APPENDIX B

REPORT OF MISSION EVENTS PANEL

CONTENTS

Part Page

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PART B1

TASK ASSIGNMENT

Panel 1 was assigned the task to develop a detailed and accurate chronology of mission events directly related to the flight of Apollo 13. This event sequence would then form a baseline of data for analytical use by Panel 1, other Panels, and the Review Board.

To provide such a chronology, Panel 1 worked to produce a consolidated sequence of all data whether derived from telemetry records, crew observations, inflight photographs, air-to-ground communications, or other sources of information. Of special significance to Panel 1 was the requirement to correlate data taken from different sources, such as crew observations and telemetry, in order to provide greater assurance of the validity of data wherever possible.

In order to provide meaningful boundary conditions for its work, Panel 1 divided its effort into three areas:

1. Preincident events, which covered the flight from countdown to the time of the inflight accident.

2. Incident events, which covered the flight from approximately 55 hours and 52 minutes to the conclusion of immediately related data events.

3. Postincident events, which covered the subsequent mission period to splashdown.

In each of the three areas the main purpose of the Panel was to provide the most efficient presentation of events for the Board's use in reviewing, evaluating, and interpreting the significance of mission events. Consequently, Panel 1 devoted a considerable portion of its time to the task of data interpretation and verification. As was intended from the Charter of the Board, the primary focus of the Panel's work was the period of time during which the service module encountered serious inflight difficulties, and its presentation of data reflects this particular emphasis.

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PART B2

PANEL ORGANIZATION

Panel 1 was chaired by Mr. Francis B. Smith, Assistant Administrator for University Affairs, NASA Headquarters, Washington, D.C. The Board Monitor was Mr. Neil Armstrong from the Manned Spacecraft Center. Additional Panel Members were:

Mr. John J. Williams, Kennedy Space Center, for preincident events

Dr. Thomas B. Ballard, Langley Research Center, for incident events

Mr. M. P. Frank, Manned Spacecraft Center, for postincident events

Although each of the above specialized in one phase of the Panel's total assignment, the Panel acted as one unit in the review and assessment of data and in the analysis and interpretation of those events identified with the accident.

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PART B3

SUMMARY OF EVENTS

Apollo 13 was launched on schedule from Kennedy Space Center at 2:13:00 e.s.t. on April 11, 1970. The crew consisted of James E. Lovell, Commander (CDR); John L. Swigert, Command Module Pilot (CMP); and Fred W. Haise, Lunar Module Pilot (LMP). The preflight countdown was routine and although some malfunctions and anomalies occurred during boost and earlier portions of the flight, none except the premature cutoff of one of the S-II engines was considered at the time to be of a serious nature.

At about 55:54, the crew had just completed a television broadcast; CMP Swigert was in the left seat of the command module, LMP Haise was in the lunar module, and CDR Lovell was in the CM lower equipment bay, when all three heard a loud bang. At about the same time in Mission Control in Houston, the Guidance Officer (GUIDO) noted on his console display that there had been a momentary interruption of the spacecraft computer. He told the Flight Director, "We've had a hardware restart. I don't know what it was." At almost the same time, CDR Lovell, talking to Mission Control, said, "I believe we've had a problem here." Also at about the same time, the Electrical, Environmental, and Communications Engineer (EECOM) in Mission Control noticed on his console display the sudden appearance of limit sensing lights indicating that a few of the telemetered quantities relating to the spacecraft's cryogenic, fuel cell, and electrical system had suddenly gone beyond pre-set limits. Astronaut Swigert in the command module, noting a master alarm about 2 seconds after the bang, moved from the left seat to the right seat where he could see the instruments indicating conditions of the electrical system, and noticed a caution light indicating low voltage on main bus B, one of the two busses supplying electrical power for the command module. At that time, he reported to Mission Control, "We've had a problem. We've had a main B bus undervolt." At the same time, however, he reported the voltage on fuel cell 3, which supplied power to main bus B, looked good and assumed that the main bus B undervolt condition had been a transient one. However, 2 or 3 minutes later, when another master alarm sounded, LMP Haise moved into the right-hand seat to recheck the fuel cells and noted that two of the three fuel cells (no. 1 and no. 3) were showing no hydrogen or oxygen flow and no electrical output and that fuel cell 2 was carrying the command module's total electrical load through bus A. Bus B was dead. In addition, several other electrical and cryogenic system abnormalities were evident.

Detailed studies and analyses of telemetry records made since the flight indicated that during the 90 seconds before the "bang", several abnormal events occurred. At about 55:53:23, within a few seconds after the crew had turned on two fan motors which stir the supercritical cryogenic oxygen in oxygen tank no. 2, electrical "glitches" (transient high-amplitude current and voltage fluctuations) occurred which could be indicative of momentary electrical short circuits. Analyses of telemetry data also indicate that first one fan motor and then the other probably became disconnected from the electrical bus concurrently with the glitches. Thirteen seconds after the first glitch (16 seconds after the fans were turned on) the pressure in oxygen tank no. 2 started to rise; during the next 24 seconds it increased from a normal value of 891 psia to 954 psia; it remained at that pressure for approximately 21 seconds and then again increased to a maximum value of 1008 psia (approximately the pressure at which the relief valve was set to open), at which point the relief valve apparently opened and pressure began decreasing. During the last 23 seconds of this period, during the second oxygen pressure increase, telemetry indicated that oxygen tank no. 2 temperature also began to increase sharply; and concurrently with the sudden temperature rise, the oxygen tank no. 2 quantity gage, which had been inoperative for the previous 9 hours, began to show fluctuating readings. At about 90 seconds after the start of the pressure rise, telemetry transmission from the spacecraft was suddently interrupted for a period of 1.8 seconds.

Putting all of this and other information together with the service module photographs taken later by the crew and with subsequent changes in the condition of the spacecraft system leads to a determination that immediately before and during this 1.8-second interval the following things happened:

1. The oxygen tank no. 2 system failed, leading to loss of all oxygen pressure.

2. The service module panel covering bay 4 blew off, possibly producing the "bang" heard by the crew.

3. The spacecraft's velocity changed by 0.5 fps.

4. Transmission of telemetry from the spacecraft was interrupted (possibly caused by the panel striking and damaging the high-gain antenna through which data were being telemetered).

5. Various valves in the reaction control systems (RCS) were shocked closed (contributing to some difficulties in maintaining automatic attitude control).

6. Valves controlling oxygen flow to fuel cells 1 and 3 were shocked closed (leading to failure of both fuel cells 2-1/2 minutes later for lack of oxygen).

7. Oxygen tank no. 1 started leaking oxygen.

8. Venting of oxygen produced forces on the spacecraft which the automatic stabilization system counteracted by firing opposing spacecraft reaction control thrusters.

9. Various sensors or their wiring were damaged to cause subsequent erroneous readings.

These changes occurred so rapidly, of course, that neither the crew nor the mission controllers could have had a clear picture of specifically what had happened.

In the Mission Control Center, after the 1.8-second data loss, the EECOM first suspected an instrumentation failure since earlier in the flight (46:40) the oxygen tank no. 2 quantity gage had failed and since other pressures, temperatures, voltages, and current readings were so abnormal (e.g., more than 100 percent or less than 0 percent of full scale) as to appear unrealistic. They appeared more indicative of an instrumentation failure than of real quantities. The Flight Director also initially believed, from the information available to him in the Control Center, that the difficulty was electrical or electronic in nature. Consequently, Mission Control Center's initial efforts during the first 3 or 4 minutes after the malfunction were to validate instrument readings and to identify a possible instrumentation failure. During the next several minutes, both the flightcrew and the ground controllers worked at switching fuel cell bus power configurations in an attempt to understand what had happened and to get fuel cells 1 and 3 back on line. They determined that fuel cell 1 had no output and disconnected it from the bus. Later they also disconnected fuel cell 3 for the same reason. For several minutes they connected the command module's entry battery to bus A to aid fuel cell 2 in supplying electrical power and to insure against further failures due to low voltage.

Shortly after the malfunction, while the Apollo 13 crew and the EECOM were trying unsuccessfully to restore electrical power output from fuel cells 1 and 3, the Guidance and Navigation Officer (GNC) reported an unusually high level of attitude control thruster activity on the spacecraft. This added to their problems, since it indicated other abnormal conditions aboard the spacecraft and used excessive thruster fuel. Consequently, during the next hour the ground control and the crew were required to pay a great deal of attention to maintaining attitude control of the spacecraft and to identifying and eliminating the cause of the instability. At the same time, the Flight Director began to suspect that the genesis of the problem might lie in the RCS, rather than in the high-gain antenna or instrumentation.

During this period (about 14 minutes after the accident) CDR Lovell reported, "...it looks to me, looking out the hatch, that we are venting something. We are venting something out into space......it's a gas of some sort." He subsequently described this venting as extremely heavy and unlike anything he had seen in his three previous space flights.

For about 1 hour 45 minutes after the accident, the crew and ground controllers wrestled with electrical problems caused by oxygen supply and fuel cell failures and with attitude stability problems caused by the venting of oxygen, the shock closing of thruster system valves, and electrical system failures. During this period they went through a series of control system reconfigurations until automatic control was finally established at 57:32. In the meantime, as it became more apparent that the loss of oxygen from oxygen tank no. 1 could not be stopped and that fuel cell 2 would soon expire, the LM was powered up (57:40), LM telemetry was turned on (57:57) and attitude control was transferred from the CM to the LM (58:34). At 58:40, 2 hours 45 minutes after the accident, the CM was completely powered down.

One of the main concerns then was to make the trajectory changes that would return the spacecraft safely to Earth within the lifetime of the onboard consumables--water, oxygen, thruster fuel, and electric power. At the time of the accident the spacecraft was on a trajectory which would have swung it around the Moon (about 21 hours after the accident) and returned it to Earth where it would have been left in a highly elliptical orbit about the Earth with a perigee (nearest approach to Earth) of about 2400 miles. Four trajectory correction burns were made during the remainder of the flight as illustrated in figure B6-9.

<u>61:30</u> - A 38 fps incremental velocity (delta V) burn using the descent propulsion system (DPS) engine and the LM primary guidance and navigation system (PGNS). This burn was performed 16 hours before they swung around the Moon, and was targeted to place the spacecraft on a trajectory which would return it to the atmospheric Earth reentry corridor rather than the 2400-mile perigee.

<u>79:28</u> - A 861 fps delta V burn using the DPS 2 hours after swinging around the Moon to speed up return to Earth by about 9 hours (143 versus 152 g.e.t.) and to move the landing point from the Indian Ocean to the Pacific Ocean where the primary recovery forces were located.

<u>105:18</u> - A 7.8 fps delta V burn using DPS to lower perigee altitude from 87 miles to about 21 miles.

<u>137:40</u> - A 3.2 fps delta V final burn using LM RCS thruster to correct for small dispersions in previous burns and assure that the spacecraft would reenter in the center of its entry corridor.

During the remainder of the flight there were several other unusual situations which the crew and Mission Control successfully contended with. The use of electrical power aboard the LM had to be managed very carefully to conserve not only the LM batteries but also the water supply, since water was used to dissipate heat generated by the electrical equipment. The LM LiOH was not adequate to remove carbon dioxide for three men for the duration of the return trip, so a method was devised to circulate the LM cabin oxygen through the CM's LiOH filters. Since the CM had to be used for reentry, its main bus B had to be checked out very carefully to assure that there were no electrical shorts and the CM entry battery which had been used earlier to supply power for the ailing CM had to be recharged from the LM batteries.

Several actions essential to reentry and landing were undertaken during the last 9 hours of the flight as illustrated in figure B6-10. The SM was jettisoned a few minutes after the last midcourse correction, about 4-1/2 hours before reentry. In viewing and photographing the SM, the crew realized for the first time the extensiveness of the physical damage (panel blown off, Mylar strips hanging from antenna, etc.). At about 2-1/2 hours before reentry, the CM's inertial platform was powered up and aligned and the LM was jettisoned about 1/2 hour later. Reentry was at 142:40 and splashdown at 142:54 g.e.t.

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PART B4

PRELAUNCH AND MISSION EVENTS PRIOR TO THE ACCIDENT

This section of the report contains significant events prior to the accident with emphasis placed on the spacecraft and particularly on the cryogenic system. It starts with the launch count (T - 98:00:00) and ends prior to the significant events of the accident (55:52:00).

LAUNCH COUNTDOWN

Countdown operations for both the command service module (CSM) and lunar module (LM) were started at approximately 10:00 a.m. e.s.t. on Monday, April 6, 1970. The start of the countdown was delayed approximately 8 hours because of a pad clear operation involving a special test of the LM supercritical helium (SHe) system. A timeline of significant countdown milestones is shown in figure B4-1.

Mechanical Build-up and Gas Servicing

Following completion of CSM powerup, water servicing, and securing of the LM SHe operation, installation of the CSM heavy ordnance initiators was started at approximately 3:00 p.m. e.s.t. The ordnance operation and remote resistance checks of the launch escape rocket initiators were completed by 9:30 p.m. e.s.t., April 6, after being slightly delayed to correct a mechanical interference problem (incorrect thread depth) with the initiator in the launch escape rocket motor. Combined CSM and LM helium and gaseous oxygen (GOX) servicing was started at 2:00 a.m. e.s.t. on April 7, and was successfully completed by noon that day. At this time, both the CSM and LM were functional at T - 66:00:00, at which point a built-in hold of 12 hours had been originally planned. As a result of the late countdown start, both the LM and CSM spacecrafts experienced only a 6-hour built-in hold.

From noon Tuesday, April 7, through 11:00 a.m. Thursday, April 9, mechanical build-up operations (panel closure, LM thermal blanket installation, etc.) were conducted on the CSM and LM. The CSM fuel cells were activated and preparations were completed for CSM cryo loading, that is, filling the cryogenic oxygen and hydrogen tanks. Details of this operation are covered below. During this time the LM SHe tank was initially loaded and a 24-hour cold soak period started. All of these operations were completed without a significant problem, with the

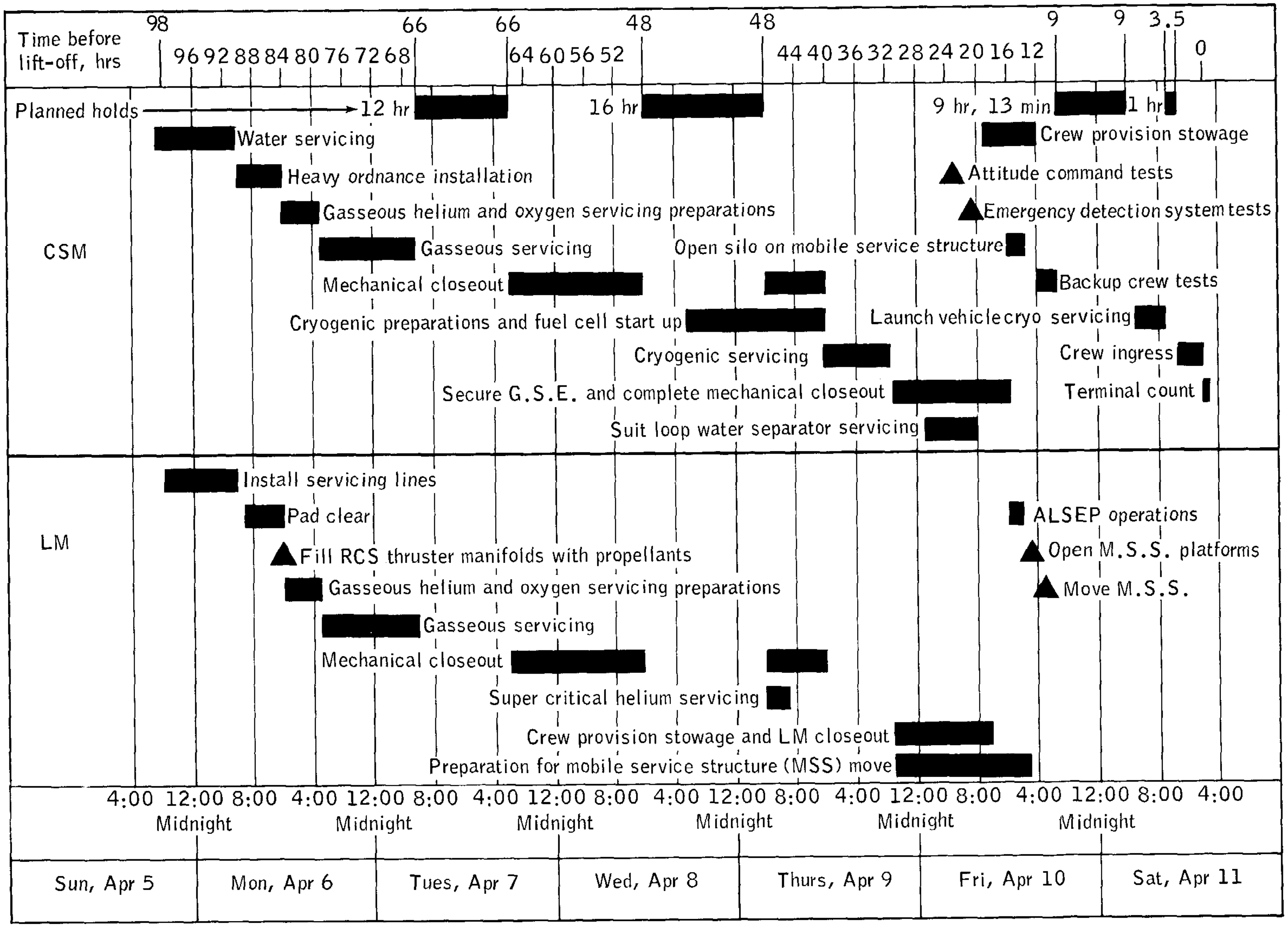

Figure B4-1.- Planned launch countdown timeline, e.s.t.

spacecraft progressing functionally from T - 66:00:00 to T - 41:00:00; including completion of the built-in hold at T - 66:00:00 and another planned 16-hour built-in hold at T - 48:00:00.

Cryogenic Servicing

CSM cryo loading or flowing liquid hydrogen and liquid oxygen was scheduled to be performed from 11:00 a.m. e.s.t. through 7:00 p.m. e.s.t. Thursday, April 9, 1970. A timeline of significant milestones, including preliminary preparations, is shown in figure B4-2. (See Appendix A, Part A5 for a description of the fuel cell and cryogenic systems.) The configuration of the cryogenic and fuel cell systems was as follows:

1. The fuel cell gaseous oxygen and hydrogen systems were at a pressure of 28 psia with oxygen and hydrogen gases. The fuel cells had been operated in the countdown demonstration test (CDDT) and were left pressurized with reactant gases (gaseous oxygen and hydrogen) to maintain system integrity between CDDT and countdown.

2. The oxygen and hydrogen tanks were at a pressure of 80 psia with oxygen and hydrogen gases. The tanks had been evacuated (less than 5mm Hg for 2 hours minimum) and serviced during CDDT, with reactant gas left in the system after detanking to maintain system integrity between CDDT and countdown.

3. The ground support equipment (GSE) lines were connected to the spacecraft and had been previously evacuated, pulse purged, and then pressurized with reactant gas to 80 psia. Purity samples taken of the gases from the GSE were within specification. The pressure-operated disconnects (POD's) that connect the GSE to the spacecraft had been leak checked at 80 psia with reactant gas and indicated no leakage.

4. The portable oxygen dewar used to service the spacecraft oxygen tanks was serviced on April 7, 1970. Liquid samples taken from the vent line of the dewar during servicing were within specification. All of the preceding activities were accomplished without undue delay or difficulty.

The first activity for the fuel cell and cryogenic system in the countdown started at approximately 3:00 p.m. e.s.t. on April 8, 1970. The move of the liquid hydrogen and liquid oxygen dewars from the cryogenic buildings to the pad had been completed. The primary oxygen, backup oxygen, and backup hydrogen dewars were located on the pad at the base of the mobile service structure (MSS) while the primary hydrogen dewar was moved to level 4A of the MSS. The hydrogen and oxygen GSE configuration is shown in figures B4-3 and B4-4, respectively.

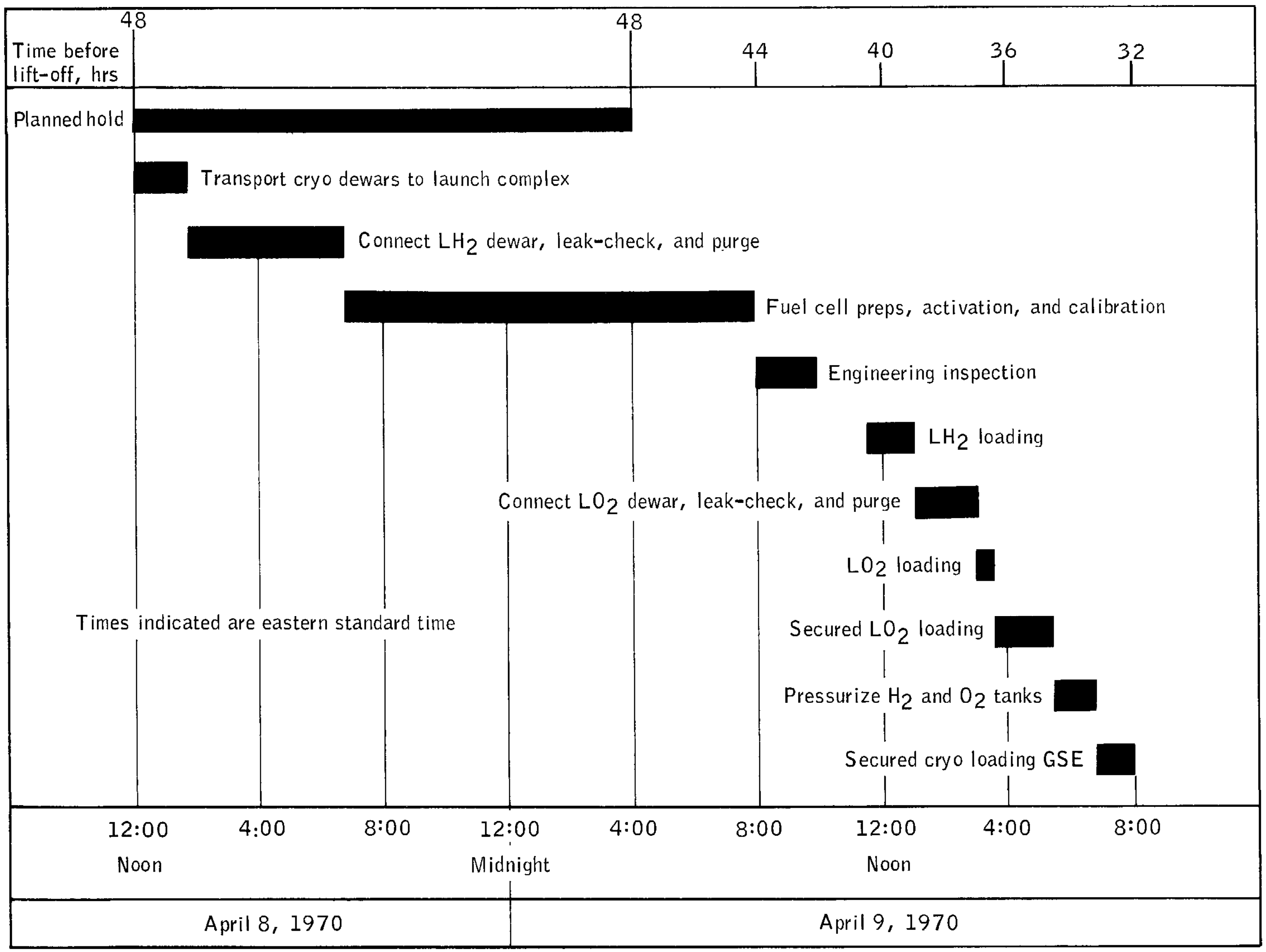

Figure B4-2.- Fuel cell activation and cryogenic servicing timeline.

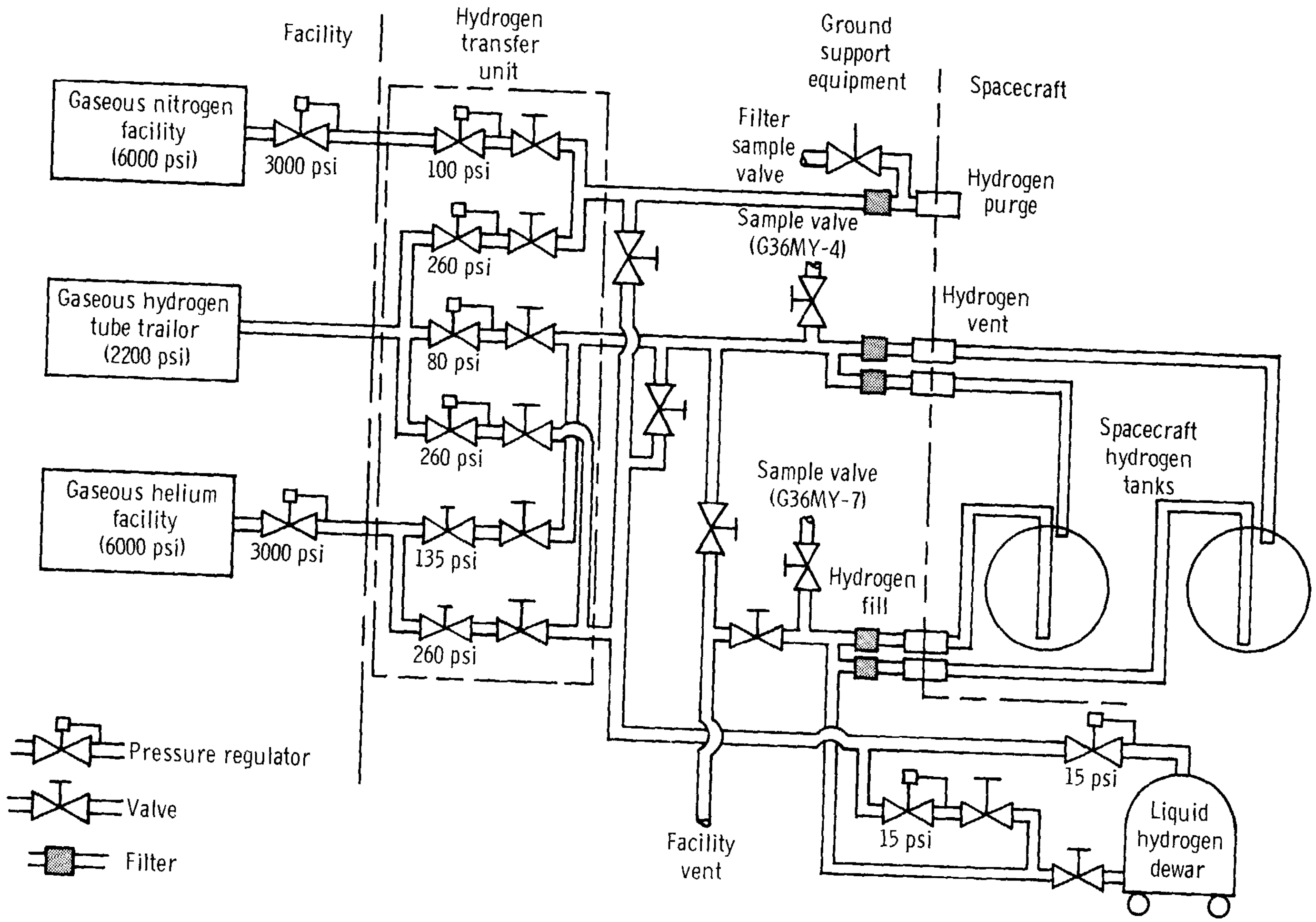

Figure B4-3.- Hydrogen servicing configuration.

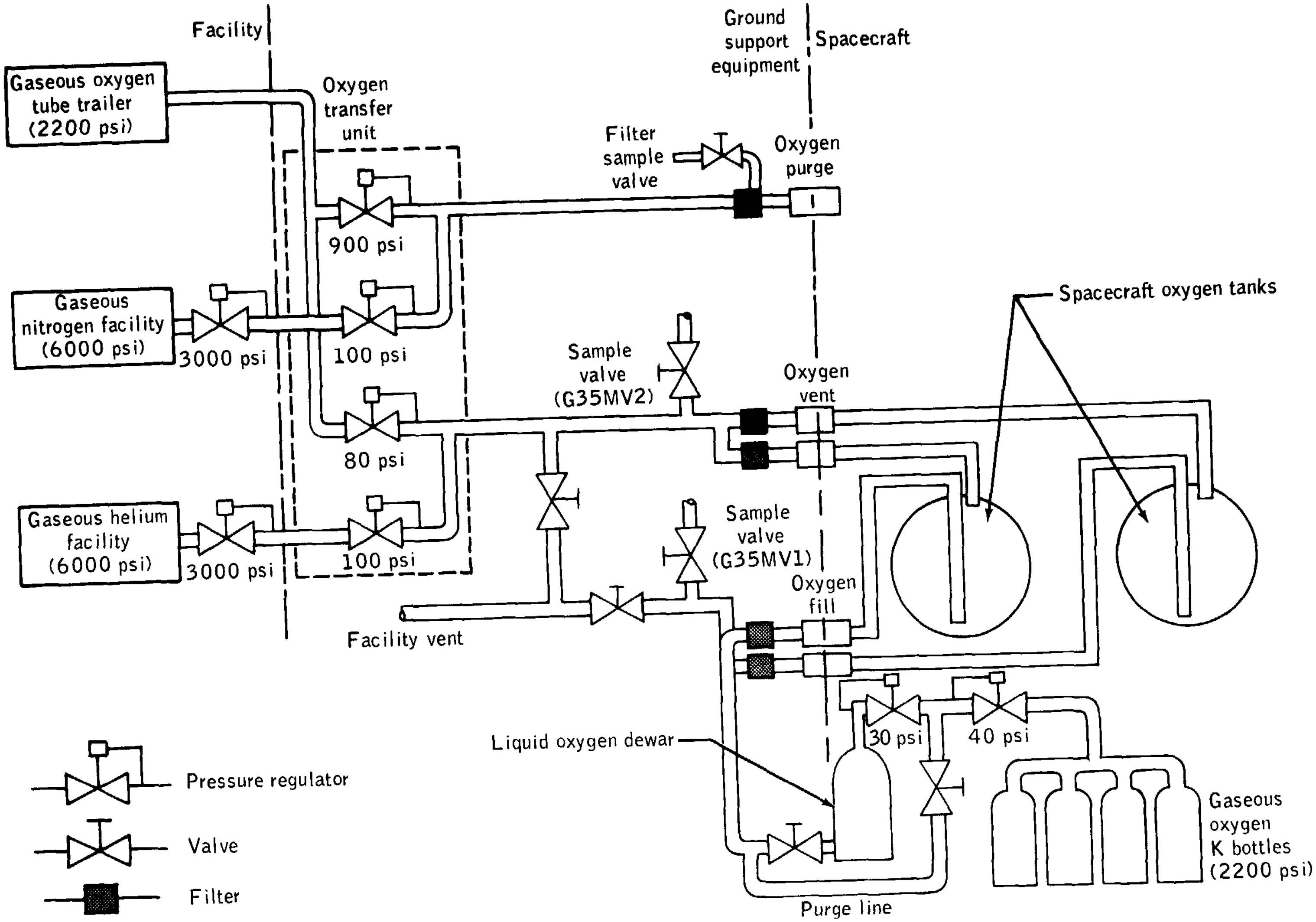

Figure B4-4.- Oxygen servicing configuration.

Pictures of the servicing dewars, valve boxes, and pressurizing equipment are shown in figures B4-5 through B4-10.

Dewpoint samples of the oxygen and hydrogen spacecraft tanks were obtained. This was accomplished by pressurizing the tanks with reactant gas to 80 psia through the vent line and then venting the tank back through the vent line and obtaining a moisture sample at the vent line sample valve. Both the oxygen and hydrogen tanks met the requirements that the moisture content be less than 25 parts per million (ppm). Oxygen tanks no. 1 and no. 2 read less than 2 ppm.

After the dewpoint samples of the tanks were obtained, sample bottles were installed on the tank vent lines. The sample bottles were flow purged with reactant gases at 80 psia for 5 minutes, followed by 10 pulse purges ranging in pressure from 80 psia to 20 psia.

The hydrogen dewar was then connected to the servicing GSE. The fill line between the dewar and the spacecraft was flow purged with 55 psia of helium gas for 15 minutes, and a moisture sample taken from the fill line. A sample result of 2 ppm was obtained. An additional flow purge using gaseous hydrogen at 55 psia was then performed for 10 minutes, followed by 13 pulse purges ranging in pressure from 55 psia to 20 psia (Note: This cleans the dead-end areas at the manifold).

The fuel cells were then pressurized to their operating pressure (62 psia oxygen and hydrogen). Heat was applied electrically to the fuel cells from external GSE to melt the potassium hydroxide. Fuel cell 3 heater current, supplied from GSE for heatup, was slightly low (1.2 amps vs. 1.4 amps). This heater current was adjusted after the heatup and calibration of the fuel cells was completed.

With the fuel cells at operating temperature (420° F) and pressures, a calibration test on each fuel cell was performed. Fuel cells were calibrated by applying loads in approximately 10-amp increments until a maximum current of 60 amps was reached while monitoring the output voltage. The fuel cell loads were supplied by GSE load banks. After calibration, the fuel cells were connected to the spacecraft busses and 40-amp GSE load applied to each cell for fuel cell water conditioning (approximately 4 hours). After these loads were removed from each fuel cell, 6-amp in-line heater loads with a 50-percent duty cycle were applied. With the fuel cells in this configuration a visual engineering inspection of the liquid hydrogen and liquid oxygen loading systems was performed with the exception of the liquid oxygen dewar, not yet connected.

Immediately prior to flowing liquid hydrogen, the spacecraft hydrogen and oxygen tank fans and quantity probe circuit breakers were

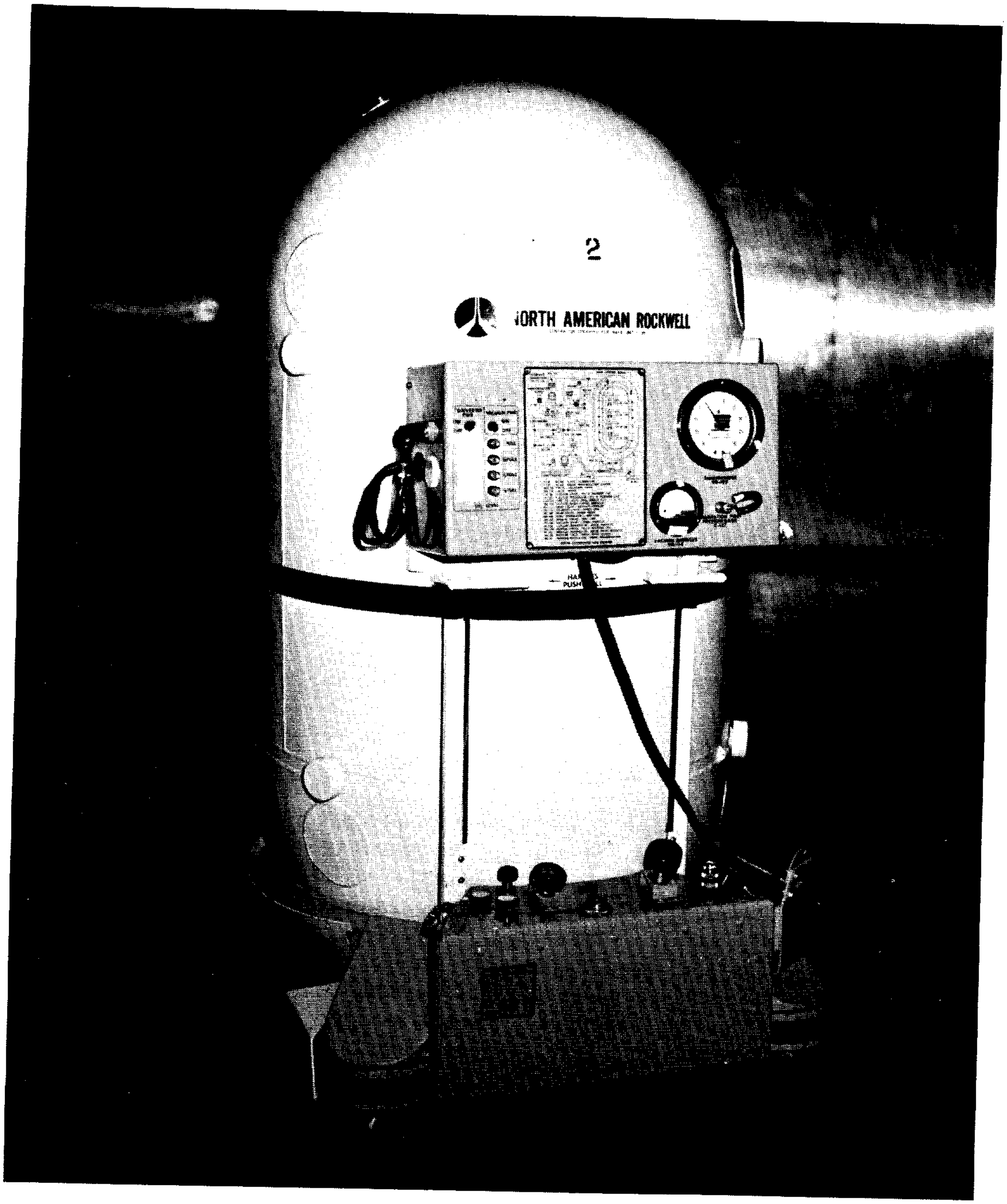

Figure B4-5.- Liquid hydrogen dewar.

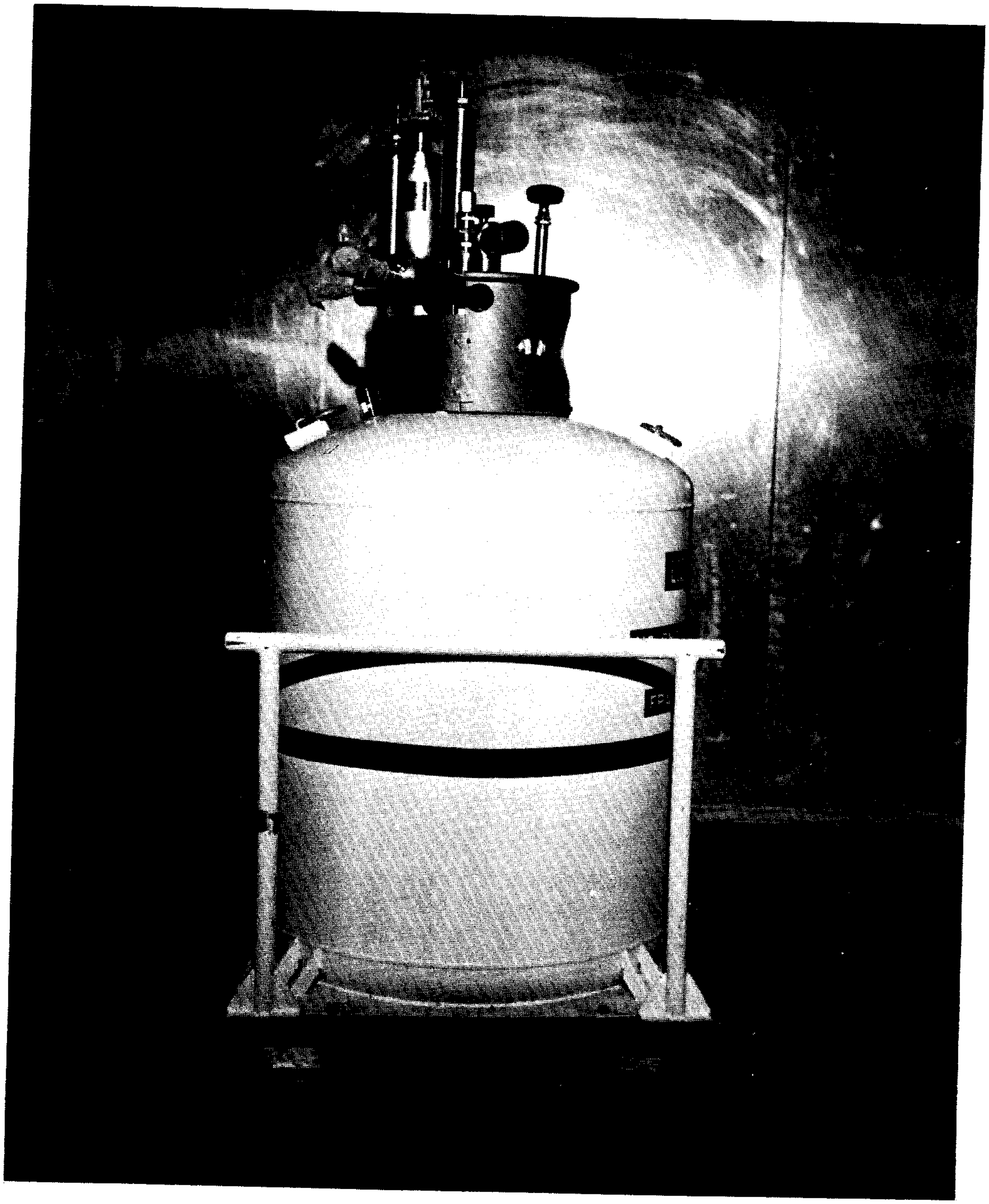

Figure B4-6.- Liquid oxygen dewar.

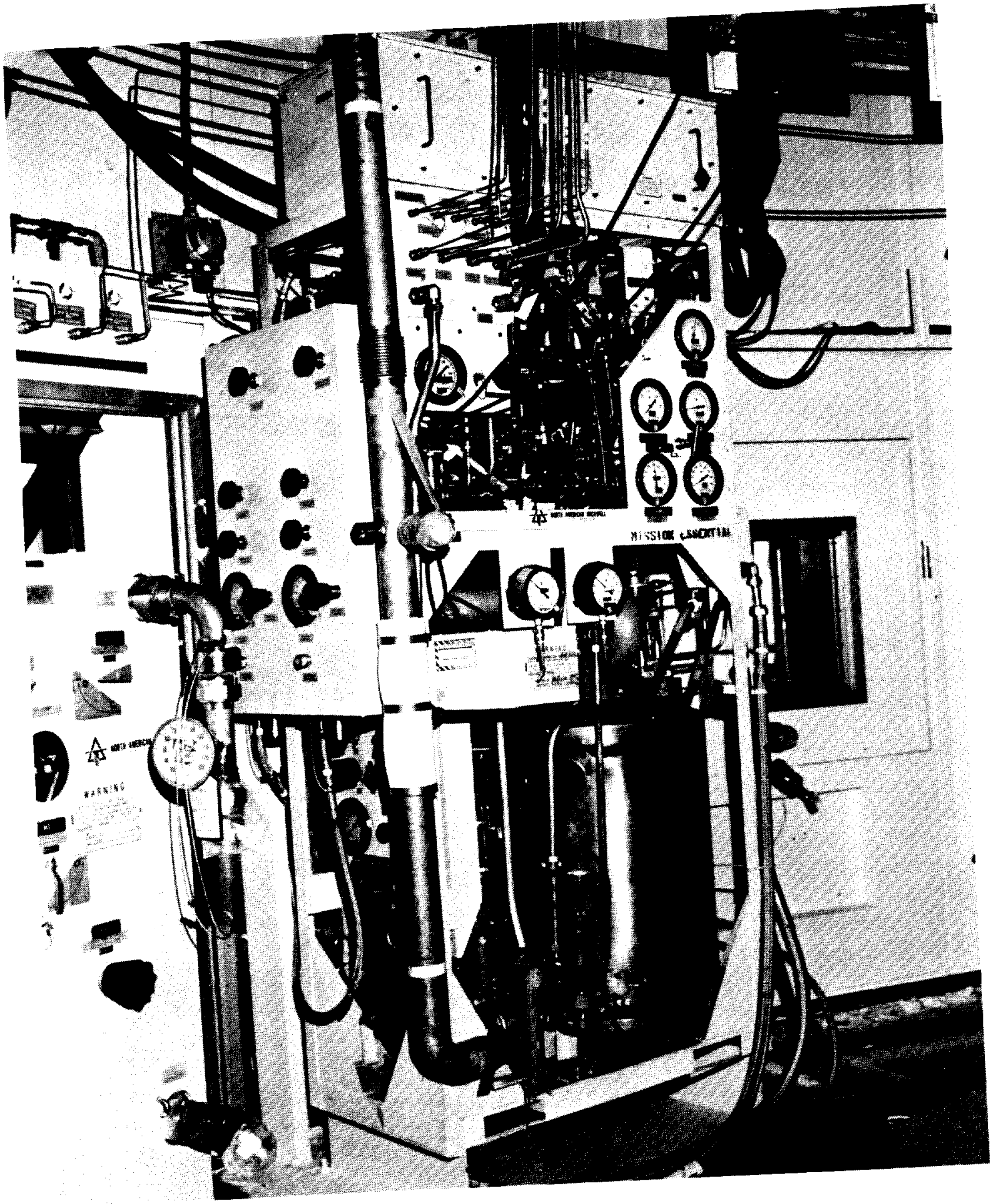

Figure B4-7.- Hydrogen valve box at Launch Complex 39.

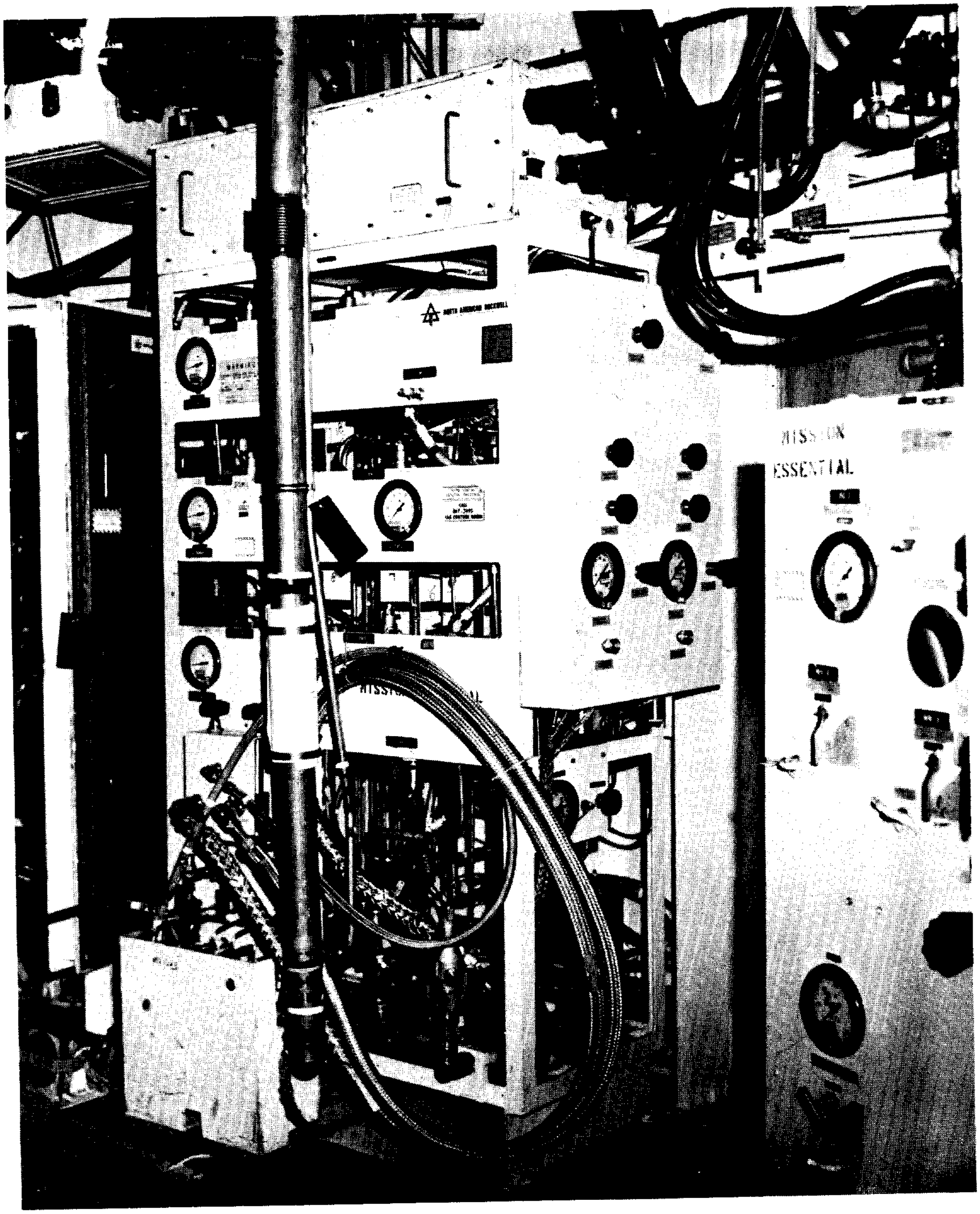

Figure B4-8.- Oxygen valve box at Launch Complex 39.

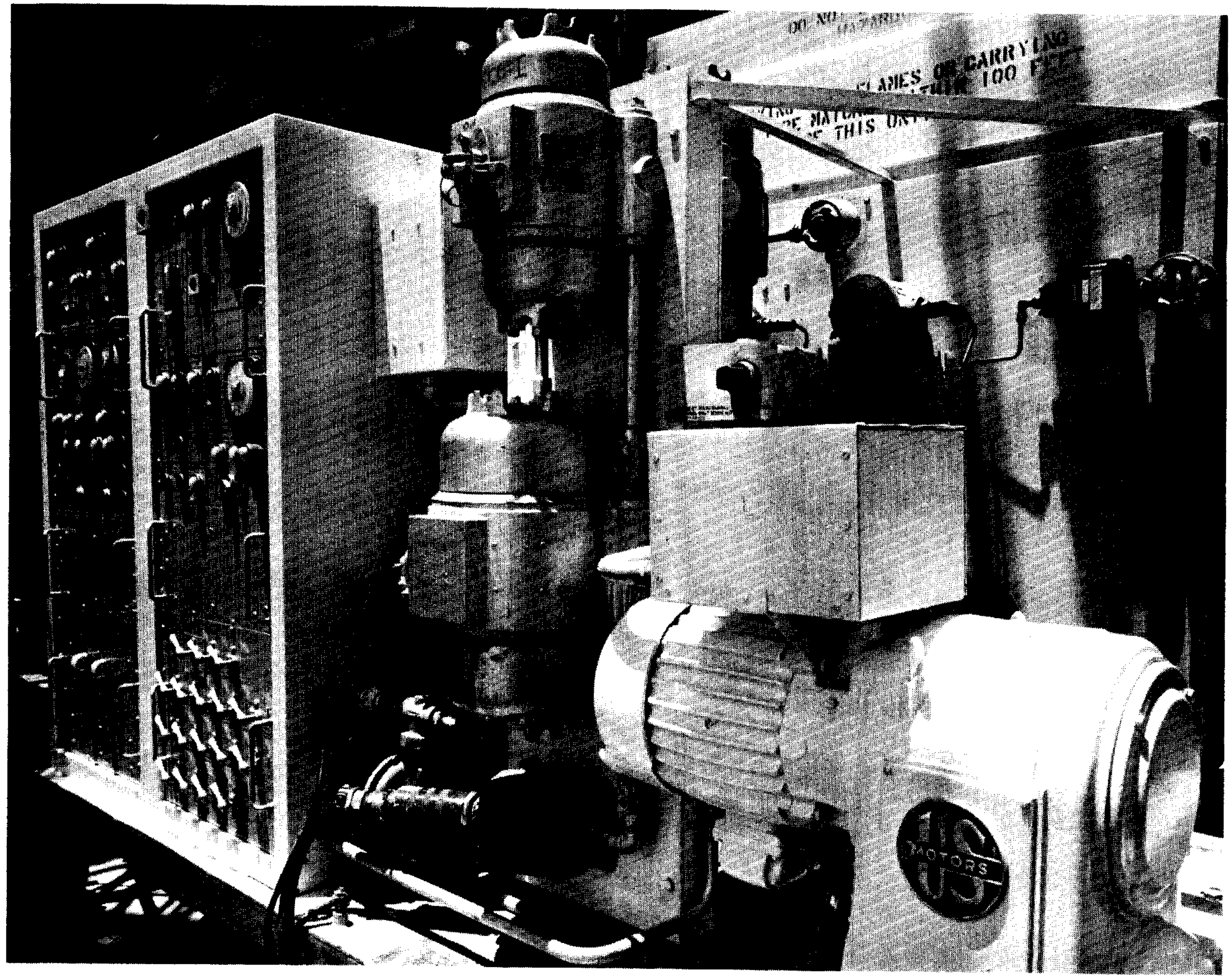

Figure B4-9.- Hydrogen transfer unit at Launch Complex 39.

