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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 14 MISSION REPORT

(NASA-TM-X-74240) APOLLO 14 MISSION REPORT (NASA) N76-78036

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MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
MAY 1971

MSC-04112

APOLLO 14 MISSION REPORT

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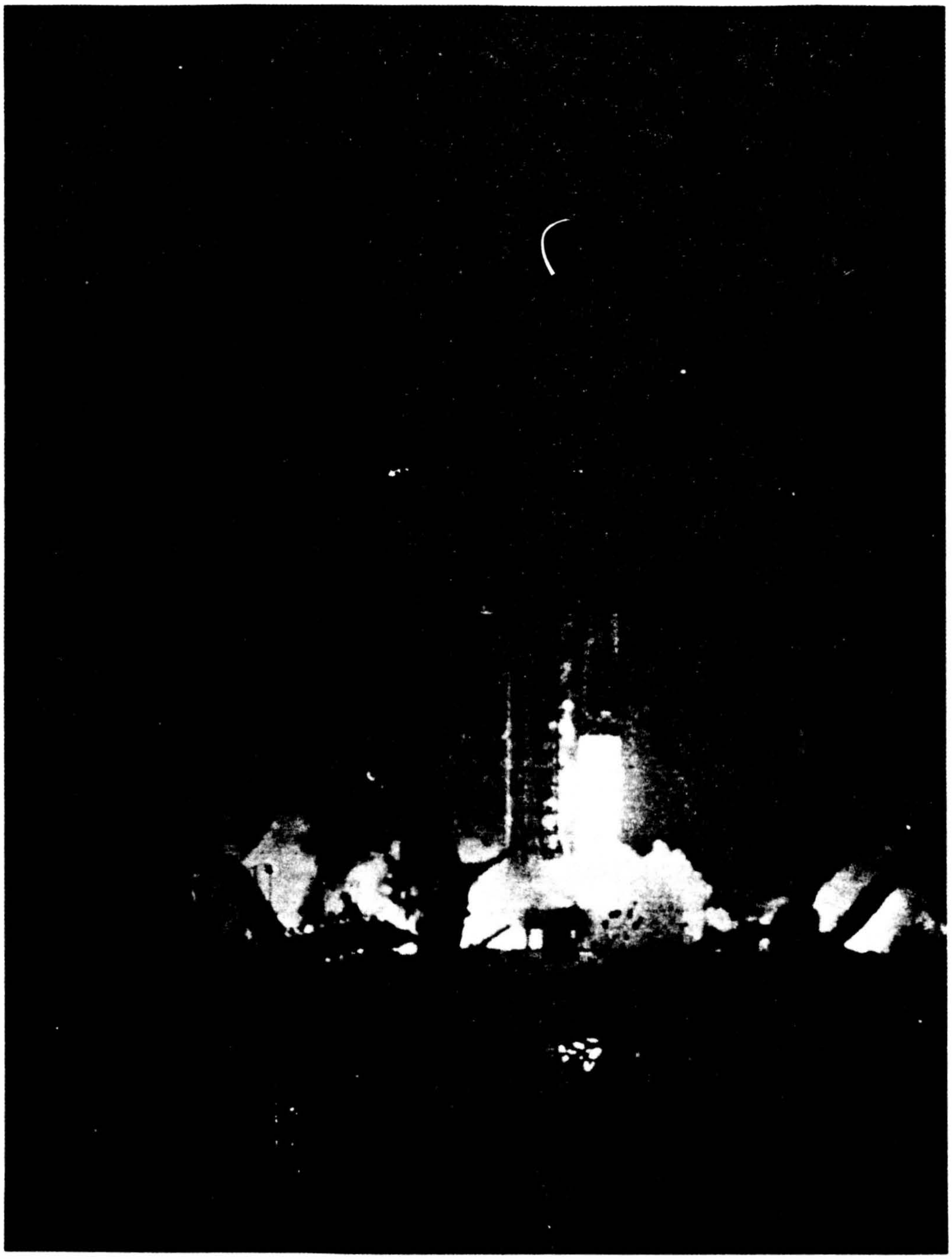
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
April 1971



Apollo 14 lift-off.

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1.0 MISSION SUMMARY

The Apollo 14 mission, manned by Alan Shepard, Jr., Commander; Stuart A. Roosa, Command Module Pilot; and Edgar D. Mitchell, Lunar Module Pilot; was launched from Kennedy Space Center, Florida, at 4:03:02 p.m. e.s.t. (21:03:02 G.m.t.) on January 31, 1971. Because of unsatisfactory weather conditions at the planned time of launch, a launch delay (about 40 minutes) was experienced for the first time in the Apollo program. The activities during earth orbit and translunar injection were similar to those of previous lunar landing missions; however, during transposition and docking following translunar injection, six attempts were required to achieve docking because of mechanical difficulties. Television was used during translunar coast to observe a crew inspection of the probe and drogue. All indications were that the system was functioning normally. Except for a special check of ascent battery 5 in the lunar module, translunar coast after docking proceeded according to the flight plan. Two midcourse corrections were performed, one at about 30-1/2 hours and the other at about 77 hours. These corrections achieved the trajectory required for the desired lunar orbit insertion altitude and time parameters.

The combined spacecraft were inserted into lunar orbit at approximately 82 hours, and two revolutions later, the descent orbit insertion maneuver placed the spacecraft in a 58.8- by 9.1-mile orbit. The lunar module crew entered the vehicle at approximately 101-1/4 hours to prepare for the descent to the lunar surface.

The lunar module was undocked from the command module at about 103-3/4 hours. Prior to powered descent, an abort command was delivered to the computer as the result of a malfunction but a routine was manually loaded in the computer that inhibited the recognition of an abort discrete. The powered descent maneuver was initiated at about 108 hours. A ranging scale problem, which would have prevented acquisition of radar data until late in the descent, was corrected by cycling the circuit breaker off and on. Landing in the Fra Mauro highlands occurred at 108:15:09.3. The landing coordinates were 3 degrees 40 minutes 24 seconds south latitude and 17 degrees 27 minutes 55 seconds west longitude.

The command and service module, after undocking and separation, was placed in a circular orbit having an altitude of approximately 60 miles to photograph the proposed Descartes landing site, as well as perform landmark tracking and other tasks required for the accomplishment of lunar orbit experiments and photography. Communications between the command and service module and earth during this period were intermittent because of a problem with the high-gain antenna.

Preparations for the initial period of lunar exploration began about 2 hours after landing. A procedural problem with the lunar module communications delayed cabin depressurization about 50 minutes. The Commander egressed at about 113-3/4 hours and deployed the modular equipment stowage assembly as he descended the ladder, providing transmission of color television. The Lunar Module Pilot egressed a few minutes later. Subsequently, the S-band antenna was erected and activated, the Apollo lunar surface experiments package was deployed, and various documented lunar samples were taken during the extravehicular period which lasted about 4 3/4 hours. A modular equipment transporter, used on this mission for the first time, assisted the crew in carrying equipment and lunar samples.

Preparations for the second extravehicular period were begun following a 6 1/2-hour rest period. The goal of the second extravehicular period was to traverse to the area of Cone Crater. Although the crew experienced difficulties in navigating, they reached a point within approximately 50 feet of the rim of the crater. Thus, the objectives associated with reaching the vicinity of this crater and obtaining the desired samples were achieved. Various documented rock and soil samples were collected on the return traverse from Cone Crater, and, upon completing the traverse, the antenna on the lunar-experiment-package central station was realigned. The second extravehicular period lasted about 4-1/2 hours for a total extravehicular time of approximately 9-1/4 hours. About 96 pounds of lunar samples were collected during the two extravehicular periods.

The ascent stage lifted off at about 141-3/4 hours and the vehicle was inserted into a 51.7- by 8.5-mile orbit. A direct rendezvous was performed and the command-module-active docking operations were normal. However, during the final braking phase, the lunar module abort guidance system failed after the system was no longer required. Following crew transfer to the command module, the ascent stage was jettisoned and guided to impact approximately 36 miles west of the Apollo 14 landing site.

Transearth injection occurred during the 34th lunar revolution at about 148-1/2 hours. During transearth coast, one midcourse correction was made using the service module reaction control system. In addition, a special oxygen flow rate test was performed and a navigation exercise simulating a return to earth without ground control was conducted using only the guidance and navigation system. Inflight demonstrations of four types of processes under zero-gravity conditions were also performed and televised to earth.

Entry was normal and the command module landed in the Pacific Ocean at 216:01:58. The landing coordinates were 27 degrees 0 minutes 45 seconds south latitude and 172 degrees 39 minutes 30 seconds west longitude.

2.0 INTRODUCTION

The Apollo 14 mission was the 14th in a series using Apollo flight hardware and achieved the third lunar landing. The objectives of the mission were to investigate the lunar surface near a preselected point in the Fra Mauro formation, deploy and activate an Apollo lunar surface experiments package, further develop man's capability to work in the lunar environment, and obtain photographs of candidate exploration sites.

A complete analysis of all flight data is not possible within the time allowed for preparation of this report. Therefore, report supplements will be published for certain Apollo 14 systems analyses, as shown in appendix E. This appendix also lists the current status of all Apollo mission supplements, either published or in preparation. Other supplements will be published as necessary.

In this report, all actual times prior to earth landing are elapsed time from range zero, established as the integral second before lift-off. Range zero for this mission was 21:03:02 G.m.t., January 31, 1971. The clock onboard the spacecraft was changed at 54:53:36 by adding 40 minutes and 2.90 seconds; however, the times given in this report do not reflect this clock update. Had the clock update not been performed, indications of elapsed time in the crew's data file would have been in error by the amount of the delay in lift-off since the midcourse corrections were targeted to achieve the prelaunch-desired lunar orbit insertion time. Greenwich mean time is used for all times after earth landing. All references to mileage distance are in nautical miles.

3.0 LUNAR SURFACE EXPERIMENTS

The experiments discussed in this section consist of those associated with the Apollo lunar surface experiments package (a suprathermal ion detector, a cold cathode gage, a passive seismometer, an active seismometer, and a charged particle environment detector), as well as a laser ranging retro-reflector experiment, a lunar portable magnetometer experiment, a solar wind composition experiment, lunar geology, and soil mechanics. Descriptions of the purposes and equipment of experiments carried for the first time on previous missions are given in the reports of those missions, and the applicable reports are referenced where appropriate. A brief description of the experiment equipment used for the first time on Apollo 14 is given in appendix A.

Lunar surface scientific activities were performed generally as planned within the allotted time periods. Approximately 5 1/2 hours after landing, the crew egressed the lunar module for the first traverse of the lunar surface. During the first extravehicular activity period, which lasted 4 hours 47 minutes 50 seconds, the crew:

a. Deployed the modular equipment stowage assembly.

b. Deployed and operated the color television camera as required to televise crew activities in the vicinity of the lunar module.

c. Transferred a contingency sample to the lunar module.

d. Erected the United States flag and the solar wind composition foil.

e. Deployed and loaded the modular equipment transporter used to aid the astronauts in transporting equipment and samples.

f. Collected surface samples including two "small-football-size" specimens weighing approximately 4.4 and 5.5 pounds.

g. Photographed activities, panoramas and equipment.

h. Deployed the Apollo lunar surface experiments package for the continuing collection of lunar scientific data via radio link.

Following a planned rest period, the second extravehicular activity period began with preparations for an extended geological traverse. The duration of the second extravehicular activity period was 4 hours 34 minutes 41 seconds, covering a traverse of approximately 1.6 miles, during which the crew:

a. Obtained lunar portable magnetometer measurements at two sites along the traverse.

b. Collected documented, core tube, and trench-site samples.

c. Collected a "large-football-size" specimen weighing approximately 19 pounds.

d. Photographed the area covered, including panoramas and sample sites.

e. Retrieved the solar wind composition foil.

f. Adjusted the antenna on the Apollo lunar surface experiments package central station.

The evaluations discussed in this section are based on the data obtained during the first lunar day — largely on crew comments and real-time information. Certain equipment difficulties mentioned in this section are discussed in greater detail in section 14.4. More comprehensive results will be summarized in a separate science report to be published when the detailed analyses are complete (appendix E). The sites at which the various lunar surface activities were conducted are shown in the figure 3-1. The specific activities at each location are identified in table 3-I.

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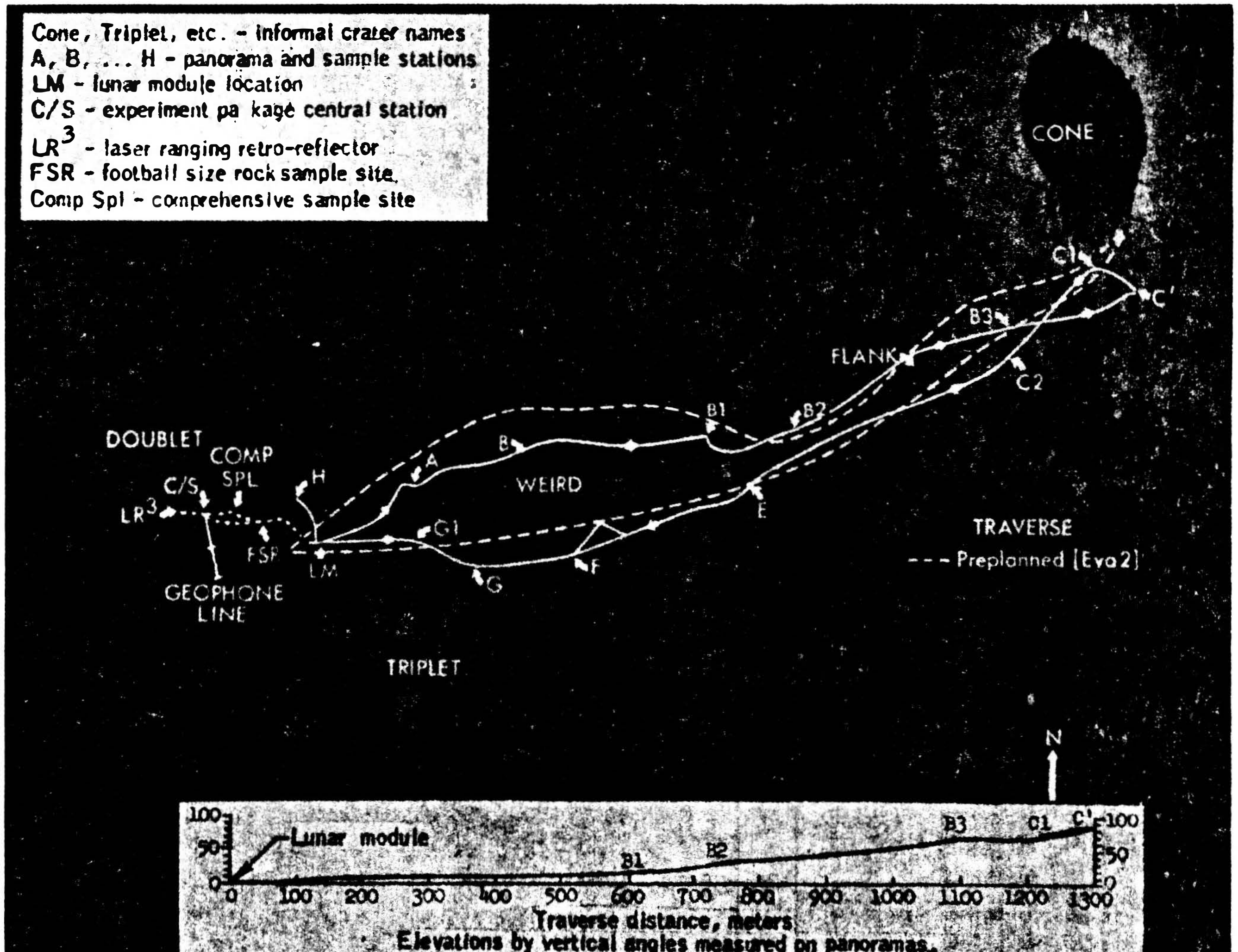

Figure 3-1.- Traverse for first and second extravehicular periods.

