In order to eliminate critical processing operations from manufacture of the connector links, the material was changed from 4130 to Inconel 718.

Based on the low probability of contact and the minimum damage anticipated should contact occur, no corrective action will be implemented for the forward heat shield. Corrective actions for the reaction control system include landing with the propellants onboard for a normal landing, and biasing the propellant load to provide a slight excess of oxidizer. Thus, for low altitude abort land landing case, burning the propellants while on the parachutes will subject the parachutes to some acceptable oxidizer damage but, will eliminate the dangerous fuel burning condition. In addition, the time delay which inhibits the rapid propellant dump may be changed from 42 to 61 seconds. This could provide more assurance that the propellant will not have to be burned through the reaction control system engines in the event of a land landing. A detailed discussion of all analyses and tests is contained in a separate anomaly report (reference 7).

This anomaly is open.

14.1.10 Data Recorder Tape Deterioration

At about 240 hours, after over 100 tape dumps had been completed, the ground was unable to recover the data contained on about the first 20 feet of tape. To alleviate the problem, that portion of the tape was not used again.

An electrical and physical examination of the flight tape was conducted. Observation of the bi-phase output of the 51.2 kilobit pulse code modulated output from the playback showed a complete deterioration of the waveform for the first 20 seconds (12-1/2 feet), together with numerous complete dropouts. After 20 seconds, the bi-phase signal gradually improved to the point where, at 30 seconds, the signal appeared normal. The 64 kilobit pulse code modulated output was similarly affected to a lesser degree.

The first 30 feet of tape was scanned under magnifications ranging from 50X to 400X. Under 50X magnification, a distinct pattern of embedded particles could be observed (**Fig. 14-14**). The deposits were quite heavy over the first 12 feet of tape, and gradually tapered out so that, at 20 feet, very few particles could be observed. Under 400X magnification, individual flakes of deposited material were observed. The portion of **Fig. 14-14** at 400X magnification shows a typical cluster of particles on the beginning portions of the tape.

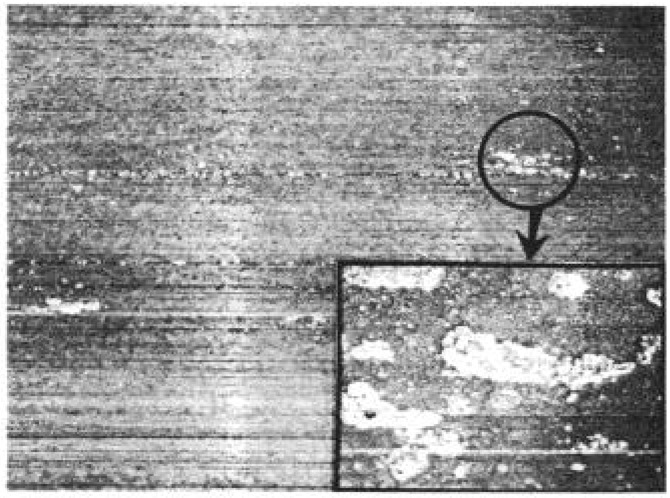


Figure 14-14.- Recording tape photographed under 50 power and 400 power magnification showing silver oxide deposits.

A 10-foot leader coated with a silver oxide compound is spliced to the beginning and end of the magnetic tape roll to activate the end-of-reel sensors on the tape transport. There has been a history of this material flaking off and affecting tape performance. Tape screening procedures were implemented by the manufacturer in 1968 to eliminate this problem. No further problems were encountered until Apollo 15. The recording method for Apollo 14 and previous missions was considerably different than that for the Apollo 15 mission. Bit packing densities for the Apollo 15 mission tape approach 9000 bits per inch while those for the previous missions were only 800 bits per inch. Abnormalities in the tape would have considerably more effect with the higher packing density. The utilization of the Apollo 15 mission recorder is also considerably higher, allowing more time for deposits to build up.

An acceptance test (except for environmental verification) with a new tape was conducted on the flight recorder and all parameters were within specification with little change in absolute values from the pre-delivery test.

Inspection on the magnetic heads under 20X magnification disclosed four scratches, one of which is shown in **Fig. 14-15**. An overlay was made of the scratches with respect to the accumulation of silver oxide on the tape; two of the four scratches aligned perfectly with the silver oxide accumulation. The scratches must have scraped loose the silver oxide on the leader. Operation of the recorder would then distribute the silver oxide particles along the tape. During the manufacture of the Apollo 16 recorder, it was discovered that the heads were being scratched by handling. The Apollo 15 recorder heads were probably also scratched during manufacture. The scratches would not have been detected during acceptance inspection since they are not visible at the 7X magnification used during that inspection.

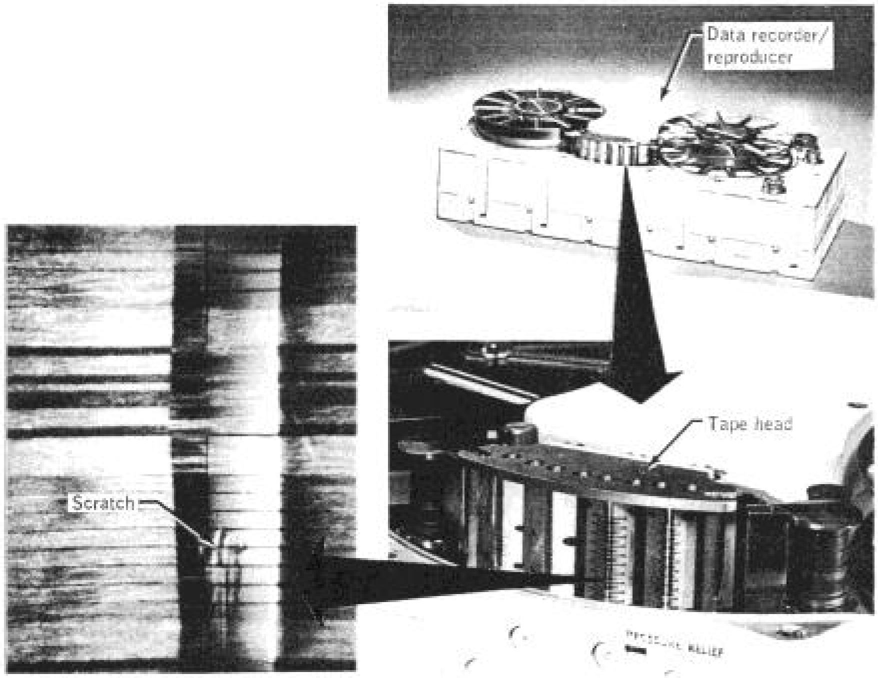


Figure 14-15.- Data recorder/reproducer and scratches on head.

Removable head covers have been provided to protect the heads from handling damage when the recorder covers are not installed. These covers have been used since early in the buildup cycle of the Apollo 16 and 17 data recorders. The recorder heads have been examined under 20X magnification and no scratches were found.

This anomaly is closed.

14.1.11 Digital Event Timer Obscured

The seconds digit of the digital event timer, located on panel 1, became obscured by a powder-like substance that formed on the inside of the glass. Postflight analysis of the unit disclosed that the substance on the window was paint which had been scraped from the number wheel by the idler gear. The idler gear is free to rotate on the shaft (**Fig. 14-16**); however, the design allows the stainless-steel shaft to also rotate. The stainless steel shaft bearing points are in the magnesium motor plate and the shaft rotation wears away the softer magnesium material.

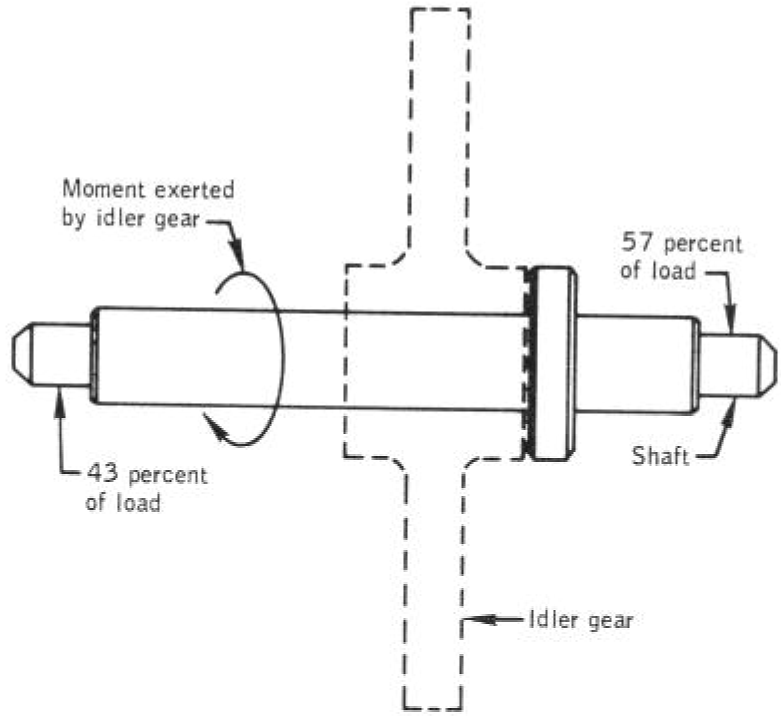


Figure 14-16.- Forces on idler gear shaft in countdown mode.

Inspection of the unit showed that the magnesium bearing points had been elongated as shown in **Fig. 14-17**. Torque from the stepping motor applied to the idler gear not only resulted in rotation of the shaft but also caused the shaft to tilt (**Fig. 14-17**). The wearing eventually allowed the shaft to tilt sufficiently to cause the gear to rub against the number wheel. 'When the timer counted down, the motor torque threw the gear teeth into the front edge of the counter wheel. Testing indicates that this bearing hole elongation occurs after approximately 500 hours running time (specification life is 1400 hours).