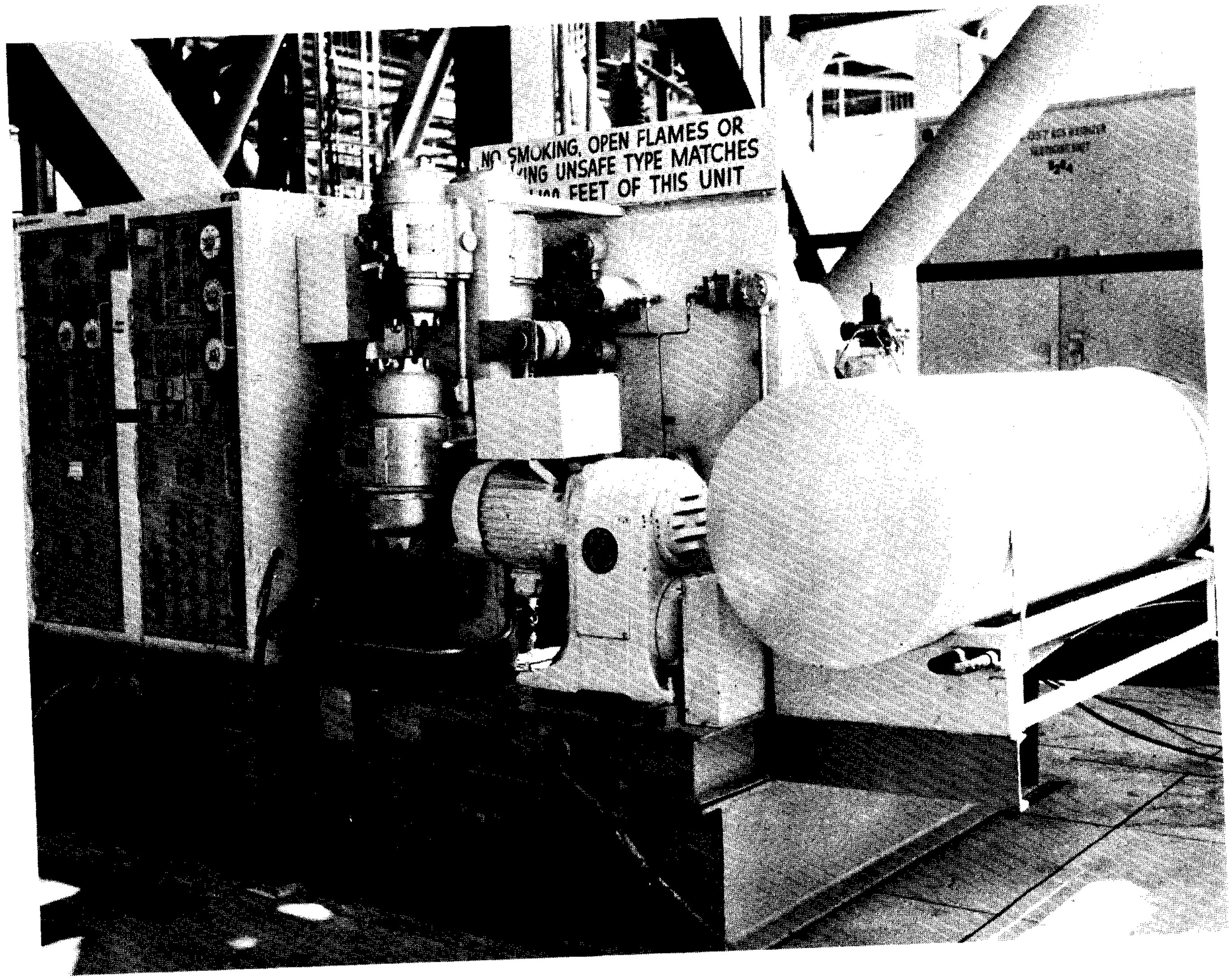

Figure B4-10.- Oxygen transfer unit at Launch Complex 39.

closed. (See Appendix A, Part A5, for description of the oxygen and hydrogen tanks.) The hydrogen dewar was pressurized to approximately 30 psia prior to servicing. Hydrogen was flowed through both tanks for 10 minutes (normal) prior to obtaining an increase in tank quantity. This period is required to chill the system. The flow rate during servicing was approximately 2.1 pounds per minute for 22 minutes (both tanks). The flow was stopped for 30 minutes when the tank quantity reached 85-percent and the dewar and spacecraft tanks vented to ambient pressure. The fans were turned off during this period. This time period is required to chill the hydrogen tank. The dewar was again pressurized to approximately 30 psia, and flow (at normal rates) began through the fill manifold detank line for 2 minutes to chill the GSE prior to then opening the spacecraft fill POD's. When the quantity gage stabilized (about 98-percent) the dewar pressure was increased to approximately 35 psia and the vent POD's closed, followed closely by the closure of the fill POD's. The GSE vent valve was closed simultaneously with the closing of the spacecraft vent POD's. This operation traps cold gas between the spacecraft vent POD's and the GSE vent valve. As the cold gas warms and expands, it is vented into the two sample containers connected to the vent line sample valve. The samples were analyzed for helium, nitrogen, and total hydrocarbons. Both samples were within specifications.

The hydrogen dewar was removed and the prime oxygen dewar was brought up to level 4A of the MSS. The oxygen dewar was connected to the servicing GSE. The fill line between the dewar and the spacecraft was flow purged with 55 psia of oxygen gas for 15 minutes, and a moisture sample taken from the fill line. A sample result of less than 2 ppm was obtained. After sampling, 13 pulse purges from a pressure of 55 psia to a slight positive pressure to maintain flow were performed. The spacecraft oxygen tank fans were turned on prior to oxygen flow. The oxygen dewar was pressurized to approximately 45 psia. Oxygen was flowed through both tanks for approximately 2 minutes (normal) before an indication was noted on the quantity probe. The flow rate during servicing was 25 pounds per minute for approximately 25 minutes (both tanks). After the tank quantity reached 100 percent, flow was continued for an additional 10 minutes, to further chill the tanks. The spacecraft vent POD's and the GSE vent were then closed, followed immediately by the closure of the fill POD's. The spacecraft tank fans were turned off at this time. The cold gas trapped in the vent line was sampled. The oxygen is sampled for helium, nitrogen, and total hydrocarbons. Both samples were within specification. The service module supply valve was opened to allow the CM surge tank to pressurize for flight.

While pressurizing the surge tank, fuel cell 1 was connected to dc bus A to minimize the usage of liquid hydrogen. A constant flow from the liquid hydrogen tanks equal to the heat gained by the tank results in minimum liquid hydrogen usage. The load on the fuel cell was approximately 20 amps. This configuration was maintained until 4 hours before launch, at which time fuel cells 2 and 3 were connected to the busses. Fuel cells 1 and 2 were connected to bus A with fuel cell 3 supplying power to bus B. The fuel cells supplied power to the spacecraft from this time through launch.

Ground electrical power was supplied to the tank heaters to bring the tanks to flight pressure. The liquid oxygen system pressurization to approximately 935 psia and the liquid hydrogen system to approximately 235 psia was completed by 6:40 p.m. on April 9, 1970. The fuel cells were supplied by onboard reactants from this period through launch. Fan motor checks were performed, and the GSE and airborne systems closed out for flight.

The entire CSM cryo loading operation was normal except that liquid hydrogen tank no. 1 was loaded to 98.7 percent instead of the desired minimum 99 percent (reason for this is still under study by both the Manned Spacecraft Center and the Kennedy Space Center) and a slight leak developed through the liquid oxygen tank no. 2 vent quick disconnect. The leak was stopped by the installation of the flight cap prior to tank pressurization. These conditions were determined to be acceptable for flight.

Spacecraft Closeout and Terminal Count

Following completion of the cryo loading operation the countdown proceeded normally from T - 32:00:00 through such milestones as: LM crew provision stowage and final closeout; LM SHe servicing; launch vehicle battery installation and electrical systems checks; CSM crew provision stowage; backup astronaut crew checks; and ALSEP fuel cask installation.

At 7:00 p.m. e.s.t. on April 10, 1970, the countdown clock was held at T - 9:00:00 for a planned built-in hold of 9 hours and 13 minutes. Following resumption of the countdown at 4:13 a.m. e.s.t. on April 11, 1970, final launch vehicle cryogenic loading preparations were completed and launch vehicle cryogenic loading was successfully conducted through 9:30 a.m. e.s.t.

The remainder of the countdown activities, including flightcrew ingress, final CSM cabin closeout, and the space vehicle terminal count, progressed normally with the exception of a minor problem with a broken key in the CSM pyro guard, and a stuck open no. 2 liquid oxygen vent valve in the S-IC stage. Both problems were satisfactorily resolved within the planned countdown time, which included a final built-in hold of 1 hour at T - 3:30:00 minutes.

LAUNCH AND TRANSLUNAR COAST PHASE PRIOR TO THE ACCIDENT

Launch and Flight Summary

The space vehicle was launched at 2:13:00 e.s.t., April 11, 1970. The only unexpected occurrence during the boost phase was an early shutdown of the S-II inboard engine. Low frequency oscillations (approximately 16 hertz) occurred on the S-II stage, resulting in a 132-second premature center engine cutoff. Preliminary analysis indicates that an engine pressure sensor detected a varying engine thrust chamber pressure resulting from a large pressure oscillation in the liquid oxygen system and turned the engine off. The four remaining engines burned approximately 34 seconds longer than normal, and the S-IVB orbital insertion burn was approximately 9 seconds longer to achieve the required velocity. The cause of the liquid oxygen system oscillation is presently being studied by the Marshall Space Flight Center. A parking orbit with an apogee of 100.2 nautical miles and a perigee of 98.0 nautical miles was obtained.

After orbital insertion, all launch vehicle and spacecraft systems were verified and preparations were made for translunar injection. The second S-IVB burn was initiated on schedule for translunar injection.

All major systems operated satisfactorily and conditions were nominal for a free-return circumlunar trajectory. With the spacecraft in a free-return trajectory, and with no further major propulsion burns, the spacecraft would pass around the Moon and reenter the Earth's atmosphere.

The command service module (CSM) separated from the service module LM adapter (SLA) at 3:06:39. The spacecraft was maneuvered and docked with the lunar module (LM) at 3:19:09 and the LM separated from the SLA at 04:01:00. The S-IVB was then maneuvered using residual propellants to impact the lunar surface. The first midcourse correction (23.1 fps), performed at 30:40:50 using the service propulsion system, inserted the spacecraft into a non-free-return trajectory with a pericynthian altitude close to the planned value of about 60 miles. Under these conditions, with no further propulsion engine burns, the spacecraft would orbit the Earth in a highly elliptical orbit. These trajectories are discussed in more detail in Part B6 of this Appendix.

The mission was routine and generally proceeded according to the timeline. Because the crew was ahead of schedule and midcourse correction number 3 was cancelled, an early entry into the lunar module was made at 55:00:00. A scheduled television broadcast to the Earth was made between 55:15 and 55:46, and at the time of the accident, both the Commander and Command Module Pilot were in the command module while the Lunar Module Pilot was just entering the command module from the lunar module.

Spacecraft Systems Operation

This section of the report will deal only with problems and events in the various systems encountered with the CSM during the powered phase, parking orbit, and translunar coast phase of the mission up to the time of the accident. The systems will be treated separately except that electrical current and voltage fluctuations associated with the operation of the fans to stir the supercritical oxygen and hydrogen will be covered under the cryogenic section.

CSM structural-mechanical.- Structural loads during boost phases of the flight were within acceptable limits. Command module structural oscillations of less than 0.1g at 16 hertz in all directions were measured during the period of S-II longitudinal oscillations (POGO) prior to the center engine cutoff. The levels of these oscillations were comparable to those measured during ground test and on previous Apollo missions.

At approximately 00:25:00 minutes, a computer program was entered into the computer to align the inertial measuring unit. During this alignment, the sextant is rotated, which in turn releases the external ablative optics covers. The optics covers are spring loaded, and held in place by clips. When the sextant is rotated, an arm located on the sextant engages a cam that releases the clips and jettisons both covers. Minor difficulty was experienced in jettisoning the two covers. The optics were rotated twice manually to 90 degrees according to the checklist, but the covers did not jettison. The optics were then rotated in the automatic mode (past 90 degrees) and the covers jettisoned. The cause of the covers not jettisoning was that the sextant was not rotated far enough in the manual mode to completely engage the cam.

After CSM/LM docking, the crew reported that two docking latches were not fully engaged. Both latches were opened and reset. There are 12 docking latches on the command module. Each latch has a trigger that is engaged when the lunar module docking ring comes in contact with the CSM docking ring. The handle has a red indicator that indicates when the latch is engaged. On several spacecraft during ground checkout one or two of the latches had to be reset manually, as in the case of Apollo 13. The prime cause is not having the two docking rings perfectly parallel at the time of engagement. The manual resetting of one or two of the latches is considered satisfactory.

The crew reported a slight "burnt" smell in the tunnel area between the CSM/LM when entering the tunnel, which is normal.

Electrical power.- The electrical power distribution and sequential system, except for the fuel cells, operated as expected until the time of the accident. The electrical parameters associated with the fan turnon and turnoff times will be discussed in Part B9.

At about 30:45:00 the fuel cell 3 condenser exit temperature pattern was observed to change to a sinusoidal ripple with a frequency of 1 cycle every 30 seconds and a peak-to-peak amplitude of 6.2° F. The oscillations continued for approximately 9 hours and then stopped. Similar oscillations had been observed on Apollo 10 during lunar orbit, and subsequent analyses and tests showed that the oscillations were not detrimental to the performance or life of the fuel cells. These transients are attributed to slugs of cold water leaving the condenser.

Instrumentation.- Four discrepancies in the instrumentation system were noted. At 46:40:06 the oxygen quantity measurement located in oxygen tank no. 2 indicated 100 percent. This anomaly will be discussed in detail in Part B9. The cabin pressure indicated 1/2 psi above the suit pressure until powerdown of the CSM after the incident. (Should be approximately the same with the crew out of the suits.) During the boost phase, when the cabin vented the transducer did not follow the cabin pressure and operated erratically for the remainder of the flight. This erratic operation was very similar to the erratic operation of the identical transducer on Apollo 12. Failure analysis of the Apollo 12 transducer indicated contamination inside the transducer.

Early in the mission (22:38 and 37:38) the potable water quantity transducer acted erratically for a brief period. This instrument has operated erratically on other spacecraft during ground checkout and flight due to oxidation of electrical winding on the transducer potentiometer. This oxidation causes intermittent contact between the wiper arm and the wiring on the potentiometer, thus giving erratic readings.

At approximately T + 32 hours, the crew reported that the spacecraft panel meters indicating fuel cell hydrogen versus oxygen flow were not exactly matched for fuel cell 3. All indications on the ground were normal. Prelaunch ground data once indicated a mismatch in panel indication on fuel cell 2. Since the instrumentation data in both cases were correct, the most probable cause was an intermittent fault in the meter circuitry causing the shift.

Communications.- At 55:05:32 the crew reported that they could not operate the high-gain antenna (HGA) in narrow beamwidth auto track or reacquisition modes. A maneuver to the passive thermal control (PTC) attitude was prescribed and as the maneuver was initiated, the crew manually positioned the antenna and acquired automatic tracking in the narrow beamwidth mode. The antenna operated normally until the accident. When troubleshooting (before lockup) both the primary and secondary electronics and both the automatic and reacquisition tracking modes were unsuccessfully attempted. Analysis indicates an effective misalignment existed between the boresight of the wide and narrow beams. The beam effective misalignment could have been caused by a defective radio frequency (RF) stripline coaxial cable, mechanical failure, or RF feed lines. A boresight shift was not indicated during antenna acceptance testing or during KSC ground checkout.

Service module propulsion and reaction control.- The service module propulsion system was used only once during the mission at 30:40:50 to place the spacecraft into a non-free-return trajectory. The engine burned for 3.6 seconds, and all parameters were nominal. The thrust chamber pressure seemed about 4 percent below preflight prediction, but within acceptable limits.

Guidance and control.- Guidance and control system performance was satisfactory, with the exception of small fluctuations of the optic shaft when in the zero optics mode and in establishing passive thermal control (PTC). At approximately 7:30:00 the crew reported difficulty in establishing PTC. The attempt resulted in a very wide and diverging coning angle. It was determined that the digital autopilot was incorrectly loaded and all roll thrusters were not enabled. The checklist did not call out the correct autopilot load and the thruster enabling was a late pen-and-ink change to the onboard checklist. Using the revised procedure, the PTC mode was successfuly established.

At about 40:00 the ground controllers noticed small fluctuations of the optic shaft when in zero optics mode. As on Apollo 12, the ground data showed a slight jitter in the optics shaft angle from 0 to 0.6 degree. A special test was conducted at 49 hours to verify the shaft oscillations. The crew compared the shaft and trunnion angles to the mechanical counters on the optics. The oscillation was evident from both sources and occurred in the optics zero mode only. The optics jitter presented no constraint to the operation of the optical system; however, at 49:51:37 the ground requested the crew to turn off optics power to guard against possible degradation of the system.

Environmental control.- No anomalies were noted in the environmental control system operation.

Thermal control.- The thermal control system of the CSM performed normally until the incident.

Cryogenic system.- Both the liquid hydrogen and the liquid oxygen systems operated satisfactorily up to the time of the accident as far as the fuel cells and environmental control systems were concerned. Because of the unbalance in hydrogen quantities during loading, and unequal usage during launch pad operation, several hydrogen low-pressure master alarms were detected on the caution and warning system. (A description of the caution and warning system is contained in Appendix A, Part A2.10.) At 46:40:08 the oxygen tank no. 2 quantity measurement indicated 100 percent quantity and remained at this value until the pressure rise at 55:53:35. With the exception of the above, system operations were normal to the time of the accident.

The following sections will describe the low hydrogen pressure master alarm and supercritical liquid hydrogen and oxygen destratification up to the time of the accident.

Hydrogen Low Pressure Master Alarm

The caution and warning system, upon receipt of a malfunction or out-of-tolerance signal, simultaneously identifies the abnormal condition and alerts the crew to its existence. Each signal (both oxygen and hydrogen pressure are on one indicator) will activate the system status indicator, light the master alarm light, and place an audio tone in the crew's headsets. The crew can turn off or reset the master alarm; however, the particular system status malfunction indicators remain lit, blocking further master alarms on this indication, until the malfunction is cleared.

At lift-off, the quantity readings for the hydrogen tanks no. 1 and no. 2 were 91 percent and 93.4 percent, respectively. This was due to initial loading values (98.7 percent for tank no. 1 and 99.4 percent for tank no. 2) and the difference in usage during countdown.

At approximately 32:00 g.e.t., a quantity unbalance of 2.38 percent existed between the hydrogen tanks, and a quantity balancing procedure was conducted to prevent tank no. 1 low-pressure master alarms during the sleep period. In the "auto" mode the tank heaters are turned on and off by pressure switches connected in series. When the pressure in either tank reaches about 260 psi, the heaters in both tanks are switched off. The heaters remain off until the opened pressure switch closes at approximately 225 psi. Since one tank pressure switch normally remains closed, the tank that controls the upper pressure will also control at the lower pressure. During the flight, tank no. 2 was controlling. Tank no. 1 pressure was almost reaching the caution and warning low pressure point (224.2 psia) prior to tank no. 2 reaching its pressure switch activation point of 233.6 psia to turn on the heaters.

Since tank no. 2 had the greater quantity, at 32:00 the tank no. 1 heaters were manually turned off by the crew while tank no. 2 remained in auto. This condition would allow the fuel cells to obtain hydrogen from tank no. 2 because of its higher pressure and in turn reduce its quantity of hydrogen. Several master alarms occurred immediately after this change (33:10, 33:41, 34:01, and 34:32).

At 36:48 the hydrogen tank no. 1 heater was placed back to auto for the sleep period. On the first "down" pressure cycle a master alarm occurred (38:00) due to hydrogen tank no. 1 pressure dropping lower than 224.24 psia, awaking the crew. The crew reset the alarm, and no master alarms occurred through the sleep period although the heaters cycled several times. To obtain a balanced condition for the next sleep period, the ground controllers devised the following plan for the next day's operation:

1. After crew wakeup, turn hydrogen tank no. 2 heater to off and leave hydrogen tank no. 1 in auto for two to three pressure cycles to determine if this will transfer heater control to tank no. 1 in anticipation of using this configuration for sleep.

2. If successful, tank no. 1 heaters will be turned off during the day and tank no. 2 heaters left in auto to create a quantity unbalance in favor of tank no. 1.

3. During the next sleep period, the tanks will be balanced by placing tank no. 1 heaters in auto and tank no. 2 heaters to off.

This plan was executed when the crew awoke the next day. At the time of the accident, tank no. 1 was in off and tank no. 2 was in auto, and the caution and warning master alarm was reset with a low hydrogen pressure indication present at 55:52:30. This hydrogen low pressure indication locked out the master alarm during the time of the increasing pressure in oxygen tank no. 2.

Cryogenic Tank Destratification

To prevent stratification in the oxygen and hydrogen tanks, two fans are located in each tank. A diagram of the oxygen tank showing the two fans and quantity gaging probe is shown in figure B4-11. The flight plan called for the fans to be operated in both the hydrogen and oxygen tanks at the following times: 3:40, 12:09, 23:12, 29:40, 37:30, and 46:39 g.e.t. The ground controllers requested the oxygen tank no. 2 fans to be operated at 47:54 and both the oxygen and hydrogen fans be operated at 51:07.

Review of cryogenic and electrical instrumentation data does not indicate that the fans were switched on at 3:40 and at 29:40. No changes in cryogenic pressures and quantities, and no indications of an increase in spacecraft current were noted. The operation of the fans during the other destratification periods were normal; however, three oxygen tank no. 2 differences were noted: (1) transients on pitch and yaw thrust vector control gimbal command parameters at fan turnon and turnoff, (2) quantity gaging probe malfunctioned just after or at the time the fans came on, and (3) ac main bus 2 indicated a 1.8-volt negative transient when the fans were turned on at 47:54.

The pitch and yaw thrust vector control gimbal command (TVC command) parameters are an excellent transient detector on ac main bus 2 when the stabilization and control system is turned off because of its sensitivity and high sampling rate (100 samples per second). The sensitivity of the system is determined by the position of the rate high/low switch and the attitude deadband maximum/minimum switch. The TVC command signals are not transmitted to the ground when the instrumentation system is in low bit rate mode.

The system was in the low sensitivity mode during two destratification periods. When oxygen tank no. 2 fans were turned on during tank destratification periods, a negative initial transient was detected and when the fans were turned off, a positive initial transient was detected on the TVC command parameters. These transients are readily detectable in the high sensitivity mode and barely detectable in the low sensitivity mode. Examination of the Apollo 11 records indicates that the system was in the high sensitivity mode once during the fan destratification periods and a similar transient occurred when the fans were turned on. The data indicate that the transients are normal for fan turnon and turnoff, and only indicate a relatively large current change on ac main bus 2.

At 47:40:08 the oxygen quantity changed from approximately 82 percent to 100 percent, or full-scale high. This change in reading or quantity system malfunction occurred just after or at the time the oxygen tank no. 2 fans were turned on. Because of the way the system recovered at the time of the accident, the data indicate that the probe or its associated wiring shorted. Since the instrumentation system was in low bit rate, it is possible to determine exactly when the oxygen

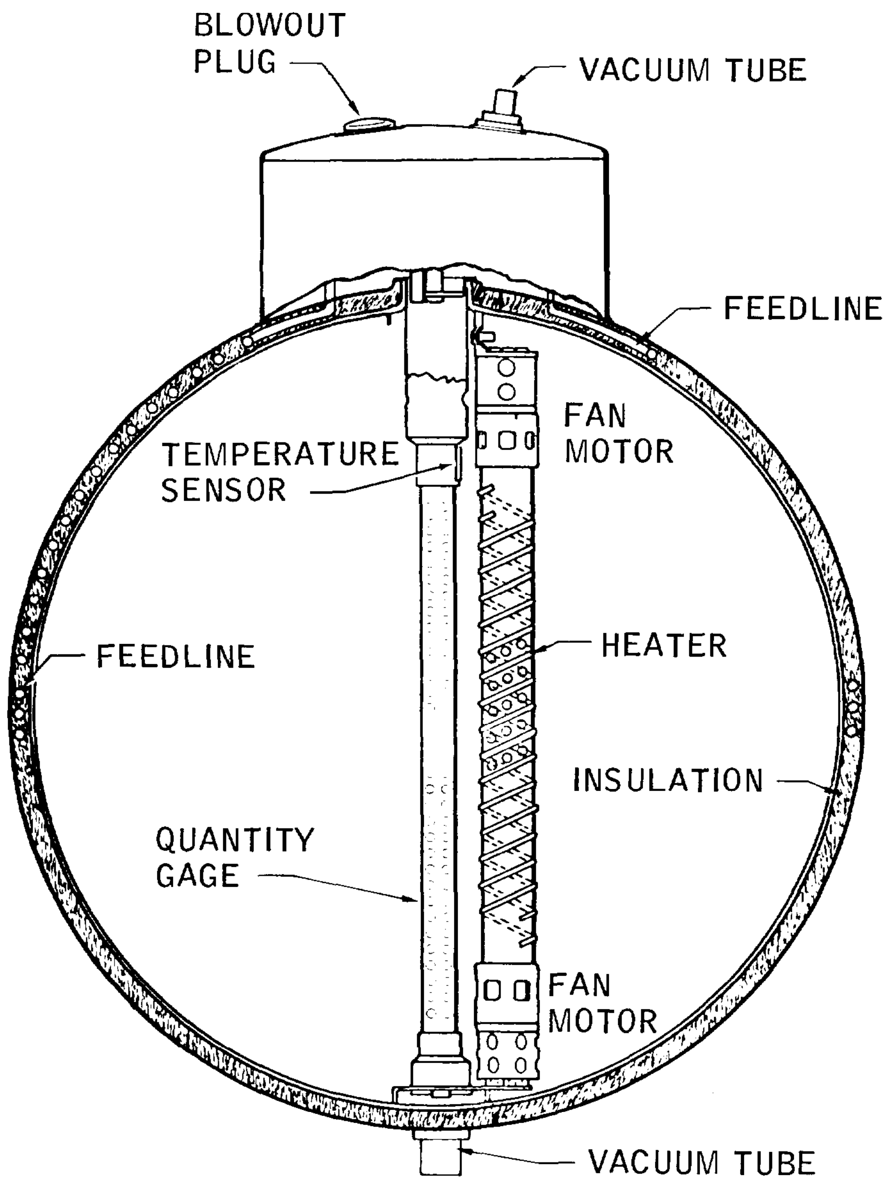

Figure B4-11.- Arrangement of components within oxygen tank.

tank no. 1 and no. 2 fans were turned on. The electrical data indicate that the oxygen tank no. 1 and no. 2 fans were turned on between the times of 47:40:05 and 47:40:08. A plot of cryogenic pressures, quantities, total CSM current, and ac main bus 2 is shown in figure B4-12. Therefore, the oxygen tank no. 1 and no. 2 fans were turned on in a period of time between 3 seconds prior to the probe malfunction and the time that the probe malfunctioned.

When the oxygen tank no. 2 fans were turned on the next time at 47:54:50, the ac main bus 2 decreased 1.8 volts for one sample (0.1 second). At the same time the TVC command parameters indicated a negative initial transient. Because of the sampling rate (10 samples per second) of the instrumentation system and the small number of fan cycles examined in the high bit rate mode, it cannot be determined if this negative initial transient is characteristic of other fan turnon's or is an indication of a deteriorating fan or wiring.

The complete oxygen and hydrogen tank destratification history prior to the accident is shown in table B4-I. Changes in oxygen and hydrogen pressures and quantities indicate normal destratification of the tanks during all fan cycles. The next destratification period occurred at 55:53:18, or when the events started leading to the accident.

References 1 through 6 and instrumentation records were used as a source of information and data in the preparation of this part of the report.

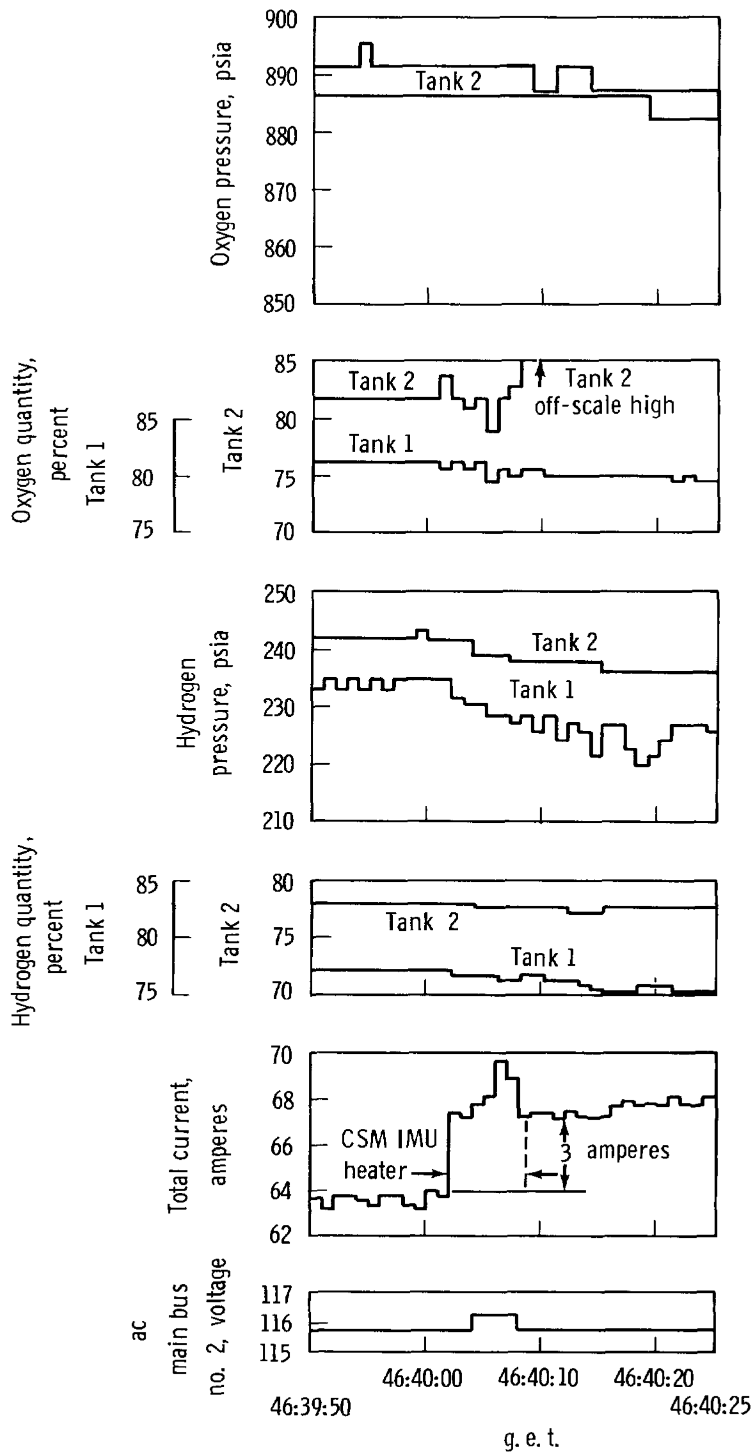

Figure B4-12.- Fan cycle at 46:40.

TABLE B4-I.- CRYOGENIC TANK DESTRATIFICATION

Planned destratification	Actual start of destratification	Verified air-to-ground	SCS transients	Remarks
3:40:00	No indications	Not verified	No indications	No indications
12:09:00 Presleep checklist	12:08:58 to 12:09:04	Not verified	12:09:03.6	Normal destratification
23:12:00 Postsleep checklist	23:10:53 to 23:11:01.5	Verified	23:11:01	Normal destratification
29:40:00	No indications	Not verified	Low bit rate	No indications
37:30:00 Presleep checklist	36:46:19 to 36:46:21	Partial verification	36:46:20.5	Normal destratification
46:39:00 Postsleep checklist	46:40:02 to 46:40:05-08	Verified	Low bit rate	Normal destratification quantity gaging system malfunctioned
Ground control request (O_2 tank no. 2 only)	47:54:50	Verified	47:54:50	Normal destratification - fans ran for approximately 11.5 minutes
Ground control request	51:07:43 to 51:07:47	Verified	Low bit rate	Normal destratification

PART B5

INCIDENT EVENTS

INTRODUCTION

This part of the report covers the significant events which took place at the time of the accident. The period covered is 55:52:00 g.e.t. to 56:00:05 g.e.t. Prior to this period, spacecraft operation had been essentially according to plan and neither the ground controllers nor the crew had any warning of the events about to occur. The first indication of a problem was a loud bang heard by all three crew members which was followed by a master caution and warning. The immediate indications in the spacecraft were that this warning had been triggered by an electrical transient. Several minutes later two fuel cells failed in the power system, and the crew became fully occupied trying to reconfigure the spacecraft electrical system. Fourteen minutes later they noticed venting and began to understand what had actually happened in the cryogenic oxygen system.

On the ground, the flight controllers first noticed that the spacecraft computer had been automatically restarted. Shortly afterwards, indication of a master caution and warning caused the flight controllers to scan their data for a problem. Since many telemetry measurements had by this time departed from their nominal values, the ground controllers' immediate reaction was to suspect an instrumentation failure. Steps were undertaken to sort the false telemetry readings from the true ones; and, simultaneously, instructions were given to help the crew handle new problems. About an hour later the ground personnel had sorted out the facts sufficiently to know that it would only be a short time before the cryogenic oxygen system would fail completely.

Reconstruction of the mission events in the detail presented in the following pages has required several hundred man-days of data analysis. Consequently, the crew and mission controllers could not possibly have understood the situation in the same depth at the time the events were actually happening. The primary sources of data for the analysis have been telemetry records, transcripts of voice communications, crew debriefings, and interviews with personnel on duty in Mission Control.

Table B5-I is a detailed chronology of the events during this time period, and figure B5-1 shows the sequence of events grouped according to spacecraft systems. For events obtained from telemetry data, where time is shown to a fraction of a second, this refers to the time at which the parameter in question was actually sampled by the telemetry system. As discussed in Part B7 of this Appendix, the characteristics of the telemetry system place an uncertainty on the time of an event. The uncertainty is a function of the telemetry system sampling rate.

The remainder of this section is a discussion of the events at the time of the accident, grouped according to the spacecraft systems involved.

TABLE B5-I.- DETAILED CHRONOLOGY FROM
2.5 MINUTES BEFORE THE ACCIDENT TO 5 MINUTES AFTER THE ACCIDENT

Time, g.e.t.	Event
	Events During 52 Seconds Prior to First Observed Abnormality
55:52:31	Master caution and warning triggered by low hydrogen pressure in tank no. 1. Alarm is turned off after 4 seconds.
55:52:58	Ground requests tank stir.
55:53:06	Crew acknowledges tank stir.
55:53:18	Oxygen tank no. 1 fans on.
55:53:19	Oxygen tank no. 1 pressure decreases 8 psi.
55:53:20	Oxygen tank no. 2 fans turned on.
55:53:20	Stabilization control system electrical disturbance indicates a power transient.
55:53:21	Oxygen tank no. 2 pressure decreases 4 psi.
	Abnormal Events During 90 Seconds Preceding the Accident
55:53:22.718	Stabilization control system electrical disturbance indicates a power transient.
55:53:22.757	1.2-volt decrease in ac bus 2 voltage.
55:53:22.772	11.1-amp rise in fuel cell 3 current for one sample.
55:53:36	Oxygen tank no. 2 pressure begins rise lasting for 24 seconds.
55:53:38.057	11-volt decrease in ac bus 2 voltage for one sample.
55:53:38.085	Stabilization control system electrical disturbance indicates a power transient.

TABLE B5-I.- DETAILED CHRONOLOGY FROM
2.5 MINUTES BEFORE THE ACCIDENT TO 5 MINUTES AFTER THE ACCIDENT - Continued

Time, g.e.t.	Event
55:53:41.172	22.9-amp rise in fuel cell 3 current for one sample.
55:53:41.192	Stabilization control system electrical disturbance indicates a power transient.
55:54:00	Oxygen tank no. 2 pressure rise ends at a pressure of 953.8 psia.
55:54:15	Oxygen tank no. 2 pressure begins to rise.
55:54:30	Oxygen tank no. 2 quantity drops from full scale for 2 seconds and then reads 75.3 percent.
55:54:31	Oxygen tank no. 2 temperature begins to rise rapidly.
55:54:43	Flow rate of oxygen to all three fuel cells begins to decrease.
55:54:45	Oxygen tank no. 2 pressure reaches maximum value of 1008.3 psia.
55:54:48	Oxygen tank no. 2 temperature rises 40° F for one sample.
55:54:51	Oxygen tank no. 2 quantity jumps to off-scale high and then begins to drop until the time of telemetry loss.
55:54:52	Oxygen tank no. 2 temperature reads -151.3° F.
55:54:52.703	Oxygen tank no. 2 temperature suddenly goes off-scale low.
55:54:52.763	Last telemetered pressure from oxygen tank no. 2 before telemetry loss is 995.7 psia.
55:54:53.182	Sudden accelerometer activity on X, Y, and Z axes.
55:54:53.220	Stabilization control system body rate changes begin.

TABLE B5-I.- DETAILED CHRONOLOGY FROM
2.5 MINUTES BEFORE THE ACCIDENT TO 5 MINUTES AFTER THE ACCIDENT - Continued

Time, g.e.t.	Event
55:54:53.323	Oxygen tank no. 1 pressure drops 4.2 psi.
55:54:53.5	2.8-amp rise in total fuel cell current.
55:54:53.542	X, Y, and Z accelerations in CM indicate 1.17g, 0.65g and 0.65g, respectively.
	1.8-Second Data Loss
55:54:53.555	Loss of telemetry begins.
55:54:53.555+	Master caution and warning triggered by dc main bus B undervoltage. Alarm is turned off in 6 seconds. All indications are that the cryogenic oxygen tank no. 2 lost pressure in this time period and the panel separated.
55:54:54.741	Nitrogen pressure in fuel cell 1 is off-scale low indicating failed sensor.
55:54:55.35	Recovery of telemetry data.
	Events During 5 Minutes Following the Accident
55:54:56	Service propulsion system engine valve body temperature begins a rise of 1.65° F in 7 seconds.
55:54:56	Dc main bus A decreases 0.9 volt to 28.5 volts and dc main bus B decreases 0.9 volt to 29.0 volts.
55:54:56	Total fuel cell current is 15 amps higher than the final value before telemetry loss. High current continues for 19 seconds.
55:54:56	Oxygen tank no. 2 temperature reads off-scale high after telemetry recovery.
55:54:56	Oxygen tank no. 2 pressure reads off-scale low following telemetry recovery.

TABLE B5-I.- DETAILED CHRONOLOGY FROM
2.5 MINUTES BEFORE THE ACCIDENT TO 5 MINUTES AFTER THE ACCIDENT - Continued

Time, g.e.t.	Event
55:54:56	Oxygen tank no. 1 pressure reads 781.9 psia and begins to drop steadily.
55:54:57	Oxygen tank no. 2 quantity reads off-scale high following telemetry recovery.
55:54:59	The reaction control system helium tank C temperature begins a 1.66° F increase in 36 seconds.
55:55:01	Oxygen flow rates to fuel cells 1 and 3 level off after steadily decreasing.
55:55:02	The surface temperature of the service module oxidizer tank in bay 3 begins a 3.8° F increase in a 15-second period.
55:55:02	The service propulsion system helium tank temperature begins a 3.8° F increase in a 32-second period.
55:55:09	Dc main bus A voltage recovers to 29.0 volts; dc main bus B recovers to 28.8 volts.
55:55:20	Crew reports, "I believe we've had a problem here."
55:55:35	Crew reports, "We've had a main B bus undervolt."
55:55:49	Oxygen tank no. 2 temperature begins steady drop lasting 59 seconds.
55:56:10	Crew reports, "Okay right now, Houston. The voltage is looking good, and we had a pretty large bang associated with the caution and warning there. And as I recall, main B was the one that had had an amp spike on it once before."
55:56:38	Oxygen tank no. 2 quantity becomes erratic for 69 seconds before assuming an off-scale-low state.

TABLE B5-I.- DETAILED CHRONOLOGY FROM
2.5 MINUTES BEFORE THE ACCIDENT TO 5 MINUTES AFTER THE ACCIDENT - Concluded

Time, g.e.t.	Event
55:57:04	Crew reports, "That jolt must have rocked the sensor on--see now--oxygen quantity 2. It was oscillating down around 20 to 60 percent. Now it's full-scale high again."
55:57:39	Master caution and warning triggered by dc main bus B undervoltage. Alarm is turned off in 6 seconds.
55:57:40	Dc main bus B drops below 26.25 volts and continues to fall rapidly.
55:57:44	Ac bus 2 fails within 2 seconds.
55:57:45	Fuel cell 3 fails.
55:57:59	Fuel cell 1 current begins to decrease.
55:58:02	Master caution and warning caused by ac bus 2 being reset. Alarm is turned off after 2 seconds.
55:58:06	Master caution and warning triggered by dc main bus A undervoltage. Alarm is turned off in 13 seconds.
55:58:07	Dc main bus A drops below 26.25 volts and in the next few seconds levels off at 25.5 volts.
55:58:07	Crew reports, "ac 2 is showing zip."
55:58:25	Crew reports, "Yes, we got a main bus A undervolt now, too, showing. It's reading about 25-1/2. Main B is reading zip right now."
56:00:06	Master caution and warning triggered by high hydrogen flow rate to fuel cell 2. Alarm is turned off in 2 seconds.

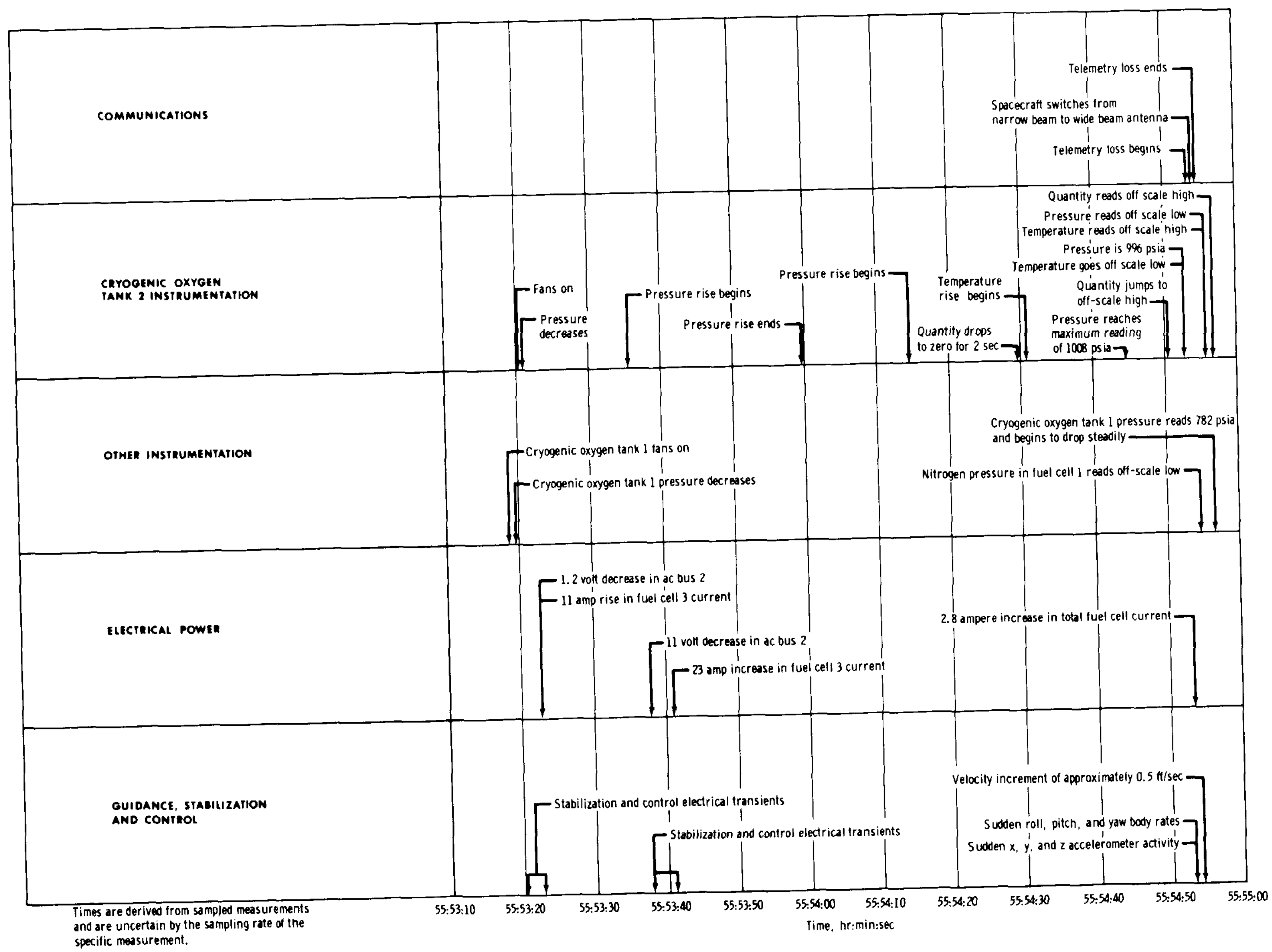

Figure B5-1.- Sequence of events immediately preceding the accident.

STATUS OF THE SPACECRAFT PRIOR TO THE ACCIDENT

At 55:52:00, just prior to the accident, the electrical system was configured as shown in figure B5-2. Fuel cells 1 and 2 were supplying main bus A; fuel cell 3 was supplying main bus B. The power for the fans in cryogenic oxygen tank no. 2 was being supplied by ac bus 2, as was power for the quantity sensor in that tank. The stabilization control system thruster vector control system was receiving its power from ac bus 2. Two quantities in this system, the pitch and yaw thrust vector control gimbal commands, though not intended for measurement of electrical system currents and voltages, are sensitive indicators of electrical transients on ac bus 2. These quantities are telemetered to the ground with a sampling rate of 100 samples per second. At 55:52:00 the telemetry system was operating in the high-bit-rate mode and the narrow beam antenna was in use.

The cryogenic oxygen tank no. 2 quantity gage had failed to a 100-percent reading in the 46th hour of the flight. (See Part B4, the subsection entitled "Spacecraft Systems Operation.") All other cryogenic oxygen instrumentation was operating normally.

The cryogenic hydrogen tank 1 pressure decreased sufficiently to trigger the master caution and warning at 55:52:31. (For a description of the master caution and warning system, see Part 2.10 of Appendix A.) The ground then requested a fan cycle, and the crew acknowledged the request. A fan cycle consists of the crew turning on the stirring fans located in both the cryogenic oxygen and hydrogen tanks and allowing them to run for approximately 1 minute. Normally, the hydrogen fans are turned on first, followed by the fans in oxygen tank no. 1 and a few seconds later by the fans in oxygen tank no. 2.

FAN TURNON AND ASSOCIATED ELECTRICAL ANOMALIES

At 55:53:18 when the two fans in cryogenic oxygen tank no. 1 were turned on by the crew, a drop in ac bus 1 voltage (fig. B5-3) and an increase of 1 ampere in total command module current indicated that the fans had been electrically energized. (Total command module current, plotted in figure B5-4, is obtained by adding the current outputs of all three fuel cells and subtracting the current drain of the lunar module.) A subsequent decrease in tank pressure and oscillations in the fuel cell flowmeters indicated that the fans had begun to stir the oxygen (fig. B5-3).

At 55:53:20 the crew turned on the cryogenic oxygen tank no. 2 fans. An increase in fuel cell current of 1-1/2 amperes, a drop in ac bus 2

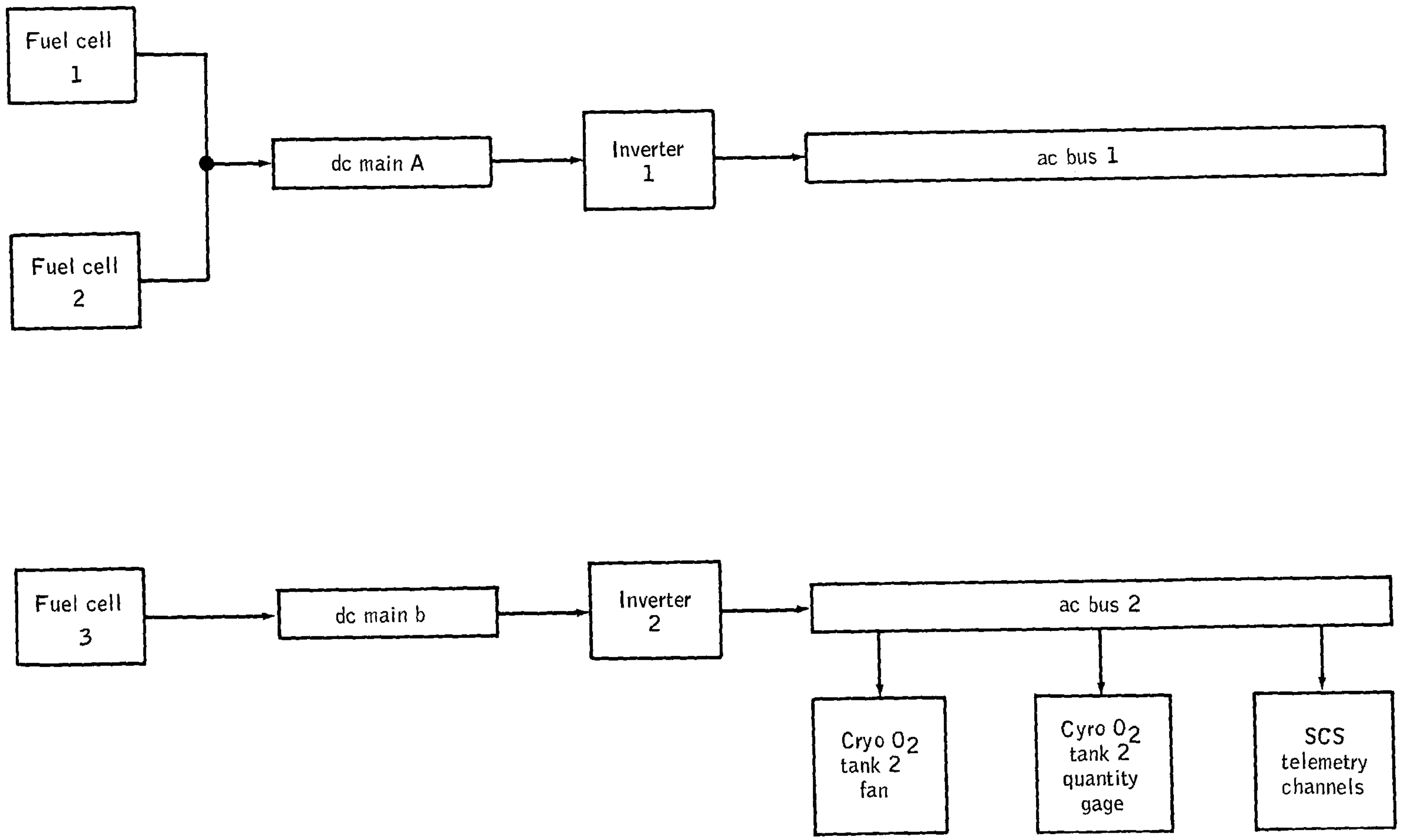

Figure B5-2.- Configuration of electrical power system at 55:53:20.

