. While all these exercises were not completed, they were valuable in that they provided flight experience and pointed out problem areas. In the case of Spacecraft 5, the rendezvous evaluation pod exercise was cancelled because of an electrical power problem, but another exercise, referred to as the phantom rendezvous, was planned and executed in real time. This exercise duplicated the maneuvers planned for the midcourse phase of the first Agena rendezvous mission and thus verified the accuracy which could be expected during this phase.

The Spacecraft 6 crew completed the first rendezvous on 15 December 1965, with Spacecraft 7. While this exercise did not include a docking, it was successful and proceeded almost precisely as planned. On the following mission, the Spacecraft 8 crew was able to successfully rendezvous and dock with an Agena Target Vehicle. Rendezvous also was performed successfully on Spacecraft 9 through 12 missions. In addition, several of these missions included more than one rendezvous so that ten rendezvous exercises in all were completed.

The closed-loop re-entry guidance planning began with guidance system qualification on the unmanned Spacecraft 2 mission. On the Spacecraft 3 flight, the closed-loop re-entry procedures were evaluated and a plan was devised for utilizing the procedures on Spacecraft 4. When a computer failure on Spacecraft 4 prevented this, the plan was rescheduled for the Spacecraft 5 mission. However, the ground complex transmitted an incorrect initialization of the re-entry mode on Spacecraft 5, once again delaying the demonstration. Finally, the Spacecraft 6 crew achieved a closed-loop re-entry. Following this, closed-loop re-entries were performed on Spacecraft 7 through 12 for a total of seven successful exercises.

In addition to developing a logical progression of missions, planning for each mission involves consideration of such constraining factors as launch windows, systems requirements, crew procedural requirements, necessary conditions for in-flight experiments and extravehicular activity, and re-entry location. Because of their great importance, the crew procedural requirements will be discussed in detail.

The four main procedural requirements are as follows:

A. Sufficient time for the crew to complete the necessary crew procedures.
B. Approach trajectories which are reasonably insensitive to insertion dispersions and errors in catchup maneuvers.
C. Lighting conditions which are compatible with back-up procedures.
D. Low terminal approach velocities and line-of-sight angular rates.

Allowing time for crew procedures affected several Gemini missions. For example, the Spacecraft 11 first orbit rendezvous was planned so that terminal phase initiation occurred near first spacecraft apogee. One of the primary factors in delaying terminal phase initiation until this point was the time required for crew procedures.

The second procedural requirement played an important part in the early rendezvous planning. It was desired to develop a plan which could effect a near nominal terminal approach trajectory in spite of insertion dispersions, spacecraft equipment degradation or ground tracking and computation errors. This desire stemmed naturally from the need to develop terminal phase procedures which could be employed if a guidance component failed. After three approaches were studied, the coelliptical approach trajectory was selected. This approach has been utilized on all of the Gemini primary rendezvous exercises except that of Spacecraft 11.

The need for lighting conditions which are compatible with back-up procedures has affected all of the rendezvous missions. The desired lighting situation for primary rendezvous, shown in Fig. 49, is that the crew be able to see the following:

A. The target by reflected sunlight prior to terminal phase initiation.
B. The target acquisition lights against a star background during the terminal transfer.
C. The target by reflected sunlight for docking in daylight.

This lighting situation allows the crew to see the target throughout the rendezvous operations and makes possible inertial line-of-sight angle measurements in the event of a platform failure. Lighting conditions are a factor in selecting the terminal phase initiation point, the central angle of the transfer, and the terminal approach angle. The desirable lighting condition for rendezvous with a passive target is very different from that for rendezvous with an active target. Since the passive target is not visible in darkness, the rendezvous is conducted in daylight. In addition, terminal phase initiation does not occur until near the midpoint of daylight. Earlier initiations place the sunline too near the line-of-sight to the target (as shown in Fig. 50) obscuring the target, while later initiations do not allow adequate daylight to complete the rendezvous.

The final requirement imposed by the need for workable onboard procedures is that the trajectory selected have a low terminal approach velocity and line-of-sight angular rate. This requirement was important in the selection of the trajectory parameters in both the coelliptical and first orbit rendezvous plans. The 130 degree transfer utilized on several missions was chosen primarily because of the low line-of-sight angular rate near intercept. For Spacecraft 11, a major reason for the selection of the biased apogee plan over a more direct approach was that the direct approach resulted in a high closing velocity.

The manner in which these constraints were involved in the trajectory planning was illustrated by the evolution of the first Gemini rendezvous plan. The first step in this evolution was the selection of a few concepts for achieving the desired objectives. As shown in Fig. 51, three different

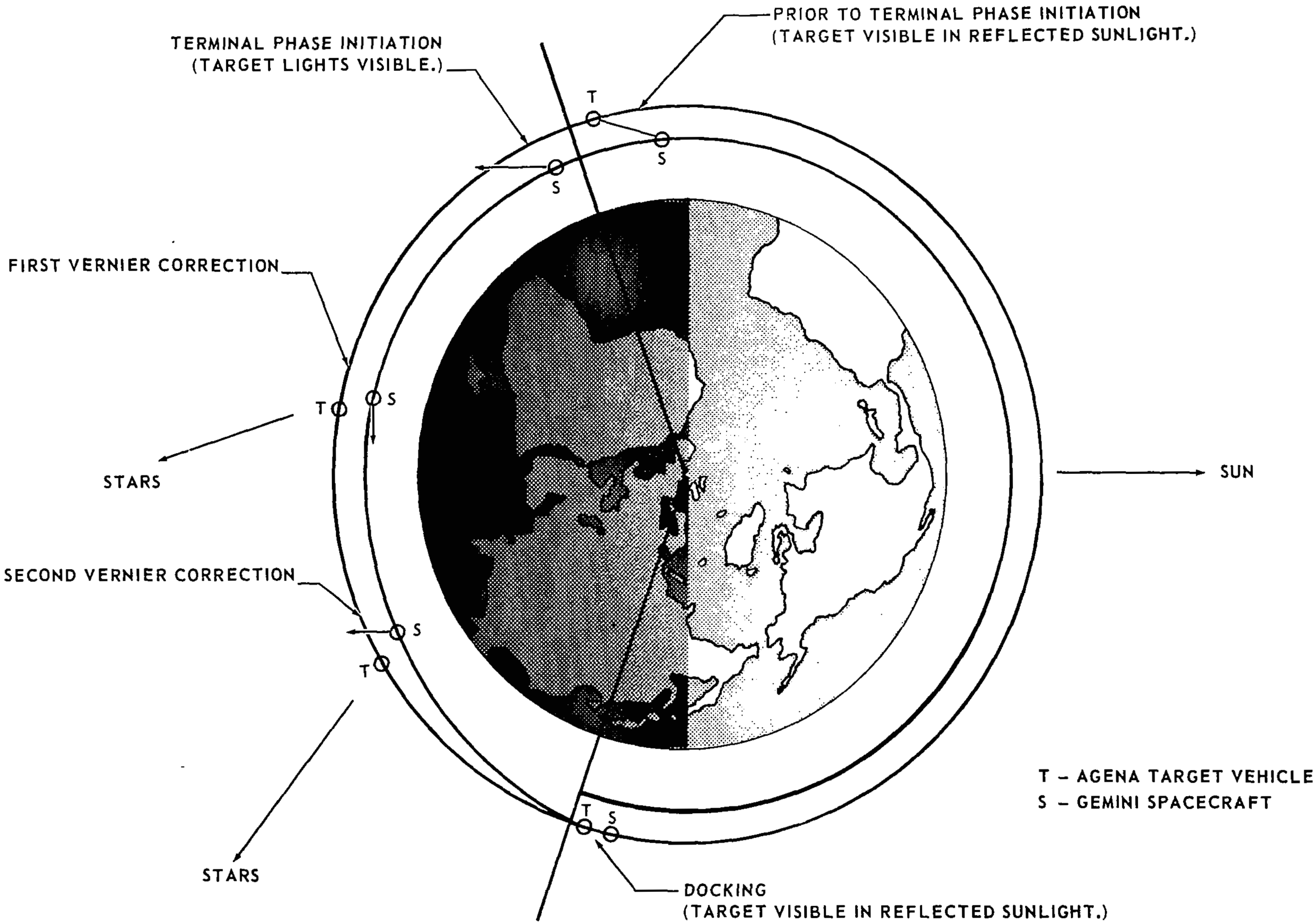


FIGURE 49 DESIRED LIGHTING SITUATION FOR PRIMARY RENDEZVOUS

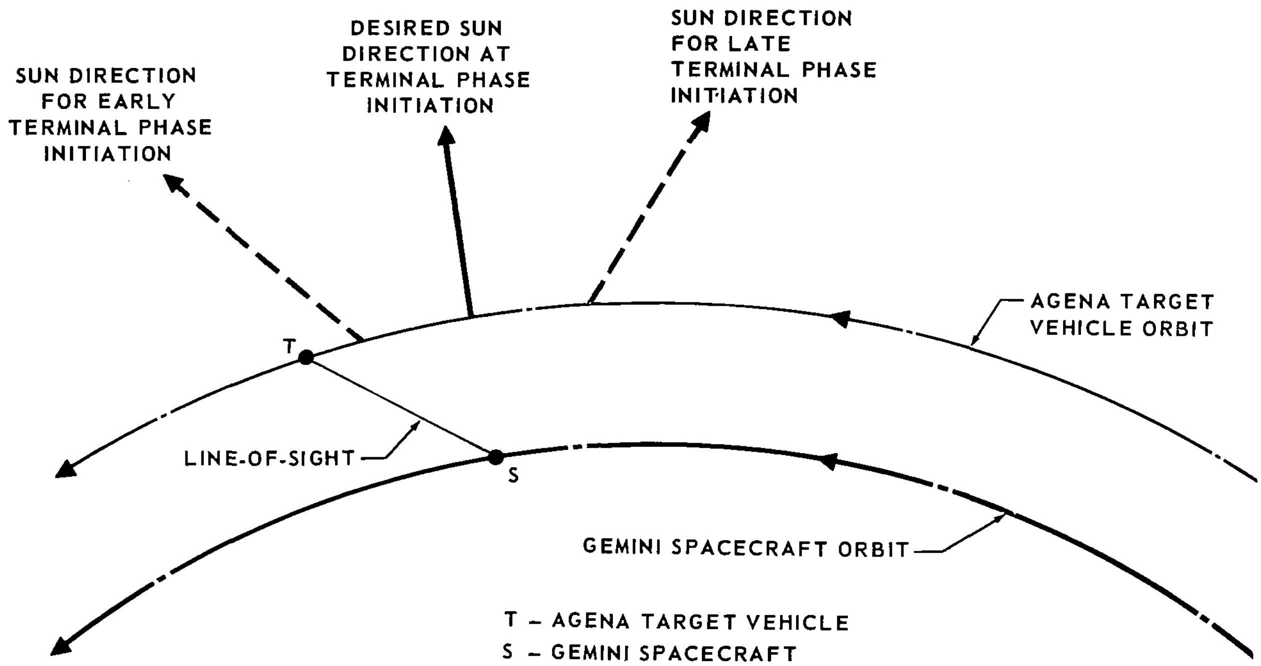


FIGURE 50 DESIRED LIGHTING SITUATION FOR PASSIVE RENDEZVOUS

concepts were selected. The first of these was the tangential plan which employed tangential approach of the spacecraft to the target vehicle following four orbits of ground commanded catchup maneuvers. The closed-loop guidance system was used in this plan only to remove trajectory dispersions since the midcourse maneuvers nominally placed the spacecraft on an intercept trajectory. The second plan has a similar catchup sequence except that the final midcourse maneuver established a coelliptical approach trajectory. The closed-loop guidance system is then used to establish a collision course. A third plan featured rendezvous at first spacecraft apogee. With this plan, the spacecraft would nominally be inserted on a collision course with the target and the closed-loop system would be used to correct insertion dispersion.

Careful analysis of a number of suitable concepts led to the selection of the plan best suited to fulfill the mission objectives. This involved trajectory error analysis, propellant and power consumption studies, an evaluation of the suitability of the plan for back-up guidance procedures and a general evaluation of the flexibility of the plan. For the first Gemini rendezvous, these studies led to the selection of the coelliptical rendezvous plan.

Primary reasons for this selection were as follows:

A. The first apogee rendezvous was eliminated from consideration because of the relatively high expected propellant consumption.

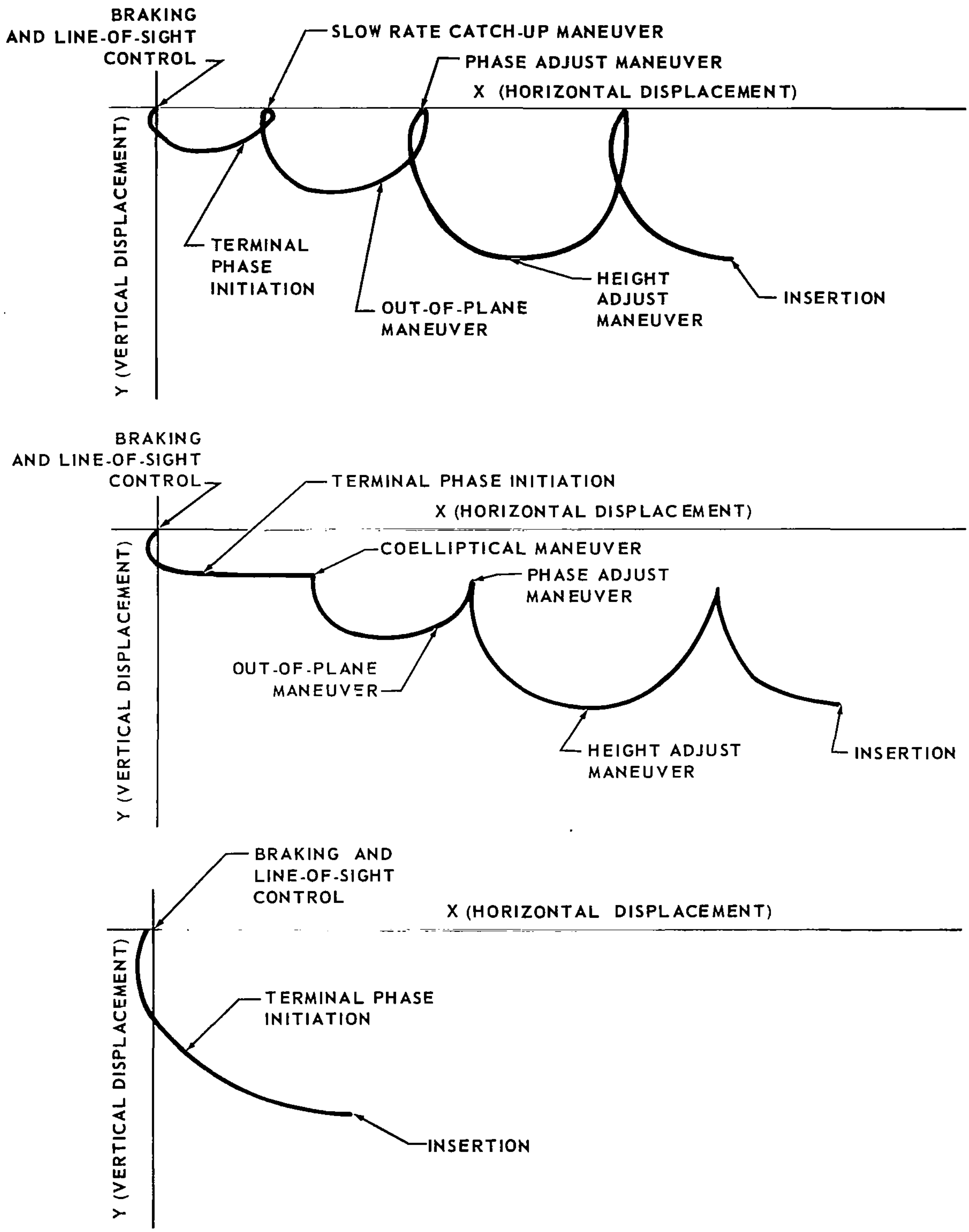


FIGURE 51 CONCEPTS PROPOSED FOR THE FIRST GEMINI RENDEZVOUS

B. The coelliptical plan was chosen over the tangential plan because the terminal phase of the coelliptical plan was much less sensitive to off-nominal trajectory conditions or equipment performance.

Development of a single operational plan involved optimization of the trajectory parameters, development and testing of the ground support programs, and the development, evaluation and crew training associated with the onboard procedures. The onboard procedures are discussed in a following section. The primary trajectory parameters to be optimized for the first rendezvous mission were the central angle of the terminal phase transfer and the coelliptical differential altitude. The central angle to transfer was chosen to satisfy several of the mission planning requirements described earlier, especially in the area of onboard procedures. First, it was desired that the transfer initiation maneuver be along the line-of-sight to the target in order to provide a back-up reference direction for the maneuver. The second requirement was a low terminal line-of-sight angular rate and low closing rate. Finally, it was desired to have the terminal phase initiation point below and behind the target vehicle and the final approach from below and ahead of the target vehicle to optimize the lighting conditions. After an evaluation of all these factors, a 130 degree transfer was selected. The selection of the coelliptical differential altitude was based on a trade-off between (1) the desire to have the range to the target at the TPI point small enough that visual acquisition could be assured, and (2) the need for a large differential altitude in order to minimize the effect of insertion dispersions and catchup maneuver errors on the location of terminal phase initiation. This trade-off resulted in the choice of a 15 nautical mile differential altitude.

This basic plan with only minor modifications was employed on every primary rendezvous of the Gemini program except that of Spacecraft 11. The first orbit rendezvous plan of Spacecraft 11 underwent the same evolutionary phases as the coelliptical plan, but with results compatible with its different objectives.

RESULTS OF MISSION PLANNING

As mentioned previously, the mission planning effort was oriented toward maximizing the probability of success for each mission. A discussion of major mission objectives and accomplishments may be found in SPACECRAFT FLIGHT PERFORMANCE, page 34.

PROCEDURES DEVELOPMENT

Development Of Hybrid Simulation

The Gemini hybrid simulations were a natural evolution of early Gemini simulations, which were either analog or digital and supported a wide range of studies as described in G & C design note 141. Since the requirement for the expanded real-time simulation capability of hybrid systems was already recognized at that time, this design note also describes the planning to achieve a hybrid capability.

After less than a year of development effort, the re-entry hybrid simulation was being used for evaluation of guidance concepts. Major problem areas encountered during this time were digital program checkout and utilization of advanced digital-analog interface equipment. Other hybrid simulations which became operational during 1964 are summarized in Table 32. While the simulations originally were intended for guidance evaluation studies, they became a useful tool for familiarizing flight crews with the required procedures.

A decision that McDonnell contribute directly to procedures development and crew training initiated additional simulation development. Simulation hardware and programs were developed in a parallel effort. Major hardware improvements included: a new digital computer with expanded capability, an additional analog computer, an improved star-horizon display, an improved reaction jet simulator, new Agena target models, and flight-qualified crew stations displays. The crew station is shown in Fig. 52. Simulation developments are outlined in Table 32.

Re-entry Simulation. - Although this simulation was developed for engineering evaluations prior to the procedures development period, only minor changes were required to simulate specific spacecraft responses. The simulation made a significant contribution to the re-entry procedure development and crew indoctrination until after Spacecraft 8, at which time re-entry training was transferred to the Gemini mission simulator.

Rendezvous Simulation. - The hybrid simulation for evaluating Spacecraft 6 rendezvous procedures was initiated by developing an all-digital program on the UNIVAC 1218 computer using a simulated guidance computer programmed with Math Flow Six. To achieve hybrid operation, an EAI 231 analog computer was programmed with the spacecraft dynamics and control systems and an adage conversion unit allowed the necessary transfer of data. To provide for man-in-the-loop, a fixed base crew station was built and a closed circuit television window display was incorporated.

The simulation was completely reprogrammed for Spacecraft 8 utilizing the CDC3200 digital computer. The greater speed and capacity of the 3200 allowed digital simulation of the platform and computation of display drive equations. This reduced the analog program by one-half but required additional digital to analog trunking capability. The trunking capability was accomplished by adding eight new conversion units and developing a multiplexing capability. Also, the math flow program was updated to Math Flow Seven module V to be accurate for Spacecraft 8 simulation. An addition to the analog program was a simulated sextant display which was used in the back-up procedures. This display was continuously improved during the rest of the program.

For Spacecraft 10, the math flow was again updated to Math Flow Seven module III and a new star display was incorporated using a spherical star

TABLE 32 GEMINI HYBRID SIMULATIONS

NAME		SEE NOTE (1) STARTED	COMPLETED	PURPOSE	SIMULATED ELEMENTS
1. RE-ENTRY	a)	3–25–64	7–12–64	TO STUDY RE-ENTRY FLIGHT CONDITIONS WITH MANUAL CONTROL.	ENVIRONMENT (6 DOF), MATH FLOW 3: RE-ENTRY GUIDANCE, PARTIAL ACME.
	b)	5– 3–65	5–21–65	TO STUDY RE-ENTRY TO DESIRED SITE WITH MANNED CONTROL.	SAME AND MATH FLOW 6: RE-ENTRY GUIDANCE.
2. BRAKING AND DOCKING	a)	9– 9–64	12–18–64	TO EVALUATE PROPELLANT AND PROCEDURES FOR BRAKING AND DOCKING.	ENVIRONMENT (RELATIVE MOTION, 6 DOF), PARTIAL ACME.
	b)	2–22–65	8– 7–65	DEVELOP RENDEZVOUS BACK-UP CHARTS AND VERIFY PROPELLANT REQUIREMENTS.	SAME
3. INSERTION VELOCITY ADJUST ROUTINE (IVAR)	a)	9–25–64	10– 2–64	EVALUATE IVAR CAPABILITY AND FAMILIARIZE ASTRONAUTS.	ENVIRONMENT (6 DOF), MATH FLOW 3 AND 6: ASCENT AND CATCH-UP, PARTIAL ACME.
	b)	6–21–65	6–22–65	ASTRONAUT TRAINING	SAME
4. ASCENT		12–30–64	2– 6–65	TO EVALUATE GEMINI ASCENT BACK-UP CAPABILITY.	ENVIRONMENT (6 DOF) MATH FLOW 3: ASCENT, RADIO GUIDANCE, PRIME AND SECONDARY AUTOPILOT, BOOSTER ENGINES, AND MALFUNCTION DETECTOR.
5. RE-ENTRY				ASTRONAUT TRAINING AND PROCEDURE DEVELOPMENT.	SEE 1b
6. RENDEZVOUS	a)	7– 6–65			ENVIRONMENT (6 DOF), MATH FLOW 6: CATCHUP AND RENDEZVOUS, PARTIAL ACME.
(STATION-KEEPING)	b)	10–11–65	10–21–65		SAME
	c)	12– 6–65			ENVIRONMENT (6 DOF & PLAT) MATH FLOW 7: MOD 5: CATCH-UP AND RENDEZVOUS, ACME.
(IVAR)	d)				SAME AND IVAR
(MODULE 3)	e)	5– 9–66		ASTRONAUT TRAINING AND PROCEDURE DEVELOPMENT.	SAME AS 5a EXCEPT MODULE 3 AND SIMULATED SEXTANT.
(MODULE 3–2)	f)	7–15–66	9–17–66		SAME AS 5e WITH ATM AND IVAR PORTIONS OF MODULE 2.
7. MANEUVER WHILE DOCKED		2–22–66	7–11–66		ENVIRONMENT (6 DOF, PLAT) CATCHUP GUIDANCE, COMBINED GEMINI–AGENA DYNAMICS, ACME. AGENA PRIMARY PROPULSION, ENCODER COMMAND SYSTEM. AGENA ASC ADDED FOR SPACECRAFT 11.
8. NAVIGATION (TOUCHDOWN PREDICT)	a)	12–27–65			TOUCHDOWN PREDICT GUIDANCE.
(ORBIT NAV. AND PREDICT.	b)	5–16–66			ENVIRONMENT (6 DOF, PLAT), MATH FLOW 7: ORBIT PREDICT AND NAVIGATION.
(MODULE 6)	c)	5–26–66			ENVIRONMENT (6 DOF, PLAT), MATH FLOW 7: MODULE 6, ACME.
(MODULE 6–3)	d)	7–15–66	8–19–66		SAME AS 8c WITH ATM AND TOUCHDOWN PREDICT.

NOTES: (1) DATES DO NOT ALWAYS REPRESENT CONTINUOUS OPERATION; (2) EXPANDED FROM 16 TO 24 D/A CHANNELS

TABLE 32 GEMINI HYBRID SIMULATIONS (*Continued*)

EQUIPMENT	DISPLAY	COMMENTS AND REFERENCES
DIGITAL: UNIVAC 1218 CONVERT: ADAGE 770 ANALOG: EAI 231	CREW STATION II HORIZON	SEE GDN 249
SAME	SAME	SEE GDN 299 PROVIDED ASTRONAUT FAMILIARIZATION.
ANALOG: EAI 231	CREW STATION I MOVING TARGET MODEL.	SEE GDN 264
SAME	SAME AND STAR DISPLAY.	PROVIDED ASTRONAUT FAMILIARIZATION.
DIGITAL: UNIVAC 1218 CONVERT: ADAGE 770 EAI DOS ANALOG: EAI 231	CREW STATION II	SEE GDN 241 USED ASCENT SIMULATION FOR INITIALIZATION.
SAME	SAME	FOR SPACECRAFT 6.
DIGITAL: UNIVAC 1218 EAI DOS CONVERT: ADAGE 770 ANALOG: EAI 231	CREW STATION II	SEE GDN 289
SEE 1	SEE 1	SEE SECTION 4.6.1.1
DIGITAL: UNIVAC 1218 CONVERT: ADAGE 770 EAI DOS ANALOG: EAI 231 (2)	CREW STATION II MOVING AGENA MODEL, STAR DISPLAY.	SEE SECTION 4.6.1.2 USED FOR SPACECRAFT 5 AND 6.
SAME	SAME EXCEPT BOOSTER MODEL.	SEE GDN 388 USED FOR SPACECRAFT 7.
DIGITAL: CDC 3200 CONVERT: ADAGE 770 (2) EAI DOS ANALOG: EAI 231	SAME EXCEPT AGENA MODE L.	USED FOR SPACECRAFT 8AND 9.
SAME	SAME	REQUIREMENT DELETED.
SAME AND EAI 131	SAME AND STAR PLANETARIUM.	USED FOR SPACECRAFT 10.
SAME AS 5e.	SAME AS 5a EXCEPT STAR PLANE-TARIUM.	USED FOR SPACECRAFT 11 AND 12.
DIGITAL: CDC 3200 EAI DOS CONVERT: ADAGE (2) ANALOG: EAI 231 EAI 131	CREW STATION II AGENA STATUS PANEL.	SEE SECTION 4.6.1.3 USED ON SPACECRAFT 9, 10 AND 11.
DIGITAL: CDC 3200 CONVERT: EAI DOS	CREW STATION II	SEE SECTION 4.6.1.4 USED FOR SPACECRAFT 8.
DIGITAL: CDC 3200 CONVERT: ADAGE 770 (2) EAI DOS ANALOG: EAI 231	CREW STATION II	USED FOR PRELIMINARY SPACECRAFT 10 WORK.
SAME AS 8b.	CREW STATION II STAR PLANETARIUM	USED FOR SPACECRAFT 10.
SAME AS 8c.	SAME AS 8c	DEVELOPED FOR SPACE-CRAFT II AND USED FOR SPACECRAFT 12.