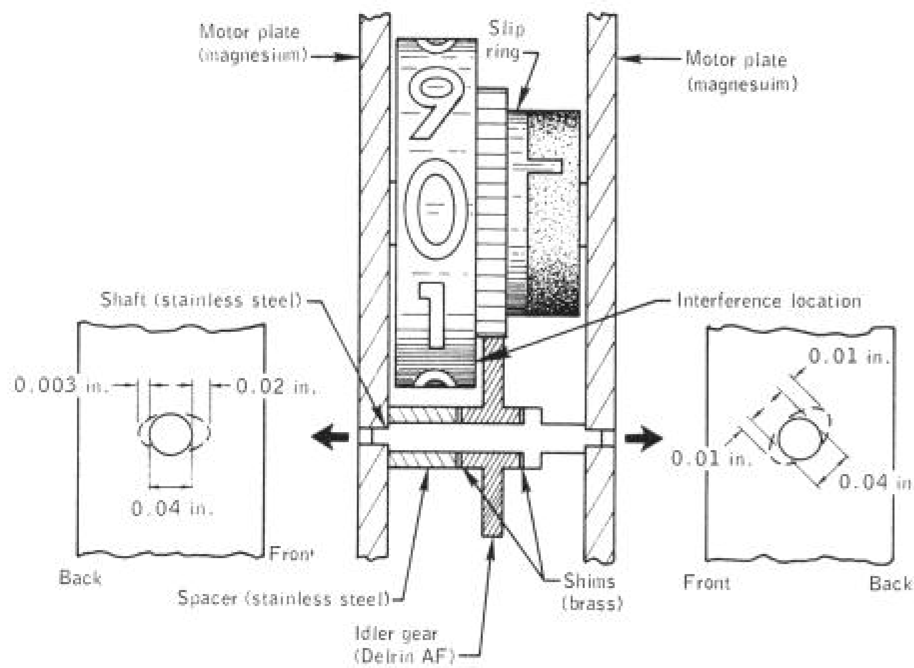


Figure 14-17.- Digital event timer mechanism.

A review of the history of the unit shows that it was built in 1966. Prior to installation in Apollo 15, the unit was modified because of failures on other timers. Brass shims (fig. 14-17) were installed to prevent the idler gear from rubbing on the number wheel.

The analysis of those failures revealed that the idler gear was rubbing paint off the number wheel and paint particles prevented the slip rings and brushes from making good contact. A review of drawing tolerances showed that an interference could occur and the addition of the shims appeared to be adequate corrective action. These failure analyses did not reveal the problem of the elongation of the bearing points since it is not obvious until the timers are disassembled.

Units for future flights will be visually inspected by looking through the window for paint flakes and signs of wear.

This anomaly is closed.

14.1.12 Crew Restraint Harness Came Apart

The restraint harness on the right side of both the center and right crew couches came apart during lunar orbit. The assemblies had become unscrewed, but the crew was able to retrieve all the parts except one cap and reassemble the harnesses satisfactorily for landing. The mating plug for the missing cap was held in place with tape.

The plug-and-cap assembly (**Fig. 14-18**), which is part of the universal assembly that attaches the restraint harness to the couch seatpan, separated. (There are a total of six plug-and-cap assemblies on the crew couch, two per man.) The plug component (bolt) has a nylon insert in the threaded portion that acts as a locking device. Back-and-forth rotation of the adjuster link can cause the plug-and-cap assembly to unscrew from each other. Checks on the four other Apollo 15 assemblies showed zero torque on two of the units and minimum specification value (2.0 in-lb) on the others. The loss of torque is apparently due to cold flow of the plastic self-locking pellet, causing a loss of friction against the mating threads.

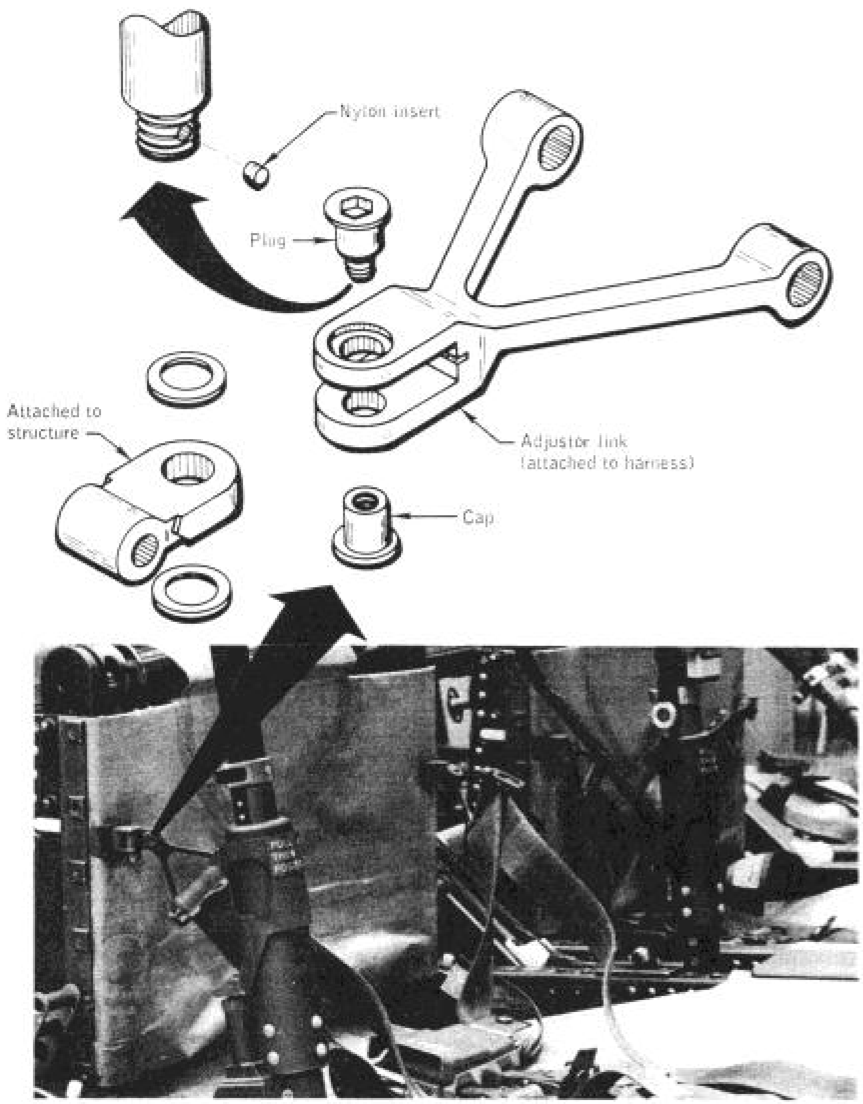


Figure 14-18.- Crew restraint harness connector.

A thread locking sealant will be used to prevent the problem on future missions.

This anomaly is closed.

14.1.13 Loose Object In Cabin Fans

During portions of the flight when the cabin fans (**Fig. 14-19**) were activated, the crew heard sounds like an object striking the blades. Cycling the fans several times allowed the object to be retained in a position that precluded it from interfering with fan operation.

Inspection of the fans revealed considerable gouging on the leading edges of the blades of both fans (**Fig. 14-19**). No marks were found on the outlet de-swirl vanes of either fan. After extensive examination, including the lunar dust filter, a 1/4-inch washer was found in the ducting.

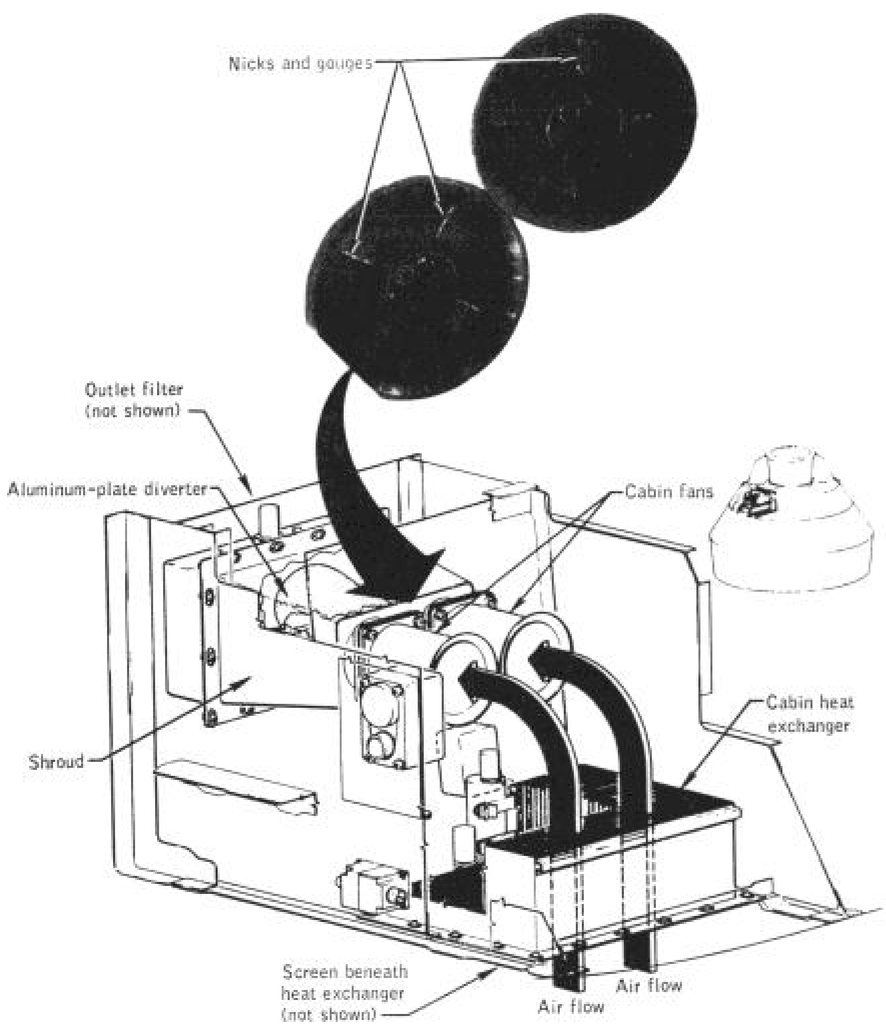


Figure 14-19.- Cabin fan installation.

Although the fan inlet is protected by screens in addition to the heat exchanger core, the outlet is relatively open and the washer could have drifted in and out when the fans were not operating. The outlet was protected only during the limited time when the lunar dust filter was installed. The washer could also have been left in the duct during assembly.

No hardware changes are contemplated. Should the anomaly occur on a subsequent flight, no detrimental effects would result.

This anomaly is closed.

14.1.14 Scanning Telescope Visibility

The crew reported that excessive attenuation of light through the scanning telescope existed throughout the flight. The telescope was adequate to perform landmark tracking while in lunar orbit, but the crew was unable to identify constellations, even though large numbers of stars could be seen by looking out the spacecraft window.

