REPORT ON THE ENGINEERING ANALYSIS OF THE CSM 109 APOLLO 13 MISSION

July, 1970

Flight Test Requirements and Evaluation Unit 093-201

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1.0 SUMMARY

The primary purposes of the Apollo 13 mission were to investigate the lunar environment, to emplace ALSEP III, to obtain lunar material samples, and to enhance the capability for manned lunar exploration. The planned landing site was in the Fra Mauro highlands. The planned mission was aborted at 56 hours because a conflagration in one of the cryogenic oxygen storage tanks resulted in loss of the oxygen in both tanks. After entering the lunar module and powering up the lunar module systems, the crew shut down all CSM systems. A circumlunar profile was executed as the most efficient manner of earth return, with the lunar module providing power and life support until two hours prior to entry.

The space vehicle, with a crew of James A. Lovell, commander, Fred W. Haise, Jr., lunar module pilot, and John L. Swigert, Jr., command module pilot, was launched from the Kennedy Space Center at 19:13:00 GMT, April 11, 1970. The only unexpected occurrence during the launch phase was an early shutdown of the center-engine of the S-II stage. The shutdown was caused by excessive vibration. The launch vehicle guidance computer successfully controlled the burning of the remaining S-II engines and the S-IVB and a normal earth orbit was achieved. Translunar injection by the S-IVB was normal. Following separation, the auxiliary propulsion system was used to maneuver the S-IVB to a trajectory which culminated in a successful impact on the lunar surface 74 nautical miles from the Apollo 12 seismic detector.

The first midcourse correction transferred the spacecraft to a nonfree-return trajectory. The mission continued smoothly until approximately 56 hours, when an electrical short in the fan motor wiring inside O_2 tank no. 2 started a fire which ruptured the vacuum dome of the tank and resulted in complete loss of all service module cryogenic oxygen and fuel cell power. The CSM systems were powered down to conserve the batteries and the lunar module was used as a lifeboat. All subsequent midcourse maneuvers were performed with the lunar module descent stage propulsion systems.

Constant vigilance by the crew and a widespread network of ground support personnel resulted in efficient rationing and use of consumables and ingenious usage of equipment for purposes for which it was not designed.

The service module was retained until 4 hours and 39 minutes before entry. The lunar module was retained until 1 hour and 10 minutes before entry. The command module systems were powered up with the entry batteries on the buses 2 hours and 30 minutes before entry. Entry was smooth and normal and the command module landed one mile from the target point in full view of the recovery ship and world-wide television.

Stable passive thermal control could not be maintained during the later part of the mission and this, in conjunction with the powered-down condition of the lunar and command modules, resulted in cabin temperatures as low as 43F and 52F for the command and lunar modules, respectively. Nevertheless, all systems performed perfectly when called upon after powerup. Although the command module IMU heaters were de-energized for 80 hours and reached an estimated low temperature of 55F, the G&N system successfully guided the command module to a near-perfect landing on target.

Low temperatures, a noisy environment, and stress, severely limited the ability of the crew to sleep during the last 96 hours of the mission. Postflight physical examinations were conducted immediately after recovery and were normal, although all crewmen were extremely fatigued and had lost weight.

The Apollo 13 mission was the first to require an emergency abort. Postflight appraisal of the conduct of the mission resulted in the following conclusions:

- 1. The lunar module systems demonstrated an emergency operational capability which, although inherent in their design, was unproven in previous flights. The lunar module systems supported the crew for a period approximately twice their intended design lifetime.
- 2. The effectiveness of preflight crew training, especially in conjunction with ground personnel, was reflected in the skill and precision with which the crew responded to the emergency.
- 3. The Mission Control Center and its network of analytical support personnel proved to be adequate in solving the unique problems associated with the total loss of redundant oxygen supplies and the attendant loss of primary power in the command and service modules. The required realtime planning was conducted in a timely manner such that a potentially catastrophic ending was avoided.
- 4. Although the mission must technically be classed as a failure, three planned experiments (lightning phenomena, earth photography, and S-IVB lunar impact) were completed; and information was obtained regarding the long-term backup capability of the lunar module and the operational characteristics of the command module systems after extended exposure to a cold-soak, powered-down, environment, which would not otherwise have been available.

This report deals only briefly with the postflight investigations into the cause of the fire in the oxygen tank. The investigations are fully reported in the NASA Report of the Apollo 13 Review Board, the NASA Apollo 13 Cryogenic Oxygen Tank 2 Anomaly Report, and the NR Apollo 13 Engineering Summary Report, SD 70-243-2.

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The mission profile is depicted in Figure 1.1





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2.0 MISSION DESCRIPTION

Apollo 13 was launched from Pad 39A, Kennedy Space Center, Florida, at 19:13:00 GMT, April 11, 1970. The final countdown was smooth and liftoff was normal. First-stage performance was normal but high-amplitude longitudinal oscillations developed during the S-II burn. S-II crossbeam oscillations reached a peak amplitude of ±33.7g and corresponding center-engine chamber pressure oscillations of ±225 psi initiated centerengine cutoff 132 seconds earlier than planned. The outboard engines burned 35 seconds longer than planned in final compensation and the S-IVB burned 9 seconds longer than planned in final compensation. An earth parking orbit of 100.2 by 98.0 nautical miles was attained. The CSM subsystems were checked out satisfactorily in earth orbit. The optics dust cover failed to jettison when the optics were driven to the angles prescribed in the crew check list but jettisoned automatically when a P52 alignment was made.

Translunar injection at 02:35:46 inserted the spacecraft into a free-return orbit with a pericynthion of 415.8 nautical miles. CSM separation, transposition and docking, and CSM/LM ejection were all conducted smoothly. An evasive maneuver conducted with the S-IVB auxiliary propulsion system (APS) removed the S-IVB from any possible collision path with the spacecraft. At 06:00:00, the APS was used again to maneuver the S-IVB to achieve impact on the lunar surface to provide calibrated seismological data.

Passive thermal control (FTC) was initiated at 7:42:02 but a divergent coning angle necessitated re-initiation 39 minutes later. The roll rate was 0.3 deg/sec (3 rev/hr). FTC was terminated at approximately 29:25:00 in preparation for the hybrid transfer burn.

A 3.5-second SPS burn at 30:40:50 transferred the spacecraft to a non-free-return trajectory with a closest approach to the lunar surface of 62 nautical miles and a closest approach to earth of 2300 nautical miles. A 49-minute color TV transmission which commenced at 30:13:00 included coverage of crew activities during the burn. PTC was initiated at 32:21:49, but an unplanned minimum-impulse engine firing necessitated re-initiation at 32:59:00. The roll rate was 3 rev/hr. PTC was terminated at approximately 55:00:00 to provide high-gain antenna coverage for TV transmission.

The quantity sensor in the number two oxygen tank failed at 46:40:09, when its output dropped to zero. This failure was not considered serious at the time because alternative means to calculate tank quantity were available.

The flight plan called for entry to the lunar module at approximately 57:00:00, but it was decided to enter three hours earlier to read the supercritical helium (SHe) tank pressure, which had been rising at an abnormally high rate during the countdown. The lunar module pilot entered

the lunar module at 54:25:00, followed shortly afterwards by the commander. The lunar module systems were checked out and the SHe tank pressure was found to be satisfactory. A 31-minute TV transmission began at 55:15:00. The lunar module pilot and the commander returned to the command module at 55:40:00.

At 55:52:58 the crew were requested to cycle the fans in the cryogenic storage tanks. At 55:54:53, the O2 tank no. 2 pressure reading fell to zero, a main bus B undervoltage alarm occurred, the crew heard a loud bang, the computer restarted, and several SM-RCS barber-pole indications occurred. Postflight analysis showed that an electrical short-circuit in the fan motor wiring in 02 tank no. 2 had started a fire in the tank. The resulting rapid increase in pressure ruptured the vacuum dome of the tank and released the pressurized oxygen into bay 4. The resulting pressure wave blew away the whole of the skin panel of bay 4. The shock also created a serious leak in the O2 tank no. 1 supply lines and closed the oxygen reactant valves of fuel cells nos. 1 and 3. These two fuel cells continued to supply power for over two minutes by using the oxygen trapped in the supply lines between the reactant valves and the fuel cells. The fuel cells failed when the oxygen was depleted and main bus B (which was tied to fuel cell no. 3) dropped to zero volts. Main bus A was tied to fuel cells no. 1 and 2 and failure of fuel cell no. 1 placed the entire spacecraft load on fuel cell no. 2. Main bus A voltage dropped to 25.5 vdc and entry battery A was placed on the bus to support fuel cell no. 2.

It was obvious at this stage that the mission had to be aborted and it was decided to use the lunar module and its life-support systems as a lifeboat for the crew. Emergency procedures were devised by ground support personnel and were read to the crew. The command module O2 surge tank was isolated from the service module to ensure retention of sufficient oxygen for entry. Lunar module power-up was completed at 57:41:00. Fuel cell no. 2 continued to operate until the pressure in O2 tank no. 1 decayed below the required inlet level of the fuel cell oxygen pressure regulator at 58:15:00. Battery A supported the fuel cell before it was shutdown and carried the reduced CSM electrical loads until CSM powerdown was completed at 58:39:00. Power was removed from all CSM systems, including the IMU heaters. The lunar module platform was co-aligned with the command module platform before CSM powerdown.

Loss of fuel cell power precluded further use of the SPS and subsequent maneuvers were performed with the lunar module descent stage propulsion systems. The main descent propulsion system (DPS) was used at 61:29:43 for a 34-second burn to change the trajectory from non-free return to free-return with a closest approach to the lunar surface of 137 nautical miles. The command module landing point for the new trajectory was in the Indian Ocean south of Mauritius. The predicted landing time was 152:00:00. The DPS was used again two hours after pericynthion to move the landing point to the Pacific Ocean and shorten the return time by nine hours. The 264-second maneuver produced a velocity change of 860 ft/sec.

PTC was established with the lunar module RCS at approximately 63:26:00 by rotating the lunar module 90 degrees in the yaw plane approximately once each hour, with attitude hold control between maneuvers. (Equivalent to CSM roll, because the lunar module yaw plane is parallel to the command module roll plane). This mode was maintained until 73:26:00, when the spacecraft was maneuvered to the attitude required for the second DPS burn. Following the second DPS burn, PTC was established by maneuvering to the required attitude, damping all rates, and using 30 clicks of right yaw to establish an estimated rotational rate of one revolution per 15 to 18 minutes. The resulting apparent moon and earth motion was horizontal with respect to the lunar module windows. After PTC was established, the lunar module was partly powered down to conserve battery energy. PTC was terminated at 104:30:00 to maneuver the spacecraft for a third midcourse correction to improve the entry angle. The correction was provided by a 14-second burn of the DFS engine at 105:18:28. PTC was reestablished at 105:38:00 with 12 clicks of right yaw, with some degree of roll-pitch coupling present. The lunar module SHe tank burst disc ruptured (as expected) at 108:54:00 and unexpectedly reversed the rotation of the spacecraft. Some pitch motion was introduced also. The command module pilot timed the resulting rotational rate as two revolutions in 3 minutes and 50 seconds (31 deg/sec). The crew reported that the rate was not uncomfortable and no attempt was made to reestablish a more stable PTC motion. The rate of rotation slowly decreased and was timed at one revolution in 11 or 12 minutes (0.5 deg/sec), at 132:53:46. The attitude had changed and the command module minus-X axis was then pointed to the sun, with the result that the command and lunar modules were completely shadowed by the service module and were cold. At approximately 134:00:00, the spacecraft was early-maneuvered to the burn attitude for the final midcourse correction to allow the sun's rays to enter the lunar module windows and provide heat. The midcourse correction was made at 137:39:52 with a 21second burn of the LM-RCS engines.

The command module IMU heaters and the command module computer were energized at 138:24:00, following a spacecraft maneuver to the service module separation attitude. The command module RCS was pressurized and the thrusters were fired individually. The service module was jettisoned at 138:01:39. A plus-X translation of 0.5 ft/sec was performed with the lunar module RCS immediately prior to separation, followed by a minus-X translation of 0.5 ft/sec immediately the pyro actuation was heard. The spacecraft was then pitched up and the service module was photographed from an initial distance of approximately 80 feet.

Electrical power had been supplied by the lunar module batteries to command module main bus B since 112:06:00. This power was used to recharge entry batteries A and B. Lunar module power was removed at 140:10:00 and the three entry batteries were connected to the command module buses. The command module systems were then reactivated for entry.

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The command and lunar modules were maneuvered to the separation attitude with the lunar module RCS. Hatch closeout in both spacecraft was normal and command module hatch integrity was tested by venting the tunnel to a differential pressure of 3.5 psi. The residual pressure in the tunnel was used to provide separation velocity at undocking. Undocking and separation were accomplished at 141:30:00.

The pre-entry check and initialization of the entry monitor system (EMS) were normal. The 0.05g light was not illuminated to show automatic activation of the EMS three seconds after 0.05g was indicated by the computer and the EMS was started manually by the crew. The guided entry was normal in all respects and the command module landed, and remained, in the Stable I position, approximately one mile from the target point. Recovery operations were expedited and the crew were aboard the recovery ship, USS Iwo Jima, 45 minutes after landing.

Because all service module oxygen and fuel cell power had been lost, all subsequent effort was devoted to ensuring the early and safe return of the crew. Consumables were rationed and monitored and detailed procedures were prepared by ground support personnel and verified in the spacecraft simulators. Food supplies were adequate. Water rationing was necessary because fuel cell water production capability had been lost. Potable water was obtained by periodically pressuring the potable tank with surge-tank oxygen and withdrawing water until the pressure equalized. The drawn water was stored in 8-ounce juice bags and used solely for drinking and rehydrating juices. Approximately 3 bags of juice per day were consumed by each crewman. No water was expended in rehydratable foods, since there was an ample supply of both prepared wetpacks and nonrehydratable foods (breads, brownies, cubes, etc.). The crew reported that the juice bags contained 15 to 20 percent gas, which made it difficult to drink from the bags but was not enough to cause distress, such as abdominal cramping or nausea. Approximately 13.5 pounds of water were drawn from the command module potable tank, which had a capacity of 37.8 pounds. The crew reported at 125:19:12 that they could not draw any more water from the tank and presumed it to be empty. (The tank was not empty, since 24.3 pounds of water were drained from it during postflight test operations). The crew drew water from the lunar module descent stage supply for the remaining 17 hours of the mission.

The lithium hydroxide (LiOH) cartridges in the lunar module were not adequate to support the aborted mission, even though allowable CO₂ levels were extended to a partial pressure of 15mm Hg. A way to adapt and use the command module LiOH cartridges was worked out and tested on the ground and detailed instructions for the adaptation were given to the crew. The adaptation consisted of taping an LiOH cartridge to each of the lunar module suit intake hoses. The commander's hoses were placed in the tunnel area to provide fresh oxygen to the command module, and the lunar module pilot's hoses were positioned in the lunar module. A second cartridge was later added in series with each of those initially installed. The improvised CO₂ removal system maintained CO₂ partial pressure below lmm Hg. It was necessary to preserve the energy remaining in the command module entry batteries for use during the period between lunar module jettison and recovery. All CSM systems were therefore powered down. The lunar module systems were partly powered down to conserve lunar module battery energy. It became apparent after a period of stabilization that sufficient lunar module power was available to recharge the command module batteries and the necessary procedures were devised and transmitted to the crew. Both battery A and battery B were successfully recharged.

One urine dump was made through the side hatch auxiliary dump nozzle. This dump clouded the hatch window, interfered with optical sightings, and perturbed trajectory dynamics. The crew was therefore requested to not make any further dump. All subsequent urine collections were stored onboard. The containers utilized were the six lunar module urine transfer bags, three command module backup waste bags, the condensate container, two water collection bags from the portable life support system, and three urine collection devices. The command module waste stowage compartment appeared to be full with only seven fecal bags stowed in the area. The stiffness of the outer fecal bags added to the stowage problem.

Following the first DPS maneuver, a schedule was provided which kept either the commander or the lunar module pilot on watch at all times. The command module was used as sleeping quarters until the cabin temperature became extremely uncomfortable. The crew then attempted to sleep in the lunar module or the tunnel, but the temperature in these areas also dropped too low for prolonged, sound, sleep. In addition, lunar module coolant pump noise, stress, and frequent communications with the ground further hindered sleep. The total accumulated sleep obtained by each crewman during the 96 hours from the end of the scheduled sleep at 47:00:00 (prior to the incident) is estimated to have been 11, 12, and 19 hours for the commander, command module pilot, and lunar module pilot, respectively.

Cabin temperatures decreased to 43F in the command module and 52F in the lunar module. The crew reported heavy condensation on the command module windows and the formation of a thin film of water on the command module walls. Moisture also appeared on the lunar module windows but disappeared shortly after powerup at approximately 135 hours. The condensation generally disappeared after parachute deployment, although the structure remained cold even after landing.

Lunar Module Impact in Pacific Ocean

The lunar module entered the atmosphere and impacted in the open sea between Samoa and New Zealand at 25.5 degrees south latitude and 176 degrees west longitude, with surveillance aircraft in the area.

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S-IVB Impact on Lunar Surface

The S-IVB was impacted on the lunar surface to provide calibrated data from the seismic equipment deployed during the Apollo 12 mission.

The S-IVB impacted at 13:09:41 GMT, April 14, 1970, travelling at a speed of 5600 nautical miles/hr. Stage weight at impact was 30,700 pounds. The coordinates of the impact point were 2.4 degrees south latitude and 27.9 degrees west longitude. Impact occurred 74 miles westnorthwest from the Apollo 12 experiments station. The energy release was equivalent to an explosion of 7.7 tons of TNT. Seismic signals were first recorded 28.4 seconds after impact, and continued for over 4 hours. An expanding gas cloud, which presumably swept over the lunar surface from the impact point, was recorded by the lunar ionosphere detector. Detection of the cloud began 8 seconds before the first seismic signal and lasted 70 seconds.

3.0 DATA COVERAGE

The Apollo 13 CSM was completely powered down at 58:39:00 and CSM data transmission was therefore limited to the period before shutdown; three minutes at 102 hours; two minutes at 123 hours; and the pre-entry and entry period from 140:10:00 to 142:45:00. The DSE was not used during entry and data were not acquired during the blackout period and between 142:45:00 and landing at 142:54:41. Lunar module data and voice transmissions were received continuously between lunar module powerup at approximately 57 hours and lunar module jettison at 141:30:00. Communications were maintained from translunar injection to 142:39:00 through the primary ground stations at Goldstone, California; Madrid, Spain; and Honeysuckle Creek, Australia. During entry, the ground station at Carnarvon, Australia, maintained communication with the command module for a few minutes after Honeysuckle Creek lost contact. One of the Apollo range instrumentation aircraft (ARIA) acquired the last few minutes of data.

The quality of the PCM data received and recorded at the various sites was generally good. The panel from bay 4 of the service module, which was blown away by the pressure wave associated with failure of the oxygen tank, struck the high-gain antenna and caused data breakup and loss for approximately two seconds immediately following the failure.