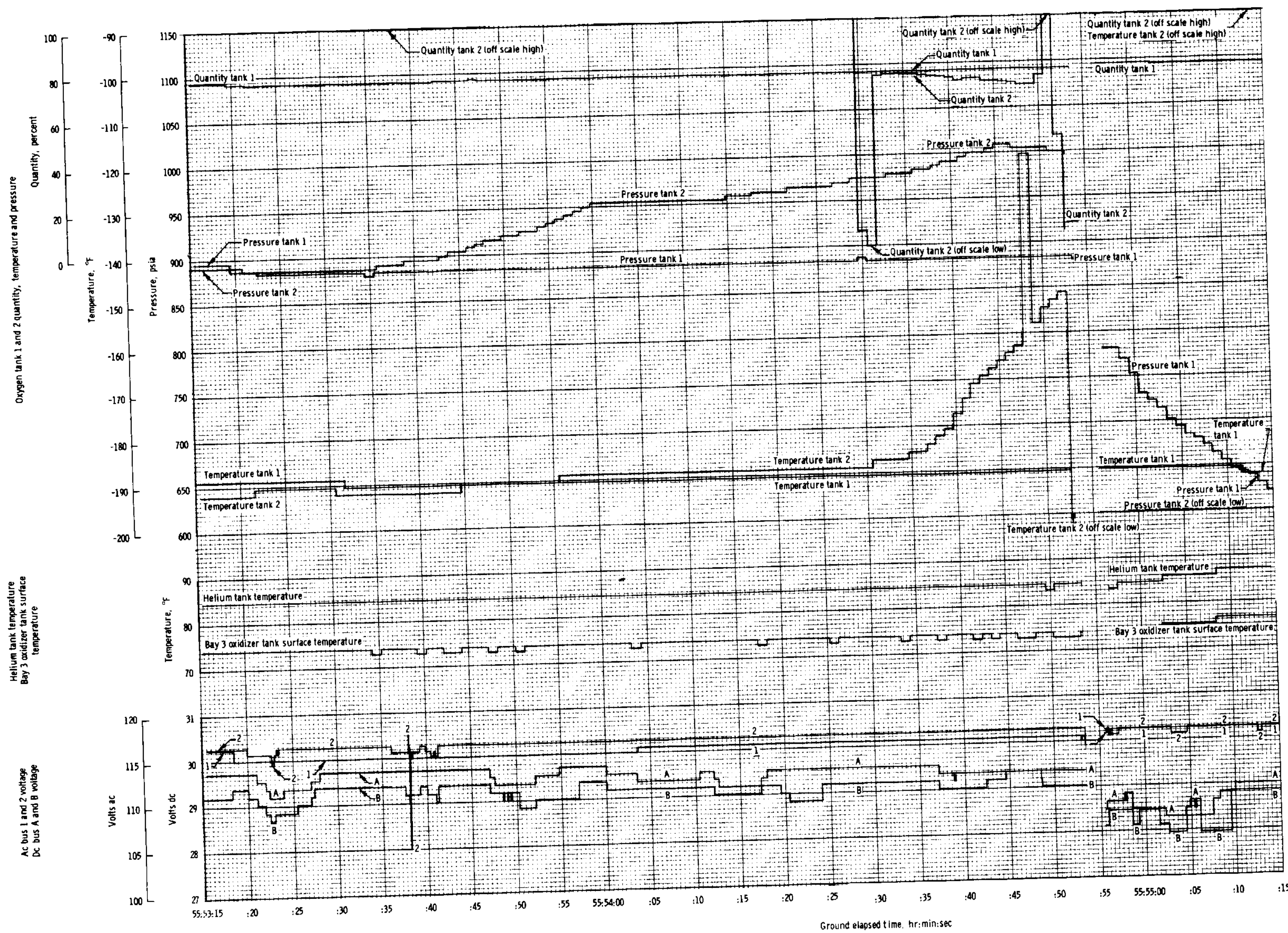

Figure B5-3(a).- Telemetered parameters during anomaly.

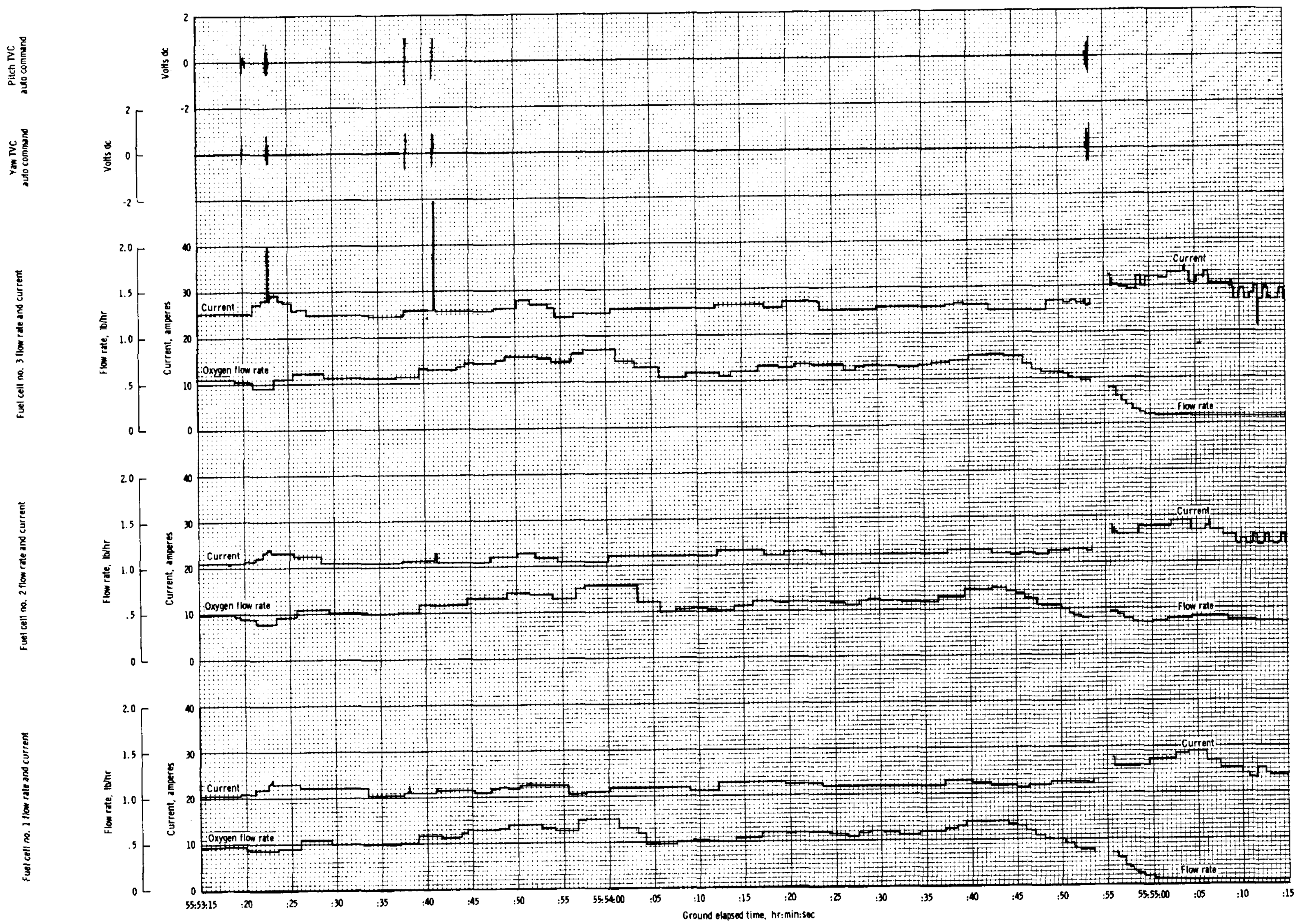

Figure B5-3(b).- Telemetered parameters, concluded.

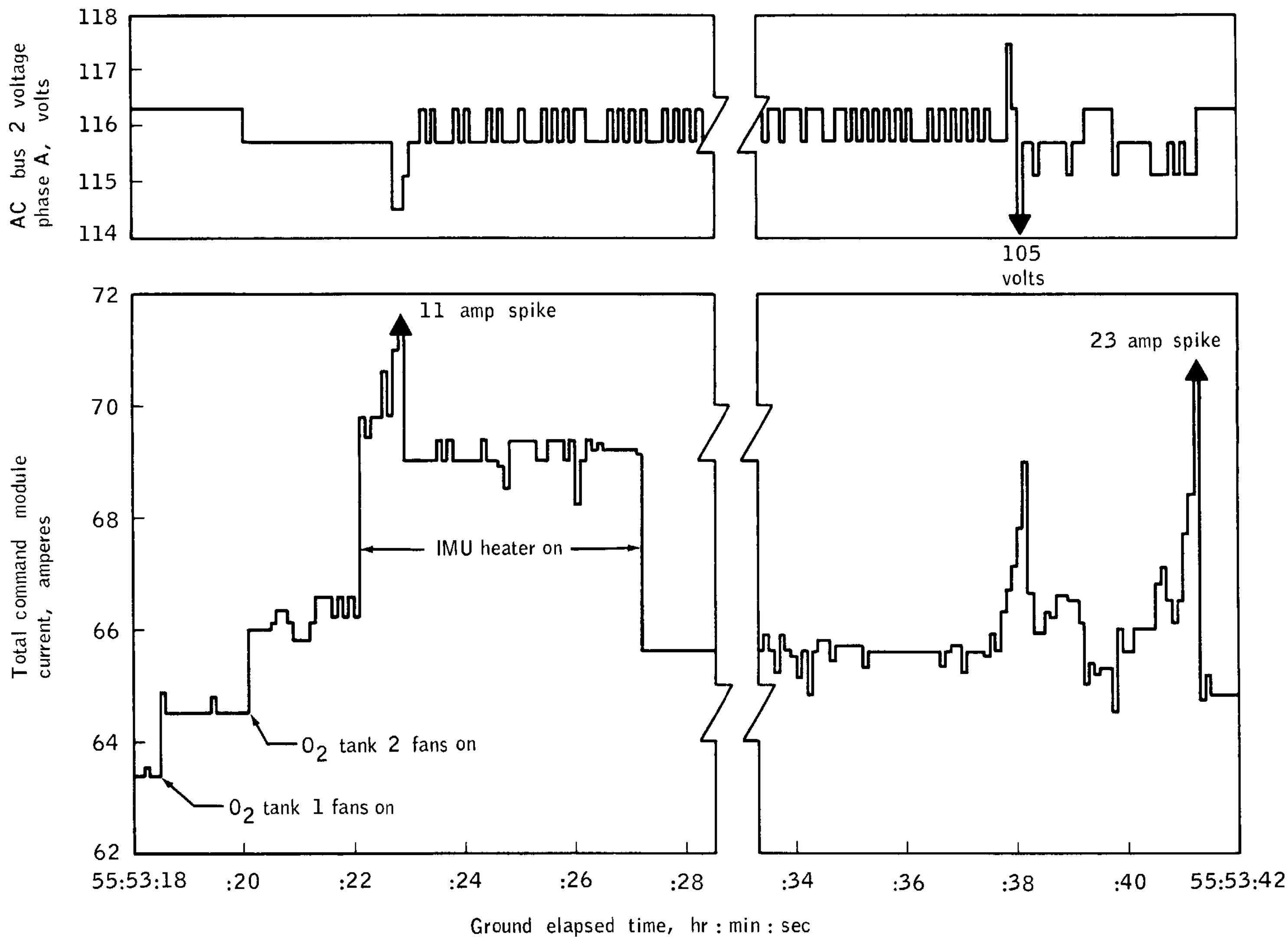

Figure B5-4.- Electrical anomalies.

voltage of 0.6 volt, and a glitch in the stabilization control system telemetry indicated that the fans had been electrically energized. These events are shown in figures B5-4 and B5-5. However, it is not certain that the fans began rotating at this time, since the tank no. 2 pressure showed a minimum observable drop and the fan motor stall current does not significantly differ from the running current. The quantity gage in tank no. 2 was already in a failed condition, and the fuel cell flowmeters were already being affected by the fan operation in tank no. 1 so that neither of these instruments could positively verify rotation of the fans in tank no. 2. During the next 20 seconds a series of electrical anomalies occurred which cannot be explained as a result of known loads in the spacecraft. These anomalies are shown in figure B5-4. The first, at 55:53:23 was an 11-amp positive spike in the output current of fuel cell 3. Several events were associated with this spike:

(a) The command module current decreased approximately 1/2 ampere immediately afterward.

(b) The ac bus 2 voltage had a transient decrease and then began to alternate between 115.7 and 116.3 volts, whereas it had been maintaining a steady value of 115.7 volts since fan turnon.

These events indicate that at the time of the 11-amp spike, a load may have been disconnected from ac bus 2. This could have been one of the fan motors in cryogenic oxygen tank no. 2.

At 55:53:38 another abnormal electrical disturbance occurred, a 3-amp spike of current and variations in ac bus 2 voltage. The ac bus 2 voltage first increased 2 volts and then dropped suddenly from 116 to 105 volts. The ac bus 2 is a three-phase electrical system, although the only voltage telemetered is phase A. The operation of the inverter which generates ac bus 2 is such that it attempts to maintain a constant average voltage among the three phases; if one phase becomes heavily loaded, the inverter will increase the voltages of the other two phases. Consequently, it is possible that the voltage rise in ac bus 2 at 55:53:37.8 was caused by a heavy load applied to phase B or phase C. The decrease in voltage immediately afterward was probably caused by loading of phase A.

At 55:53:41 a 23-amp spike occurred on fuel cell 3 output current, after which the total command module current returned to a steady value within 0.3 ampere of the value prior to turnon of cryogenic oxygen tank no. 2 fans. Also, the voltage of ac bus 2 returned to the value it had shown prior to fan turnon. At the same time transients appeared in the stabilization control system, as shown in figure B5-5.

The most probable cause of the electrical disturbances between 55:53:22 and 55:53:42 is that a short circuit occurred in the electrical system of the cryogenic oxygen tank no. 2 fans. The short circuit was sufficiently severe to result in loss of part of the fan load at 55:53:22 and the remainder at 55:53:41. Reduction of the load could have been caused by fuses blowing or by wires opening.

It should be noted that the nature of the telemetry records makes it difficult to define the exact parameters of the electrical disturbances. Since the value of fuel cell 3 current is sampled by the telemetry system at 10 times per second, the duration of the observed current spikes is in question by 0.2 second. Also, the peak values of the spikes may well have exceeded the maximum recorded values. For similar reasons a large current spike could possibly have occurred at 55:53:38 simultaneous with the 11-volt decrease of ac bus 2. If the spike were very short, less than 0.1 second duration, it could have occurred between the times of successive telemetry samples and thus not have been recorded.

The electrical anomalies ended by 55:53:42 and no further electrical disturbances were observed for the next minute.

OXYGEN TANK PARAMETERS FROM 55:53:30 UNTIL LOSS OF TELEMETRY

Thirteen seconds after the 11-amp spike and 6 seconds before the 23-amp spike, the pressure in cryogenic oxygen tank no. 2 began a steady increase at an abnormally rapid rate. The increase began at 55:53:35 and lasted 19 seconds before the pressure reached a plateau of 954 psia for 21 seconds. At 55:54:15 the pressure rise resumed, reaching a maximum value of 1008 psia 9 seconds before loss of telemetry. During this rise the master caution and warning trip level of 975 psia was exceeded, but a master alarm was not generated because of the existing cryogenic pressure warning occasioned by low hydrogen pressure. After reaching 1008 psia, the pressure decreased to 996 psia just before loss of telemetry. The oxygen flow rate for all three fuel cells declined for about 10 seconds and then began to rise just before loss of telemetry.

The pressure transducer for cryogenic oxygen tank no. 2 is not located in the tank but is connected to the tank along with a pressure relief valve through 19 feet of tubing. The relief valve is set to open fully at 1008 psia. (See figure B7-4 for a diagram of this portion of the cryogenic oxygen system and Part B7 of this Appendix for a more complete description of the cryogenic oxygen pressure sensing system). The remote location of the oxygen pressure transducer causes some time lag in the telemetered pressure data but unless there are unknown restrictions, such as clogging of the filter at the tank end of the line, this lag will not cause serious errors in the pressure reading under the conditions observed.

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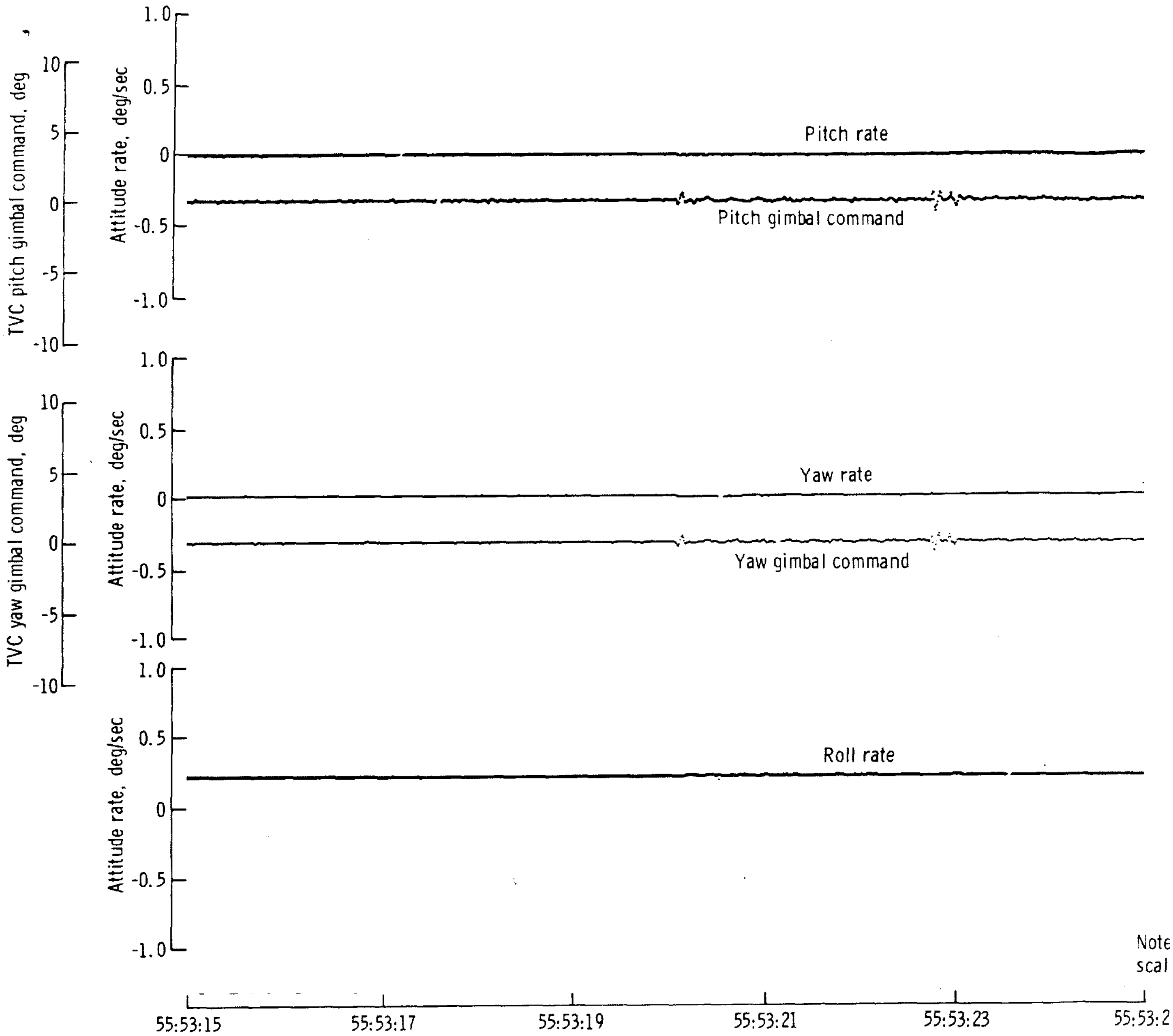
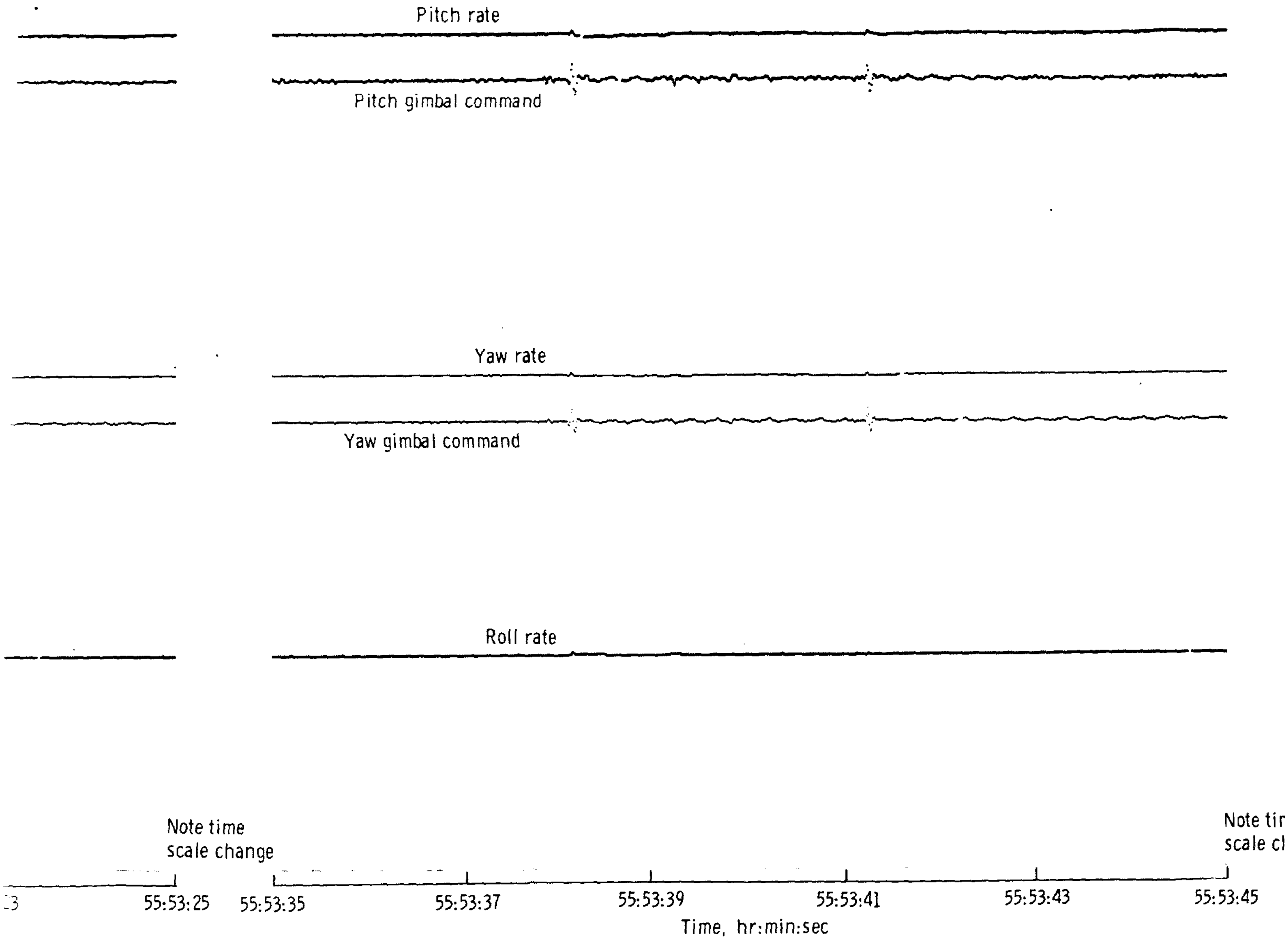


TVC pitch gimbal command, deg
10
5
0
-5
-10
Attitude rate, deg/sec
1.0
0.5
0
-0.5
-1.0
Pitch rate
Pitch gimbal command
TVC yaw gimbal command, deg
Yaw rate
Yaw gimbal command
Roll rate
Note
scal
55:53:15
55:53:17
55:53:19
55:53:21
55:53:23
55:53:2



Pitch rate
Pitch gimbal command
Yaw rate
Yaw gimbal command
Roll rate
Note time
scale change
Note tir
scale cl
55:53:25
55:53:35
55:53:37
55:53:39
55:53:41
55:53:43
55:53:45
Time, hr:min:sec

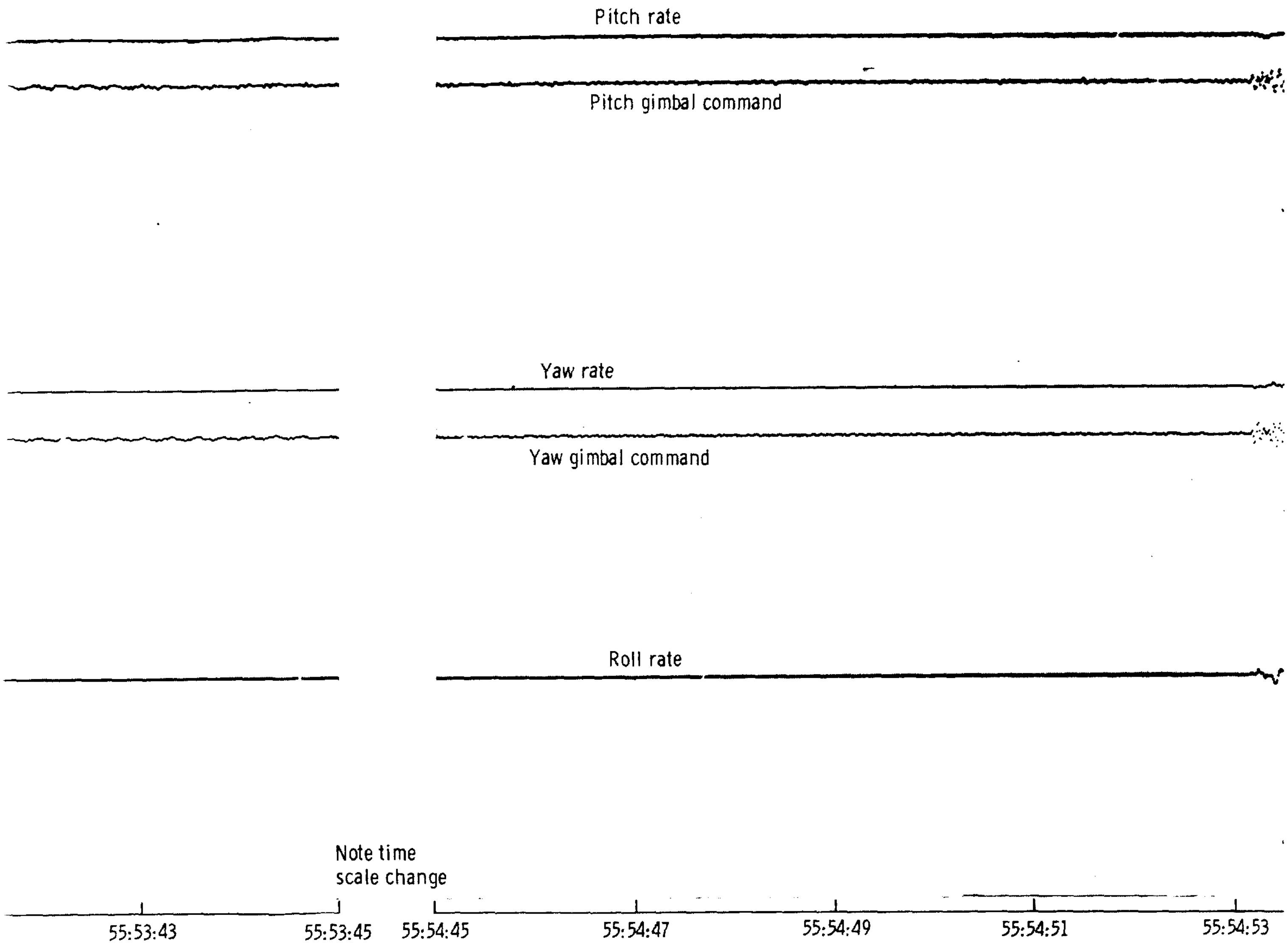


Figure B5-5.- Disturbances observed on thrust vector control gimbal command signals.

The quantity gage for oxygen tank no. 2, which had been in a failed state ever since the 46th hour, suddenly dropped to 6.6 percent and then to off-scale low at 55:54:30. These readings do not correlate with other telemetered data and are, consequently, thought to be erroneous. The gage then jumped to a 75-percent reading, which may be reliable data since it is about the value to be expected. Afterwards the quantity decreased gradually for 19 seconds until 3 seconds before telemetry loss, at which time an erratic gage output occurred. The behavior of this type of gage when a short across the capacitor probe is removed is to drop to zero for several seconds and then return to a correct reading. However, the gage has other failure modes which result in a wandering false indication. See Part B7 of this Appendix for a discussion of the quantity gage. Because of the gage's erratic behavior, it cannot be stated with complete confidence that the 75-percent reading obtained at 55:54:32 is reliable.

The temperature in cryogenic oxygen tank no. 2 remained at -190° F $\pm 2^\circ$ until 55:54:31 when a steady rise in temperature commenced. At 55:54:48 a single data sample indicated a reading 40 degrees higher than the adjacent readings. The last data sample before loss of telemetry was off-scale low, probably indicating a short circuit in the gage or wiring. As discussed in Part B7 of this Appendix, the time constant of the temperature sensor is in the order of at least tens of seconds, which means that the 40-degree jump in reading at 55:54:48 and the final off-scale reading were both due to sensor failure or telemetry system errors. Also, because of the slow gage response, the indicated rate at which the temperature rose between 55:54:31 and 55:54:52 could have been caused by an actual temperature rise of greater magnitude.

The temperature and quantity of cryogenic oxygen tank no. 1 remained steady until telemetry loss. The pressure remained nominal until 0.2 second prior to telemetry loss, when a slight drop was observed.

LOSS OF TELEMETRY

At the time of the accident the spacecraft telemetry signal was being received on both a 210-ft-diameter and an 85-ft-diameter antenna at the Deep Space Instrumentation Facility in Goldstone, California. The carrier level on the Goldstone 85-ft antenna was -100 dBm. At 55:54:53 the signal strength dropped abruptly below -160 dBm, the lower limit of the signal strength recorder, and began an erratic increase. Figure B5-6 is a plot of the carrier strength received at the 85-ft antenna, corrected by 8 dB to show the carrier strength received at the 210-ft antenna. The 210-ft antenna was not equipped with a signal strength recorder.

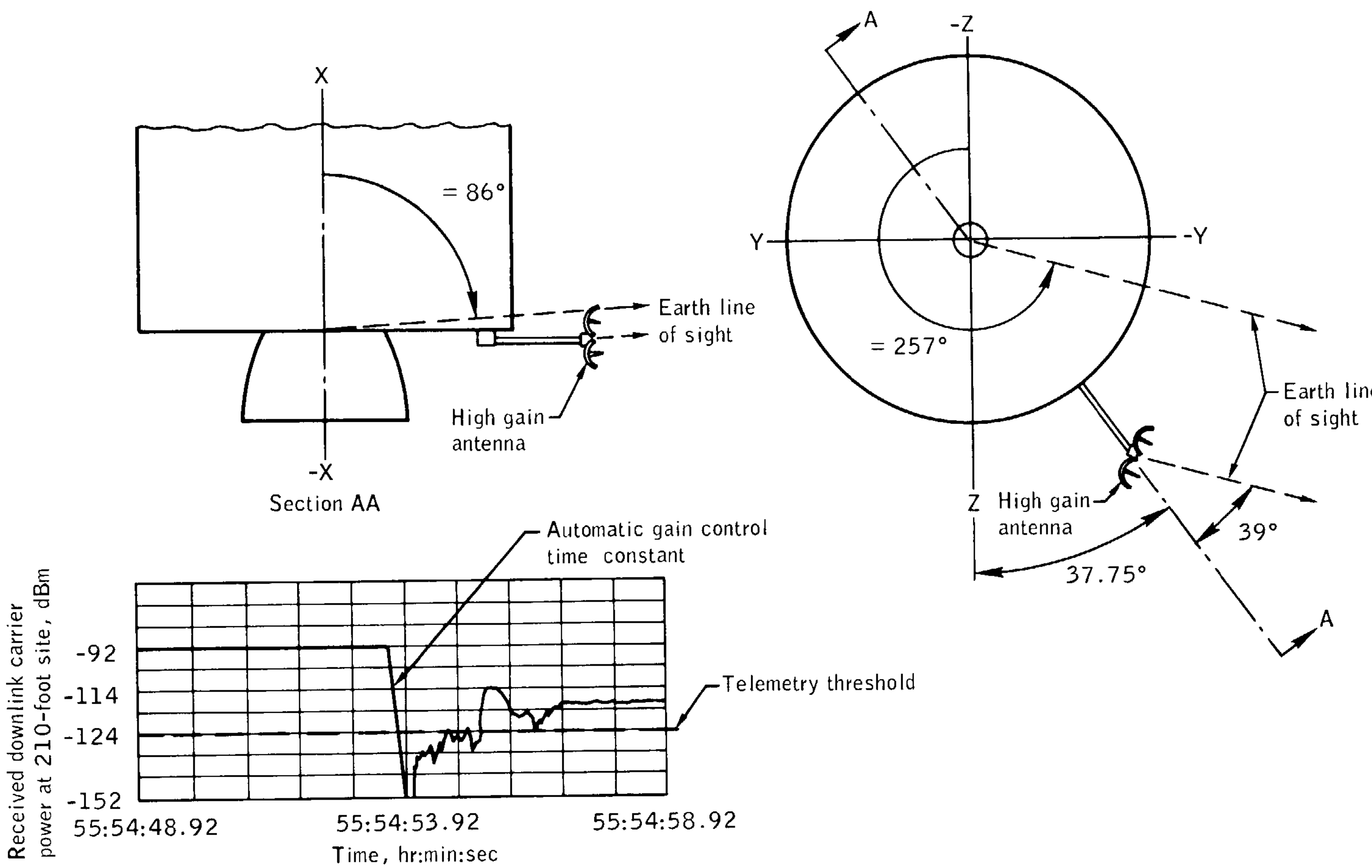


Figure B5-6.- Received S-band downlink power.

Telemetry data recovered completely 1.8 seconds after the loss of carrier power. Sporadic telemetry data are available within the 1.8-second period.

The recorded input signal to the PCM bit detector provides an indication of the rapidity with which the telemetry signal was lost. There appears to be some degradation in signal-to-noise ratio in the time period from 55:54:53.51 to 55:54:53.555. This may have been the result of attitude changes of the spacecraft causing mispointing of the high-gain antenna. At 55:54:53:555 an abrupt change in the character of the signal occurred, and the signal-to-noise ratio rapidly decreased in a period of 1 millisecond. The limitations in the available records make it impossible to definitely determine the speed with which the loss occurred, but an estimate is 1 millisecond.

Although figure B5-6 indicates that the signal required 0.3 second to decrease 60 dB, the actual time was probably much shorter. The decrease of 60 dB in 0.3 second is the same as that obtained when the input signal is abruptly removed from the receiver. This slow response is caused by long time constant circuitry in the automatic gain control.

When the telemetry signal was reacquired, the spacecraft had switched from the narrow-beam antenna to the wide-beam antenna. This has been verified by signal-strength calculations and comparisons of antenna patterns with spacecraft attitude. The spacecraft is designed to automatically switch to the wide-beam antenna if the pointing error of the narrow-beam antenna exceeds 3 degrees.

If a power supply interruption larger than 0.4 second occurs in the communication system, the system design is such that the power output will automatically drop 19 dB for a 90-second period. This power reduction cannot be observed in the received signal strength after recovery of telemetry.

SPACECRAFT EVENTS AT THE TIME OF TELEMETRY LOSS

A large number of spacecraft events took place approximately at the time of telemetry loss. These events are discussed in detail in the following sections as they relate to the various spacecraft systems. This section describes the events as an aid in understanding their interrelationship.

Within the last second prior to telemetry loss, several indications of spacecraft motion appeared on the telemetry records of body accelerometers, and roll, pitch, and yaw rate. The total fuel cell current increased by 3 amperes at the last data sample.

When telemetry data were restored at 55:54:55.35, a large number of channels associated with the electrical system, stabilization control system, and cryogenic system showed marked changes (fig. B5-3). Both dc main A and main B had dropped 0.9 volt and the master caution and warning had been triggered because of an undervoltage on main bus B. The undervoltage triggering level is 26.25 volts and the initial voltage on main B registered 28.1 volts. All three fuel cell currents had increased by 5 amperes over the values before telemetry loss. Both ac bus voltages had maintained their previous values. All telemetry readings from cryogenic oxygen tank no. 2 showed off-scale readings. The temperature was off-scale in the high temperature direction, the quantity gage read 100 percent, and the pressure gage read off-scale low. The capability of the gage is to read pressures as low as 19 psia. Cryogenic oxygen tank no. 1 had not changed temperature or quantity. However, the pressure had decreased from 879 psia to 782 psia. The regulated nitrogen pressure in fuel cell 1 dropped to zero during telemtry loss and remained at zero. The continued operation of this fuel cell indicates a sensor malfunction. As shown in figure B5-7, the wires from the nitrogen pressure sensor to the telemetry system pass along the front of the shelf which supports the fuel cells, in close proximity to the panel covering bay 4. It is quite possible that damage to these wires caused the change observed in the nitrogen pressure reading.

Approximately at the time of telemetry loss all three crew members heard a single loud bang. One or two seconds later they noted the master caution and warning caused by main bus B undervoltage and at 55:55:00 turned off the alarm. They also verified that fuel cell currents were normal at this time. Figure B5-8 is a photograph of the command module control panel showing the type of displays provided the crew. At 55:55:20 the crew reported, "I believe we've had a problem here," and at 55:55:35, "We've had a main B bus undervolt." Later they reported that a computer restart had occurred at the time of the bang, which had already been noted in Mission Control.

Photographs later taken by the crew show the panel covering bay 4, the bay containing the cryogenic oxygen tanks, cryogenic hydrogen tanks, and fuel cells, to be missing. One of these photographs is reproduced in figure B5-9 and a photograph prior to launch is shown in figure B5-10. The high-gain antenna located adjacent to bay 4 shows a misalignment of one of the four dishes. The photographs also show that the axes of fuel cells 1 and 3 have shifted 7 degrees in such a way that the tops of the fuel cells point outward. It is not possible to determine conclusively from the photographs whether or not cryogenic oxygen tank no. 2 is present, partially missing, or totally missing. It is probable that the loud bang heard by the crew was caused by the separation of the panel from the spacecraft approximately at the time of telemetry loss.

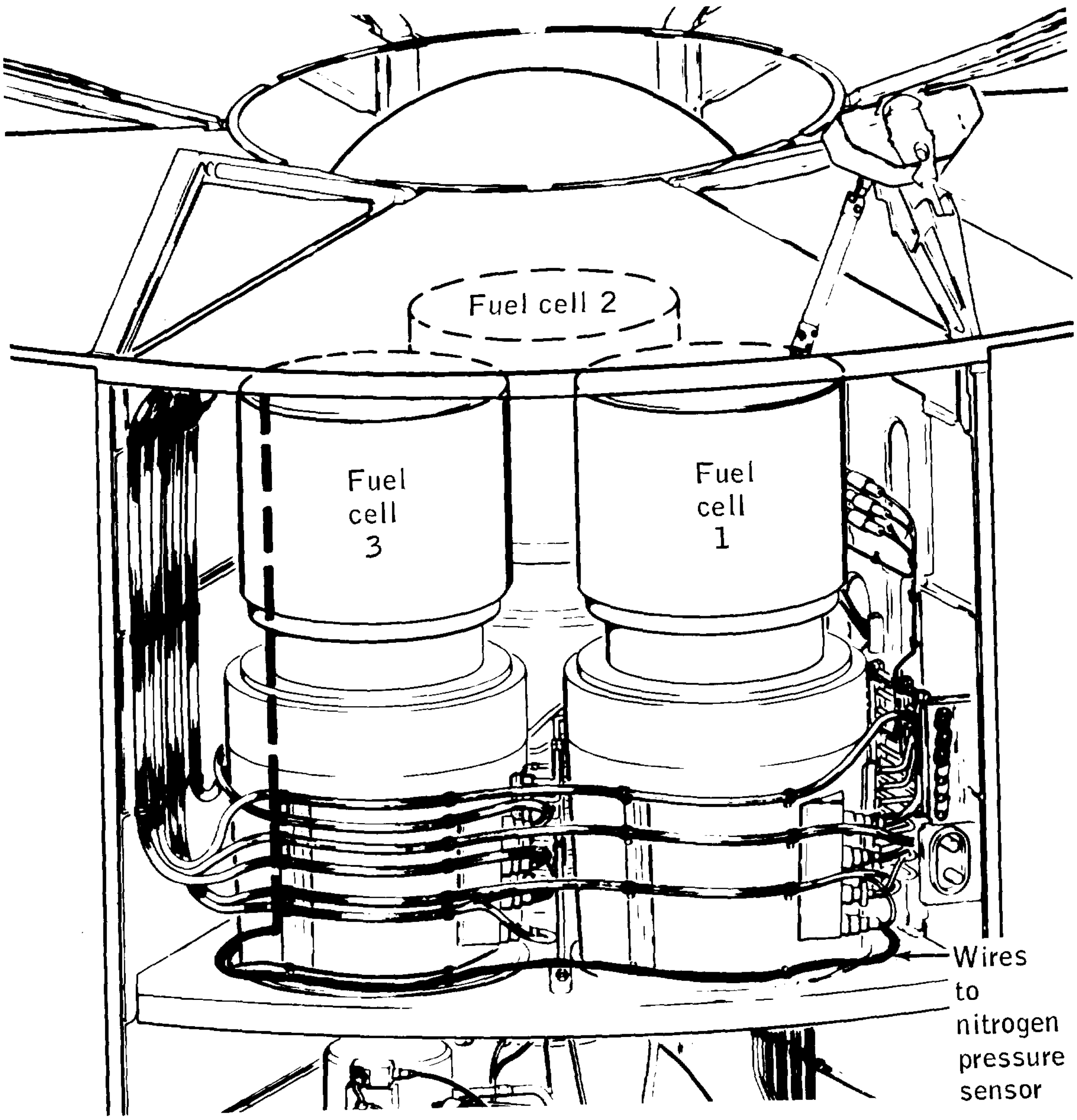


Figure B5-7.- Location of wiring to nitrogen pressure sensor in fuel cell 1.

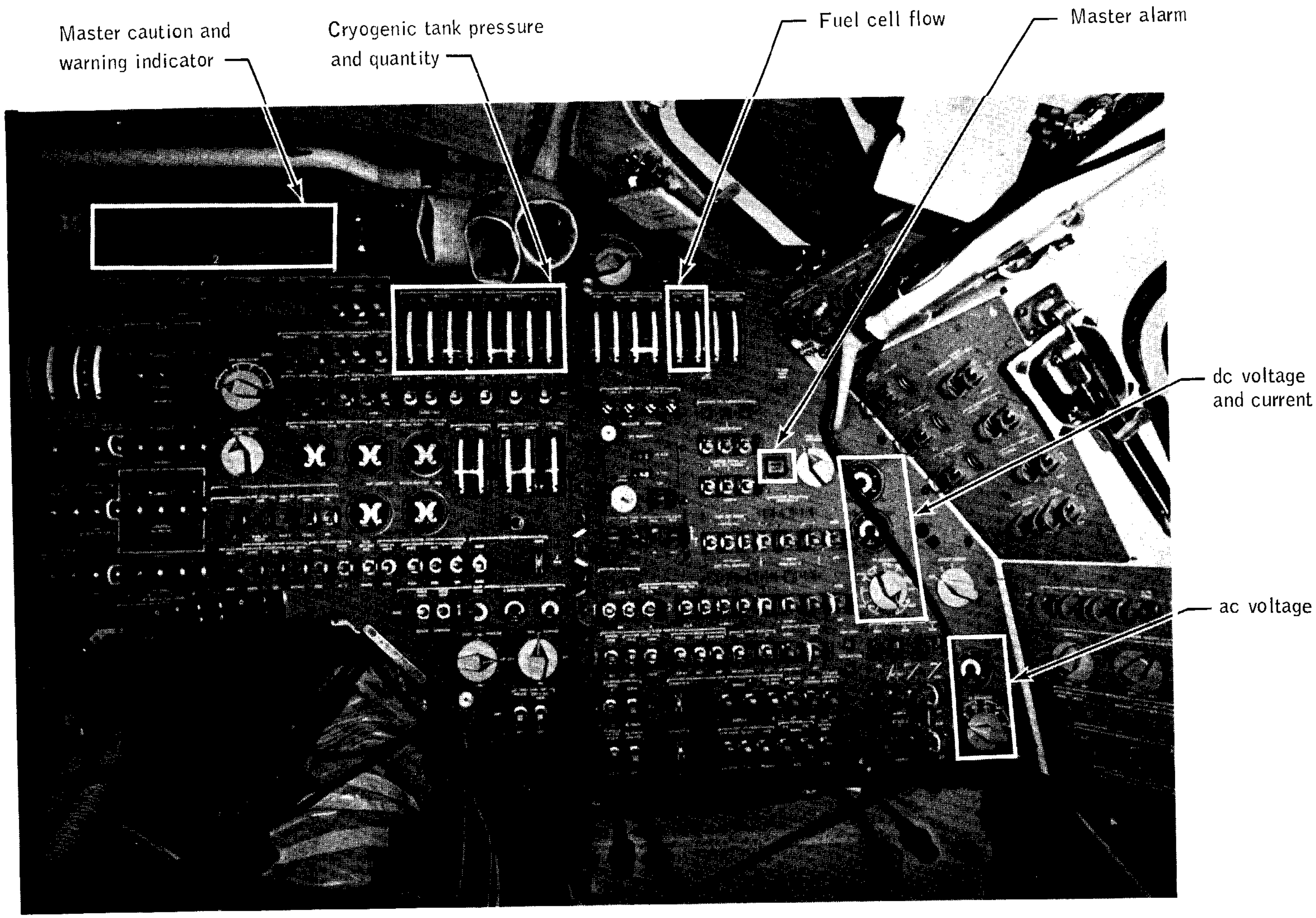


Figure B5-8.- Command module instrument panel.

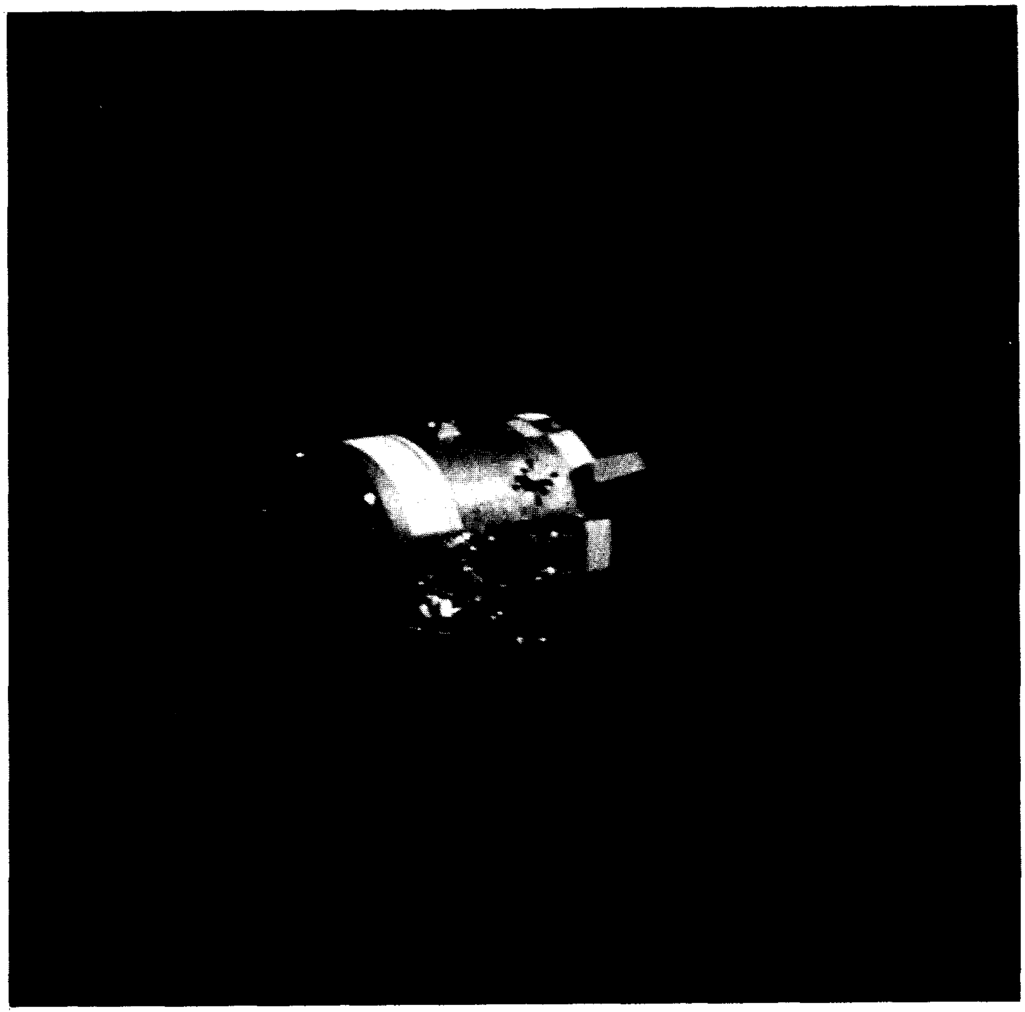


Figure B5-9.- Photograph of service module taken by crew.

