11 MISSION SUPPORT PERFORMANCE

11.1 FLIGHT CONTROL

Flight control provided satisfactory operational support for all required areas during the Apollo 15 mission. A number of the problems that were encountered are discussed elsewhere in this report. Only those problems that are unique to flight control, or have operational considerations not previously mentioned, are presented in this section.

A radial velocity error in the launch vehicle guidance system at earth-orbit insertion necessitated a navigation update to minimize the subsequent planned midcourse correction. Without the update, a 32-ft/sec velocity change would have been required at 9 hours. After updating the state vector, the actual midcourse correction was approximately 5 ft/sec (See section 6.5.)

As a result of the service propulsion system thrust light anomaly discussed previously, the crew was requested to deactivate both pilot valve circuit breakers immediately after the light was first observed. This measure was instituted to safeguard against an inadvertent firing until the problem could be thoroughly understood. To isolate the cause of the Malfunction, a test was conducted in conjunction with the first midcourse correction and the problem was resolved, including the development of workaround procedures. The crew was instructed to power down the entry monitor system scroll in order to eliminate the nuisance factor of a constant false light indication until the use of the entry monitor system was required.

During the first period of scientific instrument module activity for film advancement, the ground station (Madrid) had a problem in locking onto the FM subcarrier. This was determined to be a site procedural problem. All sites were briefed on the problem and no subsequent problems were encountered.

After lunar module ingress and the crew's description of the broken glass cover on the range/range rate tapemeter, ground tests were performed to verify that the tapemeter would function properly with the glass broken, exposing the inside of the instrument to the cabin atmosphere. A careful review of procedures was made to evaluate crew monitoring techniques during descent. A technique was developed to use the abort guidance system for displaying raw landing radar altitude data should the tapemeter and the primary guidance and navigation system fail, but the technique was not voiced to the crew.

At acquisition of signal during the 12th lunar revolution, the lunar module crew reported that they had been unable to separate from the command and service module and that the Command Module Pilot was investigating the probe umbilical integrity. An off-scale high docking probe temperature was indicative of a possible umbilical problem. The umbilicals were found to be the source of the problem, and the condition was corrected. Meanwhile, the crew had been advised by Mission Control that undocking and separation were not time critical. The separation was achieved about 36 minutes late.

Landmark tracking was deleted during the umbilical integrity problem, but adequate data were later obtained when the command and service module was in a higher orbit.

During the sleep period after the standup extravehicular activity, the descent oxygen was being depleted at a rate about 1 pound/hour greater than expected. The oxygen quantity was not critical, but the descent oxygen tank pressure was critical to allow a full portable life support system recharge for the third extravehicular activity, The crew was awakened approximately 1 hour early to locate the leak. They found that the leak was caused by the urine receptacle device being inadvertently left open. The early completion of this task allowed preparations for the first extravehicular activity to start about 20 minutes early.

11.2 NETWORK

Adequate support was provided by the Mission Control Center and Manned Space Flight Network. Although a number of minor discrepancies and problems arose during the mission, there was no interruption of mission support. The most significant problem, in terms of potential impact, was an error in a command module computer delta-velocity return update. The error was not discovered until after the load had been transmitted to the spacecraft. A different load was then generated and transmitted to correct the data in the command module computer. A correction to the software will be made for the next mission.

11.3 RECOVERY OPERATIONS

The Department of Defense provided recovery support in accordance with the mission planning for Apollo 15. Ship support for the primary landing area in the Pacific Ocean was provided by the helicopter carrier USS Okinawa. Active air support consisted of five SH-3G helicopters from the Okinawa and two HC-130 rescue aircraft staged from Hickam Air Force Base, Hawaii. Two of the helicopters, designated "Swim 1" and "Swim 2", carried underwater demolition team personnel and the required recovery equipment. The third helicopter, designated "Recovery," carried the flight surgeon and was utilized for the retrieval of the flight crew. The fourth heliccpter, designated "Photo," served as a photographic platform for both motion picture photography and live television coverage. The fifth helicopter, designated "Relay," served as a communications relay aircraft. The two HC-130 aircraft, designated "Hawaii Rescue 1" and "Hawaii Rescue 2," were positioned to track the command module after it had exited from S-band blackout, as well as to provide pararescue capability had the command module landed uprange or downrange of the target point. The inset in **Figure 11-1** indicates the relative positions of the recovery ship and HC-130 aircraft prior to landing. The recovery forces assigned to the Apollo 15 mission are shown in **Table 11-I**.

TABLE 11-I.- APOLLO 15 RECOVERY SUPPORT

Type	Number	Ship name/ aircraft staging base	Area supported
		Ships	
AFT	1	USS Salinan	Launch site area
LCU	1		
LPD	1	USS Austin	Launch abort area and West Atlantic earth-orbital recovery zone
LPH	1	USS Okinawa	Deep-space secondary landing areas on the Mid-Pacific line and the primary end-of-mission landing area
AO	1	USS Kawishiwi	Provided refueling for USS Okinawa
		Aircraft	
HC-130	[a]1	Eglin Air Force Base	Launch abort area, West Atlantic recovery zone, contingency landing area
HC-130	[a]1	Pease Air Force Base	Launch abort area
HC-130	[a]1	Lajes Field, Azores	Launch abort area, earth-orbital contingency landing area, deep-space abort landings
HC-130	[a]2	Hickam Air Force Base	Mid-Pacific earth orbital recovery zone, deep-space secondary landing area and primary end-of-mission landing area
HC-53C	3	Patrick Air Force Base	Launch site area to 1000 miles down range
SH-3G	5	USS Okinawa	Deep-space secondary landing area and primary end-of-mission landing area

[a]Plus one backup

11.3.1 Command Module Location and Retrieval

Based upon a navigation satellite (SRN-9) fix obtained at 2o46 G.m.t., August 7, the Okinawa's position at the time of command module landing was determined to be 26 degrees 12 minutes 54 seconds north latitude and 158 degrees 13 minutes 12 seconds west longitude. The ship-based aircraft were initially positioned with respect to the target point as shown in **Figure 11-1**, and they departed station to commence recovery operations after visual contact had been made with the command module. The Okinawa is shown in **Figure 11-1** as it was positioned at the time of command module landing.

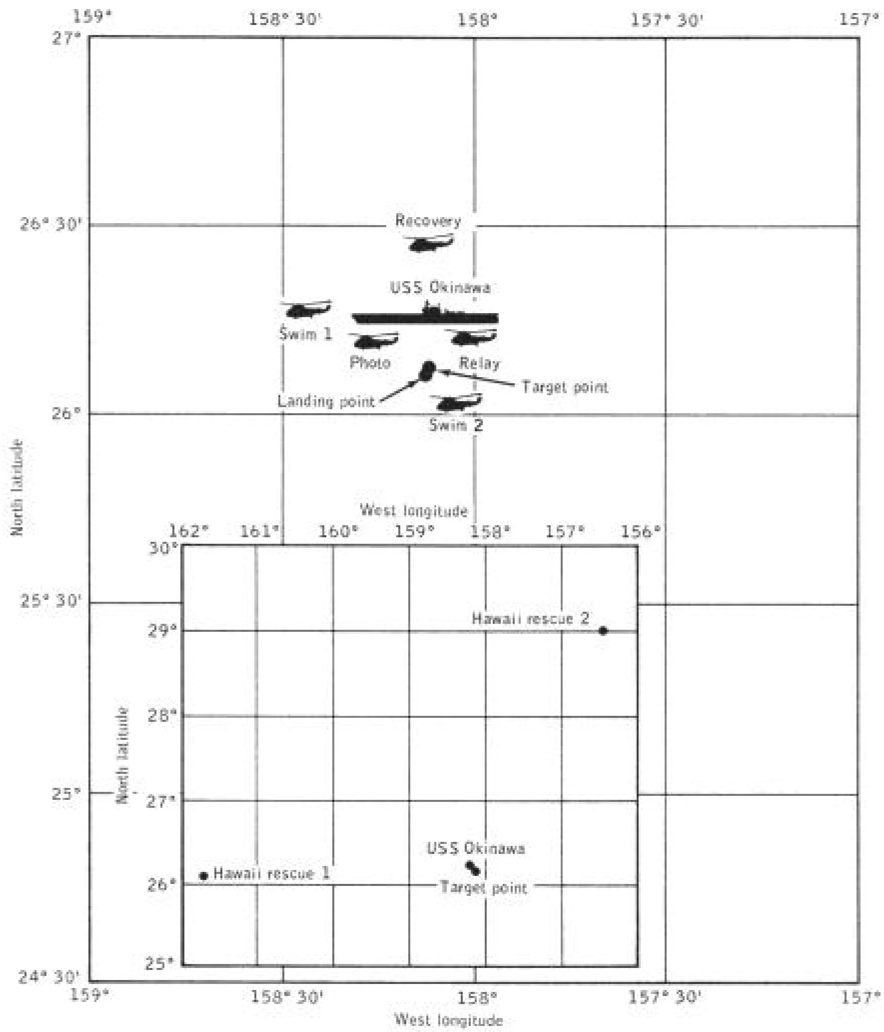

Figure 11-1.- End-of-mission recovery support.