The available PCM data were relayed from the MSFN stations to Mission Control Center at reduced sampling rates and displayed in real time. The MSFN Data were recorded by the Computing and Analysis Division and, at four-hour intervals, the recordings were bandpassed, converted to engineering units, and listed for use by the MSC and NR systems specialists who supported the mission in Building 45, MSC. A microfilm copy of each listing was made for NR and these films were shipped by air to Downey each evening for use in the Mission Support Room. The MSFN data proved adequate for preliminary analysis.

Summary TWX's listing single-sample data from the major subsystems were rebroadcast by MCC at approximately 15-minute intervals when available, and were received in the MSR-Downey

Tracking data from liftoff to S-IVB/CSM separation were processed by MSFC. Tracking data from S-IVB separation to splashdown were processed by MSC. The resulting trajectory data were supplied to NR on magnetic tape.

The ground station recordings of PCM data acquired within the North American continental area were forwarded to MSC, where they were duplicated as required by MSC or NR. The recordings of data acquired at ground stations outside the North American continental area were forwarded direct to the Goddard Space Flight Center (GSFC), Baltimore, where duplicates were made at the request of MSC. The MSFN data acquired in realtime were considered adequate for general mission analysis and duplicates of tapes were requested only when it was necessary to conduct a detailed analysis of dynamic activity or to investigate system anomalies.

Copies of 19 magnetic tapes were supplied for processing by NR at Downey. These included requested tapes covering the launch phase, TLI, MCC₁, and entry. The other tapes were supplied principally to provide additional data for investigation of the oxygen tank failure and a high-gain antenna anomaly, and data on system status at 102 and 123 hours.

The tape which covered the period around the oxygen tank failure was processed with a number of different techniques in order to extract the maximum amount of information. These included normal bandpæss filtering, zero-aperture bandpass filtering, listing of all data points without filtering, Hogan plots, and oscillograms. The PCM wavetrain for the data breakup period was reproduced on an oscillogram and, in a final effort to extract additional data points, the wavetrain was listed as a serial binary bit-stream and was manually decommutated and converted to engineering units. A few additional and valuable data points were recovered, but it was abundantly clear that most of the data transmitted were not acquired, presumably because the high-gain antenna was momentarily knocked out of alignment with the ground station.

MSC also provided NR with microfilm copies of all flight data processed in their Computing and Analysis Division (CAAD). Key data were reproduced from these microfilm and distributed to subsystem evaluators as required. These data included look angles between the ground receiving stations and the spacecraft. The look-angle data were transformed by NR to optimum high-gain-antenna pitch, yaw, and gimbal angles to facilitate investigation of the high-gain antenna anomaly.

NR was also provided with films and photographs taken onboard the spacecraft and during recovery operations; the crew log; and transcripts of air-to-ground communications, DSE voice recordings, and crew self-debriefings.

The Apollo 13 data were impounded during investigation of the cryogenic oxygen storage system failure and postflight analysis was conducted, very successfully, on a semi-boiler-room basis in the Mission Support Room (MSR). Extensive use was made of the MSC microfilmed data and the two reader/printers in the MSR.

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4.0 TEST EVALUATION

4.1 SEQUENCE OF EVEN IS FOR THE APOLLO 13 MISS	SION	
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Event	GET Hr:Min:Sec
Range zero - 19:13:00 GMT, April 11, 1970	00:00:00
Liftoff	00:00:00.6
Mach 1	00:01:08.4
Maximum dynamic pressure (651.9 psf)	00:01:21.3
Maximum aerodynamic heating	00:01:35
S-IC center-engine cutoff	00:02:15.2
S-IC outboard-engine cutoff	00:02:43.6
S-IC/S-II separation	00:02:44.3
S-II engine ignition	00:02:45.0
Interstage jettison	00:03:14.3
Launch escape tower jettison	00:03:21.0
S-II center-engine cutoff	00:05:30.6
S-II outboard-engine cutoff	00:09:52.6
S-II/S-IVB separation	00:09:53.5
S-IVB engine ignition	00:09:53.6
S-IVB engine cutoff	00:12:29.8
Earth orbit insertion	00:12:39.8
Translunar injection maneuver (S-IVB - 350.8 seconds)	02:35:46.4
S-IVB/CSM separation	03:06:38.9
Docking	03:19:08.8
CSM/LM ejection	04:01:00.8
Evasive maneuver (S-IVB APS - 80.2 seconds)	04:18:00.6

Event	GET Hr:Min:Sec
S-IVB maneuver for lunar impact (S-IVB APS - 217 seconds)	06:00:00
First TV transmission began	30:13:00
First midcourse correction (SPS - 3.5 seconds)	30:52:43.8
First TV transmission ended	31:02:00
Second TV transmission began	55:15:00
Second TV transmission ended	55:46:00
02 tank anomaly	55:54:53.2
LM powered up	57:41:00
CSM powered down	58:39:00
Second midcourse correction (IM-DPS - 34.1 seconds)	61:29:43.5
S-IVB lunar impact	77:56:39.7
Transearth injection (IM-DPS - 263.8 seconds)	79:27:39
CM PCM transmission (3 minutes)	101:59:09
Third midcourse correction (LM-DPS - 14 seconds)	105:18:28
CM PCM transmission (2 minutes)	123:09:56
Fourth midcourse correction (LM-RCS - 21.5 seconds)	137:39:51.6
Command module/service module separation	138:01:39
IMU heaters on	138:24:00
CSM powered up	140:10:00
LM/CM undocking	141:30:00.2
Entry interface (400,000 feet altitude)	142:40:45.9
Begin blackout	142:41:04
End blackout	142:44:23
Drogue deployment	142:48:54.3
CM-RCS purge terminated	142:51.40
Landing	142:54:41

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Event	Time (GMT) (April 17, 1970)
First visual sighting	18:02
Visual sighting from USS Iwo Jima	18:03
First voice contact	18:03
Splashdown	18:07
Flotation collar inflated	18:24
Command module hatch open	18:32
Crew aboard helicopter	18:42
Crew aboard USS Iwo Jima	18:53
Command module aboard USS Iwo Jima	19:36

Recovery Operations

Postrecovery Operations

Event	Date
Command module offloaded at Pearl Harbor, Hawaii	April 24, 1970
Command module arrived at Downey	April 27, 1970

Lunar module impact coordinates:			176.0° 25.5°			West Longitude South Latitude	
S-IVB lunar impact coordinates:			27.9° 2.4°			West Longitude South Latitude	
Splashdown coordinates:				165.36° 21.64°			West Longitude South Latitude
Landing att	itude: 🍐						Stable I
Sea state:	Waves: Swell:	1 ft: 6 ft:	2 5	seconds:	270° 130°	True True	

Wind:

6 knots: 270° True

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4.2 TRAJECTORY

S-IC Flight

The maximum dynamic pressure of 651.9 psf (3.5 percent less than predicted) occurred at 00:01:21.3. Maximum aerodynamic loading occurred at 00:01:14.5 with a = 6.396 degrees and a qa product of 3790 degreelbs/ft². This was well below the specification limit value of 6614. The maximum aerodynamic boost heating rate indicator (2qVas) occurred at 00:01:35 with a value of 2.42 x 10⁶ lbs/ft-sec (2.5 percent less than predicted). S-IC center-engine cutoff occurred at 00:02:15.2, with a load factor of 3.69g as predicted. S-IC outboard-engine cutoff occurred at 00:02:43.7 with a load factor of 3.93g (0.7 percent more than predicted). S-IC stage burn time was 0.3 second longer than predicted (liftoff to cutoff). The total aerodynamic boost heating load ($\int 2qVas$ dt) at S-IC/S-II separation was 140.2 x 10⁶ lbs/ft (3.8 percent less than predicted). Figure 4.2-1 presents aerodynamic boost heating parameters.

S-II Flight

S-II engine ignition occurred at 00:02:45. The launch escape system was jettisoned at 00:03:21. The S-II center engine was cut off prematurely by excessive vibration at 00:05:30.6, 132 seconds earlier than planned. The four outboard engines compensated for the early cutoff by burning 34 seconds longer than planned. Mixture-ratio shift occurred at 00:08:57. The outboard engines cut off at 00:09:52.6.

S-IVB Flight

The earth-orbit insertion burn lasted nine seconds longer than predicted in final compensation for the early cutoff of the S-II center engine. At the end of the burn, a greater than 3-signa probability of meeting translunar injection cutoff conditions existed with the remaining S-IVB propellants. A parking orbit of 100.2 by 98 n.m. was achieved.

The translunar injection burn was nominal and lasted five seconds longer than predicted. The pericynthion of the resulting free-return circumlunar trajectory was 415.8 mautical miles.

Overall Performance of the Launch Vehicle

The performance of each stage was calculated by normalizing cutoff velocity to the predicted altitude through constant energy. This method showed that the launch vehicle performance was outside the 3-sigma envelopes at the time of S-II/S-IVB separation (Figures 4.2-2 and 4.2-3) as a result of the early cutoff of the center engine of the S-II stage. The extended duration of the first S-IVB burn removed the deficit while placing the spacecraft in the proper orbit.

SPS Performance

The SPS was used once only. A 3.5-second burn at 30:40:49.6 was completed satisfactorily and placed the spacecraft on a non-free-return circumlunar trajectory with a predicted closest approach to the moon of 62 nautical miles and a closest approach to earth of 2300 nautical miles. The 0_2 tank failure at 55:54:53 resulted in loss of fuel-cell power and it was therefore not possible to use the SPS again. All remaining major maneuvers were performed with the aid of the lunar module propulsion systems.

Midcourse Corrections

The spacecraft was restored to a free-return circumlunar trajectory by a 34.1-second burn of the lunar module descent propulsion system (IM-DPS) at 61:29:43.5. The altitude of closest approach to the lunar surface was raised to 137 miles.

A second burn of the LM-DPS at 79:27:39 reduced the transearth transit time from 73 hours to 64 hours and moved the splash point from the Indian Ocean to the South Pacific Ocean.

A 14.0-second IM-DPS midcourse correction was made at 105:18:28 and a final 21.5-second correction was made with the LM-RCS at 137:39:51.6.

Spacecraft Separation Maneuvers

The service module was separated from the command and lunar modules 4 hours and 39 minutes before entry interface (EI). Separation translation was provided by the LM-RCS.

The lunar module was separated from the command module 1 hour and 11 minutes before EI. Separation translation was provided by tunnel pressure, which was vented down to a delta-pressure reading of 3.5 psia before separation to provide a hatch integrity check.

Entry Performance

The performance of the G&N system and the entry monitor system (EMS) during entry was excellent. A copy of the EMS scroll is presented in Figure 4.2-4.

The maximum load factor read from the scroll was 5.6g. No PCM entry data are available for comparison because the DSE was left deenergized to conserve battery power. The 0.05g light did not illuminate and the crew started the EMS manually. The EMS must have been started very close to 0.05g because the expected jump from a zero load factor to the load factor at the time of manual start is not visible on the G-V trace. No G&N incompatibility (i.e., no tangency) is apparent in the up phase. The trace predicted by the operational trajectory is superimposed on a copy of the actual scroll trace in Figure 4.2-4. The close agreement between the traces shows that the entry trajectory was as planned.

No evaluation of the range-to-go meter readings may be made because no crew callouts were received during entry.

EMS 0.05g Anomaly

The EMS is designed to start automatically when 0.05g is sensed by the accelerometer in the unit. When this occurs, the 0.05g lamp should illuminate, the scroll should begin to drive, and the range-to-go counter should begin to count down. The crew reported the failure of the light but did not check scroll or counter response before initiating manual backup.

The EMS was removed from the spacecraft and a complete functional test was performed. The flight anomaly could not be duplicated. The EMS was cold-soaked for 7 hours at 30F and was allowed to slowly warm while continuous functional tests were performed to ascertain if the anomaly could have been caused by thermal gradients. The EMS operated properly throughout all tests.

After the light and sensing circuits were functionally verified, the mode switch was examined in detail. Tests were performed to determine contact resistance, and the switch was examined for conductive contaminants (by X-ray) and nonconductive contaminants (by dissection). No evidence of switch problem was indicated.

Magnification of the scroll showed that the scroll operated normally at the sensing of 0.05g. A vertical line would have been traced if the scroll had not moved until three seconds after 0.05g had been sensed by the guidance and navigation system. This suggests that the mode switch was changed from automatic to manual without waiting for three seconds after the guidance system sensed 0.05g.

It is unlikely that the guidance and navigation system sensed 0.05g early. The spacecraft would have missed its targeted landing point by nearly 100 miles if 0.05g had been sensed as little as 4 seconds early. The entry trajectory shows that this did not happen.

Because of the conditions to which the crew were subjected during the mission, and because of the many last minute changes to the entry procedures, it is possible that the mode switch was actuated prematurely.



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Figure 4.2-4 Apollo 13 EMS Entry Scroll

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4.3 STRUCTURES

Structural load and vibration data for the Apollo 13 mission were limited to those provided by three linear accelerometers in the command module (CK0026A, CK0027A, and CK0028A, in the X, Y, and Z axes, respectively). These data, which were commutated and permit only a coarse analysis of loading and frequency response, indicate that the structural loads imposed on Apollo 13 were nominal throughout the mission. The spacecraft easily withstood the loads imposed during the two most critical phases - launch and entry.

Launch Phase

The maximum Y-axis acceleration was \pm 0.5g, experienced 2 seconds after liftoff. The maximum Z-axis acceleration of \pm 0.3g occurred near maximum q. Steady-state X-axis acceleration was 3.69g at S-IC centerengine cutoff (CECO) and 3.93g at S-IC outboard-engine cutoff (OECO).

The following table lists the values of the significant parameters at critical moments of the launch phase:

Event	Elapsed Time (sec)	X-axis Accel (g)	q (psf)	a (deg)	qa
Liftoff Max q <i>a</i> Max q S-IC CECO S-IC OECO	0.0 74.0 81.3 135.2 163.6	1.17 1.90 2.24 3.69 3.93	0.1 588.5 651.9 68.3 7.3	6.5 	3802

The 6.5-degree angle-of-attack at maximum qa was unusually high, but was well below the Block II design limit of 10.3 degrees.

S-II steady-state X-acceleration was 1.12g at CECO, 1.70g at mixtureratio shift, and 1.51g at OECO.

S-IVB steady-state acceleration at engine cutoff was 0.69g.

The predominant frequencies and their amplitudes during the launch phase were:

Front	X-Axis		Y-A:	xis	Z-Axis	
Evenc	g	Hz	g	Hz	g	Hz
Liftoff S-IC CECO S-IC OECO S-II CECO S-II OECO	0.45 0.48 0.93 0.12 0.16	4.0 5.2 5.0 7.0 9.3	0.49 0.07 0.13 0.10 0.16	Complex 5.5 5.0 14.0 13.0	0.17 0.05 0.13 0.08 0.12	Complex 4.5 5.0 14.0 Complex

Translunar Injection

The steady-state X-axis acceleration at engine cutoff was 1.5g.

SPS Hybrid Transfer Burn

The maximum X-axis acceleration during the single SPS burn was 0.35g.

02 Tank Anomaly Period

A sudden pressure increase in bay 4 of the service module tore the skin panel of that bay from its fastenings at approximately 55:54:53 and ejected it from the spacecraft. The pressure increase resulted from sudden failure of the cryogenic oxygen storage system. Telemetry data were interrupted simultaneously with the incident and no valid accelerometer readings were acquired. Except for the missing panel, the service module structure appears to have remained intact.

4.4 RECOVERY, DOCKING, AND ORDNANCE SYSTEMS

Recovery System

The recovery system, the recovery aids, and the uprighting system performed satisfactorily. The sea-dye-marker/swimmer-umbilical was intentionally not deployed.

Postflight inspection of the top deck at Downey showed that parachute strap R8083-3, adjacent to gusset No. 3, had been broken off. Similar damage has been observed after parachute drop tests and is considered random and not significant. No further action is recommended.

The docking-ring charge-holder was found retained in its groove by the retention springs but a photograph taken aboard the carrier showed that the charge-holder was not captured by the springs during descent. It is concluded that the charge-holder was replaced in its groove after the photograph was taken. A similar failure to retain the charge-holder was experineced during the Apollo 10 mission (CSM 106). Subsequent investigation led to the conclusion that the anomaly was caused by the presence of a positive pressure in the tunnel at the time of separation. A positive pressure of approximately 1.5 psi was deliberately permitted to exist in the Apollo 13 tunnel to provide separation translation in the absence of SM-RCS capability. The existence of this pressure accounts for the reoccurrence of the anomaly. Apollo 15 and subsequent spacecraft will have their charge holders bolted in position. No change is contemplated for Apollo 14. The Apollo 14 tunnel will be vented to zero pressure if IM separation is completed in the normal manner of Apollo 11 and 12 (and as originally planned for Apollo 13).

No other damage or anomaly was observed during the inspection. The forward heatshield lanyard was intact with all fourteen knots tied.

One main parachute and the forward heatshield were recovered. Both were examined and found to be in generally good condition. Minor damage was noticeable on both the main parachute and the forward-heatshield parachute and was attributed to recovery and handling damage from grappling hooks, etc.

Satisfactory operation of the recovery system was evidenced by television coverage of the final descent, the normal descent time from drogue mortar fire to splashdown of 346.7 seconds, and the absence of significant top deck damage. The data storage equipment (DSE) was not powered during entry and no detailed data are available for analysis.

Recovery Aids

Apollo 13 recovery aids comprised a flashing light, two VHF antennas, a sea dye marker, a swimmer umbilical and a grappling hook. The flashing light and both VHF antennas were erected but the outboard ground-plane whisker on the plus-Z side of gusset No. 1 did not deploy (see section 4.9). The sea dye marker, the swimmer umbilical, and the grappling hook were not deployed.

Uprighting System

The command module landed and remained in the Stable I attitude. The uprighting bags were inflated as a precautionary measure against inversion to the Stable II attitude.

Docking System

Docking system performance during docking with the LM/S-IVB was within normal design limits and satisfactory. This includes probepiston extension and retraction operations (including time spans) and docking latch actuations.

Probe and drogue removal was performed without difficulty. The probe and drogue were not reinstalled and remained stowed in the command module until they were transferred to the lunar module during preparation for lunar module jettison.

Ordnance System

The ordnance system satisfactorily performed all of its required functions.

The V36-596130 breech-plenum assembly of the forward heatshield thruster subsystem was removed during postrecovery operations for refurbishment and reuse. It was then noted that hot gas had escaped at the breech to plenum interface, eroding the assembly and burning a hole through the gusset side plate. The adjacent fiber glass housing was locally burned but not penetrated.

The assembly was disassembled, cleaned, and reassembled, and subjected to pneumatic pressure tests. A similar assembly from the Apollo 12 command module was subjected to hydraulic pressure tests. Both series of tests were satisfactory and no leakage occurred. The assembly procedures for the assemblies to be used in SC-110 and subsequent vehicles have been clarified and tightened to avoid reoccurrence. As a further precautionary measure, fiberglass blast shields will be added to the gussets, docking tunnel, and thruster cylinders.

4.5 MECHANICAL SUBSYSTEMS AND CREW EQUIPMENT

The mechanical and crew equipment operated satisfactorily.