Visual observations through the telescope (**Fig. 14-20**) were made at the spacecraft manufacturer's facility, and no degradation could be observed. A luminescent transmittance test was performed on the telescope before removal from the spacecraft and transmittance was calculated to be 25 percent. This compares with an acceptance test value of 55 percent. The decrease is due to the entry environment and sea water contamination. The 30-percent decrease agrees well with the expected results and is not significant as far as being able to see and recognize constellations is concerned. For comparison, the earth's atmosphere normally causes a 50 percent loss in star intensity; therefore, observing stars from earth with a telescope with a 50-percent transmittance would be equivalent to observing stars from a spacecraft in flight using a telescope with a 25- percent transmittance.

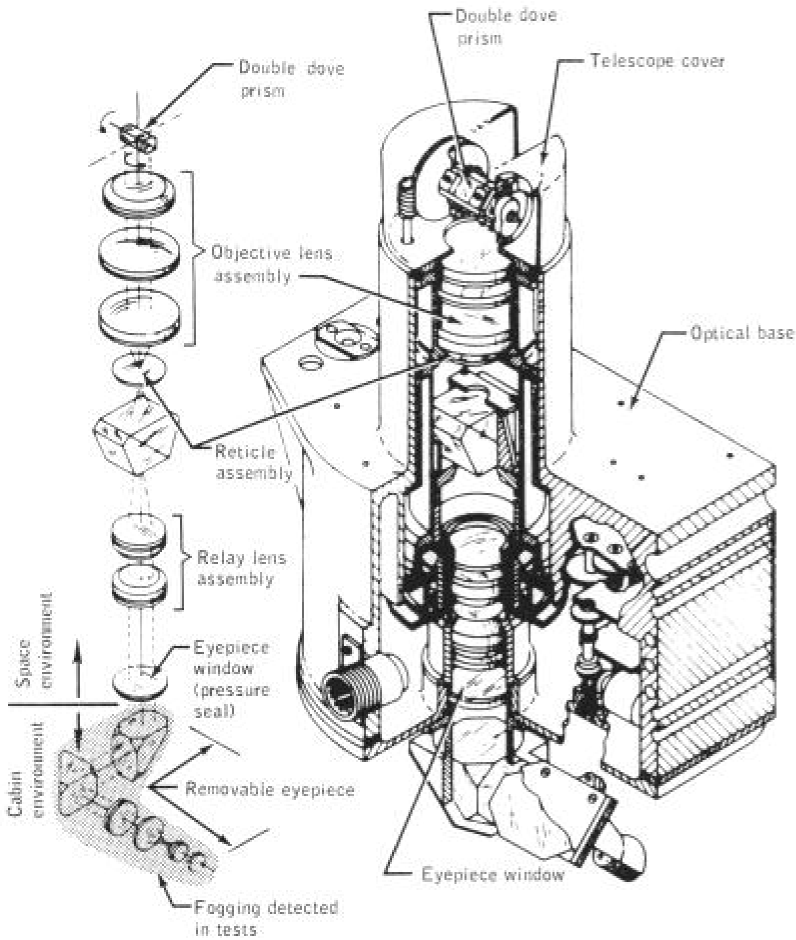


Figure 14-20.- Scanning telescope assembly.

The flight anomaly was reproduced in the laboratory by placing the optical unit assembly, the removable eye piece, and the optics panel in a chamber wherein the environmental conditions that existed in the cabin during flight were duplicated. Condensation on the eyepiece window and, to a lesser extent, on the prisms in the removable eyepiece caused the transmittance to decrease to about 4 percent.

A heater will be added to the removable eyepiece to prevent fogging in the eyepiece assembly and on the eyepiece window.

This anomaly is closed.

14.1.15 Gyro Display Coupler Roll Alignment

The crew reported that the roll axis did not align properly when the gyro display alignment pushbutton was pressed. The roll axis error was not nulled, whereas, the pitch and yaw axes were. Only by depressing the align pushbutton for progressively longer periods, and eventually, by moving the roll-axis thumbwheel, could the roll error be nulled.

For normal operation during alignment, resolvers in the gyro display coupler electronics are compared to resolvers, one for each axis, on the thumbwheels used to set desired attitude. The difference is an error signal. The error is displayed on the attitude error needles, and the signal is used to drive the resolvers to match the attitude set on the thumbwheels. The anomaly could have been caused by either of two failure modes. An intermittent open in the roll axis align loop or a low gain problem in the electronics (**Fig. 14-21**).

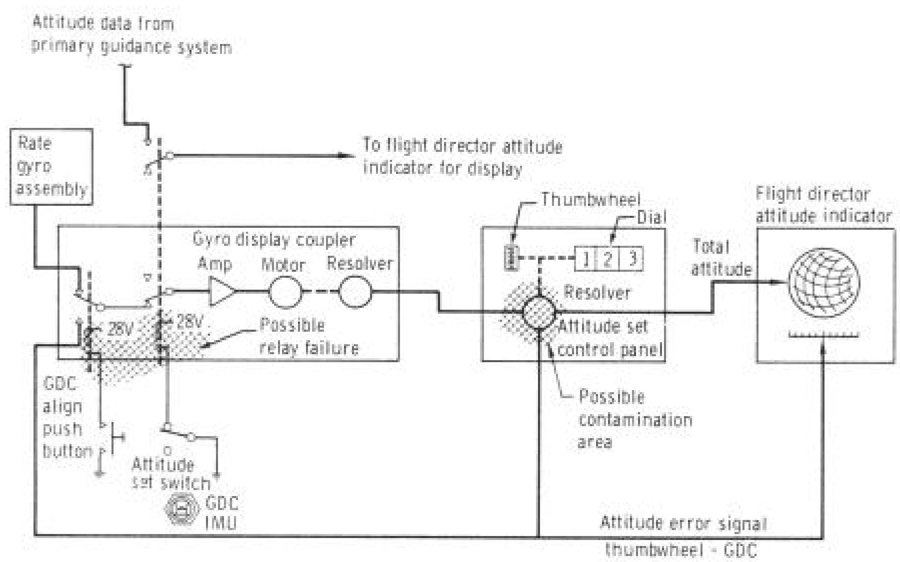


Figure 14-21.- Simplified align loop.

The gyro display coupler and attitude set control panel were put into the hardware evaluator simulator and functionally tested at the systems level in the actual spacecraft configuration in an attempt to repeat the flight problem. During this testing, an out-of-tolerance condition was observed on the attitude set control panel. This condition could have caused a gain type problem and been the cause of the flight anomaly. The measured resistance of the thumb wheel resolvers increased from the nominal in all three axes by as much as 1000 ohms. Normally, this value does not vary by more than 1 ohm. In order for this condition to have been the cause of the anomaly, a resistance change in the roll axis would have had to be an order of magnitude larger than that measured postflight. The resistance change is caused by contamination between the slip rings and the thumbwheel resolvers. As a result of the flight anomaly, several resolvers were examined and contamination was detected. The corrective action is to wipe the resolvers clean by rotating them several hundred revolutions. The attitude set control panels in Apollo 16 and 17 will be checked and the resolvers will be wiped clean, or will be replaced if necessary.

The flight condition could also have been caused by either of two golden-g" relays failing to close. Two failure modes have been determined. One failure mode is "normally open-failure to close", and other, "normally closed - failure to open", both caused by contamination. "Golden-g" relays were the subject of an extensive review in 1966 and 1967. Relays were classified as (1) critical, (2) of major importance, (3) of minor importance, and (4) having no effect. It was decided at that time to (1) make critical relays redundant, (2) improve screening tests, and (3) take no corrective action for non-critical relays. Both of the suspect relays are of major importance in that either one would cause loss of the normal alignment capability of the backup attitude reference system. The attitude reference system could be aligned, but extensive work-around procedures would have to be used.

Tests performed on the roll axis align enabling relay revealed contamination which could have caused the flight anomaly. The rationale developed during the "golden-g" relay review is applicable at this time.

This anomaly is closed.

14.1.16 Unable To Open Circuit Breaker Supplying Main A Power To Battery Charger

The circuit breaker tying the battery charger to main bus A could not be opened manually during postflight testing. This breaker was not required to be opened during the flight.

A green residue on the aluminum indicator stem at the copper mounting bushing jammed the stem and prevented operation. Some of the residue was removed for chemical analysis. The rest of the residue was dissolved by the application of distilled water, thereby freeing the breaker. The green residue was predominantly sodium-copper carbonate hydrate. Traces of sodium chloride, and other metals were also present. These products would result from salt water corrosion. Salt water could have been introduced by sea water splashing on the breaker after landing or by urine or perspiration released in the cabin during flight.

No corrective action is considered necessary.

This anomaly is closed.

14.1.17 Pivot Pin Failure On Main Oxygen Regulator Shutoff Valve

The toggle-arm pivot pin for the side-A shutoff valve of the main oxygen regulator was found sheared during postflight testing. With the pin failed, the shutoff valve is inoperative in the closed position, thus preventing oxygen flow to the regulator.

The pivot pin attaches the toggle arm to the cam holder and is retained in place by the valve housing when properly assembled (**Fig. 14-22**).