TABLE 3-I.- LUNAR SURFACE ACTIVITIES

Station	Activities
First extravehicular activity period	
Lunar module	Sampling and photography
Apollo lunar surface experiments package deployment site	Apollo lunar surface experiment activities and photography
Laser ranging retro-reflector site	Deployment of instrument and photography
Comprehensive sample site	Sampling and photography
Small-football-size rock site	Sampling and photography
Second extravehicular activity period	
A	Sampling, photography and first deployment of lunar portable magnetometer
B	Sampling and photography
B to B1	Sampling
B1	Photography
B2	Sampling and photography
B3	Photography
C'	Sampling, photography and second deployment of lunar portable magnetometer
C1	Sampling and photography
C2	Sampling and photography
C2 to E	Sampling
E	Sampling
F	Sampling and photography
G	Sampling and photography
G1	Sampling and photography
H	Sampling and photography

3.1 APOLLO LUNAR SURFACE EXPERIMENTS PACKAGE

The Apollo lunar surface experiments package was deployed with the central station positioned 600 feet west-northwest of the lunar module (fig. 3-2). No difficulties were experienced in off-loading the pallets or setting them up for the traverse other than an initial difficulty in latching the dome removal tool in the fuel cask dome. The crew installed the fuel capsule in the radioisotope thermoelectric generator and lock-on data were obtained with initial antenna alignment at 116 hours 48 minutes.

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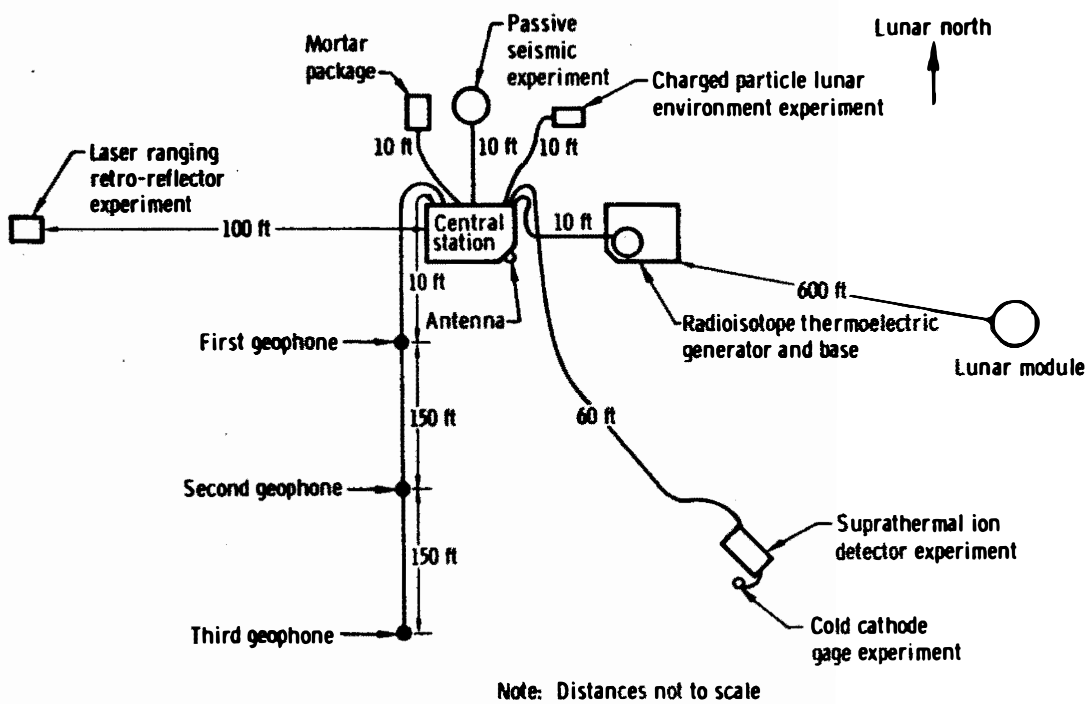

Figure 3-2.- Arrangement of the Apollo lunar surface experiments.

3.1.1 Central Station

Initial conditions of the central station (ref. 1) were normal. Power output of the radioisotope thermoelectric generator was 69.1 watts, and the central station thermal plate temperature averaged 73.8° F. A reserve power reading of 43.5 watts indicated that the basic power consumption was normal for Apollo lunar scientific experiment package startup. As the generator warmed up, the power output increased to 72.0 watts and has remained nearly constant at that level.

The transmitter signal strength at initial acquisition was lower than expected, and about 4 dB lower than that of the Apollo 12 experiment package. This was partially the result of acquisition occurring at the time of the worst-case condition of the relative earth-moon positions. In addition, lunar surface photography shows that the antenna was not fully seated in the gimbal interface socket (resulting in a misalignment with gimbal settings) and the gimbal pointing toward the earth was off the nominal pointing angle. Subsequent monitoring indicates that the signal strength obtained from the Apollo 14 unit is now equal to that of the Apollo 12 unit and that signal strength variation can be predicted based on the relative earth-moon positions.

The Apollo lunar scientific experiment package central station was commanded to the high-bit-rate mode at 116 hours 56 minutes for the active seismic experiment/thumper mode of operation, which continued until 117 hours 34 minutes. Using the high-bit-rate mode, only the active seismic experiment data and limited engineering data can be received from the central station. The other experiments were turned on following the active seismic experiment/thumper mode of operation.

During the deployment of the central station, the sunshield erected normally. However, the crew had to lift one side on three occasions because it was sagging. Lunar surface photography indicates that the sunshield had been bumped downward in a counterclockwise direction. However, the sagging condition has had no adverse effect on the central station thermal control system, and the central station has been operating within thermal limits.

The Apollo lunar scientific experiment package 12-hour timer pulses did not occur after initial central station turn-on. Subsequent tests verified that the mechanical section of the timer was not operating. The timer functions started to occur on February 11 and the timer provided 12-hour pulses thirteen times in succession before failing. Loss of the timer has no adverse effect of the Apollo lunar experiment package since all functions are being accomplished by ground command. This problem is discussed further in section 14.4.4.

The lunar dust detector of the central station is showing normal outputs from all three photoelectric cells. No changes in the outputs of these cells were observed during or after lunar module ascent, indicating that dust from the ascent engine exhaust did not settle on the central station.

3.1.2 Passive Seismic Experiment

The passive seismic experiment (ref. 2) was deployed 10 feet north of the central station (fig. 3-2). No difficulty was experienced in deploying the experiment other than the inability to make the ribbon cable lie flat on the surface under the thermal shroud skirt. All elements have operated as planned with the following exceptions.

a. The long-period vertical component seismometer is unstable in the normal mode (flat-response mode). (See section 14.4.6 for a discussion of this anomaly.) The problem was eliminated by removing the feedback filter and operating in the peaked-response mode. In this mode, the siesmometer has a resonant period of 2.2 seconds instead of the normal period of 15 seconds. Without the extended flat response, the low-frequency data is more difficult to extract. However, useful data are being obtained over the planned spectrum by data processing techniques.

b. The gimbal motor which levels the Y-axis long-period seismometer has not responded to commands on several occasions. In these cases, the reserve power status indicates that no power is being supplied to the motor. The power control circuit of the motor is considered to be the most likely cause of this problem. Response to commands has been achieved in all cases by repeating the motor drive command. (See section 14.4.5 for a more detailed discussion of this problem.)

3.1.3 Active Seismic Experiment

The active seismic experiment (appendix A, section A.4.1) was deployed during the first extravehicular period with the first geophone approximately 10 feet southwest of the central station and the geophone array extending in a southerly direction (figs. 3-2 and 3-3). The Apollo lunar scientific experiment package was commanded to the high-bit-rate mode for 28 minutes during the active seismic experiment/thumper mode of operation. Thumping operations began at geophone 3 (the furthest from the central station) and proceeded for 300 feet at 15-foot intervals toward geophone 1.

The attempts to fire the initiators resulted in 13 fired and 5 misfired. Three initiators were deliberately not fired. In some instances, two attempts were made to fire an initiator. (See section 14.4.1 for further discussion of this anomaly.)

A calibration pulse was sent prior to the last thumper firing verifying that all three geophones were operational. The mortar package, was deployed 10 feet north-northwest of the central station and aimed to fire four grenades on command from earth to distances of 500, 1000, 3000 and

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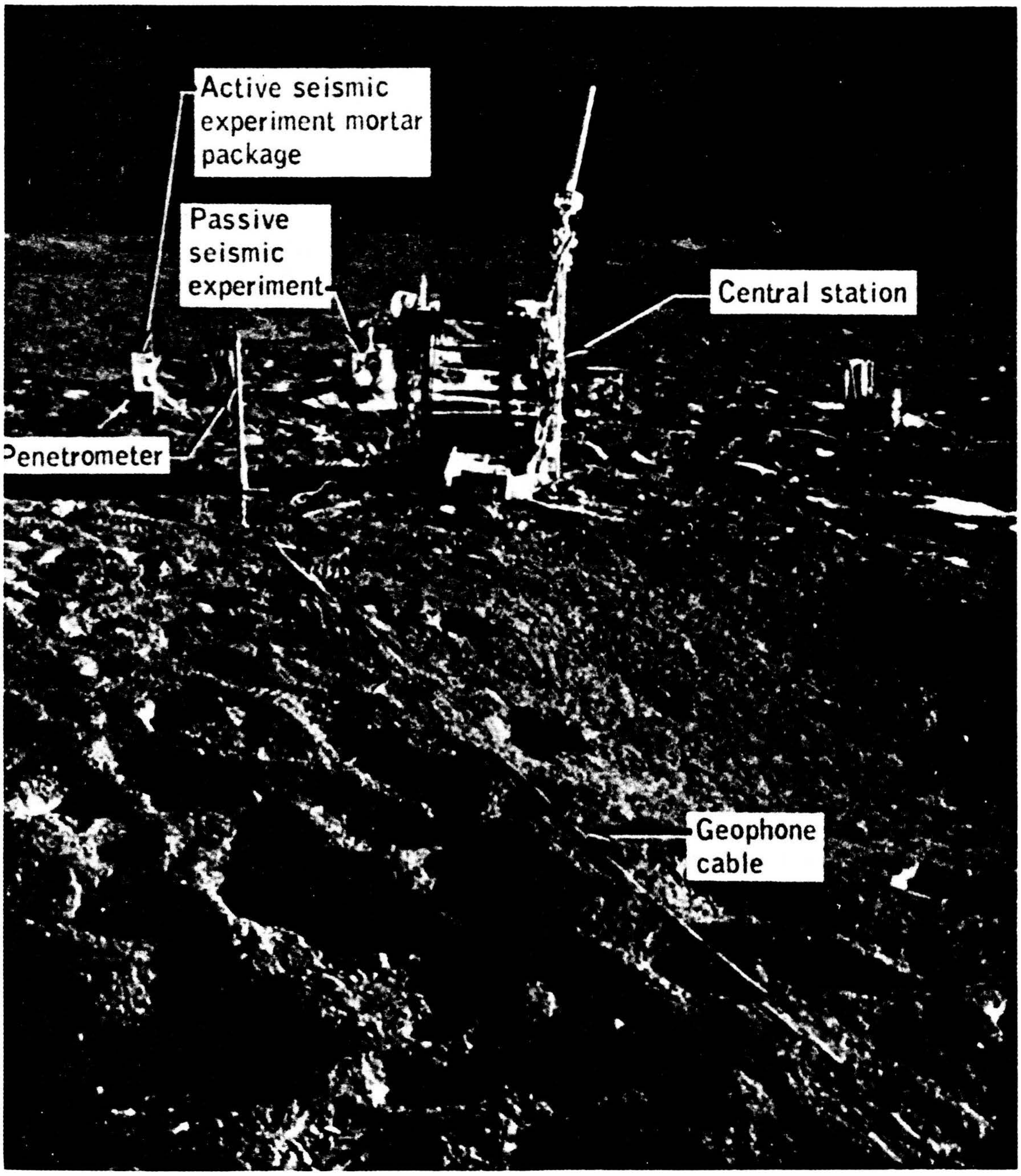

Figure 3-3.- Apòllo lunar surface experiment package components deployed on the lunar surface.

5000 feet in a northerly direction. Firing of the four mortars has not been scheduled. Postmission tests and analyses are being performed to establish the appropriate time and provisions for conducting this part of the experiment.

3.1.4 Suprathermal Ion Detector Experiment

The suprathermal ion detector experiment (ref. 2) was deployed southeast of the Apollo lunar surface experiments package central station (fig. 3-2). Noisy data were received at turn-on (section 14.4.2) but the data were satisfactory after seal break and dust cover removal. The experiment is returning good scientific data, with low background rates. Despite a large amount of lunar dust which adhered to one end of the package when it fell over several times during deployment (fig. 3-4), the temperatures throughout the lunar day and night remained within the range allowed for the instrument. Photographs show that the instrument is properly deployed and aligned.

3.1.5 Cold Cathode Gage Experiment

The cold cathode gage (ref. 2) was deployed 4 feet southeast of the suprathermal ion detector, aimed slightly southwest (figs. 3-2 and 3-4). The deployment was accomplished after several attempts in which the crewman experienced difficulty with the stiffness of the connecting cables while handling the suprathermal ion detector experiment, the cold cathode gage, and the ground screen at the same time.