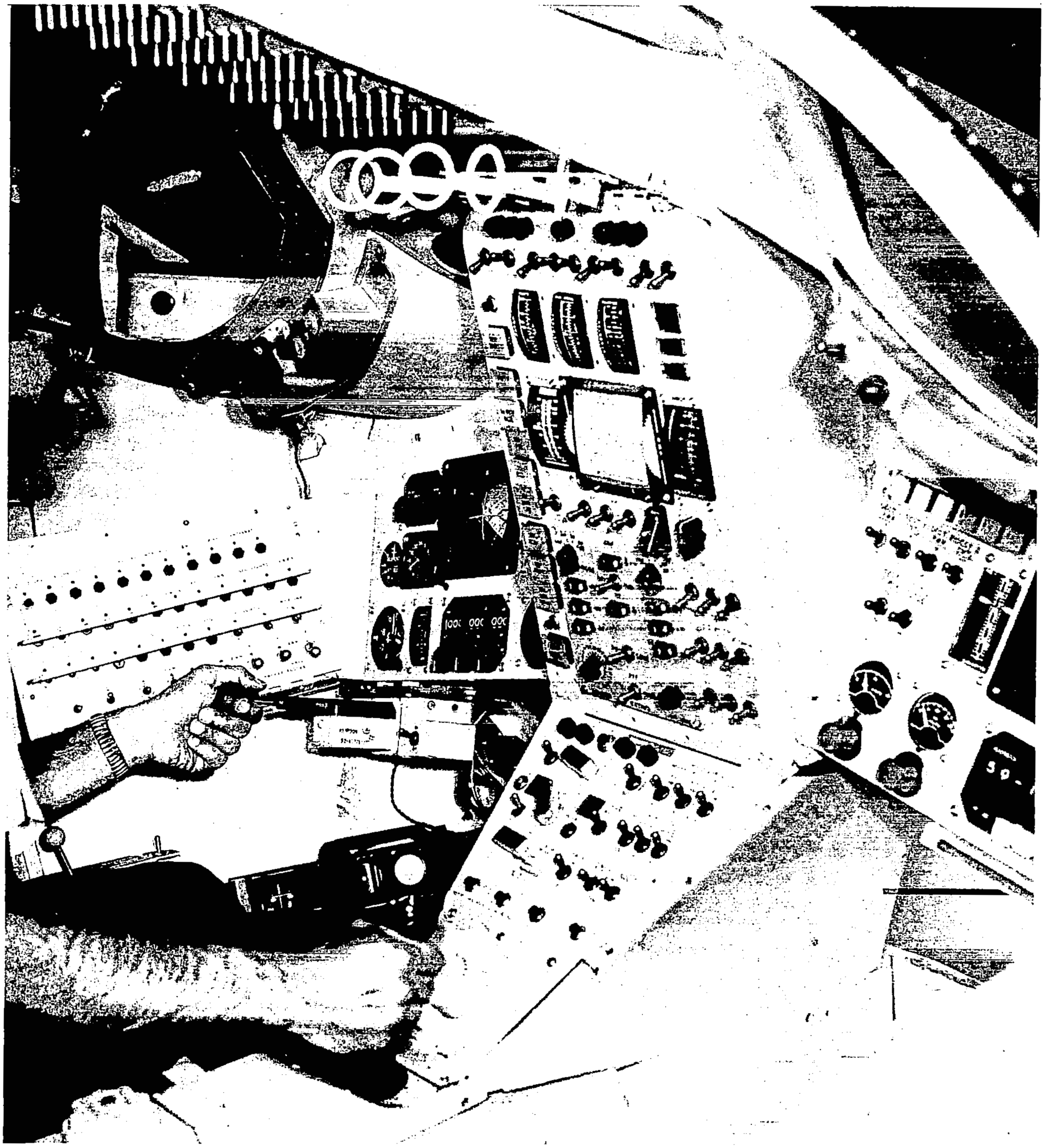


FIGURE 52 HYBRID SIMULATOR CREW STATION

field model. The new star display greatly increased the operating range and accuracy of the star display.

Spacecraft 11, the IVAR portion of module II and an auxiliary tape memory (ATM) program were added to the simulated math flow. This, along with applicable changes in the environment program, provided a simulation beginning at SECO and continuing through rendezvous in the first orbit.

Maneuver-While-Docked Simulation. - This simulation evolved from portions of the rendezvous simulation and included extensive revisions to incorporate desired Agena systems, such as primary propulsion and remote command logic. While minor changes were made on a flight-to-flight basis to maintain the desired configuration, only one major change was made. This was the extension of the Agena control system to include high order and nonlinear control system effects for Spacecraft 11.

Navigation Simulation. - The navigation simulation was composed of several related onboard guidance modes. The first to be developed was touchdown predict, which at that time was a portion of Math Flow Seven. This first simulation was developed to drive crew station displays only.

Other environment and navigation program sinulations, orbit navigation and orbit predict, were developed concurrent with Math Flow Seven. When module 6 of Math Flow Seven was defined, orbit determination and the necessary realistic star display were added to make a complete simulation. Finally, touchdown predict (module 4) was integrated with module 6 to support the onboard re-entry initialization procedures studies.

Rendezvous Procedures Development

The primary means to accomplish rendezvous with a cooperative target was closed-loop, i.e., utilizing a radar, computer, and platform. To increase the probability of mission success, an alternate or back-up to the closed-loop system which provided a monitoring capability and back-up guidance information in case of a closed-loop equipment failure. The back-up guidance commands were based on visual observations of the target with a reticle or sextant in conjunction with flight charts. The flight charts, consisting of tables, graphs, and nomograms, were derived from orbital mechanics relationships and provide the velocity change maneuvers required to rendezvous in components convenient to the crew. Flight charts of this form also were required to accomplish rendezvous with an uncooperative or passive target such as a "dead" Agena target which was used on a previous mission.

Preparation of flight charts and the crew procedures that accompany their use was followed by engineering evaluation on the hybrid simulator. During this evaluation, the flight charts were subjected to a wide variety of trajectory dispersions and their performance was compared with that of the closed-loop system. The procedural time line was refined to simplify the chart usage as much as possible and thereby reduce the probability of a procedural error. The charts and hybrid simulator then were prepared for crew training. The weeks of crew indoctrination provided both the primary and back-up flight crews with a good working knowledge of the procedures and flight charts, and procedures were further tailored to suit crew preferences. In addition, estimates of the orbit attitude and maneuver system (OAMS) propellant consumption for various rendezvous situations were obtained.

The following paragraphs describe briefly the rendezvous missions, outline the development of the procedures and flight charts for both primary and rerendezvous and present significant results of the engineering evaluation and crew training. However, it should be noted that the procedures are described as they were developed and that in some cases, these were not precisely followed in the actual flight.

Initial Rendezvous.

A. Midcourse Maneuvers - The Gemini primary rendezvous missions included a number of midcourse maneuvers which corrected insertion errors and orbit perturbations prior to the terminal rendezvous phase. The maneuvers were designed to position the terminal phase initiation (TPI) point at the correct trailing displacement and differential altitude from the target with planned lighting conditions. The following is a discussion of the various maneuvers as applied to each primary rendezvous mission including the source of the midcourse velocity corrections.

Spacecraft 6 was launched into an orbit essentially coplanar with the target vehicle and performed ground-supplied midcourse maneuvers to rendezvous during the fourth spacecraft revolution. An apogee height adjustment was performed at the first perigee following insertion at perigee to correct for in-plane insertion dispersions and orbit decay due to atmospheric drag. A phase adjustment was made at second apogee to raise perigee and reduce the catchup rate to provide the correct phase angle at third apogee. An out-of-plane correction occurred at the nodal crossing just following second apogee to place the spacecraft in the same orbital plane as the target. A vernier height adjustment was performed at second perigee to compensate for small errors in the initial height adjustment and permit a more nominal terminal phase. At third spacecraft apogee a forward ΔV was applied to raise perigee and produce an orbit coelliptical with and 15 nautical miles below the target orbit in preparation for the TPI maneuver.

Spacecraft 8 primary rendezvous midcourse maneuvers were identical to those performed during the Spacecraft 6 primary rendezvous.

Spacecraft 9 primary rendezvous included ground-supplied midcourse maneuvers which resulted in rendezvous with the target during the third spacecraft revolution. These maneuvers included a phase adjustment at first apogee to provide the correct phase angle at second apogee. A combination phasing, height adjust, and out-of-plane correction was performed approximately one quarter orbit prior to the coelliptical maneuver at second apogee. The coelliptical maneuver and a vernier out-of-plane correction were performed at second apogee.

Spacecraft 10 primary rendezvous included a procedure for computer and sextant usage to demonstrate an onboard navigation capability. The flight crew was required to calculate the midcourse maneuvers necessary to produce the desired orbit conditions at TPI by using the orbit prediction and determination modes of module VI of the Gemini onboard computer. These maneuvers were the same as those used on the Spacecraft 6 mission with the exception of the apogee height adjust at first perigee which was replaced by an insertion velocity adjust routine (IVAR) correction at insertion.

The procedures began by making the insertion velocity adjust routine (IVAR) correction to downrange and vertical velocity after which module VI was loaded and the orbit determination mode was selected and started with the ascent navigated insertion state vector. In the first darkness period, the altitude of the observed horizon was calibrated by taking sextant measurements to the selected star, and plotting the resulting computed measurement residuals on a flight chart. After the calibration, the remaining darkness was used to gather data for the computer by taking two sextant sightings and by inputting a dummy measurement. Following the first darkness period activity, the orbit predict mode of module VI was started with the ascent vector and used with flight charts to obtain solutions to the midcourse maneuvers. The second darkness period was used to complete the data table of the orbit determination mode. Two additional sextant measurements were made, and a second dummy measurement was input, thus filling the data table with the required six measurements. After processing the data contained in the six measurements, the computer produced an updated vector which attempted to eliminate any IGS errors contained in the ascent vector. Using the orbit predict mode once again, this updated vector was used with the flight charts to obtain a second set of onboard solutions to the midcourse maneuvers. With the ground solutions to these maneuvers as the basis of comparison, the onboard maneuvers were evaluated to determine whether they were within preflight established bounds. The selected maneuvers then were performed in the normal manner. An addition onboard capability similar to that planned for Spacecraft 12 was provided for calculating the circularization maneuver in which radar range data was used with a flight chart to obtain a solution.

Spacecraft 11 rendezvous in the first orbit contained two maneuvers prior to the terminal phase, a radial insertion correction and an out-of-plane maneuver. Both maneuvers were calculated on board. Because of the interrelationship between these midcourse maneuvers and the TPI maneuver a discussion of the former will be reserved for a subsequent section on the terminal phase.

Spacecraft 12 primary rendezvous was similar to that of Spacecraft 9 except for the onboard computation of the coelliptic maneuver at second apogee and out-of-plane corrections during the terminal phase. The coelliptic maneuver was calculated from radar range measurements and onboard flight charts. The charts converted the radar measurements into a trajectory fix, and supplied the ΔV to be applied at the ground determined time of second apogee to provide a coelliptic orbit with a nominal differential altitude of ten nautical miles. Out-of-plane corrections were calculated from platform yaw gimbal angles measured with the target in the center of the reticle, radar range measurements and onboard flight charts. The actual change in the yaw gimbal angle over a specified interval was compared with the required change for the appropriate angular travel to rendezvous. The results of this comparison were used in the charts to obtain the out-of-plane ΔV to be applied.

B. Terminal Phase Maneuver - The terminal phase of Spacecraft 6 primary rendezvous contained a 130 degree transfer maneuver occurring nominally one minute after entering darkness. The closed-loop guidance system was considered the primary method to accomplish rendezvous; however, back-up guidance procedures and flight charts were developed in the event of guidance equipment failure. The terminal phase initiation (TPI) maneuver began with the initiation cue which was chosen to be the elevation angle of the line-of-sight to the target. This cue provides a convenient pointing reference during thrust application and permits radar lock-on during the closed-loop maneuver. The closed-loop TPI solution was monitored with a back-up solution computed using the line-of-sight elevation angle and radar range and the chart in Fig. 53. This chart provided a line-of-sight maneuver based on the nominal forward ΔV plus a correction which was proportional to the deviation in the range measurements from the nominal. A maneuver normal to the line-of-sight which was directly proportional to both the angular variation from the nominal and the radar range was provided.

Following the TPI maneuver the command pilot remained boresighted on the target while the computer collected radar data necessary for the first vernier closed-loop solution. After a period of optional platform alignment, the pilot read radar range and elevation angle measurements from the computer and calculated the back-up solution for the closed-loop vernier correction using a chart similar to the TPI maneuver chart. This chart was similar to the TPI maneuver chart and provided velocity corrections along and normal to the line-of-sight. The IVI's displayed the closed-loop vernier correction at a point in the trajectory where 81.8 degrees central angle travel of the target remained until intercept. Upon completion of the thrusting, the first vernier was complete and an identical cycle was repeated for the second closed-loop vernier correction which occurred at 33.6 degrees central angle to go to rendezvous. Four back-up solutions were computed following the TPI maneuver to detect a trajectory error as soon as possible. The second and fourth back-up vernier corrections provided a check on the two closed-loop vernier corrections. Following the last vernier correction the command pilot observed any motion of the target with respect to the celestial background and nulled the line-of-sight motion. The pilot monitored range and range rate information from the computer to determine trajectory characteristics and give position reports. Braking thrust was applied as a function of range with initiation at 15,000 ft, braking to four fps at 3000 ft, and to one-half fps at 100 ft. Since a radar failure precluded the use of range information, a braking schedule as a function of time was followed until visual growth of the target was detected.

In the event of a radar failure, range and range rate information were not available for terminal phase back-up solutions. The TPI initiation cue was elevation angle as in the closed-loop case. However, the spacecraft was controlled to a specified attitude and when the target drifted through the

GEMINI VI RENDEZVOUS FLIGHT CHARTS
INITIAL THRUST CALCULATION

ANGULAR RATE CORRECTION

GET: θ_A ___:___ + 3:20 = GET: θ_C ___:___ → + 4:30 = GET TO STOP-RESET-START ___:___

θ_{Aa} DEG	θ_{Ca} DEG	θ_{CN} DEG	$\Delta\theta_C$ DEG	$\Delta\theta_C$ DEG	I	II NOM	III	Δt SEC	Δt UP-DWN	ΔV UP-DWN
19.5	–	22.1 =		+2.0				29	130 SEC	46 FPS
19.6	–	22.3 =		+1.0				15	67 SEC	24 FPS
19.7	–	22.4 =		+ .8				12	54 SEC	19 FPS
19.8	–	22.5 =		+ .6				9	39 SEC	14 FPS
19.9	–	22.7 =		+ .4				6	26 SEC	9 FPS
20.0	–	22.8 =		+ .2				3	13 SEC	4 FPS
20.1	–	22.9 =		0	0.0	0.0	0.0	0	0 SEC	0 FPS
20.2	–	23.1 =		– .2				3	13 SEC	4 FPS
20.3	–	23.2 =		– .4				6	26 SEC	9 FPS
20.4	–	23.3 =		– .6				9	39 SEC	14 FPS
20.5	–	23.4 =		– .8				12	34 SEC	19 FPS
20.6	–	23.6 =		–1.0				15	67 SEC	24 FPS
20.7	–	23.7 =		–2.0				29	130 SEC	46 FPS

RADAR FAILURE
POINTING COMMAND AFTER PT C:
$\Delta\dot{X}$ = 25 00284
$\Delta\dot{Y}$ = 26 90147
$\Delta\dot{Z}$ = 27 00000

COMP FAILURE: → BALL 15.5
TGT AT TOP

α_{CN} 5.1 – α_{Ca} ___ = $\Delta\alpha_C$ ___ × 2 = $\Delta\theta_C$ ___

TIME: ___:___ + 1:40 – ___:___

INITIATE BALL: 27.5

R_{Ba} ___ – R_{Ca} ___ = ___ × 2 = ΔR_a ___

INITIATE RANGE: 32.96 NM

PLAT FAILURE: → R = 41.00

ΔV OR ΔT APPLIED
FWD: ______
AFT: ______
UP: ______
DWN: ______
LT: ______
RT: ______

RANGE RATE CORRECTION

R_{Ba} ___.___
+ 2.50
R_A ___.___

	R_A NM	R_A – NM	R_C NM	ΔR_a NM	ΔR_a = NM	$\epsilon\Delta R$ NM		$\epsilon\Delta R$ NM	$\Delta t_{\Delta R}$ SEC	Δt SEC	Δt FWD	ΔV FWD
I	39.00	–	=		–4.29 =		RADAR OR COMP FAILURE ↔ APPLY NOMINAL	–.50	60 –		= SEC	47 FPS
I	40.00	–	=		–4.42 =			–.40	56 –		= SEC	44 FPS
I	41.00	–	=		–4.56 =			–.30	52 –		= SEC	41 FPS
I	42.00	–	=		–4.71 =			–.20	48 –		= SEC	38 FPS
II NOM	43.00	–	=		–4.84 =			–.10	44 –		= SEC	35 FPS
II NOM	45.45	–	=		–4.90 =			0	41 –		= SEC	32 FPS
II NOM	44.00	–	=		–4.97 =			+.10	37 –		= SEC	29 FPS
II NOM	45.00	–	=		–5.11 =			+.20	33 –		= SEC	26 FPS
III	46.00	–	=		–5.24 =			+.30	29 –		= SEC	23 FPS
III	47.00	–	=		–5.39 =			+.40	25 –		= SEC	20 FPS
III	48.00	–	=		–5.52 =			+.50	22 –		= SEC	17 FPS

FIGURE 53 INITIAL THRUST CALCULATION SHEET

center of the reticle, thrust was initiated. The nominal maneuver along the line-of-sight was applied and the maneuver normal to the line-of-sight was computed from the deviation of observed elevation angle from the nominal. Only vernier corrections normal to the line-of-sight were applied in the absence of range data.

A computer failure precluded the use of accurate radar and elevation angle information; however, the TPI initiation cue could have been obtained from the attitude ball while holding boresight. The TPI maneuver consisted of the nominal velocity correction along the line-of-sight and a calculated correction normal to the line-of-sight based on the deviation of inertial elevation angle from the nominal. The first two vernier corrections consisted of only the velocity correction normal to the line-of-sight based on inertial elevation angle variations. The second two vernier corrections also included the component along the line-of-sight which was computed using range rate data from the analog meter.

In the event of a platform failure, the TPI initiation cue was radar range obtained from the computer. The TPI maneuver and the four vernier corrections were computed from the variations in range, range rate, and inertial elevation angle.

Although the procedures development and flight chart derivation for the terminal phase of most Gemini rendezvous missions were very similar, differences in format and data measurement techniques did occur through flight experience. The evolution of the procedures and flight charts is outlined in Table 33 for Spacecraft 6, 8, 9, 10 and 12 primary rendezvous missions. This tabulation includes a comparison by spacecraft of the TPI initiation cues and data measurement techniques used to determine the back-up TPI maneuver and back-up vernier corrections for both closed-loop guidance and radar, computer, and platform failure cases. The major revision in format was the change from tabular chart form for Spacecraft 6 rendezvous to the graph and nomograph format for Spacecraft 8 and up missions as shown for the TPI maneuver in Fig. 54. This change allowed direct interpolation without calculations and an expansion of the data to include a wider range of variables. The engineering evaluation and crew training sessions for the Spacecraft 6, 8, 9 and 10 primary rendezvous are summarized in G & C design notes 317, 351, 352, 370, 389 and 409.

The development of procedures and flight charts for Spacecraft 11 direct rendezvous necessitated a reaction to requirements not imposed in previous rendezvous missions. These requirements included:

1. Compression of the time allotted for procedures and back-up guidance computations due to the limited time from insertion through intercept.

2. The necessity for a radial insertion correction to eliminate the effect of downrange and flight path angle insertion errors on spacecraft position at TPI.

3. The use of nonnominal trajectory parameters in case of closed-loop failure due to wide range of possible TPI dispersions.

In addition to the above requirements, two out-of-plane corrections were planned for the first time, one at 90 degrees of orbit travel from insertion and a second near TPI to be applied in case of closed-loop failure.

To satisfy the requirement for a procedural load which would allow rendezvous within one orbit, the following guidance and control function time line was established and shown to be practical during hybrid simulation.

GROUND ELAPSED TIME (Hrs: Min: Sec)	EVENT
00:00	Lift-off
06:00	Insertion Insertion Velocity Adjust Routine (IVAR) Radial Correction
10:00	Begin Platform Alignment Load Module 3 with ATM Compute Out-of-Plane Correction to be applied at 90 degrees of Orbital Travel
25:00	End Platform Alignment Acquire Radar Lock-On
29:00	Out-of-Plane Correction Select Rendezvous Mode Back-up Guidance Computations
49:00	TPI Back-up Vernier Computations
1:01:00	First Vernier Correction Back-up Vernier Computations
1:13:00	Second Vernier Correction Braking and Line-of-Sight Control
1:18:00	Intercept

A change in the math flow for Spacecraft 11 allowed the pilot to read the navigated downrange angle travelled at insertion. From this angle a nominal value of radius rate was determined and compared to the actual navigated radius rate read from the computer. The velocity difference was applied radially to achieve a radius rate which would result in the desired trailing displacement at TPI. This correction was applied in addition to the normal IVAR correction.

The procedures and flight charts provided a back-up to the TPI closed-loop solution based on the desired 120 degrees of orbital travel to rendezvous and the predicted position of the spacecraft at relative apogee (the point

TABLE 33 PROCEDURE SUMMARY FOR THE TERMINAL PHASE OF THE GEMINI PRIMARY RENDEZVOUS

MISSION	FAILURE	TPI CUE	TPI MANEUVER DATA ALONG LOS	TPI MANEUVER DATA NORMAL TO LOS	VERNIER CORRECTION DATA ALONG LOS	VERNIER CORRECTION DATA NORMAL TO LOS
6	NONE	ANGLE(MDIU)	CLOSED-LOOP	CLOSED-LOOP	CLOSED-LOOP	CLOSED-LOOP
	PLATFORM	RANGE(MDIU)	RANGE(MDIU)	RANGE(MDIU),ANGLE(STARS)	RANGE(MDIU)	RANGE(MDIU) ANGLE(STARS)
	COMPUTER	ANGLE(BALL)	NOMINAL	RANGE(NOM.),ANGLE(STARS)	RANGE RATE METER	RANGE(NOM.),ANGLE(STARS)
	RADAR	ANGLE (CU MODE)	NOMINAL	RANGE(NOM.),ANGLE(MDIU)	NONE	RANGE(NOM.),ANGLE(MDIU)
8	NONE	ANGLE(MDIU)	SAME AS 6	SAME AS 6	SAME AS 6	CLOSED LOOP
	PLATFORM	ANGLE (SEXTANT)				RANGE(MDIU),ANGLE(STARS)
	COMPUTER	ANGLE(BALL)				RANGE(METER),ANGLE(STARS)
	RADAR	ANGLE (CU MODE)				RANGE(NOM.),ANGLE(MDIU)
9	NONE	SAME AS 8	SAME AS 6	CLOSED-LOOP	SAME AS 6	CLOSED-LOOP
	PLATFORM			RANGE(MDIU),ANGLE(STARS)		RANGE(MDIU),ANGLE(STARS)
	COMPUTER			RANGE(NOM.),ANGLE(BALL)		RANGE(METER),ANGLE(BALL)
	RADAR			RANGE(NOM.),ANGLE(MDIU)		RANGE(NOM.),ANGLE(MDIU)
10	NONE PLATFORM COMPUTER RADAR	SAME AS 8	SAME AS 6	SAME AS 9	SAME AS 6	SAME AS 9
12	NONE PLATFORM COMPUTER RADAR	SAME AS 8	SAME AS 6	SAME AS 9	SAME AS 6	SAME AS 9

TABLE 33 PROCEDURE SUMMARY FOR THE TERMINAL PHASE OF THE GEMINI PRIMARY RENDEZVOUS (Continued)

NO. OF CORRECTIONS	DATA FOR LOS CONTROL & BRAKING	IMPROVEMENTS FROM PRIOR MISSION
2 4 4 4	ANGLE(STARS), RANGE(METER) ANGLE(STARS), RANGE(METER) ANGLE(STARS), RANGE(METER) ANGLE(STARS), TIME(GET CLOCK)	
2 4 4 4	ANGLE(STARS), RANGE(METER) ANGLE(STARS), RANGE(METER) ANGLE(STARS), RANGE(METER) ANGLE(STARS), RANGE(SEXTANT)	1. MORE ACCURATE CHART 2. CURVES USED TO REPLACE TABLES. 3. SEXTANT AVAILABLE FOR ANGLE CUE FOR PLATFORM FAILURE CASE AND RANGE DETERMINATION FOR BRAKING(RADAR FAILURE). 4. LARGER RANGE OF DISPERSIONS COVERED.
2 4 4 4	SAME AS 8	1. WIDER RANGE OF VARIABLES ON CHARTS. 2. PITCH ANGLE IS DETERMINED FROM BALL INSTEAD OF RETICLE FOR COMPUTER FAILURE. 3. CHART FOR LOS RATE NULLING ADDED. 4. CHART FOR RANGE RATE DETERMINATION USING SEXTANT ADDED. 5. CHART ADDED TO COMPENSATE FOR OFF NOMINAL PITCH
2 4 4 4	SAME AS 8	1. BETTER PLATFORM ALIGNMENT PROCEDURE 2. INERTIAL FLIGHT DIRECTOR INDICATOR AVAILABLE FOR LOS RATE NULLING
2 4 4 4	SAME AS 8	1. MONITORING CAPABILITY ADDED FOR OUT-OF-PLANE CORRECTION FOR TPI AND VERNIERS, ALSO, BACK UP OUT-OF-PLANE CAPABILITIES ADDED FOR RADAR FAILURE CASE. 2. BETTER PLATFORM ALIGNMENT AS A RESULT OF DECREASED DATE COLLECTION INTERVAL

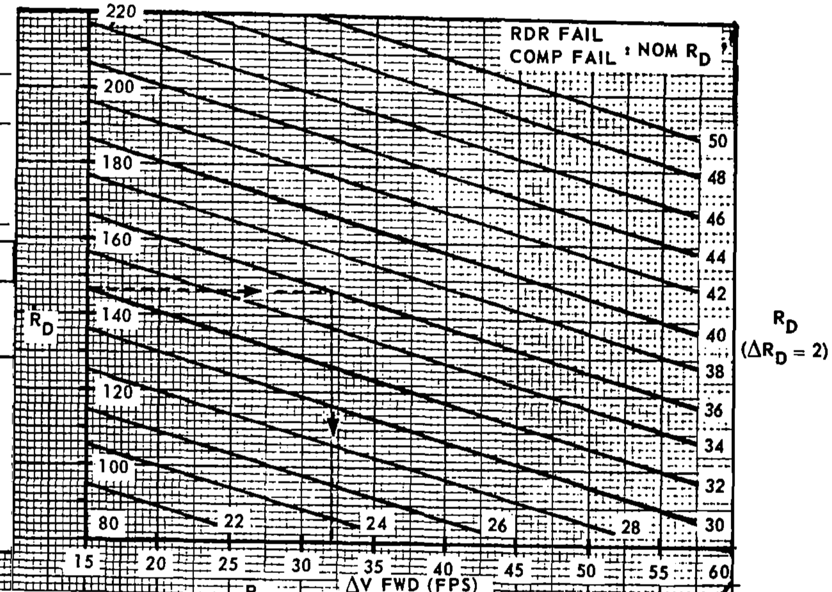
PT TIME	θ (96)	R (69)	R (35)		CL/R/C NOM		PLT
B ___:___	[.]	___.___	___.___	θ_D ___.___	24.6	α_D	___.___
+ 1:40	21.4	41.00	149.	$-\theta_B$ ___.___	21.4	$-\alpha_B$	___.___
C ___:___	___.___	___.___	___.___	$\Delta\theta$ [.]	3.2	↓	13.3
+ 1:40	22.9	38.55	148.		10.1	$-\Delta\alpha$	___.___
		R (36)				$\Delta\theta$	[.]
D ___:___	[.]	[.]	[.]				
+ 2:00	24.6	36.14	146.				