Radar contact with the command module was first reported by the Okinawa at 2037 G.m.t. This was followed by an S-band reported by Hawaii Rescue 1 at 2038 G.m.t. and VHF recovery beacon contact by the Okinawa at 2041 G.m.t. At approximately the same time, all recovery force aircraft established VHF recovery beacon contacts. Shortly thereafter, two-way voice communication was established between the Apollo 15 crew and recovery forces.

Visual sighting of the command module occurred at 2041 G.m.t. by the Swim 2, Photo, and Relay helicopters. At the time of initial sighting, the command module was descending on three normal main parachutes. At least two pilots, in different aircraft, saw one main parachute stream at about 6000 feet.

The command module with the two main parachutes properly inflated and one collapsed, landed at 2045:53 G.m.t., approximately 32 seconds earlier than predicted, and remained in the Stable I flotation attitude. The landing point was calculated using the navigation satellite fix of the shin's position at spacecraft landing and a radar sighting which established that the command module was 6.6 miles distant on a bearing of 145 degrees east of north. Based upon these data, the landing point coordinates were 26 degrees 7 minutes 30 seconds north latitude and 158 degrees 9 minutes west longitude.

After a visual inspection of the command module and assurance from the crew that they were all in good condition, the Swim 2 helicopter managed to secure one of the main parachutes before dropping swimmers who installed the flotation collar on the command module. The Swim 1 helicopter dropped swimmers to secure a life raft to the forward heat shield. One main parachute and the heat shield were retrieved.

The flight crew was delivered aboard the USS Okinawa by the recovery helicopter at 2125 G.m.t. No quarantine procedures were required for this mission. Command module retrieval took place at 26 degrees 7 minutes north latitude, 158 degrees 10 minutes 12 seconds west longitude at 2220 G.m.t. on August 7.

he flight crew remained aboard the Okinawa until 1655 G.m.t., August 8, and were then flown to Hickam Air Force Base, Hawaii. After a brief welcoming ceremony, a C-141 aircraft flew them to Ellington Air Force Base, Texas.

The command module was offloaded at North Island Naval Air Station, San Diego, California, on August 17. It was deactivated and delivered to Downey, California, on August 20.

The following page shows a chronological listing of events during the recovery and postrecovery operations. (Figure)

Event	Time, G.m.t.	Time relative to landing, days:hr:min
	August 7, 1971	
Radar contact by Okinawa	2037	-0:00:09
S-band contact by Hawaii Rescue 1	2038	-0:00:08
VHF recovery beacon contact by Okinawa	2041	-0.00:05
Visual contact	2041	-0:00:05
Voice contact with Apollo 15 crew	2043	-0:00:03
Command module landing	2046	0:00:00
Swimmers deployed to command module	2052	0:00:06
Flotation collar installed and inflated	2100	0:00:14
Hatch opened for crew egress	2111	0:00:25
Flight crew in egress raft	2112	0:00:26
Flight crew aboard helicopter	2120	0:00:34
Flight crew aboard Okinawa	2125	0:00:39
Command module aboard Okinawa	2220	0:01:34
	August 8, 1971	
First sample flight departed ship	0330	0:06:44
First sample flight arrived Hawaii	0520	0:08:34
First sample flight departed Hawaii	0604	0:09:14
First sample flight arrived Houston	1333	0:16:43
Flight crew departed Okinawa	1655	0:20:05
Flight crew arrived Hawaii	1710	0:20:20
Flight crew departed Hawaii	1746	0:20:56
Command module arrived Hawaii	1800	0:21:10
	August 9, 1971	
Flight crew arrived Houston	0219	1:05:29
	August 11, 1971	
Command module departed Hawaii	1800	3:21:10
	August 17, 1971	
Command module arrived North Island, San Diego, California	0000	9:03:10
Pyrotechnic safing complete	0740	9:10:50
	August 18, 1971	
Fuel portion of reaction control system deactivated	2330	11:02:40
	August 19, 1971	
Oxidizer portion of reaction control deactivated	2320	12:02:30
	August 20, 1971	
Command module departed San Diego	0220	12:05:30
Command module arrived Downey	1715	12:20:25

11.3.2 Postrecovery Inspection

Visual inspection of the command module in the recovery area revealed the following minor discrepancies:

One of the VHF-antenna ground planes was damaged.

The O-ring that seals the forward heat shield cover to the tunnel was loose.

The docking ring was slightly damaged, as generally occurs

A section of the aft heat shield, approximately 12 inches by 14 inches by 1/2-inch deep, was missing. (This generally occurs to some degree from the water impact.)

12 ASSESSMENT OF MISSION OBJECTIVES

The four primary objectives (ref. 4) assigned to the Apollo 15 mission were:

Perform selenological inspection, survey, and sampling of materials and surface features in a pre-selected area of the Hadley-Appenine region.

Emplace and activate surface experiments.

Evaluate the capability of the Apollo equipment to provide extended lunar surface stay time, increased extravehicular operations, and surface mobility.

Conduct inflight experiments and photographic tasks from lunar orbit.

Twelve detailed objectives (derived from primary objectives) and twenty-four experiments (listed in **Table 12-1** and **Table 12-1 Concluded** and described in reference) were assigned to the mission. Preliminary indications are that adequate data were obtained to successfully complete all objectives.

The Manned Spacecraft Center participated in two of nine approved operational tests. The two operational tests were: lunar gravity measurement (using the lunar module primary guidance system) and a lunar module voice and data relay test (a lunar module and Manned Space Flight Network test of voice and portable life support system data from the Lunar Module Pilot). Both tests were completed.

The other seven tests were performed for the Department of Defense and the Kennedy Space Center. These tests are designated as follows:

Chapel Bell (classified Department of Defense test)

Radar skin tracking

Ionospheric disturbance from missiles

Acoustic measurement of missile exhaust noise

Army acoustic test

Long-focal-length optical system

Sonic boom measurement

TABLE 12-I.- DETAILED OBJECTIVES AND EXPERIMENTS

Description	Completed
Detailed Objectives	
Lunar roving vehicle evaluation	Yes
Extravehicular communications with the lunar communications relay unit and ground controlled television assembly	Yes
Extravehicular mobility unit assessment on lunar surface	Yes
Lunar module landing effects evaluation	Yes
Service module orbital photographic tasks	Yes
Command module photographic tasks	Yes
Scientific instrument module thermal data	Yes
Scientific instrument module inspection during extravehicular activity	Yes
Scientific instrument module door jettison evaluation	Yes
Lunar module descent engine performance	Yes
Visual observations from lunar orbit	Yes
Visual light flash phenomenon	Yes
Experiments	
Contingency sample collection	Yes
Passive seismic (S-031)	Yes
Heat flow (S-037)	Yes
Lunar surface magnetometer (S-034)	Yes
Solar wind spectrometer (S-035)	Yes
Suprathermal ion detector (S-036)	Yes
Cold cathode gage (S-058)	Yes
Lunar dust detector (M-515)	Yes
Lunar geology investigation (S-059)	Yes
Laser ranging retro-reflector (S-078)	Yes
Soil mechanics (S-200)	Yes
Gamma-ray spectrometer (S-160)	Yes
X-ray fluorescence (S-161)	Yes
Alpha particle spectrometer (S-162)	Yes
Subsatellite:	
S-band transponder (S-164)	Yes
Particle shadows/boundary layer (S-173)	Yes
Magnetometer (S-174)	Yes

Description	Completed
Experiments	
Mass spectrometer (S-165)	Yes
Down-link bistatic radar observations of the moon (S-170)	Yes
Ultraviolet photography - earth and moon (S-177)	Yes
Gegenschein from lunar orbit (S-178)	[a]No
S-band transponder (command and service module/ lunar module) (S-164)	Yes
Solar wind composition (S-080)	Yes
Apollo window meteoroid (S-176)	Yes
Bone mineral measurement (M-078)	Yes

TABLE 12-1. - DETAILED OBJECTIVES AND EXPERIMENTS (Concluded)

a Fourteen 35-mm photographs were scheduled for the Gegenschein from lunar orbit experiment. None of the photographs were obtained because of an error in the spacecraft photographic attitudes used. The error was incurred during the analytical transformation of the target coordinates to spacecraft attitudes.

13 LAUNCH PHASE SUMMARY

13.1 WEATHER CONDITIONS

Gentle to moderate southerly winds extended from the surface to an altitude of 25 000 feet at launch time. The maximum wind was 37 knots at 45 000 feet. Broken thin cirrostratus; clouds covered much of the sky at 25 000 feet, but no low or middle clouds were observed. Surface temperature was 83 F and visibility was 10 miles.

13.2 ATMOSPHERIC ELECTRICITY

The Apollo 15 launch complex was struck by lightning on five different days during the checkout activities. In all, eleven separate strikes were recorded between June 14 and July 21, 1971. The direct damage incurred was limited to the command and service module ground support equipment.