The experiment was first turned on shortly before lunar module depressurization for the second extravehicular activity. Commands were sent to the instrument to turn on the high voltage and to open the cold cathode gage seal. The cold cathode gage data came off the initial full-scale indications much more rapidly than expected, indicating that the seal may have been open earlier than commanded.

Because a spontaneous change in the operational mode of the cold cathode gage and the suprathermal ion detector experiment occurred after about 1/2 hour of operation, the high voltages were switched off until after lunar sunset. When the high voltages were switched back on after lunar sunset, the response of the cold cathode gage went to the most sensitive range, indicative of the low ambient pressure. When the pressure rose at lunar sunrise as expected, the mode of operation was changed by a ground command to a less sensitive range, and the calibrate pulses appeared normal. The experiment is operating normally.

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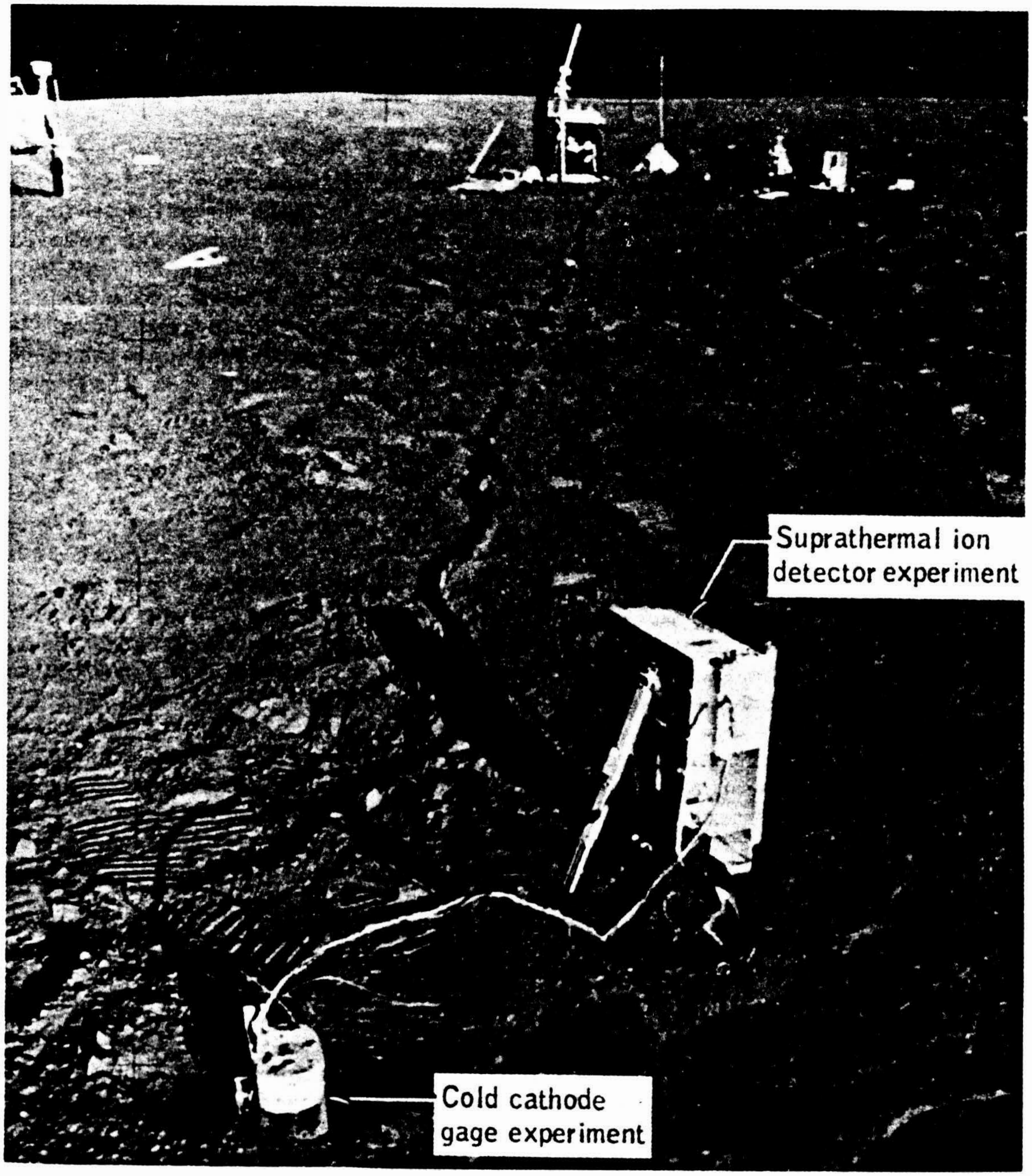

Figure 3-4.- Suprathermal ion detector experiment and cold cathode gage experiment deployed on the lunar surface.

3.1.6 Charged Particle Lunar Environment Experiment

The charged particle lunar environment experiment (ref. 3) instrument (figs. 3-2 and 3-5) was first commanded on at 117 hours 58 minutes during the first extravehicular activity for a 5-minute functional test and the instrument was normal. The complete instrument checkout showed that prelaunch and post-deployment counting rates agreed within 20 percent, with the exception of channel 6 in analyzer B. The counting rates on channel 6 were twice as high as the prelaunch values. The condition is attributed to the behavior of scattered electrons in the physical analyzers which behave quite differently in the effectively zero magnetic field of the moon compared with the 0.5-gauss magnetic field of the earth. The high counting rates on channel 6 do not detrimentally

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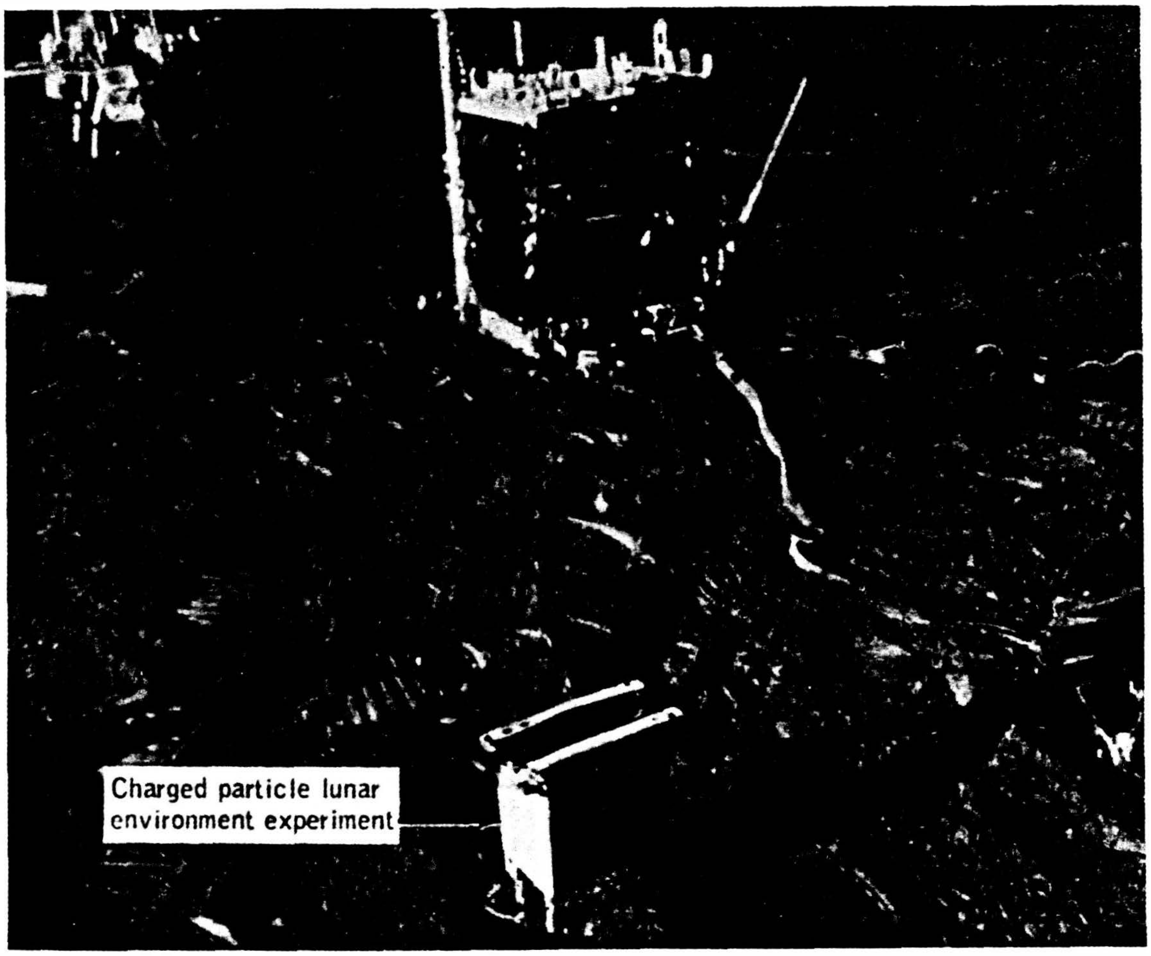

Figure 3-5.- Charged particle lunar environment experiment deployed on the lunar surface.

affect the science data. All command functions of the instrument were executed with the exception of the forced heater mode commands. Subsequent to the checkout, the experiment was commanded to standby.

After lunar module ascent, the charged particle lunar environment experiment was commanded on at 142 hours 7 minutes and the dust cover was removed about 15 hours and 20 minutes later. Operating temperatures are nominal. The maximum temperature during lunar day is 136° F and the minimum temperature during lunar night is minus 11° F. The instrument's operational heater cycled on automatically when the electronics temperature reached 32° F at lunar sunset, and was commanded on in the forced-on mode at 14° F, as planned.

The instrument, on one occasion, changed from the manual mode (at the plus 3500-volt step) to the automatic mode. The instrument was subsequently commanded back into the manual mode. There is no evidence in the data which would indicate the cause of the mode change.

3.2 LASER RANGING RETRO-REFLECTOR

The laser ranging retro-reflector (ref. 4) was deployed during the first extravehicular activity at a distance of approximately 100 feet west of the Apollo lunar scientific experiment package central station (figs. 3-2 and 3-6). Leveling and alignment were accomplished with no difficulty. The instrument was ranged on by the McDonald Observatory team prior to lunar module lift-off and a high-quality return signal was received. Ranging after lift-off, while not yet conclusive, indicates no serious degradation of the retro-reflector resulting from the effects of the ascent stage engine firing.

3.3 LUNAR PORTABLE MAGNETOMETER EXPERIMENT

The lunar portable magnetometer (appendix A, section A.4.2) was deployed at site A and near the rim of Cone Crater (fig. 3-1) during the second extravehicular activity period. The instrument operated nominally in all respects. The temperature of the experiment electronics package reached equilibrium, between 120° and 150° F. Meter readings, relayed over the voice link, indicated total fields of 102 ±10 gammas at site A and 41 ±10 gammas at Cone Crater. Vector component measurements of these readings were well within the dynamic range of the instrument. Leveling, orientation, and positioning were accomplished without difficulty; however, the experiment cable was difficult to rewind. This problem is discussed in greater detail in section 14.4.3.

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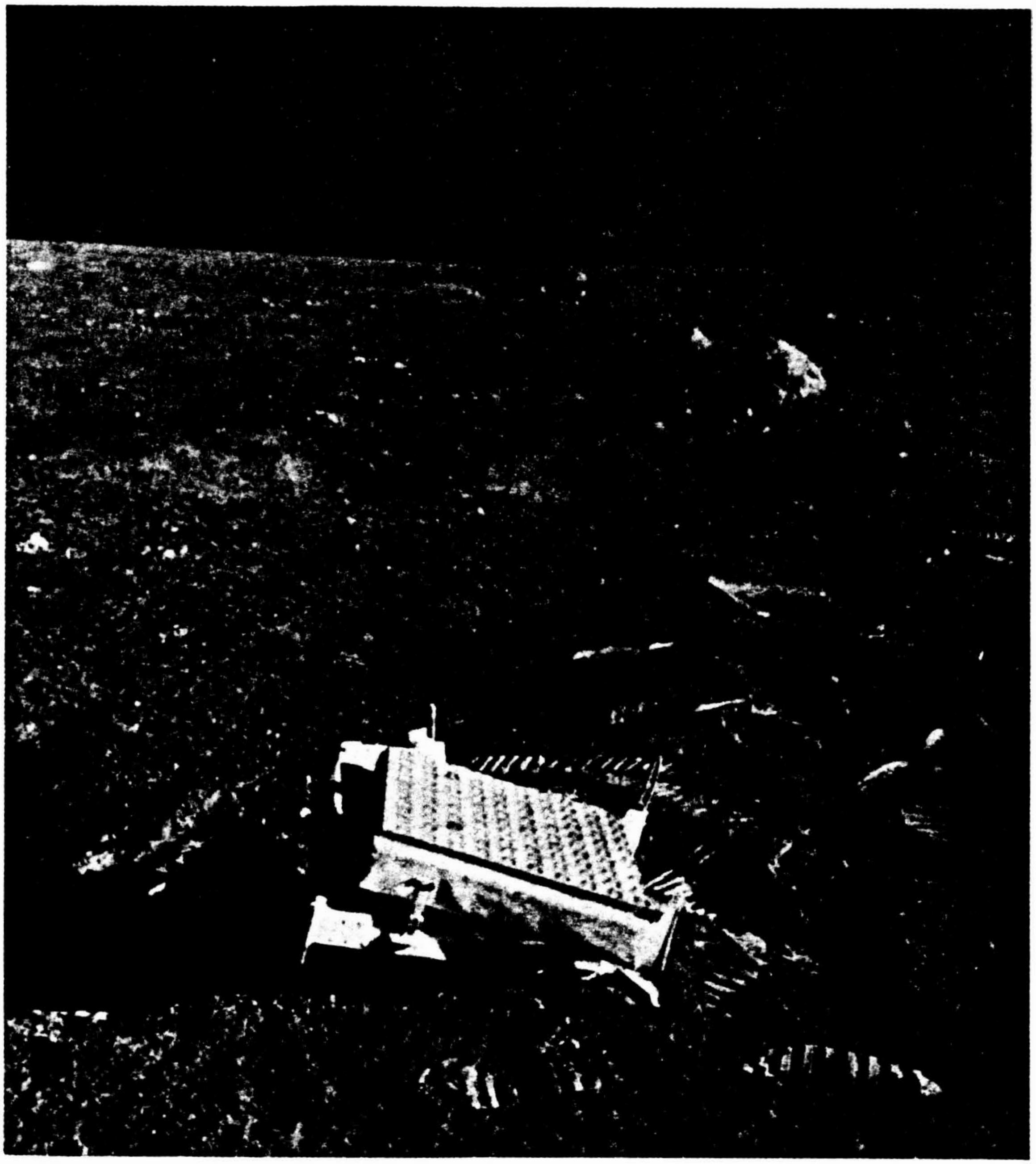

Figure 3-6.- Laser ranging retro-reflector experiment deployed on the lunar surface.