I [:]

+ :40

SET [:]

	G	IVI	ΔV	ΔT	USED
F	___	___	___	___	___
U-D	___	___	___	___	___
L-R	___	___	___	___	___



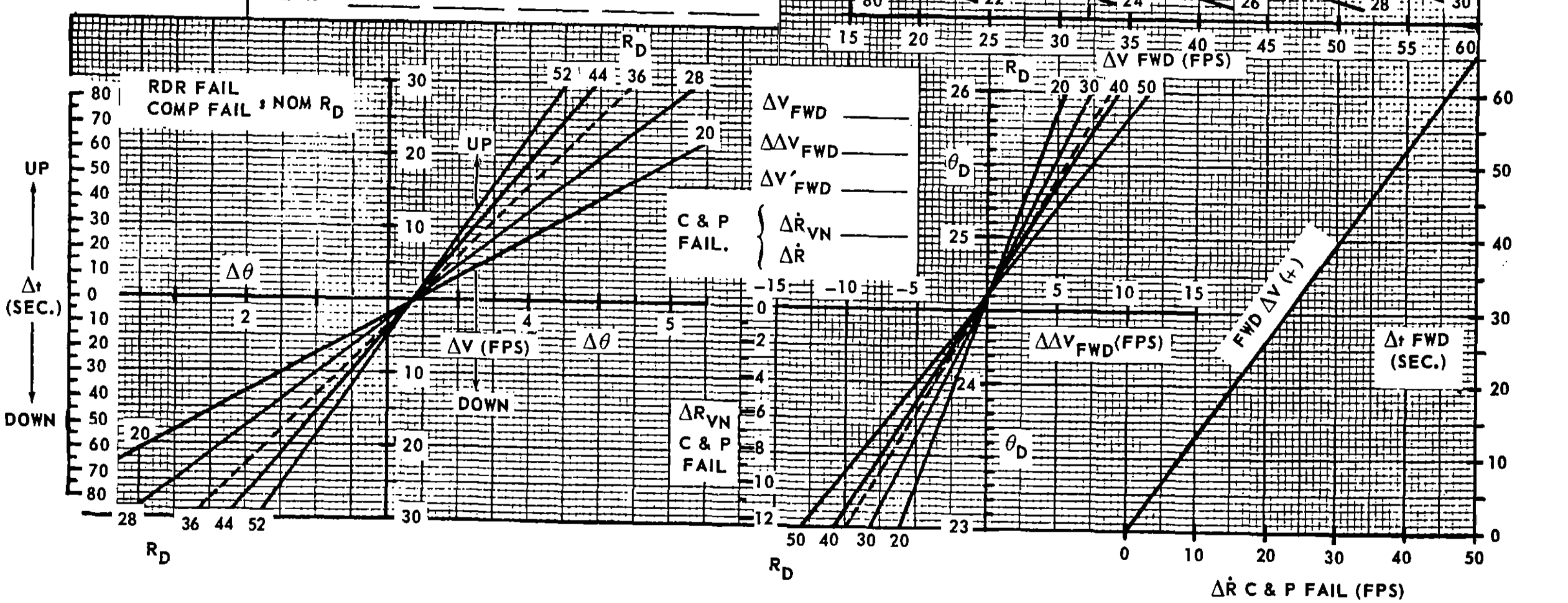


FIGURE 54 TERMINAL PHASE INITIATION

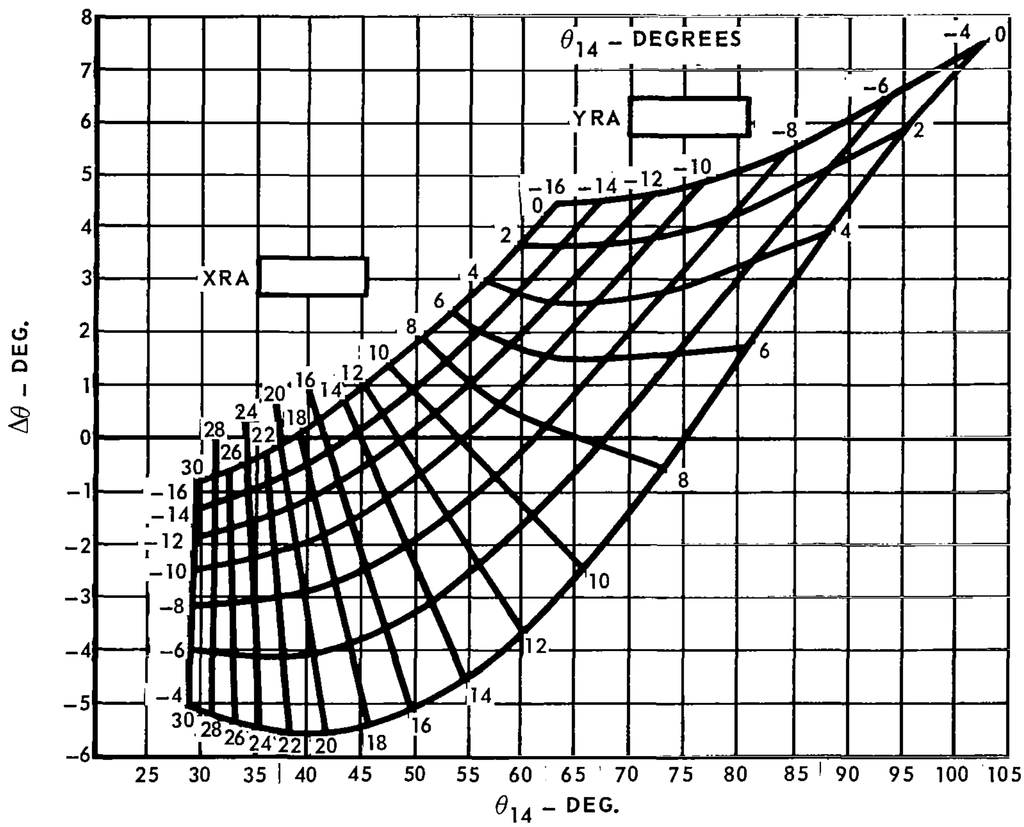
where the spacecraft vertical velocity relative to the Agena Target Vehicle (ATV) is zero). The position of relative apogee was predicted in the chart of Fig. 55 from a measurement of line-of-sight (LOS) angle and LOS angular rate taken at a fixed time from relative apogee. The chart assumed a nominal ATV orbit; however, dispersed ATV orbits were accommodated with other charts similar in format. In the case of a platform failure, the radar range and inertial LOS angle were used in a chart similar to that of Fig. 55 to predict the relative apogee point. The TPI maneuver consisted of a forward component, ΔV_X, and an up-down component, ΔV_Y, which were computed from the X_{RA} and Y_{RA} at relative apogee in the chart of Fig. 56.

This chart was derived from the Clohessy-Wiltshire equations based on a central angle to rendezvous of 120 degrees and a nominal target orbit. To account for ATV orbit dispersions the quantities ΔV_A and δV_Y were added to the ΔV_X and ΔV_Y, respectively. The ATV dispersion corrections were ground computations based on one orbit of ATV tracking and were relayed to the crew prior to spacecraft lift-off. The TPI maneuver midpoint occurs 90 sec prior to relative apogee establishing a 120 degree central angle transfer trajectory.

Two vernier corrections along and normal to the LOS were planned following TPI, the first 12 min from the TPI midpoint and the second 12 min after the first. The vernier corrections in previous rendezvous missions relied on nominal values for range and LOS angle in case of a closed-loop component failure, i.e., radar, platform, computer. However, because of the possibility of a large dispersion in the position at TPI on Spacecraft 11, the nominal values of range and LOS angle were supplied as a function of the X_{RA} and Y_{RA} at relative apogee as shown in Fig. 57.

An out-of-plane IVAR correction was planned at insertion to reduce the out-of-plane velocity error to that of the IGS. In addition, the navigated out-of-plane position at insertion was to be read from the computer at insertion and used to compute a velocity correction to be applied 90 degrees of orbit travel after insertion when the position dispersion propagated into a velocity dispersion. These two corrections reduced the out-of-plane errors to roughly those of the IGS. Another out-of-plane correction was to be computed for use at TPI in case of closed-loop failure. This correction was based on a measurement of out-of-plane angle to the target and the predicted position at relative apogee. A complete description of the procedures and flight charts utilized during Spacecraft 11 primary rendezvous is contained in G & C design notes 339 and 400. A description of the significant events occurring during engineering evaluation and crew training is given in G & C design note 410.

A measure of procedures and flight chart effectiveness for Spacecraft 6 through 11 primary rendezvous missions is presented in Table 34, which provides a comparison of the closed-loop, back-up, and ground computed solutions for the TPI maneuver during the primary rendezvous. The close agreement between the closed-loop solutions and those available from the flight charts indicates that rendezvous could have been accomplished onboard with a high probability of success on any of the missions with proper ground control up to the TPI maneuver, had a closed-loop failure occurred. Due to a



COMP FAIL ONLY

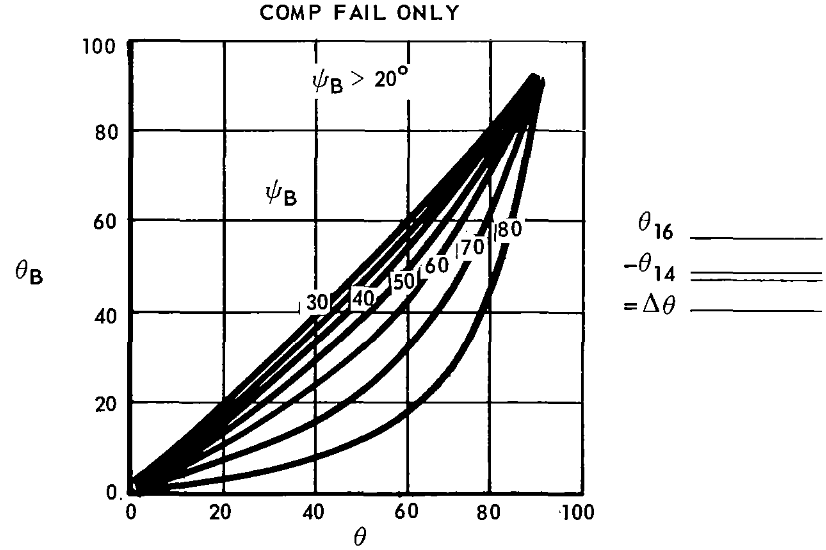


FIGURE 55 $\Delta V_A = 0$ CL, R AND C FAILURE

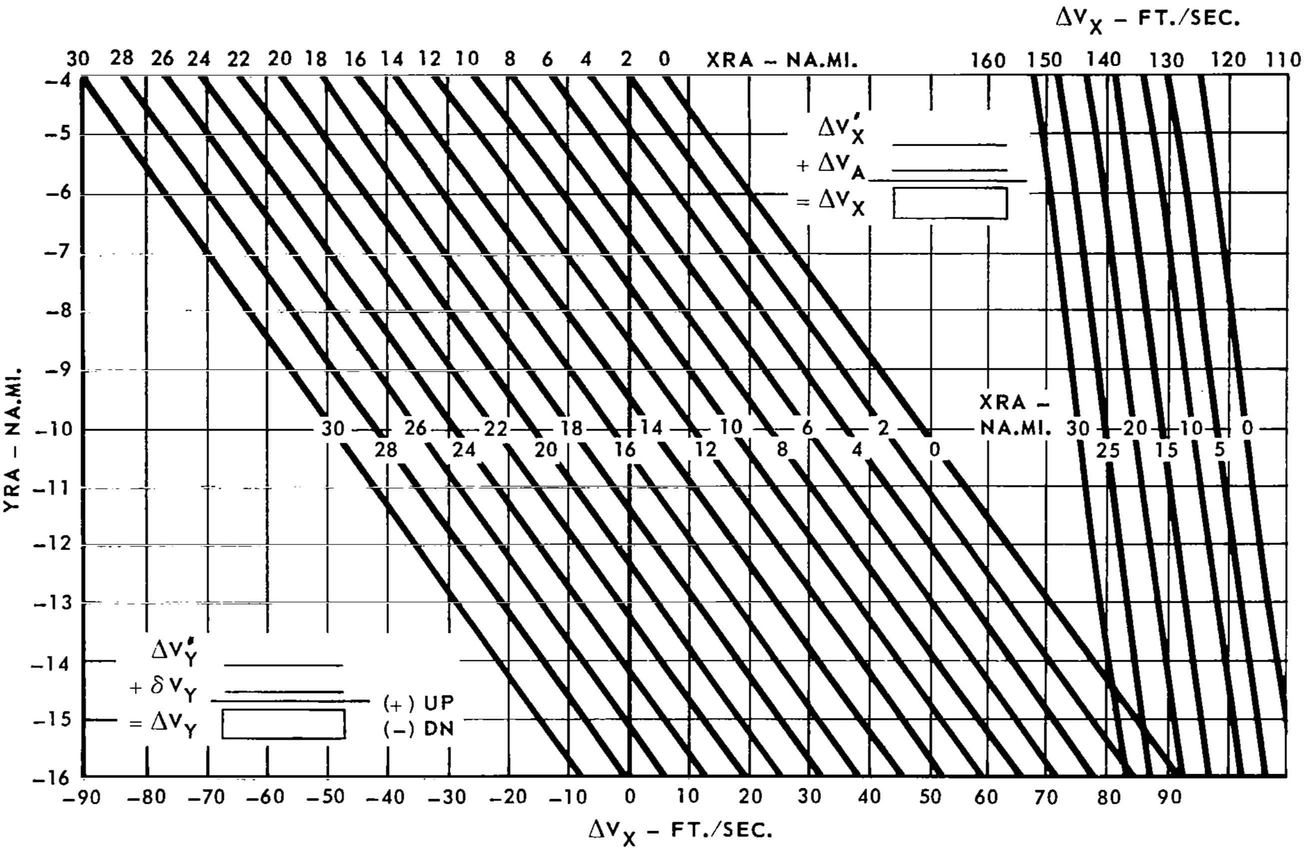


FIGURE 56 ΔVY, ΔVX

radar transponder failure on the Spacecraft 12 mission, back-up procedures were employed.

C. Mission Rendezvous Results - The following is discussion of the significant flight results including problems which arose during the primary rendezvous of Spacecraft 6 through 12 missions:

Spacecraft 6 - The closed-loop TPI maneuver was performed slightly later than nominal to insure that the spacecraft would be below and forward of the target at final rendezvous and that braking would be late, providing more daylight during final approach.

The closed-loop vernier corrections were displayed normally, agreed with available back-up solutions, and were applied. No anomalies occurred during the nulling of the small line-of-sight rates and a nominal braking schedule was followed to station-keeping. The crew commented that the flight charts should be simplified and should include a wider range of trajectory variables.

Spacecraft 8 - The radar referenced flight director indicator (FDI) needles were sonewhat erratic on this mission prior to TPI. This problem decreased with range but did result in an erroneous back-up TPI maneuver which

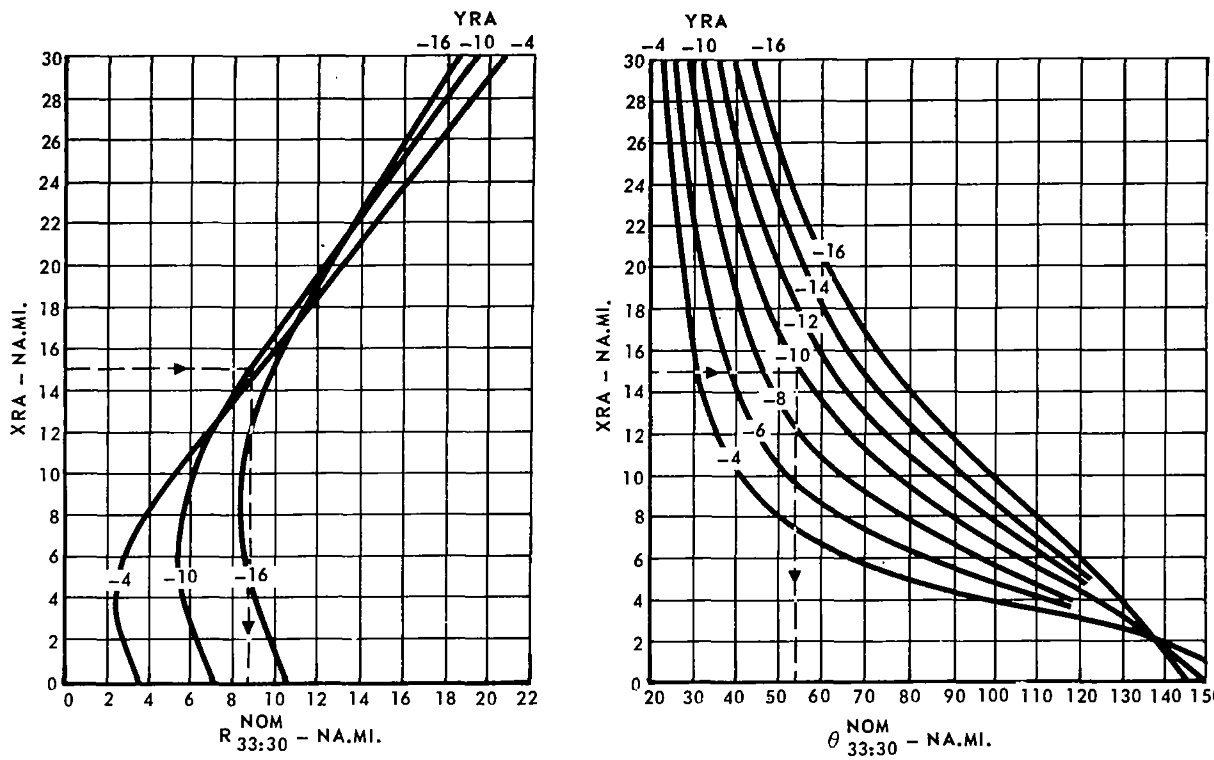


FIGURE 57 A1 (RNOM, θNOM)

called for a large down correction. The closed-loop TPI correction agreed fairly well with the ground value and was applied. The following procedural changes were recommended as a result of the flight:

1. Visual boresighting with the reticle should be used in preference to nulling the FDI's in radar reference.
2. An antenna select command should be sent via the encoder whenever lock on is first achieved.
3. Radar angle data should be locked out of the computer via the MDIU when the radar referenced FDI needles vary more than one degree from null when on visual boresight.

A post-flight evaluation of Spacecraft 8 primary rendezvous mission is contained in G & C design note 372.

Spacecraft 9 - No major problems were encountered with the closed-loop rendezvous system or with the back-up calculation requirements for the TPI and vernier corrections. Very small line-of-sight rates were observed during the terminal rendezvous indicating efficient trajectory corrections were applied.

Spacecraft 10 - The terminal phase of the initial rendezvous was normal except for the line-of-sight nulling and braking maneuvers. Apparently procedural problems in switching computer modes and obtaining inertially

TABLE 34 COMPARISON OF TPI SOLUTIONS FOR GEMINI INITIAL RENDEZVOUS MISSIONS

GEMINI MISSION	CLOSED-LOOP SOLUTION	BACK-UP SOLUTION	GROUND SOLUTION
VI	31 FWD 4 UP	32 FWD 2 UP	32 FWD 2 UP
VIII	26 FWD 3 DWN	33 FWD 0*	32 FWD 2 DWN
IX	26 FWD 8 UP	24 FWD 0	27 FWD 1 UP
X	41 FWD 1 UP	41 FWD 4 DWN	34 FWD 6 DWN
XI	140 FWD 27 DWN	140 FWD 22 DWN	140 FWD 17 DWN
XII	18 FWD** 5 DWN	22 FWD 3 DWN	23 FWD 3 DWN

*ADJUSTED FOR ERRONEOUS RADAR ANGLE DATA OBSERVED DURING THE GEMINI VIII FLIGHT.
**RECONSTRUCTED FROM POST FLIGHT DATA.

referenced FDI needles delayed early line-of-sight nulling. Both in-plane and out-of-plane line-of-sight rates built-up such that thrust was insufficient to stop the rates. Continuous thrust application resulted in an approach to the target, but with excessive propellant consumption.

Spacecraft 11 - Was inserted as targeted and an insertion correction, consisting of a 41 fps horizontal in-plane IVAR maneuver and a five fps radial maneuver computed onboard, placed the position of relative apogee 18.9 nautical miles behind and 8.6 nautical miles below the target. This was very near the desired point of 15 nautical miles behind the target and 10 miles below it. The TPI computations were accurate as demonstrated by the onboard calculation of relative apogee, based on elevation angle measurements, which was 19.3 and 8.0 nautical miles behind and below the target. The required TPI thrust as calculated onboard using the flight chart was close to the closed-loop solution as shown in Table 34. The first vernier correction was displayed and applied normally; however, prior to display of the second closed-loop vernier, the radar referenced FDI needles did not agree with visual boresighting and the back-up solution was applied. It has been determined that an intermittent radar transponder failure resulted in erroneous angle data during the last vernier correction and hence the back-up action taken by the crew was justified. Line-of-sight rate nulling using a stellar inertial reference was employed and range and range rate data were available during the braking maneuver.

Spacecraft 12 - A radar transponder failure on this mission resulted in the use of back-up rendezvous procedures. The onboard computed solution for TPI was very close to the ground computed value as shown in Table 34. Four back-up intermediate corrections were applied instead of the usual two closed-loop corrections. The velocity change requirements for these corrections was low (total of about 10 fps) and the propellant required for braking was moderate. This rendezvous verified the accuracy and dependability expected from the back-up procedures; however, this occurred after almost perfect prerendezvous maneuvers.

Rerendezvous. - Several of the Gemini missions have included more than one rendezvous. A passive rerendezvous from an equiperiod orbit was planned for Spacecraft 8 but not completed because of a spacecraft anomaly. This same rerendezvous was achieved by Spacecraft 9 and, in addition, a rerendezvous from above was conducted from a coelliptical approach. The Spacecraft 10 mission featured a rendezvous with the Agena 8 Target Vehicle following the initial rendezvous with the Agena 10 Target Vehicle. As on the primary rendezvous, the approach was coelliptical, but the terminal transfer was like that of the Spacecraft 9 passive rendezvous. The Spacecraft 11 mission included a rerendezvous from a "stable orbit" (i.e., zero differential altitude and a constant trailing displacement before initiation). A more refined version of this rendezvous was planned for Spacecraft 12 but was not included in the final mission objectives. The following paragraphs briefly describe the flight charts used in each of these rerendezvous operations and relate significant data obtained during the engineering evaluation and crew training sessions held at the McDonnell hybrid simulation facility.

A. Passive Rendezvous - As mentioned previously, three varieties of the passive rendezvous were performed. The first of these was rendezvous from an equiperiod orbit and is described here. The other two, referred to as "dual" rendezvous and "stable orbit" rendezvous are described in subsequent sections. The separation for the equiperiod passive rendezvous was initiated when the spacecraft made a 20 fps upward radial velocity change. This maneuver was executed ten min before sunset and resulted nominally in an orbit with relative altitude variations of ± 2.8 nautical miles and a maximum horizontal separation of 11.4 nautical miles. About one-half orbit after the separation maneuver, a nominally zero horizontal adjust maneuver was applied based on observation of the time at which the target vehicle appeared to be on the spacecraft local horizontal. Sunrise nominally occurred about two min after this correction, and a short time later the spacecraft was rolled heads down to shield the spacecraft windows from the sun. The crew then was prepared to determine the TPI maneuver.

The time and magnitude of the TPI maneuver were determined from the time increment between superposition of the target and horizon for sextant settings of 43 and 47 degrees as shown in Fig. 58.

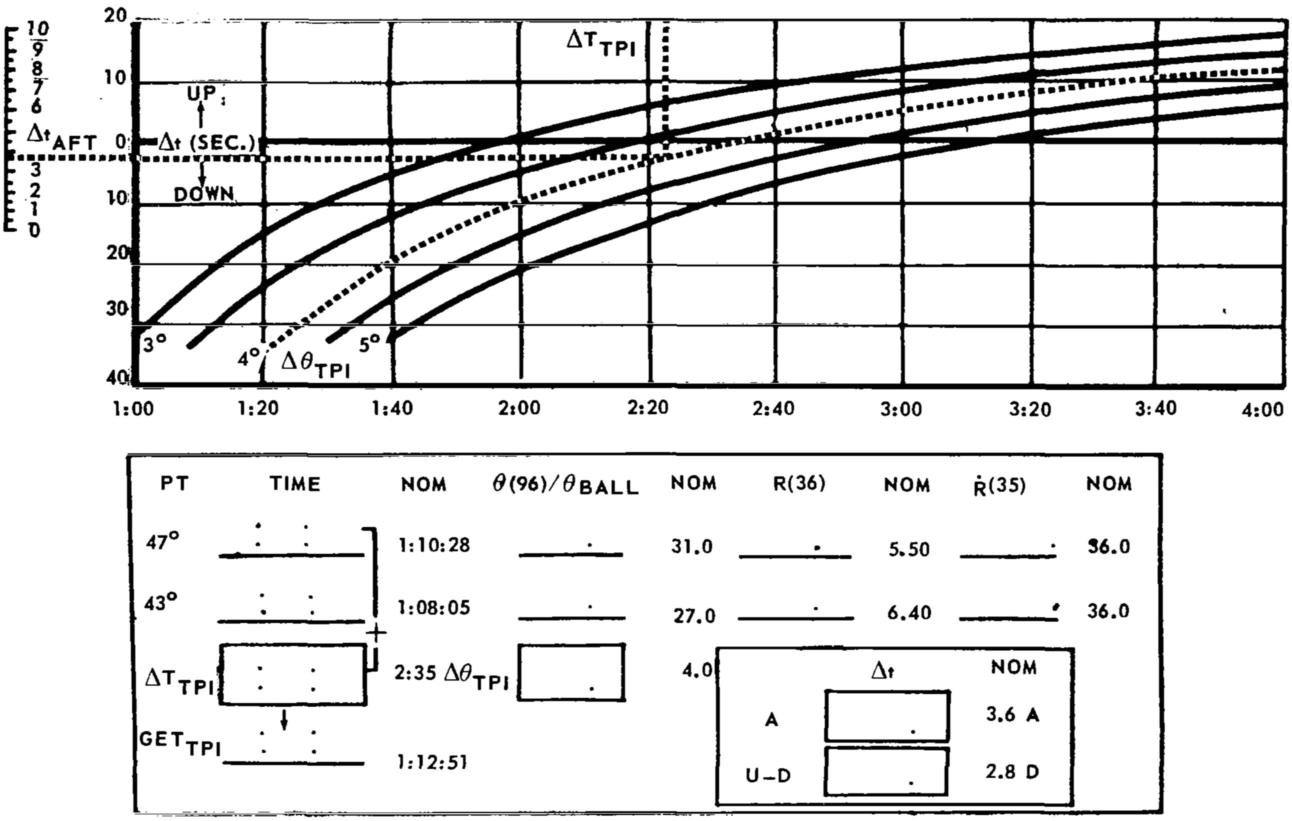


FIGURE 58 TERMINAL PHASE INITIATION

While this correction consisted of a component along and normal to the line-of-sight, the correction along the line-of-sight was just the nominal value. This was necessitated by the absence of any range data. The component normal to the line-of-sight consisted of the sum of the nominal value and a correction computed from the actual angle measurements. An alternate determination of the maneuver was obtained from the MDIU readout of the pitch angles at the sextant superposition times. Both of these determinations normally were employed but in the event of a platform failure, only the sextant was used and with a computer failure, pitch angle was read from the attitude ball to replace the MDIU information. In any case, the result was an 80-degree transfer and two intermediate corrections were made in a nearly identical manner.