Side Crew Hatch

Postflight examination showed that the shear pin on the gearbox had been sheared. There were indications of salt deposits on one lower latch roller near the hinge. The two frame-mounted jackscrew mounting rollers were rusted. The hatch checkout was conducted satisfactorily.

It was noted during removal of the hatch from the command module that the hinge pins were galled and difficult to remove. A test was conducted to measure the force required to push the hatch open. The test produced measurements ranging from 59 pounds to 74 pounds. The specification limit is 100 pounds.

Forward Hatch

The forward hatch mechanism was found to be in excellent condition when examined after the command module was returned to Downey.

Shock Attenuation Struts

Postflight inspection showed that the shock attenuation struts had not stroked and that the four lockout devices had been released.

Postlanding Ventilation Valve

The postlanding ventilation valve locking handle was found to be only partly extended after recovery. The inlet valve was closed and the lock pin was partly engaged. It is apparent that the locking handle was not fully extended following splashdown. The mechanism was found to be correctly rigged and within specification limits. The valve and its locking mechanism operated correctly during postflight tests. This anomaly is discussed further in Section 4.8.

Windows

At 03:57:14 the lunar module pilot reported "the windows came through in real good shape. Window 5 looks real clean". The side hatch auxiliary dump nozzle was used once for a urine dump following powerdown. The dump completely clouded the hatch window and rendered it unusable for photography. The CM-RCS purge deposited a brown film on the side and rendezvous windows during parachute descent.

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4.6 ELECTRICAL POWER SYSTEM

4.6.1 ELECTRICAL POWER DISTRIBUTION SYSTEM

The three fuel cells adequately provided electrical power for the CSM and the lunar module from liftoff to 55:57. Entry batteries A and B supplied supplementary power during the launch phase and the SPS burn. The fuel-cell to main-bus connections were different from those for previous Apollo missions. It had been the practice to connect fuel cells Nos. 1 and 3 to main buses A and B, respectively, and fuel cell No. 2 to both the A and B buses. This arrangement placed all three fuel cells in parallel on both buses. The effects of the lightning strike on Apollo 12 caused reconsideration of the bus-tie arrangement, and it was decided to not tie the main buses together for subsequent missions. The bus connections for Apollo 13 were fuel cells Nos. 1 and 2 to main bus A and fuel cell No. 3 to main bus B.

The dc main bus voltages ranged from 27.59 vdc to 29.72 vdc between liftoff and 55:54:53. Coincident with loss of 02 tank pressure, data were lost for just over one second. The caution and warning system main bus B undervoltage alarm was illuminated during this period, indicating that main bus B had dropped below 26.25 vdc. The bus voltages prior to data loss were 28.82 vdc on A and 28.65 vdc on B. Following data recovery, the voltages were 29.00 vdc on A and 28.83 on B. Main bus B voltage started to decrease rapidly at 55:57:27.9 and read essentially zero 12 seconds later. The voltage on ac bus No. 2 started to drop when dc main bus B reached 20.46 vdc and reached zero volts 1.4 seconds later, with main bus B at 17.61 vdc. Main bus A voltage started to decay about 20 seconds after main bus B, with an undervoltage alarm occurring when the voltage dropped below 26.25 vdc at 55:57:07. The dc bus voltage drops were caused by the loss of fuel cells Nos. 1 and 3 when their oxygen supplies were cut off by the O2 tank anomaly. Main bus A voltage stabilized at around 25.5 vdc, but was increased to around 28 vdc when battery A was placed on the bus at 56:03:07.6 and the CSM was partly powered down. The bus voltage was 28.82 vdc when battery A was removed from the bus at 56:36:07. The total current load at that time was 44 amps, of which 13.4 amps were being supplied by the battery. Following battery removal, the full load was taken by fuel cell No. 2 and the bus voltage dropped to around 27.2 vdc. Main bus A voltage ranged from 26 to 27.5 vdc until 58:04:01, when battery A was reconnected to the bus and the voltage jumped to 29.36 vdc. Fuel cell No. 2 ceased to provide power at approximately 58:15:00 for lack of oxygen and a main bus A undervoltage alarm was reported by the crew at 58:15:11. The CSM was completely powered down at approximately 58:39:15 to conserve battery energy.

Battery B was used for approximately three minutes at 102 hours to provide instrument readings for the crew and telemetry transmission. The lunar module batteries provided approximately 120 AH of electrical power to CM main bus B between 112:06 and 140:10 to recharge batteries A and B for entry. Battery C was connected to main bus A at 138:00 for pre-entry checks. Battery B was connected to main bus B at 140:10, following removal of LM power. Batteries A and C were connected to main bus A at 140:10 also. The three entry batteries provided all necessary power for the command module through entry, splashdown, and recovery.

The electrical power distribution system performed well throughout the mission. The performance of the entry batteries during pre-entry and entry was excellent, despite a low command module ambient temperature of approximately 43F at startup. The last recorded battery-voltage reading was 29.18 volts at approximately 14 minutes before splashdown.

4.6.2 FUEL CELLS

Configuration

Fuel cell Nos. 1 and 2 were connected to main bus A. Fuel cell No. 3 was connected to main bus B.

Prelaunch Operation

The fuel cells were started on April 9, 1970. All phases of the startup were normal. Fuel cell No. 1 was placed on the main buses at T-32 hours with a 22.5 ampere load. Fuel cells No. 2 and 3 were placed on the buses at T-3.9 hours. The total load on the three fuel cells was 75 amperes.

Mission Operation

Fuel cell operation was excellent up to 55:54:53, when the incident shock of the O₂ tank anomaly closed the oxygen shutoff values of fuel cells Nos. 1 and 3. These two fuel cells continued to provide normal output for approximately 2.5 minutes while they consumed the oxygen in the lines. They ceased to provide power when the oxygen was exhausted. Fuel cell No. 2 continued to operate from the depleting supply of oxygen in tank No. 1 until the residual pressure in the tank decayed below the required inlet pressure of the oxygen regulator. The internal oxygen pressure then dropped to the nitrogen pressure level and the fuel cell could no longer sustain the load. The fuel cell timeline from O₂ tank incident to final shutdown was as follows:

Mission Elapsed Time	Event
55:54:53	Abrupt and complete loss of pressure in O2 tank No. 2, accompanied by main bus B undervoltage indication.
55:58:08	Main bus A undervoltage indication. Main bus B yoltage dropping to zero.
56:00:53	Attempts to reset fuel cell Nos. 1 and 3 on main buses failed. Flowrates for these two fuel cells were zero.
56:06:24	Fuel cell No. 1 N ₂ pressure indicated zero. Fuel cell No. 2 O ₂ flowrate was 0.6 lb/hr.
56:09:58	Fuel cell No. 1 open circuited.
56:24:00	All fuel cell pumps off (during partial power- down operations).
56:25:55	Fuel cell No. 2 pump connected to ac bus No. 1.
56:34:46	Fuel cell No. 3 open circuited.
56:58:40	Inline heater on fuel cell No. 1 shutdown.
57:01:54	Fuel cell No. 3 shutdown.
57:18:20	Fuel cell No. 1 shutdown.
57:44:25	02 tank No. 1 pressure down to 100 psia.
57:56:10	Fuel cell No. 2 pump off.
58:15:00	Fuel cell No. 2 output ceased accompanied by main bus A undervoltage alarm.
58:38:42	Fuel cell No. 2 shutdown.

The zero N_2 pressure indication for fuel cell No. 1 is considered to be the result of an instrumentation failure. The other fuel cell parameters indicate that N_2 pressure remained normal.

The fuel cell currents prior to the incident were 22.1, 22.5, and 25.7 amperes for Nos. 1, 2, and 3, respectively. The total current was 70.3 amperes. The output of fuel cell No. 2 increased to over 50 amperes as the output of the other two fuel cells decayed to zero. The peak output of fuel cell No. 2 was 66.3 amperes at 23.84 vdc at 56:00:00. Battery A provided support from 56:03 to 56:36 and assumed approximately 34 percent of the load. Fuel cell output for this 33-minute period ranged from 29 to 37 amperes. Battery A output current ranged from 11 to 20 amperes.

When battery A was removed from the bus, fuel cell No. 2 output increased to 45 amperes and then ranged from 37.5 amperes at 27.75 vdc to 55.7 amperes at 25.8 vdc. Battery A was reconnected to the bus at 58:04 and assumed approximately 33 percent of the load. From then until the fuel cell ceased to provide power at approximately 58:15:00 its output ranged from 22.5 to 26.0 amperes. The bus voltage ranged from 29.00 to 29.71 vdc during this period.

The condenser exit temperature of fuel cell No. 3 varied periodically during the 56 hours prior to shutdown. Similar fluctuations have been observed during previous missions and analysis on tests conducted subsequent to the Apollo 10 mission showed that the oscillations were not detrimental to the performance or life of the fuel cells.

4.7 CRYOGENIC STORAGE SYSTEM

The cryogenic storage system satisfactorily delivered reactants for the fuel cells and oxygen for ECS operations and lunar module pressurization up to 55:54:54, at which time the pressure level of O_2 tank No. 2 dropped abruptly to zero, and pressure in O_2 tank No. 1 began to decay. O_2 tank No. 1 continued to supply oxygen to fuel cell No. 2 for approximately 2 hours and 20 minutes, i.e., until the tank pressure fell below the minimum input level of the fuel-cell oxygen-pressure regulator.

The quantity sensor in O2 tank No. 2 failed at 46:40:09.

A detanking anomaly which is believed to have led to failure of the cryogenic oxygen system in flight occurred during the countdown demonstration test.

Countdown Demonstration Test

 O_2 tank No. 2 could not be detanked by the normal procedures. This was attributed to loose or misaligned plumbing components in the dogleg portion of the tank fill path. The condition of loose plumbing was judged to be safe for flight.

Following numerous unsuccessful attempts to remove the liquid oxygen with gaseous purges and higher expulsion pressures, the fluid was boiled off by use of the tank heaters and fans, assisted by pressure cycling. The heater on-time was about 8 hours. The two thermal switches in the tank were designed to open at 80F to de-energize the heaters and prevent excessive temperature buildup. It is concluded from postflight examination of test data that both switches failed closed and that the high temperatures which resulted caused severe damage to the insulation of the fan motor wires. Failure of the switches and possible damage to insulation was not suspected prior to post-incident analysis.

Mission Performance

The cryogenic oxygen system functioned normally until 46:40:09, at which time the O_2 tank No. 2 quantity gauging sensor (CF0033Q) abruptly jumped to full scale. This malfunction was considered at the time to be a random instrumentation failure.

No further problem was observed until 55:54:53, when O_2 tank No. 2 pressure dropped abruptly to zero and O_2 tank No. 1 pressure started to decay. O_2 tank No. 1 pressure remained above the minimum level necessary for fuel cell operation for a period of approximately 2 hours and 20 minutes. Extensive analysis and testing have led to the conclusion that a fire was started in O_2 tank No. 2 by electrical short circuits in the degraded fan motor wiring. It is further concluded that the resulting pressure increase caused failure of the tank vacuum dome.

The postflight analysis and test effort is documented extensively in the NASA Report of the Apollo 13 Review Board, the NASA Apollo 13 Cryogenic Oxygen Tank 2 Anomaly Report, and the NR Apollo 13 Engineering Summary Report, SD70-243-2.

4.8 ENVIRONMENTAL CONTROL SYSTEM

The environmental control system (ECS) performed satisfactorily from system activation to powerdown at 58:39:00 and from powerup at 140:10:00 to splashdown. Between 58:39:00 and 140:10:00 the CSM was completely powered down and the ECS was non-operational. The anomalies that developed were minor and did not interfere with any planned crew activity.

There were three changes in configuration from Apollo 12. The Hycon camera installation included a provision for utilization of the urine dump line to obtain a vacuum. A lunar dust filter was supplied for installation over the cabin heat exchanger exhaust to assist in collection of free lunar dust after crew transfer from the lunar module. An extra urine filter was provided in addition to the primary and backup units.

Suit Circuit

Prior to the O₂ tank incident, the crew positioned the three suit-flow control valves in the suit full flow position during normal unsuited operations. This valve placement resulted in a suit-circuit flowrate of 54 cfm, with a compressor pressure rise of 8.31 inches of water.

From 58:39 to command module entry preparation, the lunar module ECS provided the sole means of crew life support. Command module cabin ventilation was provided by placing a lunar module suit supply hose through the tunnel into the command module. During entry, the command module ECS provided all normal life support functions.

CO2 Removal and Contamination Control

Prior to the incident, command nodule CO2 partial pressure was maintained at less than two mmHg by periodic replacement of the absorber elements. Following the incident, the cluster (command module and lunar module) CO2 partial pressure was controlled by the lunar module ECS. Emergency limits were imposed to allow the CO2 partial pressure in the cluster to exceed the normal limit of 7.6 mmHg. The CO2 partial pressure rose to 14.9 mmHg at one point before the crew changed LiOH elements. When it became apparent that there were insufficient LiCH cartridges in the lunar module to support the mission, the command module cartridges were adapted for use with the lunar module ECS. With ground instructions, a system was constructed which attached a command module LiCH cartridge to each of the lunar module suit-intake hoses. The commander's hoses were placed in the tunnel area to provide fresh oxygen to the command module, while the lunar module pilot's hoses were positioned in the lunar module environmental control area. At a later time, a second cartridge was added in series with the one installed initially. In each case, the drop in carbon dioxide levels showed satisfactory operation of the improvised system.

Temperature Control

Suit circuit temperatures were normal up to the time of the incident, with the suit heat-exchanger outlet temperature at a nominal 50F.

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Pressure Control

During normal unsuited operations, cabin pressure and suitcircuit pressure are controlled by the cabin pressure regulator, with no system demand on the suit-circuit demand regulator. For the shirtsleeves mode, the suit and cabin pressure readings should be nearly equal.

During launch the suit pressure transducer reading (CF0012P) followed cabin pressure until S-IC/S-II separation. It then dropped from 6.7 psia to 5.8 psi in four seconds. The delta pressure between the two measurements decreased to only 0.2 psi by 01:30:00, when the cabin reached its nominal regulated pressure level of 5.0 psia. During the changes in command module cabin pressure associated with the initial lunar module pressurization from about 3 to 4 hours, the suit pressure measurement responded sluggishly and indicated as much as 1 psi low while moving in a narrow band. The measurement level subsequently decayed and remained in the 4.1 to 4.3 psia range until powerdown at 58:39:00 hours.

The suit pressure measurement indicated correctly during the brief instrumentation power-up periods at 102 and 123 hours. However, the suit pressure reading was approximately 0.3 psi lower than the cabin pressure reading just prior to entry and had increased to only 7.7 psia just prior to landing, when cabin pressure was 13.9 psia.

This anomaly is similar to the Apollo 12 suit-pressure transducer anomaly and is considered minor, since alternate measurements are available for the determination of suit-circuit pressure. However, the transducer will be dismantled with the view to determining the precise cause of the malfunction.

Humidity Control

Humidity control was normal prior to powerdown, with moisture being removed from the command module atmosphere via condensation in the suit heat exchanger. Following powerdown, the only method of water removal from the cluster atmosphere was by means of the lunar module ECS water separator and condensation on the cold command module and tunnel walls. At 125:50:28 the crew reported that the command module windows were heavily coated with water.

Coolant Loop

The water-glycol coolant loop functioned satisfactorily when it was in operation. The primary coolant pumps operated as required until powerdown at 58:39:00. The primary coolant system was successfully re-activated during the entry period.
Expansion of the coolant in the primary loop increased the quantity of water-glycol in the accumulator from 34.5 percent to 50 percent. The accumulator adjustment scheduled for 30 minutes after launch was not performed. Accumulator data subsequent to the incident indicate an integral coolant loop. The accumulator quantity would have dropped sharply if the service module portion of the coolant loop had been damaged. The coolant system flowrate was approximately 10 to 20 lbs/hour lower than for any previous mission, but was well within the specification requirement of 190 lbs/hour minimum. Radiator inlet temperatures were approximately 3F higher than for Apollo 12, as would be expected with a lower system flowrate.

Coolant loop temperatures had dropped to 40F at pre-entry power up. Cabin temperature at that time was 43F.

Evaporators

All data received indicate that the primary evaporator operated satisfactorily. During the launch phase, boiling commenced at 00:01:36. Automatic operation commenced at 00:03:30. Automatic activation occurred during translunar coast at 7:57:00 when the evaporator inlet temperature exceeded 49.5F. The duration of the boiling period was three minutes. Sun look-angle data show that radiator panel 1 was oriented directly towards the sun prior to 7:43:48. The vehicle was then rolled at the rate of 17 degrees/minute, placing panel 2 normal to the sun's rays in 11 minutes. The proportioning valve redirected flow to the panels, causing the radiator outlet temperature to increase. Similar occurrences were noted during 2TV-1 thermal-vacuum tests after a rapid change of radiator environment.

The system was powered down at approximately 58:39:00. Temperature data on the evaporator measurements and radiator return temperature during the powered-down period show that the average rate of cooldown of the coolant loop was 0.342 F/hour. The average cooldown rate of the cabin atmosphere was 0.301 F/hour.

The primary evaporator was reactivated for entry at approximately 142:00:00. The outlet temperature dipped to 35.5F prior to stabilizing at 40.5F. Boiling continued until 142:47:00, when the ambient pressure increased to a level which stopped boiling. The data indicate that approximately 1.7 pounds of water were consumed.

ECS Radiator Performance

Radiator performance was satisfactory until the system was powered down at approximately 58:39:00. The radiators were not re-activated.

Comparison of primary and secondary accumulator readings prior to and subsequent to the O₂ tank incident show no loss of waterglycol, indicating no damage to the hydraulic radiator network. Two radiator system measurements, the primary radiator inlet temperature (SF0260T), and the secondary radiator outlet temperature (SF0263T), showed unusual temperature perturbations prior to, during, and after the incident.

The primary radiator inlet temperature pegged low at 55:57:53. This measurement was powered from main bus B, which dropped to zero volts at that time.

The secondary radiator outlet temperature increased from 40F to 53F between 55:18:00 and 56:33:00. Analysis of the sun look-angle data prior to and during the temperature rise reveals that the sun look angle was less than 38° between 55:00:00 and 55:12:00 and less than eight degrees between 55:12:00 and 55:48:00. Figure 4.8-1 shows the relationship between temperature response and the cosine of the true sun-angle. It is noted that the peak temperature occurred 36 minutes after the sun left the temperature transducer. The spacecraft was rotating to move the transducer out of the sun and radiator panel 2 into the sun. Because of its location, the temperature sensor was influenced by both the external environment and primary panel 2 outlet temperature. The time lag between the temperature drop and the cosine of the sun angle was due to both the heat-sink effect of the surrounding structure and the influence of the primary radiator outlet fluid temperature.

Based on the available data, it is concluded that the rise in the secondary outlet temperature readings was caused by the environmental conditions existent at the time and was in no way related to the incident.

Examination of the sun angle data at 7:57:00 further verified that the measurement is a function of the cosine of the sun angle.

Cabin Pressure and Temperature Control.

The launch data show that the cabin pressure relief valve cracked open approximately 56 seconds after liftoff. Cabin pressure at the time of reseat was 6.08 psia.