3.4 SOLAR WIND COMPOSITION EXPERIMENT

The solar wind composition experiment (ref. 4), a specially prepared aluminum foil rolled on a staff, was deployed during the first extravehicular period for a foil exposure time of approximately 21 hours. Deployment was accomplished with no difficulty; however, during retrieval, approximately half the foil rolled up mechanically and the remainder had to be rolled manually.

3.5 LUNAR GEOLOGY

The landing site in the Fra Mauro highlands is characterized by north-south trending linear ridges that are typically 160 to 360 feet in height and 6000 to 13 000 feet in width. The ridges and valleys are disfigured by craters ranging in size from very small up to several thousand feet in diameter.

The major objective of the geology survey was to collect, describe, and photograph materials of the Fra Mauro formation. The Fra Mauro formation is believed to be ejecta from the Imbrium Basin, which, in turn, is believed to have been created by a large impact. This material is probably best exposed in the vicinity of the landing site where it has been excavated from below the regolith by the impact that formed Cone Crater. The major part of the second extravehicular activity traverse, therefore, was designed to sample, describe, and photograph representative materials in the Cone Crater ejecta. Most of the returned rock samples consist of fragmental material. Photographs taken on the ejecta blanket of Cone Crater show various degrees of layering, sheeting, and foliation in the ejected boulders. A considerable variety in the nature of the returned fragmental rocks has been noted.

During the first extravehicular activity, the crew traversed a total distance of about 1700 feet. On their way back to the lunar module after deployment of the Apollo lunar scientific experiment package, the crew collected a comprehensive sample and two "football-size" rocks. The comprehensive sample area was photographed with locator shots to the Apollo lunar scientific experiment package and to the lunar module prior to sampling, and stereo photographs were taken of the two "football-size" rocks before they were removed from the surface. The location of the Apollo lunar scientific experiment package and the sampling and photographic sites for the first extravehicular activity are shown in figure 3-1.

The traverse during the second extravehicular activity covered a total distance of about 10 000 feet. The actual line of traverse is shown in figure 3-1. The crew reached a point within about 50 feet of the rim of Cone Crater. The crew was behind the timeline when they neared the rim of the crater; therefore, several of the preplanned sample and photographic stations along the route back to the lunar module were omitted. There was difficulty in navigating to several of the preplanned station points because of the undulations in the surface which prevented sighting of the smaller landmarks that were to be used.

The crew collected approximately 96 pounds of rock fragments and soil samples. Approximately 25 samples can be accurately located using photographs and the air-to-ground transcript, and the orientation of 12 to 15 on the lunar surface prior to their removal can be established.

Driving the core tubes with a rock hammer was somewhat difficult. The double and triple cores could not be driven their full length, and the material in the single core fell out upon removal of the core tube because of the granular nature of the material. Some sample material was recovered from the double and triple core tubes.

The only geologic equipment problems reported were that the contingency sample bag cracked when folded, and the vacuum seal protector on one of the special environmental sample containers came off when the container was opened.

3.6 LUNAR SOIL MECHANICS

Lunar surface erosion resulted from the descent engine exhaust as observed in previous lunar landings. Dust was first noted during descent at an altitude of 100 feet but did not hinder visibility during the final approach.

The lunar module footpad penetration on landing appears to have been greater than that observed on previous Apollo landings. Bootprint penetrations for the crew ranged from 1/2 to 3/4 inch on level ground in the vicinity of the lunar module to 4 inches on the rims of small craters. Lunar soil adhered extensively to the crewmen's clothing and equipment as in earlier Apollo missions. Tracks from the modular equipment transporter were 1/4 to 3/4 inch deep and were smooth.

The Apollo simple penetrometer (also used as the geophone cable anchor) was used for three penetration tests. In each case, the 26 1/2-inch-long penetrometer could be pushed to a depth of 16 to 19 inches with one hand and to the extension handle with both hands. No penetration interference attributable to rocks was encountered.

A soil mechanics trench was dug in the rim of a small crater near North Triplet Crater. Excavation was easy, but was terminated at a depth of 18 inches because the trench walls were collapsing. Three distinct layers were observed and sampled: (1) The surface material was dark brown and fine-grained. (2) The middle layer was thin and composed predominantly of glassy patches. (3) The lower layer was very light colored granular material.

3.7 MODULAR EQUIPMENT TRANSPORTER

The modular equipment transporter (described in appendix A, section A.2.1 and shown in fig. 3-7) was deployed at the beginning of the first extravehicular activity. Deployment was impeded by the thermal blanket which restrained the modular equipment transporter from rotating down from the bottom of the modular equipment stowage assembly. The crew released the transporter by pulling the upper pip-pins and allowing the transporter and thermal blanket to fall freely to the lunar surface. The thermal blanket was easily discarded and erection of the transporter went as planned. The tires had inflated as expected. Equipment was loaded on the transporter without difficulty. Two of the three pieces of Velcro which held the lunar maps on the transporter handles came off at the beginning of the first extravehicular activity. These pieces had been glued on a surface having a different finish than the one to which the Velcro adhered.

The modular equipment transporter stability was adequate during both traverses. Rotation in roll was felt by the crewman through the handle but was easily restrained by using a tighter grip if the rotation sensed was excessive. The jointed legs in the front of the transporter operated as expected in that they flexed when hit and would spring back to the vertical position readily. The smooth rubber tires threw no noticeable dust. No dust was noted on the wheel fenders or on top of the metal frame of the transporter.

The modular equipment transporter was carried by both crewmen at one point in the second extravehicular activity to reduce the effort required for one crewman to pull the vehicle. This was done for a short period of time because it was believed to be more effective when traveling over certain types of terrain.

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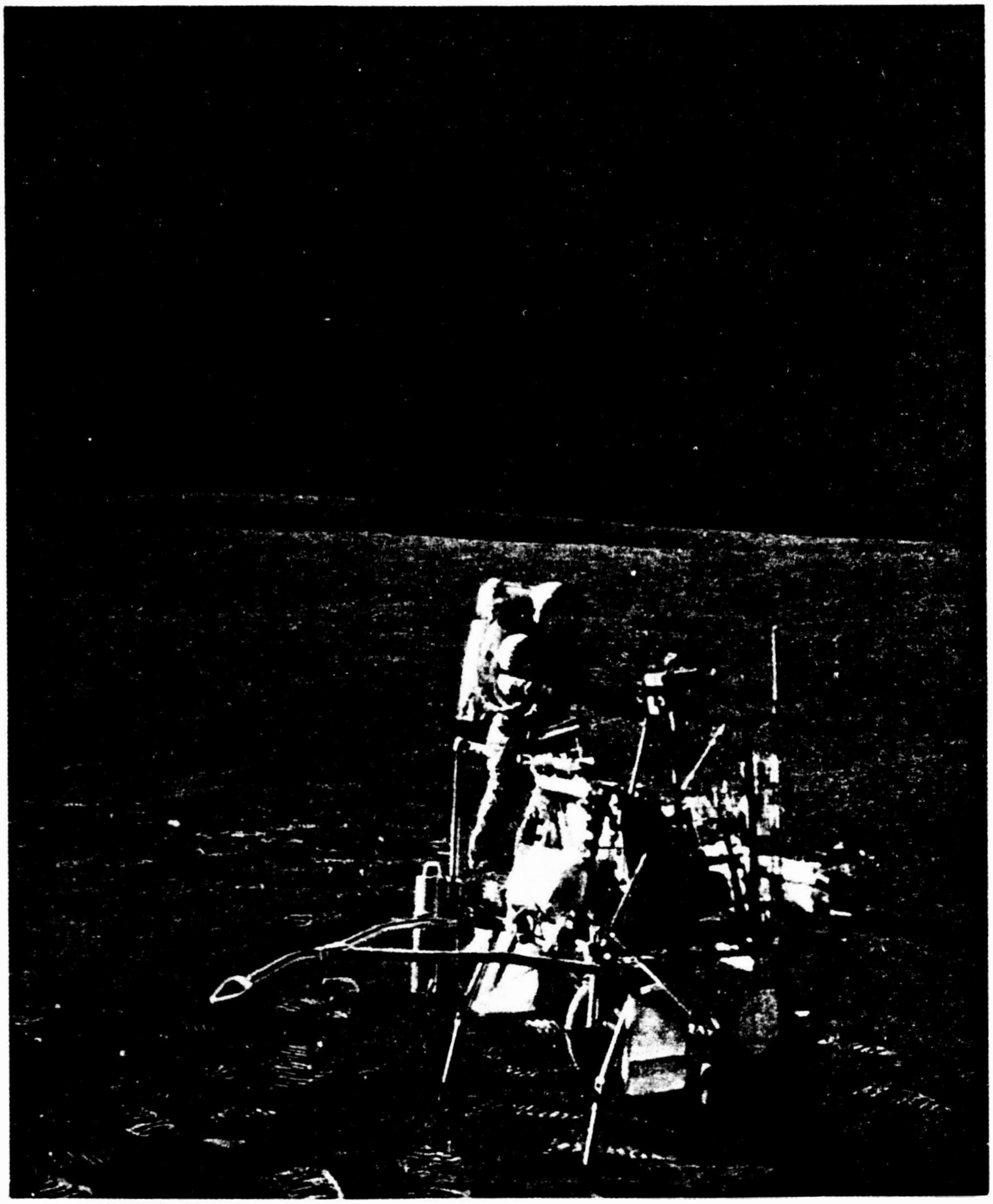

Figure 3-7.- Modular equipment transporter in use during the second extravehicular period.

3.8 APOLLO LANDING SITES

The Apollo 11 through 14 missions have placed a considerable amount of equipment on the lunar surface. Figure 3-8 shows the locations of all Apollo hardware that has been placed or impacted on the lunar surface.

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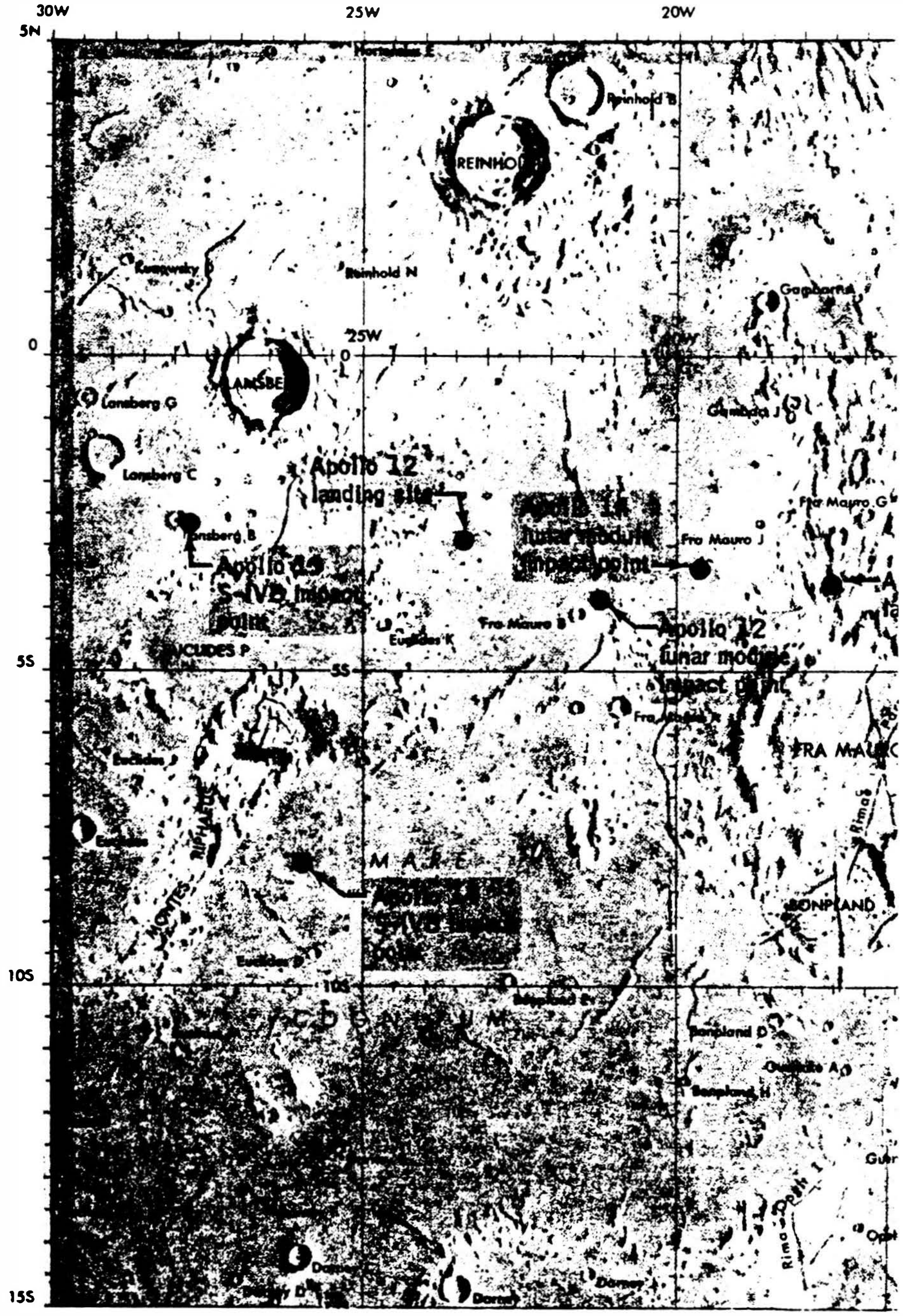

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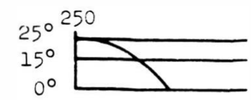