Following the intermediate corrections, the command pilot controlled line-of-sight angular rates in the ORB RATE platform mode. The pilot used additional charts to determine burn times for nulling the line-of-sight angular rate and (with the help of the sextant) for braking.

An engineering evaluation of the proposed charts and procedures was conducted from 5 to 14 January 1966. Several procedural improvements resulted from the evaluation, especially in the technique for applying the separation maneuver. One area which proved to be difficult was that of nulling the line-of-sight angular rates for the 80-degree transfer.

Crew training sessions were held for the Spacecraft 8 and 9 crews during the weeks of 24 January 1966 and 30 Match 1966, respectively. A summary of Spacecraft 8 and 9 passive rendezvous engineering evaluation and crew training is contained in G & C design note 382.

Several problems were encountered when this rendezvous was actually conducted on the Spacecraft 9 mission. The TPI time increment which was to have been measured by observing the superposition of the target and the horizon on the sextant set sequentially at 43 and 47 degrees was not made since the sextant was erroneously set to 47 degrees initially. Instead a time increment was measured by observing pitch elevation angle from the MDIU at modified times. The TPI maneuver was calculated to be two sec of aft and five sec of down thrust which approximated the nominal values of 3.6 sec aft and 2.8 sec down. The first vernier correction also was based on MDIU pitch elevation angles. However, the second vernier correction was calculated following the prescribed procedures and the sextant measurements. Both vernier corrections were near the nominal value of zero. Braking thrust calculations were based on a nominal time schedule rather than on sextant measurements. The major problems in sextant usage involved tracking the target in the sextant's small field of view and reacquiring the target after changing a sextant setting. The crew recommended that platform angles be the primary angle measurement source and that the sextant be reserved for platform failure cases.

B. Dual Rendezvous - The second passive rendezvous performed was the Spacecraft 10 - Agena 8 dual rendezvous. Following a rendezvous with Agena 10, a series of docked configuration maneuvers was made to establish gross phasing and following this, a series of undocked Gemini maneuvers was made for fine phasing. The result of all these maneuvers was a coelliptical approach with a differential altitude of seven nautical miles and a phasing such that a line-of-sight elevation angle of 33 degrees was reached at 27 min after sunrise. A 16 min platform alignment began at sunrise and following this alignment, the spacecraft was rolled heads down to shield the windows from the sun. The crew then was prepared to determine the TPI maneuver.

The data used for determining the TPI maneuver was nominally the line-of-sight elevation angle change over three min. To obtain this data, visual acquisition of the target was necessary. The spacecraft pitch angle was controlled to preselected angles as a function of time to maximize the probability of early visual acquisition. Upon acquisition, the spacecraft was controlled to visual boresight and pitch elevation angles (θ_1 and θ_2) were read from the computer three min apart. A subtraction provides the $\Delta\theta$

over the three min interval and this information was used in Fig. 59 to determine components of velocity change along and normal to the line-of-sight for an 80 degree transfer. Contingency procedures were developed for the case of late visual acquisition (using one angle measurement only or, if very late, using ground computed maneuvers) and back-up procedures were developed in the event of a platform or computer failure. Elevation angle was read from the attitude ball in the computer failure case and target-to-horizon sextant angle measurements were used in the event of a platform failure. Two intermediate corrections were made following TPI. These corrections were computed in the same manner as the TPI maneuver except that the line-of-sight component was not determined.

Another procedure that proved useful for the dual rendezvous was the use of the braking chart. This chart provided an estimate of the range rate at a range of two nautical miles, the time from TPI to a range of two nautical miles, and the terminal approach angle. This information was based on the line-of-sight elevation angle at the second pre-TPI measurement. The charts used for braking and line-of-sight control for the equiperiod passive rendezvous were not employed here since the terminal approach velocity was so high that sextant measurements were ineffectual.

An engineering evaluation of these charts was conducted during the first week of June 1966. As anticipated, line-of-sight control during the terminal approach was a problem but eventually was mastered. Several modifications to the charts resulted from the evaluation.

From 6 to 10 June 1966, 54 runs were made during the crew training session. The training resulted in an increase in the TPI-to-sunset minimum time from 25 to 28 min. This allowed more time for braking and line-of-sight control before entering darkness and was the result of a number of fly-bys caused by early darkness entry. A summary of Spacecraft 10 dual rendezvous evaluation and crew training is contained in G & C design note 390.

The only problem encountered during the dual rendezvous was the attempt to estimate ranges greater than one nautical mile using the sextant. This apparently was caused by the end-on approach to the target. At ranges less than one nautical mile stadiametric ranging with the sextant was possible and the normal braking schedule was followed.

C. Rendezvous from Above - A second rendezvous was performed during the Spacecraft 9 mission which was similar to the Spacecraft 8 coelleptic primary rendezvous. The difference was that instead of being displaced 15 nautical miles below and 142 nautical miles behind the target following circularization, the spacecraft was 15 nautical miles above and 126 nautical miles ahead of the target. The flight charts used to compute the TPI maneuver and vernier corrections during the rendezvous from above were the corresponding Spacecraft 8 flight charts rescaled to fit the Spacecraft 9 rendezvous.

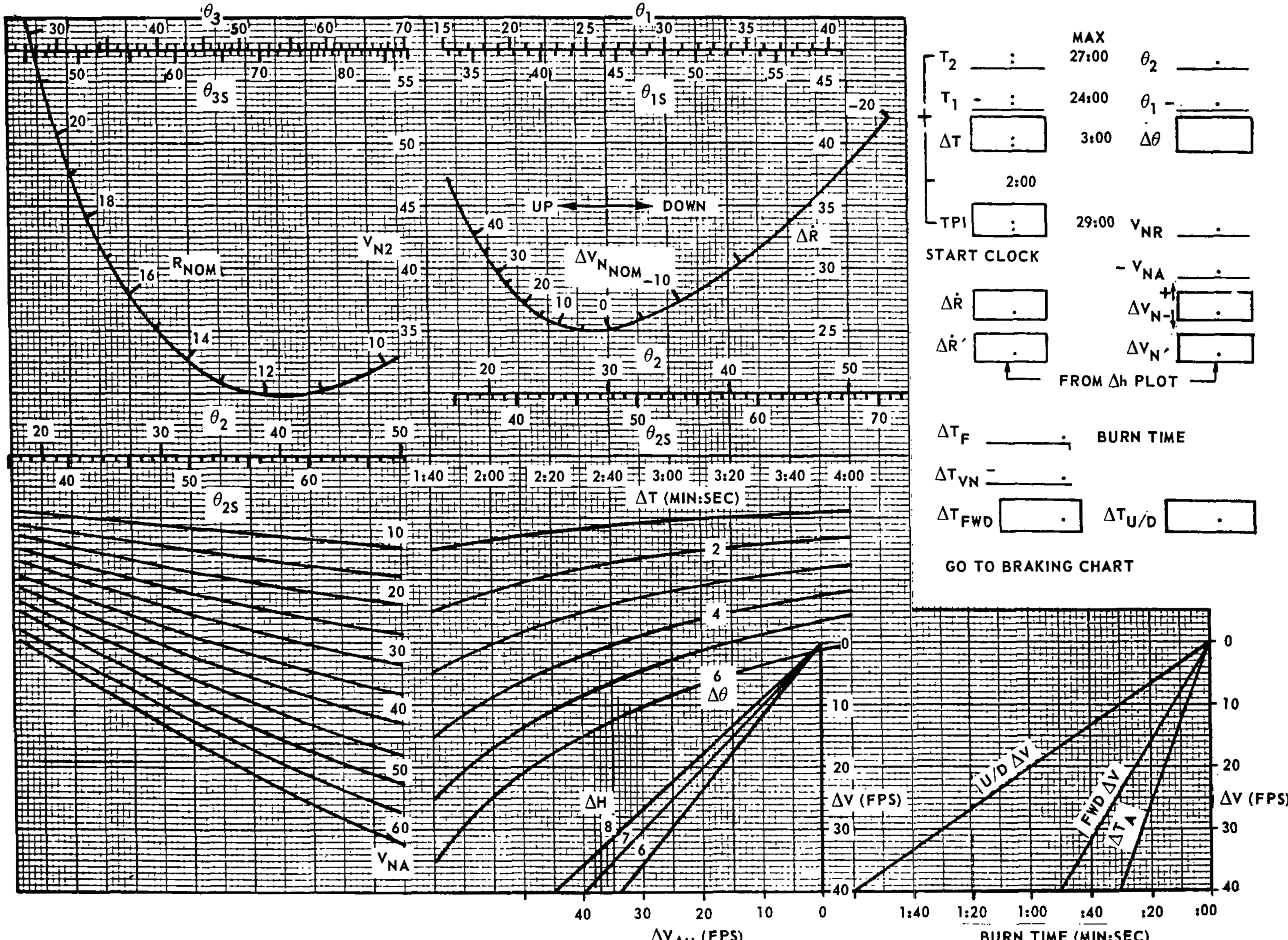


FIGURE 59 TERMINAL PHASE INITIATION

The following modifications were required:

1. Pitch angles were modified to include a 180 degree change in the line-of-sight angle and errors in the platform torquing rate which was preset for a spacecraft orbit 15 nautical miles below the target.

2. The computer constant for target orbital radius, R_T, was modified to provide minimum total ΔV for the rendezvous from above.

3. The charts were modified to indicate a thrust in the opposite direction for the same set of trajectory conditions as on the Spacecraft 8 rendezvous since the position of the spacecraft relative to the target vehicle was reversed on the Spacecraft 9 rendezvous from above.

4. An additional curve was included which provide up-down thrust in case of a platform failure during the vernier corrections. This was necessary because of the lack of a stellar inertial reference when the target vehicle was viewed against the earth background.

A summary of evaluation and crew training for the rendezvous from above is contained in G & C design note 376.

The flight data from the Spacecraft 9 rendezvous from above indicates that radar lock-on to the ATDA was achieved at 70 nautical miles. The closed-loop TPI solution was 19 FWD and 4 DWN which did not agree exactly with the back-up and ground solutions which were 16.5 FWD and 3.0 UP and 16.5 FWD and 0.3 UP, respectively. The back-up solution was applied since the closed-loop solution appeared to be erroneous. Only the closed-loop vernier corrections were applied since the radar angle data displayed on the FDI needles contained transients (which were anticipated with the rate stabilized ATDA) making $\Delta\theta$ measurements for back-up corrections impractical. The target was first seen in reflected moonlight at a range of 20 nautical miles but was lost one min later when daylight occurred precluding visual tracking of the target for back-up calculations. Visual observation again was obtained at three nautical miles, but the target was intermittently lost against the varying terrain features. Line-of-sight rate nulling was done in the orbit rate platform mode by noting small angular changes over small time intervals. Although visibility problems and unsteady FDI needles caused the indicated corrections to be inaccurate and scattered, trends were established for gross line-of-sight nulling. It was apparent that accurate radar data was a necessity for an efficient terminal phase of rendezvous from above both because of the visibility problems during back-up calculations and because of the need for accurate angle data in lieu of an inertial reference for line-of-sight rate nulling.

D. Stable Orbit Rendezvous - The Spacecraft 12 passive rerendezvous was a stable orbit rendezvous characterized by a zero differential altitude and a constant trailing displacement prior to TPI of 15 nautical miles. A nominal TPI maneuver of 5.5 fps aft and 2.9 fps down produced a transfer trajectory with intercept in 330 degrees of target central angle travel. Five nominally zero vernier corrections were planned at selected central angles to go to rendezvous. The vernier correction components normal to the line-of-sight were based on the deviation of measured elevation angles from nominal angles determined as a function of the actual relative position at TPI. The vernier correction component along the line-of-sight for the first vernier was computed from the elevation angle, nominal range, and an empirical relationship. The line-of-sight vernier component was omitted for the second and third corrections due to the lack of range data, but, this component was included in the last two corrections as sextant ranging information became available.

A stable orbit passive rerendezvous similar to that planned for Spacecraft 12 was performed during the Spacecraft 11 mission. The difference in the trajectory profile from the Spacecraft 12 rerendezvous was a 25 nautical mile trailing displacement instead of 15 nautical miles. A ground supplied TPI maneuver established the transfer trajectory and the fourth vernier correction chart of Spacecraft 8 scaled by one-third was used to verify a ground supplied vernier correction.

The Gemini XII-Agena Tethered Vehicle Gravity Gradient Operation - At 47 hr and 23 min into the Gemini XII mission, the spacecraft undocked from the Agena Target Vehicle and deployed a 100 ft tether between the vehicles. The objectives of this maneuver was to establish a mode of stabilization of the tethered system using the gradient of the earth's gravitational field, as described and analyzed in Structural Dynamics Design Note 31. Attempts by the command pilot to orient the spacecraft above the Agena with the proper initial relative velocity were impeded by OAMS thruster problems, but at 48:21 into the mission the crew reported the tether was becoming "fairly taut." At 49:28, the crew felt that a stabilized mode had been established and turned off the Agena ACS. A few minutes previous to this, the spacecraft OAMS was shut down and also the crew had reported a "tight tether." For the following 2-1/2 hr, the tethered system drifted without control system activity to determine to what extent the gravity gradient had stabilized the system about the local vertical.

Data Evaluation - The only data available for evaluating the gravity gradient operation are the spacecraft inertial platform gimbal angles (θ, ψ, ϕ). Only two of these three Euler angles are of interest here - θ (pitch) and ψ (yaw). These determine the orientation of the spacecraft longitudinal axis and should, therefore, give some indication of the orientation of the tether (or more precisely, the line between the centers-of-gravity of the two vehicles). The (0,0,0) position corresponds to the spacecraft longitudinal axis oriented so as to be in both the local horizontal and orbital planes with the spacecraft small end forward and the crew sitting upright. θ is measured first and is positive when the spacecraft is pitched up. ψ is then applied about the spacecraft yaw axis and is to the crew's right. The nominal gravity gradient orientation is therefore (270 degrees, zero degrees, ϕ).

SDDN 31 details the histories of θ and ψ from the time the Agena ACS was shut down till the Gemini docking bar was purposely blown thus terminating the tether operation.

Following the Agena ACS shutdown, the spacecraft is seen to flip over from the horizontal forward-facing position to the downward pointing position in a period of about ten min. Neither this flip nor the one occurring at 50:30 was obvious to the crew, apparently because of the associated rolling motion. Following the first spacecraft flip, a period of regular oscillatory motion occurs at a frequency readily identified from SDDN 31 with the spacecraft moment-of-inertia and tether elasticity interaction. The second spacecraft flip is seen to be coincident with a fuel cell purge which may have been a contributing factor. Despite the occasional erratic motion of the spacecraft, the crew stated that at no time after Agena ACS shutdown did the tether make an angle as great as 90 degrees to the local vertical, i.e., the total tethered system remained stabilized. The dashed line faired through the data indicates this was so. The librational motion of the total system about the local vertical is approximated by this line. It shows the system to be executing a counterclockwise coning motion when viewed from above. The pitch and yaw periods are on the order of one hour as expected with yaw somewhat shorter than pitch due to centrifugal force effects shortening the yaw period.

Conclusion - From crew observation interpretation of spacecraft angular motions, it is concluded that the tethered system was stabilized about the local vertical for a period of about 2-1/2 hr (1-2/3 orbits). The motions observed were in general agreement with the prediction of SDDN 31. A comprehensive evaluation of the Gemini XII-Agena tethered vehicle operation is given in Structural Dynamics Design Note 32.

Re-entry Procedures Development

Re-entry Monitoring Procedures. - The re-entry monitoring procedures used on Gemini missions provided the flight crews with a means for evaluating information on the operational status of the primary closed-loop guidance system. Based on this information, a decision was made onboard as to whether primary or back-up guidance should be employed. Briefly, the primary guidance procedure was to maintain the modulated bank angle commanded by the computer by nulling the displayed bank angle error. The back-up guidance procedure was to establish a ground supplied constant bank angle which nulled downrange error to the target and reverse the bank angle at a time specified by ground to null crossrange error to the target. The monitoring procedures used the flight director indicator (FDI) range error and bank error display information generated by the re-entry mode of the onboard computer.

The first event monitored was the time of arrival at a navigated altitude of 400,000 ft which gave the first indication of the status of the computer guidance and navigation functions. This event was signaled onboard by the initial deflection of the FDI bank error display in response to a computer commanded constant bank angle. An arrival time within the tolerance of +40 and -25 sec from the ground computed time of arrival indicated acceptable operation. However, a time outside the tolerance indicated that a system failure or degradation in navigation accuracy had occurred which might preclude the use of primary guidance. From the time of arrival at 400,000 ft to guidance initiation, which occurred at approximately 300,000 ft, the back-up bank angle was maintained. This was done to avoid having to switch from the computer commanded bank angle to the back-up bank angle at guidance initiation when further information on the operational status of the computer was available.

The second event monitored was the initial downrange error needle deflection at guidance initiation. An acceptable deflection was within ± 50 nautical miles of the ground computed value relayed following retrofire. If the deflection was within tolerance the back-up bank angle was ignored and the closed-loop bank angle commands maintained. If the initial deflection was outside the established limits the back-up bank angle was held and reversed at the designated time. The needle deflection was monitored continuously during re-entry to see that the range error was being driven to null. Detailed descriptions of the above procedures for each spacecraft are G & C design notes 394, 358, 353, and 334.

Touchdown Predict Procedures. - The touchdown predict procedures involved estimating a trial retrofire time which would result in a computer-predicted touchdown location within a desired landing area.

This trial retrofire time was a time-to-go to retrofire and was estimated from the difference in the longitude of the desired landing area and the longitude of the spacecraft at the current mission time, assuming the spacecraft travelled four degrees of longitude per min. If the predicted touchdown location was not desirable, the trial retrofire time was adjusted by calculating a correction time from longitude difference between the displayed and the desired longitude. Prediction was continued until the predicted landing site was acceptable at which time the retrofire initial conditions were transferred to the re-entry mode by inserting an accept code into the computer. A detailed description of the touchdown predict procedures is contained in G & C design note 348.

EXPERIMENT SUMMARY REPORT

A variety of scientific, medical, technological, and engineering experiments were conducted on Gemini missions to extend man's knowledge of space and to develop further the ability to sustain life in space.

The McDonnell Company was responsible for integrating experiment equipment into the Gemini Spacecraft, while NASA was responsible for the design, construction, and qualification of experiment equipment before flight and the analysis of the data after flight.

This summary describes the experiment objectives, experimental equipment, and the interface requirements in the spacecraft. Table 35 lists the spacecraft effectivity for each experiment.

Since experiment equipment was the responsibility of NASA, no attempt is made to describe equipment design and evaluation, state-of-the-art advances, qualification program results, reliability and quality assurance program results, results of flight missions; or any problems associated with the overall program. The results of these experiments are reported in Gemini Mid-Program Conference (NASA SP-121), and Gemini Summary Conference (NASA SP-138), and in Manned Spaceflight Experiments Interim Reports (MSC-TA-R-67-1, 2, 3, dated March, May, and August 1967 respectively).

EXPERIMENT D-1 (BASIC OBJECT PHOTOGRAPHY)

The objective of this experiment was to investigate the ability of man to acquire, track, and photograph space-borne objects, such as the rendezvous evaluation pod (REP), the Agena Target Vehicle, and natural celestial bodies.

The following equipment was required: a Zeiss Contarex camera, a 200 mm Nikkor f/4 lens, a Questar 1400 mm lens, three film packs, a telescopic sight, a periscope viewer, a window and lens bracket assembly, a film transport adapter, an optical sight, a photo event indicator, and a Gemini voice tape recorder.

The experiment equipment interface with the spacecraft was accomplished at the RH hatch window by mounting the ACFE window and lens bracket assembly at three points on the window sill. An adapter was provided on the window bracket to permit the interchange of the two lenses. All equipment was stowed in the cabin.

Electrical power was provided from the main bus for the GFAE photo event indicator.

EXPERIMENT D-2 (NEARBY OBJECT PHOTOGRAPHY)

This experiment, an extension of Experiment D-1, was intended to demonstrate the flight crew's proficiency in acquiring high-resolution photographs

TABLE 35 DOD/NASA EXPERIMENTS – NASA – GEMINI MISSIONS

EXP. NO.	EXPERIMENT TITLE		MISSION NUMBER									
		TDA (T)						3	5*	1A	6	7A
		S/C (X)	3	4	5	6A	7	8	9A	10	11	12
D–1	BASIC OBJECT PHOTOGRAPHY			X	X	X						
D–2	NEARBY OBJECT PHOTOGRAPHY				X	X						
D–3	MASS DETERMINATION							X			X	
D–4	CELESTIAL RADIOMETRY				X		X					
D–5	STAR OCCULTATION NAVIGATION						X			X		
D–6	SURFACE PHOTOGRAPHY			X	X							
D–7	SPACE OBJECT RADIOMETRY				X		X					
D–8	RADIATION IN SPACECRAFT			X		X						
D–9	SIMPLE NAVIGATION			X			X					
D–10	ION-SENSING ATTITUDE CONTROL									X		X
D–12	ASTRONAUT MANEUVERING UNIT								X			X
D–13	ASTRONAUT VISIBILITY				X		X					
D–14	UHF/VHF POLARIZATION							X	X			
D–15	NIGHT IMAGE INTENSIFICATION							X			X	
D–16	POWER TOOL EVALUATION							X			X	
M–1	CARDIOVASCULAR CONDITIONING				X		X					
M–3	INFLIGHT EXERCISER			X	X		X					
M–4	INFLIGHT PHONOCARDIOGRAM			X	X		X					
M–5	BIOASSAYS – BODY FLUIDS					X	X	X	X	X		
M–6	BONE DEMINERALIZATION			X	X		X					
M–7	CALCIUM BALANCE STUDY						X					
M–8	INFLIGHT SLEEP ANALYSIS						X					
M–9	HUMAN OTOLITH FUNCTION				X		X					
MSC–1	ELECTROSTATIC CHARGE			X	X							
MSC–2	PROTON-ELECTRON SPECTROMETER			X			X					
MSC–3	TRI-AXIS MAGNETOMETER			X			X			X		X
MSC–4	OPTICAL COMMUNICATION						X					
MSC–5	LUNAR UV SPECTRAL REFLECTANCE									X	X	X
MSC–6	BETA SPECTROMETER									X		X
MSC–7	BREMSSTRAHLUNG SPECTROMETER									X		X
MSC–8	COLOR PATCH PHOTOGRAPHY									X		
MSC–10	TWO-COLOR EARTHS LIMB PHOTOS			X								
MSC–12	LANDMARK CONTRAST MEASUREMENTS						X			X		
S–1	ZODIACAL LIGHT PHOTOGRAPHY				X			X	X	X		
S–2	SEA URCHIN EGG GROWTH		X									
S–3	FROG EGG GROWTH							X				X
S–4	RADIATION AND ZERO-G ON BLOOD		X								X	
S–5	SYNOPTIC TERRAIN PHOTOGRAPHY			X	X	X	X			X	X	X
S–6	SYNOPTIC WEATHER PHOTOGRAPHY			X	X	X	X			X	X	X
S–7	CLOUD TOP SPECTROMETER				X			X				
S–8	VISUAL ACUITY				X		X					
S–9	NUCLEAR EMULSION							X			X	
S–10	AGENA MICROMETEORITE COLLECTOR							TX	TX*	TX	X	T
S–11	AIRGLOW HORIZON PHOTOGRAPHY								X		X	X
S–12	MICROMETEORITE COLLECTION								X	X		X
S–13	UV ASTRONOMICAL CAMERA									X	X	X
S–26	GEMINI WAKE MEASUREMENT									T	T	
S–29	LIBRATIONS REGIONS PHOTO										X	X
S–30	DIM-LIGHT ASTRONOMICAL OBJECT PHOTO.										X	
S–51	DAYTIME SODIUM CLOUD PHOTO.											X
T–1	REENTRY COMMUNICATION		X									
T–2	MANUAL NAVIGATION SIGHTINGS											X
		S/C (X)	3	4	5	6A	7	8	9A	10	11	12
		TDA (T)						3	5*	1A	6	7A

*ALSO ON TDA 4 (ATDA)

of the rendezvous evaluation pod (REP). In addition, the spacecraft was to perform station keeping within 60 ft of the REP and to make an in-plane maneuver around the REP to permit an all-aspect viewing. Subsequently, the spacecraft was to be maneuvered 500 ft aft for photographs with the 1400 mm lens.

The equipment used in this experiment was the same as that for Experiments D-1 and D-6. The experiment equipment interface with the spacecraft, including stowage provisions, was also the same as that for Experiments D-1 and D-6.

EXPERIMENT D-3 (MASS DETERMINATION)

The objective of this experiment was to investigate the feasibility of a direct-contact method of determining the mass of an orbiting vehicle. The procedure involved accelerating the Gemini Agena Target Vehicle by pushing it with the spacecraft. The mass of the target vehicle would be determined by measuring the incremental velocity change and the thrusting time.

No special equipment was required because the experiment employed the existing Gemini data acquisition system. No interface with the spacecraft was required.

EXPERIMENT D-4/D-7 (CELESTIAL RADIOMETRY AND SPACE OBJECT RADIOMETRY)

The objective of the D-4/D-7 experiment was to obtain spectral irradiance information about various celestial and terrestrial backgrounds, rocket plumes, and cold objects in space.

The equipment flown on Spacecraft 5 for this experiment was: a multi-channel radiometer, an IR interferometer/spectrometer, a cryogenic interferometer/spectrometer, an FM/FM transmitter, an electronic unit, a recorder electronics unit, a tape transport, and a UHF antenna.

On Spacecraft 7, the radiometer was slightly modified and different filters were employed. With these exceptions, the equipment was the same for Spacecraft 5 and 7.

The components, location, and the functions of the items of equipment were:

The tape recorder, used to record spectrometer and commutated data, was located in the left skid-well.

The IR spectrometer consisted of two separate detectors utilizing a common optical system to provide spectral radiance measurements of targets and earth background to the tape recorder and the telemetry transmitter.

The radiometer contained filters for measuring target radiation over selected spectral regions. The IR spectrometer, the cryogenic spectrometer, and the radiometer were mounted in the retrograde section of the adapter. Jettisonable doors were provided to protect the radiometer and the spectrometer sensor elements during the launch.

Power ON/OFF switches for the tape recorder, the IR spectrometer, the radiometer, the cryogenic spectrometer, and the telemetry transmitter were located on the instrument console in the spacecraft cabin.

The electronic unit, the telemetry transmitter, and the antenna were mounted in the equipment section of the adapter.