Launch site lightning strikes have not been documented for program other than Apollo and Gemini. Incidents reported for these two programs are as follows:

On August 17, 1964, a lightning strike occurred near, but not directly on, launch complex 19. The Gemini II vehicle was visually inspected and found to be free of lightning-type markings or burns. Later, some failed components were discovered which were believed to have resulted from the lightning incident.

The first Apollo strike occurred in September 1964. The launch umbilical tower was struck on launch complex 34 while the S-I launch vehicle was being prepared for flight. No failures were reported.

The second, and final, Gemini incident was a strike near the launch complex 19 power facility during the Gemini V countdown on August 19, 1965. No lightning-relatea damage was reported.

At least one lightning strike occurred on the mobile launcher for the AS-201 vehicle. This occurred sometime between August 9 and August 27, 1965, at launch complex 39. Only the S-IB stage of the space vehicle could have been on the pad at the time and no lightning damage was reported.

At least two strikes were recorded on the mobile launcher for the AS-50OF vehicle at launch complex 39A. This was a non-flight facility test vehicle. The first incident occurred on May 27, 1966. On June 21, 1966, magnetic recording links on the mobile launcher were examined and the reading indicated at least one strike had occurred between May 27 and June 21, 1966. During this time, the complete launch vehicle for the AS-202 mission was on launch pad 37B and the complete launch vehicle for the AS-203 mission was on launch pad 34. No lightning activity associated with the latter two complexes was reported.

On July 27, 1967, a lightning strike occurred at launch complex 37B. The complete launch vehicle was on the pad at the time of the incident. Ground support equipment damage was found at a later date, which may have been caused by the lightning.

The only other lightning strike reported prior to the Apollo 15 prelaunch activities occurred during the launch phase of the Apollo 12 mission. This occurrence is documented in reference 6.

Existing weather data were examined for the May-through-July periods from 1966 to the launch of Apollo 15. Reported cloud-to-ground lightning strikes for a period of 90 days within the general vicinity of Cape Kennedy showed the daily average to be as follows:

	Daily average number of strikes					
Year	1966	1967	1968	1969	1970	1971
Daily average	0.598	0.837	0.380	0.239	1.142	1.613

(Figure)

Thunderstorms are more prevalent on the west side of the Indian River and remain west of the launch pad. During the summer of 19T1, however, the west winds prevailed more frequently than the preceding several years, thus causing the thunderstorms to move east. The lightning density in conjunction with the general easterly movement of the storms contributed to the number of strikes being higher than in the past.

13.3 LAUNCH VEHICLE PERFORMANCE

The eighth manned Saturn V Apollo space vehicle, AS-510, was launched on an azimuth 90 degrees east of north. A roll maneuver was initiated at 12.2 seconds after lift-off and the vehicle was placed on a flight azimuth of about 80 degrees. The trajectory parameters from launch through translunar injection were nominal. Earth-parking-orbit insertion conditions were achieved 4.4 seconds earlier than planned.

The performance of the S-IC propulsion system was satisfactory and the specific impulse and mixture ratios were near the predicted values. Four of the eight S-IC retromotors and all of the S-II stage ullage motors were removed for this flight; therefore, the S-IC/S-II separation sequence was revised. This sequence change extended the coast period between S-IC outboard engine cutoff and S-II engine start command by one second. The S-IC/S-II separation sequence and S-II engine thrust buildup performance was satisfactory.

The S-II propulsion system performed normally. The specific impulse and mixture ratios were near predicted values. This was the second S-II stage to incorporate a center-engine liquid-oxygen feedline accumulator as a longitudinal oscillation (POGO) suppression device. The operation of the accumulator system was effective in suppressing these types of oscillations.

The S-IVB stage J-2 engine operated satisfactorily throughout the first and second firings and had normal start and cutoff transients. The firing time for the first S-IVB firing was 141.5 seconds, 3.8 seconds less than predicted. Approximately 2.6 seconds of the shorter firing time can be attributed to higher than predicted S-IVB performance. The remainder can be attributed to S-IC and S-II stage performances. The specific impulse and engine mixture ratio were near the predicted values.

Abnormal temperatures were noted in the turbine hot gas system between the first S-IVB firing engine cutoff and second firing engine start command. Most noticeable was the fuel turbine inlet temperature. During liquid hydrogen chilldown, this temperature decreased from +130 to -100 F at the time of the second engine start command. The oxidizer turbine inlet temperature also indicated a small decrease. In addition, the fuel turbine inlet temperature indicated an abnormally fast decrease after engine cutoff for the first firing. A possible cause of the decrease in turbine inlet temperature was a small leak past the gas generator fuel inlet valve.

The S-IVB firing time for translunar injection was 350.8 seconds. Upon completion of the spacecraft separation, transposition, docking, and extraction operations, the S-IVB evasive maneuver was performed and, subsequently, the vehicle was placed on a trajectory to impact the lunar surface in the vicinity of the Apollo 14 landing site. The S-IVB/instrumentation unit impacted the lunar surface at 1 degree 31 minutes south latitude and 11 degrees 49 minutes west longitude with a velocity of 8455 ft/sec. This impact point is approximately 146 kilometers (79 miles) from the target of 3 degrees 39 minutes south latitude and 7 degrees 35 minutes west longitude. Although the impact location was not within the preferred region, scientific data were obtained from the impact.

The impact point projected from the first auxiliary propulsion system maneuver was perturbed by unplanned attitude control thrusting that occurred to counteract forces resulting from a water leak in the sublimator. Following the second auxiliary propulsion system maneuver, the small and gradually decreasing unbalanced force from the sublimator water leak continued to act for a period of 5 hours and further perturbed the point of impact.

The structural loads experienced during the S-IC boost phase were well below design values. Thrust cutoff transients experienced were similar to those of previous flights. During S-IC stage boost, 4- to 5hertz oscillations were detected beginning at approximately 100 seconds. The maximum amplitude measured at the instrumentation unit was ±0.06g. Oscillations in the 4- to 5-hertz range have been observed on previous flights. The structural loads experienced during the S-IVB stage firings were well below design values.

The guidance and navigation system provided satisfactory end conditions for the earth parking orbit and translunar injection. The control system was different from that of Apollo 14 because of redesigned filters and a revised gain schedule. These changes were made to stabilize structural dynamics caused by vehicle mass and structural changes and to improve wind and engine-out characteristics.

The launch vehicle electrical systems and emergency detection system performed satisfactorily throughout all phases of flight. Operation of the batteries, power supplies, inverters, exploding bridge wire firing units, and switch selectors was normal. Vehicle pressure and thermal environments in general were similar to those experienced on earlier flights. The environmental control system performance was satisfactory. All data systems performed satisfactorily through the flight.

More details of the launch vehicle operation and performance are given in reference 1.

14 ANOMALY SUMMARY

This section contains a discussion of the significant anomalies that occurred during the Apollo 15 mission. The discussion is divided into six major areas: command and service modules; lunar module; scientific instrument module experiments; Apollo lunar surface experiments package and associated equipment; government-furnished equipment; and the lunar roving vehicle.

14.1 COMMAND AND SERVICE MODULES

14.1.1 Service Module Reaction Control System Propellant Isolation Valves Closed

During postinsertion checks, the quad B secondary isolation valve talkback indicated that the valve was closed, and the switch was cycled to open it. Subsequently, talkbacks indicated that both the primary and secondary valves for quad D were also closed, and these valves were reopened. At S-IVB separation (approximately 3 hours 22 minutes), all the aforementioned valves closed and were reopened. Upon jettisoning of the scientific instrument module door, the quad B secondary valve closed and was reopened.

This type of valve (a magnetic latching valve, shown in **Fig. 14-1**) has, in previous missions, closed as a result of pyrotechnic shocks. Ground tests have shown that the valve will close at a shock level of approximately 80g sustained for 8 to 10 milliseconds. There were no indications of shock levels of the magnitude required to close the valve during launch.

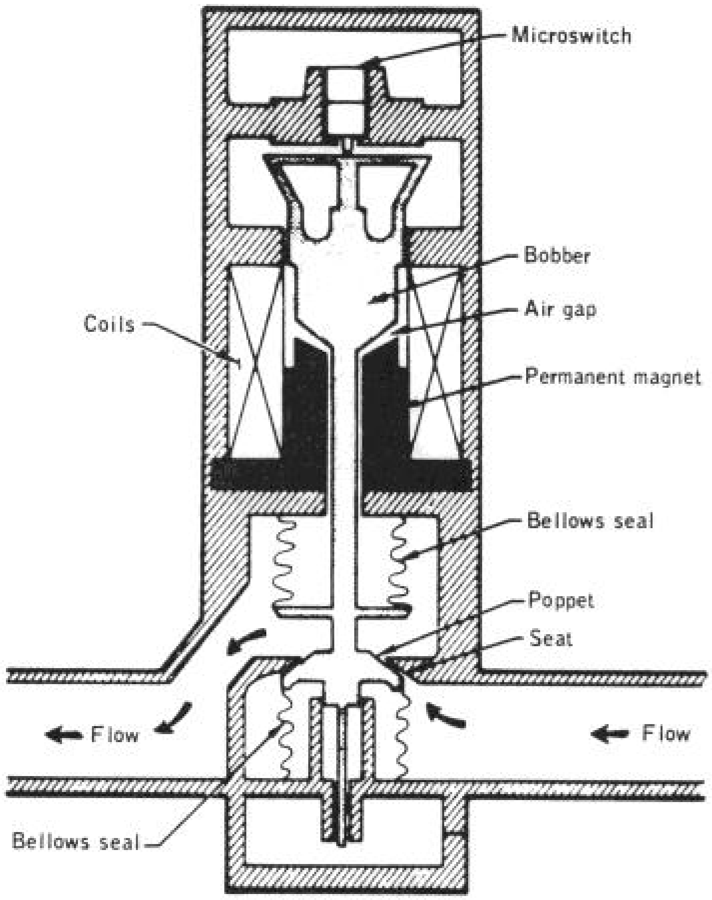

Figure 14-1.- Cross section of reaction control system isolation valve.