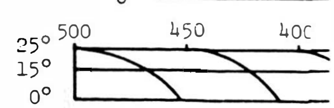

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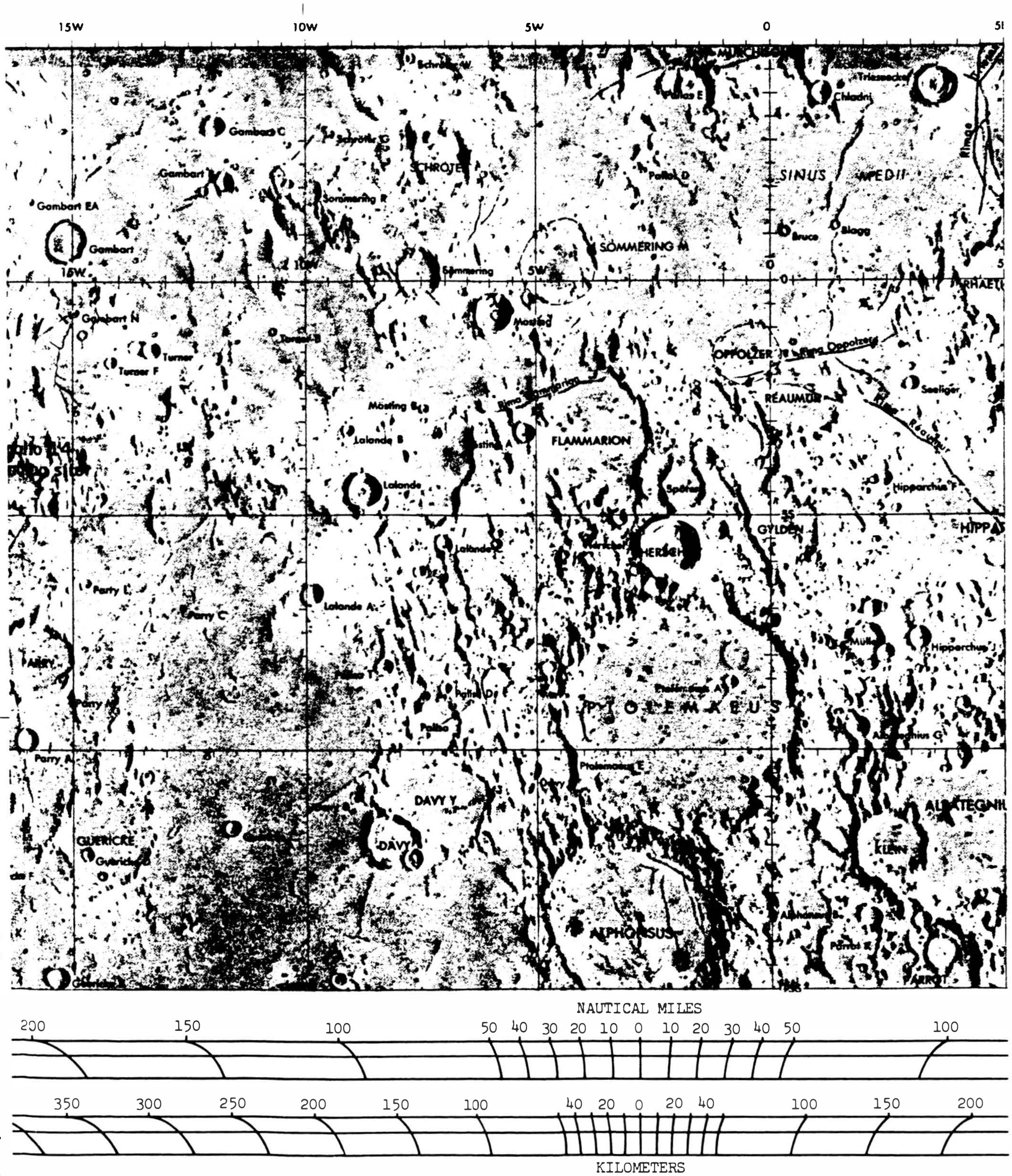
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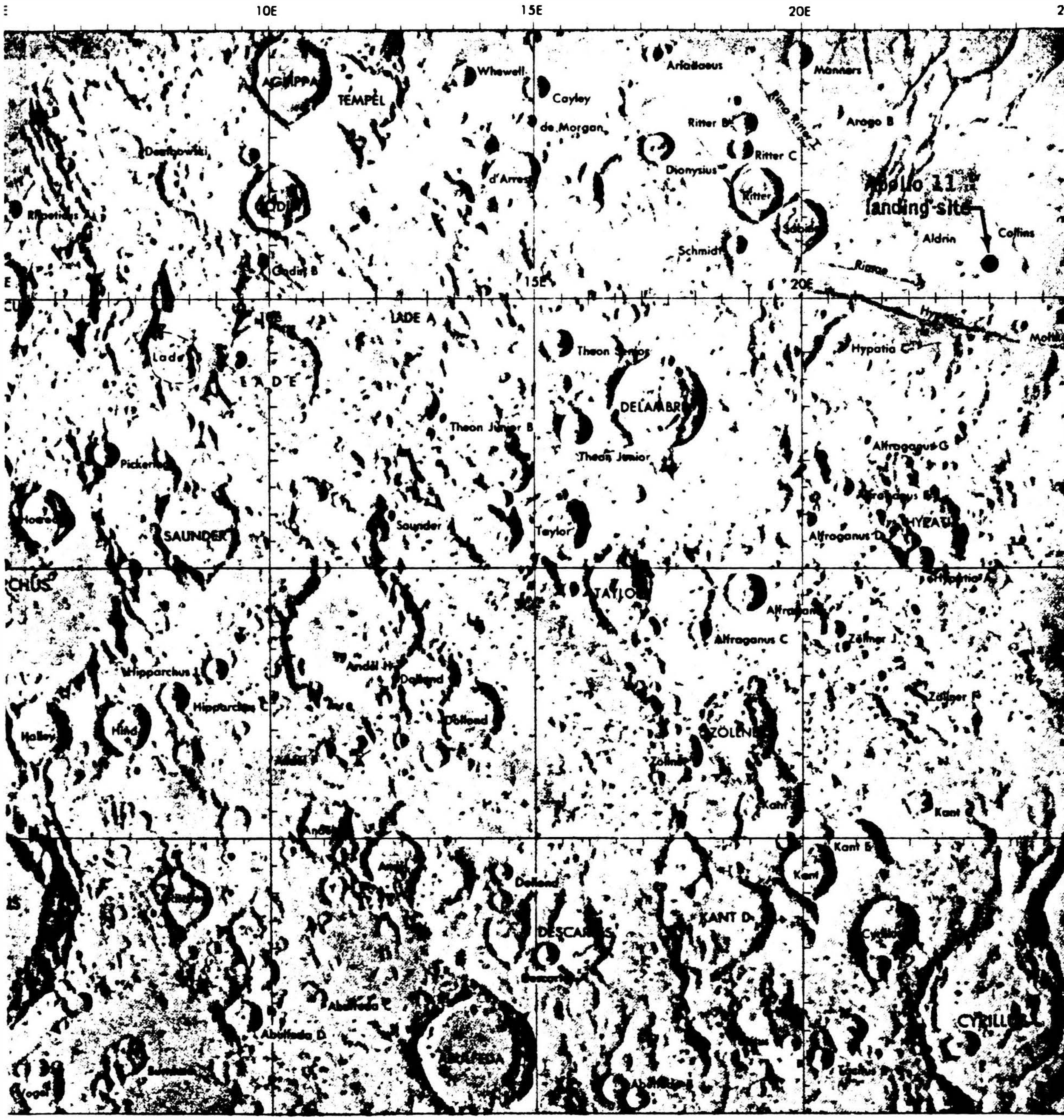
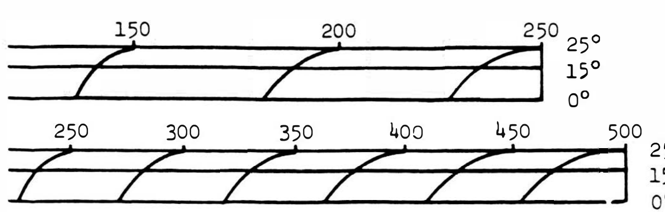
Sömmering
Gambart N
Mösting
Turner
Turner F
OPPOLZER
Rima Oppolzer
Seeliger
REAUMUR
Mösting B
FLAMMARION
Lalande B
Lalande
Hipparchus F
GYLDEN
Parry L
Lalande A
Parry C
Hipparchus
Parry M
PTOLEMAEUS
Palisa
Parry A
Ptolemaeus E
DAVY Y
GUERICKE
DAVY
KLEIN
ALPHONSUS
NAUTICAL MILES
200 150 100 50 40 30 20 10 0 10 20 30 40 50 100
350 300 250 200 150 100 40 20 0 20 40 100 150 200
KILOMETERS



Mercator Projection

EDITION 1, JULY 1969



PREPARED UNDER THE DIRECTION OF DEPARTMENT OF DEFENSE BY THE AERONAUTICAL CHART AND INFORMATION CENTER, UNITED STATES AIR FORCE FOR NATIONAL AERONAUTICS AND SPACE ADMINISTRATION.

Figure 3-8.- Apollo landing site and hardware locations on lunar surface.

4.0 LUNAR ORBITAL EXPERIMENTS

Four lunar orbital experiments were conducted on Apollo 14: the S-band transponder experiment, the downlink bistatic radar experiment, gegenschein/Moulton point photography from lunar orbit, and the Apollo window micrometeoroid experiment (a space exposure experiment not requiring crew participation). Detailed objectives associated with photography while in lunar orbit and during transearth flight are discussed in addition to the aforementioned experiments. The evaluations of the lunar orbital experiments given here are based on preliminary data. Final results will be published in a separate science report (appendix E) when the data have been completely analyzed.

4.1 S-BAND TRANSPONDER

The S-band transponder experiment was designed to detect variations in the lunar gravitational field caused by mass concentrations and deficiencies, and establish gravitational profiles of the spacecraft ground tracks. This will be accomplished by analysis of data obtained from S-band Doppler tracking of the command and service module and lunar module using the normal spacecraft S-band systems.

There were some difficulties during the prime data collection period (revolutions 3 through 14). Two-way telemetry lock was lost many times during revolutions 6 and 9 because of the high-gain antenna problem, making the data for those revolutions essentially useless. At other times maneuvers, orientations, and other operations interfered with the data. However, sufficient data were received to permit successful completion of the experiment objectives. Preliminary indications are that the mass concentrations in Nectaris will be better described and the distribution of gravitational forces associated with the Fra Mauro formation will be better known. The data will also permit other features to be evaluated.

4.2 BISTATIC RADAR

The objectives of the bistatic radar experiment were to obtain data on lunar surface roughness and the depth of the regolith to a limit of 30 to 60 feet. The experiment was also designed to determine the lunar surface Brewster angle, which is a function of the bulk dielectric constant of the lunar material. No spacecraft equipment other than the normal spacecraft systems was required for the experiment. The experiment data consists of records of VHF and S-band transmissions from the command

and service module during the frontside pass on revolution 25, with ground-based detection of both the direct carrier signals and the signals reflected from the lunar surface. Both the VHF and S-band equipment performed as required during revolution 25. The returned signals of both frequencies were of predicted strength. Strong radar echoes were received throughout the pass and frequency, phase, polarization and amplitude were recorded. Sufficient data were collected to determine, in part, the Brewster angle.

4.3 GEGENSCHEIN/MOULTON POINT PHOTOGRAPHY FROM LUNAR ORBIT

The experiment required three sets of photographs to be taken to help differentiate between two theoretical explanations of the gegenschein (fig. 4-1). Each set consisted of two 20-second exposures and

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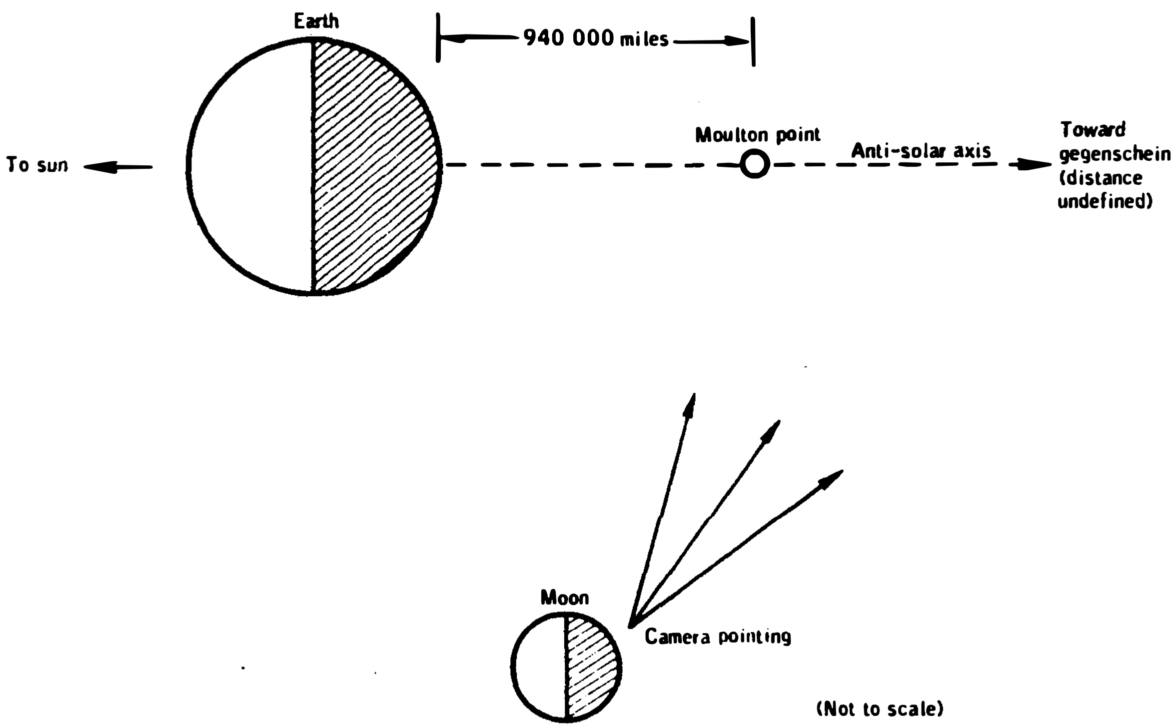

Figure 4-1.- Camera aiming directions for gegenschein/ Moulton point photography.

one 5-second exposure taken in rapid succession. One set was obtained of the earth orbit stability point in the earth-sun system (Moulton point) to test the theory that the gegenschein is light reflected from a concentration of particles captured about the Moulton point. Two additional sets were taken to test another theory that the glow is light reflected from interplanetary dust that is seen in the anti-solar direction. In this theory, the brightening in the anti-solar direction is thought to be due to higher reflectivity of particles exactly opposite the sun. For an observer on earth, the anti-solar direction coincides with the direction of the Moulton point and the observer is unable to distinguish between the theories. From the moon the observer is displaced from the anti-solar direction by approximately 15 degrees, and therefore, can distinguish between the two possible sources.

The 16-mm data acquisition camera was used with an 18-mm focal length lens. The camera was bracket-mounted in the right-hand rendezvous window with a right angle mirror assembly attached ahead of the lens and a remote control electrical cable attached to the camera so that the Command Module Pilot could actuate the camera from the lower equipment bay. The flight film had special, low-light-level calibration exposures added to it prior to and after the flight which will permit photometric measurements of the phenomena by means of photographic densitometer and isodensitrace readings during data reduction. The investigators also obtained ground photography of the phenomena using identical equipment and film prior to the time of Apollo 14 data collection.