Spacecraft power was utilized to operate this experiment.

The experiment determined the threshold of sensitivity values in absolute numbers for earth and sky background radiation. Irradiance measurements of the sun, moon, stars, cloud formations, and active and passive earth objects were taken. Measurements on the rendezvous evaluation pod (REP) and on rocket engine plumes were made during the Gemini V mission. During the Gemini VII flight, measurements were made on the second stage of the launch vehicle.

EXPERIMENT D-5 (STAR OCCULTATION NAVIGATION)

The objective of this experiment was to determine the usefulness of the star occultation technique for space navigation. An astronaut was equipped with a photoelectric occult telescope to make time and intensity measurements of stars occulted by the edge of the earth's atmosphere. The procedure involved visually locating occulting stars and manually recording the appropriate times. The photometric attenuation and the time measurements data would be recorded and transmitted via telemetry, or else recovered after spacecraft re-entry for post-flight orbit computation and comparison with ground track data.

The experiment equipment consisted of a GFAE star occultation photometer and one telemetry channel. Interface with the spacecraft involved modifying the cushion liner in the aft centerline stowage container to accommodate the photometer. An electrical connection and two utility cords were provided in the spacecraft cabin, one cord supplying photometer power and the other transmitting telemetry signals.

EXPERIMENT D-6 (SURFACE PHOTOGRAPHY)

The objective of this experiment was to investigate the technical problems associated with acquiring, tracking, and photographing terrestrial objects.

The equipment and the equipment interface with the spacecraft, including stowage provisions, were the same as those for Experiments D-1 and D-2.

The procedure involved acquisition of the object by using the optical sight and the spacecraft attitude instruments, and tracking by means of the telescope sight, the optical sight, and the periscope viewer.

For the acquisition and visual tracking modes, the command-pilot would maneuver the spacecraft; for the telescope and periscope tracking modes, the pilot would control the spacecraft. Beginning with the acquisition of the terrestrial object by the crew, four photographs would be taken.

EXPERIMENT D-8 (RADIATION IN SPACECRAFT)

The objective of the D-8 experiment was to gather reliable data on the absorbed and total radiation doses penetrating the cabin of the Gemini Spacecraft. This information is highly relevant to the accuracy of future manned-space-mission planning.

The experiment equipment consisted of two active current-mode ionization chambers, five small passive dosimeter packets, three telemetry channels, and an electrical power cable for each ionization chamber. The portable tissue-equivalent ionization chambers were to measure the dose levels during: (1) the "nonanomaly" revolutions and (2) during spacecraft passage through the anomalous region of the inner Van Allen radiation belt.

The experiment equipment interface with the spacecraft consisted of attaching the ionization chambers and the dosimeter packets inside the cabin, and providing electrical power connections and the wiring for telemetry output.

EXPERIMENT D-9 (SIMPLE NAVIGATION)

The objective of this experiment was to prove the feasibility of spacecraft navigation by manual means during flight. Star and horizon sightings and measurements were to be made using a handheld space sextant. The goal would be a simplification of orbital determination mathematics so that the orientation and the size and shape of the orbit could be obtained with the sextant and an analog computer.

The experiment equipment consisted of only the handheld sextant. The interface with the spacecraft comprised provisions for stowing the sextant within the cabin.

EXPERIMENT D-10 (ION-SENSING ATTITUDE CONTROL)

The objective of the D-10 experiment was to evaluate the use of ion sensors in determining spacecraft attitude.

The experiment equipment consisted of two ion sensors and two extendible booms. The booms were used to elevate the ion sensors above the spacecraft to measure variations of the ion flow.

The pitch sensing system consisted of two planar electrostatic analyzers mounted 90 degrees apart on an extendible boom in the spacecraft retroadapter. Each analyzer was set at an angle of 45 degrees from the plane of zero pitch. The yaw sensing system was identical to the pitch sensing system, except that the two planar electrostatic analyzers were mounted 40 degrees from the plane of zero yaw. The data acquired by the sensing systems were tape recorded and transmitted on ten high level and four lo-level telemetry channels of the adapter TM multiplexers.

Spacecraft power operated the experiment equipment. A power ON/OFF switch and an antenna EXTEND/SAFE switch were provided on the instrument console in the spacecraft cabin.

EXPERIMENT D-12 (ASTRONAUT MANEUVERING UNIT)

The objectives of this experiment were to perform EVA utilizing the astronaut maneuvering unit (AMU), and to determine the capabilities of the unit.

The experiment equipment consisted of the following GFAE: an astronaut maneuvering unit, a battery unit, a tether line, a thrust neutralizer pull-off housing, a telemetry receiver, and an ELSS unit and associated umbilical. The following equipment was CFE: an AMU cover assembly, two UHF antennae and a coaxial switch.

The experiment equipment interface with the spacecraft was as follows:

A. The AMU was installed in the equipment adapter section against a torque box and secured by a hollow tension bolt. The hydrogen peroxide and nitrogen service lines and the tension bolt were cut by a pyro guillotine activated from a cabin switch (AMU DEPLOY) on the command pilot's instrument panel.

B. The hand and foot rails and the two UHF antennas were deployed at the same time as the AMU cover was jettisoned. This was accomplished by activating the EVA BARS EXT switch on the cabin RH circuit breaker panel.

C. Other cabin displays consisted of an H_2O_2 pressure warning light on the annunciator panel and an H_2O_2 temperature and pressure gauge. The latter was read with the selector switch in the EXP position. Spacecraft power was utilized only for reading the H_2O_2 temperature and pressure gauge and for telemetry.

The telemetry receiver was located in the adapter electronics module, and was connected to the UHF antennas by the coaxial switch.

The AMU provided the astronaut with automatic attitude control and stabilization in three axes, and manual translation in two axes. Side translation could be accomplished by a combination of a roll or yaw maneuver and translation in one of the available modes. The astronaut controlled the AMU manually through the knobs located on the sidearm controllers. Controller knob movement was direction oriented.

The unit components were an oxygen subsystem, redundant propulsion and electrical packages (activated by a single handle on the unit), a telemetry transmitter, and a communications transceiver.

EXPERIMENT D-13 (ASTRONAUT VISIBILITY)

The objective of this experiment was to measure, under controlled conditions, the visual acuity of the astronauts during long space flights, in order to determine whether a prolonged spacecraft environment tended to degrade the crew's eyesight.

The experiment equipment consisted of a GFAE photometer, a GFAE vision tester with mouth supports, a CFE light trap mounted upon the forward end of the RH hatch, and one high-level telemetry channel.

The experiment equipment interface with the spacecraft consisted of stowage provisions for the equipment, the connection of the vision tester to the existing power supply, the attachment of the mouth supports to the plot board fittings under the LH and RH main instrument panels, and the connection of the photometer to the telemetry channel.

EXPERIMENT D-14 (UHF/VHF POLARIZATION)

The objectives of the D-14 experiment were (1) to measure the incongruities which exist in the electron content along the orbital path, and (2) to gain knowledge about the structure of the lower ionosphere and its temporal variations. These goals were achieved by measuring the electron content of the ionosphere below the Gemini Spacecraft with a Faraday rotation system utilizing dual-frequency transmitters operating at about 130 mc and 400 mc.

The experiment equipment consisted of a transmitter, a diplexer, a colinear dipole antenna, and an antenna boom. The transmitter generated the two signals of about 130 mc and 400 mc, which were then fed into the diplexer to be combined into a single output. The diplexer contained two passive filter circuits to prevent feedback between the two transmitter outputs. From the diplexer the signals were fed through coaxial cables to the dipole antenna, from which they were radiated with a vertical polarization.

The transmitter and diplexer were mounted in the spacecraft adapter equipment section and the antenna and antenna boom were mounted in the adapter retrograde section. The antenna assembly comprised an extendible dipole antenna, the antenna boom, and a choke to isolate the antenna from the boom. The upper element of the antenna was hinged in two places so that it could be confined within the spacecraft during the launch.

Spacecraft power operated this experiment. A power ON/OFF switch and an antenna EXTEND/RETRACT switch were located in the instrument console in the spacecraft cabin.

EXPERIMENT D-15 (NIGHT IMAGE INTENSIFICATION)

The objective of the D-15 experiment was to obtain information on the performance of the low light-level television system, which was designed for nighttime surveillance of oceans by an orbiting spacecraft.

The experiment equipment consisted of five units: a TV camera, a camera control unit, a viewing monitor, a recording monitor and photographic camera, and a monitor electronics and equipment control unit. Operation of the system was as follows: the TV camera scanned the earth scene and focused the image upon a sensor which converted the image into an electrical video signal that was supplied to the viewing monitor and the recording monitor. The camera control unit regulated the operation of the TV camera. The viewing monitor, receiving the video signal from the TV camera unit, reproduced and displayed to the astronauts the scene viewed by the camera. The recording monitor also received the video signal from the TV camera unit and reproduced the image for recording by the photographic camera. The monitor electronics and equipment control unit contained circuits to operate both the viewing monitor and the recording monitor.

The TV camera was mounted perpendicular to the spacecraft skin on a CFE coldplate located in the adapter retrograde section. A CFE mirror reflected TV camera vision forward and parallel to the spacecraft longitudinal centerline. This mirror was located externally on the spacecraft adapter retrograde section and, prior to use, was held in place by a jettisonable fairing. The mirror was extended into position upon the application of experiment standby power and was then locked into this position. The camera control unit had no external controls and was mounted on a CFE coldplate on the electronics module in the spacecraft adapter equipment section. The viewing monitor was stowed in the spacecraft cabin. The recording monitor and photographic camera were mounted on the coldplate in the right side landing gear well of the spacecraft so that they projected into the equipment bay through the damper cutout.

The monitor electronics and equipment control unit was also mounted on a CFE coldplate in the right side landing gear well.

The experiment utilized three high-level and three bilevel TM channels of the spacecraft adapter section telemetry system. A power ON-OFF-STANDBY switch and a fairing door SAFE-OPEN switch were provided on the instrument console in the spacecraft cabin. The experiment used spacecraft power.

EXPERIMENT D-16 (POWER TOOL EVALUATION)

The objectives of this experiment were to evaluate man's ability to perform maintenance tasks in free space and to appraise the performance of a minimum reaction power tool designed to overcome the effects of reactive torques and forces under zero gravity conditions.

The experiment equipment consisted of a GFAE space power tool (battery operated), a GFAE hand wrench, a GFAE space power tool restraint assembly with mounting rails and a work-site containing an array of mechanical fasteners, a GFAE knee tether, and a CFE strain gauge to obtain load-readings through telemetry.

The experiment equipment interface with the spacecraft was as follows: The restraint assembly, on its rails, was installed in the retroadapter section immediately aft of the RH hatch. A spring-loaded access door at the adapter outside skin was opened on booster separation by the action of a guillotine upon its restraining cable. The restraint assembly was unlocked by the EVA astronaut and rolled outboard radially at the start of the experiment. The power tool and the hand wrench were stowed in the power tool restraint assembly.

The knee tether was stowed in the cabin and taken out at EVA. During the experiment, the tether was attached to the astronaut's knees and to the adapter handrail.

EXPERIMENT M-1 (CARDIOVASCULAR CONDITIONING)

The objective of the M-1 experiment was to evaluate the effectiveness of pneumatic pressure cuffs around the astronaut's legs in preventing cardiovascular deterioration caused by prolonged weightlessness. The pressure cuffs were cyclically inflated, creating a stress on the venous system in the pilot's legs which would approximate the normal gradient in a low gravity environment.

The experiment equipment was as follows: a GFAE cardiovascular reflex conditioner, GFAE inflatable cuffs (in the pilot's space suit), CFE cardiovascular reflex conditioner mounting devices, a CFE manual control, and CFE hose assemblies.

The equipment interface with the spacecraft consisted of mounting provisions for the cardiovascular reflex conditioner unit between the ejection seats at the large pressure bulkhead, and a manual ON/OFF control mounted on the water management panel.

EXPERIMENT M-3 (IN-FLIGHT EXERCISER)

The objective of the M-3 experiment was to devise a method for evaluating the general physical condition of the crew during a space flight. This evaluation was based upon the response of the cardiovascular system to a calibrated work load, which for this experiment was the periodic use of an in-flight exerciser. Blood pressure was measured at the beginning and end of each exercise period; the pulse rate was continuously monitored by in-flight biomedical instrumentation.

The experiment equipment consisted of a GFAE EC30004 in-flight exerciser. The contractor's experiment equipment interface consisted of a cabin stowage provision for the exerciser.

EXPERIMENT M-4 (IN-FLIGHT PHONOCARDIOGRAM)

The objective of the M-4 experiment was to measure the functional status or the fatigue state of the astronaut's heart muscle by determining the time interval between the electrical and the mechanical systole of the heart muscle.

The experiment equipment consisted of an onboard biomedical tape recorder, an electrocardiographic signal conditioner, and a phonocardiographic transducer. The astronaut's heart sounds were detected by the phonocardiographic transducer and transmitted by a shielded cable to the signal conditioner, located in a pocket of the astronaut's undergarment. The signal was relayed from the signal conditioner to the suit bioplug and then on to the biomedical tape recorder.

The transducer was attached to the astronaut's chest on the sternum; transducer and signal conditioner were considered part of the astronaut's suit. Spacecraft power was utilized for this experiment and spacecraft wiring transported data from the signal conditioner in the suit to the GFAE biomedical tape recorder in the spacecraft cabin.

EXPERIMENT M-5 (BIOASSAYS BODY FLUIDS)

The intent of the M-5 experiment was to gather data on the effect of space flight upon various systems of the human body. This experiment concentrated upon the effects of space flight which alter the chemistries of body fluids. The method was to obtain preflight, in-flight, and post-flight samples of body fluids, which were then analyzed for electrolytes, hormones, protein, and other organic constituents that indicate the physiological status of the crew.

The principal weight loss during flight may be related to water loss, of urinary, sweat or other origin. Fluid intake and urinary output were measured, along with changes in the electrolyte and hormone concentrations in samples. Before flight, the plasma and urine samples were analyzed; during flight, only the urine was sampled.

To obtain this data, a quantity of 112 GFAE 75 cc urine sample bags was provided. Stowage provisions in the cabin constituted the experiment equipment interface.

EXPERIMENT M-6 (BONE DEMINERALIZATION)

The objective of this experiment was to establish the occurrence and degree of bone demineralization resulting from immobilization and prolonged weightlessness encountered by astronauts in space flight.

The experiment involved the making, before and after the flight, of a series of radiographs of each astronaut's left foot and left hand. The method used was radiographic bone densitometry. No in-flight equipment was required for this experiment, but a special analog computer and several standard clinical x-ray machines, standard films, and calibrated densitometric wedges were employed at Cape Kennedy, on the recovery carrier, and at the Manned Spacecraft Center.

There was no experiment equipment interface with the spacecraft.

EXPERIMENT M-7 (CALCIUM BALANCE STUDY)

The objective of this experiment was to obtain information about the effects of extended space flight upon human bone and muscle tissue. Bedrest studies have demonstrated that a prolonged immobility of the human body results in substantial losses of calcium, nitrogen, and related elements. A sizable decrease in the body's normal aggregate of these elements, due to the stresses of a long space flight, could in theory lead to a serious weakness of the bones and muscles. This experiment was conducted to determine whether these losses were large enough to require prophylactic procedures.

Equipment for this experiment consisted of urine sample bags and a supply of GFAE defecation gloves. As a back-up to the sample bags, a flowmeter was added in the urine transport system; this output signal was recorded on the voice tape recorder. The equipment interface with the spacecraft consisted of stowage provisions in the cabin for the gloves and sample bags.

EXPERIMENT M-8 (IN-FLIGHT ELECTROENCEPHALOGRAM)

The goal of this experiment was to obtain precise and objective information regarding the number, duration, and depth of an astronaut's sleep periods during flight. This information is vital in devising future work-rest cycles and in determining to what extent pilot performance would be affected by diminished alertness.

This experiment employed an electroencephalogram (EEG) to chart the brain's electrical activity, which undergoes established variations with different levels of sleep.

The experiment equipment consisted of a biomedical tape recorder, two miniature transistorized amplifiers, and two pairs of nontraumatic scalp electrodes. The electrodes monitored the electrical activity of the cerebral cortex, sending a signal which was conditioned by the amplifiers for storing in two channels of the biomedical tape recorder.

The electrodes were attached to the astronaut's scalp and both the electrodes and the amplifiers were considered part of the astronaut's suit. Spacecraft power operated this experiment and spacecraft wiring transported data from the amplifiers in the suit to the GFAE biomedical tape recorder in the spacecraft cabin.

EXPERIMENT M-9 (HUMAN OTOLITH FUNCTION)

The objectives of the M-9 experiment were twofold: (1) to assess the ability of the astronauts to determine horizontality using their spacecraft alone as a reference, and (2) to determine the possible effect of prolonged weightlessness upon equilibrium.

In weightlessness, primary gravitational cues are lost and the otolith apparatus (the organs of equilibrium) is deprived of its normal stimulus. This permits investigation of the importance of secondary gravitational cues in enabling the body to orient itself. In orbital flight, the astronaut is responsive to his spacecraft because of tactile messages, even though his eyes are closed.

Persons having a bilateral loss of otolith function are incapable of estimating the vertical and horizontal in the absence of visual cues. Prolonged weightlessness may also change the sensitivity of the equilibrium organs. For this reason, the astronauts were required to orient a luminous line daily during the flight to permit monitoring of their otolith activity.

The experiment equipment was incorporated into the GFAE vision tester used in Experiment S-8/D-13, but in all other respects the two experiments were separate. The experiment equipment interface with the spacecraft was the same as that of the S-8/D-13 experiment.

EXPERIMENT MSC-1 (ELECTROSTATIC CHARGE)

The objective of the MSC-1 experiment was to detect and measure any accumulated electrostatic charge that might have been created on the surface of the Gemini Spacecraft by ionization from attitude motor exhausts. Investigation of these theoretical charges was required before attempting rendezvous flights because of the possibility of explosion and other detrimental effects due to the discharge of electrostatic potential between spacecraft during rendezvous. The experiment equipment was also to be operated during retrofire and during the spacecraft's passage through the South Atlantic magnetic anomaly.

The experiment equipment consisted of a sensor unit and an electronics unit. The sensor unit contained a sensor disk vibrating in a direction normal to its surface, and a preamplifier to detect the electrostatic charge. The electronics unit contained an oscillator which generated the current to drive the sensing disk, an automatic ranging device extending over four decades, a demodulator to condition the amplifier output for telemetry input, and a power conditioner.

The two experiment units were mounted between stringers in the retrograde adapter section of the spacecraft, aft of the ring at station Z 94.40, with the element of the sensing unit exposed to space after spacecraft insertion. The sensing element was protected during launch by a jettisonable door, which was jettisoned at the first application of power to the sensor unit. Telemetry data were transmitted by the electronics unit on two TM channels of the high-level multiplexer in the re-entry module. Spacecraft power operated the experiment; a power ON/OFF control switch was located on the instrument console in the spacecraft cabin.

EXPERIMENT MSC-2 (PROTON-ELECTRON SPECTROMETER)

The objective of Experiment MSC-2 was to measure the proton and electron intensities outside the spacecraft and to investigate the radiation dose received by the crew during the mission. A long-range goal of this experiment was to discover how closely the measured dose agreed with a predetermined calculation, to permit more accurate predictions of the radiation to which astronauts may be exposed on space missions.

The experiment equipment was a proton-electron spectrometer. On the Gemini IV mission, the unit employed a pulse height analyzer with a plastic scintillator in an anticoincidence arrangement. On the Gemini VII mission, the scintillator was replaced by a thin dE/dx plastic wafer mounted over the instrument entrance aperture. This modification permitted the measurement of protons of energy $5 < E < 18$ MeV instead of protons of energy $25 < E < 80$ MeV.

The spectrometer, which monitored the external environment, was mounted in the Gemini equipment adapter section and fastened to the center pallet area, with the spectrometer face looking aft and covered by the adapter thermal curtain. The internal radiation dose was monitored with operational film-badge packages in the crew's underclothing. The electronics unit data were transmitted on eight bilevel and six high-level TM channels of the spacecraft PCM telemetry system in the adapter section. A power ON/OFF control switch (labeled SPT-MAG) was located on the instrument console (Agena Control section) in the spacecraft cabin. The pilot recorded the experiment ON/OFF times on the Gemini voice recorder. Spacecraft power operated this experiment.

EXPERIMENT MSC-3 (TRI-AXIS MAGNETOMETER)

The objective of the MSC-3 experiment was to determine the direction and amplitude of the earth's magnetic field during selected periods of several orbital missions. This information was vital to the support of the MSC-2 experiment, since particle intensities are strongly directional with respect to the magnetic field. A tri-axis flux-gate magnetometer was utilized to gather this information over the range of 0 to 60,000 gammas in three axes.

The experiment equipment consisted of a sensor unit, an electronics unit, and an interconnecting cable. The sensor unit contained three orthogonally mounted sensing devices for measuring vector components of the magnetic field in the three axes. The electronics unit contained a converter which supplied the necessary sensor drive currents, detected and transformed the magnetic field sensor signals and converted these signals to a 0-5 analog DC voltage.

The experiment units were mounted in the spacecraft adapter section. The sensor unit was mounted upon a contractor furnished telescopic boom which was capable of extending the unit 40 in. from the spacecraft adapter skin. The electronics unit was hard-mounted on the retrobeam in the spacecraft adapter retrograde section. From this unit, data were transmitted on three high-level TM channels of the adapter section telemetry multiplexer. A power ON/OFF control switch (labeled SPT-MAG/OFF/MAG ONLY) was located on the instrument console in the spacecraft cabin. Spacecraft power operated the experiment.

EXPERIMENT MSC-4 (OPTICAL COMMUNICATION - LASER)

The objective of this experiment was to determine the feasibility of using coherent optical beams (laser beams) for spacecraft-to-earth communication.

The spacecraft experiment equipment was a self-contained compact transmitter employing a gallium arsenide semiconductor laser as the active coherent light source. Three instrumented ground sites were established, each equipped with a flashing beacon and having the capability to collect and demodulate coded optical signals.

The transmitter, which was designed to be handheld by the astronaut, had special infrared safety (spectral) glasses and a microphone attached. A 6-power telescope, in conjunction with a 400-angstrom filter for fine tracking of the ground beacon, was integral to the unit.

The spacecraft experiment equipment was mounted on the center stowage container door before and after the experiment.

EXPERIMENT MSC-5 (LUNAR UV SPECTRAL REFLECTANCE)

The objective of the MSC-5 experiment was to determine ultraviolet spectral reflectance of the lunar surface between 2000 and 32,000 angstroms. Data from this experiment were to be used to estimate the total incident UV energy on an astronaut's face while he is on the lunar surface. These data will influence the design of lunar space suits.

The experiment equipment consisted of a 70 mm general purpose Maurer camera, a Maurer film magazine, an aiming telescope, an objective grating, an objective prism, and an extended timer.

The experiment equipment was stowed in the spacecraft cabin. A mounting bracket for the Maurer camera was supplied by McDonnell to position the camera in the opened right-hand hatch, so that the camera could be aimed by the illuminated reticle in the left-hand window. The camera could be removed from the bracket for use in other photography, such as general purpose and Experiment S-11.

EXPERIMENT MSC-6 (BETA SPECTROMETER)

The objective of the MSC-6 experiment was to check the calculational techniques used to compute the spacecraft external radiation. The data obtained were also used to update and fill voids in the knowledge of this environment as it affects manned earth orbital missions. The beta spectrometer was used in conjunction with the Bremsstrahlung spectrometer (Experiment MSC-7) and the tri-axis flux-gate magnetometer (Experiment MSC-3). This equipment provided the added abilities to determine the direction and amplitude of electrons impinging upon the spacecraft and to discover the extent of x-ray radiation.

The experiment equipment consisted of two units, a sensing unit and an electronics unit. The sensing unit contained a stack of four lithium-drifted silicon semiconductors to detect beta particles. The electronics unit conditioned the sensing unit data for transmission by the spacecraft telemetry system.

The experiment units were mounted in the spacecraft adapter retrograde section. The sensor unit was mounted flush with the adapter skin and was protected during launch by a spring-loaded door operated by a cable guillotine.

The electronic unit was mounted on the retro-rocket beam. Data from the unit were transmitted on eight bilevel and two high-level TM channels of the adapter section telemetry multiplexer. A power ON/OFF control switch (labeled SPT-MAG/OFF/MAG/ONLY) was on the instrument console in the spacecraft cabin. This switch served both the MSC-3 and the MSC-6 experiments. Spacecraft power operated this experiment.

EXPERIMENT MSC-7 (BREMSSTRAHLUNG SPECTROMETER)

The objective of the MSC-7 experiment was to measure the secondary gamma rays produced in the materials of the Gemini Spacecraft by trapped electrons while the spacecraft was passing through the South Atlantic magnetic anomaly regions. While such rays do not now appear to be biologically significant, a problem may develop in future long-duration missions, when astronauts might be required to subsist for considerable periods in high trapped-electron flux environments. For this reason, a time-differentiated measurement of these rays over a large section of the spacecraft was deemed desirable.

The Bremsstrahlung spectrometer employs a scintillator of cesium iodide and plastic. Coupled with the sensor is an electronics section composed of a photomultiplier tube and an analog-to-digital converter. Transmission is accomplished by the spacecraft telemetry system.

The experiment unit was mounted on the large pressure bulkhead, inboard of the left-hand seat rail in the spacecraft cabin. The electronics data were transmitted on eight bilevel and two high-level TM channels of the re-entry module telemetry system. A power ON/OFF switch was located on the instrument console in the spacecraft cabin. Spacecraft power operated this experiment. A synchronous pulse was provided from the spacecraft PCM system to the experiment package.

EXPERIMENT MSC-8 (COLOR PATCH PHOTOGRAPHY)

The objective of this experiment was to ascertain whether existing photographic materials accurately reproduce the colors of objects photographed in space. These materials were evaluated by photographing a target containing known colors outside the spacecraft and comparing the photographs with the target colors.

The experiment equipment consisted of a 70 mm Maurer camera, a film magazine, a color patch slate, and a three ft extension rod. The color patch slate was a titanium plate with Bureau of Standards primary colors - red, blue and yellow - and a neutral gray. The extension rod held the color patch three ft from the camera. To reduce the effect of ultraviolet energy on the film, a filter cutting off at 3500 Å was placed over the camera lens.

The experiment equipment interface with the spacecraft consisted of stowage provisions in the cabin for the camera, color slate, extension rod, adapter, and mounting clip.

EXPERIMENT MSC-10 (TWO-COLOR EARTH'S LIMB PHOTOGRAPHY)

The purpose of the MSC-10 experiment was to provide a precise definition of the earth's limb so that future astronauts would be able to use it to make a navigational fix. Black and white photographs of the limb were taken with a two-color filter mosaic placed in the magazine directly in front of the film. The central vertical portions were red-transmitting Wratten No. 92 and the side portions blue-transmitting Wratten No. 47B. These filters permitted microdensitometry measurement of the terrestrial elevation of the blue over the red portion of each photographed limb.

The experiment equipment consisted of a 70 mm Hasselblad camera, a modified 70 mm film magazine, and a voice tape recorder. The experiment equipment interface with the spacecraft consisted of stowage provisions in the cabin for the camera and the tape recorder.