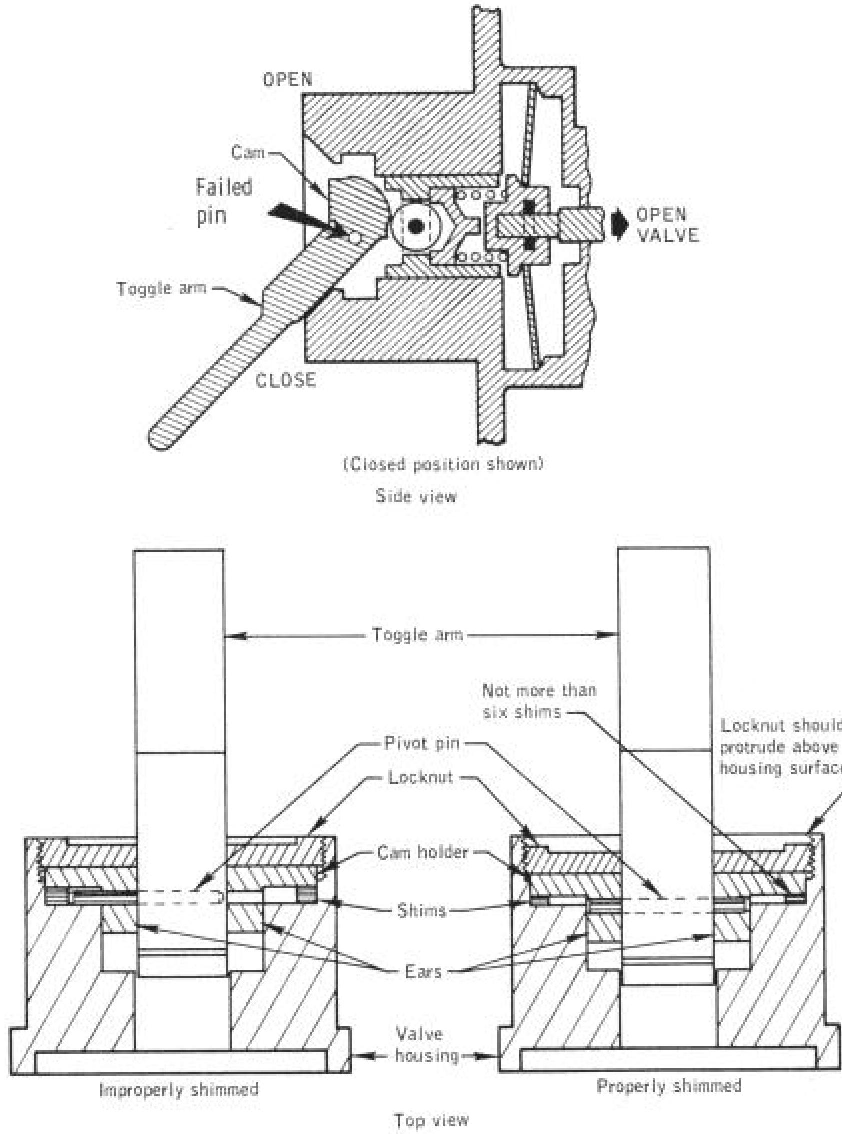


Figure 14-22.- Oxygen regulator shutoff valve.

Failure analysis showed that the pivot pin failed in single shear and bending. This failure resulted from improper shimming which allowed the pivot pin to come out of one side of the cam holder as shown in **Fig. 14-22**. Analysis and testing has shown that the pin strength is adequate in double shear, but will fail in single shear and bending with a force of about 70 pounds applied at the tip of the toggle arm when it is in the closed position. No marks were found on the toggle arm to indicate that it had been struck by some object.

Inspection criteria to assure that valves now installed in other spacecraft are properly assembled have been developed from a study of adverse tolerance buildups associated with the valve components. These criteria are that the lock nut does not protrude and the number of shims does not exceed six (fig. 14-22).

This anomaly is closed.

14.1.18 Crew Optical Alignment Sight Fell Off Stowage Mount

The crew optical alignment sight fell from its stowage mount during landing because the locking pin which secures it was not engaged. Normally, when the sight is placed into the mount, the locking pin is raised automatically by a ramp and the pin is moved into the locking pin hole by spring action (**Fig. 14-23**). Postflight examination showed that the ramp had been gouged preventing raising of the pin by the ramp. The cause of the gouge is not known.

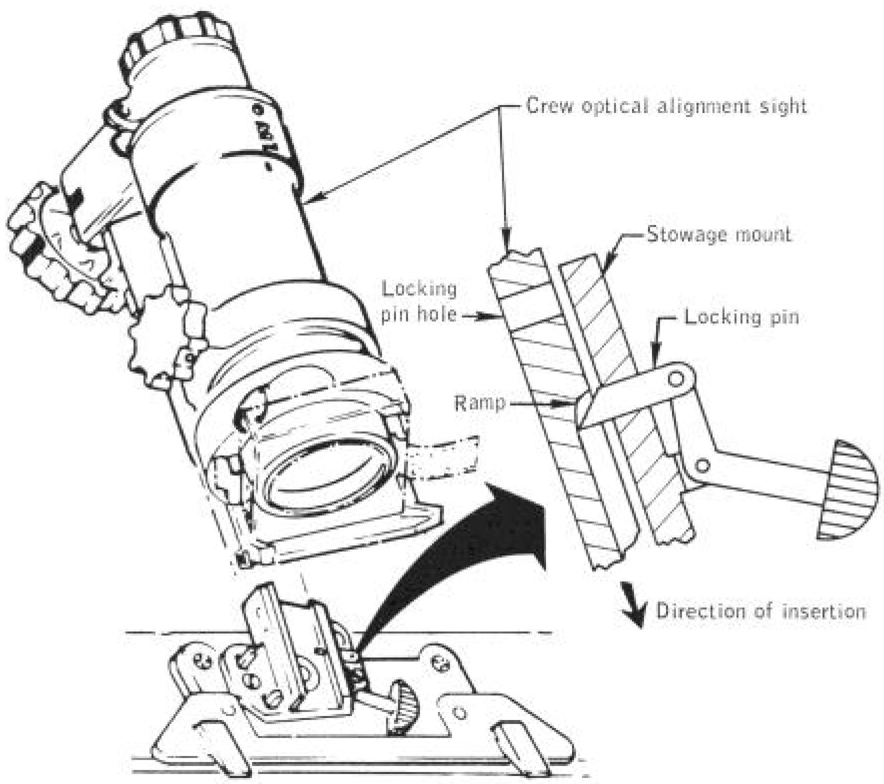


Figure 14-23.- Crew optical alignment sight stowage.

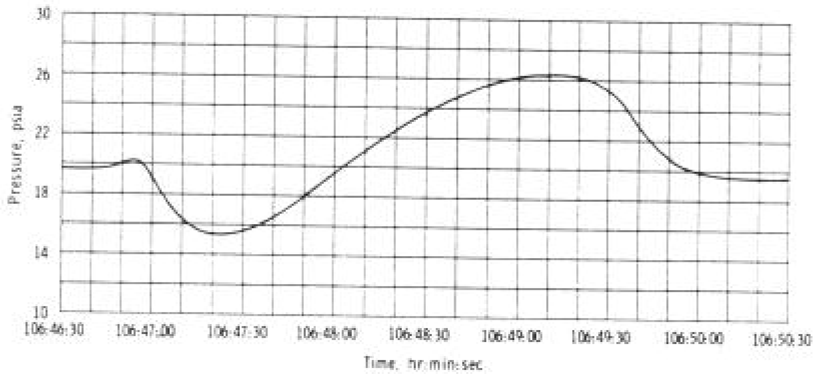
The crew optical alignment sights for Apollo 16 and future spacecraft will be fit-checked to insure proper operation of the latching mechanism. Also, the Apollo Operations Handbook and crew checklist are being revised to include verification of the latching pin engagement prior to entry.

This anomaly is closed.

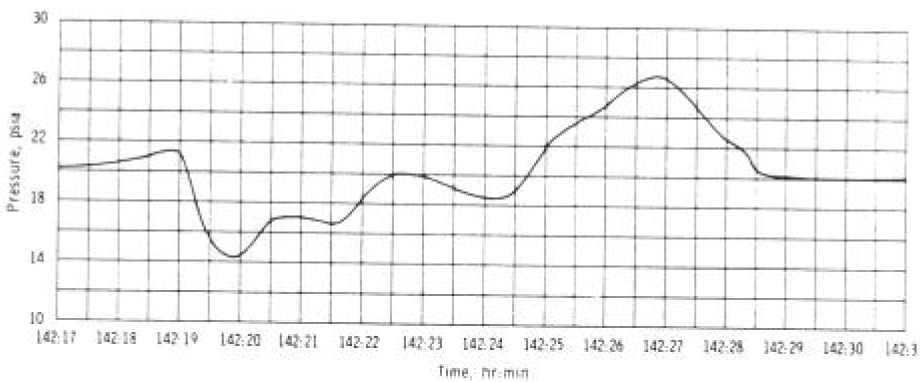
14.2 LUNAR MODULE

14.2.1 Water/Glycol Pump Differential Pressure Fluctuations

Variations were noted in the water/glycol pump differential pressure shortly after the cabin depressurizations for the standup extravehicular activity and the second extravehicular activity. The variations were similar on both occasions. The pressure differential decreased from about 20 psid (normal) to about 15 psid, then increased to about 27 psid and returned to normal (**Fig. 14-24**).



(a) Glycol pump differential pressure variations during standup extravehicular activity.



(b) Glycol pump differential pressure variations during second extravehicular periods

Figure 14-24.- Glycol pump differential pressure variations during extravehicular periods.

The total times for the cycles were 3 minutes during the standup extravehicular activity and 10 minutes during the second extravehicular activity. The pump discharge pressure remained relatively stable throughout both periods. If pressure fluctuations had taken place in the heat transport system, both the pump discharge pressure and differential pressure should have varied together.

After the second fluctuation occurred, water/glycol pump 2 was selected because of the erratic differential pressure. All parameters were normal during pump 2 operation. The crew reselected pump 1 prior to egress for the second extravehicular activity, and it operated satisfactorily. Later, after docking, the pump was turned off momentarily and the pump discharge pressure readout was verified as correct since it decreased to the accumulator pressure of 7.8 psia.

The lunar module cabin humidity was high at the initial manning for descent because the command module cabin humidity was high. Furthermore, a water spill in the lunar module cabin after the first extravehicular activity again produced high humidity. Consequently, water would have condensed on the cold 1/8-inch water/glycol sense lines between the pump assembly and the pressure transducer (**Figs. 14-25** and **14-26**), and the water would have frozen and sublimed at the next cabin depressurization. Since there is no flow in the sense lines, as little as 0.002 inch of condensed moisture on the outside of these lines would freeze the fluid and cause the fluctuations in the indicated differential pressure, but would not affect system operation. Consequently, no corrective action is required.

This anomaly is closed.

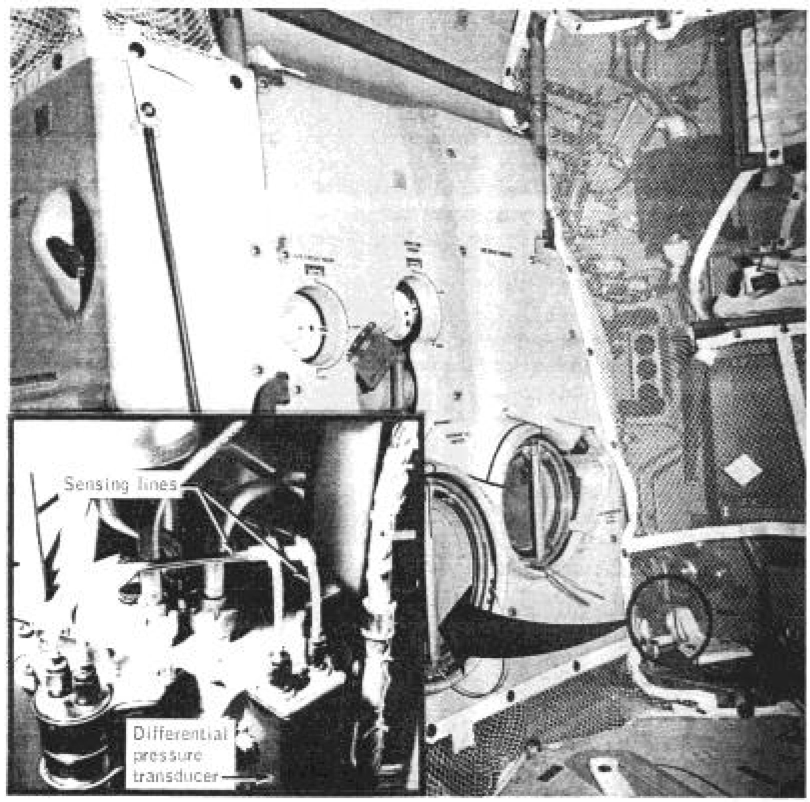


Figure 14-25.- Glycol pump sensing lines.