The cabin pressure was a nominal 5.0 psia following cabin purge with the cabin pressure regulator automatically supplying the oxygen make-up for cabin leakage and crew metabolic consumption.

Cabin temperatures ranged from 61F to 66F prior to powerdown at 58:39:00. The cabin temperature from powerdown to entry is plotted, along with selected component temperatures, in Figure 4.8-2. Following powerdown, the crew described the command module cabin as a cold, high-humidity environment. There was considerable condensation on the windows and the cabin wall.

Tunnel Vent Valve

There were no problems with the tunnel vent valve. The crew lowered tunnel pressure to 1.5 psia (3.5 psi CM/IM delta-P) prior to lunar module jettison.

Oxygen Supply

Surge-tank pressures were normal up to the time of the 02 tank incident. The pressures ranged between 853 and 900 psia, excluding lunar module pressurization. The surge tank was isolated at 56:30:00 to conserve oxygen for entry.

The drop in surge-tank pressure during isolation is of particular interest, since the drop could be indicative of leakage if no withdrawals were made. Tank pressure dropped at the rate of 55 psi per day for the first 44 hours of isolation, and 15.6 psi per day for the next 38 hours. The difference in the rates is attributed to (1) the higher tank pressure during the first interval and (2) oxygen withdrawal during water tank pressurization. A plot of surge-tank pressure during the isolation period is presented in Figure 4.8-3. The pressure decay was converted to leakage and/or consumption for the last 38.5-hour time period, and was found to be 2.88 x 10^{-3} lbs/hour. The checkout at Downey showed a leakage of 0.464 sccm nitrogen at 902 psia and 74F, which, converted to the flight conditions, yields an oxygen leakage of 6.54 x 10^{-6} lbs/hour. The difference between flight and checkout data probably represents water withdrawal attempts and amounts to 0.11 lb. during the 38.5-hour time span.

Regulated system pressure ranged from 102 to 105 psia before surge tank isolation. Regulation was also normal after the surge tank was reopened for entry.

The system oxygen flowrate was approximately 0.2 lb/hour up to the time of the incident, indicating a good hatch seal.

Rapid Repressurization System

There was no crew report of any problem with lunar module pressurization.

Water Management

Except for anomalous behavior of the potable tank quantity measurement, the performance of the water management system prior to power down was as expected.

Similar erratic operation of the potable water quantity transducer was observed during the Apollo 8 and 12 missions. Postflight examination of the disassembled transducer from Apollo 8 revealed the presence of corrosion in the transducer housing and silver deposit across the ends of the potentiometer, both problems being attributed to moisture. It was assumed that a similar condition existed inside the Apollo 12 transducer. The Apollo 13 transducer will be disassembled for further investigation.

Chlorination was accomplished more successfully than during some of the earlier flights. It was commented that, on the morning after chlorination, there was no taste and it seemed almost like the water had not been chlorinated.

The crew observed that there was "gas in the command module potable water all the way through the mission". A gas separator cartridge was provided to remove gas from the potable water, but it was not used. From crew comments, it is clear that the presence of gas in the water makes it difficult to drink, particularly when using the juice bags. The problem was considered by the commander to be about the same as on Apollo 8.

After the incident, a decision was made to withdraw water from the potable water tank and store it in eight-ounce juice bags to be available if the remaining water in the tank froze. The exact times and the numbers of bags filled at any one time is not clear. It appears that 22 bags were filled on one occasion, and that a total of 35 bags were filled. The crew reported at 125:19:12 that they had tried to draw more water from the tank, but found none left. The conclusion that the tank was empty was not correct, because 24.3 pounds of water were drawn from the potable tank during postflight test operations.

It has not been determined why the crew were unable to draw more water from the tank. Thermal analysis indicates that the temperatures in the aft equipment bay were not low enough to freeze the tank or waterlines. This analysis is supported by the fact that no water could be drawn during postrecovery operations onboard the carrier, although the command module had been through entry and had been exposed for several hours to an ambient temperature of 79F. The potable water tank is attached to the aluminum pressure-shell structure and ice should have formed on the cabin walls if the temperature of the inner structure had been low enough to freeze the water in the tank or lines. Further investigation of this anomaly would require extensive dismantling and inspection of the potable water system and appears to be unwarranted.

The performance of the waste water system was completely satisfactory. Water dumps (three) were accomplished as desired. Nozzle temperature ran between 88F and 90F, except when dumping, when it dropped to about 60F.

Waste Management

The available data indicate that any difficulty experienced with the system was procedural and brought on by the incident. Two items are noteworthy, however.

Early in the flight, it was reported that urine dumping was slow. This can be explained by the fact that the command module was being vented through the waste stowage vent valve to accelerate purging of the atmosphere to 100 percent oxygen. This purging operation reduces the capacity of the dump nozzle to handle urine. The crew recognized the problem and reported that urine dumping improved when the waste stowage vent valve was closed.

The other item concerns the auxiliary dump nozzle located in the side hatch. The crew reported that use of this nozzle completely clouds the window. This is a backup system which is not to be used unless both the urine and the waste dump nozzle are unusable.

The urine dump nozzle operated satisfactorily prior to powerdown and was not used subsequently. The temperature of the nozzle at 102 and 123 hours was -1.46F and -2.28F, respectively.

Postlanding Ventilation

The crew reported that, following splashdown, they opened the postlanding vent valve (PLVV) and operated the fan in the low mode. They also reported that ventilation was adequate.

During postrecovery operations it was found that the exhaust valve was open but the inlet valve was closed. This configuration could not provide adequate ventilation. It is believed that the low cabin temperature created an impression of good ventilation and misled the crew.

The PLVV is locked in the closed position during flight to prevent accidental opening. The mechanical lock handle was found to be jammed halfway between the stowed and fully extended (open) position. The handle has a travel of 0.75 inch and must be extended a minimum of 0.5 inch to ensure disengagement of both locking pins.

If it is not fully extended and either of the locking pins is not completely withdrawn, subsequent actuation of the valve solenoids will cause the engaged pin and the handle to jam.

Postflight investigation showed that the inlet valve pin was partly engaged. The locking mechanism and the valves were tested and found to operate correctly and without difficulty when the locking handle was fully extended. A complete dimensional check of the rigging mechanism was made and the assembly was found to be within specification limits. It is concluded that the inlet valve failed to open because the crew did not fully extend the locking handle. The crew were suffering from extreme fatigue at splashdown.

CSM Temperature Response Subsequent to Shutdown

Following powerdown, command module temperatures dropped at the rate of approximately eight degrees F per day, and were still falling at powerup for entry. No sign of thermal equilibrium was evidenced after 83 hours of unpowered flight. The cabin temperature of the active lunar module was about 10 degrees above the command module cabin. Temperature plots for the command module environment during the passive mode are presented in Figure 4.8-2.

Service module temperature plots are not shown since all ECS service module measurements were pegged low during the passive mode, indicating that some of the temperatures were less than 30F.





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Figure 4.8-2 Apollo 13 Command Module Cabin Temperature Following ECS Powerdown

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Figure 4.8-3 Apollo 13 Oxygen Surge Tank Pressure Decay During Isolation

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4.9 TELECOMMUNICATIONS

4.9.1 COMMUNICATIONS EQUIPMENT

The communications system satisfactorily supported the mission. Both S-band and VHF communications were used until translunar injection, after which the VHF was turned off and the S-band equipment was used until spacecraft powerdown at 58:39:00. S-band and VHF voice, color television pictures, and real-time and playback telemetry were satisfactory. Uplink and downlink signal strengths corresponded with preflight predictions except during three short periods when high-gain antenna anomalies occurred. Communications system management, including antenna switching, was good.

From 101:53:00 to 102:02:00 and from 123:05:00 to 123:12:00, the communications system was powered up to the extent necessary to transmit high-bit-rate telemetry data using the omnidirectional antennas. The S-band system was activated prior to lunar module jettison. The VHF/AM and VHF recovery systems were activated at parachute deployment and operated satisfactorily. -

4.9.2 VHF-AM RECOVERY ANTENNAS

The outermost radial of VHF antenna No. 2 (+Z side of gusset No. 1) failed to deploy because of dimensional interference with the housing. Deployment tests are prescribed before top deck closeout on Apollo 14 and subsequent spacecraft.

4.9.3 HIGH-GAIN ANTENNA

Three high-gain antenna (HGA) anomalies were observed during the mission.

1. 31:09:23 to 31:10:06

Signal levels were low and variable. The most probable cause is operation in the area where skin reflections cause multipath conditions. A second possibility is an earlier occurrence of the anomaly noted in the next paragraph.

2. 55:00:00 to 55:14:00

Fourteen minutes expired before narrow beam lockup could be achieved in readiness for a television transmission. The most probable cause is a malfunction in the scan limit circuitry.

3. Loss of Signal at 55:54:54

The one-second loss of signal is attributed to momentary deflection of the antenna when it was struck by the skin panel of bay 4, which had been separated from the service module by the cryogenic oxygen system failure.

Low and Variable Signal Levels from 31:09:23 to 31:10:06

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At approximately 31:09:23, while the antenna was operating in narrow beam mode, both uplink and downlink signal strengths suddenly dropped from normal levels for narrow beam operation and varied as much as 30 db below nominal for approximately 43 seconds. Figures 4.9-1 and 4.9-2 show the signal levels recorded during this period. After recovery, the signal strengths remained constant at nominal levels.

The HGA had been in use for nearly 90 minutes prior to signal drop at 31:09:23. The switch from omni antenna A was made at approximately 29:43:00 in preparation for a television program and MCC2. The antenna pitch and yaw angles at acquisition were approximately minus 68 degrees and 180 degrees, respectively. Signal strength levels were steady at expected levels for the existing range of 122,000 nautical miles. The quality of the television transmission (which started at 030:13:00) was excellent; there was no observable change in signal strength levels during the SPS burn (3.6 seconds duration beginning at 30:40:49.6); and signal strengths remained steady at expected levels as the spacecraft maneuvered. Spacecraft attitude changes after the SPS burn shifted the antenna pitch and yaw angles into the interference zone (defined in Figure 4.9-3). The spacecraft had been in attitude hold for approximately five minutes prior to the signal level drop at 31:09:23 and in an attitude that required pitch and yaw angles which are in the interference zone (minus 29.5 degrees and 24 degrees, respectively). The drop in signal level appears to have closely followed the start of a maneuver to a new attitude. Signal strengths recovered to normal levels during this maneuver, with pitch and yaw angles of minus 34 degrees and 20 degrees, respectively, at the time of recovery. Following recovery at 31:10:06, the signal strengths remained steady at predicted narrow beam levels.

Multipath problems in the interference zone are caused by reflections from the CSM and LM and are known to be severe. Multipath problems are considered to be the most probable cause of the drop in signal strength at 31:09:23. The transient increases in downlink signal strength shortly before recovery (Figure 4.9-2) are typical of those expected if narrow beam acquisition is attempted in the interference zone. Bending of the wide beam results in switching from wide beam to narrow beam when the antenna is not pointed directly toward the target. as the large track error which results from the narrow beam being offaxis causes the antenna to switch back to wide beam. If the wide-beam bending is large enough, (approximately 6 degrees) the antenna will lock-up on a narrow-beam side lobe. The initial decrease in signal strengths (which appears to have been caused by a beam switch) and the subsequent variations in signal levels, are not typical of signatures obtained previously during ground tests or flight when operating in the interference zone. However, it appears quite possible that multipath conditions could produce a transient narrow-beam error signal large enough to cause switching from narrow beam to wide beam. Once the antenna switched out of narrow beam beacuse of multipath conditions, switching and tracking problems could be expected to result from widebeam bending and scalloping.

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A possible alternative explanation for the signal level changes during the first 30 seconds of the anomaly period is that the antenna reached the scan limit. However, the signal strength signatures and antenna operation do not completely support this alternative explanation. If the track mode switch was in the REACQ position, the antenna would have slewed to the manual preset position commanded by the pitch and yaw controls, which presurably were still in the initial acquisition position of minus 68 degrees of pitch and 180 degrees of yaw. This position is nearly 180 degrees from the position of the antenna at the onset of the anomaly (minus 29.5 degrees of pitch and 24 degrees of yaw). The signal strength recordings do not indicate that the antenna slewed that far. However, if the track mode switch was in the AUTO tracking position, the antenna would have stopped tracking when it hit the scan limit and would have switched to wide beam when the trackerror signal became sufficiently large (approximately 3 degrees). Having reached scan limit the antenna would not have resumed tracking unless it was moved away from the scan limit either by drift (caused by electrical noise, inertia, or vehicle motion) or by crew switching to the manual mode. Drift is a valid possibility and the alternative hypothesis for the auto tracking mode is plausible except for the fact that the prevailing antenna pitch and yaw angles (minus 29.5 degrees of pitch and 24 degrees of yaw) were well removed from the scan limit. However, an assumption of a shift in scan limit setting such as is hypothesized for the 55-hour anomaly would restore plausibility.

In summation, the most probable cause of the 31:09:23 anomaly is the multipath phenomenon. An alternative explanation in that the anomaly resulted from a shift in scan limit setting such as is hypothesized for the 55-hour anomaly.

Inability to Lock on Narrow Beam from 55:00:00 to 55:14:00

The S-band equipment was switched from omni antenna B to the HGA at approximately 54:59:50, in preparation for television transmission. Both uplink and downlink signal strength levels were below expected levels following the switch. The antenna pitch and yaw angles given to the crew several hours earlier were 23 degrees and 267 degrees, respectively. Corrected angles of 5 degrees and 237 degrees were given to the crew at 55:02:27, approximately 2.5 minutes after the HGA was selected. The command module pilot reported at 55:05:32 that "We can't get it down to narrow. We tried to switch to AUTO track or REACQ, and yaw drives around from 270 to zero. And pitch goes from about 6 degrees around to 90. We're setting it manual now at the angles that you gave us and I'll try and get you in medium and narrow beamwidths, picking it up manually here." The signal strength levels remained substantially below required levels for approximately 14 minutes. Narrow beam lockup was achieved at 55:13:45, following a change in spacecraft attitude, and the television transmission was successfully completed. Signal-strength levels were steady and normal after lockup was achieved.

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During the anomaly, the crew reported that they had tried both the primary and secondary servo electronics, all three track modes (manual, auto, and auto-reacquisition), and all three beamwidths (wide, medium, and narrow). Except for a few instances, the antenna configuration cannot be positively correlated with signal strength signatures. Analysis of pattern data and signal strength signatures indicates that the beam select switch was in medium beam when the switch from omni B to HGA was made, rather than the expected : wide beam. It cannot be determined if the track mode switch was in the REACQ or MANUAL position at that time. The analysis also indicates that the track mode switch was in the REACQ position approximately 20 seconds after the switch to HGA was made. Low signal strengths were observed during these first 20 seconds because the antenna was in medium beam and off boresight, as it had been preset to the 23 and 267 degrees recommended some hours before. At 55:00:10, the signal strength levels began to cycle up and down as

The periodic signal level variations and sudden drops and increases appear to be the result of the antenna contacting the scan limit as it attempts to acquire the target in wide beam in the REACQ track mode. The antenna operates as follows in the auto reacquisition (REACQ) mode:

The correct initial conditions are:

shown in Figure 4.9-4.

- 1. Track mode select switch in MANUAL.
- 2. Beamwidth select switch in MEDIUM.
- 3. Pitch and yaw controls set for a position in the clear-track zone, i.e., a position outside the scanlimit and scan-limit-approach zones defined in Figure 4.9-3, and approximately 35 degrees away from the true pointing angle to earth.
- 4. The antenna is stabilized in the preset position commanded by the pitch and yaw controls. From a signal-level standpoint, this position is represented by point A in Figure 4.9-5. The unified S-Band equipment (USBE) receiver is in phase lock and an earth-present signal is therefore being delivered to the HGA electronics package.

Figures 4.9-6 and 4.9-7 illustrate the logic states existing in the electronics package for these conditions. The track mode select switch is now placed in the REACQ position. The antenna immediately switches to wide beam and auto track. Figures 4.9-8 and 4.9-9 show the new logic states and, from a signal-level standpoint, this condition is represented by point B in Figure 4.9-5. The antenna moves in the auto mode of auto reacquisition and in wide beam (the track error is large) towards the line-of-sight to earth. Velocity limiting is in effect along the A-axis. In Figure 4.9-5, this is represented by the line between B and C. Assume that the antenna hits a scan limit before the HGA can attain a small track error condition (point C in Figure 4.9-5). Logic networks I and II (Figures 4.9-10 and 4.9-11) latch up when the scan limit is reached and the antenna, which is now at point C in Figure 4.9-5, begins slewing back toward the preset manual position. The logic states for this manual slewing mode are shown in Figures 4.9-10 and 4.9-11. The A-axis is not velocity limited and the antenna stays in wide beam because medium-beam operation is inhibited by the scan-limit warning signal. The antenna leaves the scan-limit warning zone at point D in Figure 4.9-5. If the earth-presence signal persists, logic network II unlatches immediately and removes the inhibition of medium-beam operation in the manual mode. The associated logic changes are shown as circled items in Figure 4.9-11. Signal strength drops immediately, because the antenna has switched to medium beam and is considerably off boresight. The antenna tracks along the portion of the medium-beam pattern indicated by D, E, F, G, and A on Figure 4.9-12, and produces the down-link signal strength signature identified by D, E, F, G, and A' in Figure 4.9-5. Figures 4.9-6 and 4.9-7 show the logic states at this time. When the antenna reaches the manual preset position, i.e., when the manual error decreases to approximately one degree A! on Figure 4.9-5, the logic manual-error signal goes to zero, logic network I unlatches, and the antenna goes back into wide beam and auto track. The logic states are shown in Figures 4.9-8 and 4.9-9. Point B' in Figure 4.9-5 represents the down-link signal strength at this time. The antenna will now traverse a path which produces downlink signal strength signature represented by points B', C', D', E', F', G', and A" in Figure 4.9-5, in the same manner and for the same reasons that it traversed the path B, C, D, E, F, G, and A'. Thus the conditions hypothesized are sufficient to set up a repetitive cycle of antenna motion and beam switching which could produce the signal level variations shown in Figures 4.9-4 and 4.9-5. The cycling will continue until some change is made. In the case of the Apollo 13 anomaly the cycle was apparently broken by the crew switching from REACQ to AUTO. The brief dip in downlink signal strength at approximately 55:00:45 (Figures 4.9-4 and 4.9-5) is attributed to a momentary pause in the MANUAL position when changing the track-mode switch from REACQ to AUTO. The steady signal levels between 055:00:48 and 055:00:58 (Figure 4.9-4) are attributed to the antenna reaching the scan limit (in the AUTO track mode and wide beam) before the track error decreased sufficiently to cause switching out of wide beam. In this case the servo drive signals would be inhibited and the antenna would remain against the scan limit stop. The slight increase in signal levels at approximately 055:00:58, followed by what appears to be a tracing out of the antenna pattern, can be attributed to crew switching from AUTO to MANUAL. In this case the antenna would go to medium beam (assuming the beamwidth switch was in the mediumbeam position) and start slewing to the preset position of 23 degrees in pitch and 267 degrees in yaw.