EXPERIMENT MSC-12 (LANDMARK CONTRAST MEASUREMENT)

The objective of this experiment was to determine the relative visibility of terrestrial landmarks from outside the earth's atmosphere. The eventual objective was to provide a reliable source of data for Apollo onboard guidance and navigational systems.

The perception, alignment and identification of landmarks are to a marked degree based upon their luminance and contrast with surrounding areas. This contrast is reduced by increasing the amount of atmosphere between the feature and the observer. Although the visual contrast of ground targets viewed from outside the atmosphere will be considerably reduced, visual contrast is a useful criterion for target visibility. Because contrast is a ratio, the measurement is independent of long-term photometric equipment gain stability, and the effects of scattered light reception are also negligible.

This experiment utilized the same photometer provided for the D-5 experiment, with the addition of two optical filters over the photometer lens. The experiment equipment interface with the spacecraft was the same as that of the D-5 experiment.

EXPERIMENT S-1 (ZODIACAL LIGHT PHOTOGRAPHY)

The objectives of the S-1 experiment were to photograph the zodiacal light, the airglow, and other dim light phenomena such as the gegenschein.

Previous experience on the Mercury program had shown conclusively that these experiments could be performed at altitudes above 90 kilometers without airglow contamination.

Some of the questions which this experiment was intended to resolve were:

A. What is the minimum angle from the sun at which the zodiacal light could be studied without twilight interference?

B. Is the gegenschein to be found in the anti-sun position, or will it rather be found to have a westerly displacement?

C. Does the airglow layer extend beyond the "normal" 90 kilometer range?

D. What is the intensity distribution of the zodiacal light and of the Milky Way in various orientations - such as the region of the sky near Cygnus?

The experiment equipment consisted of a modified 35 mm Widelux Camera (GFAE) with a rotating lens which provided very wide angle pictures (larger than 50 degrees by 130 degrees). The camera included an electronic device which programmed the exposures according to a predetermined sequence. A special mounting bracket (CFE) was attached to the right hand hatch window for camera positioning on all affected spacecraft except Gemini IX. During the Spacecraft 9 mission, the pilot held the camera. The camera and mounting bracket are stowed in the spacecraft cabin during launch and re-entry.

EXPERIMENT S-2 (SEA URCHIN EGG GROWTH)

The objective of this experiment was to evaluate the effects of low gravity upon a simple biological system during sensitive stages of its development, such as fertilization and cell division.

The experiment equipment was a cylindrical container composed of eight specimen chambers, each divided into separate compartments for sperm, ova, and fixative solution. Florida sea urchin eggs were to be fertilized in four of the chambers shortly before launch and in the remaining chambers soon after orbital insertion. Rotation of a handle on the container allowed both the fertilization of the eggs and the release of the fixative solution to inhibit their growth at various stages of development.

The experiment package was located in the spacecraft cabin on the left hand hatch and had no system interfaces other than mounting.

EXPERIMENT S-3 (FROG EGG GROWTH)

The objective of this experiment was to determined the effects of weightlessness upon normal cell division, cell differentiation, and normal embryo formation in fertilized frog eggs. Since the frog egg is known to orient itself with respect to gravity during its early development, it was an ideal subject for this experiment.

The experiment was contained in two packages, each package having four chambers which were partitioned into a frog egg section and a fixative section. Each package was insulated and had temperature control systems to maintain an experiment temperature of close to 70°F. At specified times in the flight, fixative was injected into the egg chambers by means of an actuating handle on the outside of each package. This action would kill the eggs in two chambers of each package, thereby preserving them for microscopic study at the end of the flight. Identical hardware was utilized for control experiments on the ground.

The experiment packages were mounted one on each hatch of the re-entry module. Power for temperature control and telemetry was provided by the spacecraft electrical system.

EXPERIMENT S-4 (RADIATION AND ZERO G ON BLOOD)

The objective of the S-4 experiment was to determine whether chromosome aberrations in human white blood cells were being produced by a synergistic interaction between radiation and weightlessness during orbital flight.

The experiment plan was to irradiate a thoroughly studied biological material with a known quality (phosphorous 32 beta rays) and quantity (approximately 2 rads) of radiation during the weightless phase of flight. A duplicate control sample at Cape Kennedy was actuated concurrently and subjected to an equivalent irradiation for comparison.

Each experiment package contained ten blood samples. Incorporated into the volume of each sample were two fluoroglass dosimeters to insure that the exposure was of the required duration. Two block dosimeters were also included in each instrument package to measure the nonelectron background radiation. Irradiation of the blood samples was initiated manually during flight by means of a handle on the experiment package. The in-flight experiment package was located in the spacecraft cabin on the right hatch. Electrical power was provided from the spacecraft and one telemetry channel was supplied to monitor package temperature.

EXPERIMENT S-5 (SYNOPTIC TERRAIN PHOTOGRAPHY)

The objective of the S-5 experiment was to secure high quality, small scale photographs of selected land and near-shore areas for geologic, geographic, and oceanographic study. The experiment plan called for photographic coverage of essentially two types, (1) pictures of well known areas which would serve as standards for the interpretation of lesser known areas, and (2) clear pictures of more remote regions to extend the scope of the existing photography.

Within the limits of fuel and power reserves, the flight crew was to take vertically oriented, systematic overlapping photographs of the target areas, and good single pictures of any other areas which were cloud-free and had terrain features of possible interest.

The experiment equipment consisted of a 70 mm general purpose Hasselblad camera, 70 mm film magazines, a haze filter, an exposure meter, and a voice tape recorder. The experiment interface with the spacecraft consisted of cabin stowage provisions for the equipment.

EXPERIMENT S-6 (SYNOPTIC WEATHER PHOTOGRAPHY)

The objectives of the S-6 experiment were to secure high resolution color photographs of meteorological interest and to acquire detailed photographs of the earth's cloud cover in order to validate the information obtained from weather satellite pictures.

The experiment equipment consisted of the 70 mm Hasselblad general purpose camera, a film magazine, a haze filter, and the voice tape recorder. Stowage provisions for this experiment equipment were made in the spacecraft cabin.

EXPERIMENT S-7 (CLOUD-TOP SPECTROMETRY)

The objective of the S-7 experiment was to test the possibility of measuring cloud altitudes from orbiting satellites by means of a handheld spectrograph. This experiment was of interest to meteorologists because cloud altitude indicates atmospheric states upon which weather forecasts are based.

The experiment equipment consisted of a GFAE spectrograph fitted with a 35 mm camera body. A voice tape recorder was also used. The experiment interface with the spacecraft provided cabin stowage for the spectrometer and tape recorder.

EXPERIMENT S-8 (VISUAL ACUITY)

The objective of the S-8 experiment was to determine the astronaut's naked eye capability to distinguish small objects upon the earth's surface during daylight. A photoelectric photometer was mounted near the lower corner of the right hatch window. This instrument measured the amount of ambient light dispersed into the pilot's line of sight at the moment of observation of the ground test patterns. Data from the photometer were sent to the ground by real-time telemetry.

Ground targets, set up at several locations, consisted of a number of 2000-ft squares of plowed and graded soil which were covered with white rectangles of styrofoam-coated wallboard. Each of these styrofoam rectangles was oriented in one of several positions (i.e., east-west, north-south, or diagonal) within its square, and the placement was varied from square to square.

Advance knowledge of the placement of the rectangles was withheld from the astronauts, since it was their task to report the orientations. The plan called for a change in the placement of the rectangles between orbital passes and an adjustment in their size in accordance with the variables of slant range, elevation of the sun, and the visual performance of the astronauts on preceding passes.

The experiment equipment, in addition to the photometer and the ground site preparation, included ground instrumentation to determine atmospheric and lighting conditions. The experiment equipment interface with the spacecraft consisted of cabin stowage provisions for the photoelectric photometer, and the connection of the photometer to a spacecraft telemetry channel.

EXPERIMENT S-9 (NUCLEAR EMULSION)

The primary objective of the S-9 experiment was to broaden our knowledge of high-energy-particle physics, especially of such phenomena as cosmic rays and Van Allen radiation. To accomplish this investigation, the experiment plan called for a stack containing nuclear emulsion to be exposed on several long-duration orbital flights.

Nuclear emulsion has two principal advantages to recommend it in this type of experiment: (1) the amount of data that can be obtained for a given weight of the emulsion is high compared to the results achieved by other detectors, and (2) special properties of nuclear emulsion permit a more detailed study of energetic particles than is possible with other mediums.

The emulsion (made of gelatin and inert silver bromide) was arranged in a stack composed of two sections held together by clamping plates. When the spacecraft encountered the desired experiment environment, the upper section of the stack was designed to separate from the lower section, allowing the stack to commence its planned operation. During EVA, the experiment package, which was mounted in the retrograde section of the adapter immediately behind the right hand hatch, was to be retrieved by the astronaut; it would then be stored in a heat-insulated location inside the re-entry vehicle.

To protect the experiment from excessive radiation in a high apogee orbit, an insulated door containing 0.50 in. thick aluminum radiation shield was installed in the adapter skin, over the sensitive surface of the package. This door was held closed during the launch phase and opened when the horizon scanner fairing was jettisoned. A toggle switch was added to the right hand main instrument panel for actuation of the pyrotechnic door. The temperature-maintaining container in the re-entry module was a cold box made of welded aluminum, cooled by the spacecraft bypass coolant lines. One high-level analog channel was provided to the experiment package for telemetry.

EXPERIMENT S-10 (AGENA MICROMETEORITE COLLECTOR)

The primary objectives of the S-10 experiment were to study the micrometeorite content of the upper ionosphere and of the near-earth space environment, and to determine what effect this environment has upon biological microorganisms. These objectives were to be accomplished by exposing various samples of polished metal, glass, plastic, and meteorite material to the particle flux of the upper atmosphere. It was hoped that this procedure would yield data about impact craters, meteor erosion rates, and the deterioration of optically ground and polished surfaces.

In addition, thin film samples and biological specimens were exposed to discover penetration characteristics and the number of living microorganisms remaining upon the biological plates after exposure. Hyperclean surfaces were also provided to collect particulate material.

The experiment equipment was composed of (1) an aluminum frame which provided a mounting platform for the polished plates and collection surfaces, (2) a mounting bracket of rails to contain the collector frame, and (3) an aft fairing and a removable forward fairing to protect the sensitive experiment plates from the direct impact of airborne particles during launch and orbital insertion. A fabric pouch was also provided in the spacecraft cabin to stow the micrometeorite collector after its retrieval from the TDA. The mounting bracket and the aft fairing were attached to the outer skin of the TDA cylindrical section; both bracket and fairing were permanently installed.

The experiment method was to mount the collector on the TDA in the closed position, within the mounting rails. During EVA, the pilot was to remove the fairing cover and expose the collection surfaces to the outside environment. On the following mission, the astronaut was to close the old collector and retrieve it for stowage in the spacecraft cabin. He would then replace it with a new collection package whose cover he would open to expose the sensitive plates.

EXPERIMENT S-11 (AIRGLOW HORIZON PHOTOGRAPHY)

The objective of the S-11 experiment was to obtain more complete information about the airglow layer, a band of light lying some six to ten degrees above the visible horizon of the earth. This objective was to be achieved by means of a thorough photographic coverage, comprising both nighttime and twilight pictures of the layer.

The experiment required that an optical filter, which passed only the 5577 Å and 5893 Å lines of the neutral oxygen and sodium atoms, be fitted to the camera. Through this filter, the airglow band would continue to be visible, although all other details of the horizon would disappear; this phenomenon is due to the fact that the airglow is an emission of light by the gases of the high atmosphere. Additional exposures with the lens filter removed from the camera were to be made for comparison.

The equipment was a 70 mm general purpose camera with an f/0.95 lens, a film magazine containing the optical filter, an exposure timer and an illuminated sight, a window camera mounting bracket, and a mounting bracket adapter.

The experiment equipment interface with the spacecraft consisted of stowage provisions in the cabin for the camera, film magazine, mounting bracket and adapter. The contractor's interface requirements also included the attachment of the mounting bracket to the right hatch window.

EXPERIMENT S-12 (MICROMETEORITE COLLECTION)

The objectives of the S-12 experiment were (1) to collect and recover uncontaminated micrometeorite material in the near-earth space environment, and (2) to determine the effects of this environment upon biological micro-organisms.

The experiment method involved exposing highly polished plastic and metal plates outside the spacecraft during orbital flight. The particulate material collected, the number of living organisms remaining upon the exposure plates, and the holes and craters made in the specially polished plates constituted the data requirements of the experiment. In addition, sterile surfaces were provided for gathering microorganisms which might be emitted from the spacecraft or from expelled materials.

The collection equipment included an aluminum assembly providing 24 surfaces for the collection of data and the exposure of specimens. Two motor-operated cover doors permitted the controlled exposure of the plates and specimens by the flight crew. In addition to the collector assembly, the experiment required a mounting bracket and aft fairing, a removable forward fairing, and a cabin control switch.

The experiment equipment was mounted on the contractor-furnished mounting bracket attached to the outside moldline of the retrograde section of the adapter. Upon the experiment's completion, the collector assembly was retrieved and stowed in the spacecraft cabin on the centerline structure door.

EXPERIMENT S-13 (UV ASTRONOMICAL CAMERA)

The principal goal of the S-13 experiment was to ascertain the ultraviolet radiation spectra of stars in the wavelength between 2000 and 4000 Å. Photographs were to be made with the 70 mm general purpose camera, used in conjunction with an objective grating (Gemini X) or an objective prism and an objective grating (Gemini XI and Gemini XII). The high resolution photographs obtained with this equipment were expected to reveal absorption and emission lines in the spectra, thereby enabling us to analyze atomic excitation and ionization phenomena in these wavelength regions. Knowledge about the surface temperatures of the selected stars and the absorption effects taking place in their atmospheres will be gained as a result.

The experiment equipment consisted of the 70 mm Maurer camera, film magazines, an objective grating attachment, an objective prism, a UV lens, a cable release, and a camera mounting bracket.

The experiment equipment interface with the spacecraft consisted of the same contractor-furnished mounting bracket used in the MSC-5 experiment (Lunar UV Spectral Reflectance).

EXPERIMENT S-26 (GEMINI ION WAKE MEASUREMENT)

The goals of the S-26 experiment were (1) to determine the extent of the atmospheric disturbance produced by an orbiting Gemini Spacecraft, and (2) to achieve a more complete understanding of the structure of the ion flux resulting from the spacecraft's passage. To arrive at these objectives, the accomplishment of the following steps was necessary:

A. A mapping of the spacecraft ion density wake within specific coordinates.

B. A contour mapping of the spacecraft electron density wake made concurrently with A.

C. A determination of electron temperature within the same position coordinates.

D. A recording of the fluctuation of ambient ion and electron densities and temperatures caused by the variations in altitude and position of the docked GATV configuration.

E. A measurement of ionization oscillations due to spacecraft thruster firing.

The experiment equipment consisted of two ion detectors, an electron detector, and a programmer unit.

The four experiment units were mounted in the target docking adapter (TDA). One of the ion detectors and the electron detector were mounted upon the forward ring (adapter station 23.795) and projected toward the cone. The other ion detector and the programmer unit were mounted within the TDA. The experiment packages utilized power from the Agena Target Vehicle. The experiment was turned on at the time of Agena activation prior to launch and remained on for the duration of the mission. Experiment data were transmitted on the Agena pulse-code-modulation system utilizing 11 Agena PCM channels.

EXPERIMENT T-1 (RE-ENTRY COMMUNICATIONS)

The objective of this experiment was to investigate the feasibility of overcoming re-entry radio blackout by injecting water vapor into the ionized sheath which surrounds the spacecraft during re-entry.

The experiment equipment consisted of a GFAE water expulsion system and a CFE experiment activation switch.

The experiment interface with the spacecraft comprised the installation of the water expulsion system on a modified RH skid well door and provisions for wiring the system to the activation switch located on the RH switch/ circuit breaker panel.

EXPERIMENT T-2 (MANUAL NAVIGATIONAL SIGHTINGS)

The objective of this experiment was to evaluate the ability of a navigator to measure the angles between various celestial bodies from the Gemini Spacecraft using a handheld sextant.

The experiment equipment consisted of a space sextant, two sextant eyepieces, a battery, and the general-purpose photo event indicator. The experiment equipment interface with the spacecraft consisted of stowage provisions in the cabin for the sextant, photo event indicator, and the related equipment.

EXPERIMENT S-29 (PHOTOGRAPHIC STUDY OF EARTH - MOON L_4 AND L_5 LIBRATION REGIONS)

The objective of this experiment was a photographic investigation of the regions of the L_4 and L_5 libration points of the earth-moon system. This investigation was intended to settle the question of the existence of clouds of particulate matter which may be orbiting the earth in these regions. The libration points, which lie in the orbital path of the moon, 60° ahead of and 60° behind the moon, define an area of stable equilibrium in which centrifugal forces balance gravitational forces. These points have therefore been proposed as sites for earth orbital stations which, once in orbit, could be maintained in that condition indefinitely. The existence of particulate clouds in these regions, whether derived from interplanetary matter or from material sloughed off by the moon, would go far to demonstrate the relative quiescence of the area and hence the feasibility of the proposed space station location.

The experiment equipment consisted of one GFAE film back with special high speed black-and-white film, the Experiment S-11 mounting bracket (GFAE), the Maurer 70 mm camera with f/0.95 lens, and the extended exposure timer.

Since all the equipment listed above, except the special film back, had previously been provided for other purposes, only special cabin stowage provisions for the film back were required in order to incorporate this experiment into the mission.

EXPERIMENT S-30 (DIM LIGHT ASTRONOMICAL OBJECT PHOTOGRAPHY)

The goal of the S-30 experiment was to obtain photographs of selected faint astronomical phenomena by using the low light television camera and other equipment of the D-15 experiment. The primary targets were (1) the Lagrangian libration points, (2) the gegenschein, (3) the zodiacal light at 60° elongation, (4) the airglow layer viewed in profile, and (5) the brightest portion of the Milky Way.

The experiment was to start upon the next orbital pass after the D-15 experiment, thereby eliminating equipment warm-up time and conserving spacecraft power. The command pilot was to orient the spacecraft so that the target was centered in the field of view from the cabin window. Before and during camera actuation, the astronauts were to describe the scene on the voice tape recorder, noting the star field and any dim light phenomena visible. After each exposure sequence, the command pilot was to reorient the spacecraft to the next target.

The experiment equipment and the stowage provisions were the same as those provided for Experiment D-15 (night image intensification).

EXPERIMENT S-51 (DAYTIME SODIUM CLOUD PHOTOGRAPHY)

The aims of this experiment were (1) to determine the possibility of photographing a sodium vapor cloud during the daytime, and (2) to analyze the behavior of upper atmosphere winds at altitudes between 100 and 150 kilometers. The experiment was to be initiated by launching a French Centaur rocket from a site in Hammaguir, Algeria, on a ballistic trajectory.

At approximately 60 miles altitude, sodium vapor was to be ejected from the rocket along the flight path. Directional commands from the ground were to be given to the astronauts as they approached the cloud. When the astronauts saw the cloud, they were to photograph it as it dissipated, recording the effects of the high altitude winds upon the vapor.

The experiment required a Maurer 70 mm camera, an interference filter, one film pack containing high speed black-and-white film, and two pairs of goggles for the flight crew. Stowage provisions for the experiment equipment were made in the spacecraft cabin.

TARGET DOCKING ADAPTER AND AUGMENTED TARGET DOCKING ADAPTER

TARGET DOCKING ADAPTER

General

The target docking adapter (TDA) is combined with the Agena to form the Gemini-Agena Target (GAT) Vehicle, utilized in rendezvous and docking missions with the Gemini Spacecraft. The TDA receives the spacecraft, attenuates impact shock, and provides a rigid connection between the Gemini and the Agena. The TDA consists of two major structural components - a cylindrical shell section (adapter) and a truncated docking cone section (docking cone). The assembly is shown in Fig. 60 and the docking system is shown in Fig. 61.

Adapter. - The adapter shell is bolted to the Agena forward auxiliary rack. During the launch, a clamshell-type aerodynamic ascent shroud is mounted on a ring at the forward face of the adapter shell. This shroud is jettisoned before orbit insertion. The shell section of the adapter contains the following components:

A. Three hardpoints for rigidizing of the docking cone.
B. Attachment points for the shock attenuation system.
C. Mooring drive system.
D. Radar transponder and associated electronics.
E. Acquisition and docking cone lights.
F. Left and right status display panel (the center status display panel is mounted on a lateral damper).
G. Power and sequential electrical systems.
H. Various experiments housed within or mounted upon the adapter shell.

Docking Cone. - The docking structure, a double-skinned, ring and rib stiffened truncated cone, serves as a structural bridge between the Gemini nose hardpoints and the hardpoints in the adapter shell. The docking cone contains the following components:

A. Mooring latch.
B. Umbilical plug.
C. Three fittings which attach the shock attenuation system.
D. Electrostatic discharge device.

The docking cone is connected to and held in a fixed relationship to the adapter by the shock attenuation system. During initial contact with the Gemini spacecraft, the docking cone is in its extended or "unrigidized"

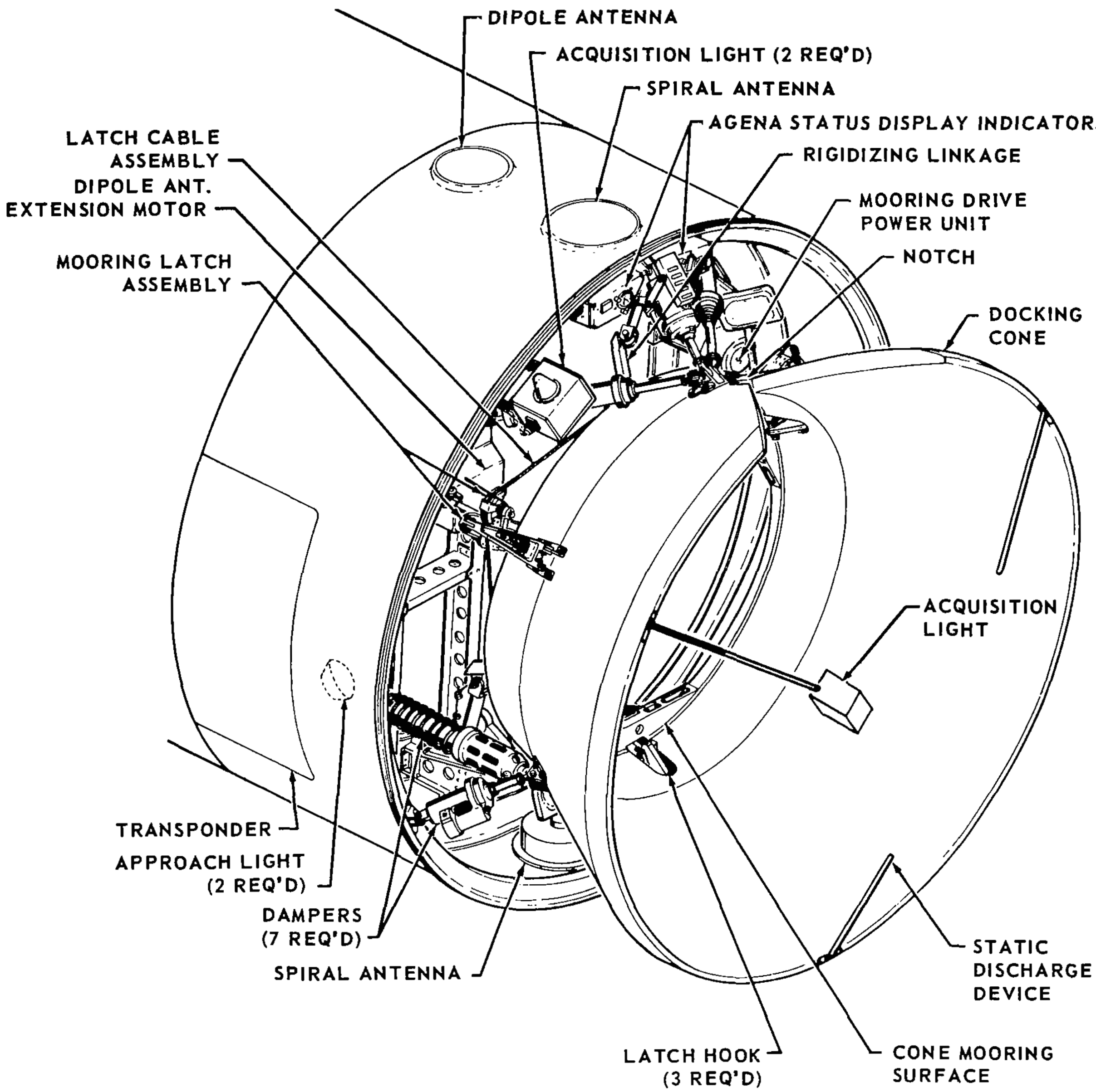


FIGURE 60 TARGET DOCKING ADAPTER ASSEMBLY

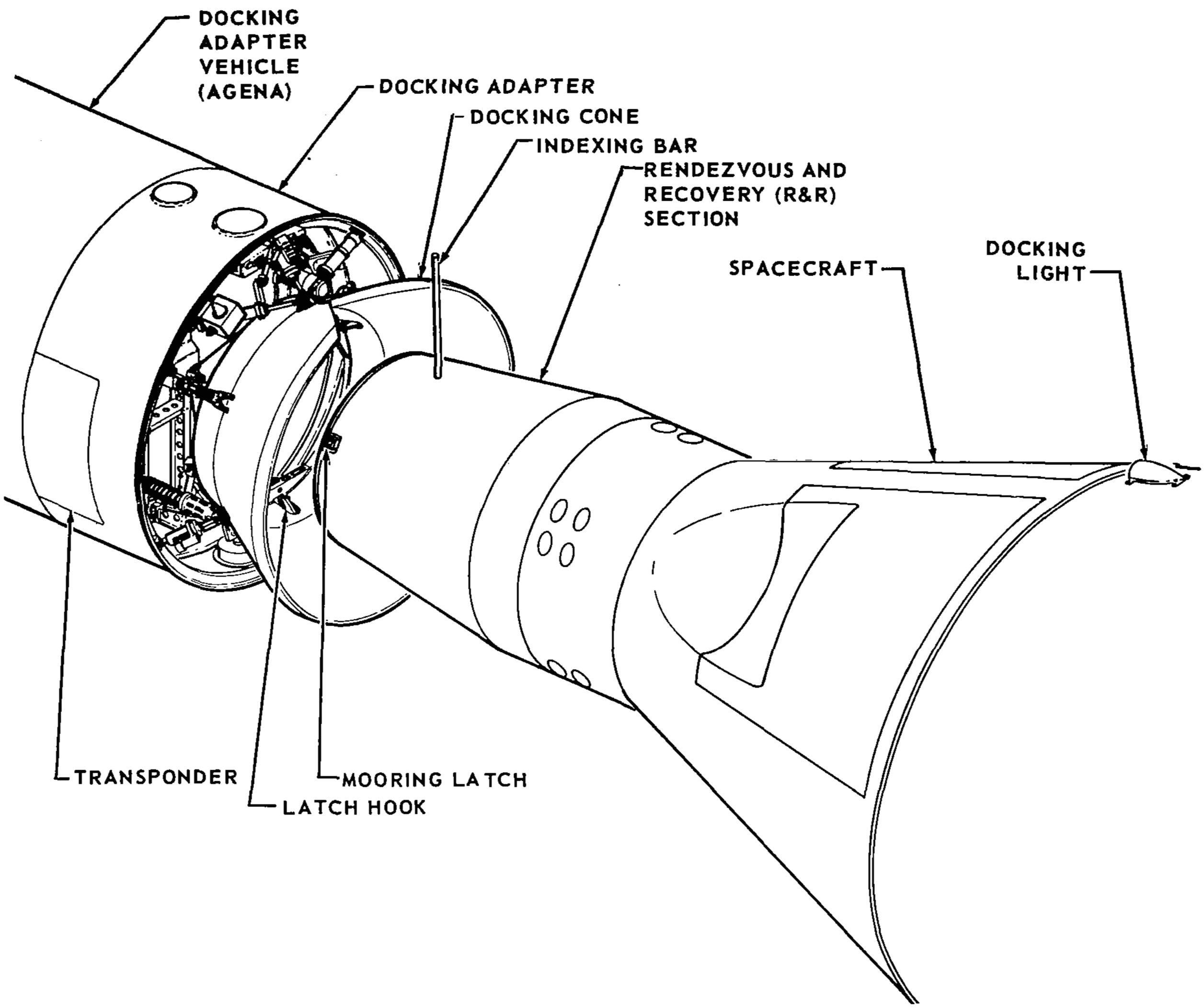


FIGURE 61 DOCKING SYSTEM

position. During the rigidizing sequence, the docking cone and the spacecraft are pulled toward the docking adapter main structure, and the base of the cone is seated firmly against hardpoints in the adapter shell.