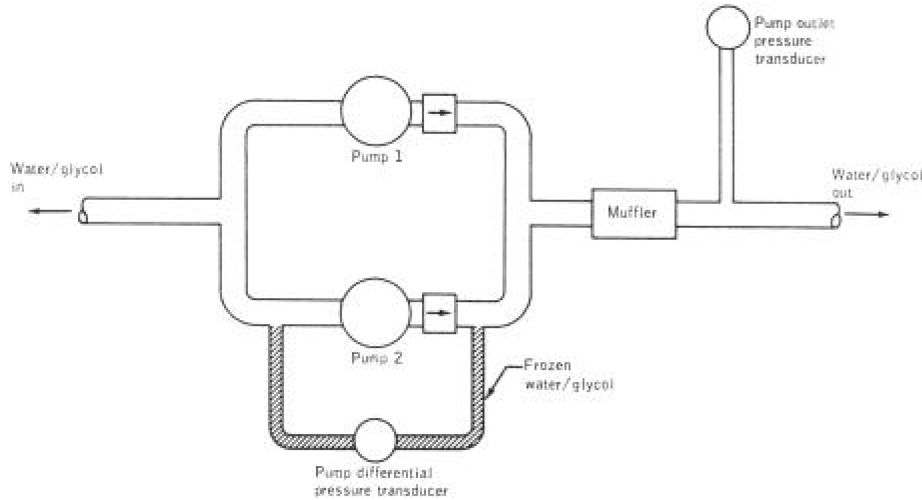


Figure 14-26.- Water/glycol pressure measurements.

14.2.2 Water Separator Speed Decrease

The speed of water separator 1 decreased to below 800 rpm and tripped the master alarm during the cabin depressurization for the standup extravehicular activity. Separator 2 was selected and operated properly at approximately 2400 rpm. After approximately 1 hour of separator 2 operation, separator 1 was reselected and performed satisfactorily throughout the remainder of the mission.

Cabin atmosphere is cooled and passed through one of the water separators (fig. 14-27) where condensed water is separated by centrifugal force and picked up by a pitot tube. The water is then piped from the pitot tube, through a check valve, to the water management system where it is used in the sublimator.

Cabin humidity was high before the standup extravehicular activity and, because the water and structure were cold, the line between the water separator pitot tube and the water management system was cold. Under these conditions, water would condense on the outside of the line (**Fig. 14-27**), and when the cabin was depressurized for the standup extravehicular activity, the water on the outside of the line would boil, freeze, and sublime, thereby freezing the water in the line.

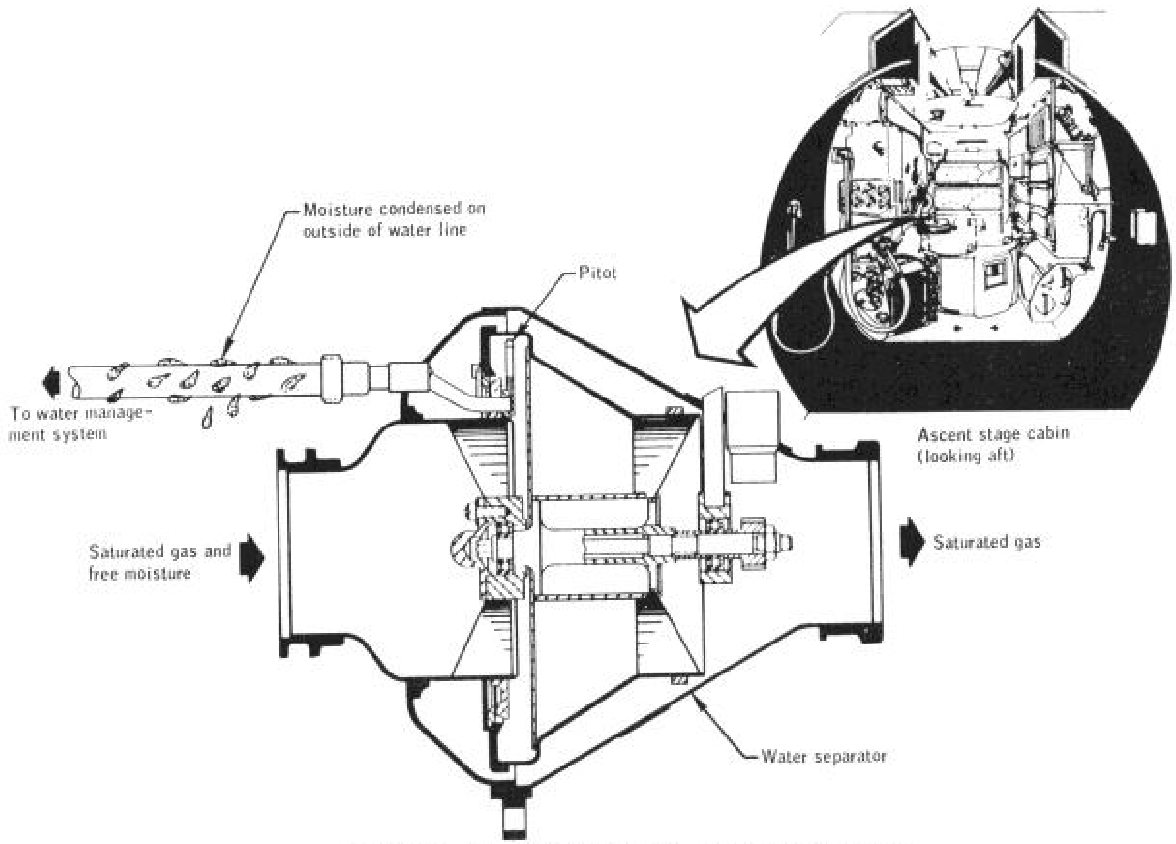


Figure 14-27.- Cross section of water separator.

Analysis shows that as little as a 0.01-inch film of water on the outside of the line will freeze the line at cabin depressurization. Freezing within the line will cause the separator to slow down and stall because of excessive water. Separator 2 had not been used. Therefore, no water was in its outlet line to freeze. Consequently, it operated successfully when activated.

Analysis and tests have shown that freezing of the line will not damage any spacecraft hardware. If such freezing occurs, the other separator would be used. Therefore, no corrective action is required.

This anomaly is closed.

14.2.3 Broken Water Gun/Bacteria Filter Quick Disconnect

A water leak occurred at the quick disconnect between the bacteria filter and the water gun (fig. 14-28) shortly after the first extravehicular activity. The leak was caused by the plastic portion of the disconnect being broken and was stopped by removing the filter.

When the water gun/bacteria filter combination is properly stowed, the gun is held in a

U-shaped boot (**Fig. 14-28**). Both the gun and filter are held in place by straps with the hose protruding above the liquid cooling assembly. Bending the hose can easily exceed the force required to break the quick disconnect between the bacteria filter and the water gun if the filter is not strapped down. A test showed that a torque of 204.6 inch-pounds caused a similar quick disconnect to break in half.

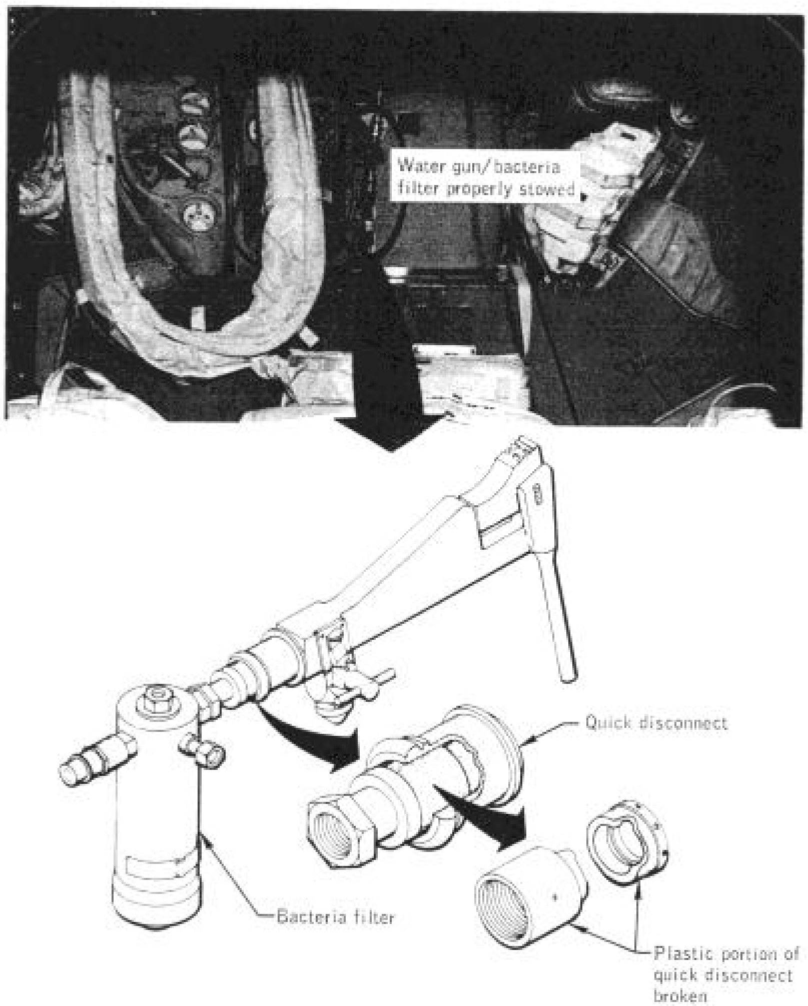


Figure 14-28.- Water gun/bacteria filter.