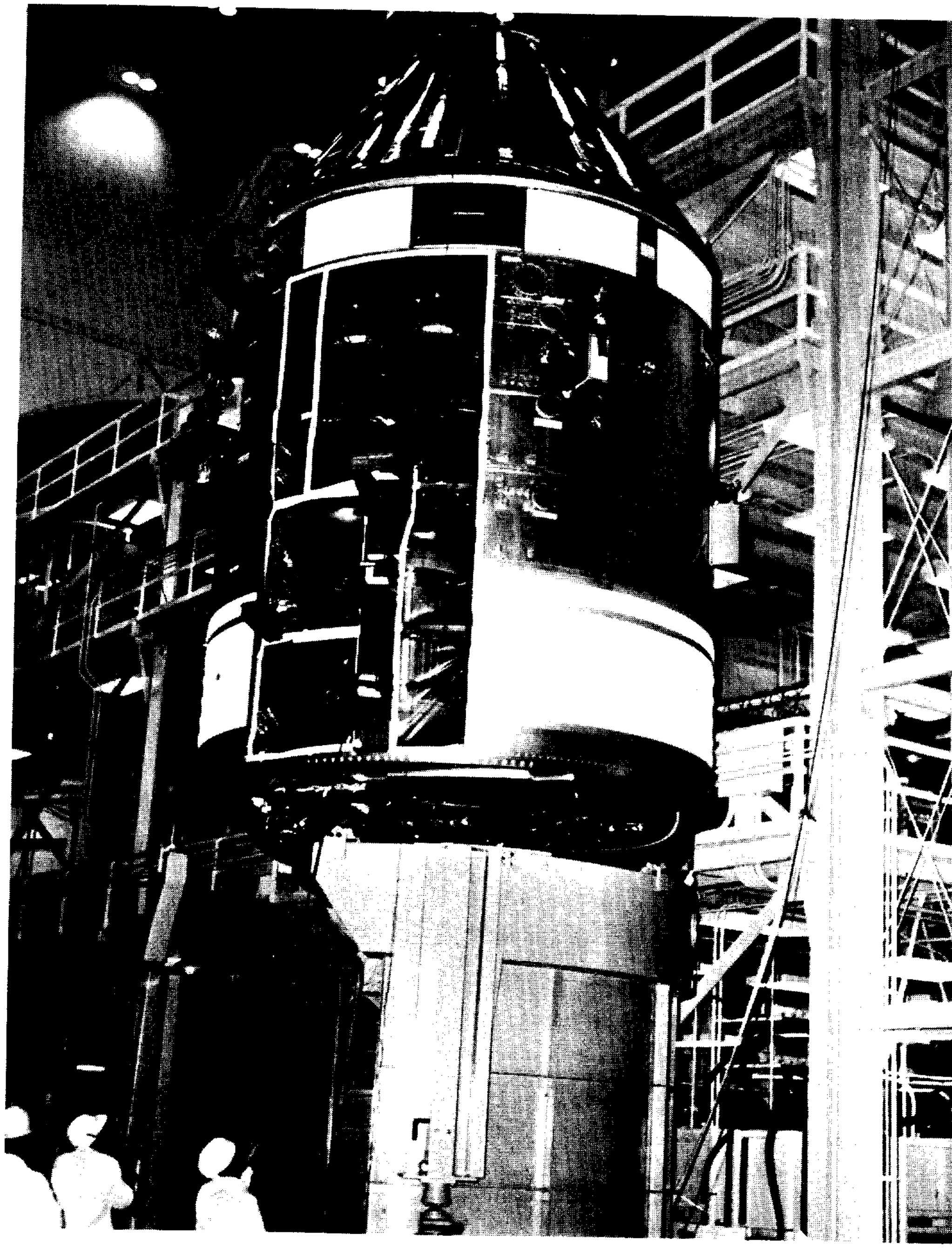


Figure B5-10.- Bay 4 of service module.

CHANGES IN SPACECRAFT DYNAMICS

At 55:54:53.182, less than half a second before telemetry loss, the body-mounted linear accelerometers in the command module, which are sampled at 100 times per second, began indicating spacecraft motions. These disturbances were erratic but reached peak values of 1.17g, 0.65g, and 0.65g in the X, Y, and Z directions, respectively, about 13 milleseconds before data loss.

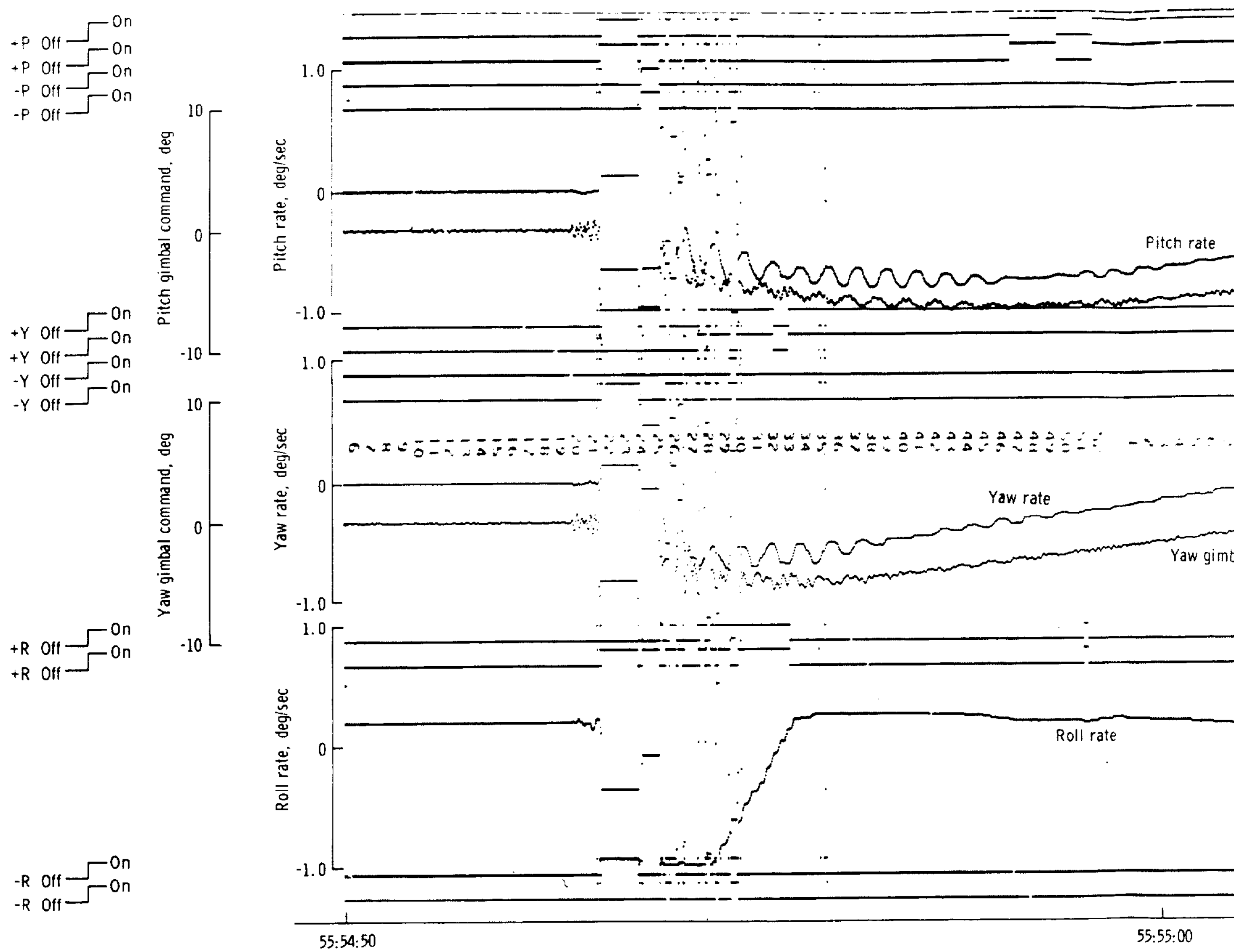
At 55:54:53.220 the pitch, roll, and yaw rate gyros indicated low-amplitude variations in output. These gyros are body mounted in the command module, have a full-scale range of ±1 degree per second, are sampled 100 times per second, and provide a fairly sensitive indication of spacecraft motions. They are also sensitive to electrical disturbances not necessarily associated with the gyros; however, the characteristics of the output at 55:54:53.220 are believed to have resulted from low-amplitude dynamic forces acting on the spacecraft. These channels were, of course, lost at 55:54:53.555, along with all other telemetered data. Figure B5-11 is a record of all three rate gyro outputs.

When telemetry was recovered at 55:54:55.35, these channels definitely indicated that moments had been applied to the spacecraft. The total change in angular moment was:

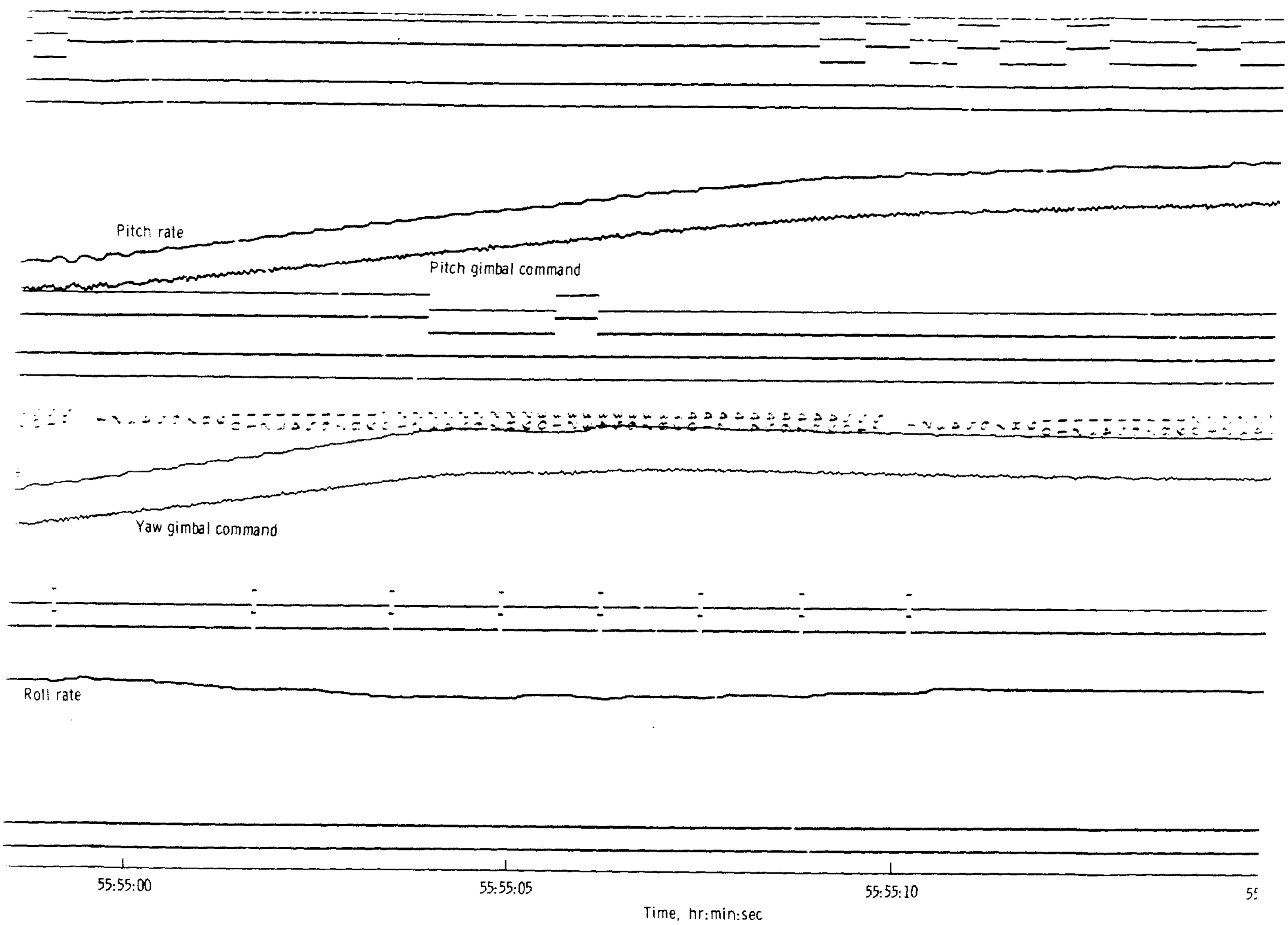
Roll	-1535 ft-lb-sec
Pitch	-6482 ft-lb-sec
Yaw	-5919 ft-lb-sec

The roll, pitch, and yaw rates were automatically compensated for by the attitude control system, as shown in figure B5-11.

The inertial platform on the command module contains three mutually orthogonal integrating accelerometers, whose outputs are telemetered with an increment value of 0.2 fps. After telemetry was recovered, a change of two increments was observed in one axis, one increment in the second axis, and zero increments in the third axis.



+P Off On
+P Off On
-P Off On
-P Off On
+Y Off On
+Y Off On
-Y Off On
-Y Off On
+R Off On
+R Off On
-R Off On
-R Off On
Pitch gimbal command, deg
10
0
-10
Yaw gimbal command, deg
10
0
-10
Pitch rate, deg/sec
1.0
0
-1.0
Yaw rate, deg/sec
1.0
0
-1.0
Roll rate, deg/sec
1.0
0
-1.0
Pitch rate
Yaw rate
Yaw gimt
Roll rate
55:54:50
55:55:00



Pitch rate
Pitch gimbal command
Yaw gimbal command
Roll rate
55:55:00
55:55:05
55:55:10
Time, hr:min:sec

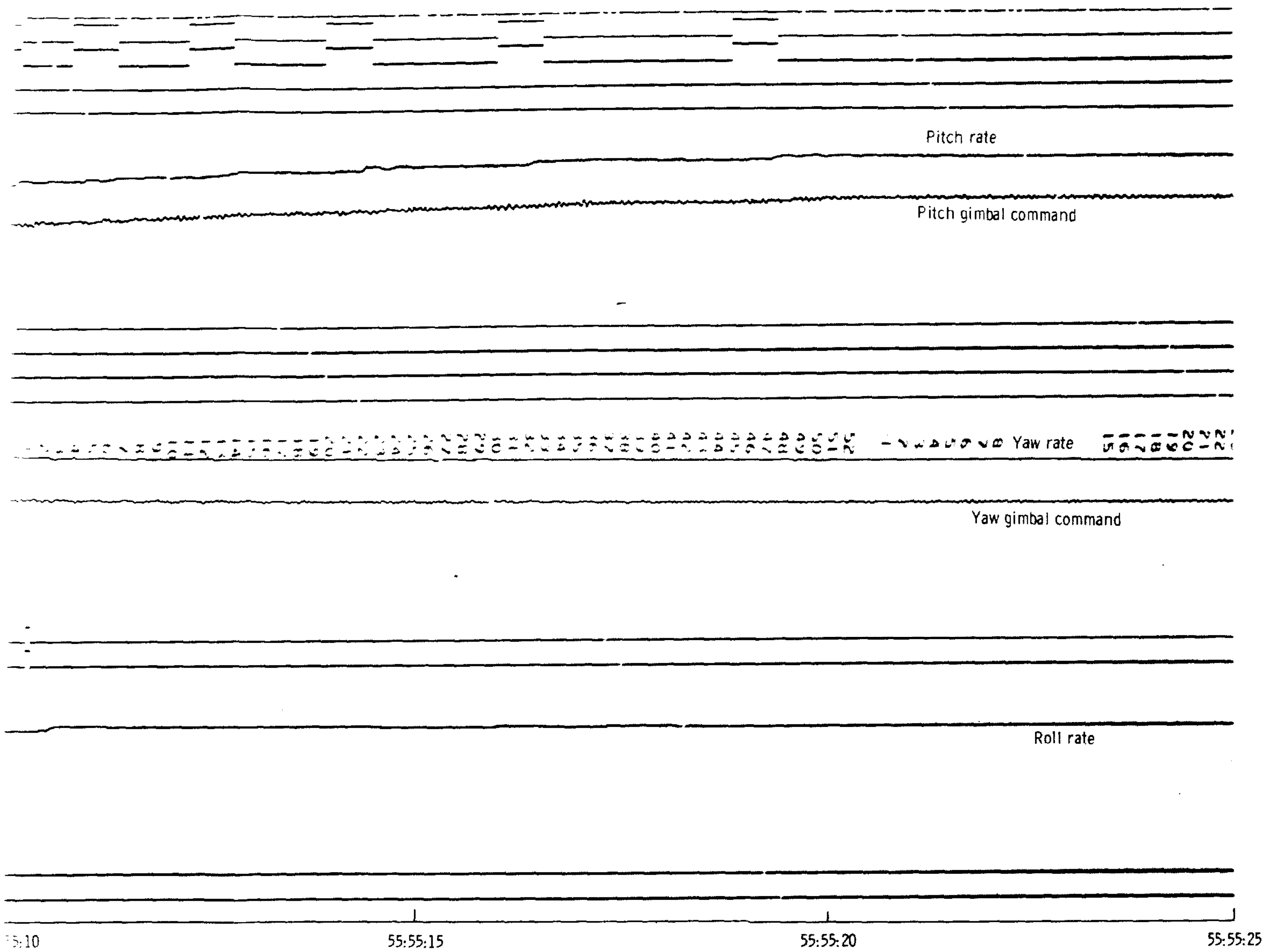


Figure B5-11.- Roll, pitch, and yaw rates.

When transformed from platform coordinates to spacecraft coordinates, this represents a velocity change of 0.4 to 0.6 fps. The uncertainty in this measurement is due to the fact that the PCM system has an increment value of 0.2 fps. The velocity change was combined with the observed roll, pitch, and yaw rates; and a single equivalent impulse acting on the spacecraft calculated. The impulse components are:

X	500 lb-sec
Y	800 lb-sec
Z	-900 lb-sec

This indicates that the force was directed generally normal to the panel covering bay 4 of the service module. The extremely coarse data upon which this calculation is based makes it impossible to better define the force acting upon the spacecraft.

After recovery of data, the integrating accelerometer on the spacecraft stable platform also began to show an abnormal output. Calculations show that the force producing the acceleration amounted to about 60 pounds in the -X direction (retro thrust) over a period of about 8 minutes. At about the same time Commander Lovell reported seeing extensive venting of gases from the service module which definitely was not a normal or expected part of the spacecraft operation at that time. He later described the venting as continuous, looking like, "......a big sheet with the sun shining on it--very heavy--like fine spray from a water hose," unlike gases and liquids vented during other planned spacecraft operations.

The radial velocity of the Apollo spacecraft relative to the Earth can be accurately determined by measuring the doppler shift of the S-band signal transmitted to Earth. Spacecraft velocity components normal to the line between the spacecraft and Earth cannot be determined by this method. The doppler velocity measurement is routinely made every 10 seconds and has an equivalent noise level of 0.015 fps.

Between 55:54:45 and 55:55:05 doppler measurements indicated a radial velocity increment of 0.26 fps. This is shown in figure B5-12. Following this abrupt change in velocity at approximately the time of telemetry loss, additional velocity changes were observed, as shown in figure B5-13. These velocity increments were caused in part by venting from the spacecraft and in part by firings of reaction control system jets.

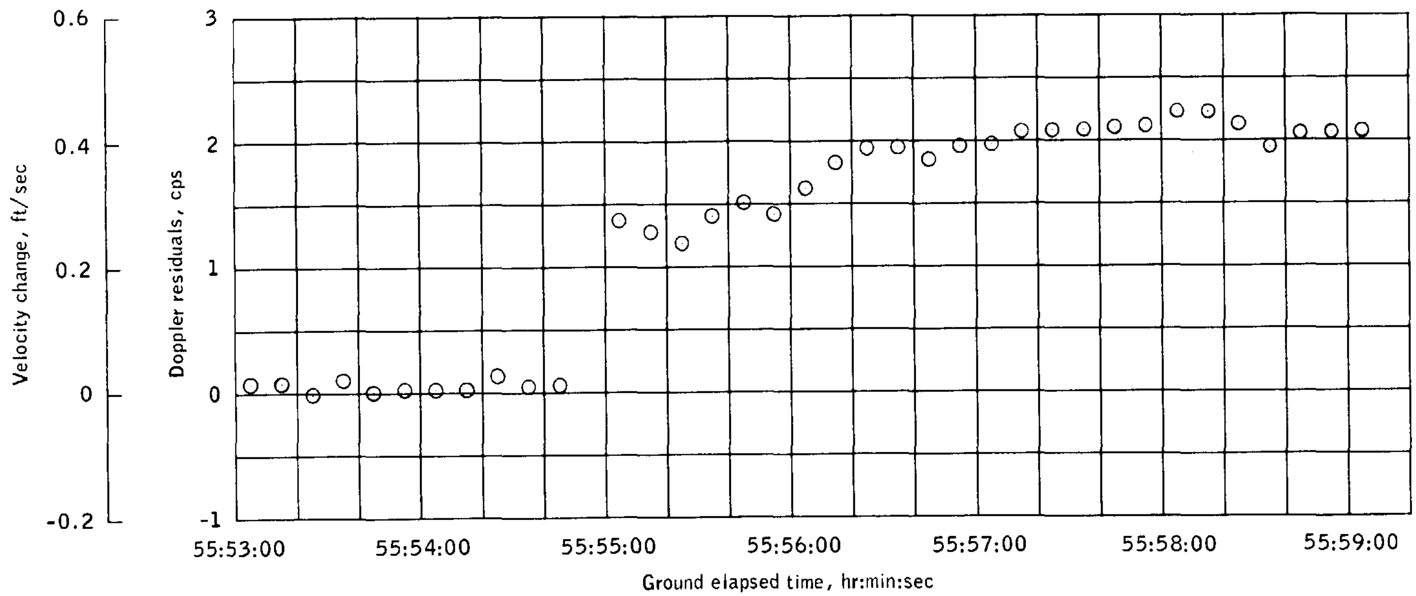


Figure B5-12.- Velocity increment at 55:54:53.

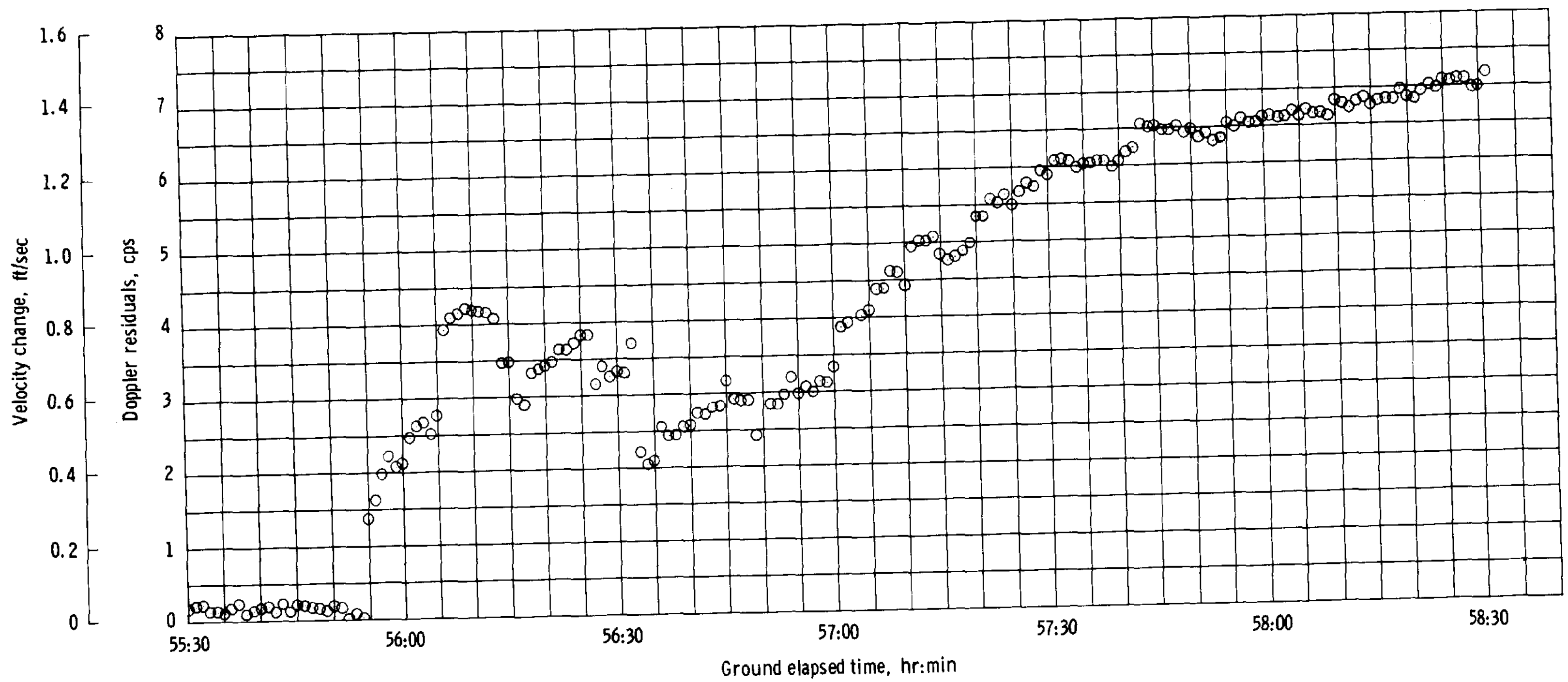


Figure B5-13.- Doppler tracking data.

The jet firings caused velocity increments rather than pure rotation rate changes because the jets did not always fire in opposed pairs. This resulted from the power system configuration in the spacecraft and closure of the quad C valves. (See Part B6 of this Appendix.)

TEMPERATURE CHANGES OBSERVED IN SERVICE MODULE

Following the recovery of telemetry data there were a number of temperature changes observed at various locations in the service module. The locations of all temperature sensors in the service module are shown in figure B5-14, and telemetered records from these sensors for the time period of 55:53:40 to 55:56:10 are shown in figure B5-15. From these temperature records the following conclusions can be drawn:

1. Both temperature measurements in bay 3, the bay adjacent to bay 4, increased after 55:54:55, whereas they had been steady prior to that time.

2. The corresponding temperature measurements in bay 5 showed much smaller increases. Bay 5 is adjacent to bay 4.

3. A change was observed in the service propulsion valve body temperature. This sensor, unlike many temperature sensors in the lower part of the service module, is not covered by multiple layers of insulation.

4. Four sensors located in close proximity on the separator between bay 4 and bay 5 showed rapid temperature rises of small magnitude immediately after the recovery of telemetry data. These sensors measure the temperatures of fuel cells 1 and 3 radiator inlets and outlets.

5. The temperature of quad C and D reaction control engines continued the same rate of rise after data loss as before data loss.

FAILURE OF CRYOGENIC OXYGEN SYSTEM

The telemetered quantities from cryogenic oxygen tank no. 2 were all off-scale following the recovery of telemetry at 55:54:55. Since no accurate assessment of the damage to this tank has been possible, the readings of the sensors within it are in doubt. The temperature was reading full-scale high and continued this way until 55:55:49,

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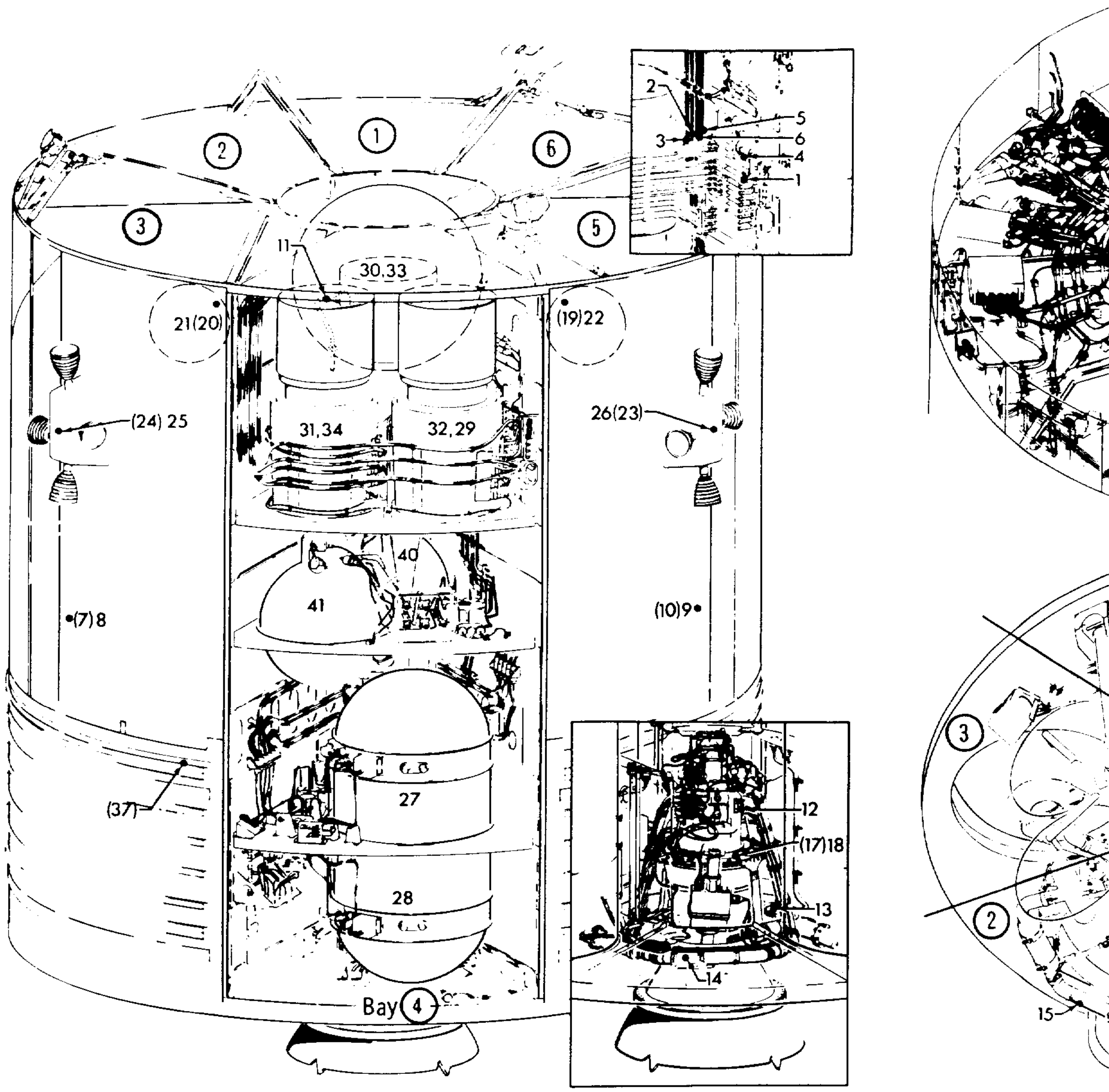
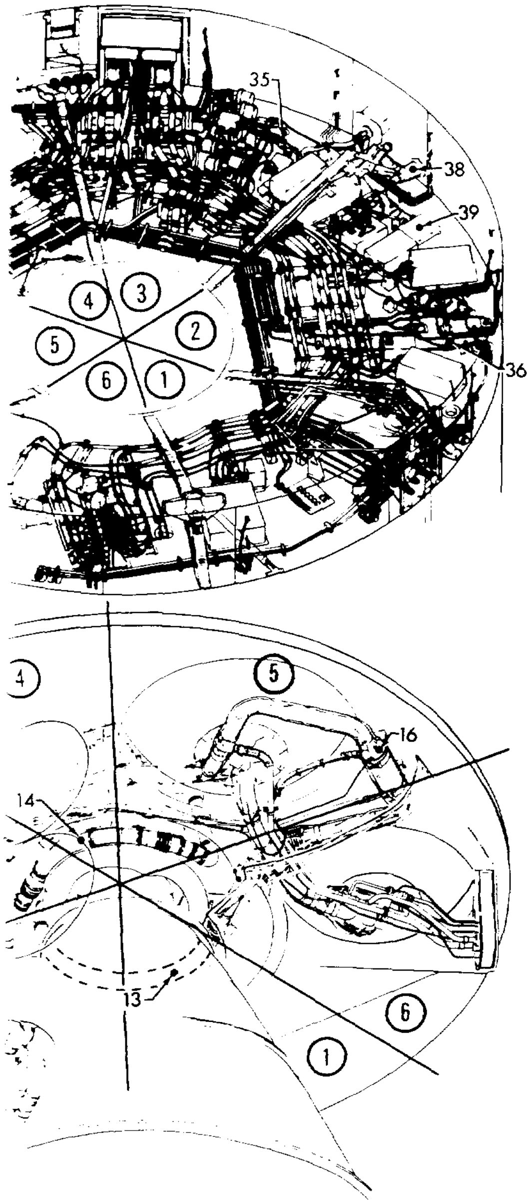


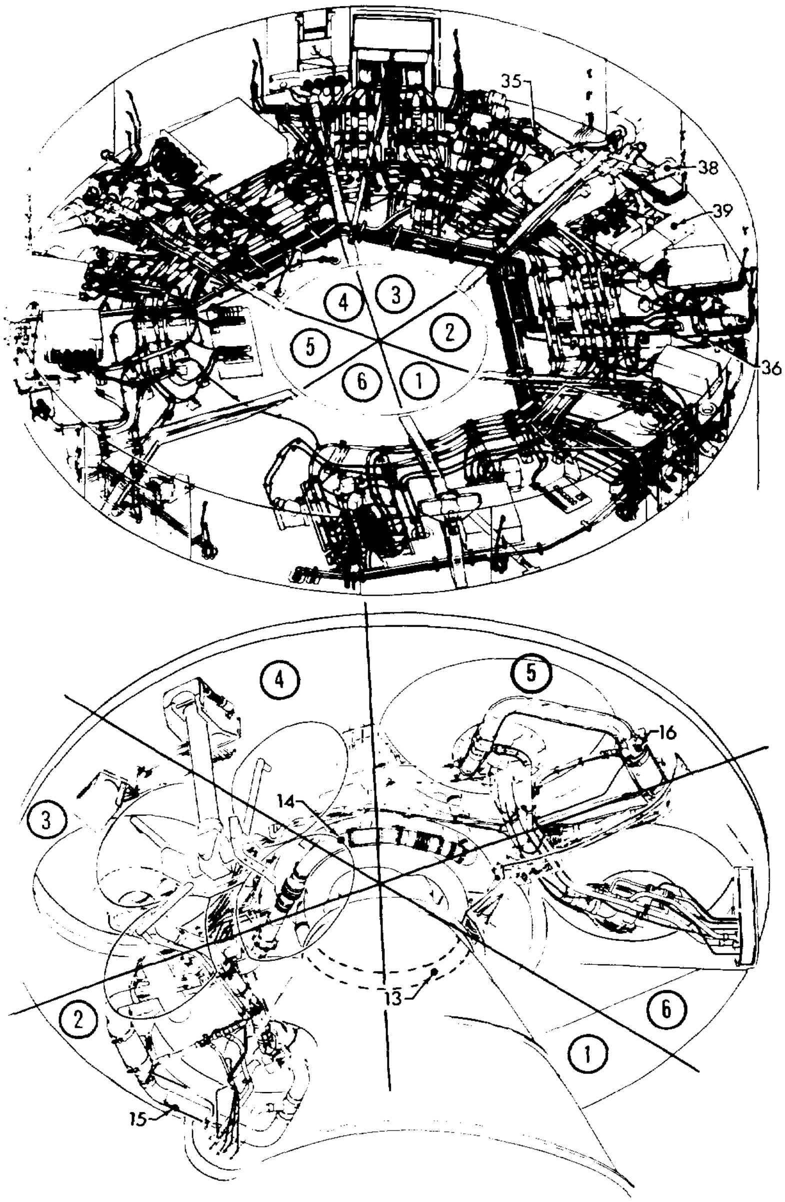
1
2
3
4
5
6
2
3
5
6
4
1
11
30,33
21(20)
(19)22
(24) 25
26(23)
31,34
32,29
40
41
•(7) 8
(10)9•
(37)
27
28
Bay 4
12
(17)18
13
14
15



Service module temperature measurements			
Number	Measurement number	Title	Remarks
1	SC2090	Fuel cell 1 radiator inlet	
2	SC2091	Fuel cell 2 radiator inlet	
3	SC2092	Fuel cell 3 radiator inlet	
4	SC2087	Fuel cell 1 radiator outlet	
5	SC2088	Fuel cell 2 radiator outlet	
6	SC2089	Fuel cell 3 radiator outlet	
7[a]	SA2377	Bay 2 oxidizer tank surface	Opposite 9
8	SA2378	Bay 3 oxidizer tank surface	
9	SA2379	Bay 5 fuel tank surface	
10[a]	SA2380	Bay 6 fuel tank surface	Opposite 8
11	SP0002	Helium tank	
12	SP0045	Service propulsion valve body	
13	SP0048	Service propulsion fuel feed line	
14	SP0049	Service propulsion oxidizer feed line	
15	SP0054	Service propulsion oxidizer line	
16	SP0057	Service propulsion fuel line	
17[a]	SP0061	Service propulsion injector flange 1	Opposite 18 on flange
18	SP0062	Service propulsion injector flange 2	
19[a]	SR5013	Reaction control helium tank quad A	Bay 6 opposite 21
20[a]	SR5014	Reaction control helium tank quad B	Bay 2 opposite 22
21	SR5015	Reaction control helium tank quad C	Bay 3
22	SR5016	Reaction control helium tank quad D	Bay 5
23[a]	SR5065	Reaction control engine package quad A	Bay 6 opposite 25
24[a]	SR5066	Reaction control engine package quad B	Bay 2 opposite 26
25	SR5067	Reaction control engine package quad C	Bay 3
26	SR5068	Reaction control engine package quad D	Bay 5
27	SC0043	Hydrogen tank 1	
28	SC0044	Hydrogen tank 2	
29	SC2084	Fuel cell 1 skin	Internal
30	SC2085	Fuel cell 2 skin	Internal
31	SC2086	Fuel cell 3 skin	Internal
32	SC2081	Fuel cell 1 condenser exhaust	Internal
33	SC2082	Fuel cell 2 condenser exhaust	Internal
34	SC2083	Fuel cell 3 condenser exhaust	Internal
35	SF0260	Environmental control primary radiator inlet	
36	SF0262	Environmental control secondary radiator inlet	
37[a]	SF0263	Environmental control secondary radiator outlet	Bay 6 opposite
38	ST0840	Nuclear particle detector	
39	ST0841	Nuclear particle analyzer	
40	SC0041	Oxygen tank 1	
41	SC0042	Oxygen tank 2	

[a] Located on opposite side of vehicle and not shown in the view.

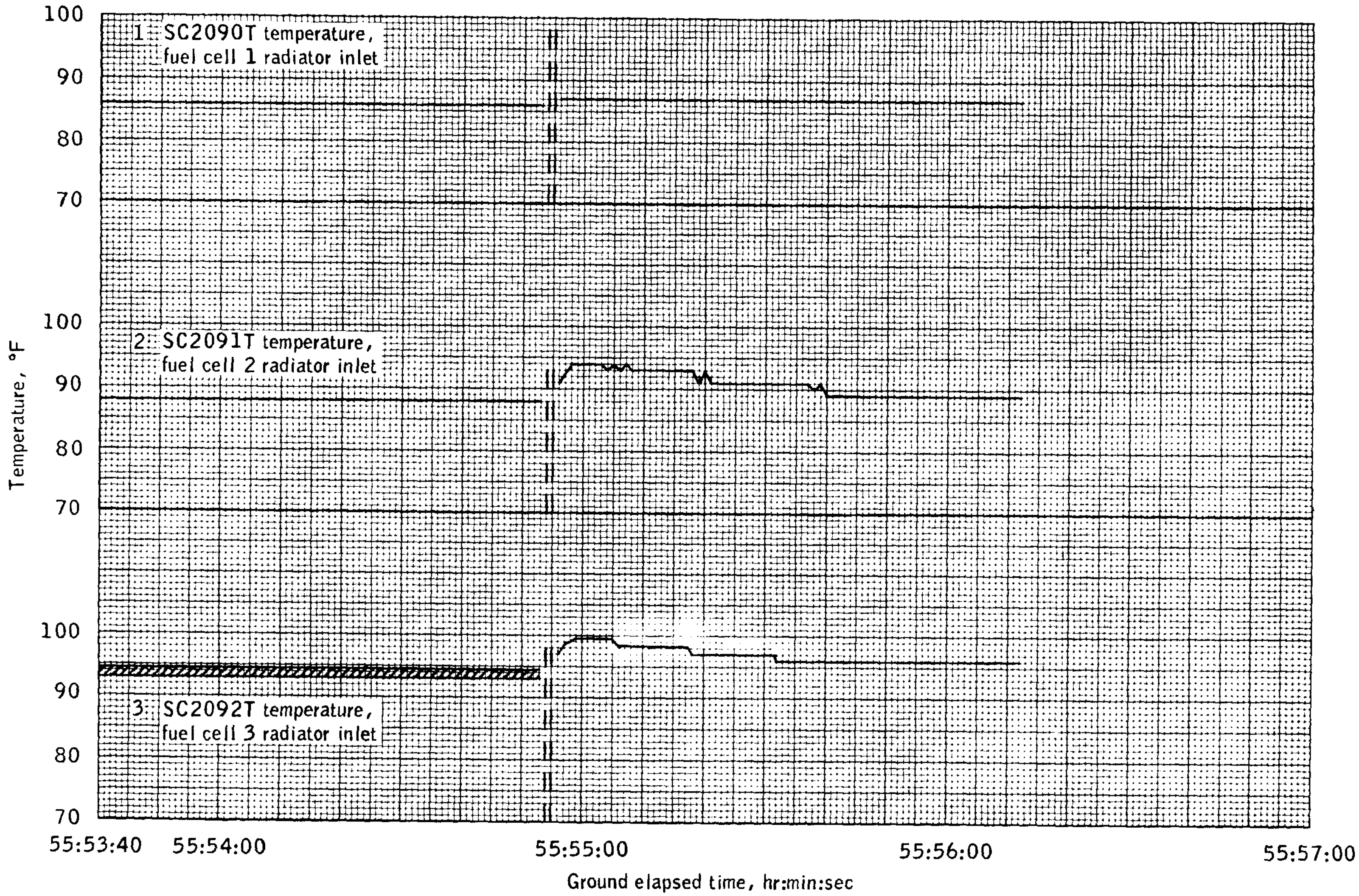
Figure B5-14.- Temperature sensor location in Apollo service module.



Number	Measurement number	Title	
Service module temperature measurements			
1	SC2090	Fuel cell 1 radiator inlet	
2	SC2091	Fuel cell 2 radiator inlet	
3	SC2092	Fuel cell 3 radiator inlet	
4	SC2087	Fuel cell 1 radiator outlet	
5	SC2088	Fuel cell 2 radiator outlet	
6	SC2089	Fuel cell 3 radiator outlet	
7[a]	SA2377	Bay 2 oxidizer tank surface	Opposi
8	SA2378	Bay 3 oxidizer tank surface	
9	SA2379	Bay 5 fuel tank surface	
10[a]	SA2380	Bay 6 fuel tank surface	Opposi
11	SP0002	Helium tank	
12	SP0045	Service propulsion valve body	
13	SP0048	Service propulsion fuel feed line	
14	SP0049	Service propulsion oxidizer feed line	
15	SP0054	Service propulsion oxidizer line	
16	SP0057	Service propulsion fuel line	
17[a]	SP0061	Service propulsion injector flange 1	Opposi
18	SP0062	Service propulsion injector flange 2	
19[a]	SR5013	Reaction control helium tank quad A	Bay 6
20[a]	SR5014	Reaction control helium tank quad B	Bay 2
21	SR5015	Reaction control helium tank quad C	Bay 3
22	SR5016	Reaction control helium tank quad D	Bay 5
23[a]	SR5065	Reaction control engine package quad A	Bay 6
24[a]	SR5066	Reaction control engine package quad B	Bay 2
25	SR5067	Reaction control engine package quad C	Bay 3
26	SR5068	Reaction control engine package quad D	Bay 5
27	SC0043	Hydrogen tank 1	
28	SC0044	Hydrogen tank 2	
29	SC2084	Fuel cell 1 skin	Intern
30	SC2085	Fuel cell 2 skin	Intern
31	SC2086	Fuel cell 3 skin	Intern
32	SC2081	Fuel cell 1 condenser exhaust	Intern
33	SC2082	Fuel cell 2 condenser exhaust	Intern
34	SC2083	Fuel cell 3 condenser exhaust	Intern
35	SF0260	Environmental control primary radiator inlet	
36	SF0262	Environmental control secondary radiator inlet	
37[a]	SF0263	Environmental control secondary radiator outlet	Bay (
38	ST0840	Nuclear particle detector	
39	ST0841	Nuclear particle analyzer	
40	SC0041	Oxygen tank 1	
41	SC0042	Oxygen tank 2	

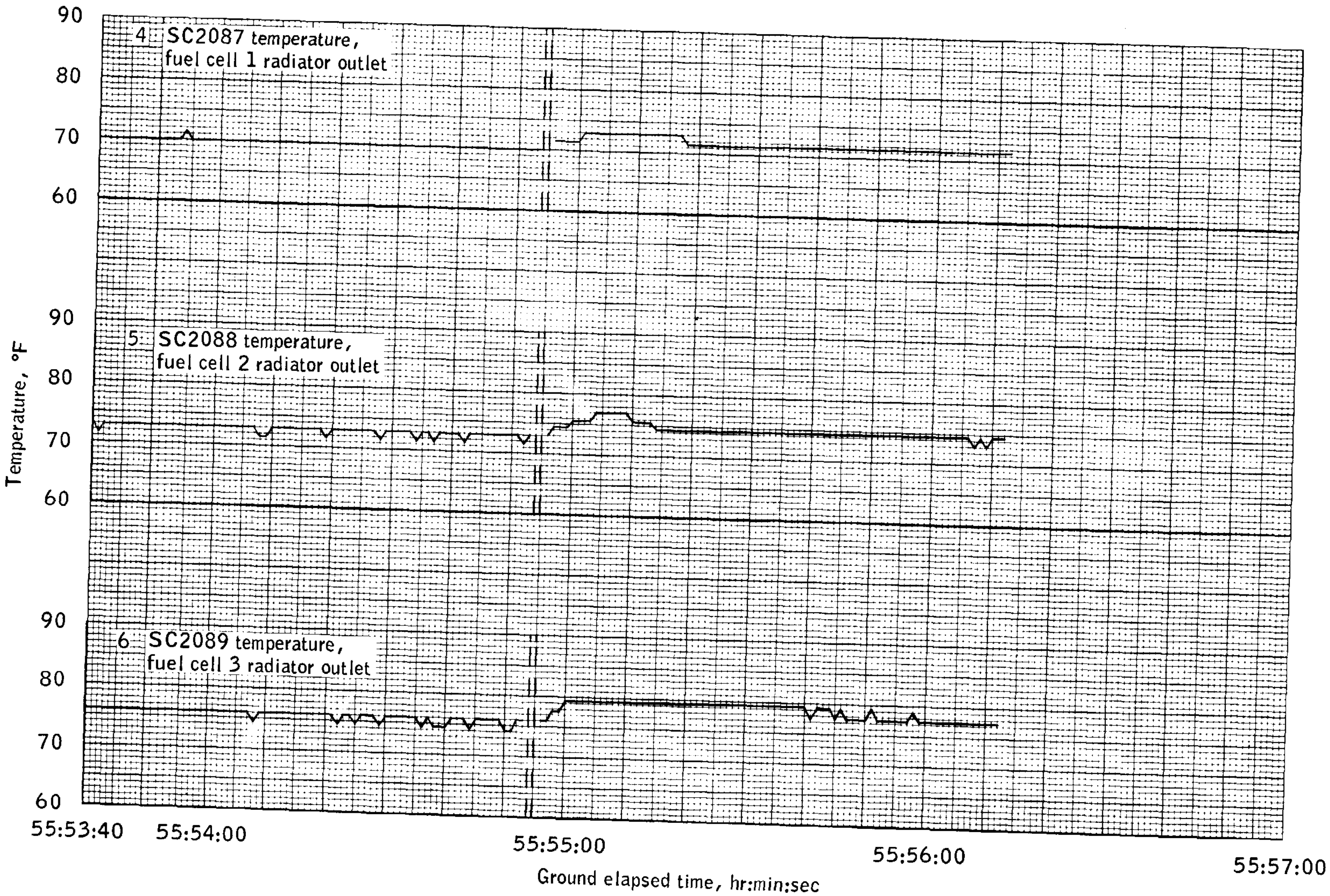
[a] Located on opposite side of vehicle and not shown in the view.

Figure B5-14.- Temperature sensor location in Apollo s



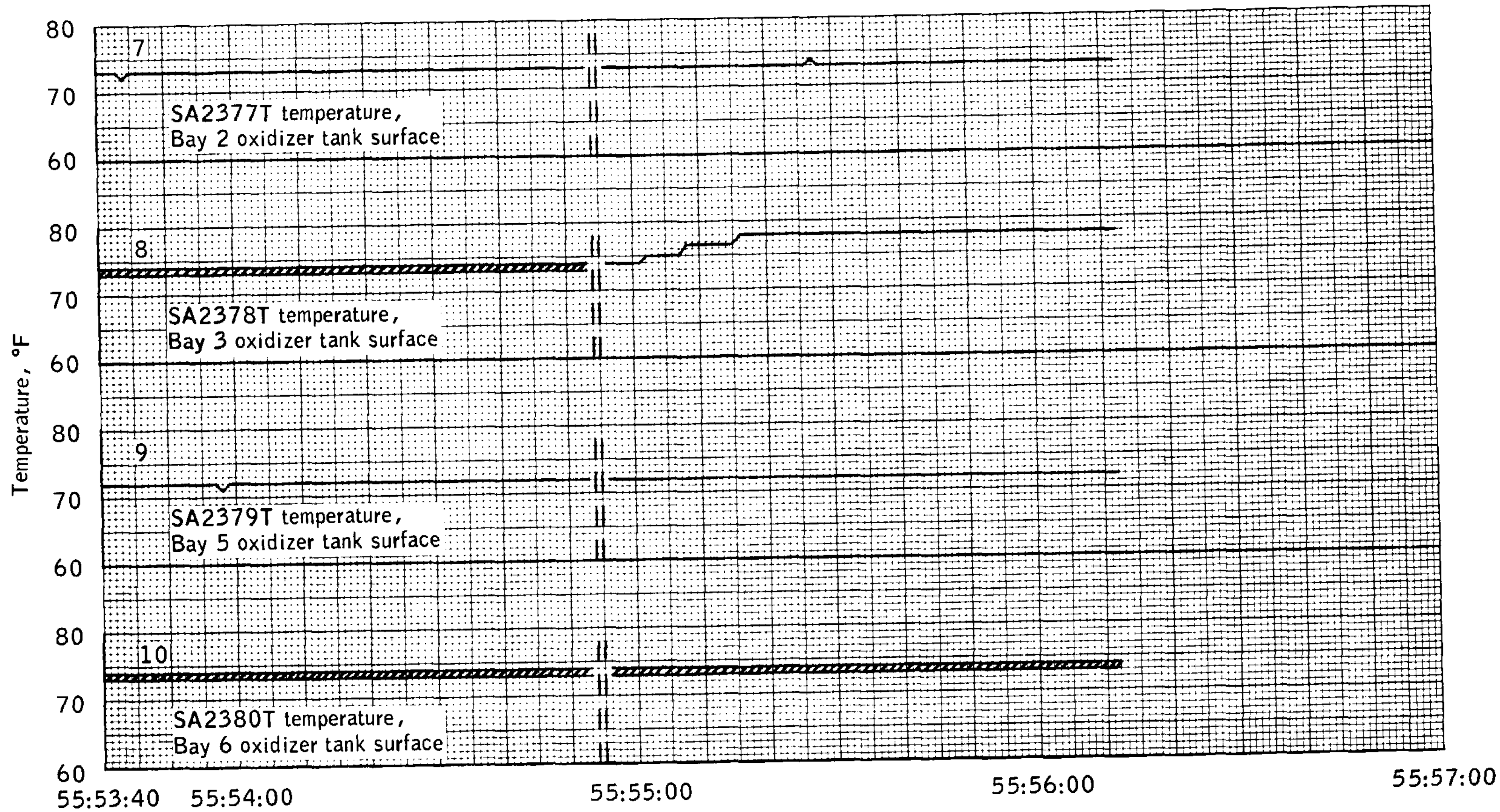
(a) Radiator inlet temperatures.

Figure B5-15.- Service module temperature history.