Structural Design Criteria

The TDA is designed for the limit loads and heating effects of the Atlas/Agena design launch trajectories. For these conditions, ultimate loads are limit loads times a safety factor of 1.25.

The design launch vibration spectrum is 11.2 g's RMS with the following distribution: 0.08 g^2/cps, 20 to 200 cps; 0.12 g^2/cps from 200 to 500 cps;

and 0.05 g^2/cps from 500 to 2000 cps. This was based on data from the Mercury program and trajectory information for the Atlas/Agena (Ref McDonnell Report 8616, Appendix A), with a conservative assumption that the Agena configuration would be comparable to Mercury in developing buffeting pressures. By comparison, the Gemini launch spectrum is 8.8 g's RMS.

Latching And Rigidizing

Docking is accomplished by sliding the fixed latch receptacles on the Gemini R & R section past three spring loaded latch hooks inside the TDA cone. When all three latches are engaged the automatic rigidizing sequence is initiated by the closure of three limit switches, one at each latch point. Rigidizing is achieved by applying tension to three over-center linkages driven by the mooring drive system. (The latter is an electromechanical device, consisting of an electric motor driving through a series of gearboxes and flexible shafts). Rotation of the over-center links compresses the dampers and preloads the spacecraft-docking cone combination against the hard-points in the cylindrical section, thus achieving structural continuity between the spacecraft and the target vehicle.

Rigidizing is completed when the docking cone, with the spacecraft attached, bottoms against structural stops within the cylindrical portion of the TDA. After the rigidizing sequence, the over-center linkage has completed its travel and has shut off the motor by means of the three limit switches. Rigidizing can be accomplished automatically (normal mode) or initiated via the UHF ground command link.

In addition to the automatic rigidizing sequence employed on the Gemini VI-TDA 2 mission, subsequent missions provided, through the umbilical plug and the single point umbilicals, a pilot hardline control of the operation. A RIGID-OFF-STOP switch on the spacecraft instrument panel is for the use of the crew during a docking anomaly, In the RIGID position the switch applies power if the automatic rigidize circuit fails. The STOP position cuts off power if the rigidize limit switch fails.

During a normal docking procedure, a Gemini/TDA umbilical mating forms a hardline electrical connection. The umbilical connector is a nine conductor receptacle-plug assembly. The receptacle mates with the docking adapter plug as the spacecraft and the TDA are being rigidized. The connector is mated by using a spring-operated insulator block. The mating of this assembly is illustrated in Fig. 62.

The TDA has flashing acquisition lights in addition to lights to illuminate the cone. The two acquisition lights (65 flashes per min, nominal), which can be seen for about 20 nautical miles, aid the pilots in visually tracking the Agena Target Vehicle. These lights are mounted at the outer edges of the adapter structure and are visible around the outer edge of the cone. They are extended at first docking cone unrigidizing and remain extended.

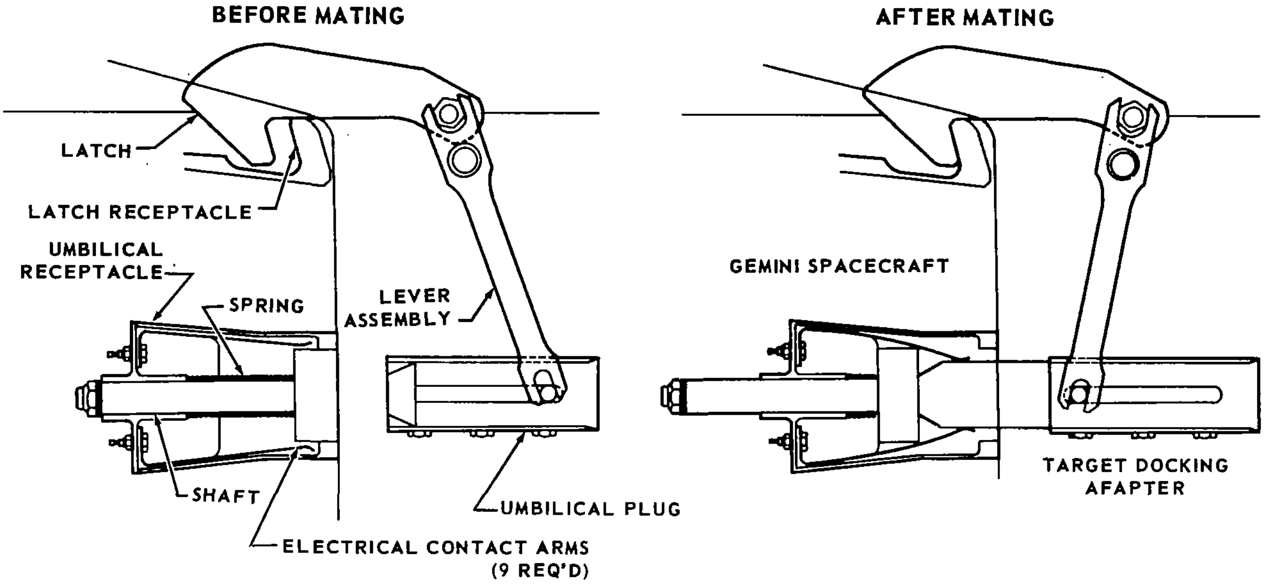


FIGURE 62 GEMINI/TDA UMBILICAL MATING

Two approach lights are mounted on the lower inside of the adapter and shine through the rear of the cone and on the upper inside surface near the V-notch. These lights may be turned off and on either by pilot or ground command.

In addition, the spacecraft is fitted with a 4000 candlepower rendezvous-docking light which is mounted externally on the retrograde section (centered above and just aft of the heads of the flight crew). This light aids in observing the Agena target during rendezvous.

Unrigidizing And Unlatching Sequence

On TDA 2, unrigidizing could be initiated by hardline or UHF command to sequentially unrigidize the mooring drive system, retract the latches to allow the spacecraft to separate from the target vehicle, and, after 30 sec, reset the latches to permit additional docking operations. On TDA 3 and up, (Spacecraft 8) the unrigidizing and unlatching were considered as separate sequences.

Unrigidize Sequence. - To back up the unrigidizing sequence by hardline or UHF command, an OFF-UNDOCK switch is mounted on the spacecraft center instrument panel. The UNDOCK position applies power through the Gemini/TDA umbilical and one of the rendezvous umbilicals to interrupt rigidize power and supply power to the unrigidize side of the mooring drive motor. Power to the mooring drive motor is cut off when two of the three limit switches sense that the unrigidizing has been accomplished.

Unlatching Sequence. - The unlatching sequence begins with the unrigidize sequence. Power is simultaneously applied to the unrigidize winding of the mooring drive motor and to the retract winding of the latch actuator. The actuator mechanism retracts the three latches by means of a system of spring cartridges, bellcranks, and a run-around cable. The slip clutch in the latch actuator slips until the load in the latches is relieved by sufficient movement of the unrigidizing system. After full retraction of the latches, the clutch will continue to slip until one of the two spacecraft free limit switches is returned to the unactuated position. At this time, power is removed from the latch actuator retract winding and applied to the extend winding. In addition, all sequential relays are reset for additional docking maneuvers. Power is removed from the extend winding of the latch actuator when an external limit switch senses that the actuator is extended and the latches are reset. Fig. 63 illustrates the latch hook retracted and extended configurations.

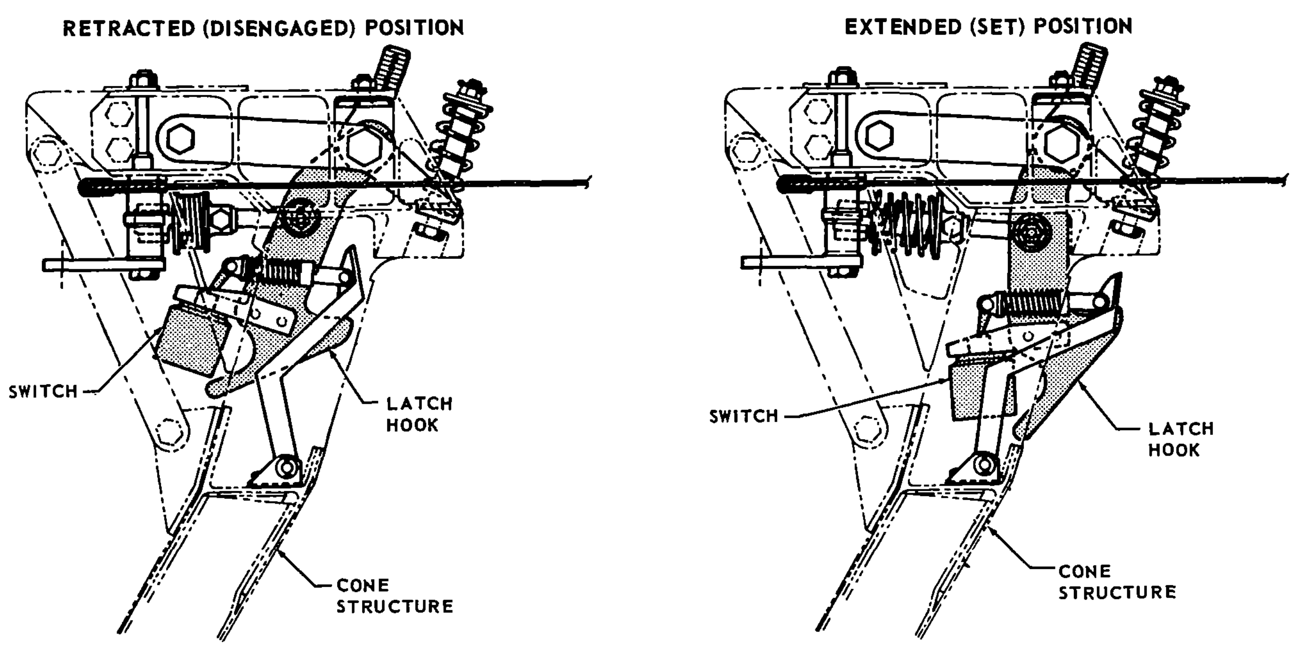


FIGURE 63 LATCH HOOK

The crew can separate the spacecraft from the Agena in an emergency by pressing an emergency switch which fires pyrotechnic igniters, breaking the latch receptacles loose from the spacecraft. Emergency separation permanently removes the docking capabilities of the spacecraft and target vehicle.

Development And Qualification Tests

Primary development and qualification tests are listed in Table 36. The 1/4 scale impact tests and the stiffness and response tests of the combined vehicles were also applied to the TDA. Additional TDA tests included the dynamic response of equipment to vibration, and element impact tests to develop the design of the docking cone shock absorber tests. As a result of tests the proposed pyrotechnically actuated transponder door was deleted prior to the first TDA flight (TDA 2) and was replaced with a fixed door for all missions.

Failure Summary And Analysis

No structural failures occurred during development or qualification testing. The following is a summary of the failures, fixes, and retests.

Evaluation of Mooring Drive System. - TR052-044.07.01 - The mooring drive system, tested in the Zaleski Chamber at -65°F, would not consistently drive to a rigidized position with minimum interface voltage. In addition, the latch release actuator failed to operate. (When the latch release actuator was replaced and the specimen was moved to the 18 ft altitude chamber for continued tests, failure analysis of the actuator did not confirm the -65°F failure. Further testing of the mooring drive system underpower condition is discussed in Mooring Drive Power Unit Comparison Test, below.)

Mooring Drive Power Unit Comparison Test. - TR052-044.07.05 - This testing was aimed at developing sufficient power in the mooring drive motor to consistently achieve a full rigidizing cycle at the minimum voltage. Power was increased 50% by a minor redesign of the brake coil, which allowed more current to flow through the armature. The greater outputs were evaluated and found satisfactory. As a result, the Agena interface maximum voltage was decreased from 29.5 volts to 26 volts. This motor was retested in TR052-044.07.06, described below.

Mooring Drive Motor Evaluation. - TR052-044.07.02 - This testing was instituted because of the impracticability of sealing the mooring drive power unit to zero leakage at an appreciable altitude. A power unit known to leak was put into a five-day altitude soak, then operated for 130 cycles. This test indicated that the motor had degraded but not failed; static torque output was down by approximately 30%. Failure analysis showed that the brushes were pitted by arcing and that a portion of the gear train has been badly eroded and had filled the gearbox with "shavings." This incident was due to improper assembly; the pitted brushes were deemed otherwise normal for the large amount of cycling in a hard vacuum. Therefore, no fixes or retests were made.

Functional Test TDA. - TR052-044.07 - This testing followed TR052-044.07.01 and was done in the 18 ft altitude chamber. Testing consisted of the application of 150 cycles at room ambient conditions, 100 cycles at 450,000 ft and 200°F, and 30 cycles at 450,000 ft and -65°F. Again the system indicated inadequate power to rigidize at minimum interface voltage.

TABLE 36 STRUCTURAL DEVELOPMENT & QUALIFICATION TESTS

T.R. NUMBER	TEST	PURPOSE
052–044.05 (TDA & S/C)	1/4 SCALE GEMINI DYNAMIC MOORING DEVELOPMENT MODEL TEST	TO VERIFY DOCKING IMPACT DYNAMICS
052–044.06 (TDA & S/C)	DYNAMIC RESPONSE OF GEMINI SPACECRAFT IN THE MOORED CONFIGURATION	VERIFY THE MODAL RESPONSES OF THE GEMINI/AGENA SPACECRAFT, GATHER EQUIPMENT RESPONSE DATA, AND VERIFY THE STABILITY OF THE RIGIDIZING LINKAGE AGAINST INADVERTENT RELEASE
052–044.07 (TDA)	LATCHING, MOORING AND LATCH RELEASE SYSTEM FUNCTIONAL AND LIFE TEST, TDA	VERIFY THE OPERATIONAL FATIGUE LIFE OF THE TDA MECHANISM SYSTEM
052–044.08 (TDA)	DEVELOPMENT OF TDA DAMPERS	DEVELOP THE METERING ORIFICE TO DESIGN REQUIREMENTS
052–052.16 (TDA)	DOCKING CONE IMPACT TEST	VERIFY TYPE OF STRUCTURAL DESIGN OF CONE SURFACE AND SUBSTRUCTURE TO SATISFACTORILY RESIST LOCAL IMPACT LOADS
052–052.40 (TDA)	DEFLECTION TEST OF TARGET DOCKING ADAPTER DOCKING CONE	DETERMINE DEFLECTION CHARACTERISTICS OF THE 52–34004 DOCKING CONE
052–052.47 (TDA & S/C)	STIFFNESS TEST OF TDA STRUCTURAL ASSEMBLY	VERIFY STIFFNESS OF COMPOSITE TDA WITH AND WITHOUT PRELOAD
052–058.34 (TDA)	DYNAMIC RESPONSE OF TDA DAMPERS	OBTAIN LOADS DATA IN DAMPERS AND DAMPING CONSTANTS CORRESPONDING TO VIBRATION FREQUENCIES
052–058.44 (TDA)	VIBRATION TEST – TARGET DOCKING ADAPTER (TDA NO. 2)	CHECK ACTUAL EQUIPMENT VIBRATION RESPONSE AND APPLY ACCEPTANCE LEVEL RANDOM VIBRATION
052–058.45 (TDA)	DEMONSTRATION OF TRANSPONDER PERFORMANCE AFTER PYROTECHNIC DOOR FIRING	TO OBTAIN DYNAMIC IMPULSE INFORMATION ON FIRING OF TRANSPONDER DOOR PYROTECHNIC
052–044.09 (TDA & S/C)	FULL SCALE MOORING SHOCK & FUNCTIONAL TEST OF GEMINI SPACECRAFT AND AGENA TDA	QUALIFY TDA, INDEX BAR AND GEMINI R & R FOR DOCKING IMPACT LOADS. DEMONSTRATE ALL FUNCTIONING, INCLUDING EMERGENCY RELEASE AFTER BEING SUBJECT TO IMPACT
052–044.11 (TDA)	TDA APPROACH LIGHT ASSEMBLY – QUALIFICATION TEST	QUALIFY TDA APPROACH LIGHT INSTALLATION. DEMONSTRATE PERFORMANCE, LIFE REQUIREMENT AND STRUCTURAL ADEQUACY FOR LAUNCH VIBRATION ENVIRONMENT
052–055.10 (TDA & S/C)	FUNCTIONAL CHECK OF GEMINI TDA UMBILICAL	TO DETERMINE THE FUNCTIONAL ADEQUACY OF THE UMBILICAL ASSEMBLY AND SUBJECT THE MECHANISM TO THE LAUNCH VIBRATION SPECTRUM
LMSC SW/30.83 (TDA)	STRUCTURAL TEST REQUIREMENTS, GEMINI-AGENA TDA	STRUCTURALLY QUALIFY TDA FOR LAUNCH CONDITIONS (ENCOMPASS ATDA CONDITIONS)
052–044.19 (TDA & S/C)	QUALIFICATION TEST OF TDA – R & R "S/C GONE" SWITCH/ UMBILICAL	TO QUALIFY FOR SPACE USAGE THE TDA – R & R "S/C GONE" SWITCH/UMBILICAL
052–044.18 (TDA)	QUALIFICATION TEST OF RE-DESIGNED TDA LATCH RELEASE ACTUATOR	TO QUALIFY THE RE-DESIGNED LATCH RELEASE ACTUATOR FOR SPACE USAGE
052–044.20 (TDA & S/C)	FUNCTIONAL EVALUATION TESTS OF MODIFIED TDA – R & R	TO EVALUATE THE DOCKING FUNCTIONS OF THE TDA – R & R FOR REVISED RIGIDIZE AND UNRIGIDIZED SEQUENTIAL MODIFICATIONS
WORK REQUEST 7677.01	QUALIFICATION TEST OF TALLEY CORP. MODEL 938T100 MOORING DRIVE SYSTEM (52–34700)	TO QUALIFY THE MOORING DRIVE SYSTEM OF THE TDA FOR SPACE USAGE
WORK REQUEST 7677.02	QUALIFICATION TEST OF TALLEY CORP. MODEL 934T100 LATCH RELEASE ACTUATOR (52–34702)	TO QUALIFY THE LATCH RELEASE ACTUATOR OF THE TDA FOR SPACE USAGE
052–052.27	ULTIMATE STRENGTH STATIC TEST OF GEMINI-AGENA MANEUVERING	TO QUALIFY CRITICAL STRUCTURE OF GEMINI AND AGENA (TDA) UNDER ULTIMATE LOADS FOR ORBITAL MANEUVERING

Also the latch release actuator would not operate at 65°F unless 28 volts were applied. Failure analysis on the actuator indicated satisfactory operation to -7° but not below. The motor brake was found to be at fault by successful operation of the actuator at -80°F with the brake removed. A brake development program, conducted by failure analysis on the actuator, evolved the present flight configuration. The first fix was to remove Drilube from the brake spline and give the brake a metal-to-metal friction surface in place of the cork-to-metal friction surface. The final corrective action was to remove the brake and reset the clutch load to permit slippage each normal cycle.

Qualification of Talley Latch Release Actuator. - WR7677.02 - The first actuator (P/N 52-34702-3, S/N 105) failed humidity tests. Much internal corrosion was discovered by failure analysis, indicating that moisture seals were inadequate. Therefore, specific sealing methods were specified, but the reworked actuator again failed salt fog testing due to inadequate sealing of the actuator shaft. This was corrected by installing a boot over the shaft after rigging the shaft to a TDA. The humidity retest was then accomplished satisfactorily.

Mooring Drive and Latch Release Actuator Flight Evaluation Qual Test. TR052-044.07.06 - This testing superseded TR052-044.07, and consisted of exposing the talley components to salt fog, and then installing them in TDA static article No. 1 for vibration, temperature, and altitude response. The components subjected to salt fog failed. The output gearboxes and the H drive had great internal corrosion. The condition of the power unit and the flex shafts was acceptable, however. It was concluded that the corroded components needed grease protection rather than Drilube. A salt fog retest using reworked gearboxes and H drive was successful.

TDA Approach Light Qual Test. - TR052-044.11 - The TDA approach light is a commercial aircraft product, General Electric P/N 4502. Vibration tests resulted in the failure of the light element, caused by a light support hinge pin which had slipped halfway out of its bracket. A retest was conducted with the hinge pin staked in, but the element again failed. Sine sweeps were made to establish resonant frequencies, and shock mounts for these ranges were then provided. With this reworked installation the third vibration test was passed.

Functional Check Gemini/Agena Umbilical. - TR052-055.10 - Testing of the Gemini/Agena umbilical disclosed that three of the nine contacts exhibited discontinuity when subjected to side load. Although failure analysis concluded that a poor quality component was responsible, design tolerances were nevertheless revised.

In a second anomaly, the dielectric sleeve material distorted at high temperature and therefore required higher than normal forces for proper operation. It was established that Fiberglas would be a better material for the fabrication of the sleeve, since it possessed the same nonconductive properties but was immune to warping at high temperatures. These redesigned features were retested for high and low temperatures under side load and humidity environments. The umbilical performed satisfactorily in its retest.

TDA Communication System

The TDA communication system responds to Gemini Spacecraft radar interrogation signals and furnishes an L-band command link. The system consists of a radar transponder, two spiral antennas, a multielement dipole antenna, a boost regulator, and a dipole antenna boom (see Fig. 64). In addition to serving as a radar beacon, the transponder can detect L-band commands via RF prior to docking, and via hardline (through the umbilical) after docking.

TDA Status Display Panel

The TDA status display panel shows Gemini pilots the critical functions of the TDA and the Agena vehicles in the near-docked and docked configurations. The panel may be set on dim, bright, or off positions. Containing nine lights and three quantity indicators, the panel provides the following information:

A. DOCK Light (green) indicates that the docking cone is in the unrigidized position and the latches are reset.

B. RIGID Light (green) indicates that the docking cone is rigidized to the adapter.

C. PWR Light (green) indicates that the Agena electrical power is sufficient to perform all operations.

D. MAIN Light, showing red with the main engine firing, indicates that the turbine has exceeded the overspeed limits, the hydraulic pressure too low, or the differential pressure between the fuel and oxidizer tanks (fuel above oxidizer) is low. When the main engine is not firing, it indicates that the differential pressure between the fuel and oxidizer tanks (fuel above oxidizer) is improper.

E. MAIN Light, showing green, indicates that the hydraulic, main fuel, and oxidizer tank pressures are satisfactory.

F. ARMED Light (amber) indicates that the engine command circuits are closed and either the main or secondary engines may be fired by command.

G. SEC HI Light (green) indicates that both Agena nitrogen spheres have enough pressure for a specified Unit II (200 lb) firing and that both propellant tank gas manifolds have a specified pressure.

H. SEC LO Light indicates that sufficient pressure remains in both Agena nitrogen spheres for a specified Unit I (16 lb) firing and that pressure also exists in both propellant tank gas manifolds.

I. ATT Light indicates that the Agena attitude control system is active.

J. Main Time Indicator shows by min and sec the (PPS) engine burn time remaining in the Agena primary propulsion system.

K. Secondary Time Indicator shows by min and sec the burn time remaining in the secondary propulsion system (SPS). The clock runs at two speeds, depending upon whether the Unit I (16 lb, SPS LO) engines or the Unit II (200 lb, SPS HI) engines are burning.

L. Attitude Gas Indicator is a synchro clock which registers the percentage of pressure remaining in the Agena attitude control system gas sphere.

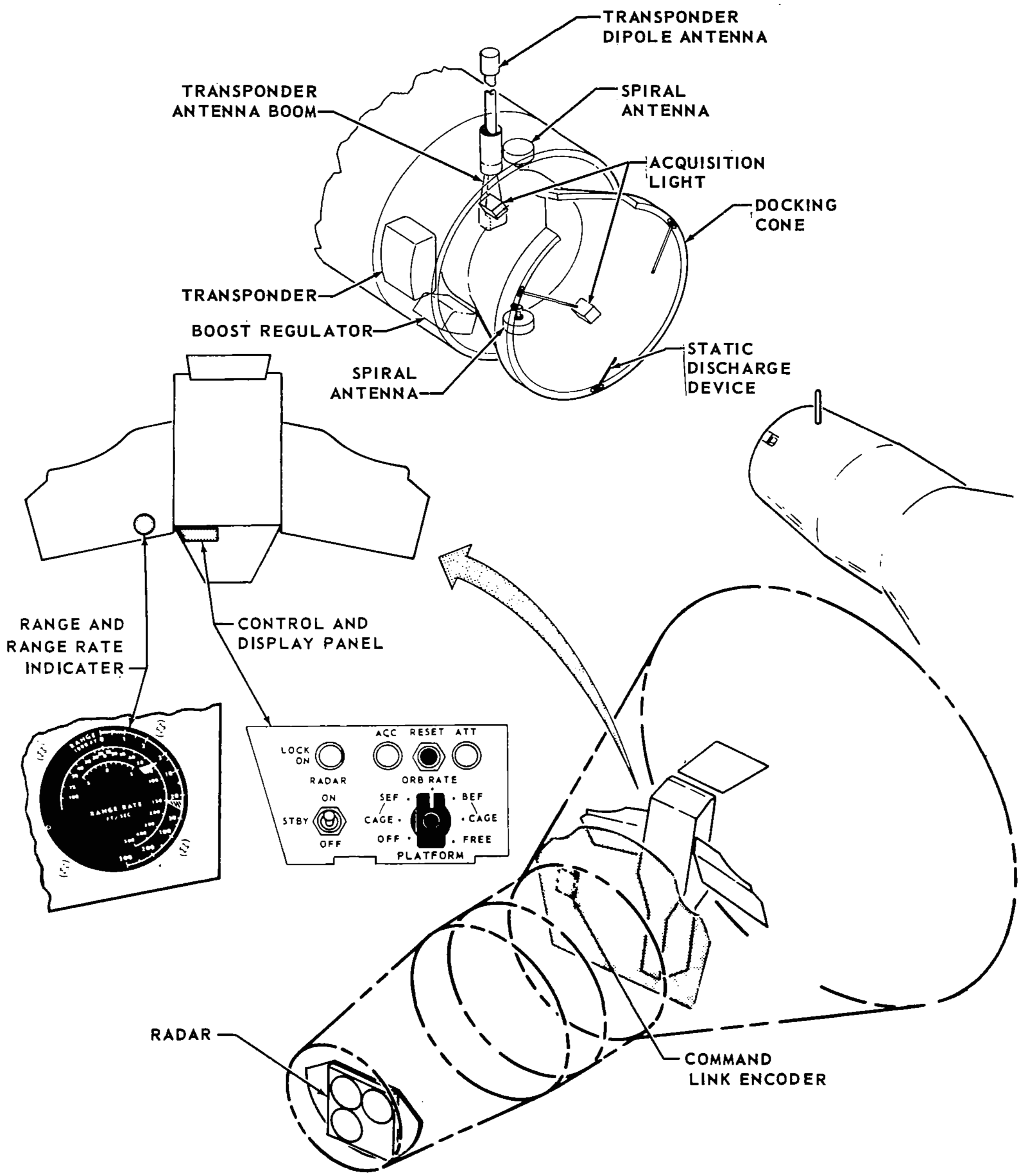


FIGURE 64 TDA COMMUNICATION SYSTEM

Electrostatic Discharge Device

A discharge device is mounted on the leading edge of the TDA docking cone to neutralize the electrostatic potential between the spacecraft and the target vehicle before final docking. On the Gemini IX and X missions (TDA 5 and TDA 1A), a monitor was used to measure this potential between the vehicles during docking; this data was transmitted to ground stations by the Agena telemetry system. Flight data revealed the electrostatic potential to be minimal; therefore the discharge device was deleted for the Gemini XI and Gemini XII missions.