Based on the torque value of 204.6 inch-pounds, a force of approximately 27 pounds applied at the hose quick disconnect interface would break the disconnect between the filter and the water gun.

The plastic parts will be replaced by steel inserts in all locations where the quick disconnects are used in the lunar module and command module.

This anomaly is closed.

14.2.4 Intermittent Steerable Antenna Operation

Random oscillations occurred while the steerable antenna was in the auto-track mode. The oscillations were small, and damped without losing auto-track capability (**Fig. 14-29**). The three times when the oscillations became divergent are shown in **Fig. 14-30**.

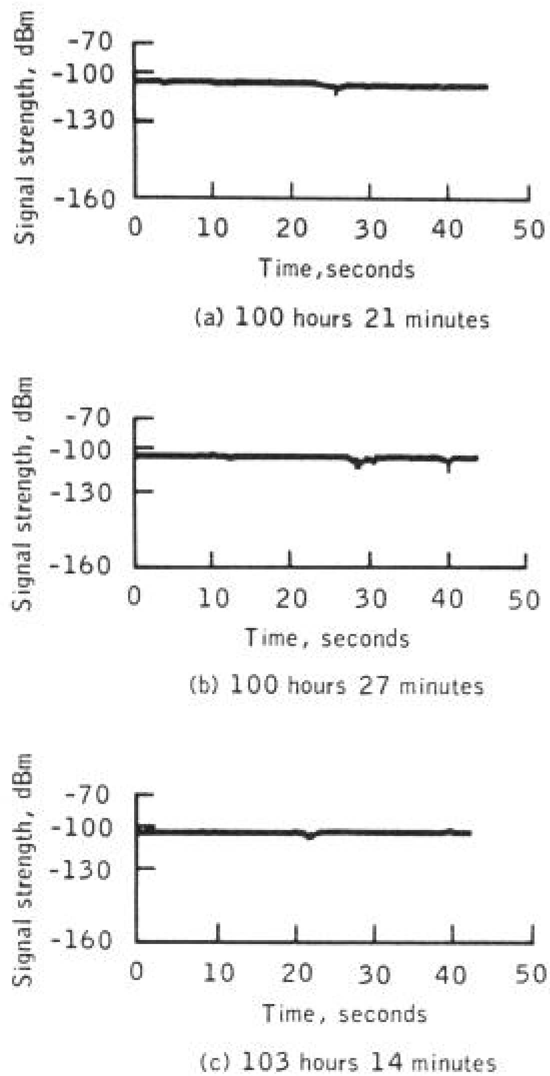


Figure 14-29.- Damped antenna oscillations at random times.

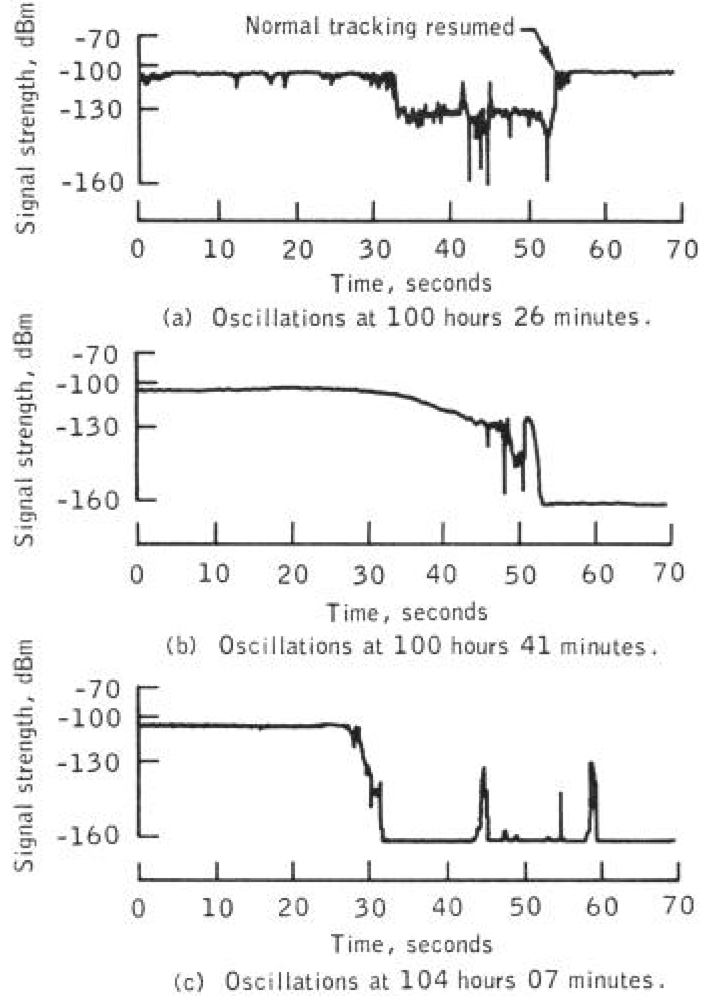


Figure 14-30.- Received signal strength plots associated with divergent antenna oscillations.

100:26 (revolution 12): This divergence occurred prior to separation and was not caused by vehicle blockage or reflections from the command and service module structure.

100:41 (revolution 12): The lunar module maneuver, performed approximately 2 minutes after separation, caused the antenna to track into vehicle blockage, which resulted in the antenna oscillation and loss of lock.

104:07 (revolution 14): Earth look angles for this time period indicate that the antenna was clear of any vehicle blockage.

All oscillations indicated characteristics similar to the conditions experienced on Apollo 14 (**Fig. 14-31**). After Apollo 14, one of the prime candidates considered as a possible cause was incidental amplitude modulation on the uplink signal. A monitor capable of detecting very small values of incidental amplitude modulation was installed at the Manned Space Flight Network Madrid site for Apollo 15. The data from this monitor indicated that no amplitude modulation existed on the uplink at the frequencies critical to antenna stability during the problem times. Consequently, incidental amplitude modulation has been eliminated as a possible cause of the antenna oscillations, and the problem must be in the spacecraft.

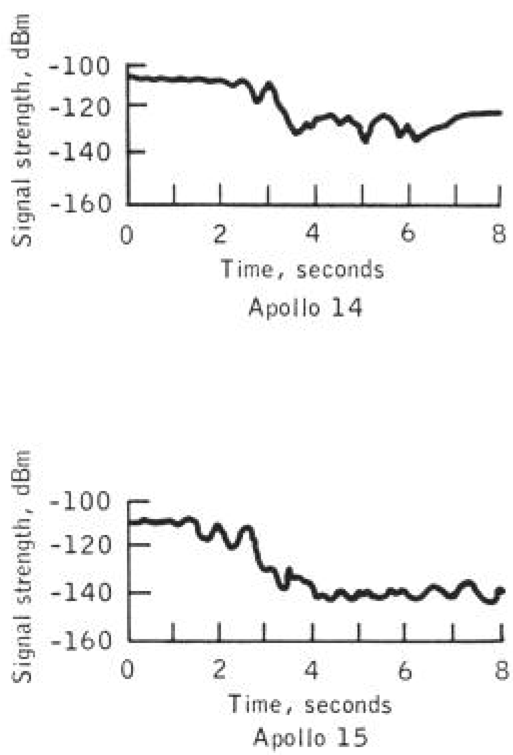


Figure 14-31.- Ground-received signal strength showing spacecraft antenna oscillation characteristics.

There are two possible causes of the problem in the spacecraft. The first is that electrical interference generated in some other spacecraft equipment is coupled into the antenna tracking loop. A test performed on the Apollo 16 lunar module showed that no significant noise is coupled into the tracking loop during operation of any other spacecraft equipment in the ground environment. The second is that temperature or temperature gradients cause an intermittent condition in the antenna which results in the oscillations. The acceptance tests of the antennas do not include operation over the entire range of environmental temperatures; therefore, a special test is being conducted on a flight spare antenna to determine its capability to operate over the entire range of temperatures and gradients.

This anomaly is open.

14.2.5 Descent Engine Control Assembly Circuit Breaker Open

The descent engine control assembly circuit breaker was found open during the engine throttle check after lunar module separation from the command and service module. The circuit breaker was closed and the check was successfully performed.

If there was a short circuit condition, it is highly unlikely that the fault would have cleared at the instant the breaker opened. Thus, when the breaker was reset, it would have reopened. Data were reviewed for current surges large enough to trip the 20-ampere breaker, but none were found. The crew stated that they may have left the circuit breaker open. This is the most probable cause of the anomaly.

This anomaly is closed.

14.2.6 Abort Guidance System Warning

Abort guidance system warnings and master alarms occurred right after insertion into lunar orbit and at acquisition of signal prior to lunar module deorbit. The first one was reset by the crew; the second persisted until lunar impact. Performance of the abort guidance system appeared normal before, during, and after the time of the alarms.

Exceeding any of the following three conditions in the abort guidance system can cause the system warning light to illuminate. In each case, the warning light would reset automatically when the out-of-tolerance condition disappears.

a. 28 ±2.8 volts dc

b. 12 ±1.2 volts dc

c. 400 ±15 hertz

A fourth condition which could cause the warning light to illuminate is the receipt of a test-mode fail signal from the computer. This condition requires manually resetting the warning light by placing the oxygen/water quantity monitor switch to the CW/RESET position.

The conditions for generating a test mode fail signal are as follows (none of these conditions was indicated in any of the computer data).

a. Computer restart: A restart would cause internal status indicators to be reset and telemetry quantity "Vdx" to be set to a prestored constant (minus 8000 ft/sec).
b. Computer self test: Computer routines perform sum checks of computer memory and logic tests. Failure of these would set a self-test fail status bit.

c. Program timing: If the computer program is executing any instruction (except a "delay" instruction) at the same time that a 20-millisecond timing pulse is generated, a test-mode fail will be generated.

Computer routines running at the time of the warning have a worst-case execution time of 18.425 milliseconds of the allowable 20 milliseconds; therefore, a timing problem should not have occurred.

After the warning at insertion, the crew read out the contents of the computer self test address 412, but there was no indication of a test-mode fail. The crew did not, however, reload all zeros into address 412 as is required to reset the flip-flop (fig. 14-32) which controls the test-mode fail output in the computer. Consequently, a second test-mode fail would not have caused an abort guidance system warning. The fact that a second warning did occur restricts the location of the failure to the output circuit of the computer, the signal conditioner electronics assembly, or the caution and warning system.