Testing has shown that if a reversed voltage of 28 volts is applied to the valve, the latching voltage will drop to a point where the valve will no longer remain latched (magnet completely degaussed). In addition, at lower voltages with reversed polarity, the magnet would become partially degaussed.

During acceptance testing of one valve for command and service module 117, the latching voltage had changed from approximately 13 volts to 3 volts. Additional testing of the spacecraft 117 valve verified the low voltage condition. Additionally, the valve stroke was proper, thereby eliminating contamination as a possible cause of the problem. During the test, the valve was disconnected from spacecraft power (28 volts) and was being supplied power through a variable power supply (approximately 20 volts, maximum, applied to the valve). The valve was most likely subjected to a reversed polarity at a voltage level which would partially degauss the magnet. This may have been the cause of the valve closures during Apollo 15 launch phase.

A magnetic latching force test was not performed on the valves after assembly into the system for the Apollo 15 command and service module, as on some previous spacecraft. A test will be performed on subsequent assemblies to verify that the valve latching forces are acceptable.

This anomaly is closed.

14.1.2 Water Panel Chlorine Injection Port Leakage

Minor leakage was noted from the chlorination injection port when the cap was removed to perform the prelaunch water chlorination. The cap was reinstalled and the leak ceased. A leak of approximately 1 quart in 20 minutes also was noted at the chlorine injection port as the crew removed the injection port cap for the third injection at about 61 hours. The crew tightened the septum retention insert (**Fig. 14-2**) and satisfactorily stopped the leakage. Leakage recurred at about 204 1/2 hours and was corrected.

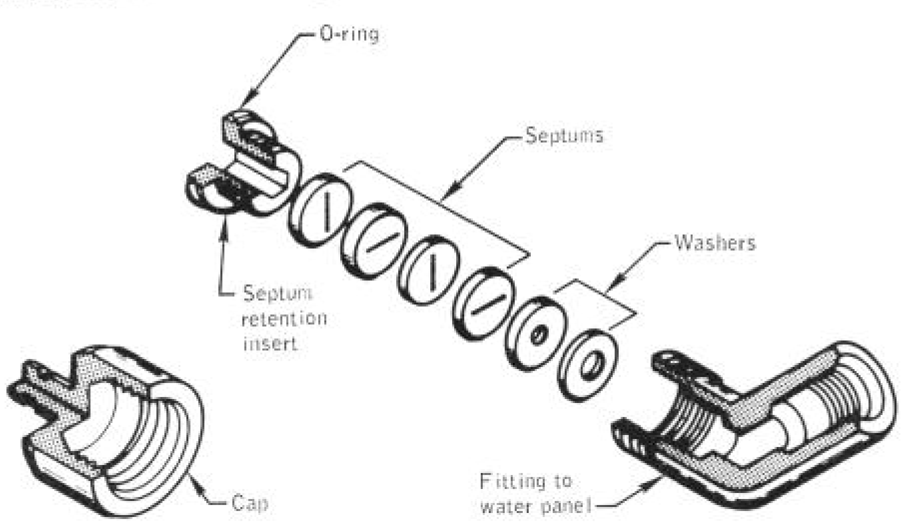

Figure 14-2.- Water panel chlorine injection port.

Postflight inspection and dimensional checks of the injection port assembly showed that all components were within established tolerances.

However, when the insert was tightened in accordance with the drawing requirements, the resulting septum compression was apparently insufficient to prevent the insert from loosening as a result of "O-ring drag" when the cap was removed. This allowed water leakage past the relaxed septums.

For future spacecraft, a shim will be installed under the insert shoulder to control the septum compression while allowing the installation torque to be increased to a range of 48 to 50 in-lb and, thus, preclude insert backout.

This anomaly is closed.

14.1.3 Service Propulsion System Thrust Light On Entry Monitor System

The service propulsion system thrust light located on the entry monitor system panel was illuminated shortly after transposition and docking with no engine firing command present. This light indicated the presence of a short to ground in the service propulsion system ignition circuitry. Ignition would have occurred if the engine had been armed.

The short was isolated to the system A delta-V thrust switch which was found to be intermittently shorted to ground (**Fig. 14-3**).

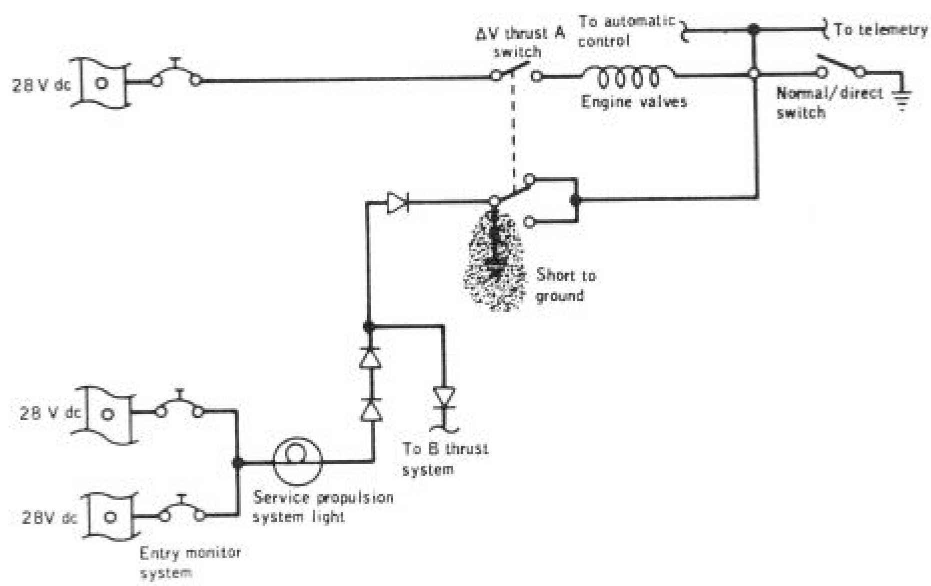

Figure 14-3.- Service propulsion system thrust light circuitry.

A test firing performed at 28:40:22 verified that the short existed on the ground side of the service propulsion system pilot valve solenoids.

The delta-V thrust switch (**Fig. 14-4**) was shorted to ground both before and after removal of panel 1 from the command module during postflight testing. After a change in panel position, the short-to-ground disappeared. The switch was then removed from the panel and X-rayed. The X- rays showed a wire strand extending from the braid strap which was thought to have caused the grounding problem. After switch dissection, an internal inspection verified that a strand extended from the braid strap; however, it did not appear to be long enough to cause a ground at any point within the switch (**Fig. 14-4**). The bracket assemblies (pivot brackets, pigtail braids, and movable contacts) and the plastic liner were removed from the switch. Microscopic examination revealed that a wire strand (approximately 0.055 inch long) was present on the flange on terminal 2 (**Fig. 14-5**). The strand appeared to be attached, but was later moved quite easily.