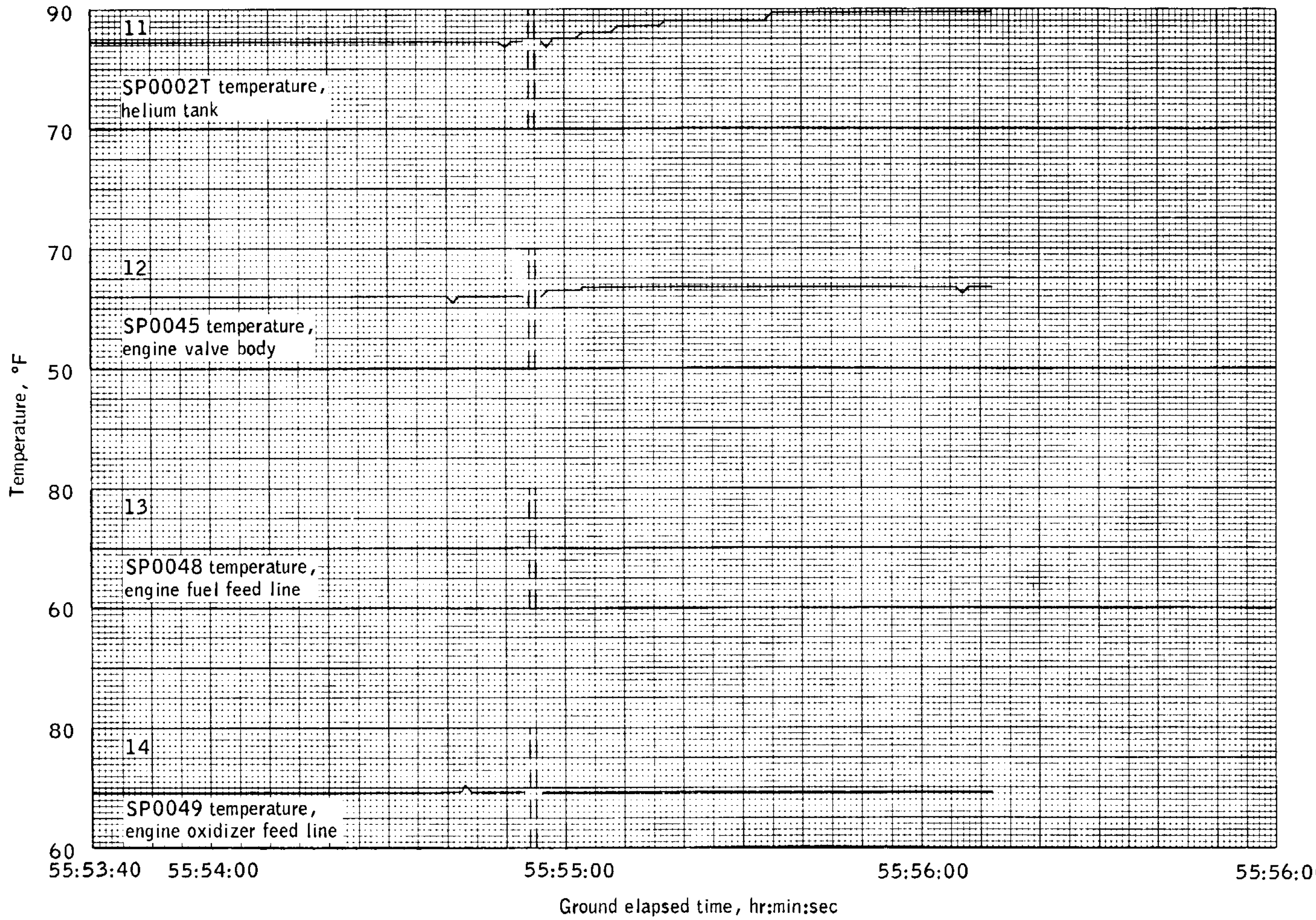
(b) Radiator outlet temperatures.

Figure B5-15.- Continued.



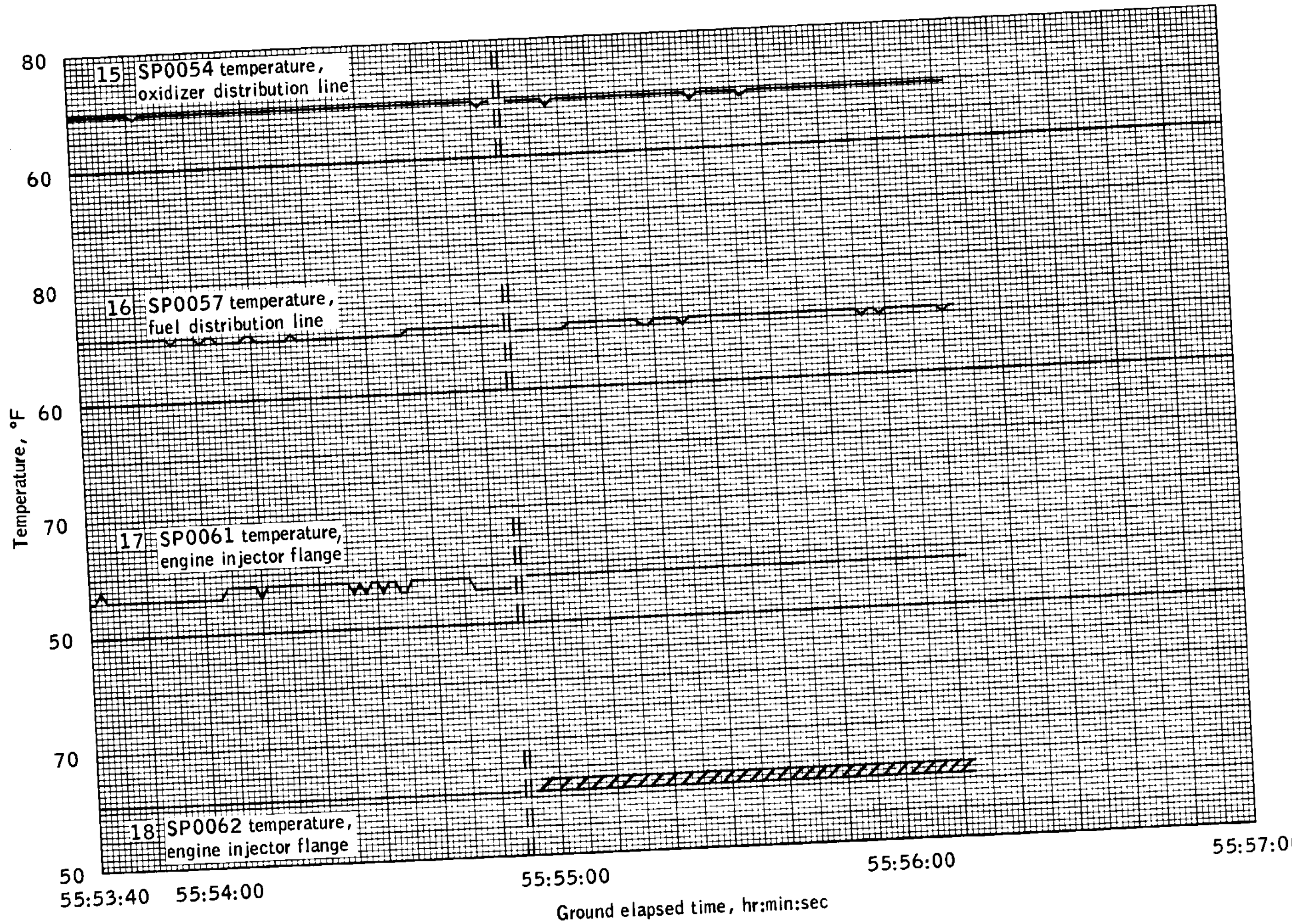
(c) Tank surface temperatures.

Figure B5-15.- Continued.



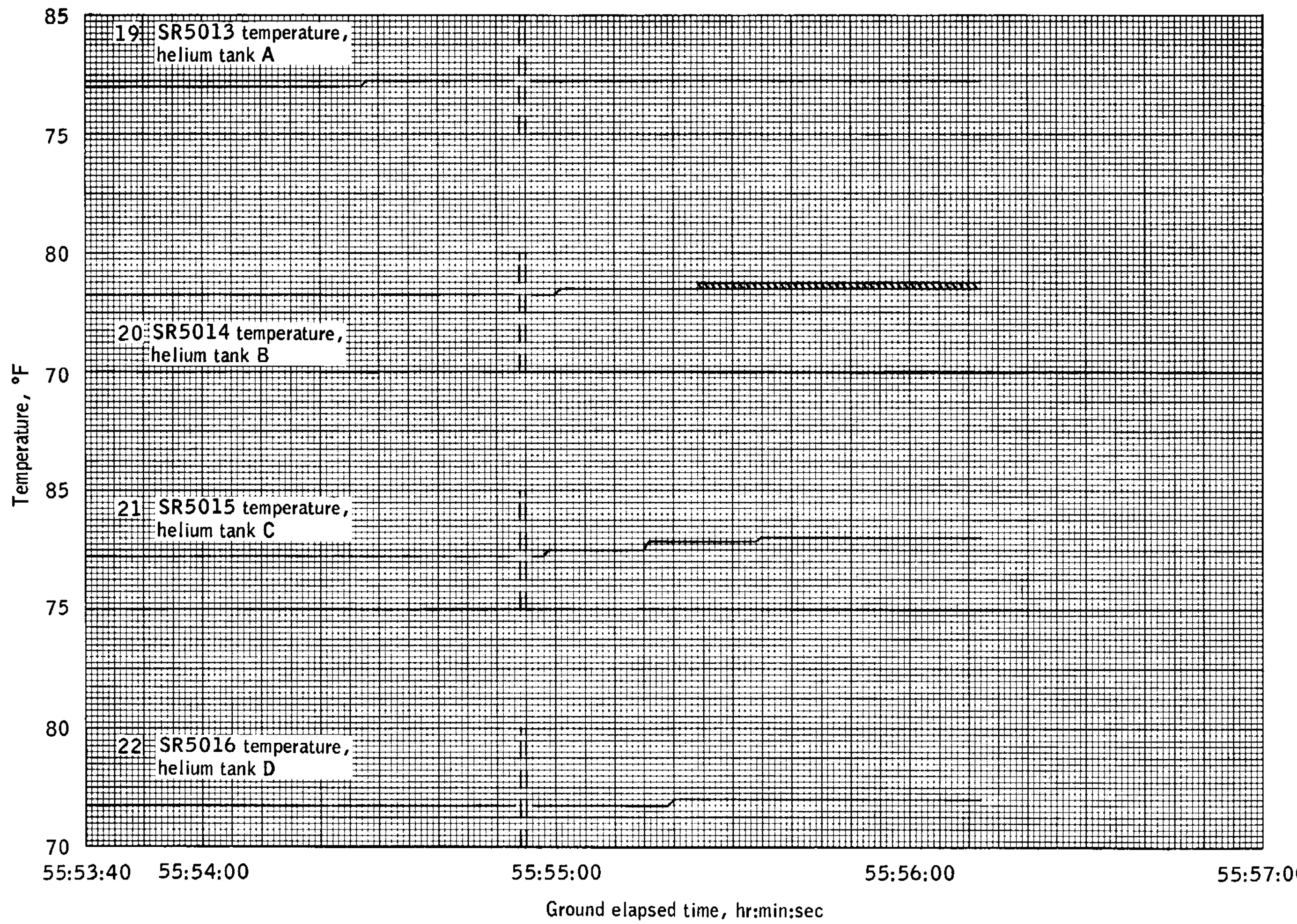
(d) Service propulsion temperatures, group 1.

Figure B5-15.- Continued.



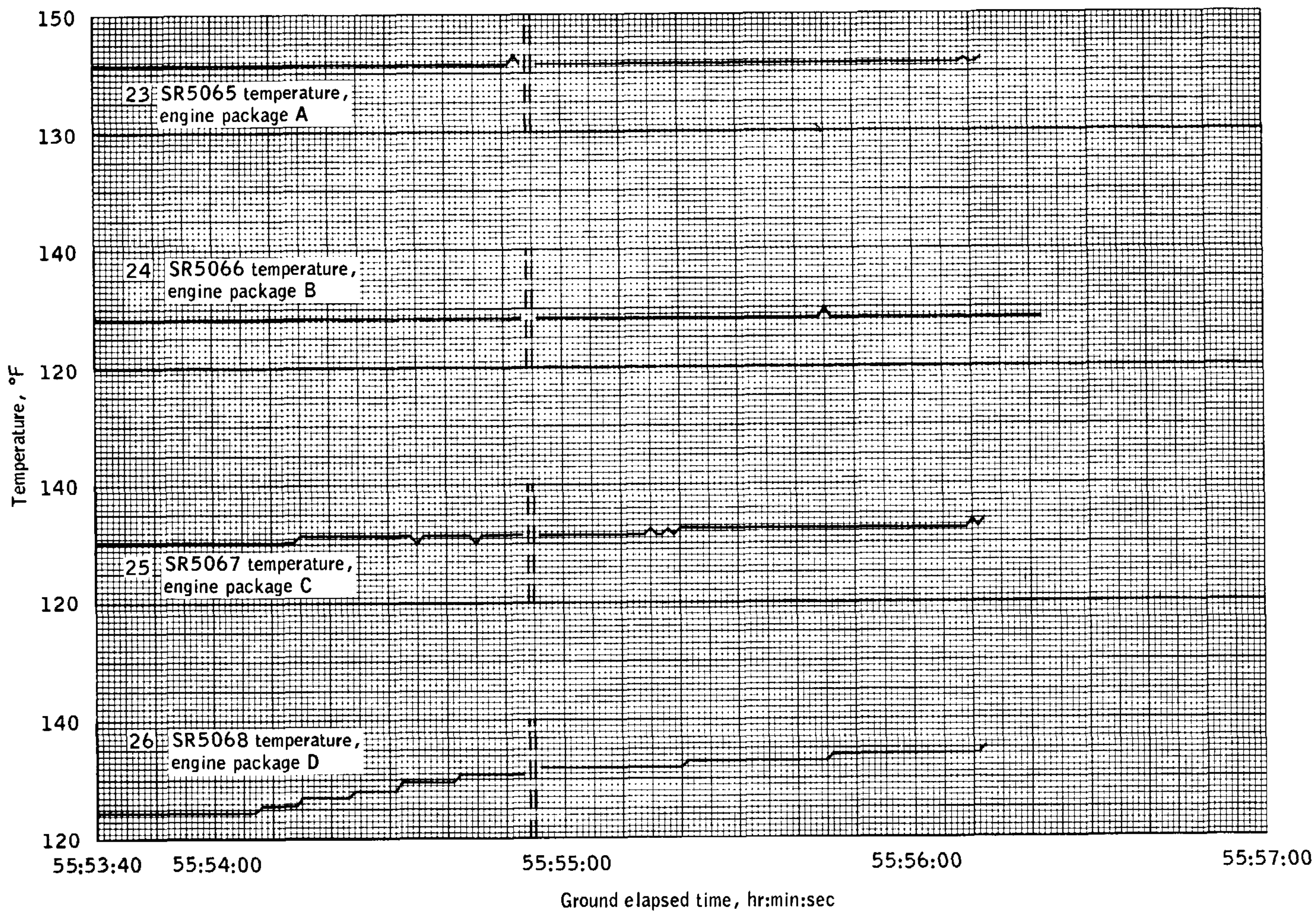
(e) Service propulsion temperatures, group 2.

Figure B5-15.- Continued.



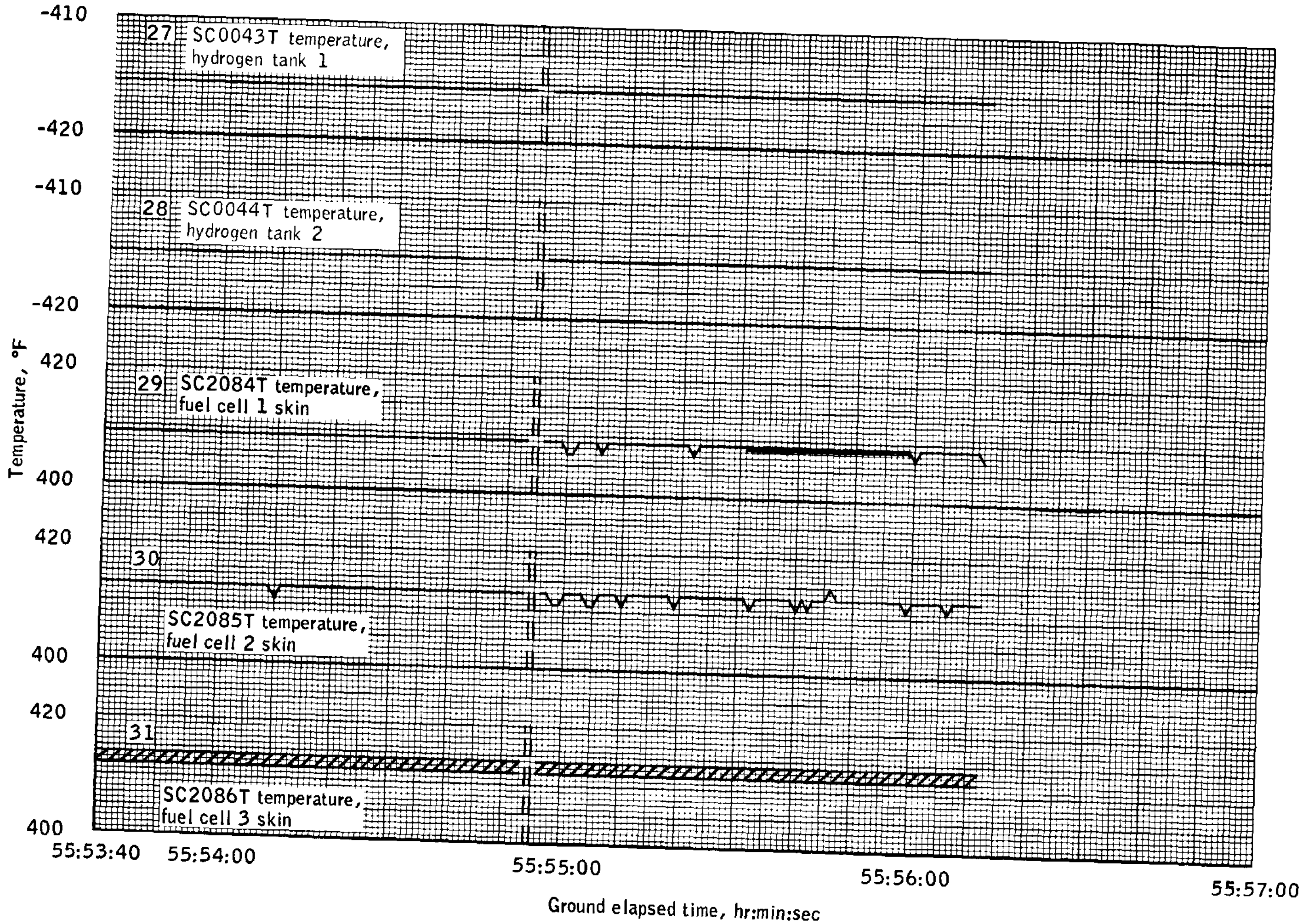
(f) Reaction control helium tank temperatures.

Figure B5-15.- Continued.



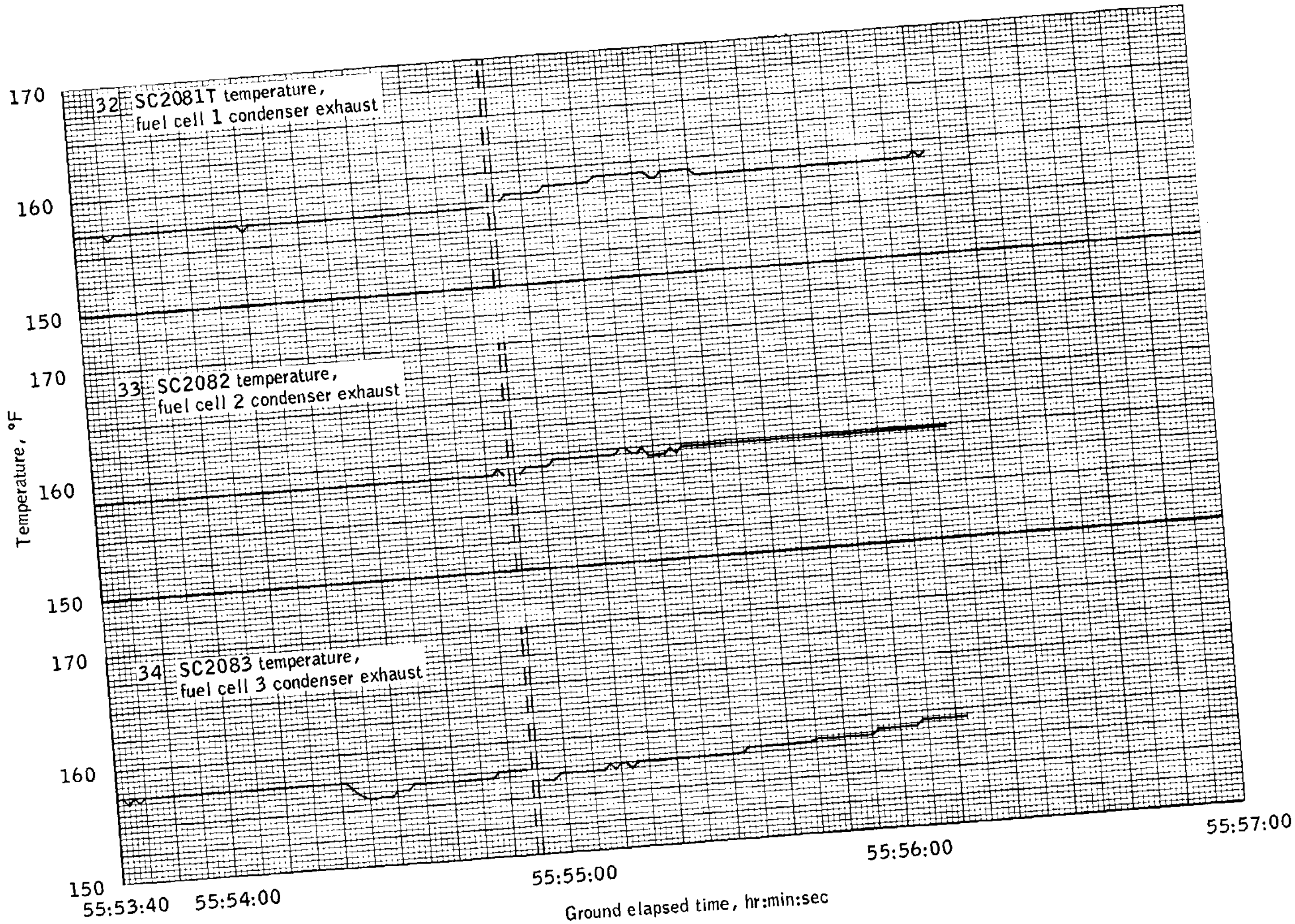
(g) Reaction control engine package temperatures.

Figure B5-15.- Continued.



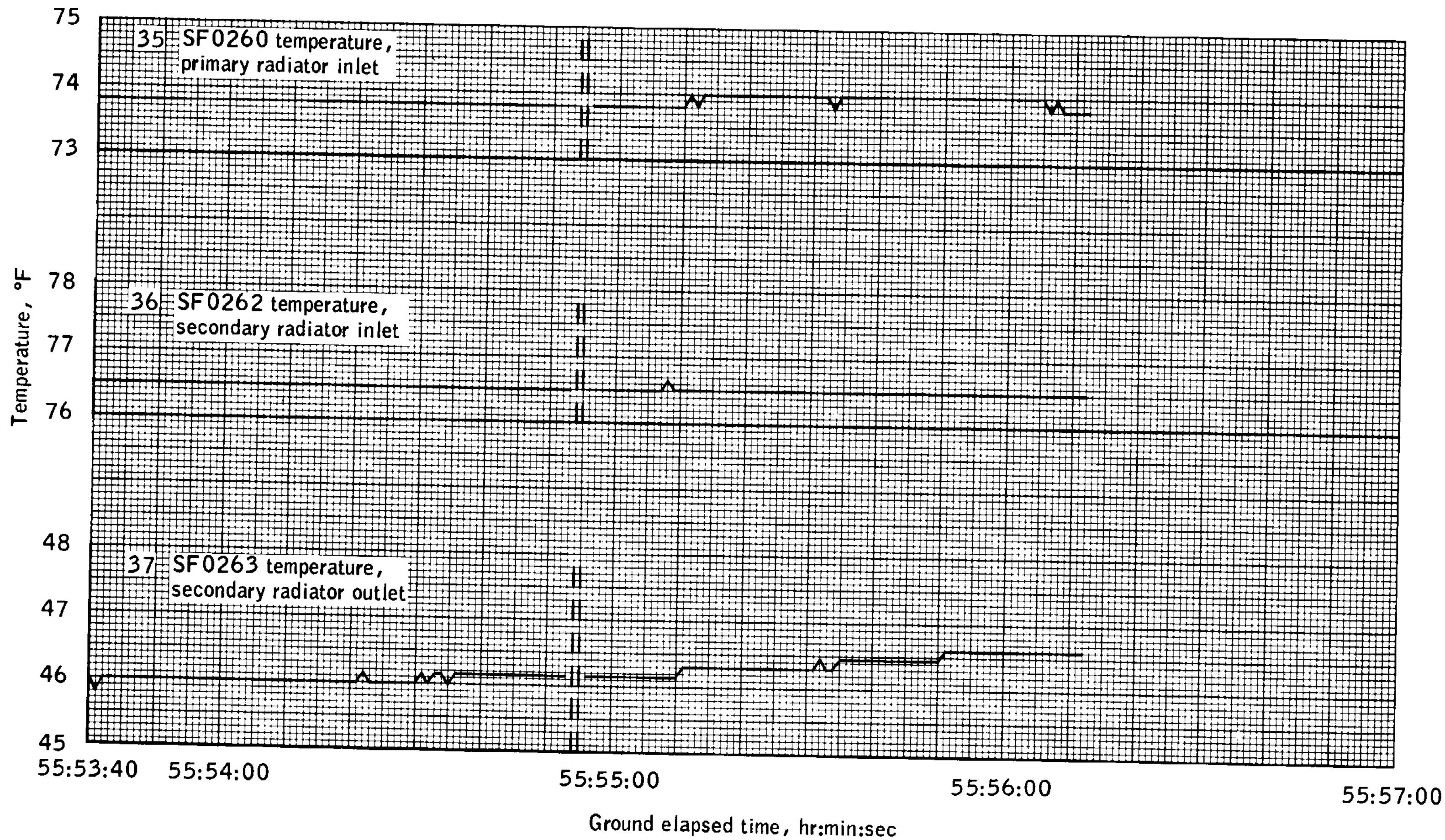
(h) Fuel cell skin temperatures.

Figure B5-15.- Continued.



(i) Fuel cell condensor exhaust temperatures.

Figure B5-15.- Continued.



(j) Environmental control radiator temperatures.

Figure B5-15.- Concluded.

when it began a steady decrease which ended at 55:56:48 in an off-scale-low reading. This behavior is a possible result of a failure of the sensor. For an explanation of possible failure modes of this sensor, see Part B7 of this Appendix. The pressure of cryogenic oxygen tank no. 2, sensed remotely, read off-scale low and continued to show this reading. Off-scale low for this sensor represents a pressure of 19 psia or below.

The quantity gage read full-scale high after telemetry recovery and continued in this state until 55:56:38 when it began oscillating in an erratic manner. The oscillation continued until 55:57:47 when the gage assumed an off-scale-low reading. The quantity gage and its failure modes are described in Part B7 of this Appendix.

The pressure in cryogenic oxygen tank no. 1 had dropped from 879 psia to 782 psia during telemetry loss. This pressure continued to drop at a slow rate for about 2 hours until it was insufficient for operation of the last remaining fuel cell.

The heaters in both cryogenic oxygen tanks were off prior to telemetry loss as a result of the high pressure in tank no. 2. After telemetry recovery the total fuel cell current indicated an increase of about 5 amperes after known loads had been accounted for. The low pressure levels in both oxygen tanks should have caused both heaters to be on at this time. The total current drain by the heaters in any one tank is about 5 amperes. It therefore appears that the heaters in one tank had come on since telemetry loss and were operating at this time. It is possible that the heaters in cryogenic oxygen tank no. 2 were either physically open-circuited before or at the time of the bang.

Additional evidence that the heaters in only one tank were on can be obtained by observing that at 56:19:03 the spacecraft dc current decreased 5 amperes. This is the time at which the crew began to power down the spacecraft according to the emergency powerdown checklist. If heaters in both oxygen tanks had been on at that time, the current should have decreased approximately 11 amperes instead of the observed 5 amperes.

OPERATION OF THE ELECTRICAL POWER SYSTEM

Following the period of telemetry loss, a high-current condition existed on the fuel cell outputs for 19 seconds. In the same time period, the two dc main voltages were approximately 0.9 volt lower than their previous values. By 55:55:14 the voltages and currents had become normal. The observed currents during the 19-second period have been correlated with reaction control system jet firings and an inertial measurement unit heater cycle. The excellent correlation indicates that no unaccountable loads were added to the power system during this time period.

The crew observed a master caution and warning signal 1 or 2 seconds after the bang, along with an indication of undervoltage on dc bus B. The master caution and warning was turned off at 55:55:00.

Within 5 seconds after the resumption of telemetry data, the oxygen flow rates to fuel cells 1 and 3 had decreased to approximately 20 percent of their prior values. These flow rates remained at a sufficiently low level to cause failure of fuel cells 1 and 3 at approximately 55:58. The most probable explanation for the reduced oxygen flow rates is that at the time of the bang a sufficiently intense shock occurred to close the valves in the oxygen lines feeding fuel cells 1 and 3.

There is sufficient volume in the oxygen lines between the supply valves and the fuel cells to maintain fuel cell operation for the observed time of about 3 minutes. The intensity of the shock is indicated by the fact that the reaction control system valves on quad C were closed. Tests on these valves have shown that 80g for 10 milliseconds will cause them to close. Tests on the oxygen supply valves have shown that a shock of 86g for 11 milliseconds will cause them to close.

The crew was not alerted to the abnormally low flow rate of oxygen to fuel cells 1 and 3 because the hydrogen supply valves had not been closed. The valve closure indicator is only activated when both the oxygen and hydrogen supply valves to a fuel cell are closed. The first indication to the crew that the power system was failing came at 55:57:39 when the master caution and warning was triggered by a main bus B undervoltage, occasioned by the failure of fuel cell 3. Main bus B voltage dropped to an unusable level within 5 seconds, causing ac bus 2 to drop to zero at 55:57:45.

The crew quickly checked the ac and dc voltage levels, recognized that ac bus 2 had failed, and responded by switching ac loads from ac bus 2 to ac bus 1. This heavier load on ac bus 1 was reflected as a heavier load on dc main A, causing it to drop in voltage. At 55:58:06, a dc main A undervoltage master caution and warning was triggered as the main voltage dropped to between 25 and 26 volts. Shortly afterwards, at approximately 55:58:06, fuel cell 1 failed, placing the entire load of dc main A on fuel cell 2. Fuel cell 2 was now called upon to supply a current of 50 amperes.

Fuel cell 2 remained the major source of electrical power in the command module for the next 2 hours. During this time, telemetry continued to indicate a decreasing cryogenic oxygen pressure in tank no. 1. At 58:04 battery A was connected to main bus A and fuel cell 2 was removed from operation when oxygen flow became insufficient.

PART B6

POSTINCIDENT EVENTS

The description of postincident events is presented in two sections. The first, entitled "Immediate Recovery," describes the flightcrew and flight controller actions during the 2-1/2-hour period following the incident. This section is primarily concerned with actions of the flightcrew and flight controllers during this period in response to the immediate problems caused by the spacecraft failures. The long-term problems addressed by Mission Control are described in the second section, entitled "Plans and Actions Taken to Return the Crew to Earth."

IMMEDIATE RECOVERY

The first indication in the Mission Control Center of any problem in the spacecraft came from the Guidance Officer who reported that he had observed a "hardware restart." This term describes the action of the onboard computer when certain computer electrical problems occur, such as a reference voltage or an oscillator frequency getting out of tolerance. When this occurs, the computer stops its computations and recycles to a specified location in the program. Computations will not resume until the out-of-tolerance condition is cleared. At 55:55 this event occurred so rapidly that the flight controllers did not observe the computer halt; they only saw that it had occurred.

The report of the hardware restart was followed almost immediately by the crew's report, "I believe we've had a problem here." This was followed quickly by a statement from the crew that they had a main bus B undervolt indication from the master caution and warning (MC&W) system. Flight controllers responsible for the electrical, environmental control, and instrumentation systems immediately searched their displays, but at that time there were no indications of any electrical problems, all voltages and fuel cell currents appeared normal. Apparently, the main bus B undervolt problem was a transient that had cleared up, for the crew next reported the bus voltages were "looking good." However, the flight controllers knew that all was not well because the oxygen tank measurements indicated some major problems in this system, or its instrumentation.

The next report in Mission Control was from the flight controller responsible for the communication systems. He stated that the high-gain antenna on the spacecraft had unaccountably switched from narrow beam width to wide beam width at approximately the same time the problem had occurred.

In sorting out these pieces of information, the flight controllers initially suspected that there had been an instrumentation failure. However, with the subsequent failure of main bus B and ac bus 2 it became more obvious that a serious electrical problem existed. The flight controllers considered the possibility that a short had occurred, and that this was in some way related to the unusual behavior of the high-gain antenna. The rapid rate at which so many parameters in the electrical and cryogenic system had changed state made it impossible to tell which were causes and which were effects.

The Mission Control Center response to the situation is described in this section of the report. The time interval covered is from 55 hours 58 minutes ground elapsed time (55:58 g.e.t.) to 58:40 g.e.t., when all power was removed from the command module (CM). The major portion of the activities of both the flightcrew and the flight controllers in this time period was directed toward (1) evaluation and management of the electrical and cryogenic oxygen problems; (2) maintenance of attitude control; and (3) activation of the lunar module (LM). A chronological listing of all significant actions is presented first. This is followed by a more detailed description of the three categories of activities mentioned above.

Chronology of Spacecraft Reconfiguration Actions

This listing was obtained from transcripts of air-to-ground voice records (ref. 2) and the "Flight Director" loop in the Mission Control Center. Additional information was obtained from interviews with members of the flight control team. Some editing has been done to eliminate the description of routine actions which obviously have no significance to this investigation: examples are omni antenna switching and the loading of weight and inertia information in the digital autopilot. The times at which specific actions are listed are only approximately correct, (±1 minute) since there was no precise time correlation available.

<u>55:59 - Fuel cell main bus connection</u>.- Mission Control requested the crew to connect fuel cell 1 to main bus A and fuel cell 3 to main bus B. Although there was no direct evidence the crew had changed the fuel cell and main bus configuration, the flight controller believed that this might be the case. The configuration prior to the loss of main bus B was as follows: fuel cell 1, main A; fuel cell 2, main A; and fuel cell 3, main B.

<u>56:03 - Entry battery on line</u>.- The crew placed entry battery A on main bus A to increase the bus voltage. Mission Control was just about to ask that this be done. The bus voltage was approximately 25 volts, which is about 1-1/4 volts below the MC&W trip limit.

56:08 - Open circuit fuel cell 1.- Mission Control requested the crew to open circuit fuel cell 1. Flight controllers did not understand the problems with the fuel cells; the data were confusing and incomplete. In an effort to get some new information, the controllers decided to take all loads off fuel cell 1 to see if it would behave any differently. It was not putting out any power so there was no reason to leave it connected to the main bus.

56:11 - Power RCS jets from main bus A.- Mission Control requested the crew to position some RCS jet select switches to main A power. All of the quad C jets and B-3 and B-4 jets had been powered from main bus B and since that bus had no power on it, they could not fire except by the "Direct" coils. By switching these jets to main bus A, there was at least one jet available for automatic control in each direction about each axis.

56:14 - Start emergency powerdown.- Mission Control advised the crew to use page 1-5 of their Emergency Powerdown Checklist, part of the Flight Data File (ref. 7) carried by the crew. Mission Control wanted to get the current on main bus A reduced by at least 10 amps, and then take the entry battery A off-line. The list down to "BMAG #2-off" was to be turned off; it included the following: all cryo tank heaters and fans, G&N optics power, potable water heater, SPS line heater, SPS gaging, suit compressor, all fuel cell pumps, SMRC heaters, ECS radiator heaters, and SPS gimbal motors.

56:23 - Power AC bus 2 with inverter (INV) 1.- The crew was requested to power both ac busses with inverter no. 1. The primary purpose was to get telemetry data from oxygen tank no. 2 which is powered by ac bus 2 only.

56:24 - Turn fuel cell no. 2 pump on.- The crew had turned the pumps off in following the emergency powerdown list. The pumps circulate glycol and hydrogen for internal cooling in the fuel cells. They could have been left off for an hour or more, but fuel cell performance would have been degraded.

56:30 - Select main bus A power to RCS jet A-3.- The spacecraft was drifting in pitch without any apparent control. Quad C, which should have been controlling pitch, did not seem to be firing at all. To try to regain control in pitch, the quad A-3 jet was switched to main bus A power.

56:33 - Open circuit fuel cell no. 3.- Same reason as for open circuit fuel cell no. 1.

56:33 - Reconfigure quad B and D thrusters.- Flight control felt that a quad B thruster might be causing the spacecraft attitude deviations, and asked the crew to take off all power to the quad B jets. To compensate for quad B being off, all jets in quad D were selected to be powered from main bus A.

56:34 - Battery A taken off line.- The bus loads had been reduced sufficiently to allow fuel cell 2 alone to keep the bus voltage up. It was highly desirable to use the battery as little as possible, because there was no guarantee it could be recharged.

56:35 - Isolate the surge tank.- The crew was directed to isolate the CM oxygen surge tank. The purpose was to preserve an oxygen supply for reentry.

56:38 - Oxygen tank no. 1 heaters and fans.- Mission Control requested the crew to turn on the heaters in cryogenic oxygen tank no. 1 in an effort to build up the tank pressure. The current was observed to increase about 5 amperes, indicating they did come on. About 2 minutes later, since there was no increase in pressure, the crew was asked to turn on the fans in this tank.

56:45 - BMAG 2 off.- In an effort to further conserve power, the second BMAG was powered down.

56:51 - Turn off thruster C-1.- Thruster C-1 seemed to be firing very frequently without any apparent reason. The crew was requested to turn off all power to this thruster. The attitude disturbances were noted to have been virtually ended at about 56:40.

56:57 - Fuel cell no. 3 shutdown.- Fuel cells 1 and 3 had been open circuited earlier because they were not putting out any power. With the cryogenic oxygen leaking at its present rate, there would be no reactants for the fuel cells within a short time. Because there was a possibility that the oxygen was leaking down stream of one of the fuel cell reactant valves, it was decided to shut off these valves in an effort to save the oxygen remaining in tank no. 1. Fuel cell 3 was selected because it had been the first of the two to fail.

57:03 - Main bus A power to thruster A-4.- The crew was told to put power to thruster A-4 by connecting to main bus A. The spacecraft had a positive pitch rate and the crew was unable to stop it with quad C thrusters. With A-4 activated, pitch control was regained.

57:18 - Fuel cell no. 1 shutdown.- Shutting down fuel cell 3 did not effect the oxygen leak rate, so the reactant valves to fuel cell 1 were closed in an effort to try to stop the leak.

57:22 - Charge battery A.- The crew was directed to charge battery A. The fuel cell 2 was maintaining main bus A voltage at an adequate level to support the battery charger. Mission Control decided to charge battery A for as long as possible. Since the oxygen was still leaking, it was obvious that all fuel cell operation would be lost within about an hour.

57:29 - Disable power to quad C.- It appeared that quad C was not thrusting, although it was receiving firing signals. The explanation of this was that the propellant isolation valves had been closed by the "bang" at 55:55 and no propellant was being fed to the thrusters. Since these valves are powered by the main bus B, they could not be opened without getting power to this bus. The firing signals to quad C therefore were a useless drain of power on bus A, and the crew was directed to disconnect the thrusters from it.

57:39 - Fans on in oxygen tank no. 2.- In a final effort to try to increase the pressure in oxygen tank no. 2, the crew was directed to turn on the fans in that tank.

57:40 - LM power on.- The crew reported, "I've got LM power on."

57:49 - Stopped charging battery A.- In order to be ready to bring battery A on-line when fuel cell 2 failed, it was decided to terminate the charge. A total of about 0.75 amp-hours had been restored.

57:53 - CSM glycol pump off.- To reduce the main bus A loads, the crew was directed to turn off the glycol pump and to bypass the environmental control system radiators.

57:55 - Turn off oxygen tank no. 2 fans.- To further reduce the load on main bus A, the pumps in fuel cell 2 and the fans in oxygen tank no. 2 were turned off.

57:57 - LM data received.- Low-bit-rate telemetry data were received in the Mission Control Center at this time.

58:04 - Battery A on.- The crew powered main bus A with battery A in anticipation of the loss of fuel cell 2. The pressure in oxygen tank no. 1 was approximately 65 psi at this time.

58:07 - CSM communication reconfiguration.- The Command Module Pilot (CMP) was directed to turn off the CSM S-band primary power amplifier and to select low bit rate and down-voice backup. This was to reduce the load on battery A and maintain adequate circuit margins on the communication downlink.

58:18 - CSM guidance and navigation powerdown.- The CSM inertial platform (IMU) alignment had been transferred to the LM and verified by Mission Control. The crew was directed to turn off the CSM computer, the IMU, and the IMU heaters.

58:21 - Powerdown CM attitude control.- In an effort to reduce electrical power requirements in the CM, the CMP was directed to turn off "SCS Electronics Power," and "all Rotational Control Power Off." This completely removed all attitude control capability from the CM.

58:22 - LM RCS activation.- The LM crew was advised to pressurize the RCS, turn on the thruster heaters, and power up the attitude reference system.

58:27 - Activate "Direct" attitude control.- It was discovered that neither module was configured to provide attitude control. The quickest way to regain it was to have the CMP power up the rotational hand controller and the Direct coils.

58:36 - Fuel cell 2 shutdown.- The pressure of the oxygen being fed to this fuel cell had dropped below the operating level at 58:15 and it had stopped supplying current. As part of the CSM "safing," the fuel cell was disconnected from the bus and the reactant valves were closed.

58:40 - CSM powered down.- Battery A was disconnected from main bus A at this time, removing all power from the CSM.

Evaluation of Electrical and Cryogenic Oxygen Problems

The failure of fuel cell 3 resulted in the interruption of electrical power to several components in the spacecraft, including part of the telemetry signal conditioning. Main dc bus B was being powered only by fuel cell 3, so when its output dropped from about 25 amperes to less than 5 amperes, the bus voltage dropped from the normal 28 volts to less than 5 volts (fig. B5-2). Inverter no. 2, supplying power to ac bus 2, was being driven by main bus B and dropped off the line when the bus B voltage fell below about 16 volts. The bus failures, coupled with the cryogenic oxygen tank indications and some questionable instrumentation readings in fuel cells 1 and 3 (nitrogen and oxygen pressures), caused some initial uncertainty in the Mission Control Center.

The initial reaction was that there possibly had been a problem with major related instrumentation discrepancies. It was not clear that the telemetry quantities of cryogenic oxygen tank measurements or the fuel cell parameters were valid indications of conditions. For instance, the indication of no reactant flow and no fuel cell currents was compatible with fuel cells 1 and 3 having become disconnected from the main busses. Therefore, there was no reason to believe that they could not be reconnected. The lack of power output from the fuel cells could not be explained by the available information, i.e., the rapidity with which the fuel cells had failed. An additional factor that had to be considered was that the high-gain antenna had unaccountably switched from narrow to wide beam width at about this same time. Some trouble had been experienced earlier in getting this antenna to "lock on" in narrow beam width, and the possibility of a short in the antenna electronics could not be ruled out.

The first direction given to the crew was at 56:00 to return the bus power configuration to the normal operating mode; that is, fuel cell 1 powering bus A and fuel cell 3 powering bus B. The primary purpose of this direction was to get the spacecraft in a known configuration and determine if the fuel cells could be reconnected to the main busses. There are no telemetry parameters which show which fuel cells are supplying power to which busses, but the flight controllers were of the opinion that some reconfiguration might have been done by the crew.

In operating with split busses, that is, with two fuel cells powering main bus A and one fuel cell powering main bus B, the amount of equipment tied to bus A represents approximately twice the load as that to bus B. When fuel cell 1 failed, fuel cell 2 had to take up the additional load on bus A. In doing so, the voltage dropped to about 25 volts, which is low enough to cause a caution and warning indication. There was no particular harm in the bus voltage being this low, but if it dropped any lower the performance of some of the telemetry equipment would be affected and the flight controllers and crew were concerned. Normal bus voltage is above 27 volts, and the master caution and warning indication is triggered at 26-1/4 volts or less. Had fuel cell 2 been tied to both main busses as on previous missions, the total spacecraft current of 73 amperes would have driven both busses as low as 21 volts. The crew put entry battery A on bus A at 56:03 to bring the bus voltage up. Mission Control concurred in this action.

In an effort to obtain more data for troubleshooting the situation, the crew was asked to read out the onboard indications of oxygen pressure and nitrogen pressure in fuel cells 3 and 1, respectively. At 56:08 the crew was requested to disconnect fuel cell 1. This fuel cell was not supplying any power, so to disconnect it should have no effect on the bus voltage, but there was a possibility that it might give some different indications in the fuel cell telemetry parameters. There was no change in the fuel cell parameters when it was disconnected and the onboard readouts of nitrogen and oxygen pressure were the same as those on the ground, which did not add to the understanding of the situation.

Efforts to sort out the various telemetry indications and crew reports continued for the next several minutes. The next direction given to the crew was to proceed with the emergency powerdown of the electrical system, using page EMER 1-5 of the CSM Emergency Checklist which is part of the Flight Data File carried in the CSM (ref. 7). It was important to reduce the electrical loads to a low enough value for the single operating fuel cell to be able to supply all the necessary power. Mission Control was anxious to get entry battery A back off line to preserve as many amp-hours as possible.

The next step in the attempt to determine what was happening was to get power back to ac bus 2. Flight controllers considered powering ac bus 2 with inverter 3 driven from main bus A. Further consideration, however, led to the decision to simply tie ac bus 2 to inverter no. 1 which was already powering ac bus 1. Mission Control was interested in getting power to ac bus 2, since this is the only bus that powered the cryogenic oxygen tank no. 2 quantity and temperature telemetry. A temperature measurement was needed to confirm the zero pressure indication. The indications from oxygen tank no. 1 were that pressure and quantity were decreasing at a relatively high rate and it was imperative to immediately establish the condition of tank no. 2. It was not until after ac bus 2 had been powered up and oxygen tank no. 2 indicated empty, that the extreme seriousness of the situation was clear.

In proceeding through the emergency powerdown, the crew had placed the fuel cell pump switch to the "off" position in the one remaining good fuel cell; however, the pumps actually went off with loss of main bus B/ac bus 2 power. At 56:24, the Lunar Module Pilot (LMP) pointed this out to Mission Control, who in turn directed him to turn the pump back on. The only problem associated with leaving it off as much as an hour is that the fuel cell power output would start to degrade and no harm was done. But in the situation that existed, it is not inconceivable that had the crew not advised Mission Control of the fuel cell pump being off it would have been overlooked until a rise in the fuel cell 2 loop temperatures gave this indication.

Further direction in the management of the electrical system was not given until about 56:33. At this time the crew was directed to open circuit fuel cell 3 for the same reason as fuel cell 1 was open circuited earlier. At 56:35 the crew was requested to isolate the surge tank and at approximately this same time Mission Control also directed the crew to remove battery A from main bus A. The emergency powerdown had resulted in a load reduction such that the fuel cell alone could maintain bus voltage above 27 volts.

It had become apparent that the operation of fuel cells 1 and 3 probably could not be regained under any condition, and that with oxygen tank 1 quantity decreasing at its then present rate, the service module would soon become incapable of providing any life support or electrical power. The heaters and fans in this tank were turned on at 56:38 in an effort to increase the pressure, but to no avail. Because there was a possibility that a rupture had occurred in one of the inoperative fuel cells and the oxygen was leaking through it, Mission Control decided to shut down the cryogenic inputs to fuel cell 3 to see if this would stop the leak, and the reactant valves to it were closed at 57:00. It should be pointed out that this is an irreversible step; once a fuel cell is shut down, it cannot be restarted in flight. Fuel cell 3 was shut down first since its internal oxygen pressure indication was zero; there was no change in the oxygen tank pressure decay rate, however, and the reactant valves to fuel cell 1 were closed at 57:18, with equally negative results. Mission Control made one last attempt to increase oxygen pressure by directing the crew to turn on the fans in tank no. 2. At about 57:22, the crew was directed to initiate charging of battery A. By this time it became clear, with the leaking oxygen tank no. 1, that fuel cell 2 could continue to operate only for a short period of time. Since the fuel cell was maintaining an adequate bus voltage and could provide the additional power to operate the battery charger, it was decided to charge battery A as long as possible. The charging of battery A was stopped after 22 minutes. At this time the oxygen tank no. 1 pressure had decayed to a point where continued operation of fuel cell 2 was questionable. Battery A was to be connected to main bus A at the first indication that the output of the fuel cell was decaying. Since the battery cannot be connected to power a bus while it is being charged, it was necessary to terminate the charging in anticipation of the fuel cell failure.

In preparation for using the entry battery to power main bus A, a further reduction of the loads on this bus was performed. The following equipment was turned off: glycol pump, oxygen tank no. 2 fans, and fuel cell no. 2 pumps.

The pressure in oxygen tank no. 1 was approximately 65 psi at 58:04 when the crew connected battery A to main bus A. This is below the minimum operating pressure for the fuel cell. This battery continued to power main bus A until about 58:40. By this time, the LM had been activated and the inertial platform alignment transferred from the command module.

The attempts to determine the cause of the problem in the electrical power system were confused by the misleading symptoms that resulted from the cryogenic tank failure. The failure in the electrical power system and cryogenic oxygen was so massive that by itself it would have created some initial confusion and made the flight controllers skeptical of the data, but in addition to fuel cell output dropping to zero and bus voltages dropping to zero, there were other indications that had to be considered. The attitude excursions (now presumed to have been caused by escaping oxygen) and the peculiar RCS thruster firings added to the confused situation. The RCS problems are discussed in more detail in the following section, but regardless of how quickly the problem in the electrical power system was resolved, there was nothing that could have been done to correct it. The only thing the crew and Mission Control could do under the circumstances was to preserve as much capability as possible for reentry and to power down in an orderly manner to allow time for LM activation.

Maintenance of Attitude Control

Within 3 minutes after reporting the large bang, the Commander (CDR) reported some of the "talkback" indicators for the service module reaction control system (SMRCS) were showing "barberpole." His report indicated that the helium isolation valves to quads B and D were closed, and the secondary propellant fuel pressurization valves to quads A and C were closed (fig. B6-1). These valves have a history of inadvertant closure when the spacecraft is subjected to a large "jolt" in flight, such as the spacecraft separation from the S-IVB. This phenomenon was first encountered on Apollo 9. To reopen a valve that has closed in this manner, it is necessary to cycle the position selector switch to "close" and then back to the "open" position. All of the switches in this system have momentary "open" and "close" positions, and are springloaded to a center neutral position.

The valve position indicators in the spacecraft are the flag type which show gray when the valve is open and gray-and-white stripe ("barberpole") when closed; there is no telemetry indication of the valve position. Each valve and its respective indicator are powered from the same main dc bus and cannot be selected to the other bus. The valves in the propellant system for quads B and D are powered from main bus A and quads A and C are powered from main bus B. Therefore, there was no way to determine the status of the RCS propellant and pressurization systems of quads A and C, and there was no way to reposition the valves without powering up main bus B. The ability to open the isolation valves in quads B and D was not affected by the loss of main bus B.