The abort electronics assembly test mode fail output drives a buffer (**Fig. 14-32**) in the signal conditioner electronics assembly which, in turn supplies the test mode fail signal to the caution and warning system. The test mode fail buffer is the only application in the lunar module where a transistor supplies the input signal to the buffer and where the low side of the input to the buffer is not grounded. Tests have shown that some buffers feed back 10-kHz and 600-kHz signals to the buffer input lines. These signals are completely suppressed when the low side of the buffer input is grounded.

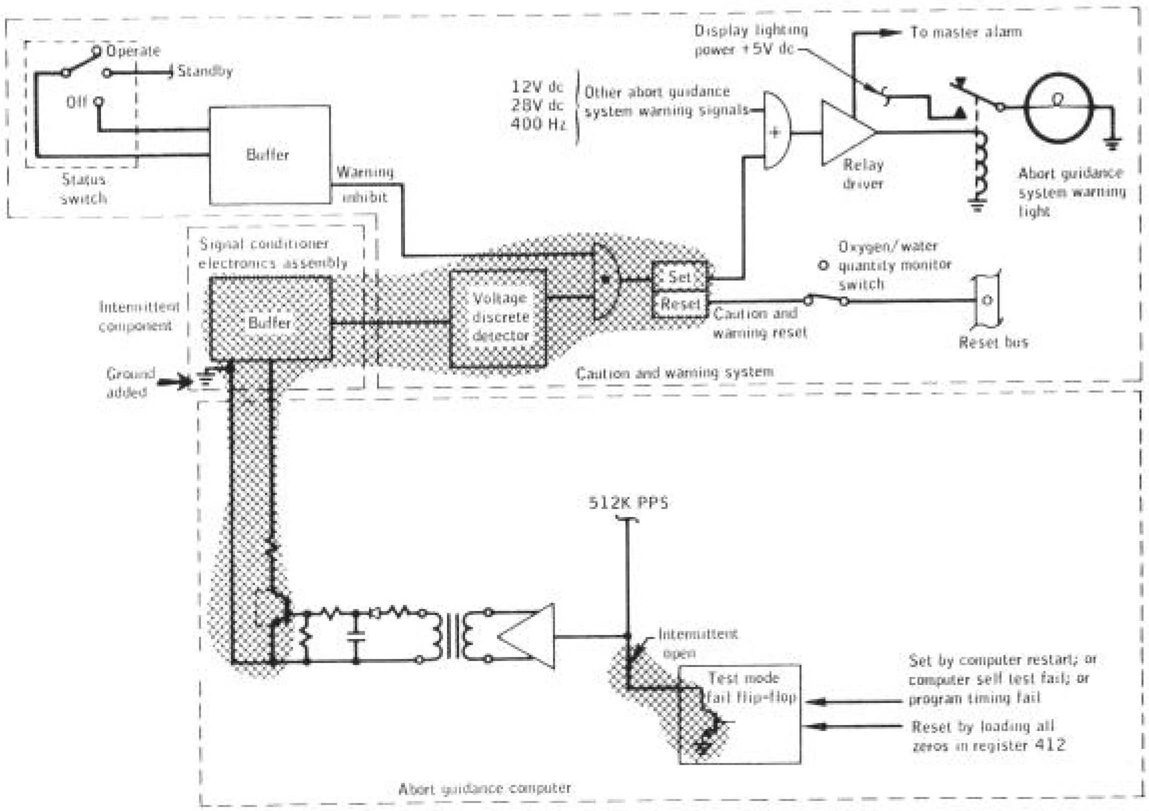


Figure 14-32.- Abort guidance system test mode fail circuitry.

With the noise signal voltages present, the test mode fail driver is more susceptible to electromagnetic interference. An analysis indicates that,, with worst-case noise signals present, an induced voltage spike of only about 3.5 volts will cause the test mode fail driver to momentarily turn on and latch the caution and warning system master alarm and abort guidance system warning light on.

For future spacecraft, the low side of the input to the buffer will be grounded.

This anomaly is closed.

14.2.7 No Crosspointer Indication

There was no line-of-sight rate data on the Commander's crosspointers during the braking phase of rendezvous. The existence of line-of-sight rates was verified by observing the position of the command module relative to the lunar module. The scale switch was in the low position; however, none of the other switch positions was verified. The power fail light was off, indicating that the Commander's crosspointer circuit breaker was closed.

The rate error monitor switch is used to select either rendezvous radar data or data selected by the mode select switch for display. The mode select switch selects velocities from the primary guidance system, the abort guidance system, or the landing radar for display. No telemetry data are available on the position of the rate error monitor switch and the mode select switch, and the Commander reported that he did not look at the flight director attitude indicator.

Four possible conditions could have affected the display of radar antenna rate data (**Fig. 14-33**).

a. The rate error monitor switch could have been in the wrong position. If the LDG RD/CMPTR position was selected, lateral velocity data from the abort guidance system would have been displayed, but only if the mode select switch had been in the AGS position. However, lateral velocity at the time of the problem would have been near zero. If the mode select switch had been in the PGNS position no data would have been displayed.

b. Conductive contamination between two contacts in the rate error monitor switch could have had the same effect on the crosspointer display as condition "a". The switch in this lunar module was X-rayed and screened before installation, and no contaminant was found; however, it should be pointed out that present screening techniques might not detect a single wire strand between two contacts.

c. An open in the return line would cause loss of rate data to one or both meters, depending upon the location of the open. The signal return lead from the Commander's meter in panel 1 is routed to panel 2 where it is connected to the signal return from the Lunar Module Pilot's meter and routed to the rendezvous radar.

d. An open in the wire in the rendezvous radar electronics assembly which connects 15 volts at 400 hertz to two velocity filters (one each for shaft and trunnion rate) could cause the loss of shaft and trunnion rate data to both sets of crosspointers.

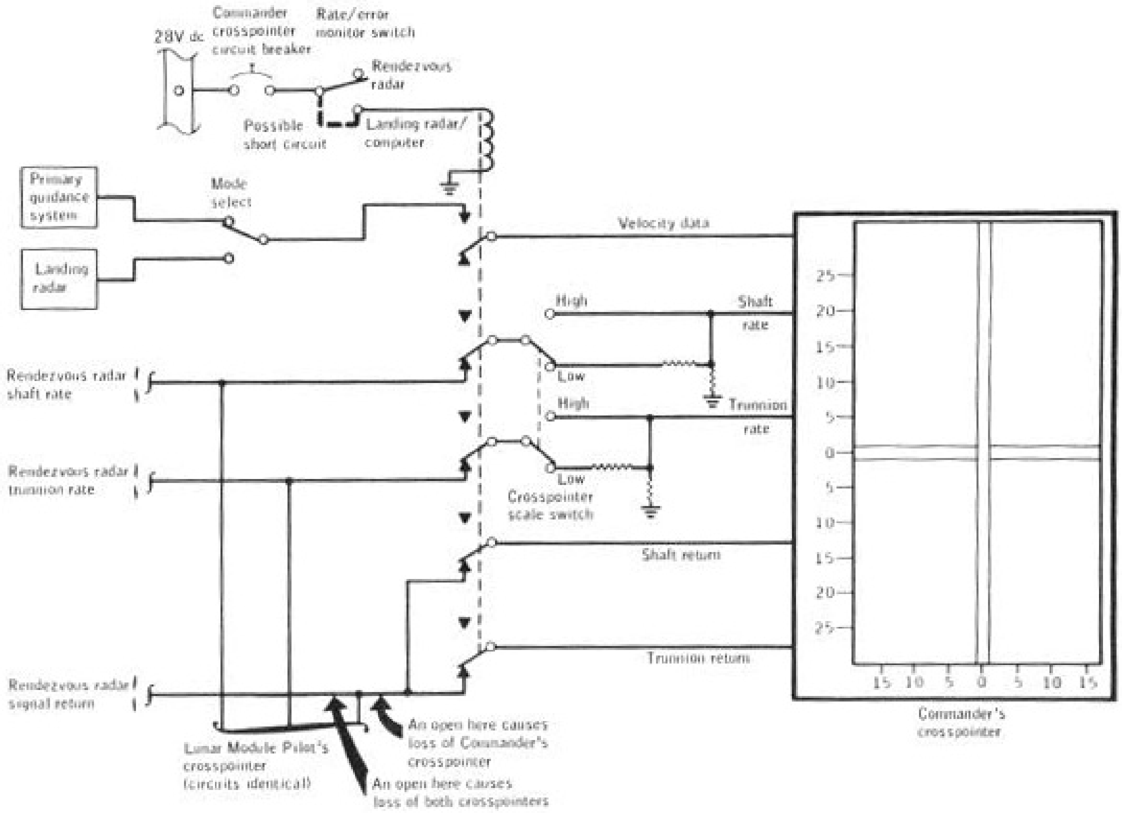


Figure 14-33.- Simplified schematic of crosspointer circuits.

The most probable failure is an open in the signal return line. Information could still be deduced from antenna position data which is displayed on the flight director attitude indicators. Ground tests and checkout may not show this type of failure. If the open is temperature sensitive, a complete vehicle test involving vacuum, temperature, and temperature gradients would be required to insure that failures of this type would not occur in the flight environment. This type of testing is not practical at the vehicle level. Consequently, no corrective action is planned.

This anomaly is closed.

14.2.8 Broken Range/Range Rate Meter Window

Sometime prior to ingress into the lunar module, the window of the range/range-rate meter broke (**Fig. 14-34**). Upon ingress, the crew saw many glass particles floating in the spacecraft, presenting a hazardous situation.