The experiment was accomplished during the 15th revolution of the moon. The aiming and filming were excellent and the experiment has demonstrated that long exposures are practicable.

4.4 APOLLO WINDOW METEOROID EXPERIMENT

The objective of this experiment is to determine the meteoroid cratering flux for particles responsible for the degradation of glass surfaces exposed to the space environment. The Apollo command module windows are used as meteoroid detectors. Prior to flight, the windows are scanned at 20× to determine the general background of chips, scratches and other defects. During postlfight investigations, the windows will again be scanned at 20× to map all visible defects. The points of interest will then be magnified up to 765× for further examination. The Apollo 12 and 13 side windows and hatch windows were examined following those flights and the results were compared with preflight scans. No meteoroid impacts larger than 50 microns in diameter were detected on the Apollo 12 windows although there was an increase in the number of chips and other low-speed surface effects. The Apollo 13 left-hand-side window had a suspected meteoroid impact 500 microns in diameter.

4.5 DIM-LIGHT PHOTOGRAPHY

Low-brightness astronomical light sources were photographed using the 16-mm data acquisition camera with the 18-mm lens. The sources included the zodiacal light, the galactic light, the lunar libration region (L_4) and the dark side of the earth.

All star fields have been readily identified and camera pointing appears to have been within one degree of the desired aiming points with less than one-third of a degree of image motion for fixed positions. This is well within the limits requested prior to flight, and it confirms that longer exposures, which had been originally desired, will be possible for studies such as these on future Apollo missions. The zodiacal light is apparent to the unaided eye on at least half of the appropriate frames. The galactic light survey and lunar libration frames are faint and will require careful work. Earth-darkside frames of lightning patterns, earth-darkside photography during transearth coast, and S-IVB photographs were overexposed and are unusable.

4.6 COMMAND AND SERVICE MODULE ORBITAL SCIENCE PHOTOGRAPHY

This photography consisted of general coverage to provide a basis for site selection for further photography, interpretation of lunar surface features and their evolution, and identification of specific areas and features for study. The Apollo lunar missions have in the past obtained photographs of these areas as targets-of-opportunity or in support of specific objectives.

The Apollo 13 S-IVB impact area was given highest priority in orbital science photography. The target was successfully acquired on revolution 34 using the Hasselblad camera with the 500-mm lens, and the crew optical alignment sight to compensate for the spacecraft's motion. Second priority was given to the lunar module landing target which was obtained with the lunar topographic camera on revolution 14. However, the camera malfunctioned and subsequent photography with this camera was deleted (section 14.3.1).

A total of eight photographic targets was planned for hand-held photography using color film; three were to be taken with the 500-mm lens (a total of 35 lunar degrees), and five with the 250-mm lens (a total of 130 lunar degrees). The 500-mm targets were successfully acquired. Three of the five 250-mm targets were deleted in real-time for operational reasons (60 lunar degrees), and two were successfully acquired (70 lunar degrees). A total of 65 percent of off-track photography has been successfully acquired.

The earthshine target was successfully acquired using both the Hasselblad data camera with the 80-mm lens and the 16-mm data acquisition camera with the 18-mm lens.

4.7 PHOTOGRAPHS OF A CANDIDATE EXPLORATION SITE

High-resolution photographs of potential landing sites are required for touchdown hazard evaluation and propellant budget definition. They also provide data for crew training and onboard navigational data. The photographs on this mission were to be taken with the lunar topographic camera on revolution 4 (low orbit), and 27 and 28 (high orbits). During revolution 4, malfunction of the lunar topographic camera was noted by the Command Module Pilot. On revolutions 27, 28, and 30, the 70-mm Hasselbald camera with the 500-mm lens (lunar topographic camera backup system) was used to obtain the required photography. About 40 frames were obtained of the Descartes region on each revolution using the crew optical alignment sight to compensate for image motion. The three targets were successfully acquired.

To support the photography, a stereo strip was taken with the Hasselblad data camera with the 80-mm lens from terminator-to-terminator including the crew optical alignment sight maneuver for camera calibration.

4.8 VISIBILITY AT HIGH SUN ANGLES

This photography was accomplished to obtain observational data in the lunar environment for evaluating the ability of the crew to identify features under viewing and lighting conditions similar to those that would be encountered during descent for a T plus 24 hour launch. The results will have a bearing on decisions to land at higher sun angles, which, in turn, could ease launch and flight constraints. Photography of the lunar surface in support of this detailed objective was obtained using the Hasselblad data camera and the 80-mm lens. This was done for three targets, two on the moon's far side and one on its near side.

4.9 TRANSEARTH LUNAR PHOTOGRAPHY

Photographs were taken of the visible disc of the moon after transearth injection to provide changes in perspective geometry, primarily in latitude. The photographs will be used to relate the positions of lunar features at higher latitudes to features whose positions are known through landmark tracking and existing orbital stereo strips. The photography was successful using the Hasselblad data camera with the 80-mm lens and black-and-white film. Additional coverage with the 70-mm Hasselblad camera and the 250-mm lens using color film was also obtained.

5.0 INFLIGHT DEMONSTRATIONS

Inflight demonstrations were conducted to evaluate the behavior of physical processes of interest under the near-weightless conditions of space. Four categories of processes were demonstrated, and segments of the demonstrations were televised over a 30-minute period during transearth flight beginning at approximately 172 hours. Final results of all four demonstrations will be published in a supplemental report after analysis of data has been completed. (See appendix E.)

5.1 ELECTROPHORETIC SEPARATION

Most organic molecules, when placed in slightly acid or alkaline water solutions, will move through them if an electric field is applied. This effect is known as electrophoresis. Molecules of different substances move at different speeds; thus, some molecules will outrun others as they move from one end of a tube of solution toward the other. This process might be exploited to prepare pure samples of organic materials for applications in medicine and biological research if problems due to sample sedimentation and sample mixing by convection can be overcome.

A small fluid electrophoresis demonstration apparatus (fig. 5-1) was used to demonstrate the quality of the separations obtained with three sample mixtures having widely different molecular weights. They were: (1) a mixture of red and blue organic dyes, (2) human hemoglobin, and (3) DNA (the molecules that carry genetic codes) from salmon sperm.

Postmission review of the filmed data reveals that the red and blue organic dyes separated as expected; however, separation of the hemoglobin and DNA cannot be detected. Postflight examination of the apparatus indicates that the samples were not released effectively to permit good separation, causing the dyes to streak. However, the fact that the dyes separated supports the principle of electrophoretic separation and shows that sedimentation and convection effects are effectively suppressed in the space environment. The hemoglobin and DNA samples did not separate because they contained bacteria that consumed the organic molecules prior to activation of the apparatus.

5.2 LIQUID TRANSFER

The liquid transfer demonstration (fig. 5-2) was designed to evaluate the use of tank baffles in transferring a liquid from one tank to

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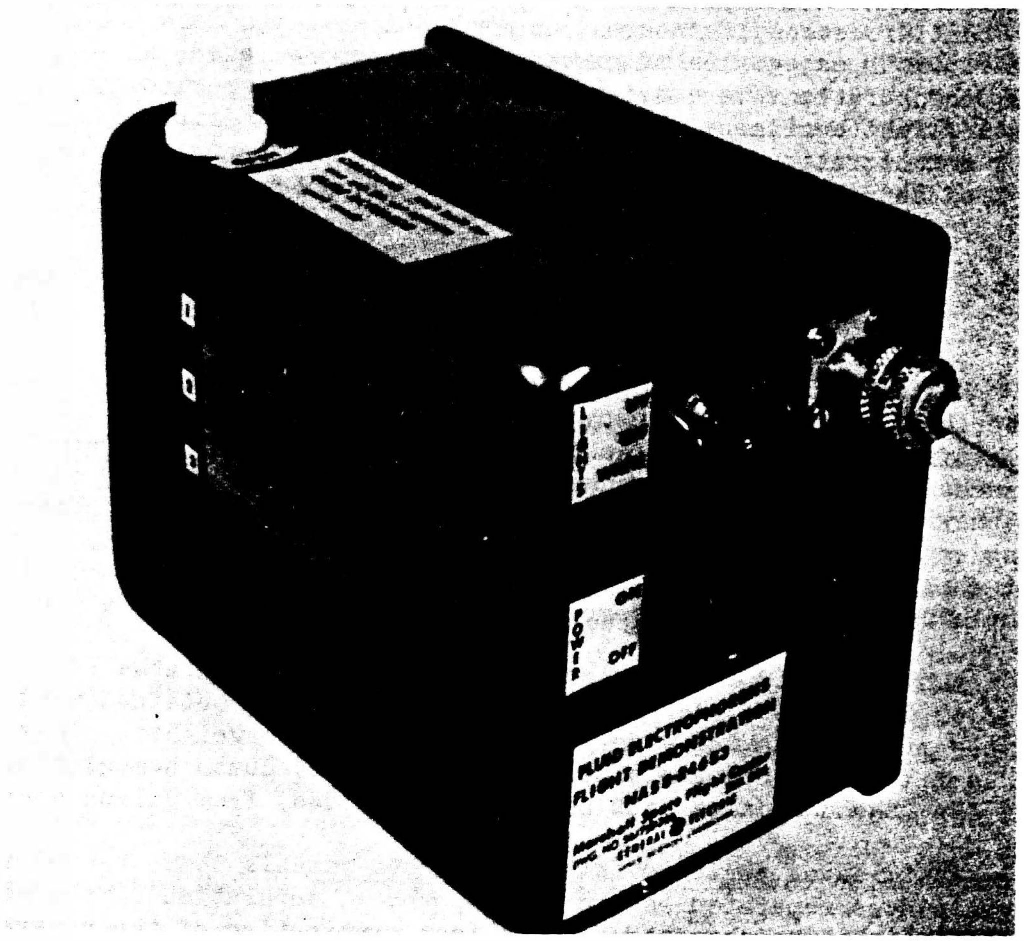

Figure 5-1.- Electrophoresis demonstration unit.

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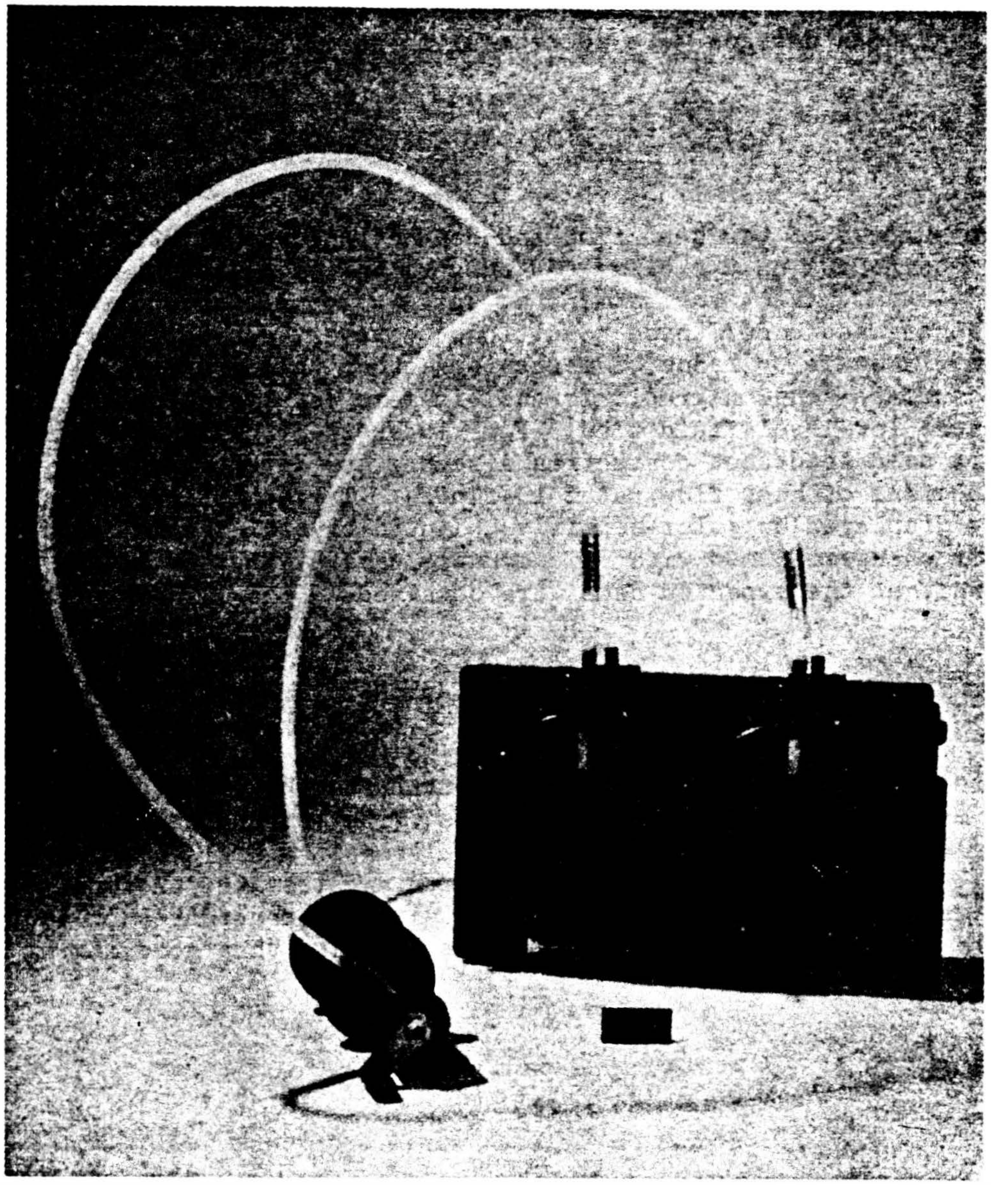

Figure 5-2.- Liquid transfer demonstration unit.

another under near-zero-gravity conditions. The demonstration was conducted using two sets of tanks, one set containing baffles and the other without baffles. Transfer of liquid between the unbaffled tanks was unsuccessful, as expected. Transfer between the baffled tanks demonstrated the effectiveness of two different baffle designs. Photographic data indicate that both designs were successful in permitting liquid transfer.

5.3 HEAT FLOW AND CONVECTION

The purpose of the heat flow and convection demonstration (fig. 5-3) was to obtain data on the types and amounts of convection that can occur in the near-weightless environment of space. Normal convective flow is almost suppressed under these conditions; however, convective fluid flow can occur in space by means of mechanisms other than gravity. For instance, surface tension gradients and, in some cases, residual accelerations cause low-level fluid flow. Four independent cells of special design were used to detect convection directly, or detect convective effects by measurement of heat flow rates in fluids. The heat flow rates were visually displayed by color-sensitive, liquid crystal thermal strips and the color changes filmed with a 16-mm data camera. Review of the film has shown that the expected data were obtained.