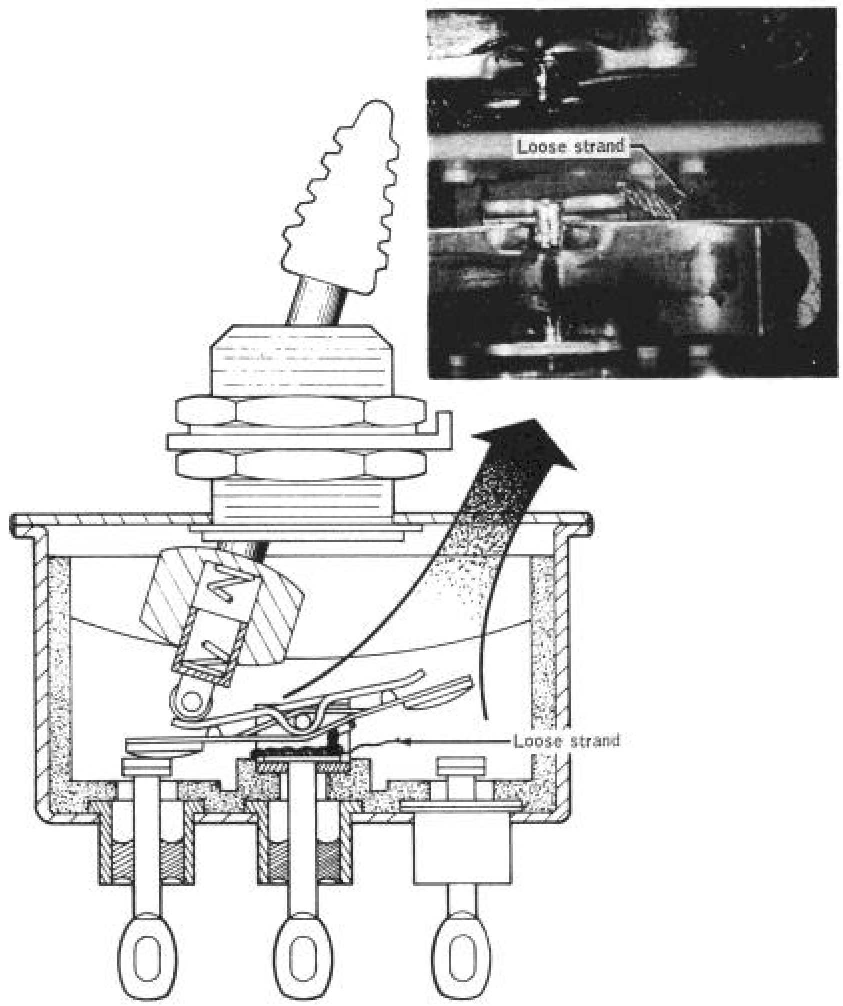

Figure 14-4.- Service propulsion system delta-V thrust switch.

The bottom of the plastic case liner was examined, and showed no evidence of a scratch or deformation conforming to the shape of the wire strand. A sample wire strand was placed on a feed-through flange of a scrap switch unit, and the plastic case liner was pressed on as would occur during normal switch assembly. When the scrap switch was disassembled an indentation in the plastic case liner was readily apparent. This test indicated that the strand could not have been trapped between the case liner and the flange surface; therefore, it is postulated that it might have been enclosed in the cavity of feed-through terminal 2 (**Fig. 14-5**). The maximum clearance between the interior of the feed-through terminal wall and the terminal itself is 0.040 inch. A 0.055-inch-long wire strand could easily have bridged this distance, and yet is short enough to move quite freely within the feed-through terminal cavity. In fact, the strand subsequently fell into the cavity. Examination of the strand and cavity wall showed evidence of arcing. The strand could not be detected on the X-rays because that area was obscured by other poles in the switch.

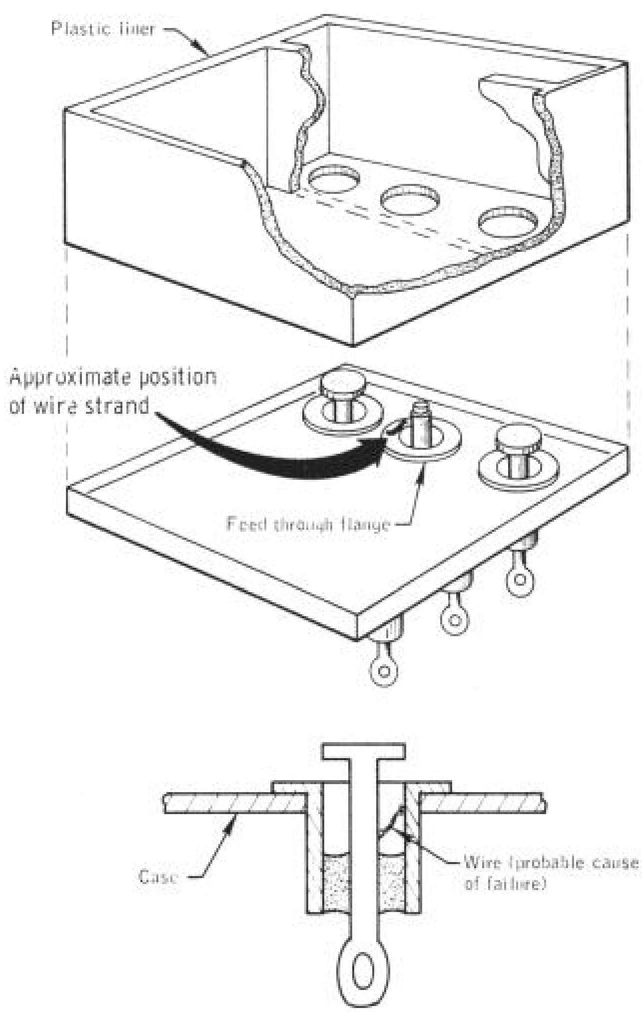

Figure 14-5.- Disassembled service propulsion system thrust switch.

Most of the switches on Apollo 16 and 17 spacecraft (3000 or 4000 series) required for crew safety or mission success were screened according to the following procedures.

Additional inspection points were employed during manufacturing.

The switches were X-rayed prior to acceptance testing.

The acceptance vibration test was 3-axis random (4000 series) or single-axis sinusoidal (3000 series) test.

The switches were X-rayed after acceptance testing.

The following switches for Apollo 16 were of an earlier series and have been replaced with 4000 series switches:

a. Up telemetry data/back-up voice
b. VHF ranging
c. Battery charger
d. Thrust vector control servo power
e. Postlanding ventilation
f .Crew optical alignment sight power
g. Optics power
h. Inertial measurement unit power
i. Rendezvous radar transponder power

Switches required for crew safety and mission success for Apollo 17 which had not been screened according to the aforementioned procedures will also be replaced. In addition, two science utility power switches are to be disabled and stowed, and two circuit breakers are to be added to provide series protection for the command and service module/lunar module final separation function.

This anomaly is closed.

14.1.4 Integral Lighting Circuit Breaker Opened

The a-c bus 2 and the d-c bus B under-voltage alarms occurred and, subsequently, the integral lighting circuit breaker opened.

A short circuit sufficient to cause the circuit breaker to open would also cause the alarms. As a result of the problem, some display keyboard lights, the entry monitor system scroll lighting, and various other backlighting were not used for the remainder of the mission.

Postflight testing of the vehicle disclosed that the short circuit was in the mission timer. The timer was removed from the vehicle and returned to the vendor for further analysis. Teardown analysis revealed a shorted input filter capacitor.

The capacitor is rated for 200-volt d-c applications and is being used in an a-c application at voltages up to 115 volts. Since the dielectric in the ceramic capacitor is a piezolectric material (barium titanate), the 400-cycle a-c voltage actually causes the materials in the capacitor to mechanically vibrate at that frequency. Over a period of time, the unit could break down because of mechanical fatigue. This may have been the cause of failure of this capacitor.

There are two mission timers on the command module and one on the lunar module. The unit on the lunar module is separately fused. Fuses will be added to the units in the Apollo 16 and 17 command modules. Appropriate action will be taken to correct the timer design and an inline change will be made on both the command module and lunar module.

This anomaly is closed.

14.1.5 Battery Relay Bus Measurement Anomaly

At approximately 81-1/2 hours, the battery relay bus voltage telemetry measurement read 13.66 volts instead of the nominal 32 volts, as evidenced by battery bus voltage measurements. The crew verified that the same low voltage reading was present on the panel 101 systems test meter. When the crew moved the systems test meter switch, the reading returned to normal.

Postflight testing of the vehicle and all of the involved components revealed no anomalous condition which could have caused the problem but did isolate the problem to the instrumentation circuitry and verify that the functional operation of the bus was not impaired. Analysis indicates that the only way to duplicate the flight problem would be to connect a resistance of 2800 ohms from ground to the battery relay bus measurement circuit (**Fig. 14-6**). No resistance near this magnitude was measured during postflight testing. The most probable cause of the anomaly was that insulation resistance at the output terminal of the switch was lowered because of humidity.

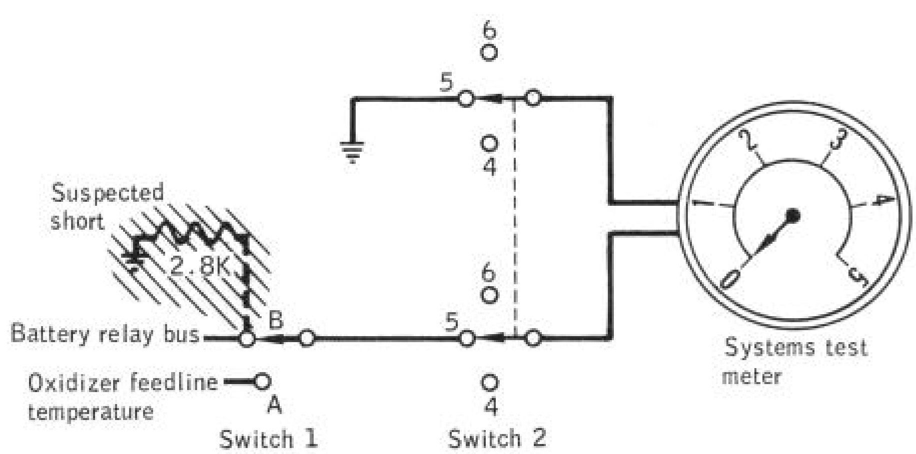

Figure 14-6.- Systems test meter switch.