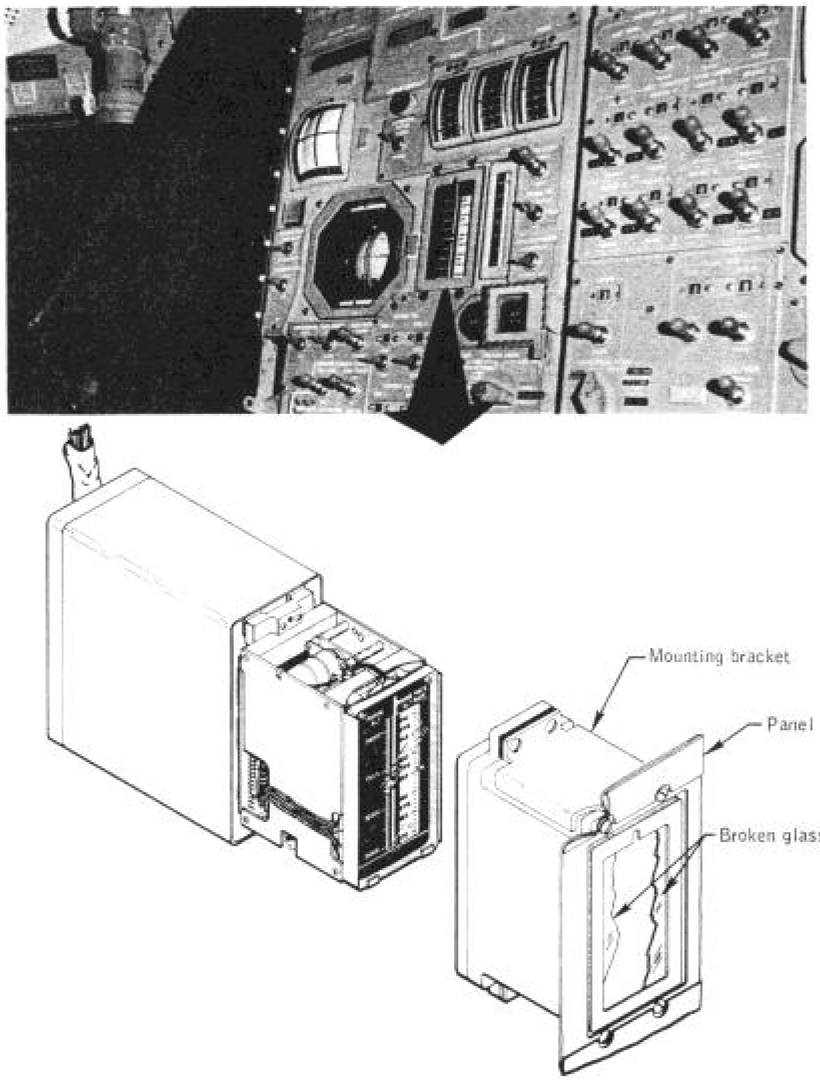


Figure 14-34.- Range/range rate meter mounting.

The window is an integral part of the meter case and is made of annealed soda-lime glass 0.085-inch thick. The meter case is hermetically sealed and pressurized with helium to 14.7 psia at ambient temperature. At the maximum meter operating temperature and with the cabin at vacuum, the pressure differential can be as high as 16.1 psi. This pressure differential is equivalent to a stress level of 6680 psi in the glass.

Glass will break when there is a surface flaw large enough to grow at the stress levels present. The threshold flaw size in a dry environment is about the same as the critical flaw size and immediate breakage occurs. The critical flaw size remains the same in a humid environment, but the threshold flaw is much smaller. For annealed soda-lime glass at a stress level of 6680 psi, the critical flaw depth is 0.0036 inch, and for a humid environment, the threshold flaw depth is 0.000105 inch.

A surface flaw deeper than the threshold depth for the glass operating stress must have existed on the outside of the meter window at launch. The flaw started growing as the cabin depressurized during the launch phase, and finally grew large enough for the glass to break.

For future missions, an exterior glass doubler will be added to the flaw depth to 0.0036 inch and the critical flaw depth to 0.032 inch. This should prevent future fatigue failures since there are no reported fatigue failures in soda-lime glass at stresses below 2000 psi. In addition, all similar glass applications in the lunar module and command module were reviewed and changes are being made. In the command module, transparent Teflon shields will be installed on the:

a. Flight director attitude indicators.

b. Service propu1sion system gimbal position and launch vehicle propellant tank pressure indicator.

c. Service propulsion system oxidizer unbalance indicator, and the oxidizer and fuel quantity indicators.

d. Entry monitor system roll indicator.

In the first three above applications, the shields will be held in place with Velcro and will be installed only when the cabin is to be depressurized. The shield on the entry roll monitor indicator will be permanently installed.

In the lunar module, tape will be added to the flight director attitude indicators and an exterior glass shield will be installed over the crosspointers to retain glass particles. The data entry and display assembly window was previously taped to retain glass particles.

This anomaly is closed.

14.3 SCIENTIFIC INSTRUMENT MODULE EXPERIMENTS

14.3.1 Panoramic Camera Velocity/Altitude Sensor Erratic

Telemetry received from the first panoramic camera pass on revolution 4 indicated that the velocity/altitude sensor (**Fig. 14-35**) was not operating correctly.

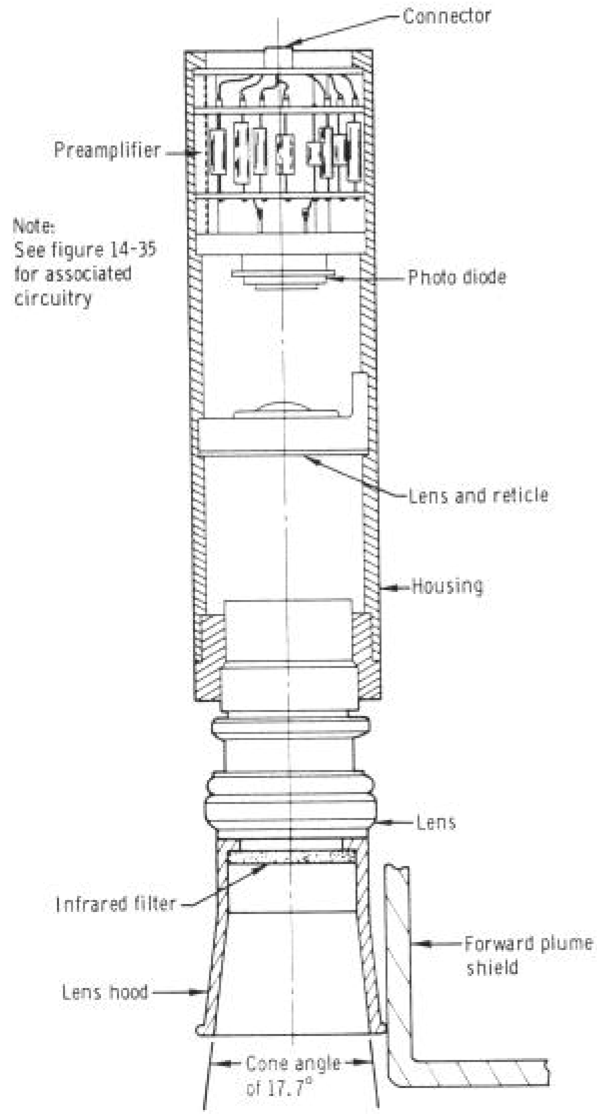


Figure 14-35.- Velocity/altitude sensor assembly.

The velocity/altitude sensor measures the angular rate of travel of the spacecraft relative to the lunar surface. The sensor output is used to control the cycling rate of the camera, the forward motion compensation, and the exposure. The sensor normally operates in the range of 45 to 80 miles altitude. If, at any time, the indicated velocity/altitude is out of this range, the sensor automatically resets to the nominal value of 60 miles. The sensor operated properly for brief periods of time, but would drift off-scale high (saturate), and then reset to the nominal value corresponding to a 60-mile altitude.

Breadboard tests and circuit analyses of the velocity/altitude electronics (**Fig. 14-36**) did not indicate failure. Tests were conducted in which endless belts of lunar scene photography from Apollo 8 and 15 were passed in front of velocity/altitude sensors. Sensors from the prototype and qualification units, and flight unit number 1 were used. By varying the illumination level, sensor performance somewhat similar to the Apollo 15 anomaly could be obtained.

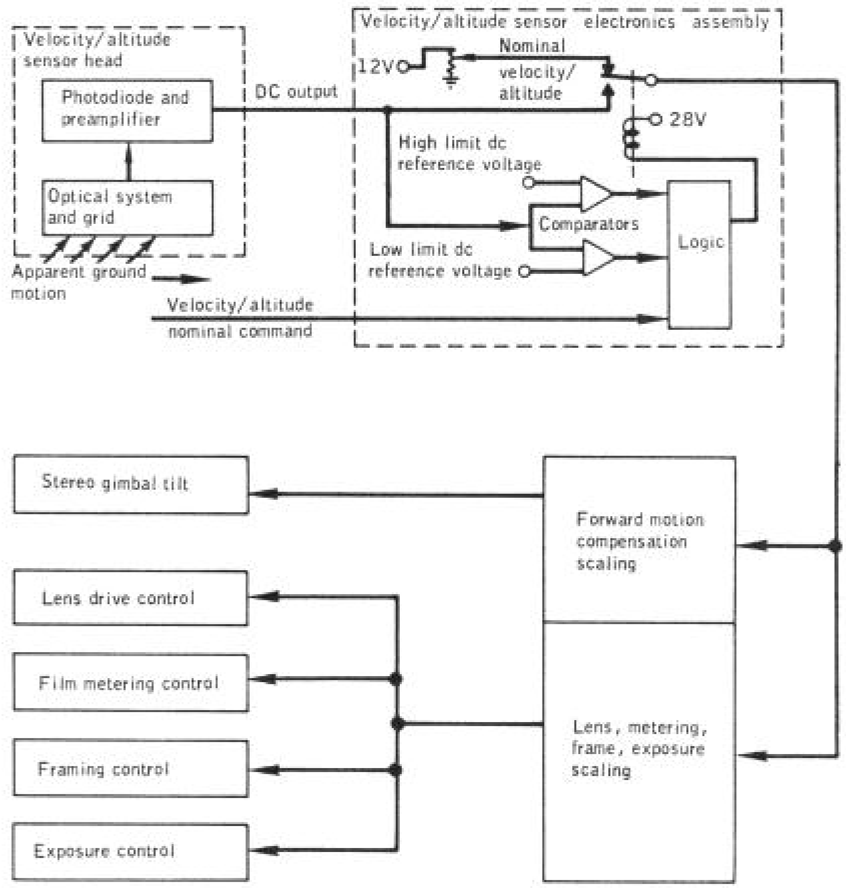


Figure 14-36.- Velocity/altitude output circuitry.

The results of the tests, coupled with analyses of the basic sensor design, indicate that the problem is related to the optical signal-to-noise ratio. The remaining flight hardware will be modified to improve this ratio. The optical signal will be enhanced by increasing the lens aperture from f/4.0 to f/3-5 and by deleting the infrared filter. The optical noise (reflections) will be reduced by increasing the length of the lens hood and by repositioning the sensor so that the camera's forward plume shield will not be in the field of view of the sensor. In addition, a manual override of the velocity/altitude sensor will be provided on the remaining flight units. By using a three-position switch, two preselected velocity/altitude ratios will be provided, as well as the automatic function.

This anomaly is closed.