The preceding discussion shows that periodic variations in signal level can be induced if the target is in the shadow zone and the HGA REACQ mode is selected. However, Figure 4.9-3 shows that the bearing to the earth was actually well outside the normal shadow zone. Thus the scan-limit explanation of the periodic signal-level changes shown in Figures 4.9-4 and 4.9-5 implies that a fault in the scan limit circuitry resulted in a significant displacement, and probably distortion, of the scan-limit and scan-limit-warning boundaries. It can be deduced that the scan-limit displacement/ distortion was such that the HGA look-angle of pitch, 5 degrees, and yaw, 239 degrees, was approximately 5 degrees inside the shadow zone, and that the look-angle of pitch, 13 degrees, and yaw, 247 degrees, was outside the shadow zone, on the following basis:

- 1. During the periodic signal-strength variations from approximately 055:00:15 to 055:00:45, the maximum levels were within approximately 1 db of expected onaxis levels for wide beam.
- 2. When the antenna switched to medium beam at approximately 055:00:57, presumably as a result of the crew changing the track-mode switch from AUTO to MANUAL after the antenna reached a scan limit, the down-link signal strength was approximately 3 db below the expected on-axis level for medium beam.
- 3. Narrow beam lockup was achieved in the AUTO mode at approximately 055:13:45 when the HGA look-angles were approximately pitch, 13 degrees, and yaw, 247 degrees.

Possible malfunctions resulting in displacement of the scan limits such that the HGA pointing angles of Pitch, 5 degrees, and yaw,239 degrees, are inside the shadow zone include the following:

- 1. A-axis function generator malfunction.
- 2. C-axis induction potentiometer malfunction.
- 3. Electronics package malfunction. A malfunction in the positive 5-volt power supply and/or the scan-limit circuitry is the most probable, but a faulty excitation voltage for either the A-axis function generator or the C-axis induction potentiometer could affect the scan limit.

4. A short, or partial short in innerconnect wiring.

Although both the primary and the secondary electronics were used, the uncertainty as to when the secondary was used precludes elimination of an electronics package malfunction on the basis that it is improbable that similar malfunctions occurred in two independent circuits. The signal-strength recordings for the period from 55:02:30 to 55:05:00 are shown in Figure 4.9-13.

The crew reported at 55:05:32 that the antenna was in MANUAL and wide beam and that they were going to try medium beam and narrow beam. Figure 4.9-14 shows the signal strength records for the period from 55:06:00 to 55:08:00. The 14 db decreases in down-link signal strength occasioned by the switch from wide beam to medium beam, and the slight increase from the medium beam levels when narrow beam was used, indicate that the antenna was pointing 13 to 17 degrees off boresight. The off-boresight hypothesis is also supported by a 4 db increase in signal level, in approximately 7.5 seconds, as the spacecraft rolled just prior to acquisition (see Figure 4.9-15). It is believed that the antenna was in MANUAL and medium beam during this 7.5-second interval. The vehicle rollrate was 0.3 deg/sec.

Although MANUAL pointing of the antenna should normally result in less than the 13 to 17 degrees error apparently encountered just after 55:05:32, circuit tolerances and meter reading errors could introduce an error of that magnitude. The meters and circuitry were intended to give only gross pointing information since the antenna can readily acquire from a position at least 60 degrees off boresight. Medium and narrow beamwidths have been used while in MANUAL track mode but tweaking (i.e., adjusting the manual position controls to maximize received-signal level) is required. There are no positive indications that tweaking was attempted in medium beam or narrow beam during the anomaly. Thus it is concluded that the low signal levels in MANUAL and medium beamwidth were most probably caused by acceptable antenna pointing error.

Figure 4.9-15 shows the recorded signal-strength levels at narrow beam lockup, which occurred at 55:13:45 in AUTO track mode with primary electronics.

Investigation of the acquisition anomaly is continuing. However, on the basis of experience to date, and the analysis which indicates that the problem resulted from a scan-limit circuit malfunction, it is believed that the malfunction was caused by a quality-type defect in manufacture or assembly, and no design or hardware change is contemplated.

Loss of Communication at 55:54:54

At approximately 55:54:54, while using the HGA in narrow beam, communication was lost for slightly more than one second. Both uplink and downlink signal levels suddenly dropped below measurement threshold and then recovered to levels approximately equal to those expected for wide beam operation. After recovery, the signal strengths remained nearly constant for approximately 6.5 minutes and then started to gradually decrease as the vehicle manuevered into the omni antenna C region of coverage. Omni C was selected, upon ground request, at approximately 56:04:19. The HGA was not used again during the mission. Following service module jettison, the crew reported that the skin panel was missing from bay 4 of the service module and that one of the paraboloidal reflectors of the HGA was damaged. Photographs taken at the same time have confirmed the damage. It is believed that the antenna damage and loss of communication resulted from the panel striking the antenna. The low signal levels for the 6.5 minutes following temporary loss of signal were probably the result of the HGA locking up on the first side-lobe of the distorted (due to damage) narrow beam. The gradual decrease in signal levels as the vehicle maneuvered into the omni C region of coverage most likely occurred because the HGA stopped tracking as a result of reaching a scan limit stop. On the basis of the crew report of HGA configuration at 55:49:25 it is believed that the track mode during and after the 055:54:54 incident was AUTO.

In the AUTO mode, the antenna stops tracking when it reaches a scan limit and either remains in the position in which it was halted or drifts slightly due to noise.

4.9.4 INSTRUMENTATION SUBSYSTEM

The instrumentation subsystem performed satisfactorily throughout the mission. With the exception of two brief transmissions at 102 and 123 hours, no data were transmitted from 58:39:00 to 140:10:00 because the CSM subsystems, including the instrumentation subsystem, were powered down.

PUGS Point-Sensor Failure

The point-sensors in the SPS propellant utilization gauging system (PUGS) failed during the countdown demonstration test. Failure is attributed to leakage of fuel through the point-sensor lead feedthrough. The condition was accepted for flight. The malfunction is discussed in detail in Section 4.12.

SPS Oxidizer Engine Interface Pressure Sensor Drift (SPOCO3P)

The SPS oxidizer engine interface pressure readings drifted from three-psi low before liftoff to eight-psi high at 123:12:06. The drift is attributed to outward leakage of the reference pressure behind the sensing diaphragm of the transducer. The malfunction is discussed in Section 4.12.

Fuel Cell No. 1 Regulated No Pressure (SC2060P)

The output of the fuel cell No. 1 regulated nitrogen pressure sensor dropped abruptly to zero at 56:06:24. The other fuel cell No. 1 data indicate that nitrogen pressure continued to be maintained at the proper level and that the sensor indication was incorrect. The failure was attributed to the disturbance in bay 4 that was created by failure of the vacuum dome of the O₂ tank No. 2. The failure is discussed in Section 4.6.2.

Suit Pressure (CFO012P)

The suit pressure sensor readings dropped from 6.7 to 5.8 psia at approximately 00:02:45 and almost recovered by 01:30:00. The readings, which should have remained relatively constant until entry, subsequently decayed from 4.8 psia to 4.1 psia at 58:20:00. Sensor output had recovered to 4.8 psia at 142:25:00 but rose to only 7.7 psia during descent. The expected reading at that time was 13.9 psia.

Erratic performance of the suit pressure transducer was observed during the Apollo 12 mission. Postflight testing determined the cause to be internal contamination. The sensor from the Apollo 13 command module has been returned to the vendor for disassembly and failure analysis. The malfunction is also discussed in Section 4.8.

Potable Water Tank Quantity (CFO010Q)

The potable water tank quantity sensor behaved erratically from 22:41:00. Similar behaviour was observed during the Apollo 8 mission and postflight testing traced the malfunction to moisture contamination. The Apollo 13 sensor will be disassembled for further malfunction analysis. The malfunction is also discussed in Section 4.8.

Data Storage Equipment

The data storage equipment (DSE) operated satisfactorily whenever its use was required. To conserve battery power, the DSE was not energized during entry.



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Figure 4.9-3 High-Gain Antenna Scan and Warning Limit, Yaw-Pitch Coordinates

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Figure 4.9-4 Apollo 13 Signal Strengths - 55:00:00 to 55:02:00

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Figure 4.9-6 HGA Switching Logic States in Manual Mode Prior to Auto-Reacquisition Mode

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Figure 4.9-7 HGA Switching Logic States in Manual Mode Prior to Auto-Reacquisition Mode



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Figure 4.9-8 HGA Switching Logic States in Auto Mode of Auto-Reacquisition Prior to Small Track Error

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Figure 4.9-10 HGA Switching Logic States at Start of Manual Slewing in Auto-Reacq. Mode at Scan Limit Departure



Figure 4.9-11 HGA Switching Logic States at Start of Manual Slewing in Auto-Reacq at Scan Limit Departure



1. Path C-D-E-F-G-A results from manual slewing in auto-reacquisition mode.

2. Path A-B results from change from manual to auto mode of auto-reacquisition.

3. Path B-D-C results from motion of high-gain antenna in auto mode of auto-reacquisition mode.

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Figure 4.9-12 Apollo 13 HGA motion to produce gain excursions shown in Figure 4.9-4.

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Note: Numbers show signal strength in db below 1 milliwatt

Figure 4.9-13 - Apollo 13 Signal Strengths - 55:02:30 to 55:05:00



Note: Numbers show signal levels in db below 1 milliwatt

Figure 4.9-14 Apollo 13 Signal Strengths - 55:06:00 to 55:08:00



Downlink - Receiver 2



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Numbers show signal levels in db below 1 milliwatt

Figure 4.9-15 Apollo 13 Signal Strengths - 55:13:00 to 55:14:00

4.10 GUIDANCE AND CONTROL

The performance of the guidance and control (G&C) system was excellent. The complete G&C system, including the IMU heaters, was de-energized at 58:30:00. The IMU heaters were reactivated at 138:24:00 and the computer was turned on. The powerdown procedure left the IMU unheated and in an unusually-cold ambient environment (down to 43F) for approximately 80 hours. Despite the long coldsoak, the G&C system responded perfectly through entry and guided the command module to a safe landing 1 mile from the target point.

CSM/S-IVB Separation and Docking

Separation from the S-IVB was performed at 03:06:38.95 by thrusting for 4.28 seconds to impart a velocity change of 0.86 ft/sec. After a manual pitch maneuver, the command and service modules were docked with the lunar module. Rate disturbances noted at docking were 0.16 deg/sec peak in pitch and yaw, and 0.60 deg/sec peak in roll.

Passive Thermal Control

The passive thermal control modes established at 7:43:02 and 32:21:49 were not successful and had to be reinitiated. The attempt at 7:43:02 resulted in a divergent coning angle because the roll rate was established with only one roll engine instead of two engines to provide a couple as required by the checklist. In addition, an incorrect roll rate was loaded into the digital autopilot.

The attempt at 32:21:49 resulted in a divergent coning angle because an unplanned minimum impulse engine firing occurred 13 seconds after initiating the roll rate. The engine firing command (two negative roll engines) was generated when the roll manual attitude switch was changed from the rate-command position to the acceleration-command position. The engine firing could have been avoided by disabling all engines before doing any control system switching.

The passive thermal control maneuver attempted at 32:21:49 is compared with the subsequent, normal, maneuver in Figure 4.10-1.

Ortics Dust Cover Jettison

The command module pilot reported that he twice performed the optics dust cover jettison procedure as outlined in the crew check list but the cover did not jettison. The cover jettisoned later when he performed a P52 alignment.

The check list procedure calls for shaft angle settings of 80 degrees for the telescope and 30 to 40 degrees for the sextant. Investigation has shown that these angles are not adequate and can cause the cover to hangup. The Apollo Operations Handbook and Crew Check List requirements have now been changed to call for 150 degrees minimum angle for the telescope and 40 degrees minimum angle for the sextant.

Sextant Shaft Fluctuations in the Zero Optics Mode

Fluctuations of up to 0.3 degree were observed in the computer readout of the sextant shaft angle, beginning at approximately 40 hours. The optics system had been in the zero optics mode since the star horizon navigation sightings were made at 31 hours. Crew observation confirmed that the fluctuations were caused by actual shaft motion. The optics system behaved similarly during the Apollo 12 mission.

Extreme sea water corrosion prevented postflight testing of the mechanical drive system, but analysis of other tests led to the conclusion that the fluctuations were caused by slip-ring contact resistance which develops in the half-speed resolver under flight conditions. The slip rings are wiped clean when the optics are rotated and the resistance disappears, so no corrective action is necessary. The accuracy of the zeroing operation is not affected by this temporary resistance buildup. The anomaly is discussed in greater detail in Section 5.2.

Computer Restart

A computer restart was generated by the voltage transients which accompanied the O_2 tank No. 2 incident. The digital autopilot reinitialized the attitude to which it was referenced and reduced the rate and attitude errors to the nulled condition in 75 seconds.

Powerup and Realignment for Entry

The IMU heaters were energized at 138:24:00 and the computer was turned on. The command module platform was coarse aligned to the lunar module platform and was fine aligned by using automatic optics positioning to locate suitable stars. This technique was used because it was difficult to recognize constellations through the scanning telescope as a result of reflections from the lunar module and obscuration from vented particles.

It is estimated that the minimum temperature of the platform during the powered down period was 55F. The only significant coefficient shift observed after the 80-hour cold soak was in Z-accelerometer bias. The bias was compensated for at 141 hours by an update from minus 0.04 cm/sec² to minus 1.66 cm/sec². Although no gyro measurements were obtained immediately prior to entry, the precision of the landing indicated that no significant misalignment existed.

Entry Monitor System 0.05g Light

The crew reported that the entry monitor system (EMS) 0.05g light did not illuminate within three seconds after 0.05g was sensed by the guidance system. The EMS was then started manually by switching to standby. Postflight testing has failed to duplicate the reported malfunction and or disclose any fault in the EMS. Testing is continuing. The anomaly is discussed in more detail in Section 4.2.

Translation Hand Controller Cable Routing

The crew reported that the translation hand controller cable interfered with opening of the rear door of the pantry. Postflight inspection confirmed the interference. The cable did not have the service loop defined in V36-781510, and it was not contained within the O₂ umbilical strap.

Occurrence of 220 Alarm

A 220 alarm occurred during the course of a P52 alignment. The IMU was being coarse aligned using three gimbal angles provided from the ground. By aligning the IMU in this manner the REFSMMAT FLAG was not set and the 220 alarm indicated that the IMU orientation was unknown. This was corrected by setting the REFSMMAT FLAG manually and then continuing with P52, which was completed satisfactorily.


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4.11 REACTION CONTROL SYSTEM

SM-RCS

Prelaunch Operations

Fuel loading was accomplished on March 9, 1970, and oxidizer loading was accomplished on March 12, 1970. The loaded weights were as follows:

Quad	Oxidizer	Fuel (1bs at 70 F)								
The Difference	(105 at 07 F)	Primary	Secondary							
A	225.6	70.2	40.2							
В	225.5	69.7	39.8							
C	225.4	70.1	40.0							
D	226.2	70.3	39.8							

Helium servicing was accomplished on April 7, 1970. The quad pressures at the time of coupling closeout were as follows:

Quad	Helium Tank Pressure (psia)	Helium Manifold Pressure (psia)	P/T (percent)
A	4310 @ 81 F	195	98
В	4220 @ 74 F	197	100
C	4190 @ 72 F	195	100
D	4260 @ 73 F	195	100

Mission Performance

The SM-RCS was activated only from liftoff until the CSM was powered down at 58:39:16, following loss of fuel cell power (section 4.6). All quad helium, fuel, and oxidizer manifold pressures were normal up to the time that O₂ pressure was lost (55:54:53). The helium tank temperatures ranged from a low of 61F on Quad A to highs of 83F on Quads B and C prior to 55:54:53. This range is normal and is the result of satisfactory passive temperature control (PTC) rotation of the spacecraft. The lowest quad temperature recorded during the mission was 52 F on Quad B, observed when the CSM PCM system was powered up for a status check at 123:10:00.

SM-RCS Effects Attributed to the O2 Tank Anomaly

Heavy usage of the SM-RCS was observed at approximately 55:55:00, immediately after the oxygen tank incident. Figures 4.11-1 through 4.11-8 show the SM-RCS flight parameters for a 35-minute interval starting at 55:50:00. At 55:54:56, the helium oxidizer and fuel manifold pressures of Quads B and D dropped 6 to 14 psi. It can be assumed that the shock associated with the incident caused several of the helium and propellant isolation valves to close. Engine firing while the helium isolation valves were closed would cause the observed decay in manifold pressure. At 55:55:00, the oxidizer manifold pressure of Quad C rose 5 psi, possibly as the result of an ambient temperature increase. At 55:55:12, the manifold pressure of Quad B returned to normal.

Quad	Helium Is Valve	solation	Primary Propellant	Secondary Propellant
	No. 1	No. 2	Valve	Valve
A				Closed
В	Closed			
C				Closed
D	Closed	Closed		

At 55:57:44, the crew reported the isolation valve status to be:

The valve position indicators for Quads A and C are powered from main bus B. These indicators were indicating their power-off springloaded positions and were not indicative of system status since main bus B power had been lost. The valve position indicators for Quads B and D were powered from main bus A and were valid. Helium isolation valves 1 on Quad B and 1 and 2 on Quad D were probably closed by a shock wave associated with the incident.

At 55:57:44, Quad C helium manifold pressure rose 3 psi; and at 55:58:07, Quad D helium tank temperature rose 1.5 degrees. These two occurrences could also indicate an ambient temperature increase.

Just prior to 56:00:00, the helium pressures of Quads A and B began dropping, indicating considerable propellant usage - probably in response to the venting effects. It can therefore be concluded that Quads A and B were functional after the incident. Although Quad D helium tank pressure remained steady, Quad D manifold-pressure instrumentation indicated decaying pressures after the incident. This is consistent with the reported helium isolation valve closures and shows that the quad operated for a short period on residual ullage pressure. The Quad D helium isolation valves were apparently reopened at 56:00:49, as indicated by a drop in helium tank pressure and a simultaneous rise in manifold pressures. Since Quad C showed neither helium-pressure decay nor manifold-pressure decay in the period following the incident, it must be assumed that shock caused closure of the two primary and two secondary propellant-isolation valves. This conclusion is also validated by the fact that it was necessary to place the Quad A negative-pitch jet in AUTO at 57:05:08 to control a persistent pitch rate. It may therefore be assumed that the pitch engines of Quad C, which were in AUTO at that time, were unable to produce thrust because the Quad C propellant isolation valves were closed.

At 56:04:06, Quad C helium and oxidizer manifold pressures decreased to their normal levels indicating a return to normal ambient temperatures.

At 56:05:48, the Quad C package temperature fell below the normal operating range, indicating that the heaters were inoperative. These heaters were connected to main bus B, which had lost power at that time.

At 56:11:00, Quad C propellant consumption was noted to have been approximately zero since the incident. There was no indication of propellant usage by this quad for the remainder of the mission.

At 56:39:05, a caution and warning system alarm was received for SM-RCS Quad B. The alarm was found to have been caused by a high package temperature (201 F). This temperature increase was caused by heat soakback resulting from heavy engine usage.