This is the only time that a problem of this type has occurred during the Apollo Program and the probability of recurrence is considered to be very low. If the problem does occur again, other measurements are available for the determination of the battery relay bus voltage.

This anomaly is closed.

14.1.6 Mass Spectrometer Boom Talkback Indicated Half-Barberpole On Retract

The mass spectrometer boom did not fully retract on five of twelve occasions. Data analysis, supported by the crew debriefing, indicates that the boom probably retracted to within about 1 inch of full retraction. Cold soaking of the deployed boom and/or cable harness preceded each anomalous retraction. In each case, the boom retracted fully after warmup.

The deploy/retract talkback indicator is normally gray when off, when the boom is fully retracted, or when it is fully extended. The indicator is barberpole when the boom is extending or retracting, and will show half barberpole if the drive motor stalls. The crew noted this last condition on the incomplete retractions.

An inflight test of the Apollo 15 boom indicated that the problem was a function of temperature. Testing and examination of the Apollo 16 spacecraft showed that the failure was possibly caused by pinching of the cable harness during the last several inches of boom retraction. The cable could have been pinched between the bell housing and rear H-frame bearing (**Fig. 14-7**), or a cable harness loop was jammed by a boom alignment finger against the bell housing (**Fig. 14-8**).

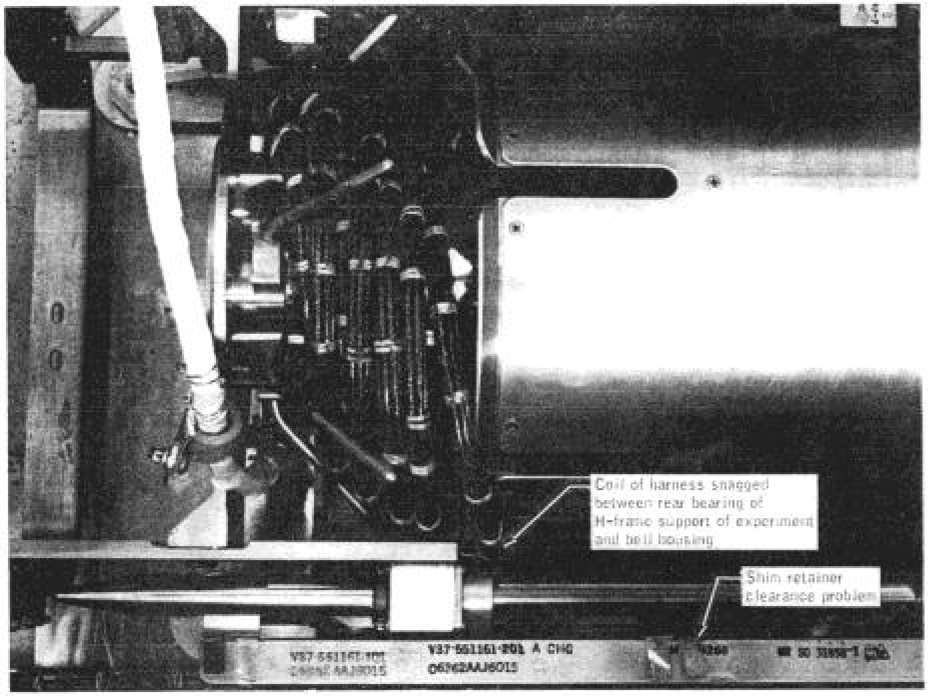

Figure 14-7.- Bottom view of mass spectrometer
deployment mechanism (experiment removed).

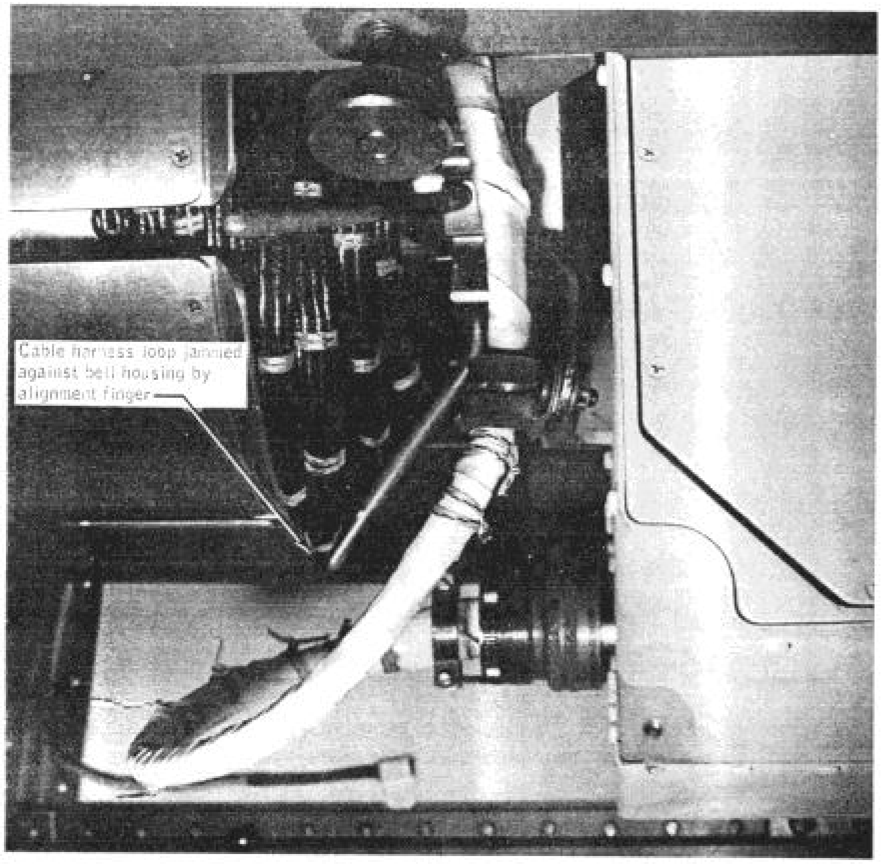

Figure 14-8.- Side view of mass spectrometer deployment mechanism (experiment attached).

The mass spectrometer boom mechanism was qualified by similarity to the gamma ray boom mechanism. There are significant differences between the two designs and they are:

a. When extended, the mass spectrometer boom is 1 foot 10 inches shorter than the gamma ray spectrometer boom.

b. The mass spectrometer cable harness contains 6 more wires and, therefore, is larger in cross section than the gamma ray spectrometer cable. In addition, the harness coil diameter on the mass spectrometer is 1/2 inch larger (6.7 inches compared to 6.2 inches).

c. The mass spectrometer cable harness terminates with an in-line connector; whereas, the gamma ray spectrometer harness terminates with a 90-degree connector.

d. The mass spectrometer rear H-frame bearings retract past the lip of the bell housing; whereas, the retracted bearing position for the gamma ray experiment boom is even with the bell housing lip. Therefore, the lip on the sides of the mass spectrometer bell housing is relieved about 1/2 inch for bearing clearance.

The differences between the two configurations are now considered to be significant enough to have required separate testing for the mass spectrometer boom assembly. Accordingly, a delta qualification test will be instituted and a thermal vacuum environmental acceptance test will be performed on each flight unit.

Additional failure modes revealed during the testing of the Apollo 16 unit are:

a. Insufficient clearance between the spectrometer rear H-frame bearings and the boom housing bearings in relation to the rail support bean shim retainers. This could have been significant on Apollo 15, had a jettison been attempted.

b. Misalignment between the right-hand guide rail forward floating section and the rigid rear section.

If the boom does not retract to within approximately 12 inches of full retraction, it will be jettisoned prior to the next service propulsion system firing. Tests have shown that the boom will not buckle during a service propulsion system firing when retracted to within 14.5 inches of full retraction.

Corrective actions for Apollo 16 are as follows:

a. A thermal vacuum test will be added to the acceptance test requirements.

b. The brackets supporting the service loop at the experiment end of the cable harness will be redesigned.

c. The existing finger guides will be extended.

d. The bell mouth housing will be extended.

e. Lead-in ramps will be added to the inboard bearing housings.

f. Rail support beam shim retainer movement will be corrected by using anti-roll pins in place of shim retainers.

g. A proximity switch modification kit will be installed to show when the boom has reached to within about 1 foot of full retraction.

This anomaly is closed.

14.1.7 Potable Water Tank Failure To Refill

The potable water tank quantity began to decrease during meal preparation at approximately 277 hours and failed to refill for the remainder of the flight. The waste water tank continued to fill normally and, apparently, accepted fuel cell water for this period. A similar occurrence had been noted earlier, at 13 1/2 hours, when the potable tank quantity decreased as the crew used the water, and remained constant until a waste water dump was performed at 28 1/2 hours. This decrease had been attributed to a closed potable tank inlet valve until the crew verified in their debriefing that the valve had been open during this time. The amount of water drained from the tank verified that the tank instrumentation was reading correctly.