5.4 COMPOSITE CASTING

This demonstration was designed to evaluate the effect of near-zero-gravity on the preparation of cast metals, fiber-strengthened materials, and single crystals. Specimens were processed in a small heating chamber (fig. 5-4) and returned for examination and testing. A total of 11 specimens was processed. No problems with the procedures or equipment were noted. An x-ray of the samples verified that good mixing occurred.

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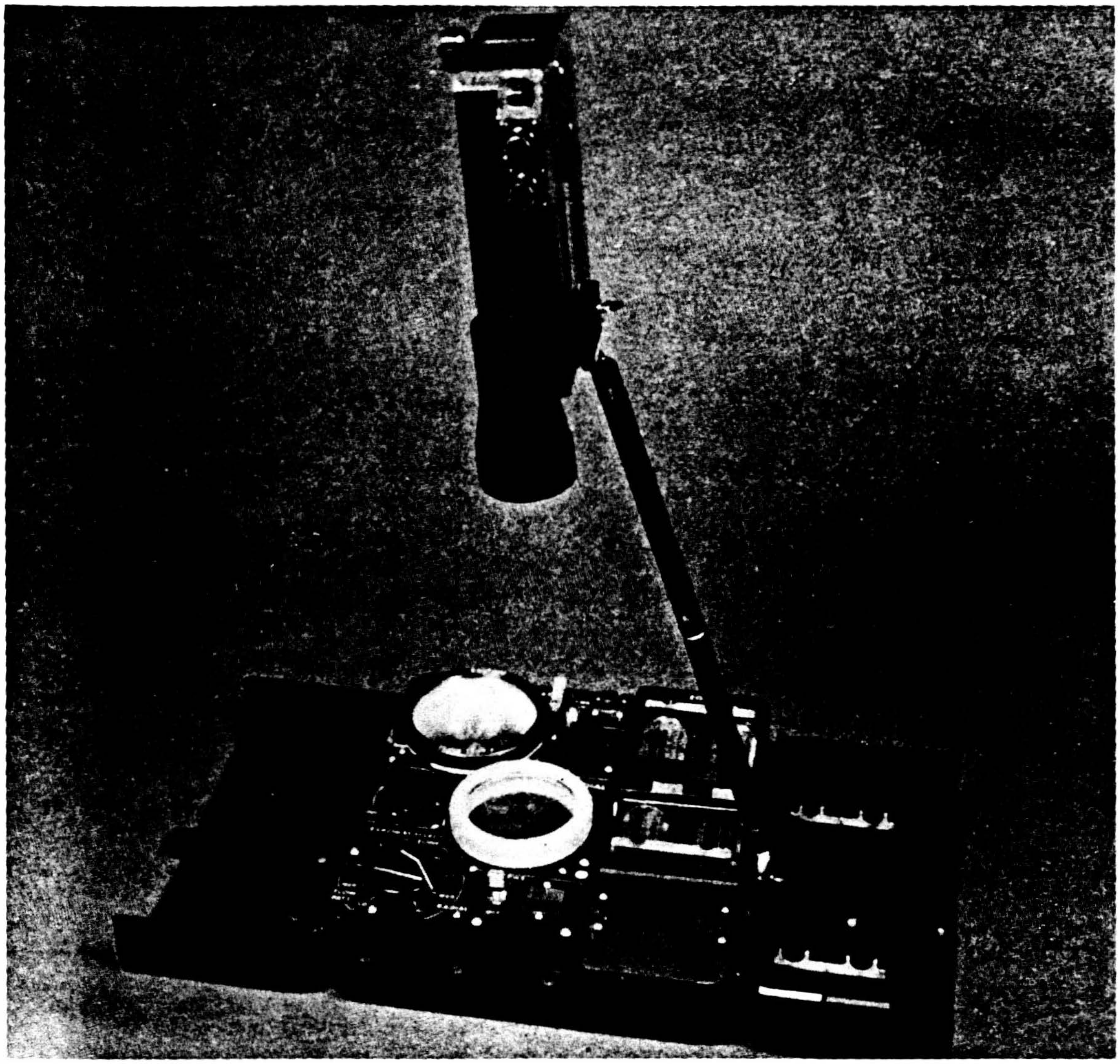

Figure 5-3.- Heat flow and convection demonstration unit.

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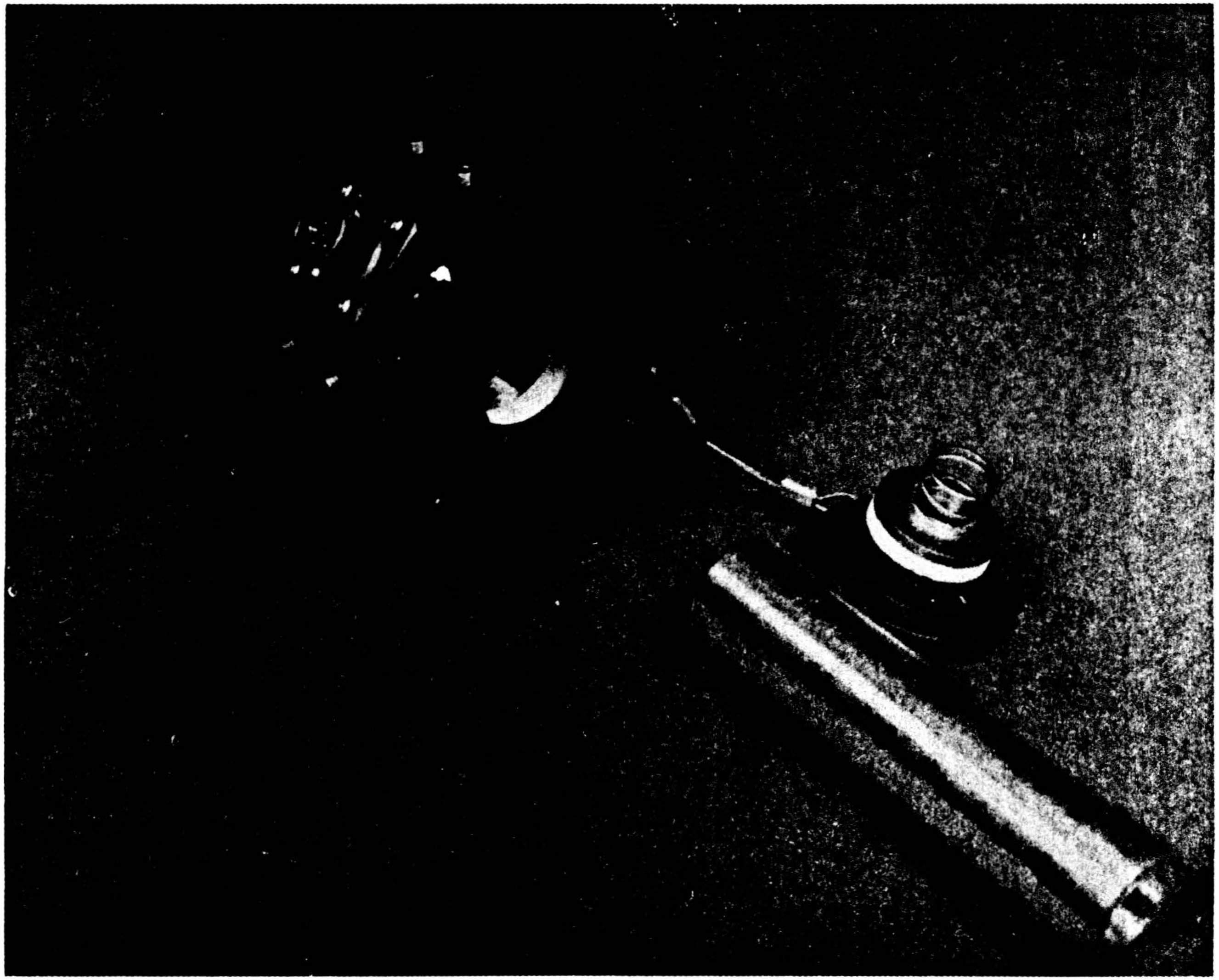

Figure 5-4.- Composite casting demonstration unit.

6.0 TRAJECTORY

The general trajectory profile of this mission was similar to that of previous lunar missions except for a few innovations and refinements in some of the maneuvers. These changes were: (a) The service propulsion system was used to perform the descent orbit insertion maneuver placing the command and service modules in the low-perilune orbit (9.1 miles). (b) A direct rendezvous was performed using the ascent propulsion system to perform the terminal phase initiation maneuver. Tables 6-I and 6-II give the times of major flight events and definitions of the events; tables 6-III and 6-IV contain trajectory parameter information; and table 6-V is a summary of maneuver data.

6.1 LAUNCH AND TRANSLUNAR TRAJECTORIES

The launch trajectory is reported in reference 5. The S-IVB was targeted for the translunar injection maneuver to achieve a 2022-mile pericynthion free-return trajectory. The command and service module/lunar module trajectory was altered 28 hours later by the first midcourse correction which placed the combined spacecraft on a hybrid trajectory with a pericynthion of 67.0 miles. A second midcourse correction, 46 hours later, lowered the pericynthion to 60.7 miles.

After spacecraft separation, the S-IVB performed a programmed propellant dump and two attitude maneuvers that directed the vehicle to a lunar impact. The impact coordinates were 8 degrees 05 minutes 35 seconds south latitude and 26 degrees 01 minute 23 seconds west longitude; 156 miles from the prelaunch target point but within the nominal impact zone.

6.2 LUNAR ORBIT

6.2.1 Orbital Trajectory

The service propulsion system was used to perform the lunar orbit insertion maneuver. The orbit achieved had an apocynthion of 169 miles and a pericynthion of 58.1 miles. After two lunar revolutions, the service propulsion system was again used, this time to perform the descent orbit insertion maneuver which placed the combined spacecraft in an orbit with a pericynthion of 9.1 miles. On previous missions, the lunar module descent propulsion system was used to perform this maneuver. The use of the service propulsion system allows the lunar module to maintain a

TABLE 6-I.- SEQUENCE OF EVENTS[a]

	Elapsed time, hr:min:sec
Range zero - 21:03:02 G.m.t., January 31, 1971	
Lift-off - 21:03:02.6 G.m.t., January 31, 1971	
Translunar injection maneuver, Firing time = 350.8 sec	02:28:32
Translunar injection	02:34:32
S-IVB/command module separation	03:02:29
Translunar docking	04:56:56
Spacecraft ejection	05:47:14
First midcourse correction, Firing time = 10.1 sec	30:36:08
Second midcourse correction, Firing time = 0.65 sec	76:58:12
Lunar orbit insertion, Firing time = 370.8 sec	81:56:41
S-IVB lunar impact	82:37:52
Descent orbit insertion, Firing time = 20.8 sec	86:10:53
Lunar module undocking and separation	103:47:42
Circularization maneuver, Firing time = 4 sec	105:11:46
Powered descent initiation, Firing time = 764.6 sec	108:02:27
Lunar landing	108:15:09
Start first extravehicular activity	113:39:11
First data from Apollo lunar surface experiment package	116:47:58
Plane change, Firing time = 18.5 sec	117:29:33
Complete first extravehicular activity	118:27:01
Start second extravehicular activity	131:08:13
End second extravehicular activity	135:42:54
Lunar lift-off, Firing time = 432.1 sec	141:45:40
Vernier adjustment maneuver, Firing time = 12.1 sec	141:56:49
Terminal phase initiation	142:30:51
Terminal phase finalization	143:13:29
Docking	143:32:51
Lunar module jettison	145:44:58
Separation maneuver	145:49:43
Lunar module deorbit maneuver, Firing time = 76.2 sec	147:14:17
Lunar module lunar impact	147:42:23
Transearth injection, Firing time = 149.2 sec	148:36:02
Third midcourse correction, Firing time = 3.0 sec	165:34:57
Command module/service module separation	215:32:42
Entry interface	215:47:45
Begin blackout	215:48:02
End blackout	215:51:19
Drogue deployment	215:56:08
Landing	216:01:58

[a]See table 6-II for event definitions.

TABLE 6-II.- DEFINITION OF EVENT TIMES

Event	Definition
Range zero	Final integral second before lift-off
Lift-off	Instrumentation unit umbilical disconnect
Translunar injection maneuver	Start tank discharge valve opening, allowing fuel to be pumped to the S-IVB engine
S-IVB/command module separation, translunar docking, spacecraft ejection, lunar module undocking and separation, docking, and command module landing	The time of the event based on analysis of spacecraft rate and accelerometer data
Command and service module and lunar module computer-controlled maneuvers	The time the computer commands the engine on and off
Command and service module and lunar module non-computer-controlled maneuvers	Engine ignition as indicated by the appropriate engine bilevel telemetry measurement
S-IVB lunar impact	Loss of S-band transponder signal
Lunar module descent engine cutoff time	Engine cutoff established by the beginning of drop in thrust chamber pressure
Lunar module impact	The time the final data point is transmitted from the vehicle telemetry system
Lunar landing	First contact of a lunar module landing pad with the lunar surface as derived from analysis of spacecraft rate data
Beginning of extravehicular activity	The time cabin pressure reaches 3 psia during depressurization
End of extravehicular activity	The time cabin pressure reaches 3 psia during repressurization
Apollo lunar surface experiment package first data	Receipt of first data considered to be valid from the Apollo lunar surface experiment package telemetry system
Command module/service module separation	Separation indicated by command module/service module separation relays A and B via the telemetry system
Entry interface	The time the command module reaches 400 000 feet geodetic altitude as indicated by the best estimate of the trajectory
Begin and end blackout	S-band communication loss due to air ionization during entry
Drogue deployment	Deployment indicated by drogue deploy relays A and B via the telemetry system
Earth landing	The time the command module touches the water as determined from accelerometers

TABLE 6-III.- TRAJECTORY PARAMETERS[a]