At 56:50:02, a second status check of the quad isolation valves was made by the crew. At that time, only the secondary propellant valve indicators for Quads A and C showed barberpole (closed) condition. The indicators were not providing valid signals since main bus B was not supplying power.

SM-RCS Thermal Control System

The SM-RCS engine-package temperatures were within the expected ranges prior to the incident. The highest temperatures recorded prior to that time were 167 F, 155 F, 175 F, and 152 F for Quads A, B, C, and D, respectively. These temperatures were recorded at 3:30:00 and were the result of soakback following the extensive firings of the transposition, docking, and LM ejection maneuvers. Similar temperature increases were noted on Apollo 9 after extensive engine firing.

Three of the four package temperatures rose sharply after the O₂ tank incident and the Quad B caution and warning light was activated at 56:39:05. The indicated temperatures of Quads A, B, and D rose to 182 F, 201 F, and 203 F, respectively, in the time period between 56:30:00 and 57:43:00. These temperature increases were probably caused by soakback from engine firings. The quad heaters were turned off at 58:02:00.

SM-RCS engine-package temperatures were obtained at the time of the PCM transmissions at 101:59:09 and 123:10:00. While the differing solar environment for the four quads resulted in significant variation between the quads, the average temperatures were 72 F and 71 F. Specific engine-package temperatures were:

Quad	101:59:09	123:10:00
A (SR5065T)	85 F	50 F
B (SR5066T)	80 F	47 F
C (SR5067T)	78 F	96 F
D (SR5068T)	44 F	92 F

Propellant Utilization

A time history of propellant remaining is presented in Figure 4.11-9. The quantities of propellant used during the mission were 86, 67, 34, and 104 pounds for Quads A, B, C, and D, respectively; with a total usage of 291 pounds. The conditions for switchover to secondary propellants were not reached.

CM-RCS

Prelaunch Operations

Fuel loading was accomplished on March 9, 1970 and oxidizer loading on March 11, 1970. The loaded weights were as follows:

System	Oxidizer (lbs at 69F)	Fuel (1bs at 70F)
1	77.8	44.2
2	78.5	44.6

Helium servicing was accomplished on April 4, 1970. The helium tanks were pressurized to the following values at the time of coupling closeout:

System	Pressure (psia)	Temperature (deg F)
1	4140	65
2	4140	67

Mission Performance

At liftoff, the helium bottle pressures were 4179 and 4177 psia for systems 1 and 2, respectively. The corresponding helium tank temperatures were 72 F and 68 F. The helium manifold pressures were 89 and 72 psia for systems 1 and 2, respectively.

The time history of CM-RCS parameters is presented in Figure 4.11-10. Continuous data transmission ceased when the CSM was powered down at 58:39:16. System status data were received when the telemetry system was energized briefly at 102 and 123 hours, approximately. Data transmission was resumed at 140:21:00 when the command module systems were powered for entry.

The small variations in helium tank pressures and temperatures during the 58 hours prior to CSM powerdown are attributed to changes in environmental heating or cooling resulting from changes in vehicle orientation. All CM-RCS parameters remained within their specified ranges during this period. The helium pressures show no sign of decay due to leakage. A decline in helium pressure occurred after the CMS was powered down. This decline is attributed to cooldown of the helium tanks.

CM-RCS Thermal Control

The off-nominal conditions of the Apollo 13 mission made it necessary to perform a CM-RCS valve preheating sequence for the first time on any Apollo flight.

The data obtained at 102:09:01 indicated that two of the injector temperatures were below the redline limit for preheat (28 F). Four of the six temperature readings were below 28 F at 123:22:44, and it was decided to preheat the valves with LM power at entry interface minus 6.5 hours. The preheat was initiated at 136:23:00 and was maintained for 20 minutes. The temperatures reported by the crew were:

A	Temperature (Degrees F)														
Engine	System	102:09:01	141:04:57												
-Pitch	2	42.3	29.7	31.8		36.0									
+Yaw	2	21.1	14.8	25.3	All	44.2									
-Roll	1	31.7	29.4	21.1	Above	38.0									
-Pitch	1	44.1	23.2	33.6	29.5F	37.8									
-Yaw	1	30.8	20.4	35.0		22.5									
+Roll	2	27.3	18.9	27.3		23.1									

None of the positive-pitch injectors on the Block II CM-RCS engines have temperature sensors. The nearest injector temperature sensors are on the roll engines nearest the pitch engines. The temperature response of the roll and yaw engines indicate that the positive-pitch engine area may have been cold-biased immediately prior to the valve warmup sequence. Previous analysis has indicated that the maximum engine-to-engine temperature

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difference is 7 F. On this basis, the positive-pitch values could have been as low as 9 F just prior to the value warmup sequence. However, it can be concluded from the roll engine injector temperatures at 141:04:57 that the positive-pitch values, in company with all of the other CM-RCS engine values, remained above their minimum operating limit of 20 F from preheat to splashdown.

Entry Phase (System Activation to Touchdown)

To conserve power, it was decided that the data storage equipment (DSE) should not be powered during entry. The data available for evaluation of CM-RCS entry performance are therefore very sparse. The data that are available indicate that both of the CM-RCS systems operated satisfactorily throughout the entry phase.

A successful test firing of all 12 engines was reported at 137:57:48, shortly before service-module jettison. The data acquired when telemetry transmission resumed at 140:19:00 showed that the helium tank pressures dropped 582 psi and 571 psi for systems 1 and 2, respectively, at system activation. The helium manifold pressures were reading within normal range at 293 psia and 292 psia, respectively.

Figures 4.11-11 and 4.11-12 present interpolated time histories for the propellant burn and system purge period. The purge was terminated at approximately 142:51:40, with a residual pressure of 223 psia locked in the system. The purge was probably terminated by returning the purge switch to the off position (the postrecovery check list shows that the purge switch was found in the off position). The system 1 fuel isolation valve was found in the open position during postrecovery decontamination operations. This open valve would have permitted the helium tanks to bleed down to atmospheric ambient level if only the propellant isolation valve switches had been actuated. The isolation valve switches must have been actuated per crew check list during the final stages of descent, because the other three isolation valves were found closed.

CM-RCS Propellant Utilization

Propellant usage was estimated by using the pressure-temperature propellant consumption nomogram from the CSM Operational Data Book. It is estimated that 9 pounds of propellant were used from system 1 prior to reaching entry interface. From entry interface to splashdown, an additional 52 pounds were used out of system 1. System 2 was not used between the test firing and propellant depletion firing.

Postflight Operations

The aircraft carrier USS Iwo Jima delivered the Apollo 13 command module to Pier Bravo, Pearl Harbor, Hawaii, at 0900 Hawaii time, on April 24, 1970. The command module was off-loaded at 1000, and was transported to Hangar 5 at Hickam Field at 1155.

During the postflight inspection preparatory to initiating decontamination procedures, the system 1 fuel isolation valve (C19LV4) was found in the open position. The system 1 oxidizer isolation valve and both of the system 2 propellant (fuel and oxidizer) isolation valves were in the closed position, which is the normal postflight position. The system 1 fuel isolation valve appeared to operate in a normal manner when it was controlled by GSE during the decontamination operation. The protective covers on the relief valves were still in place. There was an estimated helium pressure of 165 psia remaining in the system, which was bled through the helium by-pass and the fill couplings in 1 minute and 35 seconds. There was no residual fuel or oxidizer. The engines contained sea water.

Decontamination of the fuel system and the oxidizer system was completed at 0857, April 26, 1970. The command module arrived at Downey on April 28, 1970.



Figure 4.11-1 SM-RCS Quad A Manifold Pressures and Package Temperature





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Figure 4.11-5 SM-RCS Quad C Manifold Pressures and Package Temperature



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Figure 4.11-6 SM-RCS Quad C Helium Tank Temperature and Pressure

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Figure 4.11-7 SM-RCS Quad D Manifold Pressures and Package Temperature



Figure 4.11-8 SM-RCS Quad D Helium Tank Temperature and Pressure

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Ground Elapsed Time - Hours

Figure 4.11-9 Apollo 13 SM-RCS Propellant Usage

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4.12 SERVICE PROPULSION SYSTEM

The SPS was fired for 3.6 seconds at 30:40:49.6 to transfer the spacecraft from a free-return orbit to a non-free-return orbit. The oxygen storage system failure which occurred at 55:54:53 resulted in loss of fuel-cell power and precluded further use of the SPS.

Prelaunch Operations

Propellant servicing was performed between March 10 and March 12, 1970. The quantities of oxidizer and fuel loaded were 25,084 pounds and 15,685 pounds, respectively. Propellant system pressurization to flight pressure levels was accomplished during helium servicing on April 7. Helium absorption by the propellants prior to pressurization caused decreases in the oxidizer and fuel tank pressures of 14 psi and 4 psi, respectively. These decrements are approximately the same as experienced on all previous fully-loaded spacecrafts.

An anomaly was observed in the propellant utilization and gaging system (PUGS) primary fuel-quantity readout on March 10. Before any propellant was loaded, the fuel sump-tank-quantity telemetry signal and the fuel total-quantity display in the command module oscillated for two short periods. Similar short-duration oscillations were observed during the PUGS preservicing checkout, but the source of the problem was not identified. The oscillations occurred only in the fuel sump-quantity reading; they did not occur with propellant in the tank; and did not occur during the first nine minutes of operation. The last two observations tended to indicate that the problem would not reoccur in flight, since sump-tank propellant is never depleted and maximum PUGS usage time was planned to be six minutes only.

Another PUGS problem, apparently not related, was the loss of the point sensors in the auxiliary sensing system. All of the point sensors operated properly throughout the fuel loading which was performed on March 10. However, when the PUGS was powered up two days later for oxidizer loading, point sensor No. 2 (second from the top in the storage tank) was improperly indicating an uncovered condition. On the next day, when the PUGS was powered up for alignment of the auxiliary integrators, the fuel point-sensor No. 2 was still out, and Nos. 8 and 9 (the two top sensors in the sump tank) were also out. On March 23, when the PUGS was powered up for preflight checks in the Countdown Demonstration Test, the point-sensor system failed completely. When power was first applied, sensors Nos. 8, 9, 10, 11, and 12 were out. Within minutes, all of the remaining point sensors became inoperative.

This point-sensor system failure occurred in approximately the same sequence as the failure on Apollo 10. Both failures are believed to have been caused by leakage of fuel through the point-sensor lead feedthrough into the hermetically-sealed tower tube which is located in the center of each tank. Fuel which leaks into a tower tube collects on unprotected printed circuits, and results in electrical shorting. Tests have shown that this condition does not create a hazard. This failure mode does not affect operation of the PUGS primary system.

Pressure Decay After Launch

A large decrease in propellant tank pressures has been observed during the first several hours after launch of each fully-loaded spacecraft. On Apollo 13, the oxidizer tank pressure decreased from 183.6 psia to 168.5 psia between liftoff and 18:00:00. Fuel tank pressure decreased from 183.0 psia to 176.6 psia between liftoff and 9:00:00. The three factors contributing to the decrease are (1) tank stretch resulting from removal of atmospheric pressure on the outside of the tanks; (2) cooling of ullage gas as propellant surrounds the bubble in zero-g; and (3) absorption of helium by the propellant, again as propellant surrounds the bubble in zero-g.

During the Apollo 13 mission there were additional decreases in fuel and oxidizer pressure during the long period between the SPS burn and service module separation. These further decreases are attributed to (1) low temperatures in the service module resulting from shutdown of the fuel cells, loss of the Sector 4 panel, damage to insulation, and overall cooling of the spacecraft during the transearth coast; and (2) additional helium absorption caused by mixing of helium and propellants during the LM-DPS firings.

Second Midcourse Correction (MCC-2)

The single SPS firing was for MCC-2. Ignition occurred at 30:40:49.6. The burn duration was 3.46 seconds and the resultant velocity change was 23.2 feet per second. The burn was performed on engine valve-bank A.

The propellant utilization (PU) valve was in INCREASE during this burn. It had been placed in INCREASE on the launch pad. In the planned mission, the PU valve would have remained in INCREASE until propellant crossover to compensate for fuel-rich engine operation. After crossover, which occurs during LOI-1, the propellant unbalance meter would have been monitored to determine optimum PU valve position.

Start, shutdown, and steady-state data were normal for first-burn, single-bank, operation. A 30-psi spike, which increased the oxidizer tank pressure readout to 202 psia, occurred 0.1 second after fire switch 1 indication. This transient increase in transducer readout is attributed to the presence of a quantity of liquid oxidizer in the helium line between the storage tank and the check valve. The recovery of the tank pressure to normal level within 0.1 second indicates that the ullage gas pressure itself did not change. The caution and warning system upper limit for oxidizer tank pressure is approximately 200 psia, with the actual trigger level varying with individual transducers. The caution and warning system was not triggered by the spike.

Gimbal performance was normal. The differential clutch current data recorded during the steady-state portion of the burn indicate thrust misalignments of 0.063 inch in pitch and 0.024 inch in yaw. These misalignments are smaller than the average of those experienced for the first SPS burn of previous Apollo missions. Overall error in steering prediction was quite small. Pitch error was the larger with an overall system error of 0.188 degree. The allowable angular error is 0.50 degree.

Oxidizer Interface Pressure Drift

The ratio of indicated versus actual pressure for the engine interface pressure measurement for oxidizer (SP0931P) drifted upward throughout the mission. The actual error can be estimated by comparing the interface pressure data with the tank pressure data. The interface pressure data were biased approximately three psi low at the time of final pressurization for flight (five days prior to liftoff). The error decreased to zero at approximately 20:00:00 and then increased in the positive direction throughout the mission. The last SPS data were transmitted at 123:12:06, when the oxidizer interface pressure reading was approximately eight psi high.

The most probable cause of the upward drift is leakage from the cavity behind the sensing diaphragm of the pressure transducer. The cavity is pressurized to ambient atmospheric level during assembly. Subsequent leakage from the cavity causes an increase in pressure differential across the diaphragm, and results in an increase in indicated pressure.

SPS Reactions to the Oxygen Tank Anomaly

Pressure in oxygen tank No. 2 in Sector 4 of the service module was lost at 55:54:53.

A review of SPS data revealed that two SPS measurements reacted to the incident. Helium tank temperature (SPOOO2T), which was 83F before the incident, increased rapidly to 92 F. It then decreased steadily and was indicating 82 F when high-bit-rate (HBR) data transmission was discontinued at 58:06:16. The sensor is mounted on the 0.5-inch diameter stainless-steel helium supply line approximately 2.5 inches from the upper helium-tank outlet boss. It is thirteen inches from fuel cell No. 2, and is behind the inner edge of beam No. 3. There is an open path between sector 4 and the area of the measurement. The measurement is responsive to the compartment temperature between fuel cell No. 2 and the upper SPS helium tank in the vicinity of beam No. 3 when the SPS pressurization system is not operating. Since the SPS was not operating at the time, it is apparent that the rapid temperature increase was caused by heat released within the service module.

The other SPS measurement which reacted to the cryogenic tank anomaly was the oxidizer storage-tank surface temperature (SA2378T). That measurement increased sharply from 73 F to 78 F at the time of the incident. It then decreased steadily and was indicating 60 F when HBR data was discontinued. The sensor is located halfway up the tank and on the outboard side. As with the helium temperature measurement, the rapid temperature increase at the time of the incident must have been caused by heat released within the service module. The subsequent decrease to 60 F indicates that the tank insulation over the sensor was damaged. Figure 4.12-1 graphically illustrates the reactions of the two temperature sensors.

Staining of the Nozzle Extension

Following service module separation, the crew reported that they could see dark-brown streaked stains on the SPS engine-nozzle extension. This streaking was the result of melting of the vinyl coating applied for protection of the nozzle during handling. The dark green vinyl coating (TEC704) is applied to the entire Columbium (Niobium) portion of the nozzle extension from the chamber flange to approximately 57 inches aft of the flange. The coating burns off during the initial engine firings. It flows and discolors to black or brown shades at approximately 300 F. The coated portion of the nozzle would have been heated to about 700 F during the one, short, SPS burn.





4.13 THERMAL CONTROL

The temperatures of all instrumented passive elements of the Apollo 13 CSM were observed to be within specified limits from liftoff to power down at 58:39; during the two short powered periods commencing at 101:59 and 123:07; and from 140:21 to splashdown. It appears that the temperatures remained within limits during the remainder of the mission, when the CSM was powered down and data were not available.

Figures 4.13-1 through 4.13-5 present temperature histories for (1) the command module side heatshield at four locations, (2) the SM-RCS helium tanks, (3) the SPS propellant tank skins, (4) the docking probe cylinder, (5) the CM-RCS helium tanks, and (6) the SPS distribution lines.

Service Module Temperatures Prior to the O2 Tank Anomaly

The service module structure and component temperatures prior to 55:16 were similar to those recorded during previous missions. A postflight simulation of this phase has been performed for the purpose of comparing measured and predicted temperatures. The simulation utilized the service module 3-D thermal math model and the actual sun-look-angle attitude history. The results of the simulation are plotted with the flight data in Figures 4.13-3 and 4.13-4. The excellent correlation indicates nominal performance of the service module thermal control system up to the time of the O₂ tank incident. The bay 4 structure and insulation temperatures obtained from the simulation were normal and as expected.

SPS Oxidizer Storage-Tank Skin Temperature

The SPS oxidizer storage-tank skin-temperature sensor (SA2378T) exhibited a rapid and unprecedented rate of change of temperature immediately following the O_2 tank incident. The remaining SPS tank skin temperature sensors responded normally. It is concluded that the pressure wave which tore the outer panel from bay 4 dislodged the aluminized mylar insulation blanket from the SPS storage tank to the extent that the SA2378 sensor became exposed to the ambient conditions of bay 3. Figure 4.13-6 illustrates the SPS tank insulation placement and the location of the sensor under the blanket.

Predictions of the expected response from an uninsulated sensor were made and are compared with those for an insulated sensor and the Apollo 13 mission data in Figure 4.13-7. The bay 3 outer shell temperature at the time that the response of SA2378T varied sharply was inferred from data from the sun sensors mounted on the fairing on either side of bay 3 (SA1831T and SA1832T) and was set at 0°F. The single-dimensional thermal math model used for the prediction is also illustrated in Figure 4.13-7. Good correlation was obtained between the measured and predicted values for an uninsulated sensor. The data acquired during the two short status-check periods at 102 and 123 hours confirm that the insulation was dislodged. This conclusion is based on the fact that, with the exception of SA2378T, all SPS-tank skin temperatures appear to have slowly decreased, although remaining close to their initial temperatures at liftoff (Figure 4.13-4). The slow cooling trend is as expected in view of the powered-down status of the CSM. However, SA2378T did not exhibit this trend and, in fact, was warmer at 123 hours than at 102 hours. This implies that SA2378T was more sensitive to changes in the external thermal environment than the remaining SPS tank sensors; a condition which would be expected after the insulation had been dislodged.