During a postflight fill operation, with the waste tank inlet valve closed, and water introduced at the hydrogen separator, both the potable and waste water tanks filled.

The check valve between the fuel cell and waste tank dump leg (**Fig. 14-9**) was tested and found to leak excessively. A tear-down analysis of the check valve was performed and a piece of 300-series stainless steel wire (approximately 0.0085 by 0.14 inch) was found between the umbrella and the seating surface (**Fig. 14-9**). This contaminant could cause the umbrella to leak and yet move around sufficiently to allow adequate seating at other times. The wire most probably came from a welder's cleaning brush and was introduced into the system during buildup. Safety wires and tag wires are of a larger diameter than the one found. The check valve at the potable water tank inlet is of a different configuration and is spring loaded closed. The 1-psi pressure required to open this valve is a large pressure drop compared to the other components at the low flow of 1-1/2 lb/hour, and would, therefore, cause the water to flow to the waste tank.

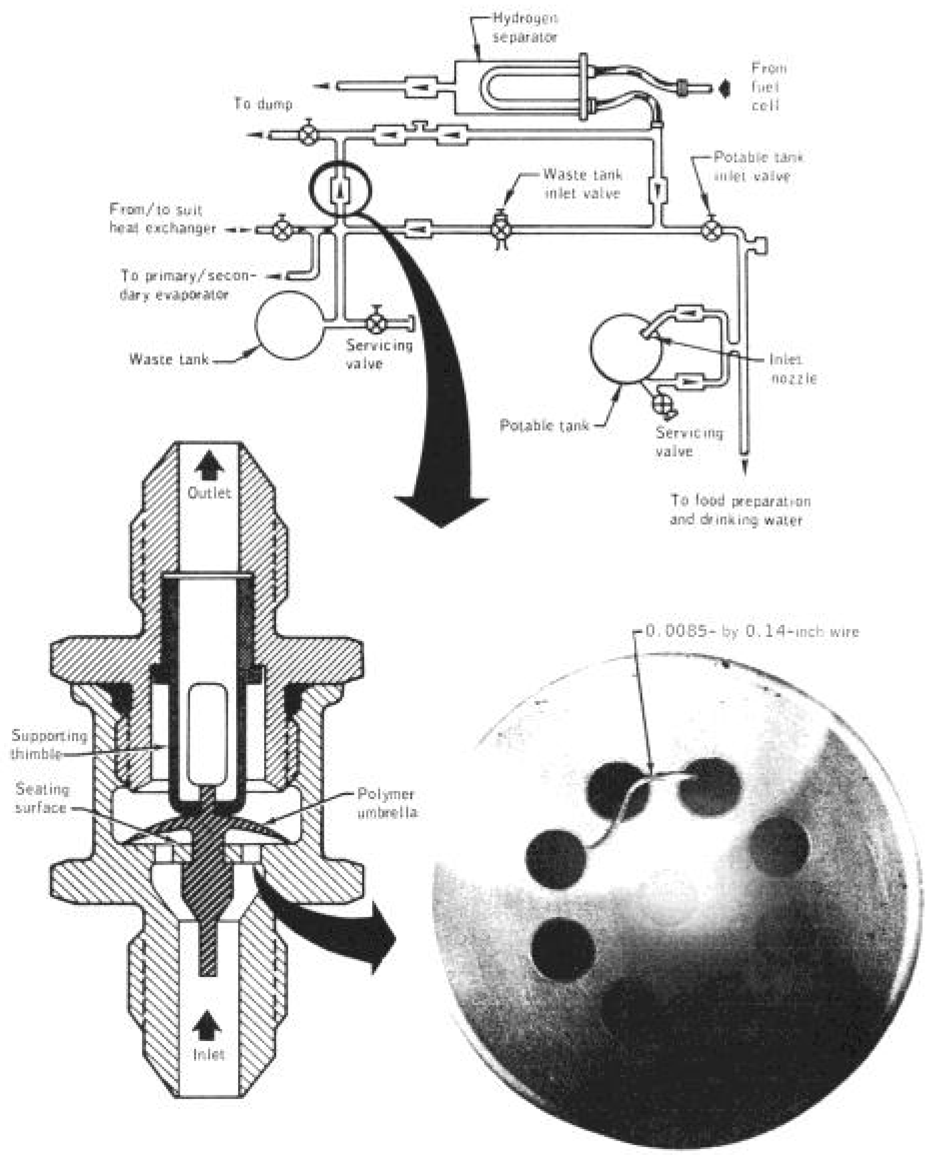

Figure 14-9.- Water management system and failed check valve details.

The potable water tank inlet check valve was found to be contaminated with aluminum hydroxide, a corrosion product, of aluminum and the buffer. The potable water tank inlet nozzle was clean and free of corrosion. The check valve corrosion is not believed to have caused the problem, but could have contributed by increasing the crack pressure of the valve.

No corrective action is considered necessary since the contamination is considered to be an isolated case. If the problem should recur, the potable tank will start to fill when the waste tank is full.

This anomaly is closed.

14.1.8 Mission Timer Stopped

The panel 2 mission timer stopped at 124:47:37. Several attempts to start the clock by cycling the start /stop/reset switch from the stop to the start position failed (**Fig. 14-10**). The timer was reset to 124:59:00 using the hours, minutes, and seconds switches, and the timer again failed to start when the switch was cycled. The switch was then placed in the reset position. The timer reset to all zeros and started to count when the switch was placed in the start position. The timer was then set to the proper mission time using the hours, minutes, and seconds switches and operated properly for the remainder of the mission.

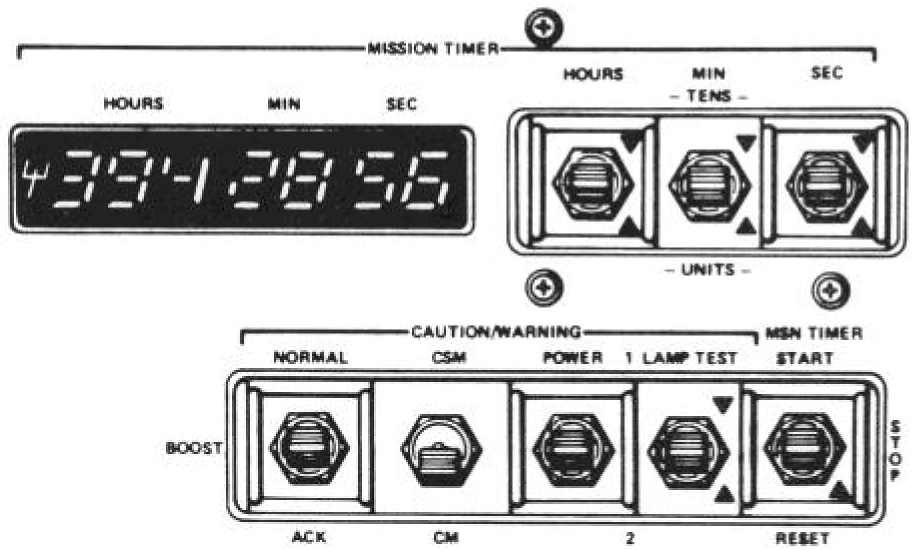

Figure 14-10.- Mission timer.

The timer and all associated equipment were still operating properly after the flight. Thermal , vacuum, and acceptance tests were performed and the cause of the failure could not be determined. Circuit analysis showed that the problem could be caused by one of five integrated circuits on the mounting board circuitry. These suspect components were removed and tested with negative results.

The failure was most probably caused by an intermittent problem within a component which later cured itself. If the problem occurs on a future mission and the timer will not restart, mission time can be obtained from the other timer in the command module, or from mission control. The failure would be a nuisance to the crew.

This anomaly is closed.

14.1.9 Main Parachute Collapse

One of the three main parachutes was deflated to approximately one fifth of its full diameter at about 6000 feet altitude. The command module descended in this configuration to landing. All three parachutes were disconnected and one good main parachute was recovered. Photographs of the descending spacecraft indicate that two or three of the six riser legs on the failed parachute were missing (**Fig. 14-11**).