Jet-firing signals, received at each individual thruster, open fuel and oxidizer valves by energizing a coil. There are two coils at each thruster. One, called "Auto," receives its signal from either the computer or the two rotational hand controllers (RHC's) and can be powered from either main dc bus, selected by the "Auto RCS Select" switches. There are 16 switches; one for each individual thruster that can be positioned to "off," "main A," or "main B." The other coil at the thruster is called "Direct" and receives its signal from the rotational hand controllers when they are rotated sufficiently far from the null detent. There are several ways of configuring the RHC's to power the Direct coils. Each RHC is limited as to which main bus and thruster combination it can be tied. Typically, the RHC's are powered so that half the jets are fired by main bus B and the other half by main bus A. As per normal procedure, the auto RCS select switches were configured so that single-jet authority in roll, pitch, and yaw attitude control would be available without reconfiguring if either main bus were lost. This protection can only be obtained if all four quads are functional. The loss of capability resulting from the failure of a main bus would be compounded by the concurrent closing of propellant isolation valves. Control about one or more axes would be lost

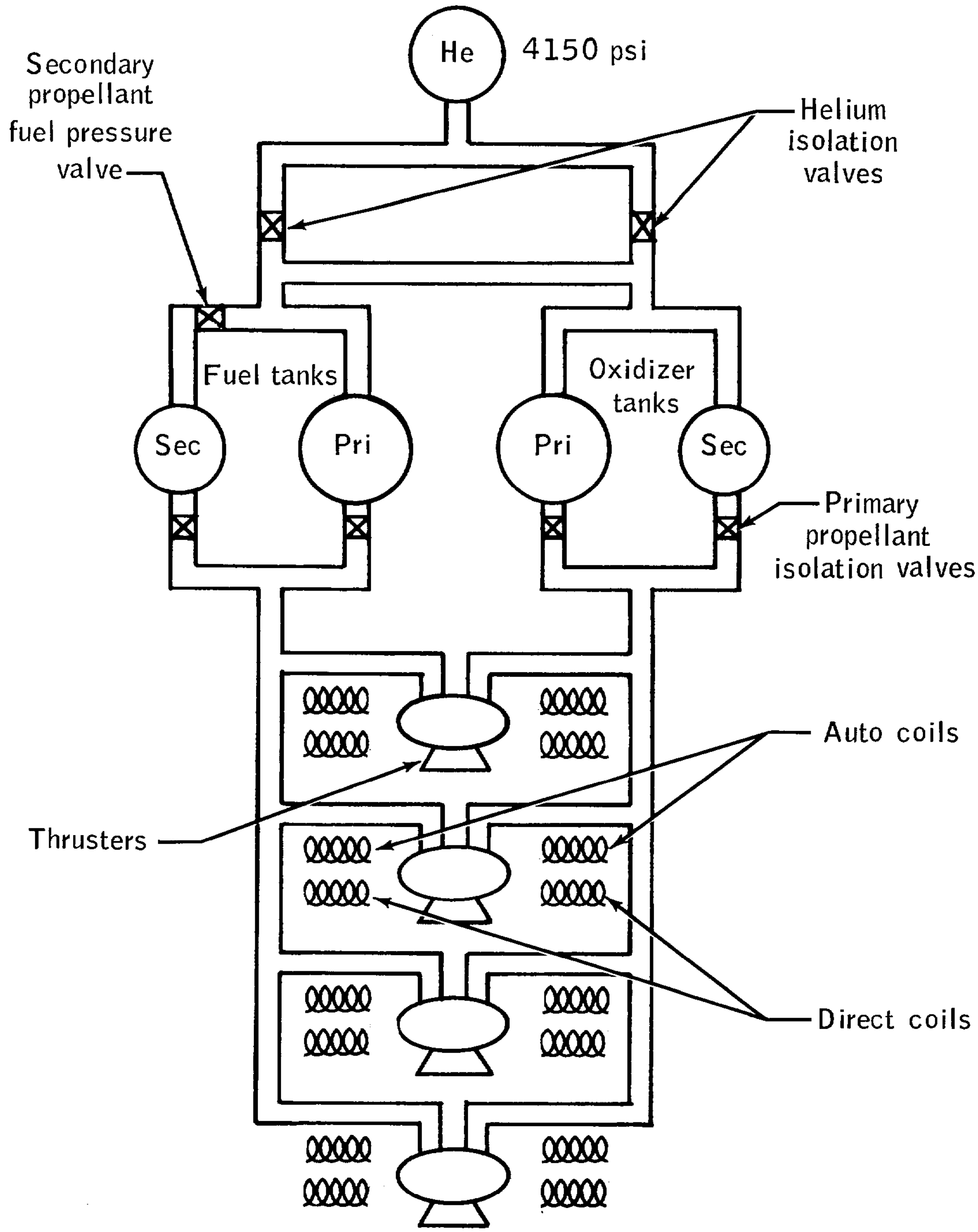


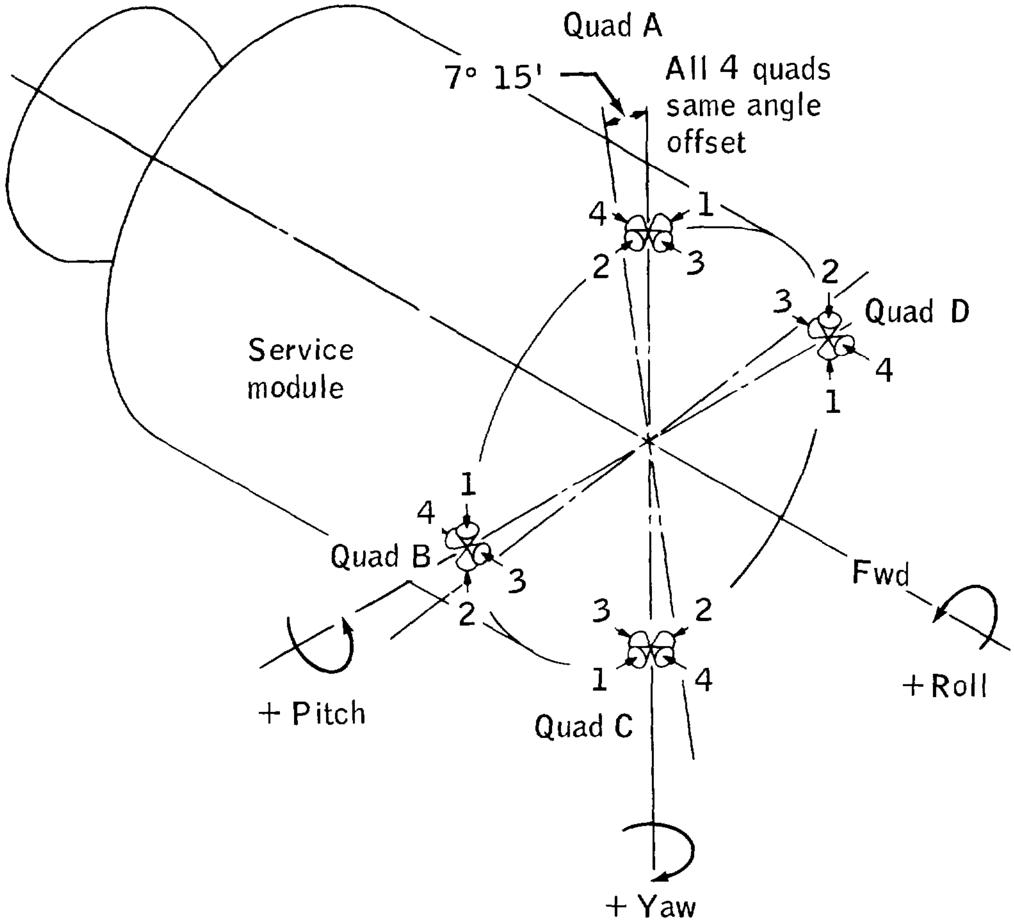
Figure B6-1.- Service module reaction control system one quad (typical of all four).

until some reconfiguration could be accomplished. Because power to the talkback indicators would also be lost, it would take some effort to determine the status of the control system.

At the time of the accident, the spacecraft was performing a computer-controlled roll maneuver and maintaining pitch and yaw attitude hold. The digital autopilot began firing RCS thrusters to counteract the attitude perturbations presumably caused by the oxygen tank no. 2 failure, and attitude was completely controlled until main bus B was lost. Soon after the loss of main bus B, Mission Control noted the spacecraft began to rotate about the pitch and yaw axes. It was also noted that the fuel and oxidizer pressures in quad D were decreasing and the crew was asked to verify that they had opened the helium isolation valves which had previously been reported as closed. Although the crew did not acknowledge this request, the pressures were observed to increase to the normal operating values shortly thereafter. The pressures had decreased in this quad because the helium pressurization valves had been jolted closed and subsequent firings of the thrusters had used some of the propellant. This increased the ullage volume and resulted in a noticeable decrease in tank pressures. The flight controllers correctly diagnosed the cause and were not mislead into thinking the tanks were leaking.

At 56:07 Mission Control noted that the crew had turned off all Auto RCS Select switches, because they were concerned that unwanted thruster firings were causing the continuing spacecraft attitude excursions. At about 56:13 the spacecraft was observed to be approaching gimbal lock of the inertial platform. Gimbal lock is a condition in which the inertial platform loses its reference alignment. To prevent a gimbal lock, the spacecraft attitude relative to the inertial platform must be kept out of certain regions. Mission Control advised the crew of this situation, and in an effort to achieve positive control about all axes of the spacecraft, the crew was directed to reconfigure the RCS Auto Select switches for thrusters 3 and 4 in quad B and all thrusters in quad C to be powered from main bus A. This would provide single-jet control authority about each axis (fig. B6-2). The other jets were not switched to main bus A power in order not to drag down the main bus A voltage any more than necessary. The LMP acknowledged and the drift toward gimbal lock was arrested, although all rotations were not stopped.

At 56:22 the CMP reported that the spacecraft was being subjected to pitch and yaw rates and that he had to use direct control with the rotational hand controller to stop them. The rates would start to increase again as soon as he stopped the direct control. He asked if the ground could see any spurious jet firings that might be causing the rates. Although the data available in Mission Control were not complete (the position of the propellant system valves in quads A and C was unknown and firing signals to the Direct coils are not on telemetry), it appeared to the flight controllers that the jet firings were not causing the



Axis	Direction	Thruster
Roll *	+ –	A-1, B-1, C-1, D-1 A-2, B-2, C-2, D-2
Pitch	+ –	A-3, C-3 A-4, C-4
Yaw	+ –	B-3, D-3 B-4, D-4

* Autopilot can be configured to use quads B and D for roll or A and C.

Figure B6-2.- SM RCS quad location and thruster numbering system.

spurious rates. It was observed that thruster 3 in quad C was receiving firing signals almost continuously, but was having no success in stopping the negative pitch rate. In an effort to gain control over the negative pitch rate, at 56:32 Mission Control requested the crew to put the Auto RCS Select switch of thruster 3 of quad A on main bus A. It was suspected that C-3 thruster was not really firing because there was no perceptable reduction in quad C propellant.

At about 56:35 the crew was requested to remove all power from the quad B thrusters auto coils and to power all quad D thrusters from main bus A. This request was made in an effort to determine if quad B thrusters were causing the unwanted pitch and yaw rates. Mission Control continued to monitor the RCS thruster firings and the spacecraft attitude response, trying to determine the status of the system. During the next 10 minutes, the crew pointed out that the quad temperature indications for A and B were out of the normal operating range, and Mission Control assured the crew that they were within acceptable operating limits. In this same time period the ground had noticed numerous firing signals of thruster C-1. Since the flight controllers could see no explanation for this, the crew was requested to remove all power from the C-1 auto coil at 56:53. About 10 minutes later, the CMP reported no negative pitch capability, and requested clearance to enable thruster A-4. Mission Control responded immediately to "bring A-4 on," and the pitch rate was stopped within a few seconds. At 57:20, Mission Control noted a discrepancy in the roll control jet configuration. The autopilot was configured to use quads A and C for roll control, but the auto coils for these jets were turned off. The crew was directed to configure the autopilot to use quads B and D for roll control.

Based on a close observation of firing signals to quad C and the resulting spacecraft response, the flight controllers thought that the quad C propellant isolation valves had been jolted closed by the incident that caused the loud bang. The computer was still sending firing signals to the auto coils, but they were apparently having no effect and propellant was not being used by this quad. Therefore, to save the small amount of electrical power that was being spent by sending firing signals to the coils, at 57:29 Mission Control directed the crew to turn off the auto coils to this quad.

Complete attitude control appeared to be established at this time and all further attitude control support to the CSM was directed toward transferring control to the LM. The overall LM activation support is described in more detail in the following section; however, establishment of the attitude control of the LM is briefly summarized as follows:

1. Mission Control referred the crew to specific pages in the LM Activation Checklist (part of the Flight Data File, ref. 7) for the

procedure to transfer the inertial platform alignment from the CSM to the LM.

2. The CMP was directed to power down all of the guidance, navigation, and control systems after the LM platform had been properly aligned.

3. Mission Control assisted the LM crew in getting attitude control established by pointing out specific circuit breakers that needed to be closed and switches that needed to be positioned.

It was approximagely 1-1/2 hours after the initial incident before complete automatic attitude control was established, although the crew had manual control capability at all times. The information on the ground was incomplete and was confused by the intermixing of automatic control and manual direct control. Furthermore, the major concern was the electrical and oxygen problems, and the only mandatory action in the control system area was to maintain a safe posture in the systems and avoid gimbal lock. These mandatory tasks were accomplished and in due time complete attitude control was established.

Lunar Module Activation

It was recognized at about 45 minutes after the accident that the LM might have to be used to provide the necessary life support, and the LM activation was started about 1-3/4 hours after the crew first reported the loud bang in the CSM. The first hour and 45 minutes were spent in regaining positive attitude control in the CSM, in troubleshooting the electrical problems in the CSM, and in attempting to halt the loss of oxygen from the service module. Since LM activation did not begin until the lifetime of the one functioning fuel cell was predicted to be about 15 minutes, there was a strong motivation to complete the LM activation and CSM powerdown as soon as possible.

The first order of business for LM activation was to get electrical power and the communications sytems operating. A specific procedure for this was read to the LMP at 57:37. Although three checklists for LM activation were available as part of the Flight Data File in the spacecraft, Mission Control did not direct the crew to follow any of them. These checklists were designed for three different situations at LM activation. The first, entitled "Apollo XIII LM-7 Activation Checklist" (contained in ref. 7), contains the nominal mission sequences from initial LM manning to undocking prior to the lunar landing. The other two activation checklists are in the "LM Contingency Checklist" (contained in ref. 7). They were written to cover the situations of having to use the LM to perform an Earth-return abort maneuver for the docked CSM/LM configuration. One checklist includes activation of the primary guidance and navigation system (inertial platform alignment, etc.) and is called the "2-Hour Activation List" because it was designed to be completed at a comfortable pace in time to execute a descent propulsion system maneuver in 2 hours elapsed time. The other contingency list is called the "30-Minute Activation List," and serves the same purpose, except that many steps, including the G&N activation, are omitted. There was no LM activation checklist available which was designed to cover the specific situation resulting from this incident. The features that were different are as follows:

1. The need to get the LM totally activated as soon as possible--including attitude control as well as supplying life support, communications, and electrical power.

2. The desire to power down the CSM as soon as possible in order to preserve all available battery power for reentry.

3. The LM was to serve as a "lifeboat" supplying oxygen, water, electrical power, and attitude control for 80 or 90 hours.

This presented a paradoxical situation in which almost total LM capability was required, but at the same time its consumables had to be conserved as much as possible. In responding to the situation, the flight controllers referred the crew to specific pages in the normal "LM Activation Checklist," augmented with additional instructions. The purpose was to bypass all steps that were not absolutely necessary for getting the LM power, communication, and environmental control system in operation. The total instructions given to the crew referred to only 4 pages of the 59 in the checklist. There were three single instruction additions to this shortly afterward which completed the LM configuration for supplying oxygen to the cabin. Although this particular contingency had never been simulated in the training exercises in preparation for the mission, similar cases had been considered, and Black Team personnel, including the Flight Director, Glynn Lunney, had prepared procedures and criteria for using the LM to augment the CSM. The simulations had been limited to cases where the LM ascent stage was to be retained following rendezvous in lunar orbit. These same personnel had participated in these simulations for the preceeding missions of Apollo 10, 11, and 12, and therefore were familiar with the problems.

The next procedure given to the crew was designed to get the LM guidance and navigation system operating and to get the LM inertial platform aligned to a known reference. Again, Mission Control referred the crew to specific pages in the "LM Activation Checklist," along with certain necessary circuit breaker closures which were not listed on those pages. Although the necessary circuit-breaker panel configuration for LM activation is shown on two pages in the checklist, the crew was not referred to those pages by Mission Control. In order to save time, only the necessary circuit breakers were given as part of each set of special instructions. The omission of a necessary circuit breaker closure later caused some delay in establishing LM attitude control.

Throughout this period of LM powerup, the CMP was given frequent instructions on the CM configuration to reduce power requirements. The crew completed an alignment of the LM IMU to the CSM IMU at 58:09. The platform gimbal angles for both spacecraft were read to the ground for computation of the fine-align torquing angles for the LM. As soon as the LM IMU was aligned, the CMP was directed to power down the CM computer and the IMU, including the IMU heaters.

At about 58:17 the temperature of the coolant loop in the LM began to rise and the LM crew was advised to activate the sublimator, referring to the appropriate page in the "LM Activation Checklist." During the next 2- to 3-minute period there was an unusually high density of conversation, both in the Mission Control Center and on the air-to-ground frequency between the CAPCOM and crewmen in both spacecraft modules. The CMP reported powering down the CM control system; the CDR reported he had no attitude reference system and requested permission to "close the FDAI circuit breakers so we could have a ball to see if we go to gimbal lock"; both the CMP and the LMP reported conditions and asked questions regarding configuration items; and on the ground the CSM flight controllers were trying to get their systems powered down as much as possible while the LM flight controllers were trying to "get through" to the LMP to pressurize the LM RCS and to turn the thruster heaters on.

At approximately 58:21, the CMP was told to continue his powerdown by turning off the power to the rotational hand controller almost simultaneously with the LM crew being directed to power up the FDAI and the RCS heaters, pressurize the RCS, and open the main shutoff valves. After about 5 minutes, when it became clear that neither spacecraft had control of the attitude, the CMP was directed to reactivate the CSM Direct attitude control capability. This was done and the LM crew then proceeded, following instructions from the ground, to pressurize the RCS and to perform the steps necessary to get the attitude reference system operating in the LM. Mission Control at 58:32 gave the LM crew the inputs for the onboard computer which set the proper system gains for the LM autopilot to control the docked spacecraft configuration. The LM achieved complete automatic attitude control capability at 58:34, when the crew received direction from Mission Control to close an essential circuit breaker that had been previously overlooked. The position of this circuit breaker is not indicated on telemetry, but the flight controller correctly diagnosed the problem when the crew stated they still did not have automatic control at 58:33.

After it was definitely established that the LM had attitude control, the CMP was given final instructions for completely powering down the CM, and work toward getting the LM configured for the long trip home proceeded. Mission Control gave the crew the LM IMU torquing angles to get the platform fine aligned to the reference orientation. Discussions were held between the ground and the spacecraft concerning the ability of the crew to use the stars as a reference for platform realignment. It was concluded that this would be difficult if not impossible to do, and the current alignment should be preserved until after the abort maneuver.

An abnormally high pressure reading was noted in one of the LM ascent stage oxygen tanks shortly after telemetry data were received in Mission Control, and the crew was directed to use oxygen from this tank instead of the descent tank. Later it was diagnosed that the shutoff valve leaked, allowing the higher pressure oxygen from the manifold to leak into this ascent tank. The condition in itself was not a problem; the net effect was that this ascent tank was raised to a slightly higher than normal pressure which was well within the tank limits. This degraded the system redundancy, however, and had a subsequent leak developed in this tank, the LM oxygen supply would have been depleted (fig. B6-3).

The next phase of activity was devoted to reducing the power drain from the LM batteries to as low a value as practical. This included turning off many of the displays in the LM and put Mission Control in the position of monitoring system parameters for the crew. The crew was also given all the information required to execute a return-to-Earth abort maneuver 2 hours after passing the point of closest approach to the Moon (pericynthion). Providing this data well in advance is a normal procedure which gives the crew the capability to perform the abort if communications are lost with the ground.

PLANS AND ACTIONS TAKEN TO RETURN THE CREW TO EARTH

After the crew had powered down the CM and activated the LM, the immediate situation had stabilized, and Mission Control could direct its full resources to the long-term problem of getting the crew safely home. The first item of concern was to determine an expected LM consumables lifetime and to develop a trajectory plan that would return the spacecraft to Earth within this lifetime. Also it was mandatory to reduce the expenditure of battery power and water as much as practical.

Subsequent efforts by Mission Control in support of the crew were varied and extensive. Much of this activity, however, is normally part of the routine functions of Mission Control. Such items as monitoring systems performance via telemetry parameters; keeping accurate records of consumables usage, and predicting future consumption rates; scheduling crew rest periods; and orbit determination are only some of the examples of this normal activity. However, only the special activities which were unique to this mission failure or which were of major importance to the

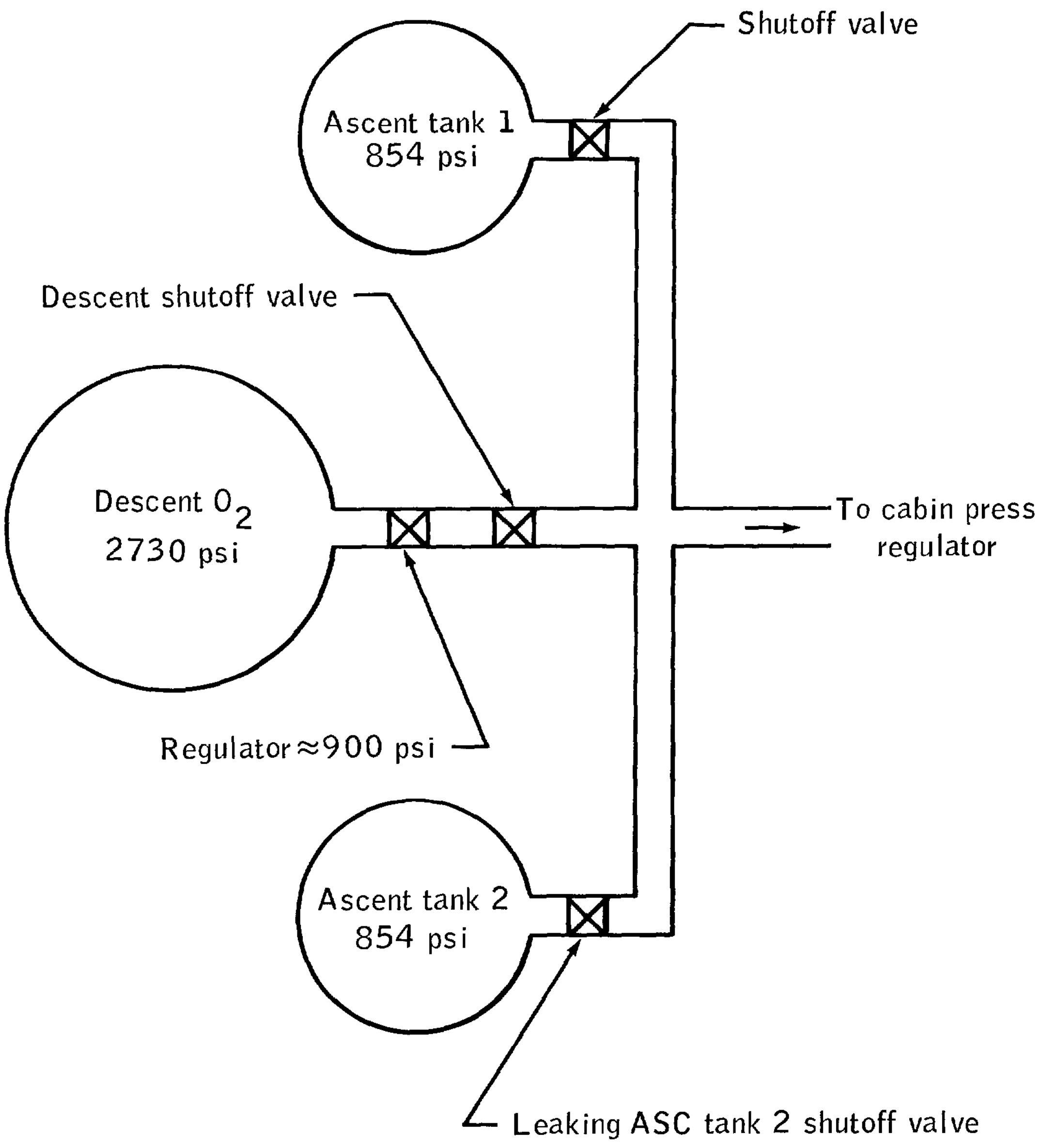


Figure B6-3.- Schematic of LM oxygen storage system.

successful return of the crew will be described. These activities are grouped in three categories in this report and described as independent subjects. These categories are consumables and system management, return-to-Earth trajectory control, and definition of procedures and checklists for reentry preparation. No attempt is made to describe the events chronologically. The Mission Operation Report (ref. 5) contains a comprehensive documentation of these events.

Consumables and Systems Management Actions

Consumables and systems management of both the LM and the CM were of vital importance and generated much activity in Mission Control.

Lunar module.-

Electrical power system: All LM electrical power is supplied by batteries. There are four in the descent stage with a total rated capacity of 1600 amp-hours and two in the ascent stage with a total rated capacity of 592 amp-hours. After the LM activation, analyses of power requirements and lifetime capability were completed. These analyses showed that after the abort maneuver at 61:30, the LM could be powered down to a total current requirement of about 27 amps and still keep the inertial platform aligned. This was extremely important because it made it possible to perform a guidance-controlled abort maneuver at 79:30 which could be used to reduce the return time back to Earth from 152 hours to 143 hours g.e.t. The analyses also indicated that if the guidance system was completely powered down after 79:30, the total power requirement could be reduced to about 17 amps, stretching the battery lifetime to approximately 165 hours g.e.t. This was a comfortable margin, even if the return time could not be reduced below 155 hours.

The flight controllers provided the crew with a list of specific switches to close and circuit breakers to open which would reduce the electrical load to the minimum possible consistent with safe operation. The fact that virtually all of the onboard displays were turned off is an indication of how extensively the spacecraft was powered down. Mission Control kept an accurate account of the switch and circuit breaker configuration, and was able to insure that the necessary equipment was powered up again when the subsequent trajectory maneuvers were made. The full powerdown configuration actually required only 12 amperes, instead of 17. The basis for this powerdown was contained in the LM Contingency Checklist (ref. 7). The Emergency Powerdown Checklist was developed for the case of the LM in lunar orbit awaiting rescue by the CSM. Some additions to this listing of turned-off equipment were made by Mission Control.

As soon as the electrical power system configuration was established and apparently performing well, Mission Control began planning for what actions to take if a LM battery failure were to occur. These plans included listing the few remaining items of equipment which could be taken off line in the powered-down condition. Since the current was already down to less than 17 amperes, there was not much left that could be removed except the communications equipment, but certain equipment could have been operated on a periodic basis rather than continuously. A schedule for this kind of operation was planned in case it became necessary.

At 97:14:26 the LMP called Mission Control to report an anomaly that he had observed in the LM. This anomaly was a "little thump" that was heard but not felt, and it seemed to come from the vicinity of the LM descent stage. The LMP also observed a "new shower of snowflakes come up that looked like they were emitted from down that way." The venting appeared to be going radially outward, perpendicular to the X-axis in the +Y, +Z quadrant, and it continued for approximately 2 minutes. Neither the flight controllers nor the LMP observed any anomalous behavior in the data. The LMP closed the essential display circuit breakers in order to scan his instruments. The flight controllers searched the various displays of telemetry data. Since no unusual readings were noted, the investigation of the "thump" incident was not pursued further at that time. A postflight review of the data indicates that at about the time of the "thump," a large, momentary increase in LM battery output occurred. The surge was of 2 to 3 seconds duration, and was experienced by all four descent batteries. The behavior of the four battery currents is summarized in the table:

Battery	Current output, amps		
	Before surge	Peak	After surge
1	3	37.5	3
2	2	Off-scale high 60 amps	6
3	3	36.8	1
4	3	30.5	1

The MSC investigation of this anomaly is still in progress, and the exact cause of the current increase, the "thump," and the venting is not known. It does appear that they were all related, but not connected with the previous service module failure.

At 99:51 g.e.t. a descent battery no. 2 malfunction warning light illuminated. Because the display system on board was powered down except for the caution and warning panels, the analysis of the problem was done in Mission Control where telemetry was available. There were three possible valid causes of the warning light: an overcurrent, a reverse current, or a battery overtemperature condition. The troubleshooting systematically eliminated all three, and Mission Control concluded the problem was a faulty temperature sensor. The crew was advised to reconnect the battery about an hour later. No problems with the battery ever developed, but the sensor indication later became erratic, causing several MC&W alarms. A plot of total usable amp-hours remaining in the LM batteries is contained in figure B6-4.

Coolant system: It was as essential to power down the LM as much as possible in order to reduce the cooling requirements as it was to reduce the battery amp-hours expended. The LM coolant loop uses the action of ice sublimination to take heat away from the spacecraft. Feed water for the sublimator is stored in tanks, and the rate of water usage to provide this cooling is proportional to the amount of electrical power expended because of the heat generated. The analysis showed that for the above-mentioned electrical power requirements, the LM water supply was most critical and would be depleted about 155 hours g.e.t. This analysis was based on data obtained several hours after the initial LM activation. Estimates based on the usage rate immediately after activation indicated the LM would be depleted of water by 94 hours g.e.t. As expected, the rate reduced drastically, however, after the initial cooling down was accomplished.

During the mission period before the postpericynthion abort, when the spacecraft was on a trajectory with a 155-hour g.e.t. landing time, efforts were made to find a method of increasing the LM water margin by means other than a further powerdown. Two procedures were developed as a result of this effort. The first allowed the crew to get drinking water from the CM potable water tank, and the second was a method of transferring water to the LM tanks for use in the LM coolant loop. The latter procedure involved the use of the portable life support systems (PLSS) water tanks as an intermediate container for transporting the water from the CM waste tank. Although it did not become necessary to use the second procedure, it was tested on the ground by engineering personnel at MSC, and was available in Mission Control. A plot of the usable water remaining in the LM is shown in figure B6-5.

Oxygen supply and carbon dioxide removal: The oxygen supply in the LM was adequate for more than 200 hours g.e.t., and was of no concern (fig. B6-6). This included a supply in the systems normally used for the lunar extravehicular activity (EVA). The initial problem with the ascent oxygen tank 2 had stabilized to the condition that the pressure in the tank was about 100 psi above the normal operating range. Engineering support personnel had advised Mission Control that this was no problem, and no further actions were taken in this area.

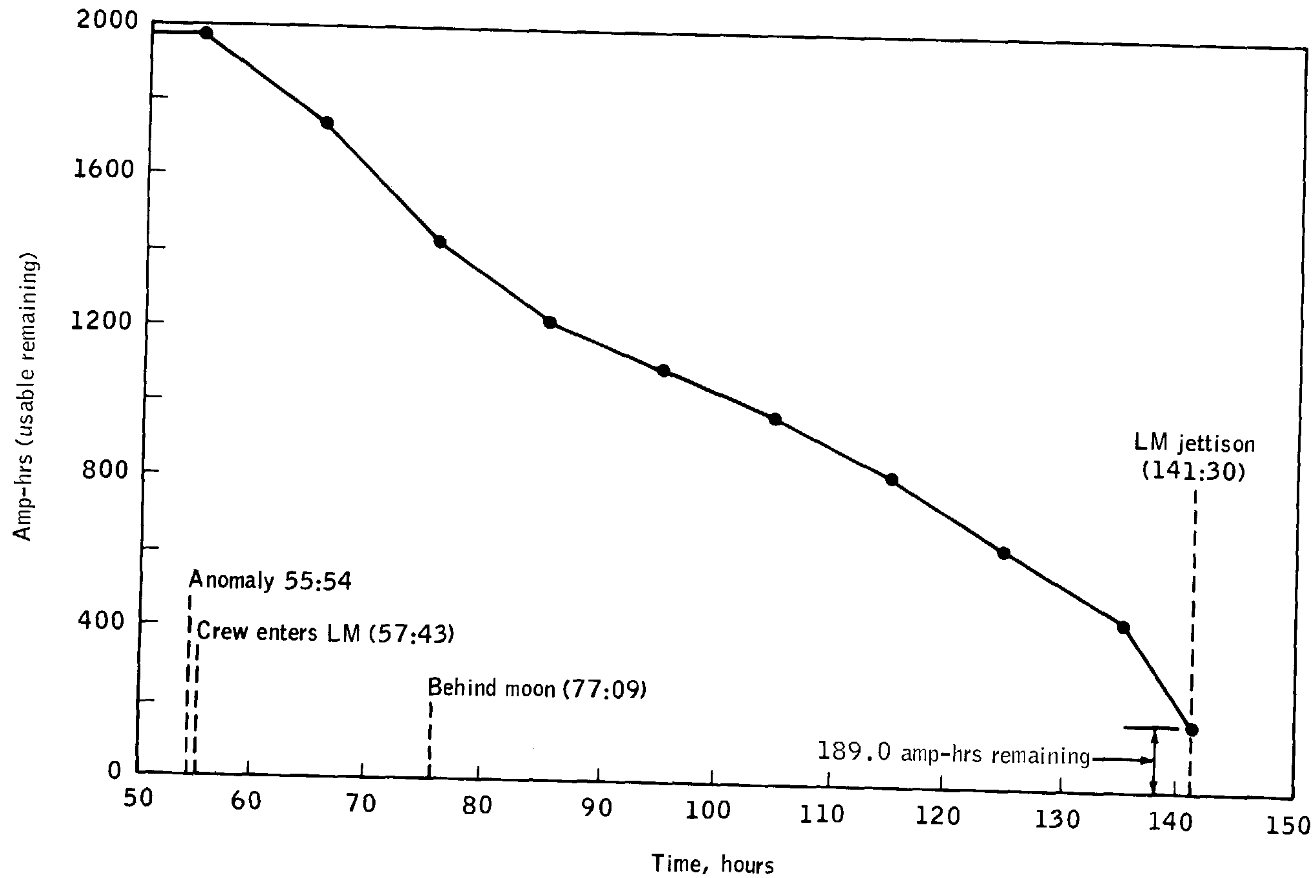


Figure B6-4.- Electrical power system consumables status.

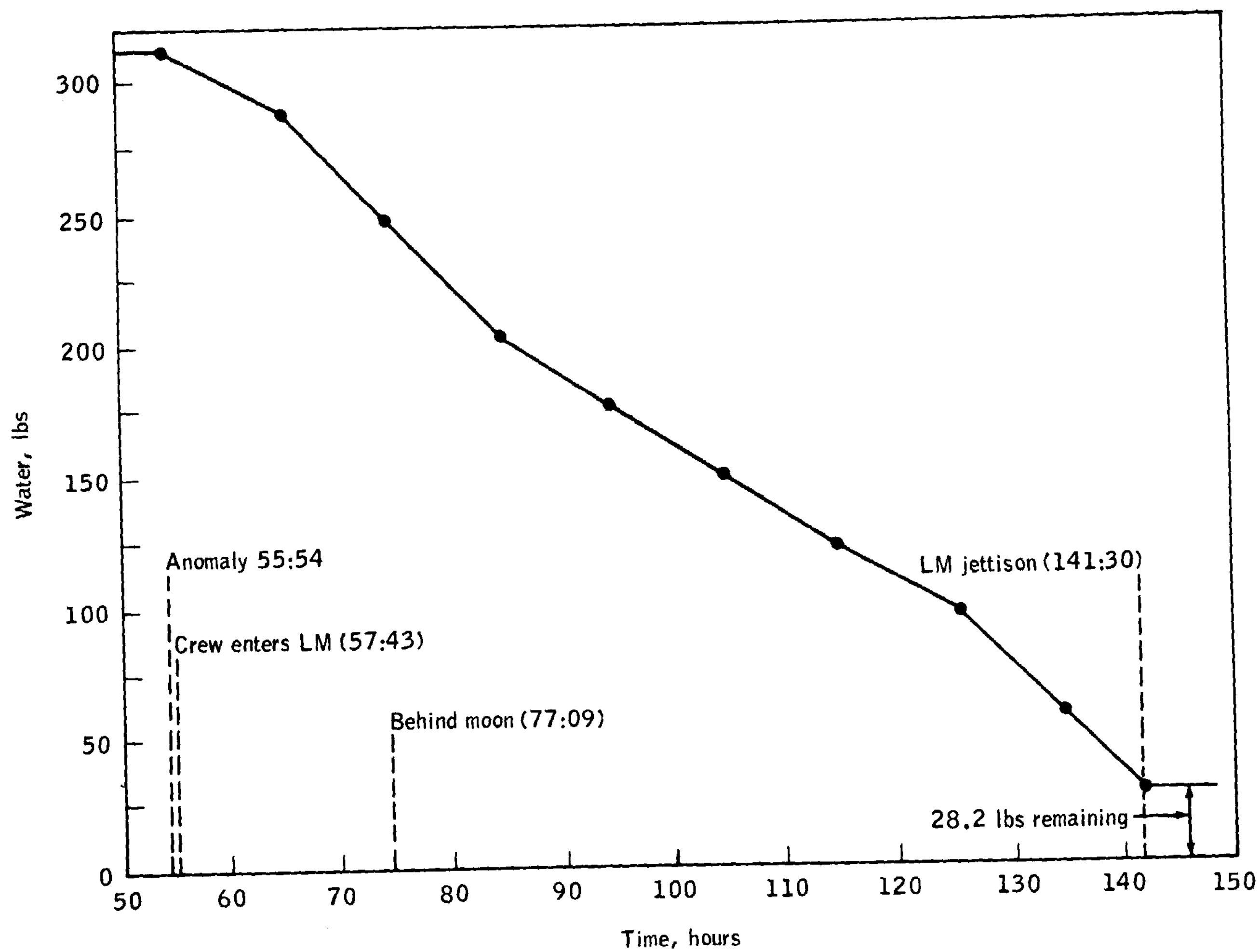


Figure B6-5.- Usable remaining water.

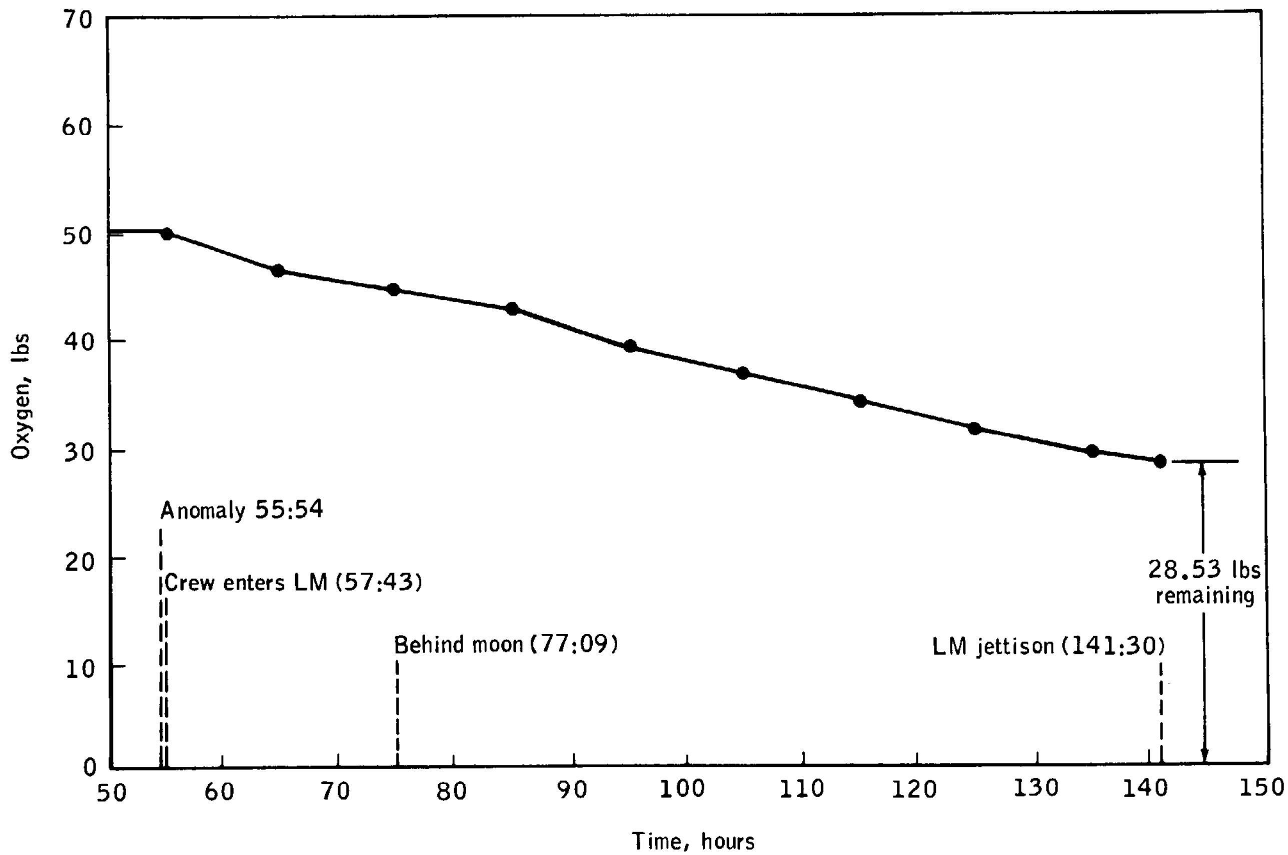


Figure B6-6.- Usable remaining oxygen.

The problem of removing carbon dioxide from the cabin oxygen was a serious one. The LM, like the CSM, uses lithium hydroxide (LiOH) cartridges to scrub the recirculated oxygen to remove odors and carbon dioxide. The LiOH cartridges are rated for a specified total man-hours capacity, and eventually must be replaced when they become saturated. The LM cartridges were not adequate for carbon dioxide removal for three men for the duration of the Earth-return trip. There were more than adequate cartridges in the CM, but they would not fit in the LM canisters. There were several methods suggested for solving the problem, including powering up the CM system to circulate cabin oxygen through its LiOH canisters. The method that was actually used was developed by Crew Systems Division personnel at MSC. It consisted of using tape, flight data file cards, and plastic bag material to connect the CM LiOH canisters to the LM oxygen circulation system. The crew implemented the modification and it worked very well. The partial pressure of carbon dioxide reading indicated by the onboard gage dropped rapidly from 8mm Hg to 0.1mm Hg soon after the rig was completed at 94 hours g.e.t. The modification was not tried until this time in order to get maximum use from the LM cartridges. About 20 hours later, the carbon dioxide partial pressure reading had increased to 1.8mm Hg, and a procedure for putting two additional cartridges in series to those in the CM canisters was given to the crew. This procedure was also developed by engineers at MSC (fig. B6-7). After this second modification was completed, the carbon dioxide partial pressure remained below 2mm Hg for the rest of the mission, without any further modifications necessary.

The modifications to the oxygen circulation systems were evaluated in the simulators at MSC before they were accepted by mission operations personnel. This included tests in the pressure chamber. As mentioned earlier, there were other methods that could have been adopted had this one proved to be unacceptable.

Reaction control system: The LM reaction control system (LMRCS) propellants were another consumable that had to be managed carefully. Maintaining attitude control of both the CSM and the LM, with a total weight in excess of 90,000 pounds, can be done by the LMRCS, but is a particularly taxing job. The LM control system was not designed to perform this task, and does not do it efficiently in terms of propellant expenditure. This was aggravated by the fact that there is some control moment loss and some cross coupling when the LM is in control due to thrust plume deflectors designed to protect the LM descent stage from extended thruster firings.

Shortly after the LM assumed attitude control, Mission Control gave the crew a procedure which increased the attitude excursion tolerance in the computer. This increased the attitude error tolerance and caused

Figure B6-7.- View of CM LiOH cannister modification as installed in the LM.

less thruster firings to be commanded by the computer which was maintaining automatic attitude control. The simulators at KSC and MSC were used to evaluate different techniques for maneuvering the spacecraft under manual control as well as automatic. Manual maneuvers became necessary after the LM inertial platform and computer were powered down after the post-pericynthion abort maneuver. Backup and support crews performed the evaluations and recommended certain techniques.

Mission Control kept a close watch on the RCS propellant consumption and was prepared to have the crew revert to an uncontrolled, drifting flight mode if necessary. This would have been requested if the RCS propellant decreased below the "red line" value. The flight controllers had computed a "red line" which provided enough propellant for meeting the midcourse correction maneuver requirements and the requirements to maneuver in preparation for the reentry sequence.

Command service module.- After the CM powerdown at 58:40 there was very little system management that could be or needed to be done. The electrical power system, however, did require some attention. The first action was to get the CM into a known configuration. So much had happened so quickly during the period following the accident, that neither the crew nor Mission Control had a complete knowledge of the switch configuration in the CM. Therefore, a checklist was developed which listed the desired position of every switch, circuit breaker, and actuator handle in the spacecraft. The lift-off configuration in the CSM launch checklist portion of the Flight Data File served as the baseline for this list, and the modifications were read to the crew. The crew then configured the CM as defined by this list.

The next task was to determine the status of main dc bus B. Because power had not been applied to the bus since the failure of fuel cell 3 at 55:58, it was not certain that a major short did not exist on it. Mission Control defined a procedure which used entry battery B to apply power to the bus. The procedure contained 12 steps, and the displays the crew should monitor were defined, along with the expected indications. The baseline configuration described in the preceding paragraph insured that all loads were isolated from the bus. The procedure was implemented at 94:21 hours and verified that there were no shorts on the bus.

After the CM had been powered down for about 24 hours, it began to cool down to a temperature well below the minimum expected operating temperatures. Engineering support personnel became concerned about the motor switches which are normally used to connect the battery busses to the main dc busses. When it was realized that the CM was going to get unusually cold before the initiation of the entry sequence, the ability of the batteries to provide sufficient potential to drive these switches was questioned. The analysis of the situation was difficult because of the uncertainty as to how cold the battery compartment would get, and it could not be proven that a problem would exist. However, to circumvent the situation, it was decided to close the bus tie motor switches after the main bus B checkout. Subsequently, the appropriate circuit breakers would have to be used as switches to connect and disconnect the batteries from the busses (fig. B6-8). A step-by-step procedure was defined and read to the crew and the bus tie switches were closed at 94:21 g.e.t.

A procedure was also developed for charging the CM entry batteries with the LM electrical power system. Approximately 20 amp-hours of the 40 amp-hours capacity had been used from entry battery A during the period immediately following the accident; a much smaller amount had been taken from battery B since that time. Since the LM battery capacity provided a comfortable power margin for the return to Earth, Mission Control decided to invest some of that power in charging the CM batteries. Preliminary examinations of an entry preparation sequence indicated that in order to not rush the crew, the CM powerup should be initiated about 6 hours before entry. To do this demanded that all three CM batteries be fully charged. The procedure to charge the CM batteries was defined in complete detail by Mission Control. In its most basic terms, it was simply a procedure that used the LM/CM electrical umbilical to get power to the CM main bus B. Then the CM battery charger was tied to this bus and the battery to be charged. The procedure as read to the crew consisted of four typewritten pages of notes and a step-by-step switch position definition. The battery charging was initiated at about 112 hours g.e.t. to demonstrate that it could be done and was completed at 128 hours after 18 of the 20 amp-hours had been replaced. This was done well before the reentry preparation, to allow the entry planning to proceed with the assurance that all batteries would be fully charged at the beginning of the entry preparations.

Return to Earth Trajectory Control

All trajectory determination and maneuver targeting for getting the crew back to Earth was performed by the Mission Control Center. This is the normal procedure, but usually the crew also has the capability to do this. This serves as a backup in case communications are lost with the ground. However, with the command module G&N system completely shut down, the crew was totally dependent on Mission Control for navigation, and abort and midcourse correction maneuver targeting. There was no backup.

There were four trajectory change maneuvers performed to return the spacecraft to the recovery area in the mid-Pacific Ocean following the command module powerdown (fig. B6-9). The first, performed at 61:30 g.e.t., placed the spacecraft on a safe reentry trajectory. The second, performed at 79:28 g.e.t., adjusted the Earth landing point to

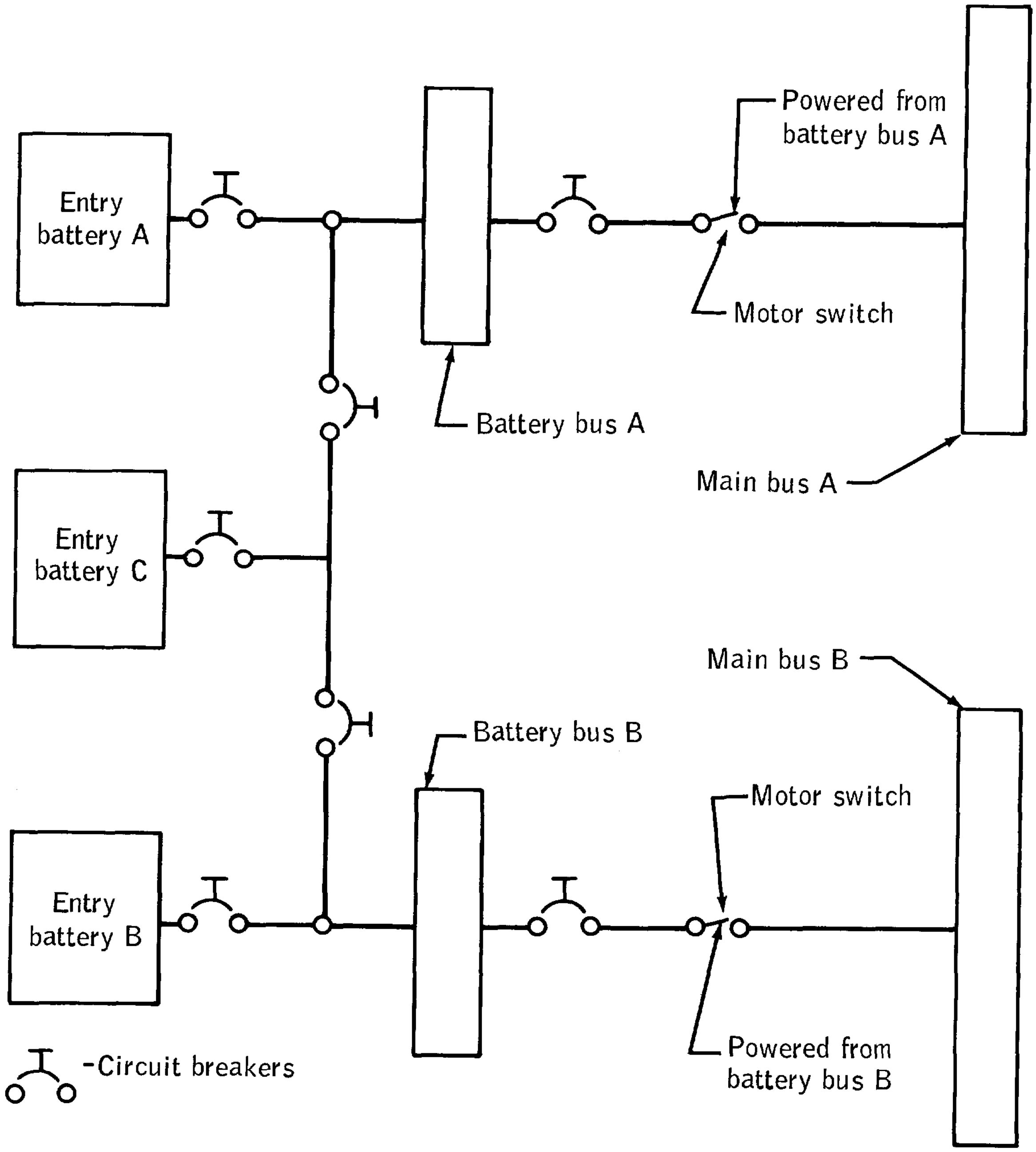


Figure B6-8.- Simplified wiring diagram showing battery power to main busses.