Command Module Temperature Response Following Powerdown

Thermal control of the structure and components located within the unpressurized aft equipment bay is strongly dependent on heat dissipation by electronic equipment located within the cabin. This equipment maintains a relatively-warm cabin wall temperature which, in turn, elevates aft equipment bay temperatures. Following shutdown of the electronic equipment and the ECS system, temperature control of the aft equipment bay became entirely dependent on external environmental heating to the conic heatshield and thermal inertia. Since the command module external temperature coating is a cold-biased coating ($a_s/\epsilon = 0.40$ to 0.54), an overall net heatloss from the command module occurs when internal temperature drops. Figures 4.13-1 and 4.13-2 show that, on the average, conic heatshield surface temperatures ranging from -20F to +20F resulted during PTC with the CSM powered down.

A simulation of the cooling trend following powerdown was made with the command module 3-D thernal math model. It was necessary to modify the model to the extent that components within the pressure vessel which are normally treated as constant temperature heat sinks had to be treated as floating, or variable, temperature components. Figures 4.13-8 through 4.13-12 present the results of the simulation for selected locations in the aft equipment bay. For the purpose of the simulation, it was assumed that the command module X-axis was perpendicular to the vehicle-sun line and that the spacecraft was rotated 90 degrees about the X-axis every hour, with attitude hold used between the maneuvers.

Figures 4.13-8 and 4.13-9 show the predicted response of the CM-RCS helium tanks and the actual data points which were acquired. Comparisons between the predicted and measured temperatures of the helium tanks show that the math model produced predicted temperatures that were five to ten degrees lower than those measured. The application of a 5F to 10F correction factor to the predictions for the temperature-critical RCS and ECS tanks shown in Figures 4.13-10 and 4.13-11 results in predicted RCS tank temperatures between 40F and 49F, and water-tank temperatures between 39F and 52F, at entry interface. These predicted temperatures are all within specified limits. Estimated RCS and ECS line temperatures can be derived from the cabin-wall temperatures shown in Figure 4.13-12. Application of the 5F to 10F correction factor to these predictions produces RCS line temperatures at entry interface ranging from 30F to 40F. The lower limit for the feedlines is 20F. The waterlines in the aft equipment bay are located on the +Z/-Y side of the command module. Application of the correction factor to the wall predictions in that area of the compartment produces predicted waterline temperatures ranging from 34F to 39F at entry interface. It is unlikely that freezing of the waterlines occurred.

It is calculated that aft equipment bay temperatures at entry interface would have been from 10F to 30F colder if the service module had been jettisoned soon after the O2 tank incident. The colder temperatures would have been a direct result of exposing the aft heatshield to the deep space environment.

Atmospheric Entry

Postrecovery inspection of the basic heatshield verified satisfactory performance of the thermal protection system during atmospheric entry.

The general appearance of the heatshield ablator was similar to that of previous lunar-return spacecraft. The degree and distribution of the fully-blackened aft toroidal heatshield appeared normal. Surface striations visible in the pad downstream areas are like those observed on the heatshield of Apollo 8. Char damage in the torus area was caused by recovery and postrecovery handling. Above the CM-SM umbilical, the windward conic heatshield was only slightly blackened. The leeward conic heatshield was virtually undegraded. Decals on the side crew hatch and the black paint on the rendezvous window wells were scorched.

It appears that the command module thermal protection system was not affected by the O_2 tank incident.

The heatshield components that were examined (pads, windows, S-band antennas, dump plugs, umbilical, and astrosextant assembly) were in a thermally satisfactory condition. The EVA handrail was again found to be only locally melted in the region downstream of the astrosextant assembly.



Ground Elapsed Time - Hours

Figure 4.13-1 Apollo 13 CM Heatshield Surface Temperature Histories



Figure 4.13-2 Apollo 13 CM Heatshield Surface Temperature Histories

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Figure 4.13-3 Apollo 13 SM-RCS Helium Tank Temperature Histories

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Figure 4.13-4 Apollo 13 SPS Tank Skin Temperature Histories

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Figure 4.13-5 Apollo 13 Selected Structural Temperature Histories

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Figure 4.13-6 Location of SA2378T and SPS Tank Insulation



Figure 4.13-7 Effect of Mylar Insulation on SPS Oxidizer Tank Skin Temperature Response





. Figure 4.13-8 Temperature History of Apollo 13 CM-RCS Helium Tank A (CROOQ3T) .

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Temperature (Degrees F)

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Figure 4.13-10 Temperature History of Apollo 13 CM-RCS Propellant Tanks

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Temperature (Degress F)





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Figure 4.13-12 Temperature History of the Aft Equipment Bay Aluminum Cabin Wall

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4.14 CONSUMABLES

Electrical Power - General

The three fuel cells were the prime source of dc power from liftoff to 55:54:53 and were supplemented by batteries A and B during launch and the single SPS burn.

The O₂ tank incident at 55:54:53 cut off the oxygen supplies of fuel cells Nos. 1 and 3 and the output of those two cells dropped to zero. Fuel cell No. 2 continued to supply power for 2 hours and 44 minutes, i.e., until the residual pressure in O₂ tank No. 1 approached the minimum level for fuel cell operation. Battery A provided supplementary power for two periods of 33 and 35 minutes, respectively.

Batter	Battery A Battery B		Batte	ry C	Bumaga	
From	To	From	То	From	То	I ut.bose
00:00	00:13	00:00	00:13			To support launch
30:35	30:42	30:35	30:42			To support SPS burn
56:03	56:36					To maintain bus voltage
58:04	58:39					To maintain bus voltage
		101:59	102:02		State 24	Instrument Readings
140:10	Splash	140:10	Splash	138.00	Splash	Entry

The entry batteries were used for the following periods:

The batteries were recharged for the following periods:

Batte	ry A	Battery B				
From	То	From	То			
23:15 57:27 112:07 123:21	25:50 57:49 122:47 125:58	04:25 52:30 126:00	12:35 55:51 127:59			

The lunar module batteries provided approximately 120 AH of electrical power to the command module buses between 112:06 and 140:10 to recharge batteries A and B for entry.

Spacecraft Electrical Loads

The electrical loads on the fuel cells averaged 71.6 amperes up to the time of the O₂ incident. This value was calculated by statistically averaging the fuel cell output current readings and compares favorably with the average load value of 71.9 amperes computed on the basis of fuel-cell hydrogen consumption. Based on the averaged load data and on hydrogen consumption, the energy output was 4002 and 4027 ampere-hours, respectively. The average kilowatt-hours, with an average main dc bus voltage of 29.2 volts and average loads of 71.6 and 71.9 amperes, amounted to 116.9 and 117.5 KWH, respectively. Total fuel cell current immediately prior to the O₂ incident was 70.3 amperes. The output of fuel cell No. 2 increased to over 50 amperes as the output of the other two fuel cells decayed. The peak output of fuel cell No. 2 was 66.3 amperes at 23.84 vdc at 56:00:00. Battery A was connected to the bus at 56:03:08 and assumed 34 percent of an approximately 64-ampere load. Fuel cell No. 2 output ranged between 29 and 37 amperes during the 33-minute period during which battery A was connected to the bus. Battery A output ranged between 11 and 20 amperes. Battery A was reconnected to the bus at 58:04:01 and assumed 33 percent of the load. From then until fuel cell No. 2 ceased to provide power at approximately 58:15:00, the fuel cell output ranged from 22.5 to 26.0 amperes and battery output ranged from 10.6 to 14.6 amperes.

Battery output from 55:58:15 until CSM powerdown was completed at 58:39:15 ranged downwards from 34.3 to 12.6 amperes.

Battery B was connected to the CM buses at 101:59 for three minutes to provide power for a systems status check and telemetry transmission.

Battery C was connected to main bus A at 138:00 for pre-entry checks. Batteries A and B were connected to main buses A and B, respectively, at 140:10. Battery C was tied to main bus B at the same time.

CSM Power to Lunar Module

The CSM supplied electrical power to the LM heater loads from approximately 03:50:00 to 55:58:00. The loads ranged from 0.4 to 6.75 amperes and averaged 1.73 amperes. The total energy supplied was approximately 2.82 KWH.

Load Sharing by Fuel Cells and Batteries

Event	Fuel C Load (per	ell/Batt Sharing cent)	tery	Battery Energy (AH)		Remarks			
	Fuel Cells	Batts.	A	B	C				
Launch	88	12	1.02	0.90	-	Batt A & B			
SPS Burn	75	25	1.95	1.34	-	Batt A & B			
56:03 to 56:36	66	34	18.98	-		Batt A			
58:04 to	67	33	2.20	-	-	Batt A			
58:15 to	0	100	6.2		-	Batt A			
101:59 to 102:02	0	100	-	0.80	· .	Batt B			
138:00 to Splash	0	100	25.9	31.3	33.8	Batt A, B & C			

The CSM loads were shared as follows:

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Main DC Bus Voltages

The main dc bus voltages ranged between 27.6 and 29.72 volts from liftoff to 55:54:53. The average bus voltage during this period was 29.2 vdc. Main bus B dropped to zero at approximately 55:58:00 when fuel cell No. 3 ceased to provide power. Main bus A dropped to 25.5 vdc at the same time because fuel cell No. 1 also ceased to provide power. Fuel cell No. 2 maintained bus A at approximinately 25.5 volts until Battery A was placed on the bus and increased the voltage level to around 28 vdc. The CSM was partly powered down and bus A voltage was maintained between 28.6 and 29.0 vdc until the battery was disconnected at 58:36:07. The bus voltages ranged between 26.0 and 27.5 vdc between 58:36:07 and 58:04:1, when battery A was reconnected to the bus. Bus voltage was then maintained around 29.18 vdc until fuel cell No. 2 ceased to provide power at approximately 58:14:00. The bus voltage then decreased to 27.2 but gradually rose to 29.18 vdc as the CSM was powered down. Powerdown was completed at 59:39:15.

Hydrogen Consumption

The total quantity of hydrogen consumed was 10.4 pounds. The two tanks shared the mission requirements with a 47:53 ratio. Approximately 0.05 pound of hydrogen was used for fuel-cell purging. The net hydrogen consumption of 10.35 pounds equates to a theoretical average power production of 71.9 amperes. This value compares well with the 71.6 amperes obtained from the fuel-cell current measurements and indicates a conversion efficiency of 99 percent. The significant hydrogen usage parameters were:

	Tank No. 1 (pounds)	Tank No. 2 (pounds)	Total (pounds)					
Loaded Less prelaunch usage	27.7 2.1	27.9 <u>1.6</u>	55.6 <u>3.7</u>					
Remaining at liftoff Less flight usage	25.6	26.3	51.9 * <u>10.4</u>					
Remaining at 55:54	20.7	20.8	41.5					
Average flight usage rate = $\frac{10.35 \text{ lbs}}{55.9 \text{ hrs}}$ = 0.185 lb/hr								
Equivalent Load	= 0.18	$\frac{5 \text{ lb/hr}}{257 \text{ lb/AH}} = 7$	1.9 amperes					
Equivalent Energy	$=\frac{10.3}{0.00}$	$\frac{5 \text{ lbs}}{257 \text{ lb/AH}} = 4$	027 ampere-hours					

*Includes 0.05 pound used for purging.

Oxygen Consumption

The loss of the O_2 tank No. 2 quantity measurement at approximately 46 hours precluded accurate determination of quantity after that time. The following oxygen consumption analysis therefore only covers the period up to 46:05:15.

The total quantity of oxygen consumed up to 46:05:15 was approximately 91.4 lbs. The tank sharing ratio was 51:49. Approximately 0.12 pound of oxygen was used for purging the fuel cells. An estimated 63.1 pounds were consumed by the fuel cells for generation of electrical energy; 14.4 pounds (0.312 lb/hr) by the ECS; and 6.9 pounds were used for pressurizing the LM and the tunnel. The significant oxygen usage parameters were:

	Tank No. 1 (pounds)	Tank No. 2 (pounds)	Total (pounds)
Loaded Less prelaunch usage	325 .3 15.5	323.4 <u>14.5</u>	648.7
Remaining at liftoff Less flight usage	309.8	308.9 <u>44.4</u>	618.7 <u>91.4</u>
Remaining at 46:05:15	262.8	264.5	527.3
Total flight usage Less fuel-cell usage			91.4 <u>63.1</u>
Other usage			28.3
Less ECS usage			14.4
LM usage and cabin purg	ge	n an an an Ariana. An Ariana	13.9
Average fuel cell usage	e rate = $\frac{62}{46}$	<u>3.1 lbs</u> = 1.36 5.1 hrs	8 lb/hr
Average other usage rat	te = $\frac{28}{46}$	<u>3.3 lb</u> = 0.61 6.1 hrs	3 lb/hr
Average ECS usage rate	$= \frac{11}{46}$	4.4 lbs = 0.31	2 lb/hr
Average total usage rat	te = $\frac{91}{46}$	$\frac{1.4 \text{ lbs}}{6.1 \text{ hrs}} = 1.98$	2 lb/hr

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Potable Water

The potable-water tank was filled to a proximately 80 percent (29 pounds) at T-27 hours and was valved off until T-3 hours. The water was chlorinated at T-26 and a 4-pound sample was extracted. The water inlet valve was opened at T-3.9 hours, allowing fuel-cell generated water to enter the tank. The fuel cells were at a load of 75 amps and produced an estimated 5.2 pounds of water, all of which went into the potable tank. The potable tank contained 30.5 pounds at liftoff and was full at 04:00:00. The tank remained full until the crew started to transfer drinking water to the lunar module. A total quantity of 24.3 pounds of water was drained from the potable tank during postflight testing operations, including approximately 0.5 pound from the lines. The net quantity of 23.8 pounds agrees fairly well with the quantity reading of 22.7 pounds which appreared in telemetry transmissions during entry. The crew apparently transferred approximately 14 pounds of water to the lunar module.

Waste Water

The waste-water tank was serviced at T-40.5 hours and the system was configured to allow fuel-cell water production to enter the wastewater tank until T-3.9 hours. The waste-water quantity measurement indicated approximately 25.5 pounds at liftoff, however, the one-g effect causes a significant error in the measurement. A liftoff quantity of 29.5 pounds was estimated based on fuel-cell operation during countdown and backward extrapolation of zero-g data.

Three waste-water dumps were conducted during the mission with an estimated 91 pounds of surplus water being dumped overboard. It is estimated that the waste water tank contained approximately 16.4 pounds at entry interface.

Water Balance

The estimated water balance for the mission was:

Liftoff	Pounds	Pounds
Potable water Waste water	30.5 29.5	60.0
Produced in Flight		
Fuel Cells LiOH Metabolic Oxygen	94.5 5.9 <u>5.8</u>	<u>106.2</u>
Total Loaded and Produced		166.2
Usage		
Transferred to LM Boiloff Urine and Feces Loss	14.0 8.2 <u>19.8</u>	42.0
Remaining at Entry Interface		
Potable water Waste water	23.8 14.7	38.5
Total Usage and Remaining		80.5
Dumped Surplus		
Estimated from water balance Estimated from flight data		85.7 91.0
Difference		5.3

4.15 MASS CHARACTERISTICS

The postflight-determined mass properties data for Apollo 13 do not vary significantly from those presented in the preflight-published Actual Weight and Balance Data document SD69-676, dated January 26, 1970.

Weight Data Comparison

Tt om	Weight (Pounds)				
Trein	Preflight	Postflight			
Command Module Service Module SLA Attach Ring	12531.6 10531.3 98.0	12627.9 10519.7 98.0			
CSM less tanked SPS propellant	23160.9	23245.6			
SPS propellant - Tanked - Usable -Unusable	40012.6 581.1	39985.5 581.1			
CSM with tanked SPS propellant	63754.6	63812.2			
SLA less ring LM-7	3946.7 33447.8	3946.1 33493.0			
Total injected	101149.1	101251.3			
Launch Escape Subsystem	9011.8	9011.8			
Total launched	110160.9	110263.1			

The hypersonic aerodynamic entry L/D after LM jettison was 0.307, based on the center-of-gravity coordinates associated with the preflight aerodynamic requirements.

The difference between the preflight and postflight weights for the command module is mainly attributable to the substitution of John Swigert for Thomas Mattingly as Command Module Pilot and to stowable equipment increases.

Figures 4.15-1 through 4.15-4 present summaries of mass properties at liftoff, after service module and LM separations, and after entry.

Tt em	Weight (Pounds)	Weight Center of Gravity (Pounds) (Inches)			Moments of Inertia (Slug-ft ²)			Products of Inertia (Slug-ft ²)		
Trem		Xa	Y	Z	Ixx	Iyy	Izz	Ixy	Ixz	Iyz
Command Module Service Module SLA attach ring	12627.9 10519.7 98.0	1041.17 918.91 835.70	0.07 -5.85 2.00	5.98 10.88 -6.60	5906 7123 120	5409 11571 65	4859 11077 56	47.9 -127.7 -0.4	-437.1 416.2 1.6	27.3 -582.8 -2.5
Total without SPS prop. SPS propellants tanked	23245.6 40566.6	984.98 905.43	-2.60 7.71	8.14 5.75	13227 20509	36069 22708	35028 25681	801.4 -5.5	-714.2 -15.9	-595.4 3912.0
Total-CSM with tanked prop. SLA less attach ring	63812.2 3946.1 33493.0	934.41 640.37 584.80	3.95 1.69 -0.10	6.62 0.16 -0.20	34094 9933 22460	78980 12557 24844	81231 12496 24916	-1819.3 -107.7 160.0	-121.9 41.0 426.0	3237.8 -13.3 377.0
Total injected into Earth orbit Launch Escape System	101251.3 9011.8	807.30 1298.10	2.53	4.11 0.50	66799 827	720762 27826	722869 27797	5076.3 8.3	12232.8 707.9	3735.6 0.2
Total launched	110263.1	847.41	2.32	3.82	67661	1178869	1180936	2870.9	9774.0	3752.1

Note: The Apollo coordinate system used to define the center-of-gravity locations is described in MD-V14-10.

Figure 4.15-1 Apollo 13 Mass Properties Summary - Launch Configuration

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Item .	Weight (Pounds)	Centers c (Inc	of Gravity ches)	
		Xa	·Y	Z ·
Service module - launch	10519.7	918.91	-5.85	10.88
Plus: SLA attachment ring SPS tanked propellant	98.0 40566.6	835.70 905.43	2.00 7.71	-6.60 5.75
Total	51184.3	908.06	4.91	6.78
Less: SPS propellant for hybrid transfer RCS propellant usage Hydrogen usage Oxygen usage	225.0 274.0 11.0 627.0	969.80 941.80 876.10 920.80	3.40 0.00 -40.70 -24.10	11.00 0.00 41.20 40.70
Total - SM prior to first LM-DPS burn	50047.3	907.45	5.32	6.36
Plus: Command module - launch	12627.9	1041.17	0.07	5.98
Total - CSM prior to first LM-DPS burn	62675.2	934.39	4.26	6.29
Plus: IM in docked configuration CDR transferred to IM CMP transferred to IM LMP transferred to IM	33489.8 173.0 196.0 157.0	1237.30 129.00 129.00 129.00	-0.20 62.10 0.00 13.10	0.00 -11.30 10.40 -11.30
Less: IM DPS propellant for MCC-3	8195.2	1262.10	0.30	-0.10
Total - CSM and LM docked after MCC-3	87969.8	1019.95	3.08	4.45
CMP from LM to CM for SM separation	196.0	-129.00	0.00	-10.40
Less: LM-DPS propellant usage IM APS-RCS propellant usage SM at separation	48.0 350.0 50047.3	1262.80 1146.10 907.45	0.00 0.00 5.32	0.00 0.00 6.36
<u>Total - CM and LM docked after SM Sep</u>	37524.5	1167.84	0.12	1.94

Figure 4.15-2 Apollo 13 Mass Properties Summary - Command Module and LM Docked Configuration after Service Module Separation.