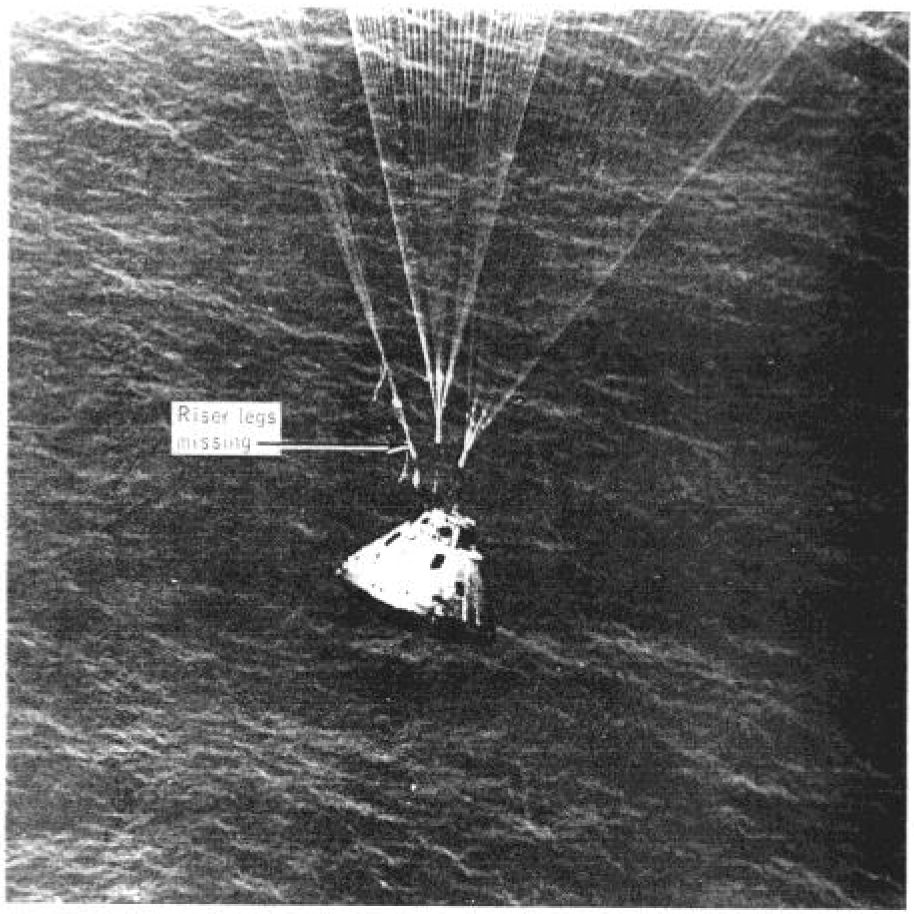

Figure 14-11.- Parachute riser damage noted during final descent.

Three areas that were considered as possible causes are:

The forward heat shield, which was in close proximity to the spacecraft flight path.

A broken riser/suspension line connector link which was found on the recovered parachute (**Fig. 14-12**).

The command module reaction control system propellant firing and fuel dump.

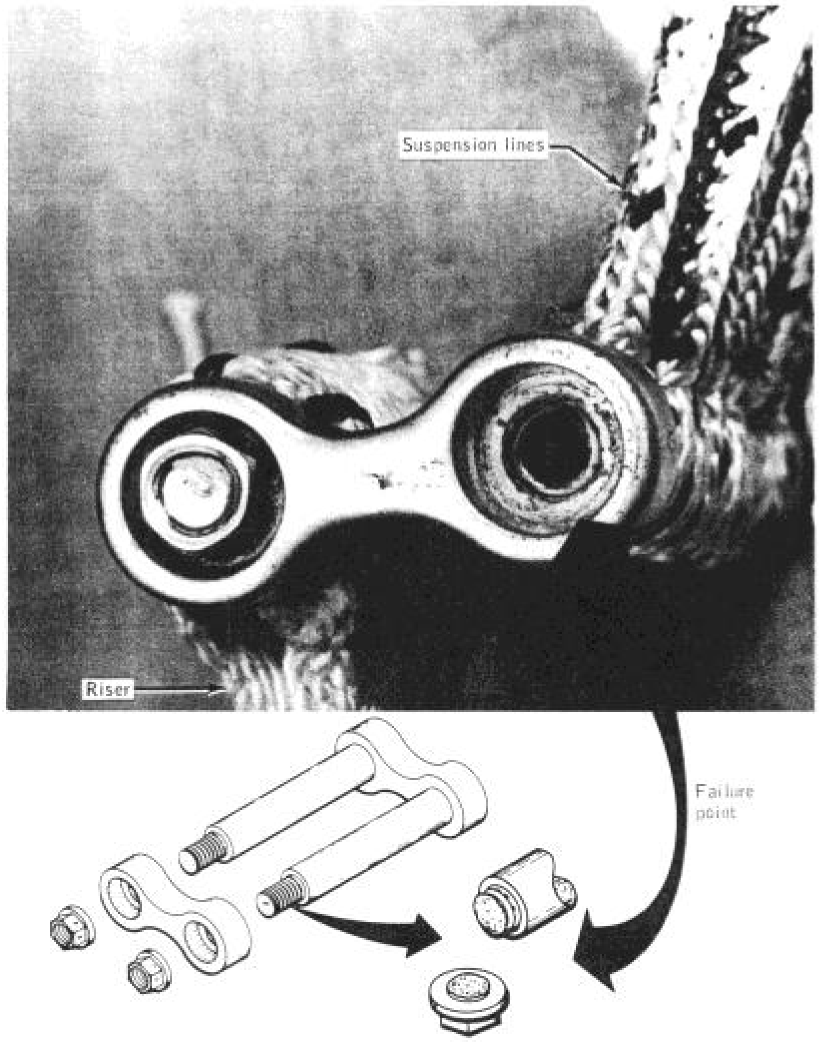

Figure 14-12.- Main parachute connector link failure.

Onboard and photographic data indicate that the forward heat shield - was about 720 feet below the spacecraft at the time of the failure. The failed link on the recovered parachute implies the possibility of a similar occurrence on the failed parachute. Based on parachute tow tests, however, more than one link would have had to fail to duplicate the flight problem. The two possible causes have been identified as hydrogen embrittlement or stress corrosion.

The command module reaction control system depletion firing was considered as a possible candidate because of the known susceptibility of the parachute material (nylon) to damage from the oxidizer. Also because the oxidizer depletion occurred about 3 seconds prior to the anomaly, and fuel was being expelled at the time the anomaly occurred (**Fig. 14-13**). Further, the orientation of the main parachutes over the command module placed the failed parachute in close proximity to the reaction control system roll engines. Testing of a command module reaction control system engine simulating the fuel (monomethyl hydrazine) dump mode through a hot engine resulted in the fuel burning profusely; therefore, the fuel dump is considered to be the most likely cause of the anomaly.

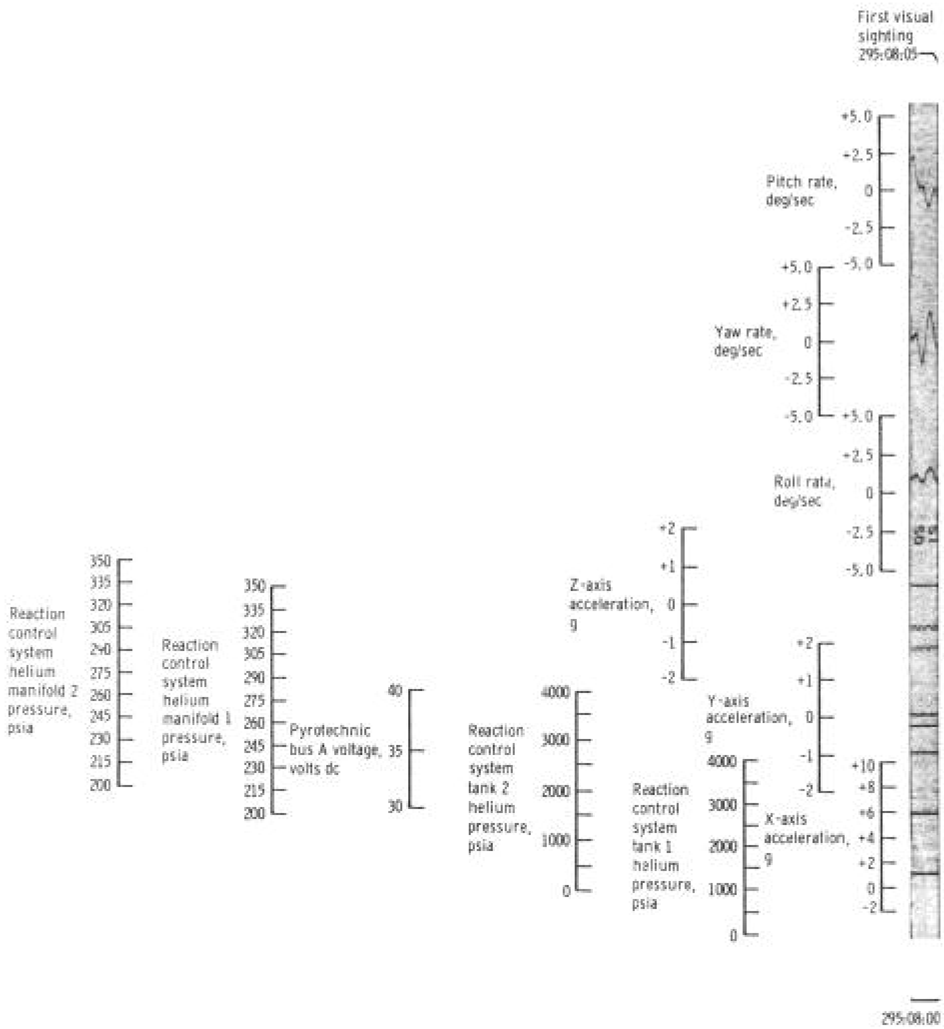

Figure 14-13.- Sequence of events during descent on the main parachutes.