Velcro Patches

On Gemini VIII and later missions, three velcro patches were mounted on the adapter structure in line with the top acquisition light to provide EVA handholds.

Experiments

During the Gemini program, two experiments were mounted on the TDA, the Agena micrometeorite collector (S-10) and the Gemini ion wake measurement (S-26).

S-10 Experiment. - The S-10 experiment is mounted on the TDA for the Gemini VIII through the Gemini X missions and for Gemini XII (TDA 1A, 3, 5 and 7A, and on TDA 4 for the ATDA). This experiment is described in EXPERIMENT S-10 (AGENA MICROMETEORITE COLLECTOR), page 318.

S-26 Experiment. - The S-26 experiment is mounted on the inside of the adapter structure on the Gemini X and the Gemini XI missions (TDA 1A, and 6). This experiment is described in EXPERIMENT S-26 (GEMINI ION WAKE MEASUREMENT), page 320.

Special Instrumentation

Temperature Sensors. - All TDA missions had four temperature sensor skin-mounted around the circumference of the TDA. The Gemini VI (TDA 2) and the Gemini VIII (TDA 3) missions had three additional sensors mounted on the TDA. On these latter missions, the three additional sensors were mounted in line with one of the other four sensors to measure launch temperature.

Accelerometers. - To measure acceleration during docking the Gemini VI (TDA 2) and the Gemini VIII (TDA 3) missions had three accelerometer/amplifier units, one to measure longitudinal acceleration and two to measure lateral acceleration.

Configuration Changes To TDA 7A For EVA Activity

A. A work station was added to the back surface of the docking cone. This work station contained a torque wrench, a test bolt, electrical disconnects, and fluid disconnects.

B. Four tether attachment rings were added to the lip of the docking cone.

C. Two handrails were added to the TDA/shroud fairing mating ring.

D. Sixteen chamfered holes were drilled in the TDA skin and a pip-pin attachment fitting was installed in each hole.

E. A receptacle was added to TDA cone for an extendible handrail from the spacecraft.

F. Five pip-pins were stowed on the TDA cone.

G. Two portable handholds were stowed on the TDA cone.

Gemini/Agena Tether. - The TDA was modified for the Gemini XI and XII missions by the addition of a Dacron strap which served as a tether. One end of the white Dacron tether line hooks to a redesigned TY damper attachment fitting on the Agena docking cone. The other end loops over the Gemini docking bar. The tether is stored on the Agena target docking adapter to permit its connection to successive spacecraft while docked. Deployment of the tether is accomplished automatically when the Gemini is separated from the Agena.

The purpose of the tether was to allow the performance of extended station-keeping operations between the Gemini and Agena vehicles. Such operations were performed by inducing an angular velocity about the common center-of-gravity of the system or by utilizing gravity gradient with the Agena toward the earth. Either method eliminated the possibility of contact between the two vehicles or their drifting apart.

TDA Mission Performance Summary

Gemini VI Mission. - TDA 2, used for the Gemini VI mission, performed as specified until the anomaly occurred on the Agena vehicle.

Gemini VIII Mission. - TDA 3, used for the Gemini VIII mission, performed as specified throughout the life of the Agena Target Vehicle. One problem was encountered with the unrigidizing and latch reset indications on Agena telemetry. After the spacecraft had unrigidized and separated from the target vehicle during the spacecraft thruster anomaly, Agena telemetry indicated that the TDA was neither rigidized nor unrigidized. T/M also indicated that the status display panel was on the BRIGHT setting, which meant that proper data regarding rigidizing should have been received. However, subsequent debriefing proved that the panel was in the DIM condition because the astronauts had not sent the BRIGHT command. After the crew cycled the mooring-drive system with the status panel on the BRIGHT setting, the proper indication of latch reset was received. The Agena T/M indication was therefore anomalous and no changes were made to the TDA as a result of this discrepant reading.

The astronauts reported a blurring condition which at times made reading of the status display panel difficult. Mission evaluation demonstrated, however, that this was an optical problem caused by the window configuration and was not ascribable to electrical anomaly.

Gemini IX Mission. - TDA 5, used on the Gemini IX mission, performed as specified up through the anomaly on the Atlas vehicle.

Gemini X Mission. - TDA 1A, used on the Gemini X mission, performed as specified.

Gemini XI Mission. - TDA 6, used on the Gemini XI mission, performed normally except for two anomalies. The transponder transmitter output began to degrade in the latter stages of first rendezvous and later failed. This anomaly is described in Spacecraft 11, page 92. At the time of the second undocking, an apparent mooring drive anomaly occurred. Post-separation telemetry data indicated that the TDA latches had not reset. This was confirmed by crew observation of the DOCK light on the GATV status panel. The crew transmitted the RF command to unrigidize the TDA, and the proper reset indication was obtained on the status panel through telemetry data. An investigation of data and an analysis of the system indicate that the latch actuator limit switch probably stopped after power interruption but before the actuation of the DOCK light and the telemetry limit switch. The possibility of the plunger's stopping consistently in this deadband region (approximately 0.005 inch) was considered highly unlikely; there is therefore no danger of actuator damage and the proper status indications can be recovered. No corrective action was consequently taken.

Gemini XII Mission. - TDA 7A, used on the Gemini XII mission, performed normally except that the transponder transmitter anomaly on TDA 6 reoccurred on TDA 7A. This anomaly is described in Spacecraft 12, page 92.

AUGMENTED TARGET DOCKING ADAPTER

General

The augmented target docking adapter (ATDA) provides a target vehicle for rendezvous and docking with the Gemini Spacecraft when an Agena target is not available. The ATDA is launched into its orbit by the Atlas standard launch vehicle (SLV).

The ATDA consists of five major structural sections: a launch shroud, a TDA, an equipment section, a modified Gemini Spacecraft re-entry control system (RCS), and a battery module (See Fig. 65 and 66).

Launch Shroud. - The launch shroud for the ATDA is the same as that used on the Gemini Agena Target (GAT) Vehicle, except for minor modifications to accommodate the slightly different launch trajectory. The shroud is jettisoned two sec prior to separation from the booster.

TDA. - The TDA used on the augmented target docking adapter is the same configuration as that used for the Gemini VIII mission (TDA 3), except that it has only four mutually independent temperature sensors mounted radially

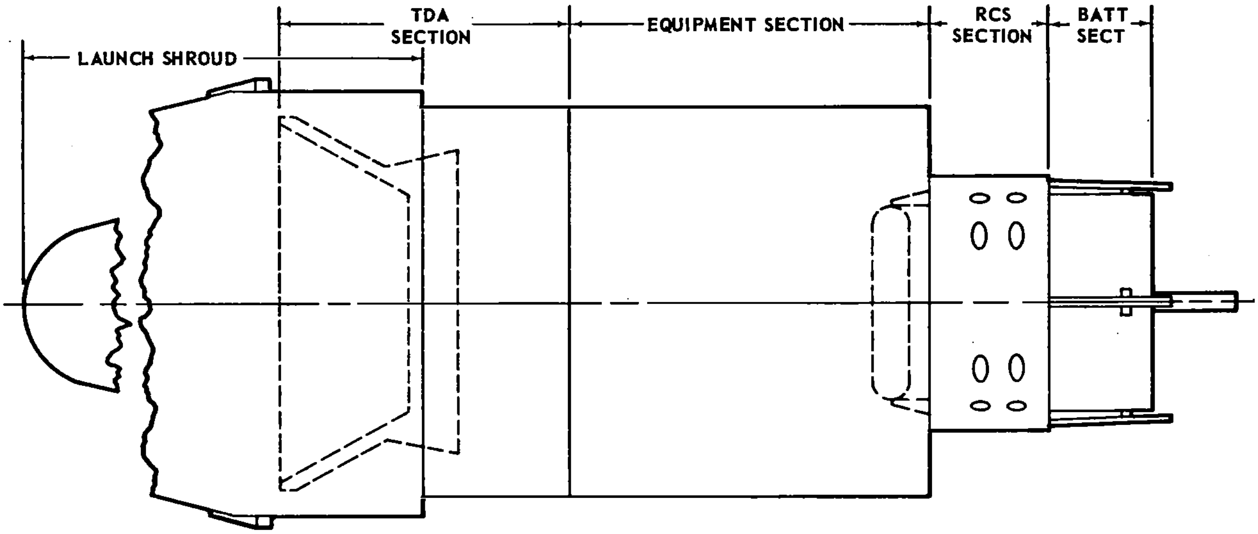


FIGURE 65 ATDA GENERAL ARRANGEMENT

on the adapter skin. Also the accelerometer and the amplifier sets were deleted and an ascent antenna is mounted on the docking cone. The docking cone approach light circuit is wired in series to reduce the voltage level.

Equipment Section. - The equipment section is a ring-stiffened cylindrical shell which is semimonocoque at the forward end (TDA) and monocoque aft. Two internal I-beams, extending across the diameter of the cylinder and intersecting at right angles near the vehicle centerline, support the ATDA electrical equipment. This equipment consists of flight qualified components which are mounted and wired by approved methods. Space for various experiments is provided in the forward end. (ATDA equipment locations are shown in Fig. 67.)

RCS Section. - The RCS section is similar to that used in the Gemini Spacecraft re-entry control system, except that its operation is modified by the different functions of the ATDA sequence. The section is qualified by Gemini Spacecraft standards.

Battery Section. - The battery section is of cross-beam construction and has individual compartments for each of the three primary batteries and the two squib batteries. The section is covered by a cylindrical can which protects the batteries and supports the separation system and the separation guide rails.

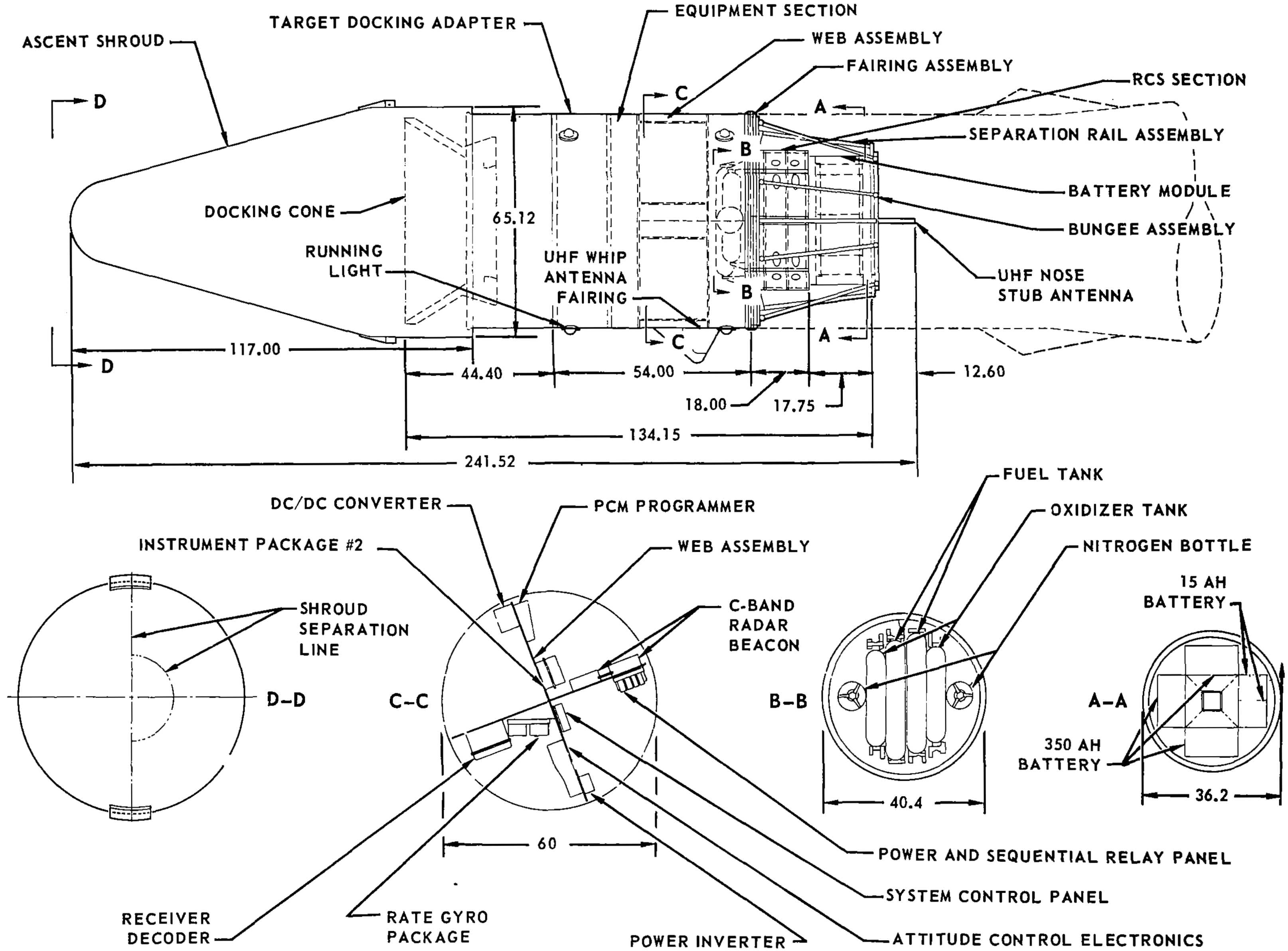


FIGURE 66 AUGMENTED TARGET DOCKING ADAPTER

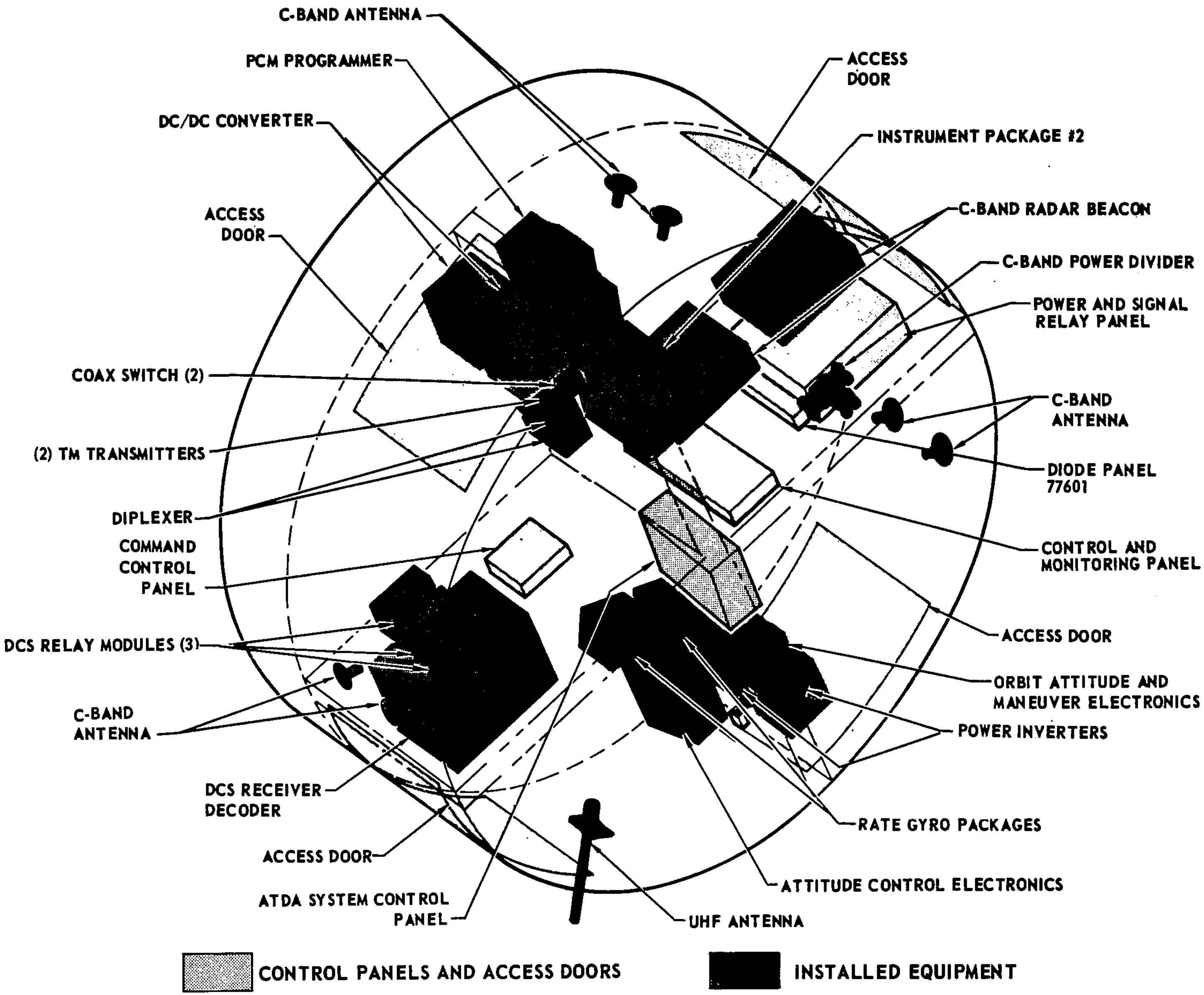


FIGURE 67 AUGMENTED TARGET DOCKING ADAPTER EQUIPMENT

Separation System

The ATDA separation system consists of two parts, (1) a flexible linear shaped charge (FLSC) at the interface of the ATDA shell and the booster adapter shell, and (2) an eight preloaded bungee system which is fixed to the booster adapter portion of the FLSC ring structure. (See Fig. 68.)

ATDA Display System

The display system provides for visual acquisition, indication, and monitoring of various ATDA functions.

Acquisition and Docking Cone Lights. - The lights are identical to those used on the TDA for the Gemini Agena Target Vehicle missions. The acquisition lights obtain electrical power from the ATDA common bus; the docking cone lights obtain power from the ATDA main bus.

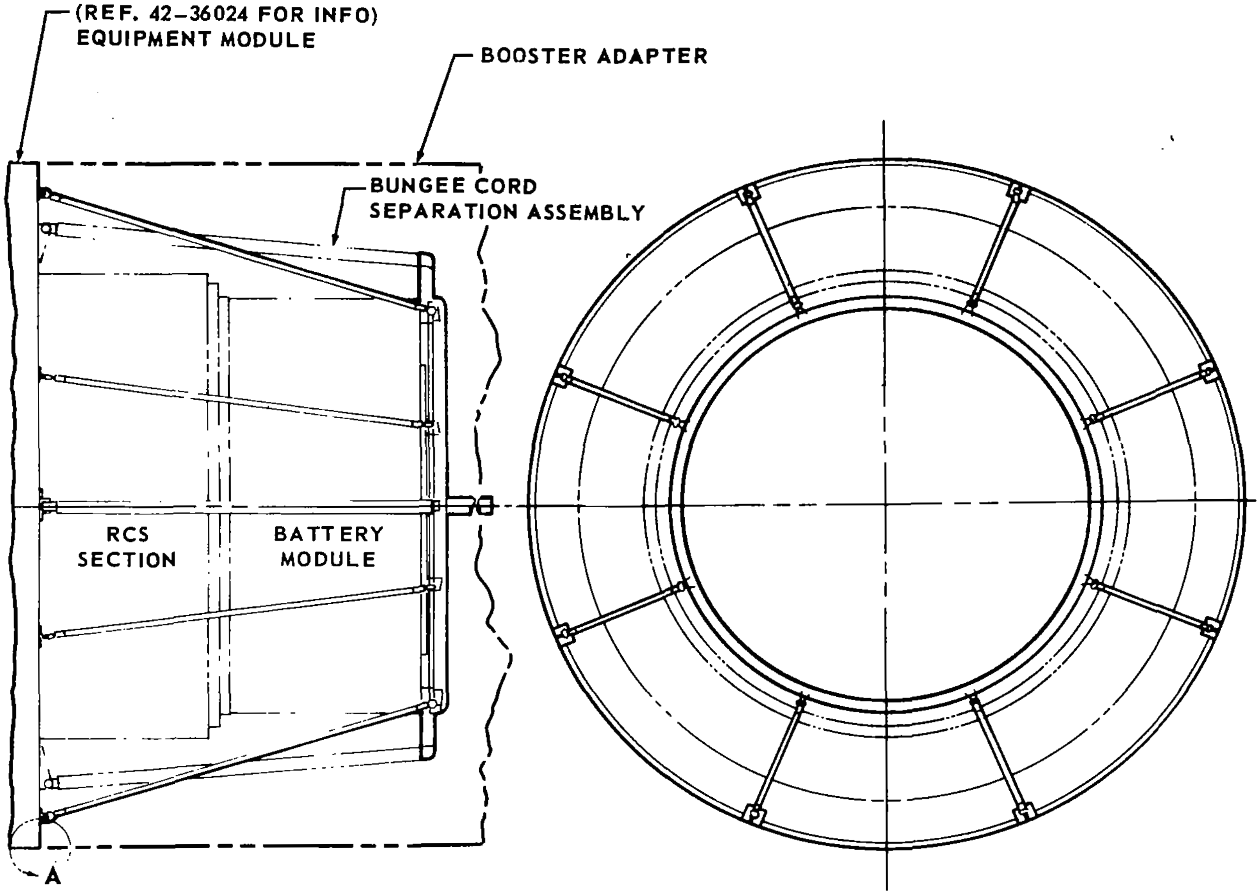
Running Lights. - Six running light assemblies, two red, two green, and two amber, are mounted on the ATDA equipment section to approximate the configuration used on the GAT vehicle.

Status Display Panel. - The ATDA status display panel is physically identical to the panel used on the TDA for the GAT missions and indicates only the following functions:

A. Dock status light (green) indicates that the docking cone is in the unrigidized position and that the latches are reset.
B. Rigid status light (green) indicates that the docking cone is in the rigidized position.
C. Armed status light (amber) indicates that ring A of the re-entry control system has been activated.
D. ATT status light (green) indicates that 0 degree per sec rate control has been selected in all three axes.

Target Stabilization System

The target stabilization system (TSS), in conjunction with the RCS, provides three axis rate stabilization of the ATDA. The TSS controls the ATDA to fixed turning rates about the pitch and roll axes and to rate damping about the yaw axis in the biased rate damping mode. The TSS rate damps the ATDA about all three axes in the normal rate damping mode. The rate damping characteristics in the biased rate damping mode for roll are $2.2^\circ \pm 0.5^\circ$/sec; for pitch, $1.8^\circ \pm 0.25^\circ$/sec; and for yaw, $0^\circ \pm 0.25^\circ$/sec. In the normal rate damping mode the characteristics for roll are $0^\circ \pm 0.50^\circ$/sec; for pitch and yaw they are $0^\circ \pm .25^\circ$/sec.



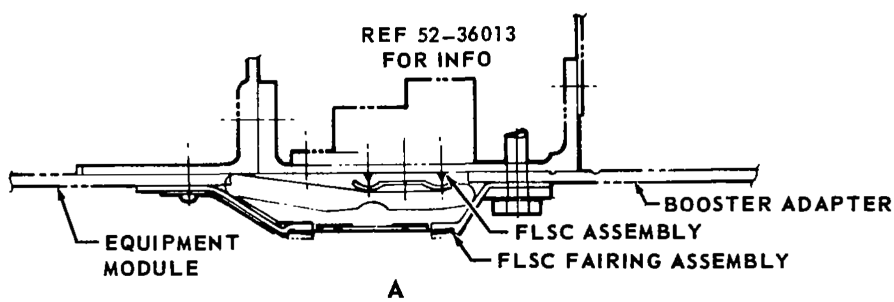


FIGURE 68 ATDA SEPARATION ASSEMBLY

Communication System

The communication system consists of a tracking system, a telemetry system, a digital command system, and a rendezvous radar transponder system. The communication system provides information, acquisition, and command links between the spacecraft and the ATDA, and between the ground and the ATDA.

Tracking System. - The tracking system consists of redundant C-band radar transponders and antennas. The transponder responds to the ground tracking radar stations and either transponder can be energized by ground command via the digital command system.

Telemetry System. - The ATDA telemetry system consists of two T/M transmitters operating with two diplexers, three coaxial switches and three antennas to accomplish real-time T/M transmission to ground stations. The components in the T/M system are energized by commands from the digital command system (DCS).

Digital Command System. - The digital command system provides a real-time command link for the spacecraft and permits ground control of the ATDA. The command system consists of a receiver-decoder unit and three relay units.

Rendezvous Radar Transponder System. - The rendezvous radar transponder system is identical to that used on the GAT vehicle except that there is only a single command capability in both hardline link and RF link.

Development And Qualification Tests

The development and qualification tests listed in Table 36 apply also to the TDA used on the ATDA. In addition, the following tests were performed on this TDA. (TDA 4) Results were satisfactory.

Structural Development Test. - TR052-005.21 (determination of load relaxation of bungee cord) was instituted to determine the work output capability of bungee cord after exposure to various temperatures and durations of sustained loading.

Structural Qualification Tests. - TR052-080.09 (SST vibration test of the simplified target vehicle (ATDA)) was run to demonstrate the capability of the ATDA to function properly during and after a normal launch vibration test.

Antenna Patterns. - Antenna patterns were run at St. Louis on the ATDA VHF antennas using a one-third scale mock-up. Patterns were also run at Cape Kennedy on the C-band antenna, using a full scale mock-up. Testing showed all antenna patterns to be satisfactory for the mission.

ATDA Mission Performance Summary

The ATDA was launched as the target vehicle for the Gemini IXA mission after the Agena Target Vehicle for the Gemini IX mission failed to achieve orbit.

The ATDA was placed in a nearly circular (161 nautical mile) orbit, but a shroud separation indication was not received. Examination by the Gemini IXA crew determined that the shroud had not properly ejected, and had failed to expose the docking cone of the TDA. The shroud was partly open but the forward band was held by the pyrotechnic wire bundle to each igniter. The two halves of the shroud were still attached to the TDA by the wire bundles between the shroud and the docking adapter section of the TDA. The docking maneuver of the Gemini IXA spacecraft with the ATDA was therefore not possible.