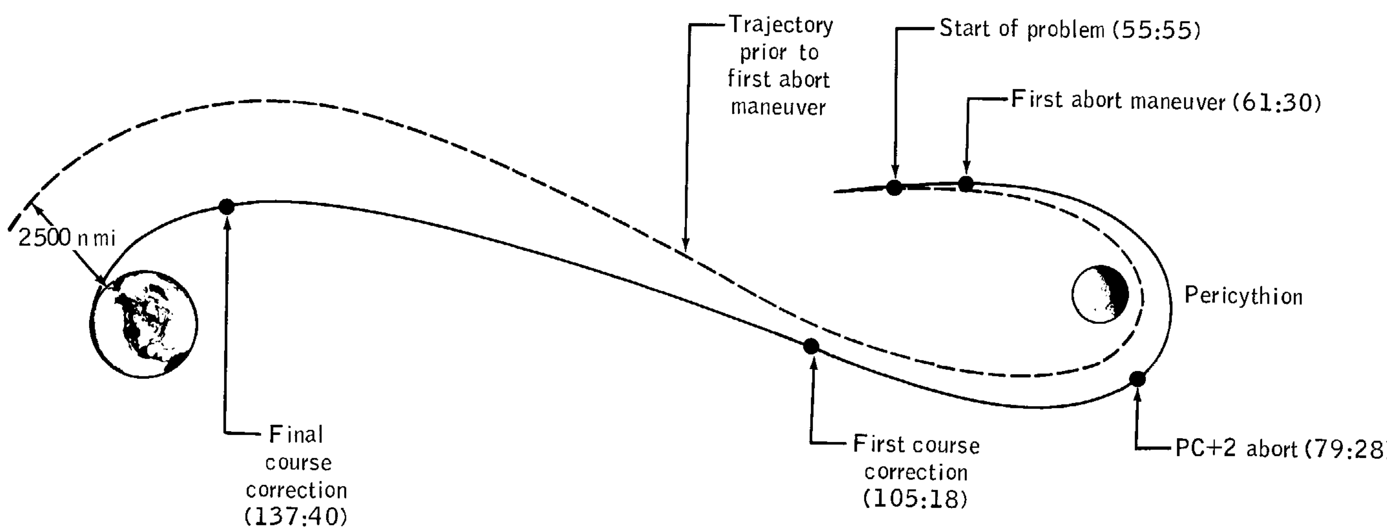


Figure B6-9.- Return-to-Earth trajectory control.

the mid-Pacific recovery area. The last two maneuvers, performed at 105:18 and at 137:40 g.e.t., were course corrections which adjusted the entry conditions to be in the middle of the safe entry corridor. These maneuvers and the decisions related to the choice of specific course changes are described in the following paragraphs.

Abort maneuver at 61:30 hours.- Soon after the failure in the CSM it became obvious that the lunar landing mission could not be achieved and that all effort would have to be focused on getting the crew back to Earth as soon as possible. At the time, the spacecraft was not on a trajectory that would return to a safe reentry of the Earth's atmosphere --so a trajectory change was mandatory. The following questions needed to be answered: What path should be followed back to Earth? When should the trajectory-changing maneuver be executed?

Because the spacecraft was on its way to the Moon, there were two basic types of abort paths that could have been followed: (1) a direct abort in which the trajectory would be turned around and the spacecraft returned to Earth without circumnavigating the Moon; and (2) a circumlunar abort in which the spacecraft would follow a path around the Moon before it returned to Earth. The disadvantage of the circumlunar abort path is that the flight back to Earth takes a longer time than for direct aborts. However, circumlunar aborts require much less velocity change and consequently much less propellant to perform, and part of the flight time can be made up by executing an additional "speedup" maneuver after the spacecraft has passed the Moon.

The direct abort was ruled out for Apollo 13 because the propellant requirements were so large. It would have been necessary to jettison the LM in order to reduce the spacecraft weight so that the service propulsion system (SPS) engine could make the necessary velocity change. The LM was essential to the crew's survival, and must not be jettisoned. Therefore, the choice was narrowed to the circumlunar abort which could be executed with the LM descent propulsion system (DPS), but there were still some decisions to be made. The options were as follows:

1. Do nothing until after the spacecraft passed the Moon; then execute a maneuver to place it on an Earth-return trajectory.

2. Execute a maneuver as soon as practical to place the spacecraft on an Earth-return trajectory and power down the LM immediately thereafter.

3. The combination of both the above: Get on an Earth-return trajectory as soon as practical, and after the spacecraft passed the Moon, perform a maneuver to speed up the return to Earth.

Option 2 was selected. The principal reason was that the LM systems necessary for executing the maneuver were working at the time, and they might not be working 20 hours from then when the spacecraft was in position to do option 1. Another consideration was the fact that the velocity requirement to get on an Earth-return trajectory would increase from 40 fps to 160 fps, making it impossible to perform with the RCS system if this became necessary. So even though option 1 would have allowed an immediate partial LM powerdown, saving some electrical power and water, it was decided that the risk was not worth the savings. Also, option 2 left option 3 available if the guidance and navigation system could be powered up to perform the second maneuver.

The decision having been made to perform a circumlunar abort, and to perform as soon as possible the maneuver to place the spacecraft on a safe reentry trajectory, the only question remaining open was what Earth landing point to target for. Because of the LM consumables status, getting back to Earth as soon as possible was the overriding factor. The quickest return resulted in a landing in the Indian Ocean at 152 hours g.e.t. This meant giving up the ability to bring the spacecraft down in the vicinity of the prime recovery force in the Pacific, although at least a water landing was provided. This was considered to be acceptable because the abort maneuver after passing the Moon probably could be used to decrease the flight time and to land in the prime recovery area.

Post-pericynthion abort maneuver.- Although the spacecraft was placed on a reentry trajectory by the abort maneuver at 61:30 with a landing at 152 hours g.e.t. in the Indian Ocean, it was decided that a post-pericynthion abort maneuver (PC + 2) should be performed. There were two reasons: (1) to reduce the return time to increase the LM consumable margin (the prediction at the time indicated only a 3-hour margin); and (2) to change the landing point to the mid-Pacific where the recovery force could be on station.

During the first few hours after LM activation, detailed analysis of LM consumable usage had shown that the guidance and navigation system could be kept powered up until after the PC + 2 abort maneuver at 79:30 g.e.t. It was predicted that all consumables would last at least until 155 hours g.e.t. even if the LM powerdown to 15 amperes total current were delayed until after 80 hours g.e.t.

There were several options available for decreasing the flight time, but only the three listed in the following table provided a landing in the mid-Pacific.

Option	Delta V, fps	Engine used	Landing time, hours g.e.t.
1	850	DPS	142
2	4000	DPS	118
3	4000	SPS	118

Option 1 was selected even though it resulted in the longest flight time, because of some very undesirable characteristics of options 2 and 3. The problem with option 2 was that it would be necessary to jettison the service module in order to be able to get a 4000 fps velocity change with the LM descent propulsion system. Such a maneuver would almost deplete the descent propellant, leaving a very limited capability should subsequent maneuvers be necessary. There was a high probability that a large course correction would have to be made later. Option 2 was seriously considered, but eventually rejected because it left the CM heat shield exposed to the space environment for such a long period of time, and the possible thermal degradation that might result from this was an unknown risk. The heatshield capability to withstand reentry might be compromised by the prolonged period of cold temperature it would experience. Option 3 was rejected because of the unknown status of the SPS; it was thought that the SPS or the SM might have been damaged by whatever had caused the "bang" and that the SPS should not be used unless absolutely necessary.

Since option 1 provided a comfortable consumables margin and allowed retention of the service module, it was selected. Option 1 also allowed a descent propulsion system delta V capability of approximately 1000 fps to be retained after the abort maneuver.

Part of the preparation for each mission is the establishment of "ground rules" and maneuver monitoring criteria for each planned maneuver. The "ground rules" are general statements which define what should be done if certain events occur. The maneuver monitoring criteria define explicitly the conditions under which the crew will deliberately terminate the maneuver early. The criteria are not the same for all maneuvers because there is a wide variation in the seriousness of the effect of dispersions, and in the seriousness of the effects of early or late engine shutdown. The trajectory and mission situations for the post-pericynthion abort burn were different from any of those for which criteria had been defined; therefore, it was necessary to establish these "rules."

The pertinent characteristics that would affect the rules were as follows:

(a) The spacecraft was on a safe reentry trajectory, although small course corrections probably would be required before reentry.

(b) The primary purpose of the maneuver was to place the landing point in the vicinity of the recovery force.

(c) The secondary purpose of decreasing the flight time was of major importance.

(d) The LM inertial platform had not been fine aligned for approximately 20 hours.

(e) The maneuver could be delayed for 2 hours with an increase in delta V of only 24 fps.

(f) The LM descent propulsion system was to be used.

The following ground rules based on these characteristics were established by the Mission Control team and were given to the crew:

(a) If the engine does not light, do not attempt any emergency start procedures.

(b) If the primary guidance and navigation system (PGNS) has failed, do not perform the maneuver.

(c) Do not attempt to null the indicated velocity errors after engine shutdown.

(d) If an engine shutdown occurred, a subsequent midcourse correction would be performed no sooner than 2 hours later.

The criteria for early termination of the maneuver were defined as follows:

1. Propulsion System Parameters

(a) Engine chamber pressure	≤85 psi (TM)
	≤77 percent thrust (on board)
(b) Inlet pressure	≤150 psi (TM)
	≤160 psi (on board)
(c) Delta P fuel/oxidizer	>25 psi (ground monitored)

2. Guidance and Control System Parameters

(a) Attitude rate — >10 deg/sec (except during start transient)

(b) Attitude error — >10 degrees

(c) Engine gimbal light

(d) Inertial platform failure with a program alarm

(e) Computer warning light

(f) Control electronics system dc fail light

A final rule that was defined stated that if an early engine shutdown was experienced not due to any of the above, a relight should be attempted, using the engine-start pushbutton and the Descent Engine Command Override switch.

A contingency LM activation checklist had been defined prior to the mission and was part of the crew's Flight Data File. This checklist was designed to prepare the LM for a docked descent propulsion system burn from a completely dormant state. The majority of this checklist had been accomplished with the initial LM powerup at 58 hours g.e.t. The flight controllers reviewed the list in detail and defined a modified list of steps necessary to prepare the LM for the abort maneuver. The modification was basically a deletion of steps already accomplished or not necessary; however, there was one change which revised the time at which the helium regulator shutoff valve was to be closed. This was done to preclude the possibility of a shift in the regulator operating pressure causing a freezing of the propellant lines after this burn. Such an event would prevent further use of the descent engine and it was mandatory to maintain this engine for probable subsequent trajectory changes.

Midcourse correction maneuver.- Postmaneuver tracking data indicated that the second abort maneuver had placed the spacecraft grossly on the right path. However, because the LM inertial platform could not be fine aligned prior to the maneuver, the execution errors were larger than normal and the spacecraft was not on a safe reentry trajectory. This was expected and subsequent corrections were planned for in the LM consumables budget. The correction delta V magnitude was projected to be about 7 fps if executed at 104 hours g.e.t. Unlike the abort maneuver, the course correction maneuvers are not extremely sensitive to pointing accuracy, and with the delta V of only 7 fps it could probably be executed with sufficient accuracy without the inertial guidance system. A special team, composed of off-duty flight controllers and members of the backup flightcrew, was formed to define the maneuver ground rules and procedures to be followed for the course correction maneuver. A detailed crew checklist was to be developed also. None of the procedures or checklists in the Flight Data File were applicable because of the unique situation that existed for this case.

The major issues addressed by this team were as follows:

1. How to get the spacecraft aligned in the proper direction for the maneuver? Was it necessary to power up the inertial platform?

2. Which engine should be used, descent propulsion system or LM RCS?

3. What burn monitoring criteria should be used?

4. What attitude control modes should be used?

The team determined that it was unnecessary to use the inertial platform for the maneuver. The spacecraft could be oriented in the proper pitch direction by sighting on the center of the Earth with the Crew Optical Alignment System (COAS) fixed along the LM +Z axis. The approximately correct azimuth could be achieved by aligning the sunset terminator parallel to the LM Y-axis. This procedure had been developed in the preparation for Apollo 8 when it was discovered that course correction maneuvers could best be made in a local horizontal attitude (that is, perpendicular to a vector from the center of the Earth to the spacecraft). It could easily be applied to the LM-active maneuver, and would give adequate thrust pointing accuracy, so it was not necessary to power up the LM G&N system and try to align its inertial platform.

It was decided to use the descent propulsion system for the maneuver instead of the RCS engines, because the engine-on time for an RCS maneuver would exceed a constraint which protects the LM RCS plume deflectors. The engine was to be left at the low throttle point (about 12.6 percent of full thrust) to give the crew more time to monitor the burn and the lower acceleration should increase the shutdown accuracy. The engine shutdown criteria were the same as for the previous burn. It was decided to monitor the delta V with the backup guidance system accelerometers, but to shut the engine down at a fixed delta time specified by Mission Control. Studies had shown that the burn time computed by Mission Control was very accurate. Since the accelerometers had not been maintained at their proper temperature (heaters had been turned off to reduce consumables expenditure), their status was questionable and the team decided to not use the backup guidance system as an engine shutdown cue. However, if this system appeared to perform nominally during the maneuver, it would be used to null the velocity residuals in the X direction. Velocity errors in either Y or Z direction had an insignificant effect on the entry conditions and were not to be nulled.

Attitude control of the docked vehicle with the backup system required both the CDR and the LMP to actively participate, and Y- and Z-axis translation thrusters had to be used to get adequate control torque. The team defined the modes and procedures to be used in getting the spacecraft in the correct attitude and in controlling the attitude during the engine burn. A procedure to return the spacecraft to the passive thermal control condition was also defined.

All plans were completed after two lengthy sessions. A subgroup from the team defined a detailed crew checklist to be followed in preparing for the maneuver and in preparing for the coasting flight following the maneuver. The checklist was evaluated by members of the backup crew in mission simulators at MSC and some minor modifications were made as a result. The checklist and the procedures were reviewed by the on-duty Mission Control team and then read to the crew approximately 5 hours prior to the scheduled course correction. This allowed the crew ample time to study them and to rehearse their roles.

Entry Procedures and Checklist Definition

After the situation in the spacecraft was stabilized, one of the several parallel activities that was initiated was the definition of procedures for the pre-reentry phase. The total loss of electrical power in the service module forced some major revisions to the activities and the crew procedures for this part of the mission. The most significant consequences of this loss were the following: (1) SM RCS engines would not continue to fire to separate it from the CM after jettison; and (2) LM electrical power and RCS should be used to conserve the CM batteries and RCS propellant as much as possible. This meant that the LM should be retained through as much of the pre-entry sequence as possible, and that a plan for jettisoning the SM and the LM had to be worked out.

A first iteration plan for the pre-entry phase was available as early as 12 hours after the LM activation. This plan called for CM powerup 2 hours before arrival at the entry interface (EI - 2 hours), and required the total remaining capacity from the CM entry batteries, 98 amp-hours. After the plan was thoroughly reviewed by all elements of the operations team, including mission planning and flight crew support personnel, several modifications and additions were considered necessary. The principal difficulty was that the crew would probably be rushed, and there was little or no extra time allowed for contingencies. It was evident that the timeline needed to be extended and the CM batteries would have to be recharged to at least 115 amp-hours. The recharging was accomplished and the procedure is described in Part A2 of Appendix A.

The White Team, one of the four flight control teams assigned to the mission, was taken off its normal rotation of duty in order to devote full attention to developing the reentry preparation sequence of events, crew procedures, and checklists. With this flight control team as the lead element, all MSC organizations normally involved in this type of premission activity were enlisted in this effort. In the course of defining the procedures, extensive use was made of the spacecraft simulators at MSC and KSC. These simulations, performed by members of the backup crews, served two essential purposes. The first was simply to evaluate them--to determine if they were practical, safe, efficient, and adequate. The second purpose was to determine the time required to complete certain parts of the procedures. The latter was important because a completely defined timeline had to be given to the crew in order to insure that everything was accomplished on time. It was essential that this timeline be realistic because the crew could not afford to get behind and fail to complete it, but neither could they start too early and use too much power from the CM batteries.

Another source of data used to develop the procedures was a series of contingency separation studies that was performed prior to the flight by mission planning personnel. These studies had examined the trajectory-related considerations for several different methods of jettisoning the SM and the LM. They had defined the effects of different attitudes, time, and velocity of jettison on the subsequent separation distances. It was only necessary to verify that these studies were valid for the Apollo 13 conditions, and then select the one with the most optimum characteristics.

The planning and evaluation of the pre-entry activities continued for approximately 2 days. At the end of this time, a complete plan had been defined and thoroughly reviewed. It was read to the crew at about 120 hours g.e.t., which gave them about a day to study and rehearse their procedures.

The pre-entry sequence plan (fig. B6-10) called for initiating the powerup at EI - $6\frac{1}{2}$ hours, with the LM supplying power to main bus B in the CM and entry battery C supplying power to main bus A. A total of 115 amp-hours was required of the CM entry batteries, including a 23 amp-hour allowance for contingency after splashdown. A detailed expected battery current profile was plotted and used during the actual preparation to verify that a safe power margin was maintained throughout the reentry preparations. Battery utilization was planned so that all three entry batteries would be available throughout the entry phase. It

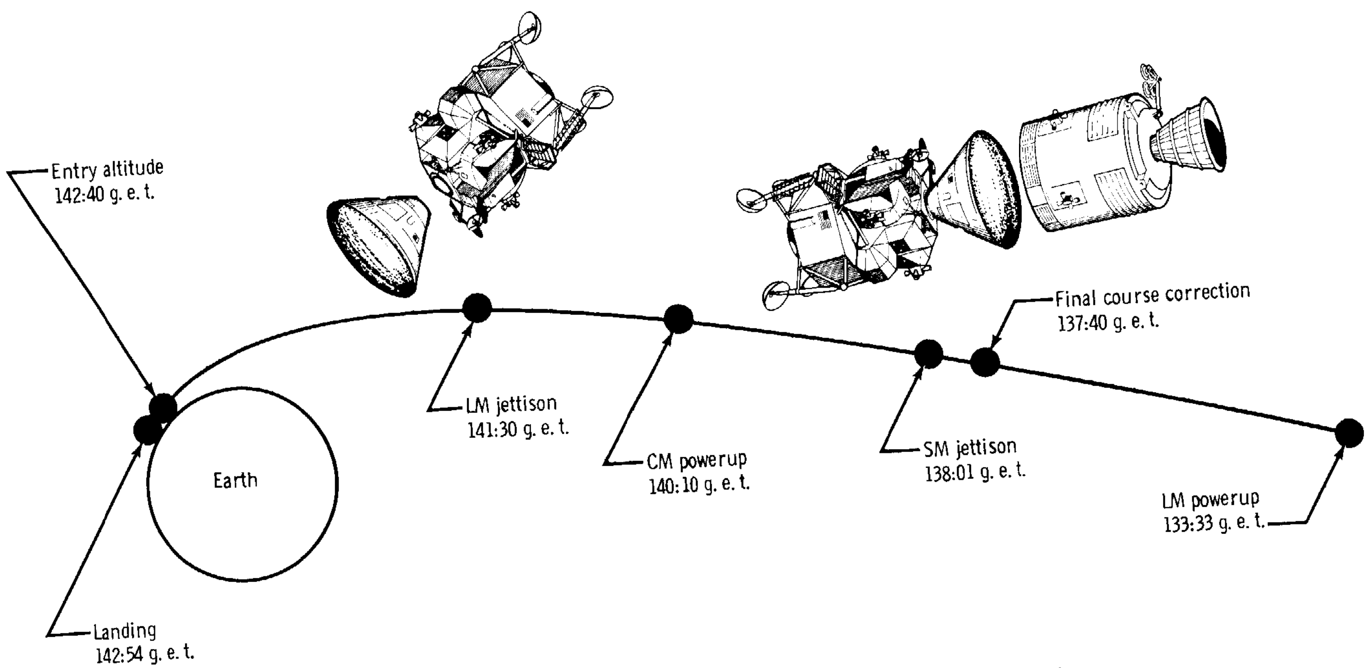


Figure B6-10.- Pre-reentry sequence milestones.

was predicted that battery C would be depleted after deployment of the main chutes, and in fact it was. This left the redundant capability of two batteries available to inflate the uprighting bags after splashdown.

The initial part of the reentry preparations, LM powerup, was performed about 3 hours earlier than planned. The crew was not resting comfortably due to the cold environment, and since there was ample margin in the LM batteries and water tanks, it was decided to turn on some equipment to try to warm up the spacecraft.

After activating the LM guidance and control system, the first major milestone in the entry sequence was to execute the final course correction to place the spacecraft on a trajectory that was in the center of the safe entry corridor. Prior to the final course correction, the trajectory had an entry angle error of about +0.5 degree, which is a safe condition, but slightly shallow (fig. B6-11). It is a standard practice to perform a final trim maneuver a few hours prior to entry to try to remove any entry angle error greater than ±0.1 degree, and this course correction was incorporated in the timeline before it was known whether or not it could be required.

The planned procedures for the final course correction were the same as for the earlier one performed at about 104 hours g.e.t., including the alignment procedure which only required sighting the Earth through the COAS. Manual control of the actual delta V maneuver was also planned. However, since the LM powerup was started 3 hours earlier than originally expected, it was decided to use part of this time to align the LM inertial platform. This was done with the crew sighting on the Moon and the Sun for orientation determination. A further modification to the planned procedures of using the primary guidance system to perform the course correction had to be abandoned, because the attitude error indications did not behave properly. It was suspected that there might be something wrong with the guidance computer, so the crew performed the maneuver manually, following the original plan. Subsequent analysis has shown that the attitude error indications were not indicative of a system problem, but were a result of the guidance system activation procedures. These same indications did not show up in the simulator evaluations performed before the crew was given the procedures because of the limitations of simulator initialization.

The service module jettison was the next major milestone in the pre-entry sequence. It was performed at about 4-1/2 hours prior to reentry. The techniques used and the attitude and delta V requirements for it were obtained from premission studies. Basically, the technique was very similar to that used by a railroad switch engine to get rid of the end boxcar. The spacecraft was given an impulse with the LM RCS that caused a velocity change in the desired direction of about 0.5 fps; the CM/SM separation pyrotechnics were fired, physically disconnecting

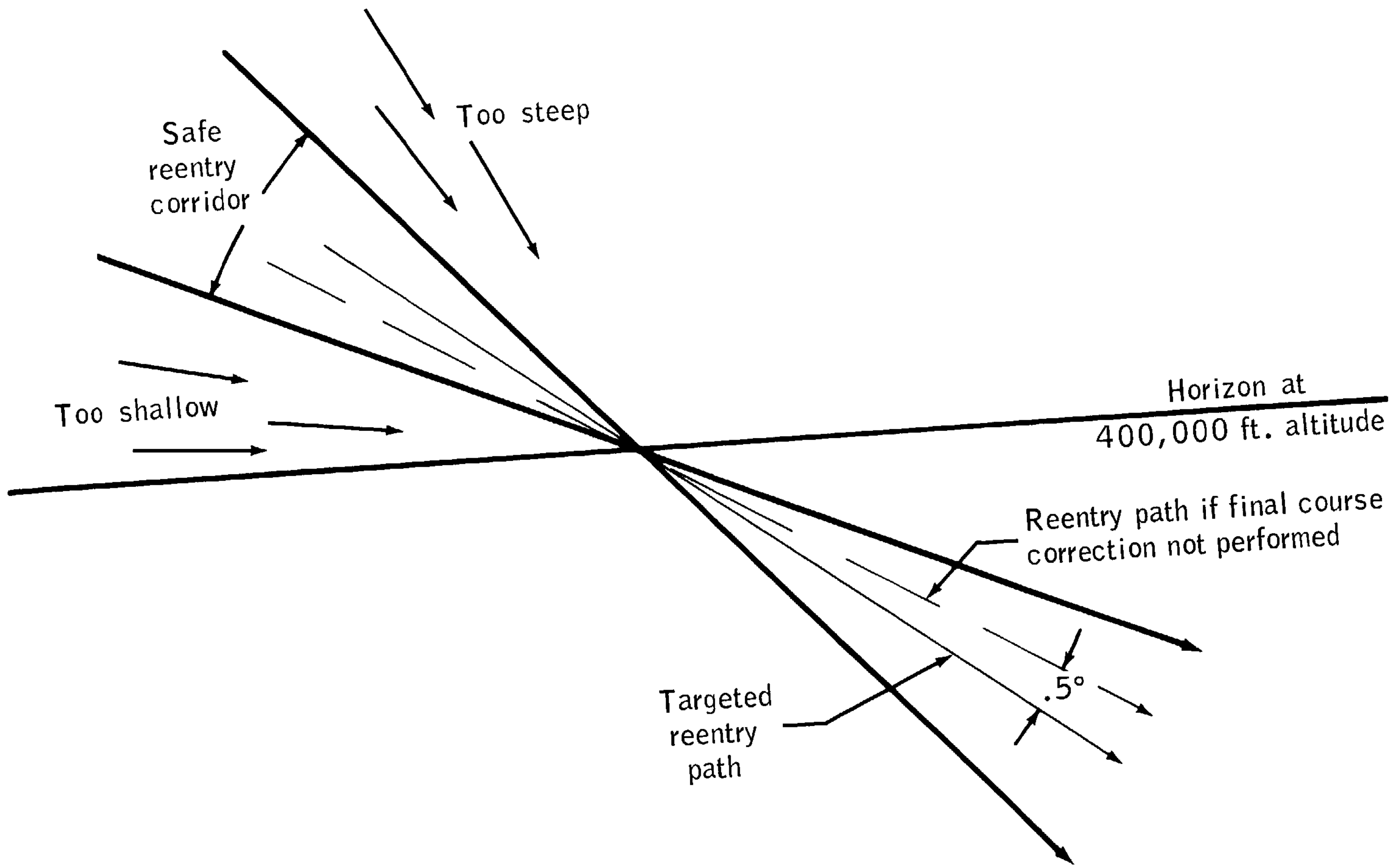


Figure B6-11.- Flight path angle and the safe reentry corridor.

the two modules; and a velocity change of the LM and CM was accomplished by reverse thrust from the LM RCS. The service module continued to translate relative to the manned modules, and separated from them at a rate of 0.5 fps. The normal method of using SM RCS jets to drive the SM away would not work because there was no way to get electrical power to keep the jets firing after CM/SM separation. The fuel cells which normally perform this function were inoperative.

The next major step was to get the CM inertial platform aligned. An automatic guidance controlled reentry was planned, which meant that the platform needed to be aligned to a known reference direction. There were several methods that could have been used to accomplish this, and a considerable amount of time was spent by the White Team in determining the best one. The selected plan used the docked align transfer procedure to get the CM platform coarsely aligned to the LM platform. The CM platform was then very accurately aligned to the desired direction by optical sightings with the CM sextant. Mission Control was standing by with an alternate procedure in case stars could not be seen through the CM optics; however, this was not necessary.

There was much interference on the voice and telemetry communication signals during this time period, which was later diagnosed to be due to the spacecraft attitude. Apparently the spacecraft was oriented so that the LM structure was blocking the signal from all of the omni antennas arrayed around the CM, and the received signal strength was very low. The antenna blockage problem was not recognized and several reconfigurations of the communication equipment were made to try to correct the problem, none of which were successful. In order to maintain adequate signal strength, it was necessary to receive data at the low bit rate only. This was not a major handicap, but it did cause some delay in completing the preparation of the CM guidance system for reentry.

The LM jettison from the CM was accomplished at about 1 hour prior to reentry. The attitude was based on premission studies, but no technique had been defined for achieving the actual separation with LM jettison from the CM only (no service module). The technique was defined by the White Team and consisted of using pressure in the LM/CM tunnel to impart a relative velocity to the two modules when the final separation pyrotechnics were fired. This method of separation had inadvertantly occurred at the LM final jettison on Apollo 10 and was known to give sufficient separation velocity.

It was planned to jettison the LM in a direction 45 degrees south of the spacecraft plane of motion; however, the crew maneuvered the spacecraft to an attitude 65 degrees north of this plane. Mission Control was monitoring the spacecraft attitude, but did not realize the mistake until the crew was in the process of final closeout of the LM. Flight controllers quickly analyzed the situation and determined that, although the 65 degrees north attitude did not give as much separation, it was acceptable. The major problem in being in error by 110 degrees was that it placed the CM in an attitude much closer to gimbal lock than is normally done. The crew had to be especially alert during the jettison and to use manual control of the CM to avoid gimbal lock.

The remainder of the sequence, from LM jettison to splashdown, followed normal procedures. The only difference was that the CM was completely independent of other spacecraft components at 1 hour prior to reentry instead of the usual 15 minutes.

PART B7

INSTRUMENT SYSTEM CHARACTERISTICS

Part 7 provides additional technical information of systems design and characteristics which are pertinent to interpretation of data presented in earlier parts of this Appendix. The following systems are discussed:

Oxygen Tank Temperature Instrumentation

Oxygen Tank Quantity Instrumentation

Oxygen Tank Pressure Instrumentation

Apollo PCM Telemetry System

Mission Control

OXYGEN TANK TEMPERATURE MEASUREMENT

The temperature measurement is made with a platinum resistance thermometer (R/T) encased in an Inconel sheath attached to the Teflon insulator part of the quantity probe (fig. B7-1). The resistance of the R/T and the transducer output voltage increase with temperature. The signal conditioner which serves as a reference voltage generator and amplifier is located on the oxygen tank shelf. An electrical schematic of the transducer is shown in figure B7-2.

The system electrical and performance parameters can be summarized as follows:

Data sample rate	one per second
Range	-320° F to $+80^{\circ}$ F
Corresponding R/T values	71 to 553 ohms
Output voltage	0 to 5 V dc
Accuracy	±2.68 percent or $\pm 11^{\circ}$ F
Output impedance	5000 ohms

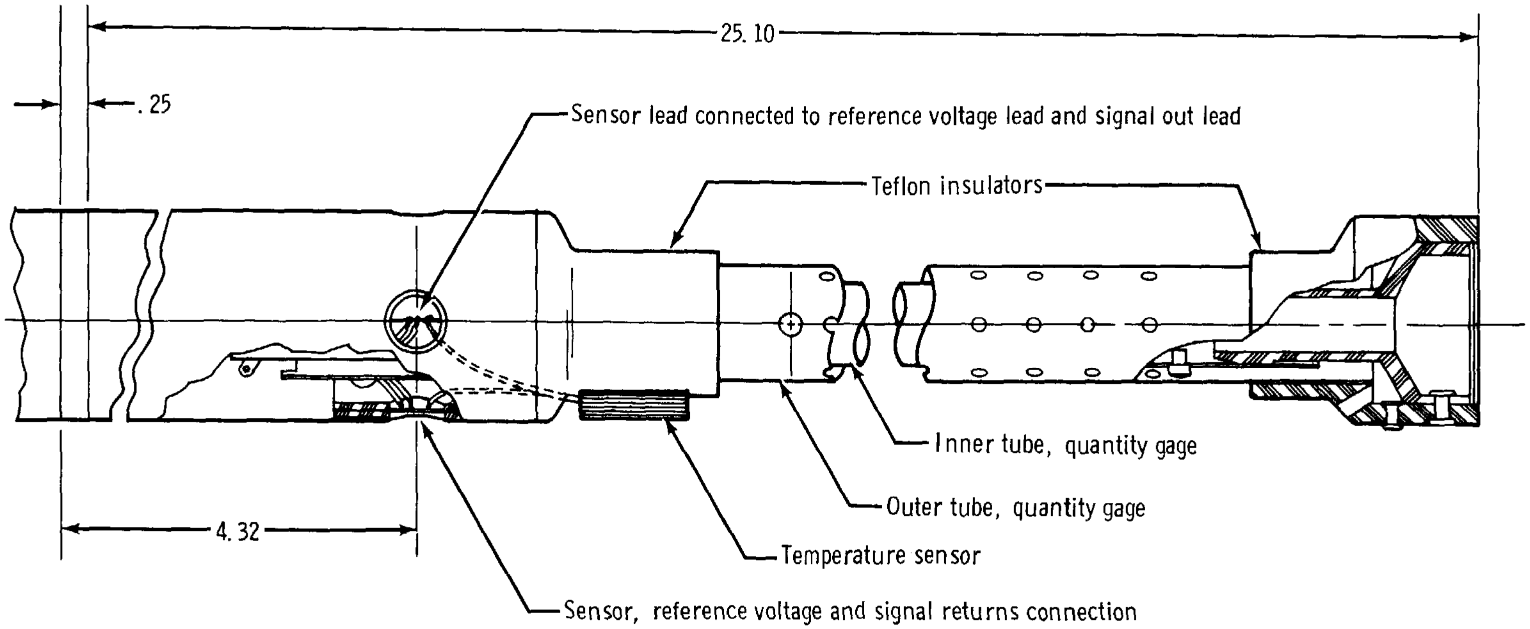


Figure B7-1. - Oxygen quantity gage and temperature sensor location.

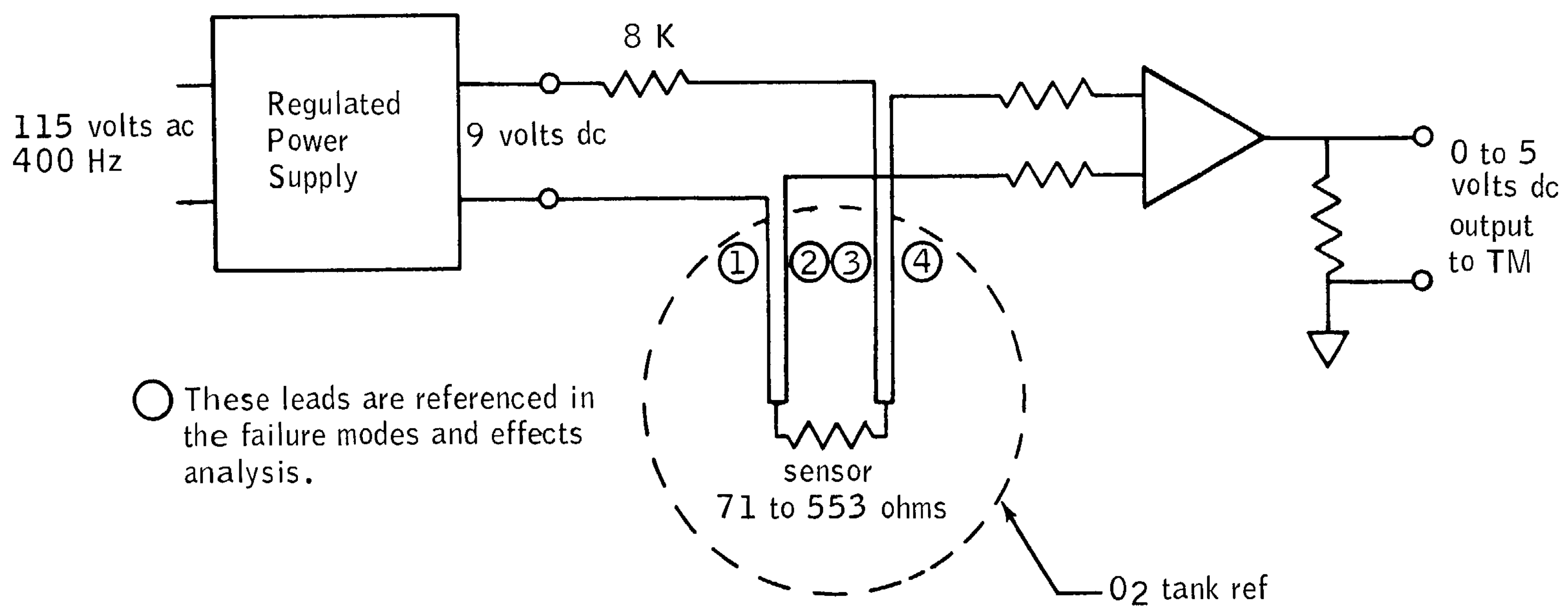


Figure B7-2.- Oxygen tank temperatures sensing circuit.

Power	1.25 watts, 115 V ac, 400 cps
Time constant in liquid nitrogen or alcohol	Approximately 20 seconds

The 20-second time constant was measured by plunging the sensor first into liquid nitrogen at -317° F and then into dry ice/alcohol at -91° F. Tests were made under one-g and 1 atmosphere and are not applicable to supercritical oxygen and zero-g.

Telemetry would indicate the temperature of the sensor itself, but under rapidly changing conditions the sensor could remain almost unaffected by local temperature changes in other parts of the tank. The effect of various failure modes on the transducer and its output signal are presented in table B7-I.

OXYGEN TANK QUANTITY INSTRUMENTATION

The oxygen tank quantity gage is shown in figure B7-1. This gage senses the average dielectric constant of oxygen in the cylindrical annular volume between two concentric aluminum tubes. The dielectric constant is proportional to density, which in turn is proportional to the quantity of oxygen in the tank. The gage is approximately 2 feet long; the outer tube is about 0.85-inch ID and the inner tube is about 0.65-inch OD to form two plates of a capacitor with 0.10-inch spacing. The gage mounts in the center of the tank.

The gage capacitance is connected in series with a reference capacity to form a capacitive 400-cycle ac voltage divider as shown in figure B7-3 and is adjusted to apply zero volts input to the amplifiers when the tank is empty. As the tank is filled, the gage capacity increases, applying a voltage to the amplifier input. This voltage is amplified and rectified to provide an output signal voltage which increases to 5 volts dc when the tank is full.

The reactive voltage developed across the probe capacitance will change as rapidly as capacitance changes. The rectifier filter on the output of the signal conditioner introduces a time constant of about 0.022 second in the instrument response.

TABLE B7-I.- FAILURE MODES OF THE OXYGEN TANK TEMPERATURE TRANSDUCER

Failure mode	Indication	Resulting damage
Any of the four temperature sensor leads shorted to 115 V ac line (1, 2, 3, 4)	Full scale output followed by zero output	Would fail signal conditioner amplifier, sensor element, and pulse code modulation gate
Temperature sensor shorted to the density probe element	*No change in output	Probably no circuit element damage
Temperature sensor shorted to ground (either side)	*Zero output signal	No circuit damage
Dc power shorted to temperature sensor	Full scale output	Would fail signal conditioner output
Either or both sensor leads open	Full scale output	None
400 Hz power input to power supply disconnected.	Output drifts to zero as charge in power supply filter capacitors discharge.	None
Temperature sensor leads shorted to each other	*Zero output	None
Any one of leads 1, 3, or 4 broken (open) (fig. B7-2)	Zero output	No circuit damage
Open lead 2 (fig. B7-1)	*Immediate rise to full scale followed by a linear decay to zero in approx. 10 msec	No circuit damage

*Indication verified by test

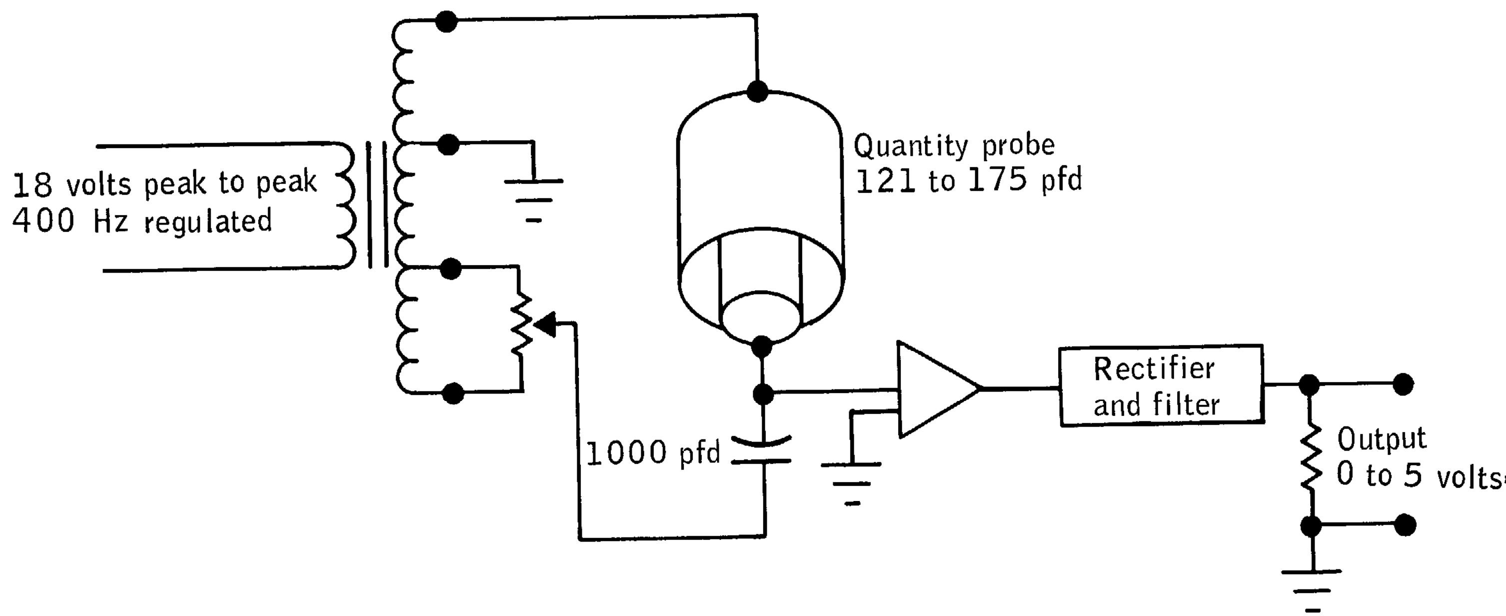


Figure B7-3.- Oxygen quantity gage block diagram.

Gage parameters are as follows:

Tank condition	Empty	Full
Density	0	69.5 lb/ft^3
Dielectric Constant	1.0	1.45
Capacitance	121	175 picofarads
Output voltage	0	5 V dc
Output impedance	500 ohms	
Power	2-1/2 watts	
Supply voltage	115 V, 400 cycles	
Accuracy	2.68 percent full scale	
Value of fixed capacitance	1000 picofarads	
Data sampled	Once per second	

This method of gaging works well for single-phase fluids in any gravity environment so long as the fluid is uniformly mixed with no significant density variations. But under zero-g, density and temperature variations can exist in the fluid, especially when heat is added without any fluid movement (convection). Under these conditions, the gage measures the average density of the oxygen between the two tubes which may or may not be representative of the average density in the tank.

If the gage is either opened or shorted, the signal conditioner is overdriven and a greater-than-100-percent quantity is indicated. Other malfunction characteristics follow.

Failure Mode	Effect
1. Elements of probe shorted to each other	Full scale output
2. Wire to either element disconnected from probe	Full scale output
3. Outside element of probe or its lead wire shorted to ground	Measurement indicates some value between zero and full scale
4. Inside element of probe or its lead shorted to ground	Random output tending towards zero
5. Clear shorted probe	Output decreases to zero, remains 0.7 second, then increases to correct value in about 1-1/2 second
6. Clear open probe fault	Output assumed correct value within 1/2 second
7. Intermittent shorts, any combination	Output becomes irregular sawtooth

OXYGEN TANK 2 PRESSURE INSTRUMENTATION

The location of the oxygen tank pressure measuring instrumentation is shown schematically in figure B7-4. Pressure transducers for both tanks are located in a valve module assembly along with the pressure switches and pressure relief valves as shown in figure B7-5. The valve module assembly is connected to the oxygen tanks by 19-foot lengths of 1/4-inch and 3/16-inch OD tubing.

The pressure transducer consists of a diaphragm 0.2 inch in diameter and 0.015 inch thick to which are attached 4 chips of strain-sensitive semi-conductor materials electrically connected into a bridge circuit. When pressure is applied, deflection of the diaphragm changes the electrical resistance of the semi-conductor clips to unbalance the bridge and develop an electrical output proportional to the applied pressure. This output is amplified so that full-scale pressure of 1050 psia gives a 5 V dc output which is indicated on the CM instrument panel and telemetered to the ground through the PCM telemetry system.

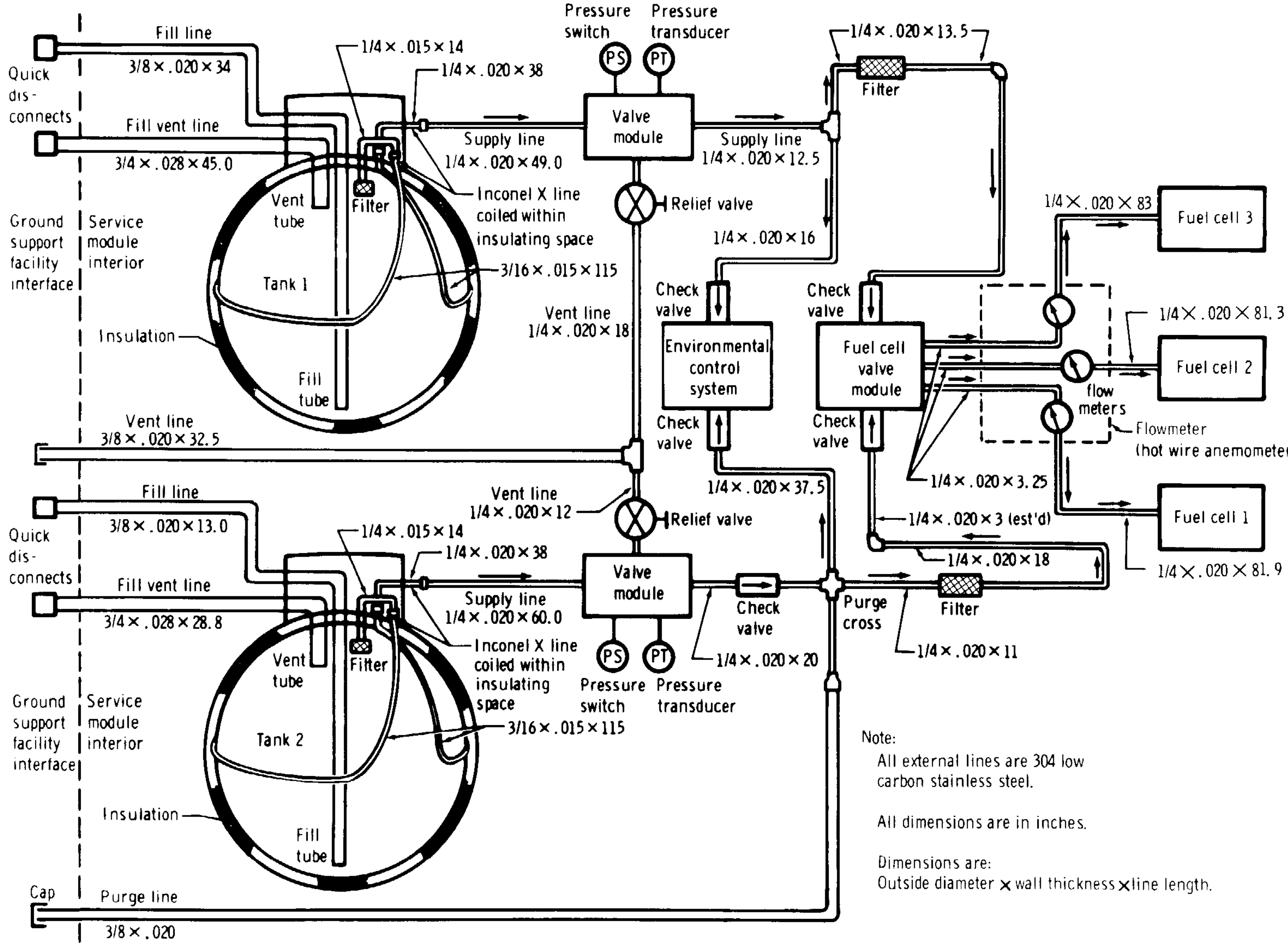


Figure B7-4.- Oxygen system.

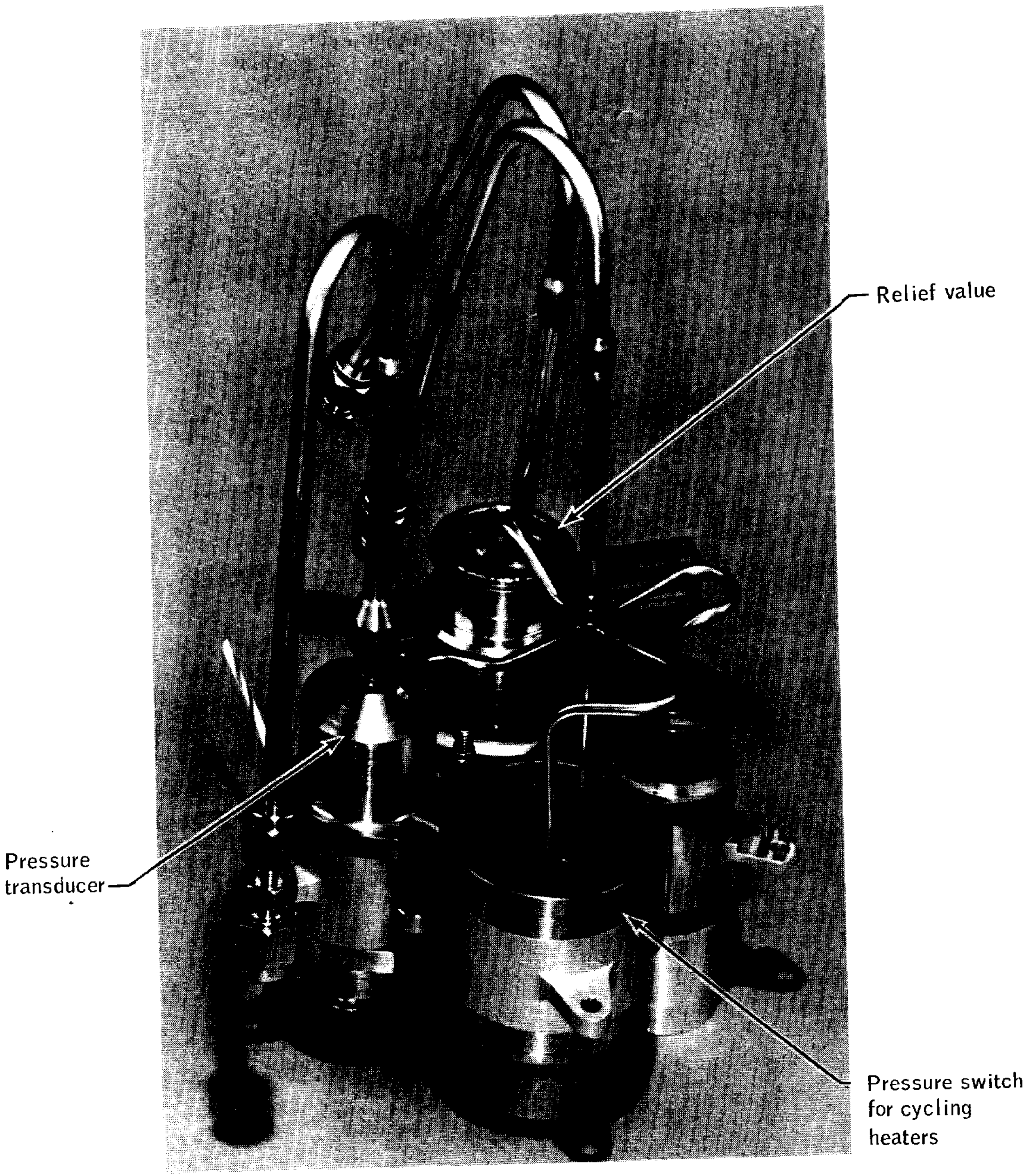


Figure B7-5.- Pressure transducer, relief valve, and pressure switch.