Item	Weight (Pounds)	Centers of Gravity (Inches)				
		X	Y	Z		
CM after SM separation and prior to LM separation (CMP on board)	12297.8	1041,12	0.11	6.42		
Less items transferred from CM to LM: Cabin fan filter and bag Liquid-cooled garments (2) LiOH canisters (4) from B6 Jettison bag Decontamination bags Docking mechanism	4.0 8.4 20.0 0.9 8.0 199.3	1015.00 1033.00 1031.00 1024.00 1012.00 1110.30	0.00 23.00 13.00 45.00 22.00 0.00	-20.00 -50.00 39.00 -26.00 -23.00 0.00		
Plus items transferred from LM to CM: CDR from LM to CM LMP from LM to CM DSEA (R13) Flag kit and 3 PPK's Black and white TV camera Reseau cameras (2) (B6) Water in baggies (B1)	173.0 157.0 2.3 2.4 7.6 7.8 9.0	1043.00 1043.00 1024.00 1012.00 1020.00 1031.00 1050.00	-24.50 24.50 45.00 22.00 25.00 13.00 -27.00	-10.40 -10.40 -26.00 -23.00 7.00 39.00 39.00		
Mission changes prior to LM separation: CO ₂ absorbed - partial Fecal waste Food consumed Potable water consumed Relocate items for unsuited entry Relocate two PGA's CMP to LH couch for entry CDR to center couch for entry	15.0 5.0 42.2 36.0 158.9 96.0 196.0 173.0	1023.50 1039.00 1049.16 1022.60 -16.10 -3.46 0.00 0.00	7.60 47.00 -39.22 -63.50 4.30 -19.02 -24.50 24.50	31.30 12.00 24.62 -16.40 -1.20 0.00 0.00 0.00		
Total - Command Module after LM separation	12358.2	1039.86	0.23	6.14		

Figure 4.15-3 Apollo 13 Mass Properties Summary - CM Pre-Entry Configuration

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Tt em	Weight (Pounds)	Center of Gravity (Inches)			Momer (:	nts of I Slug-ft ²	nertia)	Products of Inertia (Slug-ft ²)		
Trem		Xa	Y	Z	Ixx	Iyy	Izz	İ _{xy}	I _{xz}	Iyz
Command Module at LM sep.	12358.2	1039.86	0.23	6.14	5855	5193	4626	46.6	-414.0	22.6
Less: RCS propellant usage	16.8	1022.60	-5.60	57.00	-4	0	-4	0.0	0.0	0.0
Command Module at entry	12341.4	1039.88	0.23	6.08	5842	5182	4621	46.2	-410.9	23.6
Less: RCS propellant	19.4	1022.60	-5.60	57.00	-4	0	-4	0.0	0.0	0.0
Ablator burnoff Entry coolant water	150.0 6.0	1013.10 1022.60	0.00 -19.70	7.40 62.50	-103 0	-81 0	-78 0	0.0	0.0 0.0	0.0
Total at recov. threshold	12166.0	1040.25	0.26	5.95	5719	5060	4514	45.1	-404.6	26.4
Less: Forward heat-	310.0	1094.30	-0.50	0.80	-64	-26	-23	0.0	0.0	0.0
Drogues and disconnects	80.8	1089.00	0.00	-23.90	-1	1	0	0.0	0.0	0.0
Total at Main deploy	11775.2	1038.49	0.28	6.29	5637	4772	4245	48.2	-359.1	25.9
Less: Pilots and risers	44.5	1089.90	5.90	-5.80	-2	-2	-1	0.0	0.0	0.0
Main chutes RCS propellant usage	401.4 209.8	1089.10 1022.60	-0.40 -5.60	8.50 57.00	-62 -46	-22 -5	-44 -43	0.0	0.0 0.0	0.0 0.0
Total at splashdown	11119.5	1036.76	0.39	5.30	5404	4356	3889	44.6	-330.5	40.5

Note: The Apollo coordinate system used to define the center-of-gravity locations is described in MD-V14-14-10.

Figure 4.15-4 Apollo 13 Mass Properties Summary - Command Module Post-Entry Configuration

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5.0 ANOMALY SUMMARY

The significant anomalies from the Apollo 13 mission are discussed in brief below. Additional details may be found in the applicable evaluation sections of this report.

5.1 Loss of Cryogenic Oxygen Tank No. 2 Pressure

At 55:54:53 the oxygen pressure in tank No. 2 abruptly dropped to zero. The pressure loss was accompanied by a loud bang, a computer restart, a main bus B undervoltage alarm, barber-pole indications from five SM-RCS isolation valves, shock closure of the oxygen reactant valves of fuel cells Nos. 1 and 3, a sharp drop in oxygen tank No. 1 pressure, and data dropout.

Fuel cells Nos. 1 and 3 ceased to supply power when the oxygen trapped in the lines between the reactant values and the fuel cells was consumed. (After approximately 2.5 minutes). It was later discovered that the skin panel of bay 4 of the service module had been blown away and had struck the high-gain antenna and caused slight damage to one of the dishes.

Postflight investigation showed that an electrical short-circuit in the fan-motor wiring in oxygen tank No. 2 had started a fire which resulted in failure of the vacuum dome of the tank and sudden loss of the oxygen in the tank. The supply lines of oxygen tank No. 1 were affected to the extent that the oxygen in that tank became depleted in 2 hours and 20 minutes. The pressure wave generated by the failure blew the skin panel away from panel 4. The departing panel struck the high-gain antenna and caused data breakup for approximately 2 seconds. The results of the investigation are discussed in full detail in separate reports.

The cryogenic oxygen storage subsystem will be redesigned and the modifications will be incorporated in Apollo 14 and subsequent spacecraft. The anomaly is closed.

5.2 Postlanding Ventilation Valve Malfunction

It was found during postrecovery operations that the postlanding ventilation inlet valve was closed and the exhaust valve was open. The mechanical locking handle was jammed between the closed and extended positions.

The postlanding ventilation value is locked in position during flight to prevent accidental opening. The locking handle has a travel of 0.75 inch and must be extended a minimum of 0.5 inch to ensure disengagement of both locking pins. If it is not fully extended and if either of the locking pins is not completely withdrawn, subsequent actuation of the value solenoids will cause the engaged pin and the handle to jam. Postflight investigation showed that the inlet valve pin was partly engaged. The locking mechanism and the valves were tested and found to operate correctly and without difficulty when the locking handle was fully extended. A complete dimensional check of the rigging mechanism was made and the assembly was found to be within specification limits. The anomaly is closed.

5.3 Shaft Fluctuations in Zero Optics Mode

Beginning at approximately 40 hours, fluctuations of up to 0.3 degree were observed in the computer readout of the optics shaft angle. The system had been powered up throughout the flight and had been in the zero optics mode since the star horizon navigation sightings at 31 hours. Crew observation of the manual readout subsequently confirmed that the fluctuation was actually caused by motion of the shaft. The circumstances and time of occurrence were almost identical with a similar situation during the Apollo 12 mission.

An investigation conducted after the Apollo 12 mission did not identify a definite source of the problem, since extreme corrosion from sea water prevented examination of the mechanical drive system, and restricted testing to the power and servo assembly, which contains the major electronic components.

The recurrence of the problem under almost identical circumstances on Apollo 13 indicates that the cause is more likely generic than random and that it is time and/or vacuum dependent. The susceptibility of the shaft rather than the trunnion axis also tends to absolve components common to both axes, such as the electronics and the motor drive amplifier. The shaft loop has been shown to be more sensitive than the trunnion to harmonics of the 800-hertz reference voltage introduced into the forward loop; however, because the level of the required null offset voltage is well above that available by induction, this mechanism is considered unlikely.

The most likely candidate is the half-speed resolver, which is used only in the shaft axis and only to provide an unambiguous zero reference. The reference voltage is applied to the rotor through slip rings. The cosine winding is not used and is normally shorted out. However, if there is any resistance in the common ground path through the slip ring, then a portion of the reference voltage will appear across the cosine winding and the apparent output null will be offset from zero degrees. Tests indicate that a resistance of 50 ohms will cause an offset of 0.5 degree.

Some evidence of susceptibility to vacuum was exhibited in this class of resolver when variations of approximately 5 ohms were observed in slip-ring resistance during thermal-vacuum testing. However, the tests were run with the units rotating at 1 rpm, and the momentary changes in resistance disappeared with the wiping action. The slip-ring resistance mechanism meets all the bounds and constraints on the problem. It is unique to the shaft axis, since none of the other resolvers in the system use slip rings. This resolver is in the optics head, which is vented to a vacuum. The rotation of the optics head in a normal operation would wipe the slip rings clean and explain the delay in the fluctuations for some hours after selecting zero optics. No corrective action is required because accurate zeroing is unaffected and there is no affect in operational modes. The anomaly is closed.

5.4 High-Gain Antenna Acquisition Problem

Difficulty was experienced in achieving lockup on high-gain antenna narrow beam for the television transmission scheduled at approximately 55 hours. The high-gain antenna was selected at 54:59:50 but lockup was not achieved until 55:13:45, and not until the spacecraft had been maneuvered to a new attitude. Lockup was attempted in all three modes (manual, auto track, and auto-reacquisition), all three beamwidths (wide, medium, and narrow), and with both primary and secondary servo electronics.

The anomaly is still under investigation, but results to-date lead to the conclusion that a quality-type defect in manufacture or assembly caused a shift in scan-limit setting in the servo electronics. The anomaly is discussed in detail in Section 4.9-3. The anomaly is open.

5.5 Entry Monitor System 0.05g Light Malfunction

The entry monitor system 0.05g light did not illuminate within 3 seconds after 0.05g was sensed by the guidance system. The entry monitor system is designed to start automatically when 0.05g is sensed by the entry monitor system accelerometer. When this event occurs, the 0.05g light should illuminate, the scroll should begin to drive, and the range-to-go counter should begin to count down. The crew reported the failure of the light but did not know whether or not the scroll or the counter responded before manual backup was activated.

The anomaly could not be duplicated during postflight testing. The results of the tests are discussed in Section 4.2. The anomaly is closed.

5.6 Forward Heatshield Thruster Subsystem Gas Leak

The V36-596130 breech-plenum assembly of the forward heatshield thruster subsystem was removed during postrecovery operations for refurbishment and reuse. It was then noted that hot gas had escaped at the breech-to-plenum interface, eroding the assembly and burning a hole through the gusset side plate. The adjacent fiberglass housing was locally burned but not penetrated. The assembly was disassembled, cleaned, and reassembled, and subjected to pneumatic pressure tests. A similar assembly from the Apollo 12 command module was subjected to hydraulic pressure tests. Both series of tests were satisfactory and no leakage occurred. The assembly procedures for the assemblies to be used in SC-110 and subsequent vehicles have been clarified and tightened to avoid reoccurrence. As a further precautionary measure, fiberglass blast shields will be added to the gussets, docking tunnel, and thruster cylinders. The anomaly is closed.

5.7 Potable Water Quantity Reading Fluctuations

The potable water quantity measurement fluctuated briefly on two occasions during the mission. At 22:41:00, the reading decreased from 98 percent to 79 percent for about 5 minutes and then returned to a normal reading of approximately 102 percent. Another fluctuation was noted at 37:38:00, at which time the reading decreased from its upper limit to 83.5 percent, and then returned to the upper limit over a period of 7 seconds.

Preflight fluctuations of from 2 to 6 percent near the full level were observed once during the countdown demonstration test. Data review indicated a fluctuation of about 4 percent at the half-full level during the flight readiness test.

Potable water transducers have operated erratically on previous missions. Apollo 8 postflight testing traced the failure during that mission to moisture contamination within the transducer. Apollo 12 postflight testing revealed a minute quantity of undetermined contamination on the surface of the resistance wafer. Tests using aluminum hydroxide as a contaminant reproduced the anomalous transducer operation.

Postflight calibration of the Apollo 13 transducer provided readings well within tolerance envelopes. Initial examination after dissassembly disclosed no visible contamination. The investigation is continuing. The anomaly is open.

5.8 Suit Pressure Transducer Malfunction

During launch, the suit pressure transducer reading followed cabin pressure until 00:02:45, when it suddenly dropped from 6.7 to 5.7 psia. The difference between the two measurements decreased to only 0.2 psi by 01:30:00, when the cabin reached its nominal regulated pressure of 5.0 psia. For the shirtsleeve mode, the suit and cabin pressure readings should be nearly equal. The suit pressure measurement responded sluggishly during the normal changes in cabin pressure associated with the initial lunar module pressurization, and indicated as much as one psi low. The measurement remained in the 4.1 to 4.3 psia range until deactivation at 58:39:00.

The measurement indicated correctly during the brief instrumentation power-up periods at 102 and 123 hours. However, the suit indication was approximately 0.3 psi lower than cabin pressure just prior to entry and, prior to landing, had increased to only 7.7 psia although the cabin pressure reading was 13.9 psia. The Apollo 12 suit transducer showed similar erratic operation. Postflight testing determined the cause to be internal contamination from particles which either remained in the transducer from improper cleaning after electroless nickel-plating or were self-generated.

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The Apollo 13 transducer was disassembled by the vendor and electroless nickelplate flakes were found in the gap. Particles of silicone lubricant were found in the sensing port. The investigation is continuing. The anomaly is open.

6.0 POSTFLIGHT TEST PROGRAM

The Apollo 13 command module was offloaded from the USS Iwo Jima on April 24, 1970, at Pearl Harbor, Hawaii, and spacecraft deactivation was accomplished at Hickam AFB. Inspection verified that all normally activated command module ordnance devices had fired: the remainder were safed by removal of the initiators. Residual RCS propellants were expelled into ground support equipment. Deactivation was completed on April 26, 1970, and the command module was transferred to NR, Downey, arriving there on April 28, 1970.

The conduct of the overall test effort was defined by ATR 522700, Block II Postflight Baseline tests, SC-109. That document established the general requirements for inspection, visual survey, ordnance removal, battery removal, water sampling, water-glycol sampling, and heatshield removal. Specific tests to support evaluation of flight anomalies and discrepancies were conducted in accordance with NASA-approved Apollo Spacecraft Hardware Utilization Requests (ASHUR's). Some of the tests were performed at Downey; the remainder were performed at MSC. The results of the tests performed at Downey were documented by individual engineering summary reports (ESR's), which serve as the bases for resolution of the discrepancies and anomalies discussed in Sections 4.0 and 5.0.

Summaries of Apollo 13 ASHUR's and ESR's are contained in Figures 6-1 and 6-2 respectively.

ASHUR Number	Title Subject	ESR Numb er
109000	Stowage Item Disposition	None
109001	Caution and Warning Tone Booster Reuse	None
109002	Hand Controller Handles	None
109003	External Thermal Protective Covering (H-Film)	None
109004	Aft Heatshield Plugs	None
109005R1	Cancel Removal of Radioluminescent Discs at MSC	None
109007R1	ECS 900 psi System, Contaminates/Damage	Pending
109008R1	ECS 900 psi System Gas Sampling and Analysis	SD70-212-04
109009R1	Lunar Topographic Camera System to Hycon	None
109010	Remove Live Ordnance	None
109011	Remove Aft Heatshield	None
109012R1	Remove 35 Radioluminescent Discs	None
109013R1	EPS/SCS Transient Simulation Test	None
109014	Hatch Window Removal	None
109015R1	G&N Optics Anomaly	Pending
109016	Post Landing Vent Valve Anomaly	SD70-212-03
109017	Return Recovered Parachute and Equipment to MSC	None
109018R1	EMSA - Investigate 0.05g Anomaly	SD70-212-10
109019	Remove Stowed Items	None
109020 Rl	Determine Cause of Suit Pressure Transducer Failur	e SD70-212-09
109021 R1	Determine Cause of Potable Water Transducer Failur	e SD70-212-08
109022	Evaluation of Medical Accessories Kit	None
109023	Remove Crew Couch Stabilizer Brace	None
109024R1	G&N 212 Assemblies - Reuse Determination	None
109025	Disposition two DSKY's and Signal Conditioner Assembly	None
109026	Review Inflight Coverall Garments	None
109027	Disposition Displays and Controls Assemblies	None
109028	Verify Integrity System 1 RCS Prop. Isolation Valv	e SD70-212-11
109030	Ship Monocular Assembly 10 x 40 to MSC	None
109031	Food Containers Plus Contents	None
109032	Disposition of Uprighting System	None
109033	Disposition of Postlanding Equipment	None
109034	Refurbish Emergency Oxygen Mask Assembly	None
109035	Test and Analyze Pressure Garment Assemblies	None
109036	Window Shade Light Leak Test at MSC	None
109037R1	Crew Couch Armalong and GNIC Panel 122 Noun List	None
1.09038	Disposition Hasselblad Electric Cameras	None
109039	Investigate Two-Speed Interval Timer Anomalies	None
109040	Disposition and Test 16 mm Data Acquisition Camera	s None

Figure 6-1 (Sheet 1) Summary of Apollo 13 ASHUR's

A SHUR Number	Title Subject	ESR Number
109041	Disposition 16 mm Data Acquisition Camera	None
109042	Disposition 16 mm Data Acquisition Camera	None
109043	Window Shade Fit and Light Leak Test at NR	None
109044	Investigate Medical Accessories Kit Anomaly	None
109047R2	Examination of Breech-Plenum Assy.	Pending
109048	Survey Heatshield Windows for Meteoroid Impact	None
109049	Release Scissors to Astronaut for Evaluation and Use	None
109500	Dry Main Parachute - Inspect	Pending

Figure 6-1 (Sheet 2) Summary of Apollo 13 ASHUR's

ESR Number	Description	Test Requirement
SD70-212-01	Inverter/Lunar Mapping Camera Evaluation	ASHUR 020040
SD70-212-02	Fuel Cell Power Evaluation Test	MCR 12598
SD70-212-03	Postlanding Vent Valve Investigation	ASHUR 109016
SD70-212-04	ECS 900 psi O ₂ System Gas Sampling	ASHUR 109008
SD70-212-05	ECS Water-Glycol Sample Analysis	ATR 522702
SD70-212-06	ECS Potable Water Quantity	ATR 522703
SD70-212-07	ECS Waste Water Quantity	ATR 522704
SD70-212-08	ECS Potable Water Quantity Transducer	ASHUR 109021-R1
SD70-212-09	Suit Inlet-Pressure Transducer	ASHUR 109020-R1
SD70-212-10	EMS Postflight Analysis	ASHUR 109018-R1
SD70-212-11	CM-RCS System 1 Isolation Valve	ASHUR 109028

Figure 6-2 Summary of Apollo 13 ESR's

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