Other pressure transducer parameters are as follows:

Range	19 to 1080 psia
Accuracy	±2.68 percent fuel range
Output voltage	0 to 5 V dc
Output impedance	500 ohms
Power	1.5 watts
Voltage supply	28 V dc
Data sampled	Once per second

Under normal operating conditions oxygen flow through the 19 feet of tubing is about 1.5 pounds per hour and the pressure drop through the line is negligible.

The physical dimensions and electronic characteristics of the pressure transducer are such that its time lags are negligible as compared with acoustical lags of the tubing. If the relief valve opens (normally set at 1008 psia) or if the pressure in the tank changes suddenly, the delta P is communicated through the tube at sonic velocity (813 fps at 288° R) so that a delay of about 23 msec would be expected exclusive of pressure drops due to flow through the tubing. Tests run at MSC show that when a step pressure increase is applied at the tank end of the system, pressure indicated by the transducer begins to change in about 16 msec and reaches 63 percent of the pressure change in about 40 msec.

PULSE CODE MODULATION SYSTEM DESCRIPTION

The instrumentation system on the Apollo spacecraft interfaces with a pulse code modulation (PCM) telemetry system. In such a system, measurements are not presented continuously, but are sampled in time and quantitized in amplitude. Signal conditioners standardize the outputs from all sensors to a range of 0 to 5 volts. This voltage is fed into the PCM system where it is sampled and encoded for transmission to the ground.

The PCM system basically consists of a number of electronic input switches and an analog-to-digital encoder, all of which are controlled by a programmer. The analog switches, through programmer control, are sampled sequentially with a sample period of 40 microseconds for each input. The sampled voltage is then converted by the encoder into an 8-bit binary word which is subsequently transmitted to the ground. The sampling rate for each channel is selected on the basis of the rapidity with which that channel value changes under normal operation. Programmer sampling rates are 200, 100, 50, 10, and 1 sample per second. The end result of this operation when the system is in the high-bit-rate mode is a serial stream of data consisting of 128 eight-bit words in each frame with 50 frames transmitted each second. This corresponds to a bit rate of 51,200 bits per second. In the low-bit-rate mode, 1600 bits per second are transmitted and the measurements are made at a reduced sampling rate.

In evaluating telemetry data, consideration must be given to the fact that the system samples data in time and quantitizes in amplitude.

Figure B7-6 depicts an analog signal and its equivalent digital representation to illustrate several limitations of PCM telemetry systems.

1. Fast transients which happen to occur between the sample times will not be recorded.

2. A long transient whose peak amplitude occurs between sample times will be recorded with an incorrect peak amplitude.

3. A low-amplitude transient may go completely unrecorded even if its peak amplitude occurs at a sample time.

4. A change of one count in a parameter does not necessarily mean that the analog quantity has changed by an amount equal to the difference in count values. If the analog quantity happens to be very close to the switchover point between counts, a small change can cause the count to change.

5. If the analog quantity remains for a long time close to the switchover point from one count to the next, the output may toggle (jump back and forth) from one count to another. This does not indicate that the analog value is actually changing this rapidly but is characteristic of the system when noise is present.

6. In addition to the phenomena illustrated in figure B7-6, it must be recognized that noise in the RF link may cause erroneous data to be received on the ground. Such errors usually appear in the data as values which differ greatly from adjacent outputs from the same channel.

Table B7-II lists the measurements telemetered from the Apollo 13 command and service modules as well as their ranges, sampling rates, and value of one count.

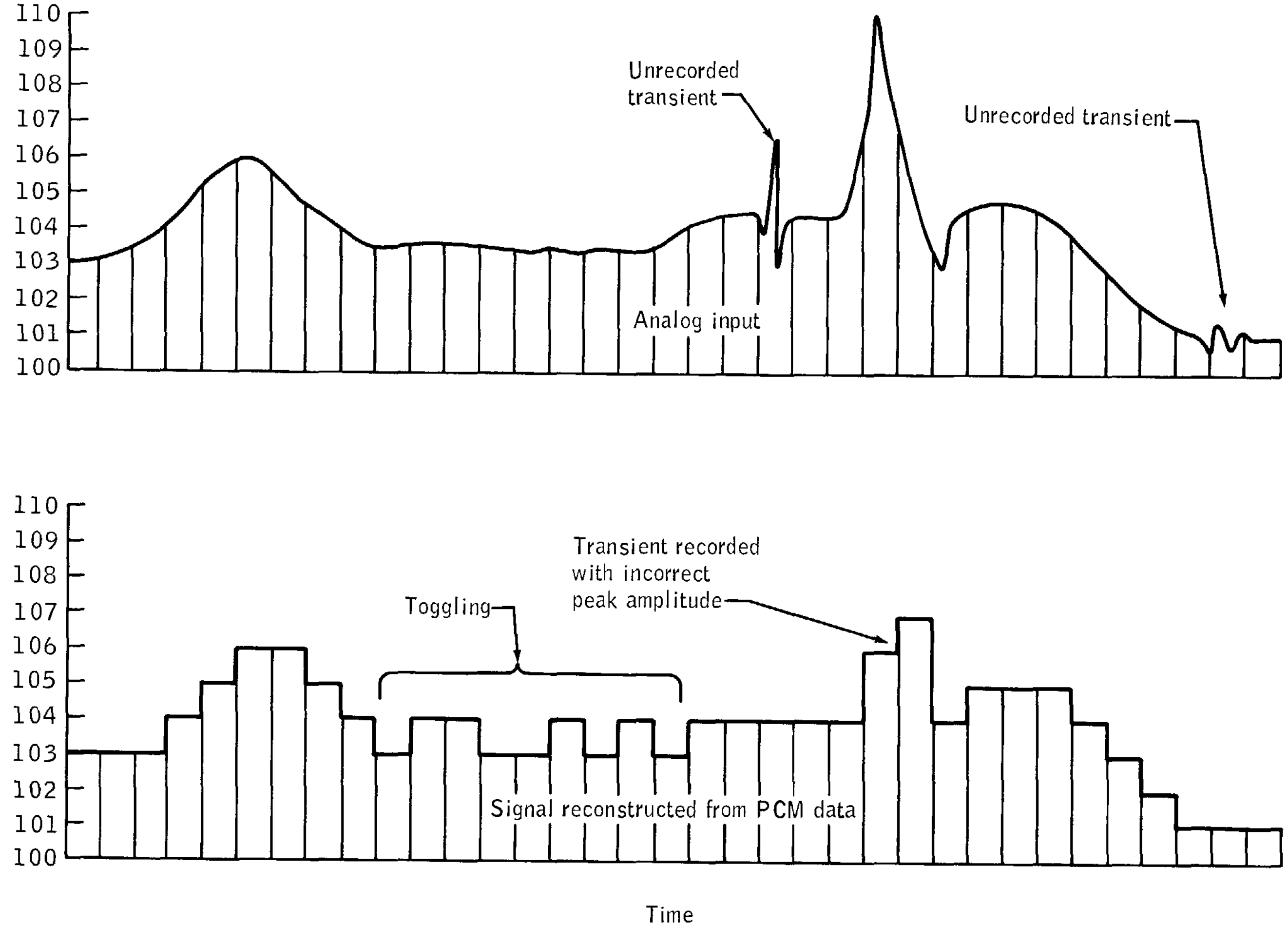


Figure B7-6.- Digital coding and reconstruction of analog signal.

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
CA1820T	TEMP CREW HS ABL SUR LOC 1A	°F	-300	+850	1	-	4 - *NL
CA1821T	TEMP CREW HS ABL SUR LOC 4A	°F	-300	+850	1	-	4 - NL
CA1822T	TEMP CREW HS ABL SUR LOC 7A	°F	-300	+850	1	-	4 - NL
CA1823T	TEMP CREW HS ABL SUR LOC 10A	°F	-300	+850	1	-	4 - NL
SA1830T	TEMP SM SKIN SURF LOC 1A	°F	-120	+270	1	-	1.5 - NL
SA1831T	TEMP SM SKIN SURF LOC 4A	°F	-120	+270	1	-	1.5 - NL
SA1832T	TEMP SM SKIN SURF LOC 7A	°F	-120	+270	1	-	1.5 - NL
SA1833T	TEMP SM SKIN SURF LOC 10A	°F	-120	+270	1	-	1.5 - NL
SA2377T	TEMP BAY 2 OX TANK SURFACE	°F	-100	+200	1	-	1.2
SA2378T	TEMP BAY 3 OX TANK SURFACE	°F	-100	+200	1	-	1.2
SA2379T	TEMP BAY 5 FUEL TANK SURFACE	°F	-100	+200	1	-	1.2
SA2380T	TEMP BAY 6 FUEL TANK SURFACE	°F	-100	+200	1	-	1.2
SC0030Q	QUANTITY H2 TANK 1	PCT	0	100	1	1	0.4
SC0031Q	QUANTITY H2 TANK 2	PCT	0	100	1	1	0.4
SC0032Q	QUANTITY O2 TANK 1	PCT	0	100	1	1	0.4
SC0033Q	QUANTITY O2 TANK 2	PCT	0	100	1	1	0.4
SC0037P	PRESS O2 TANK 1	PSIA	20	1080	1	1	4.0
SC0038P	PRESS O2 TANK 2	PSIA	20	1080	1	1	4.0
SC0039P	PRESS H2 TANK 1	PSIA	0	350	1	1	1.5
SC0040P	PRESS H2 TANK 2	PSIA	0	350	1	1	1.5
SC0041T	TEMP O2 TANK 1	°F	-325	+80	1	1	1.6
SC0042T	TEMP O2 TANK 2	°F	-325	+80	1	1	1.6
SC0043T	TEMP H2 TANK 1	°F	-425	-200	1	1	1.0
SC0044T	TEMP H2 TANK 2	°F	-425	-200	1	1	1.0
CC0175T	TEMP STATIC INVERTER 1	°F	+32	+248	1	-	1
CC0176T	TEMP STATIC INVERTER 2	°F	+32	+248	1	-	1
CC0177T	TEMP STATIC INVERTER 3	°F	+32	+248	1	-	1 - NL
CC0200V	AC VOLTAGE MAIN BUS 1 PHASE A	VAC	0	+150	10	1	0.6
CC0203V	AC VOLTAGE MAIN BUS 2 PHASE A	VAC	0	+150	10	1	0.6
CC0206V	DC VOLTAGE MAIN BUS A	VDC	0	+45	10	1	0.18
CC0207V	DC VOLTAGE MAIN BUS B	VDC	0	+45	10	1	0.18

NL - Non Linear

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
CC0210V	DC VOLTAGE BATTERY BUS A	VDC	0	+45	10	1	0.18
CC0211V	DC VOLTAGE BATTERY BUS B	VDC	0	+45	10	1	0.18
CC0215C	DC CURRENT BATT CHARGER OUT	AMP	0	+5	10	1	0.02
CC0222C	DC CURRENT BATTERY A	AMP	0	+100	10	1	0.4
CC0223C	DC CURRENT BATTERY B	AMP	0	+100	10	1	0.4
CC0224C	DC CURRENT BATTERY C	AMP	0	+100	10	1	0.4
CC0232V	DC VOLTAGE BATTERY RELAY BUS	VDC	0	+45	10	1	0.18
SC2060P	N2 PRESSURE FC 1 REGULATED	PSIA	0	75	10	-	0.3
SC2061P	N2 PRESSURE FC 2 REGULATED	PSIA	0	75	10	-	0.3
SC2062P	N2 PRESSURE FC 3 REGULATED	PSIA	0	75	10	-	0.3
SC2066P	O2 PRESSURE FC 1 REGULATED	PSIA	0	75	10	-	0.3
SC2067P	O2 PRESSURE FC 2 REGULATED	PSIA	0	75	10	-	0.3
SC2068P	O2 PRESSURE FC 3 REGULATED	PSIA	0	75	10	-	0.3
SC2069P	H2 PRESSURE FC 1 REGULATED	PSIA	0	75	10	-	0.3
SC2070P	H2 PRESSURE FC 2 REGULATED	PSIA	0	75	10	-	0.3
SC2071P	H2 PRESSURE FC 3 REGULATED	PSIA	0	75	10	-	0.3
SC2081T	TEMP FC 1 COND EXHAUST	°F	+145	+250	1	1	0.4
SC2082T	TEMP FC 2 COND EXHAUST	°F	+145	+250	1	1	0.4
SC2083T	TEMP FC 3 COND EXHAUST	°F	+145	+250	1	1	0.4
SC2084T	TEMP FC 1 SKIN	°F	+80	+550	1	1	2
SC2085T	TEMP FC 2 SKIN	°F	+80	+550	1	1	2
SC2086T	TEMP FC 3 SKIN	°F	+80	+550	1	1	2
SC2087T	TEMP FC 1 RADIATOR OUTLET	°F	-50	+300	1	1	1.4
SC2088T	TEMP FC 2 RADIATOR OUTLET	°F	-50	+300	1	1	1.5
SC2089T	TEMP FC 3 RADIATOR OUTLET	°F	-50	+300	1	1	1.5
SC2090T	RAD INLET TEMP FC 1	°F	-50	+300	1	-	1.5
SC2091T	RAD INLET TEMP FC 2	°F	-50	+300	1	-	1.5
SC2092T	RAD INLET TEMP FC 3	°F	-50	+300	1	-	1.5
SC2113C	DC CURRENT FC 1 OUTLET	AMP	0	+100	10	1	0.4

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
SC2114C	DC CURRENT FC 2 OUTPUT	AMP	0	+100.0	10	1	0.4
SC2115C	DC CURRENT FC 3 OUTPUT	AMP	0	+100.0	10	1	0.4
SC2139R	FLOW RATE H2 FC 1	LB/HR	0	.2	10	-	..001 - NL
SC2140R	FLOW RATE H2 FC 2	LB/HR	0	.2	10	-	.001 - NL
SC2141R	FLOW RATE H2 FC 3	LB/HR	0	.2	10	-	.001 - NL
SC2142R	FLOW RATE O2 FC 1	LB/HR	0	1.7	10	-	.005 - NL
SC2143R	FLOW RATE O2 FC 2	LB/HR	0	1.7	10	-	.005 - NL
SC2144R	FLOW RATE O2 FC 3	LB/HR	0	1.7	10	-	.005 - NL
SC2160X	PH FACTOR WATER COND FC 1		NORM	HIGH	10	1	
SC2161X	PH FACTOR WATER COND FC 2		NORM	HIGH	10	1	
SC2162X	PH FACTOR WATER COND FC 3		NORM	HIGH	10	1	
CC2962C	CSM TO LEM CURRENT MONITOR	AMP	0	+10	10	1	0.04
CD0005V	DC VOLTAGE PYRO BUS A	VDC	0	+40	10	-	0.15
CD0006V	DC VOLTAGE PYRO BUS B	VDC	0	+40	10	-	0.15
CD0023X	CM-SM RELAY CLOSE A			SEP	10	1	
CD0024X	CM-SM SEP RELAY CLOSE B			SEP	10	1	
CD0123X	SLA SEPARATION RELAY A			SEP	10	1	
CD0124X	SLA SEPARATION RELAY B			SEP	10	1	
CD0130X	HAND CONTROLLER INPUT A			ABORT	10	1	
CD0131X	HAND CONTROLLER INPUT B			ABORT	10	1	
CD0132X	EDS ABORT LOGIC INPUT NO 1		VOTE/ OFF	ARM	10	1	
CD0133X	EDS ABORT LOGIC INPUT NO 2		VOTE/ OFF	ARM	10	1	
CD0134X	EDS ABORT LOGIC INPUT NO 3		VOTE/ OFF	ARM	10	1	
CD0135X	EDS ABORT LOGIC OUTPUT A			ABORT	10	1	
CD0136X	EDS ABORT LOGIC OUTPUT B			ABORT	10	1	
CD0170X	RCS ACTIVATE SIG A			ENABLE	10	1	
CD0171X	RCS ACTIVATE SIG B			ENABLE	10	1	
CD0173X	CM RCS PRESS SIG A			PRESS	10	1	
CD0174X	CM RCS PRESS SIG B			PRESS	10	1	
CD0200V	DC VOLTAGE LOGIC BUS A	VDC	0	+ 40	10	-	0.15

NL - Non Linear

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
CD0201V	DC VOLTAGE LOGIC BUS B	VDC	0	+ 40	10	-	0.15
CD0230X	FWD HS JETTISON A			JETT	10	1	
CD0231X	FWD HS JETTISON B			JETT	10	1	
CD1154X	CSM-LEM LOCK RING SEP RELAY A			SEP	10	1	
CD1155X	CSM-LEM LOCK RING SEP RELAY B			SEP	10	1	
CE0001X	DROGUE DEPLOY RELAY CLOSE A			DEPLOY	10	1	
CE0002X	DROGUE DEPLOY RELAY CLOSE B			DEPLOY	10	1	
CE0003X	MAIN CHUTE DEPL DRG REL RLY A			DEPLOY	10	1	
CE0004X	MAIN CHUTE DEPL DRG REL RLY B			DEPLOY	10	1	
CE0321X	MAIN CHUTE DISCON-NECT RELAY A			DISC	10	1	
CE0322X	MAIN CHUTE DISCON-NECT RELAY B			DISC	10	1	
CF0001P	PRESSURE CABIN	PSIA	0	17	1	1	0.067
CF0002T	TEMP CABIN	°F	+40	+125	1	1	0.3 - NL
CF0003P	PRESS O2 SUIT TO CABIN DIFF	IN H2O	-5	+5	10	-	0.04
CF0005P	PRESS CO2 PARTIAL	MM HG	0	30.00	1	1	0.12 - NL
CF0006P	PRESS SURGE TANK	PSIA	30	1080	10	1	4
CF0008T	TEMP SUIT SUPPLY MANIF	°F	+20	+95	1	1	0.3
CF0009Q	QUANTITY WASTE WATER TANK	PCT	0	100	1	1	0.4 - NL
CF0010Q	QUAN POTABLE H20 TANK	PCT	0	100	1	1	0.3 - NL
CF0012P	PRESS SUIT DEMAND REG SENSE	PSIA	0	17	10	1	0.07
CF0015P	PRESS SUIT COM-PRESSOR DIFF	PSID	0	1.00	10	1	0.0035 - NL
CF0016P	PRESS GLYCOL PUMP OUTLET	PSIG	0	60	10	1	0.24
CF0017T	TEMP GLYCOL EVAP OUTLET STEAM	°F	+20	+95	1	-	0.3
CF0018T	TEMP GLY EVAP OUTLET LIQUID	°F	+25	+75	1	1	0.2
CF0019Q	QUANTITY GLYCOL ACCUM	PCT	0	107	10	1	0.5 - NL
CF0020T	TEMP SPACE RADI-ATOR OUTLET	°F	-50	+100	1	1	0.6 - NL
CF0034P	BACK PRESS GLYCOL EVAPORATOR	PSIA	0	0.25	1	-	0.0008
CF0035R	FLOWRATE ECS O2	LB/HR	0.16	1	10	-	0.003 - NL
CF0036P	PRESS OUTLET O2 REG SUPPLY	PSIG	0	150	10	1	0.6

NL - Non Linear

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
CF0070P	PRESS SEC GLYCOL PUMP OUTLET	PSIG	0	60	10	1	0.24
CF0071T	TEMP SEC EVAP OUTLET LIQUID	°F	+25	+75	10	1	0.2
CF0072Q	QUANTITY SEC GLYCOL ACCUM	PCT	0	100	10	1	0.8 - NL
CF0073P	PR SECONDARY EVAP OUT STEAM	PSIA	0.05	0.25	1	-	0.0008
CF0120P	PRESS H2O AND GLYCOL TANKS	PSIA	0	50	1	-	0.2
CF0157R	RATE GLYCOL FROM THERMAL LOAD	LB/HR	45	330	10	-	0.9 - NL
CF0181T	TEMP GLYCOL EVAP INLET	°F	+35	+100	1	-	0.3
SF0260T	TEMP PRIMARY RADI-ATOR INLET	°F	+55	+120	1	1	0.25
SF0262T	TEMP SECONDARY RADIATOR INLET	°F	+55	+120	1	1	0.25
SF0263T	TEMP SEC RADIATOR OUTLET	°F	+30	+70	1	1	0.15
SF0266X	RADIATOR FLOW CONT SYS 1 OR 2		SYS 1	SYS 2	10	1	
CF0460T	TEMP URINE DUMP NOZZLE	°F	0	+100	1	1	0.4
CF0461T	TEMP WASTE WATER DUMP NOZZLE	°F	0	+100	1	1	0.4
CG1040V	120 VDC PIPA SUPPLY DC LEVEL	VDC	+84	+135	1	-	0.2
CG1110V	2.5 VDC TM BIAS	VDC	0	5	1	1	0.02
CG1201V	IMU 28V .8KC 1 PCT	VRMS	0	30	1	-	0.12 - NL
CG1331V	3.2KC 28V SUPPLY	VRMS	0	30	1	-	0.12 - NL
CG1513X	28V IMU STANDBY		OFF	STBY	10	1	
CG1523X	28V CMC OPERATE		OFF	OPR	10	1	
CG1533X	28V OPTX OPERATE		OFF	OPR	10	1	
CG2112V	IG 1X RESOLVER OUT-PUT SIN	VRMS	-21	+21	10	-	0.17
CG2113V	IG 1X RESOLVER OUT-PUT COS	VRMS	-21	+21	10	-	0.17
CG2117V	IGA SERVO ERROR IN PHASE	VRMS	-3	+3	100	-	0.025
CG2142V	MG 1X RESOLVER OUT-PUT SIN	VRMS	-21	+21	10	-	0.16
CG2143V	MG 1X RESOLVER OUT-PUT COS	VRMS	-20	+40	10	-	0.16
CG2147V	MGA SERVO ERROR IN PHASE	VRMS	-3	+3	100	-	0.024
CG2172V	OG 1X RESOLVER OUT-PUT SINE	VRMS	-21	+21	10	-	0.16

NL - Non Linear

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
CG2173V	OG 1X RESOLVER OUT-PUT COS	VRMS	-21	+21	10	-	0.16
CG2177V	OGA SERVO ERROR IN PHASE	VRMS	-3	+3	100	-	0.025
CG2300T	PIPA TEMPERATURE	°F	+119	+140	1	1	0.08
CG3721V	SHAFT CDU DAC OUT-PUT	VRMS	-12	+12	10	-	0.09
CG3722V	TRUNNION CDU DAC OUTPUT	VRMS	-12	+12	10	-	0.09
CG5040X	CMC WARNING		WARN		10	1	
CH3500H	FDAI CM/SM ATT ERROR PITCH	DEG	-5 -15	+5 +15	50	-	MM
CH3501H	FDAI CM/SM ATT ERROR YAW	DEG	-5 -15	+5 +15	50	-	MM
CH3502H	FDAI CM/SM ATT ERROR ROLL	DEG	-5 -50	+5 +50	100	-	MM
CH3503R	FDAI SCS BODY RATE PITCH	DEG/SEC	-1 -5 -10	+1 +5 +10	100	-	MM
CH3504R	FDAI SCS BODY RATE YAW	DEG/SEC	-1 -5 -10	+1 +5 +10	100	-	MM
CH3505R	FDAI SCS BODY RATE ROLL	DEG/SEC	-1 -5 -50	+1 +5 +50	100	-	MM
CH3517H	GIMBAL POSITION PITCH 1 OR 2	DEG	-5	+5	100	-	0.04
CH3518H	GIMBAL POSITION YAW 1 OR 2	DEG	-5	+5	100	-	0.04
CH3546X	RCS SOLENOID ACT C3/13/X		FIRE/OFF	ARM	200	-	
CH3547X	RCS SOLENOID ACT A4/14/X		FIRE/OFF	ARM	200	-	
CH3548X	RCS SOLENOID ACT A3/23/-X		FIRE/OFF	ARM	200	-	
CH3549X	RCS SOLENOID ACT C4/24/-X		FIRE/OFF	ARM	200	-	
CH3550X	RCS SOLENOID ACT D3/25/X		FIRE/OFF	ARM	200	-	
CH3551X	RCS SOLENOID ACT B4/26/X		FIRE/OFF	ARM	200	-	
CH3552X	RCS SOLENOID ACT B3/15/-X		FIRE/OFF	ARM	200	-	
CH3553X	RCS SOLENOID ACT D4/16/-X		FIRE/OFF	ARM	200	-	
CH3554X	RCS SOLENOID ACT B1/11/Z		FIRE OFF	ARM	200	-	
CH3555X	RCS SOLENOID ACT D2/22/Z		FIRE/OFF	ARM	200	-	

MM - Multiple Mode Calibration

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
CH3556X	RCS SOLENOID ACT D1/21/-Z		FIRE/ OFF	ARM	200	-	
CH3557X	RCS SOLENOID ACT B2/12/-Z		FIRE/ OFF	ARM	200	-	
CH3558X	RCS SOLENOID ACT A1/Y		FIRE/ OFF	ARM	200	-	
CH3559X	RCS SOLENOID ACT C2/Y		FIRE/ OFF	ARM	200	-	
CH3560X	RCS SOLENOID ACT C1/-Y		FIRE/ OFF	ARM	200	-	
CH3561X	RCS SOLENOID ACT A2/-Y		FIRE/ OFF	ARM	200	-	
CH3574X	TRANSLATIONAL CONTROLLER XC MD		OFF	ON	10	1	
CH3575X	TRANSLATIONAL CONTROLLER-XC MD		OFF	ON	10	1	
CH3576X	TRANSLATIONAL CONTROLLER YC MD		OFF	ON	10	1	
CH3577X	TRANSLATIONAL CONTROLLER -YC MD		OFF	ON	10	1	
CH3578X	TRANSLATIONAL CONTROLLER ZC MD		OFF	ON	10	1	
CH3579X	TRANSLATIONAL CONTROLLER -SC MD		OFF	ON	10	1	
CH3582V	SCS TVC AUTO COM-MAND PITCH	VDC	-10	+10	100	-	0.08
CH3583V	SCS TVC AUTO COM-MAND PITCH	VDC	-10	+10	100	-	0.08
CH3585H	ROT CONTROL/MTVC PITCH	VDC	-10	+10	50	-	0.078
CH3586H	ROT CONTROL/MTVC YAW CMC	VDC	-10	+10	50	-	0.08
CH3587H	ROT CONTROL/MTVC ROLL CMC	DEG	-11	+11	50	-	0.087
CH3588X	ATTITUDE DEADBAND MINIMUM		MAX	MIN	10	1	
CH3590X	HIGH PRO RATE LIMIT		LOW	HIGH	10	1	
CH3592X	FDAI SCALE ERROR 5, RATE 5		OFF	ON	10	1	
CH3593X	FDAI SCALE ERROR 50/15, RT50/10		OFF	ON	10	1	
CH3600X	SCS DELTA V CG-LM/CSM POS		CSM	LM/ CSM	10	1	
CH3601X	DIR RCS SW NO 1 ENABLE POS		OFF	ENABLE	10	1	
CH3602X	DIR RCS SW NO 2 ENABLE POS		OFF	ENABLE	10	1	
CH3604X	SPS SOLENOID DRIVER NO 1		FIRE/ OFF	ARM	10	1	
CH3605X	SPS SOLENOID DRIVER NO 2		FIRE/ OFF	ARM	10	1	

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
CH3606X	LIMIT CYCLE SW OFF POS		ON	OFF	10	1	
CH3607X	SC CONTROL SOURCE SWITCH		CMC	SCS	10	1	
CH3609X	ROLL MAN ATT SW ACCEL CMD POS		OFF	ON	10	1	
CH3610X	R MAN ATT SW MIN IMP CMD POS		OFF	ON	10	1	
CH3612X	PITCH MAN ATT SW ACCEL CMD POS		OFF	ON	10	1	
CH3613X	P MAN ATT SW MIN IMP CMD POS		OFF	ON	10	1	
CH3615X	YAW MAN ATT SW ACCEL CMD POS		OFF	ON	10	1	
CH3616X	YAW MAN ATT SW MIN IMP CMD POS		OFF	ON	10	1	
CH3623X	GYRO 1 COMB SPIN MTRS RUN DET		LOW	NORM	10	1	
CH3624X	GYRO 2 COMB SPIN MTRS RUN DET		LOW	NORM	10	1	
CH3635X	BMAG MODE SW-ROLL ATT 1 RT 2		OFF	ON	10	-	
CH3636X	BMAG MODE SW-ROLL RATE 2		OFF	ON	10	-	
CH3638X	BMAG MODE SW-PITCH ATT 1 RT 2		OFF	ON	10	-	
CH3639X	BMAG MODE SW-PITCH RATE 2		OFF	ON	10	-	
CH3641X	BMAG MODE SW-YAW ATT 1 RT 2		OFF	ON	10	-	
CH3642X	BMAG MODE SW-YAW RATE 2		OFF	ON	10	-	
CH3666C	TVC PITCH DIFF CURRENT	MAMP	-800	+800	200	-	
CH3667C	TVC YAW DIFF CURRENT	MAMP	-800	+800	100	-	
CJ0060J	EKG COMMANDER LH COUCH	MV	NA	NA	200	-	
CJ0061J	EKG COMMANDER CTR COUCH	MV	NA	NA	200	-	
CJ0062J	EKG LM PILOT RH COUCH	MV	NA	NA	200	-	
CJ0200R	RESP RATE CMD, CM/LM PILOT	OHM	NA	NA	50	-	
CJ0201R	RESP RATE CM PILOT CTR COUCH	OHM	NA	NA	50	-	
CJ0202R	RESP RATE LM PILOT RH COUCH	OHM	NA	NA	50	-	
CK0026A	CM ACCEL X-AXIS	G	-2	+10	100	-	0.05
CK0027A	CM ACCEL Y-AXIS	G	-2	+2	100	-	0.016

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
CK0028A	CM ACCEL Z-AXIS	G	-2	+2	100	-	0.016
CK1051K	RADIATION DOSIMETER 1	VDC	0	5	10	-	0.02 - NL
CK1052K	RADIATION DOSIMETER 2	VDC	0	5	10	-	0.02 - NL
CK1053R	DOSIMETER RATE CHANGE	VDC	0	5	1	-	0.02 - NL
CK1043	70mm HASSELBLAD		OFF	ON	100	-	
CK1044	70mm LUNAR PHOTOGRAPHY		OFF	ON	100	-	
SP0001P	HE PRESS TANK	PSIA	0.	5000	10	1	21
SP0002T	HE TEMP TANK	°F	-100	+200	1	-	1.2
SP0003P	PRESS OXIDIZER TANKS	PSIA	0	250	10	1	1
SP0006P	PRESS FUEL TANKS	PSIA	0	250	10	1	1
SP0022H	POSITION FUEL/OX VLV 1 POT B	DEG	0	90	10	-	0.46
SP0023H	POSITION FUEL/OX VLV 2 POT B	DEG	0	90	10	-	0.46
SP0024H	POSITION FUEL/OX VLV 3 POT B	DEG	0	90	10	-	0.46
SP0025H	POSITION FUEL/OX VLV 4 POT B	DEG	0	90	10	-	0.46
SP0045T	TEMP ENG VALVE BODY	°F	0	+200	1	-	0.8
SP0048T	TEMP ENG FUEL FEED LINE	°F	0	+200	1	1	0.8
SP0049T	TEMP ENG OX FEED LINE	°F	0	+200	1	1	0.8
SP0054T	TEMP 1 OX DISTRIBUTION LINE	°F	0	+200	1	-	0.8
SP0057T	TEMP 1 FUEL DISTRIBUTION LINE	°F	0	+200	1	-	0.8
SP0061T	ENG INJECTOR FLANGE TEMP NO 1	°F	0	600	1	-	2.3
SP0062T	ENG INJECTOR FLANGE TEMP NO 2	°F	0	600	1	-	2.3
SP0600P	ENG VLV ACT SYS TANK PR PRI	PSIA	0	5000.	1	-	21
SP0601P	ENG VLV ACT SYS TANK PR SEC	PSIA	0	5000.	1	-	21
SP0655Q	QUAN OX TANK 1 PRI-TOTAL AUX	PCT	0	50	1	-	0.2
SP0656Q	QUAN OX TANK 2	PCT	0	60	1	-	0.2
SP0657Q	QUAN FUEL TANK 1 PRI-TOTAL AUX	PCT	0	50	1	-	0.2
SP0658Q	QUAN FUEL TANK 2	PCT	0	60	1	-	0.2
SP0661P	PRESS ENGINE CHAMBER	PSIA	0	150	100	-	0.6
SP0930P	PRESS FUEL SM/ENG INTERFACE	PSIA	0	300	10	-	1.3
SP0931P	PRESS OX SM/ENG INTERFACE	PSIA	0	300	10	-	1.3

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
CR0001P	HE PRESS TANK A	PSIA	0.	5000	1	1	21
CR0002P	HE PRESS TANK B	PSIA	0.	5000	1	1	21
CR0003T	HE TEMP TANK A	°F	0	+300	10	1	1.2
CR0004T	HE TEMP TANK B	°F	0	+300	10	1	1.2
CR0035P	PRESS CM-RCS HE MANIFOLD 1	PSIA	0	400	10	1	1.7
CR0036P	PRESS CM-RCS HE MANIFOLD 2	PSIA	0	400	10	1	1.7
SR5001P	HE PRESS TANK A	PSIA	0	5000	1	1	21
SR5002P	HE PRESS TANK B	PSIA	0	5000	1	1	21
SR5003P	HE PRESS TANK C	PSIA	0	5000	1	1	21
SR5004P	HE PRESS TANK D	PSIA	0	5000	1	1	21
SR5013T	HE TEMP TANK A	°F	0	+100	10	1	0.4
SR5014T	HE TEMP TANK B	°F	0	+100	10	1	0.4
SR5015T	HE TEMP TANK C	°F	0	+100	10	1	0.4
SR5016T	HE TEMP TANK D	°F	0	+100	10	1	0.4
SR5025Q	QUAN SM RCS PRO SYS A	VDC	0	5	1	1	0.02
SR5026Q	QUAN SM RCS PRO SYS B	VDC	0	5	1	1	0.02
SR5027Q	QUAN SM RCS PRO SYS C	VDC	0	5	1	1	0.02
SR5028Q	QUAN SM RCS PRO SYS D	VDC	0	5	1	1	0.02
SR5065T	TEMP ENGINE PACKAGE A	°F	0	+300	1	-	1.2
SR5066T	TEMP ENGINE PACKAGE B	°F	0	+300	1	-	1.2
SR5067T	TEMP ENGINE PACKAGE C	°F	0	+300	1	-	1.2
SR5068T	TEMP ENGINE PACKAGE D	°F	0	+300	1	-	1.2
SR5729P	A HE MANIFOLD PRESS	PSIA	0	400	10	1	1.7
SR5733P	OX MANIFOLD PR SYS A	PSIA	0	300	10	-	1.3
SR5737P	FUEL MANIFOLD PR SYS A	PSIA	0	400	10	1	1.7
SR5776P	B HE MANIFOLD PRESS	PSIA	0	400	10	1	1.7
SR5780P	OX MANIFOLD PR SYS B	PSIA	0	300	10	-	1.3
SR5784P	FUEL MANIFOLD PR SYS B	PSIA	0	400	10	1	1.7
SR5817P	C HE MANIFOLD PRESS	PSIA	C	400	10	1	1.7
SR5820P	OX MANIFOLD PR SYS C	PSIA	0	300	10	-	1.3
SR5821P	OX MANIFOLD PR SYS D	PSIA	0	300	10	-	1.3

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Continued.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
SR5822P	FUEL MANIFOLD PR SYS C	PSIA	0	400	10	1	1.7
SR5823P	FUEL MANIFOLD PR SYS D	PSIA	0	400	10	1	1.7
SR5830P	D HE MANIFOLD PRESS	PSIA	0	400	10	1	1.7
BS0080X	EDS ABORT REQUEST A		NORM	ABORT	10	1	
BS0081X	EDS ABORT REQUEST B		NORM	ABORT	10	1	
CS0150X	MASTER CAUTION-WARNING ON		WARN/OFF	NORM	10	1	
LS0200H	ANGLE OF ATTACK	PSID	0	5	10	-	0.017
CS0220T	TEMP DOCKING PROBE	°F	-100	+300	1	-	1.7
CT0012X	DSE TAPE MOTION MONITOR		OFF	MOTION	10	1	
CT0015V	SIG COND POS SUPPLY VOLTS	VDC	0	22	10	1	0.09
CT0016V	SIG COND NEG SUPPLY VOLTS	VDC	-22	0	10	1	0.09
CT0017V	SENSOR EXCITATION 5 VOLTS	VDC	0	5.5	10	1	0.02
CT0018V	SENSOR EXCITATION 10 VOLTS	VDC	0	11.	10	1	0.04
CT0120X	PCM BIT RATE CHANGE 8 BIT		LOW	HIGH	1	1	
CT0125V	PCM HI LEVEL 85 PERCENT REF	VDC	0	+5	10	1	0.02
CT0126V	PCM HI LEVEL 15 PERCENT REF	VDC	0	+5	10	1	0.02
CT0262V	UDL VALIDITY SIG 4-BIT		NA	NA	50	10	
CT0340X	PCM SYNC SOURCE EXT OR INT		INT	EXT	10	-	
CT0620E	S-BAND REC 1-2 AGC VOLTAGE	COUNTS	1	254	10	1	1 - NL
CT0640F	S-BAND RCVR 1-2 STATIC PH ERR	COUNTS	1	254	10	-	1 - NL
ST0820K	PROTON COUNT RATE CHANNEL 1	KHz	0	100	10	-	0.015 - NL
ST0821K	PROTON COUNT RATE CHANNEL 2	KHz	0	10	10	-	0.0015 - NL
ST0822K	PROTON COUNT RATE CHANNEL 3	KHz	0	10	10	-	0.0015 - NL
ST0823K	PROTON COUNT RATE CHANNEL 4	KHz	0	10	10	-	0.0015 - NL
ST0830K	ALPHA COUNT RATE CHANNEL 1	KHz	0	10	10	-	0.0016 - NL
ST0831K	ALPHA COUNT RATE CHANNEL 2	KHz	0	10	10	-	0.0015 - NL

TABLE B7-II.- COMMAND AND SERVICE MODULE TELEMETRY DATA SUMMARY - Concluded.

Measurement					Samples/Second		Units/Count
Number	Title	Unit	Approx. Range		High Bit Rate	Low Bit Rate	
			Low	High			
ST0832K	ALPHA COUNT RATE CHANNEL 3	KHz	0	10	10	-	0.0015 - NL
ST0838K	PROTON-ALPHA INTEGR COUNT RATE	KHz	0	100	10	-	0.015 - NL
ST0840T	TEMP NUCLEAR PARTICLE DET	°F	-120	+200	1	-	1.2 - NL
ST0841T	TEMP NUCLEAR PARTICLE ANALYZER	°F	-120	+200	1	-	1.2 - NL

NL - Non Linear

MISSION CONTROL

The Flight Director in Mission Control is supported by a team of specialists who are responsible for different aspects of spacecraft operation. These specialists are located in Mission Control and sit in front of console displays which provide real-time telemetry data. Each specialist is in voice contact with a group of support personnel in adjacent rooms who also have access to real-time telemetry data. See Appendix A, Part A4 for a description of the organization of Mission Control.

The display console for the CSM Electrical and Environmental Engineer (EECOM) is shown in figure B7-7 and is representative of the type of displays available to all the specialists in the Mission Control Center. The two television monitors on the console are used to display real-time telemetry data. Although various data formats are available to the EECOM, the two displays most frequently in use are shown in figures B7-8 and B7-9. These displays are updated once per second.

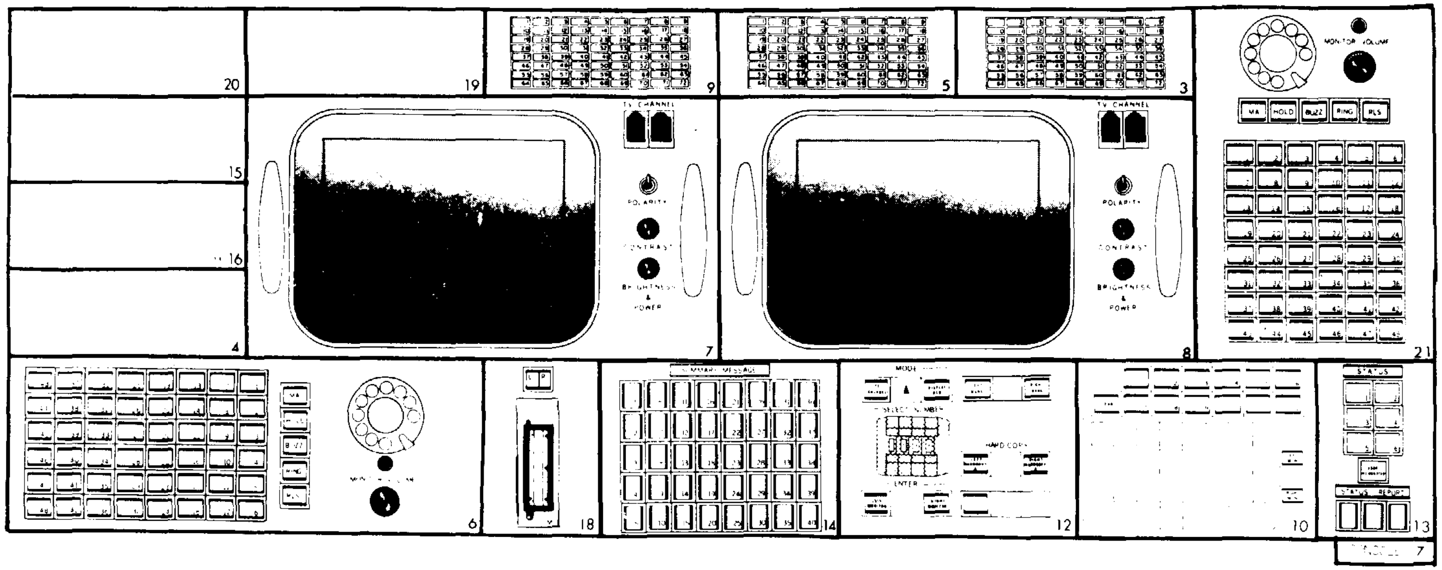
As an aid in recognizing out-of-tolerance parameters and spacecraft events, three groups of event indicators are provided at the top of the console. The lights on these panels which alert the EECOM to out-of-tolerance parameters are referred to as limit sense lights. A limit sense light comes on whenever the parameter in question falls outside of high and low limits which are manually set by the EECOM for that particular parameter. Among the 72 lights on panel 3, there are a total of 12 limit sense lights for pressure, temperature, and quantity in each cryogenic oxygen and hydrogen tank. In normal operation, the EECOM sets fairly tight limits on the limit sense lights in order to get an immediate indication of parameter variations. Consequently, it is not unusual for several limit sense lights to be burning.

Besides the limit sense lights, there are lights which indicate spacecraft events. One of these, located in the upper row of panel 9, indicates the presence of a master caution and warning in the spacecraft.

The following is a list of the system specialists in Mission Control:

(a) Retrofire Officer (RETRO) - responsible for abort planning, deorbit/entry times, and landing point prediction.

(b) Flight Dynamics Officer (FIDO) - responsible for coordinating and participating in mission planning and the control of the trajectory aspects of the mission, including powered flight trajectory, abort, and orbital GO/NO GO decision.



Location	Description	Location	Description
3	Event indicator	10	Display request keyboard
5	Event indicator	12	Manual select keyboard
6	Voice communication position	13	Status/status report
7	Precision TV monitor	14	Summary message inable keyboard
8	Precision TV monitor	18	Analog meter
9	Event indicator	21	Voice communication position

Figure B7-7.- CSM EECOM engineer console.

LM1885 CSM EPS HIGH DENSITY 0518

CTE 055:46:51 ( ) GET 55:46:53 ( ) SITE

DC VOLTS		
CC0206	VMA	29.5
CC0207	VMB	29.4
CC0210	VBA	36.4
CC0211	VBB	39.5*
CC0232	VBR	35.8
CD0200	VMLA	0.15
CD0201	VMLB	0.15
CD0005	VMQA	0.15
CD0006	VMQB	0.15
DC AMPS		
	TOT SC	67.7
	TOT FC	67.6
	FC PCT SC	100.0
	TOT BAT	0.0
	BAT PCT SC	
SC2113	FC 1	21.4
SC2114	FC 2	21.3
SC2115	FC 3	24.9
CC0222	BAT A	0.0
CC0223	BAT B	0.0
CC0224	BAT C	0.0
CC0215	CHRGR	1.12*
CC2962	LM	1.6
SC2160	PH 1	Low
SC2161	PH 2	Low
SC2162	PH 3	

AC VOLTS			
CC0200	AC	1	115.6
CC0203	AC	2	115.7
FC PSIA			
SC2060	1	N2	55.8
SC2061	2	N2	53.9
SC2062	3	N2	54.4
SC2066	1	O2	64.6
SC2067	2	O2	62.7
SC2068	3	O2	63.5
SC2069	1	H2	64.7
SC2070	2	H2	62.9
SC2071	3	H2	63.4
1	O2-N2	ΔP	8.8
2	O2-N2	ΔP	8.8
3	O2-N2	ΔP	9.1
1	H2-N2	ΔP	8.9
2	H2-N2	ΔP	9.0
3	H2-N2	ΔP	9.1
FC LB/HR			
SC2139	1	H2	.0659
SC2140	2	H2	.0679
SC2141	3	H2	.0739
SC2142	1	O2	0.488
SC2143	2	O2	0.507
SC2144	3	O2	0.550

FC °F			
SC2084	1	SKN	409.1
SC2085	2	SKN	412.7
SC2086	3	SKN	414.5
SC2081	1	TCE	158.0
SC2082	2	TCE	158.9
SC2083	3	TCE	157.1
FC RAD °F			
SC2087	1	OUT	70
SC2088	2	OUT	71
SC2089	3	OUT	75
SC2090	1	IN	86
SC2091	2	IN	88
SC2092	3	IN	95
PCT TOTAL FC LOAD			
	FC	1	31.6
	FC	2	31.6
	FC	3	36.9
INST			
CT0120	PCM		HBR
CT0125	4.25		4.249
CT0126	0.75		.731
CT0340	TMG		CTE
CT0015	+20		20.1
CT0016	-20		-20.0
CT0017	+5		5.03
CT0018	+10		10.1
CT0620	SS		
CS0220	PROBE		312 *

* Batt B Charging

	1	2	3
CC0175/76/77 INV TMPS	90	88	73

(09-54)

Figure B7-8.- Electrical power system parameters display.

LM1839 CSM ECS-CRYO TAB 0613

CTE 055:46:51 ( ) GET 055:46:53 ( ) SITE

LIFE SUPPORT

GF3571	LM CABIN P	PSIA	
CF0001	CABIN P	PSIA	5.1
CF0012	SUIT P	PSIA	4.3
CF0003	SUIT ΔP	IN H2O	-1.68
CF0015	COMP ΔP	P PSID	0.30
CF0006	SURGE P	P PSIA	891
	SURGE QTY	LB	3.67
	O2 TK 1 CAP ΔP	PSID	21
	O2 TK 2 CAP ΔP	PSID	17
CF0036	O2 MAN P	PSIA	105
CF0035	O2 FLOW	LB/HR	0.181
CF0008	SUIT T	°F	50.5
CF0002	CABIN T	°F	65
CF0005	CO2 PP	MMHG	1.5

H2O

CF0009	WASTE	PCT	24.4
	WASTE	LB	13.7
CF0010	POTABLE	PCT	104.5
	POTABLE	LB	37.6
CF0460	URINE NOZ T	°F	70
CF0461	H2O NOZ T	°F	72

PRIMARY COOLANT

CF0019	ACCUM QTY	PCT	34.4
CF0016	PUMP P	PSID	45.0
SF0260	RAD IN T	°F	73.8
CF0020	RAD OUT T	°F	35
CF0181	EVAP IN T	°F	45.7
CF0017	STEAM T	°F	64.9
CF0034	STEAM P	PSIA	.161
CF0018	EVAP OUT T	°F	44.2
SF0266	RAD VLV 1/2		ONE
CF0157	GLY FLO	LB/HR	215

SECONDARY COOLANT

CF0072	ACCUM QTY	PCT	36.8
CF0070	PUMP P	PSID	9.3
SF0262	RAD IN T	°F	76.5
SF0263	RAD OUT T	°F	44.6
CF0073	STEAM P	PSIA	.2460
CF0071	EVAP OUT T	°F	66.1
CF0120	H2O-RES	PSIA	25.8
	TOTAL FC CUR	AMPS	

CRYO SUPPLY

			O2-1	O2-2	H2-1	H2-2
SC0037-38-39-40	P	PSIA	913	908	225.7 (03-1)	235.1
SC0032-33-30-31	QTY	PCT	77.63	01.17	73.24	74.03
SC0041-42-43-44	T	°F	-189	-192	-417	-416
	QTY	LBS	251.1	260.0	20.61	20.83

Figure B7-9.- Cryogenic system display.

(c) Guidance Officer (GUIDO) - responsible for the utilization of the guidance and navigation system, correlation of inertial alignment, and evaluation of terminal phase actions in support of rendezvous.

(d) CSM Electrical, Environmental, and Communications Engineer (EECOM) - responsible for monitoring and evaluating the performance of the electrical power, environmental control, instrumentation, and sequential systems of the command and service modules.

(e) CSM Guidance and Navigation Officer (GNC) - responsible for monitoring and evaluating the performance of the guidance and navigation, propulsion, and stabilization and control systems of the command and service modules.

(f) LM Electrical, Environmental, and EMV Officer (TELMU) - responsible for monitoring and evaluating the performance of the primary guidance and navigation, abort guidance, control electronics, ascent propulsion, descent propulsion, and reaction control systems of the lunar module.

(g) LM Control Officer (CONTROL) - responsible for monitoring and evaluating the performance of the electrical, communications, instrumentation, sequential, and environmental control systems of the lunar module.

(h) Instrumentation and Communication Officer (INCO) - responsible for monitoring and evaluating the performance of spacecraft communications systems.

(i) Procedures Officer (PROCEDURES) - responsible for the detailed procedures implementation of Mission Control.

(j) Flight Activities Officer (FAO) - responsible for the detailed implementation of the flight plan and its revision.

(k) Aeromedical Officer (SURGEON) - directs all operational medical activities concerned with the mission.

The following table lists the members of the White and Black Mission Control teams. The White Team was on duty at the time of the accident, and many of the Black Team members were in Mission Control preparatory to their going on duty about an hour later.

Position	White	Black
Flight Director	E. F. Kranz	G. S. Lunney
Asst. Flt. Dir.	J. M. Leeper	L. W. Keyser
RETRO	B. T. Spencer	T. E. Weichel
FIDO	W. M. Stoval	W. J. Boone
GUIDO	W. E. Fenner	J. G. Renick
EECOM	S. A. Liebergot	W. C. Burton
GNC	B. N. Willoughby	J. A. Kamman
TELMU	R. H. Heselmeyer	W. M. Merritt
CONTROL	L. W. Strimple	H. A. Loden
INCO	G. B. Scott	T. L. Hanchett
PROCEDURES	J. R. Fucci	E. W. Thompson
FAO	E. B. Pippert	T. R. Lindsey
SURGEON	W. R. Hawkins	G. F. Humbert

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4. Anon.: Mission Director's Summary Report, Apollo 13. Manned Spacecraft Center, April 20, 1970.

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