Event	Reference body	Time, hr:min:sec	Latitude, deg	Longitude, deg	Altitude, mile	Space-fixed velocity, ft/sec	Space-fixed flight-path angle, deg	Space-fixed heading angle, deg E of N
Translunar phase								
Translunar injection	Earth	02:34:31.9	19.53 S	141.72 E	179.1	35 514.1	7.48	65.59
Command and service module/S-IVB separation	Earth	03:02:29.4	19.23 N	153.41 W	4 297.0	24 089.2	46.84	65.41
Docking	Earth	04:56:56	30.43 N	137.99 W	20 603.4	13 204.1	66.31	84.77
Command and service module/lunar module ejection from S-IVB	Earth	05:47:14.4	30.91 N	144.74 W	26 299.6	11 723.5	68.94	87.76
First midcourse correction								
Ignition	Earth	30:36:07.9	28.87 N	130.33 W	118 515.0	4 437.9	76.47	101.98
Cutoff	Earth	30:36:18.1	28.87 N	130.37 W	118 522.1	4 367.2	76.95	102.23
Second midcourse correction								
Ignition	Moon	76:58:12.0	0.56 N	61.40 W	11 900.3	3 711.4	-80.1	295.57
Cutoff	Moon	76:58:12.6	0.56 N	61.40 W	11 899.7	3 713.1	-80.1	295.65
Lunar orbit phase								
Lunar orbit insertion								
Ignition	Moon	81:56:40.7	2.83 N	174.81 W	87.4	8 061.4	-9.97	257.31
Cutoff	Moon	82:02:51.5	0.10 N	161.58 E	64.2	5 458.5	1.3	338.18
S-IVB impact	Moon	82:37:52.2						
Descent orbit insertion								
Ignition	Moon	86:10:53.0	6.58 N	173.60 W	59.2	5 484.8	-0.08	247.44
Cutoff	Moon	86:11:13.8	6.29 N	174.65 W	59.0	5 279.5	-0.03	246.94
Command and service module/lunar module separation	Moon	103:47:41.6	12.65 S	87.76 E	30.5	5 435.8	-1.52	241.64
Command and service module circularization								
Ignition	Moon	105:11:46.1	7.05 N	178.56 E	60.5	5 271.3	-0.1	248.58
Cutoff	Moon	105:11:50.1	7.04 N	178.35 E	60.3	5 342.1	0.22	248.36
Powered descent initiation	Moon	108:02:26.5	7.38 S	1.57 W	7.8	5 565.6	0.08	290.84
Landing	Moon	108:15:09.3						
Command and service module plane change								
Ignition	Moon	117:29:33.1	10.63 S	96.31 E	62.1	5 333.1	-0.04	237.61
Cutoff	Moon	117:29:51.6	10.78 S	95.40 E	62.1	5 333.3	0.01	241.79
Ascent	Moon	141:45:40						
Vernier adjustment	Moon	141:56:49.4	0.5 N	37.1 W	11.1	5 548.5	0.52	282.1
Terminal phase initiation	Moon	142:30:51.1	11.1 N	149.6 W	44.8	5 396.6	0.73	265.0
Terminal phase final	Moon	143:13:29.1	11.3 S	76.7 E	58.8	5 365.5	-0.002	265.5
Docking	Moon	143:32:50.5	10.18 S	161.87 W	58.6	5 353.5	0.11	268.06
Lunar module jettison	Moon	145:44:58.0	3.21 S	21.80 W	59.9	5 344.6	0.133	281.9
Command and service module separation	Moon	145:49:42.5	0.62 N	39.58 W	60.6	5 341.7	0.119	282.3
Lunar module ascent stage deorbit								
Ignition	Moon	147:14:16.9	11.92 S	67.43 E	57.2	5 358.7	0.018	267.3
Cutoff	Moon	147:15:33.1	12.12 S	63.53 E	57.2	5 177.0	0.019	267.7
Lunar module ascent stage impact	Moon	147:42:23.4	3.42 S	19.67 W	0.0	5 504.9	-3.685	281.7
Transearth injection								
Ignition	Moon	148:36:02.3	7.41 N	81.55 W	60.9	5 340.6	-0.17	260.81
Cutoff	Moon	148:38:31.5	6.64 N	168.85 E	66.5	8 505.0	5.29	266.89
Transearth coast phase								
Third midcourse correction	Earth	165:34:56.7	25.77 N	46.43 E	176 713.8	3 593.2	-79.61	124.88
Command module/service module separation	Earth	215:32:42.2	31.42 S	94.38 E	1 965.0	29 050.8	-36.62	117.11
Entry and landing phases								
Entry	Earth	215:47:45.3	36.36 S	165.80 E	66.8	36 170.2	-6.37	70.84
Landing	Earth	216:01:58.1						

[a]See table 6-IV for trajectory and orbital parameter definitions.

TABLE 6-IV.- DEFINITION OF TRAJECTORY AND ORBITAL PARAMETERS

Trajectory parameters	Definition
Geodetic latitude	The spherical coordinate measured along a meridian on the earth from the equator to the point directly beneath the spacecraft, deg
Selenographic latitude	The definition is the same as that of the geodetic latitude except that the reference body is the moon rather than the earth, deg
Longitude	The spherical coordinate, as measured in the equatorial plane, between the plane of the reference body's prime meridian and the plane of the spacecraft meridian, deg
Altitude	The distance measured along a vector from the center of the earth to the spacecraft. When the reference body is the moon, it is the distance measured from the radius of the landing site to the spacecraft along a vector from the center of the moon to the spacecraft, ft or miles
Space-fixed velocity	Magnitude of the inertial velocity vector referenced to the body-centered, inertial reference coordinate system, ft/sec
Space-fixed flight-path angle	Flight-path angle measured positive upward from the body-centered local horizontal plane to the inertial velocity vector, deg
Space-fixed heading angle	Angle of the projection of the inertial velocity vector onto the body-centered local horizontal plane, measured positive eastward from north, deg
Apogee	The point of maximum orbital altitude of the spacecraft above the center of the earth, miles
Perigee	The point of minimum orbital altitude of the spacecraft above the center of the earth, miles
Apocynthion	The point of maximum orbital altitude above the moon as measured from the radius of the lunar landing site, miles
Pericynthion	The point of minimum orbital altitude above the moon as measured from the radius of the lunar landing site, miles
Period	Time required for spacecraft to complete 360 degrees of orbit rotation, min
Inclination	The true angle between the spacecraft orbit plane and the reference body's equatorial plane, deg
Longitude of the ascending node	The longitude at which the orbit plane crosses the reference body's equatorial plane going from the Southern to the Northern Hemisphere, deg

TABLE 6-V.- MANEUVER SUMMARY

(a) Translunar

Maneuver	System	Ignition time, hr:min:sec	Firing time, sec	Velocity change, ft/sec	Resultant pericynthion conditions				
					Altitude, miles	Velocity, ft/sec	Latitude, deg:min	Longitude, deg:min	Arrival time, hr:min:sec
Translunar injection	S-IVB	2:28:32.4	350.8	10 366.5	1979	5396	4:14 N	172:24 W	82:15:19
Command and service module/lunar module separation from S-IVB	Reaction control	5:47:14.4	6.9	0.8	1980	5550	2:56 N	173:52 W	82:11:20
S-IVB evasive maneuver	S-IVB auxiliary propulsion	6:04:20	80.0	9.5	0	8368	2:05 N	131:52 W	82:01:01
First midcourse correction	Service propulsion	30:36:07.9	10.1	71.1	67	8130	2:21 N	167:48 E	82:00:45
Second midcourse correction	Service propulsion	76:58:12	0.65	3.5	61	8153	2:12 N	167:41 E	82:40:36

(b) Lunar orbit

Maneuver	System	Ignition time, hr:min:sec	Firing time, sec	Velocity change, ft/sec	Resultant orbit	
					Apocynthion, miles	Pericynthion, miles
Lunar orbit insertion	Service propulsion	81:56:40.7	370.8	3022.4	169.0	58.1
Descent orbit insertion	Service propulsion	86:10:53	20.8	205.7	58.8	9.1
Command module/lunar module separation	Service module reaction control	103:47:41.6	2.7	0.8	60.2	7.8
Lunar orbit circularization	Service propulsion	105:11:46.1	4.0	77.2	63.9	56.0
Powered descent initiation	Descent propulsion	108:02:26.5	764:6	6639.1	-	-
Lunar orbit plane change	Service propulsion	117:29:33.1	18.5	370.5	62.1	57.7
Lunar orbit insertion	Ascent propulsion	141:45:40	432.1	6066.1	51.7	8.5
Vernier adjustment	Lunar module reaction control	141:56:49.4	12.1	10.3	51.2	8.4
Terminal phase initiation	Ascent propulsion	142:30:51.1	3.6	88.5	60.1	46.0
Terminal phase finalization	Lunar module reaction control	143:13:29.1	26.7*	32.0*	61.5	58.2
Final separation	Service module reaction control	145:49:42.5	15.8	3.4	63.4	56.8
Lunar module deorbit	Lunar module reaction control	147:14:16.9	76.2	186.1	56.7	-59.8

*Theoretical values.

(c) Transearth

Event	System	Ignition time, hr:min:sec	Firing time, sec	Velocity change, ft/sec	Resultant entry interface condition				
					Flight-path angle, deg	Velocity, ft/sec	Latitude, deg:min	Longitude, deg:min	Arrival time, hr:min:sec
Transearth injection	Service propulsion	148:36:02.3	149.2	3460.6	-7.3	36 127	27:02 S	171:30 W	216:26:59
Third midcourse correction	Service module reaction control	165:34:56.7	3.0	0.5	-6.63	36 170	36:30 S	165:15 E	216:27:31

higher descent propulsion system propellant margin. Both vehicles remained in the low-pericynthion orbit until shortly after lunar module separation. After separation, the pericynthion of the command and service modules was increased to 56 miles and a plane-change maneuver was later executed to establish the proper conditions for rendezvous.

6.2.2 Lunar Descent

Preparations for lunar descent.- The powered descent and lunar landing were similar to those of previous missions. However, the navigation performed in preparation for powered descent was more accurate because of the command and service modules being in the 58.8- by 9.1-mile descent orbit for 22 hours prior to powered descent initiation. While in this orbit, the Network obtained long periods of radar tracking of the unperturbed spacecraft from which a more accurate spacecraft state vector was determined. The position of the command module relative to a known landmark near the landing site was accurately determined from sextant marks taken on the landmark. Corrections for known offset angles between the landmark and the landing site were used to compute a vector to the landing site. This vector was sent to the lunar module. Also, the Mission Control Center propagated this vector forward to the time of landing to predict errors due to navigation. This procedure was performed during the two revolutions before powered descent and a final landing site update of 2800 feet was computed and relayed to the crew. After ignition for the powered descent, the crew manually inserted the update into the computer.

Powered descent.- Trajectory control during descent was nominal, and only one target redesignation of 350 feet left (toward the south) was made to take advantage of a smoother landing area. After manual takeover, the crew flew approximately 2000 feet downrange and 300 feet north (fig. 6-1) because the targeted coordinates of the landing site given to the lunar module computer were in error by about 1800 feet.

Coordinates of the landing point are 3 degrees 40 minutes 24 seconds south latitude and 17 degrees 27 minutes 55 seconds west longitude, which is 55 feet north and 165 feet east of the prelaunch landing site (fig. 6-2). (Further discussion of the descent is contained in section 8.6.)

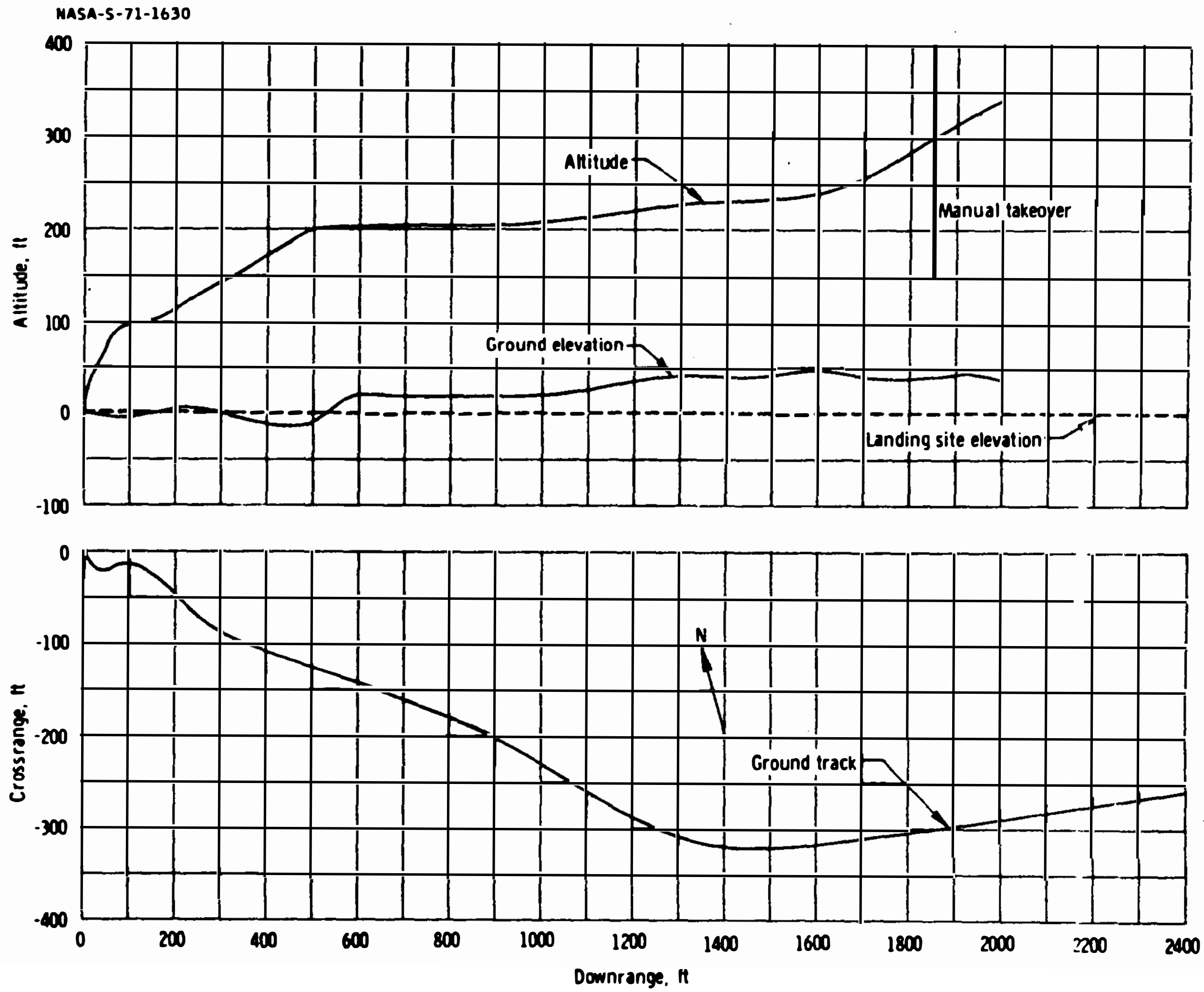

Figure 6-1.- Crossrange and altitude plotted against downrange during final phase of descent.

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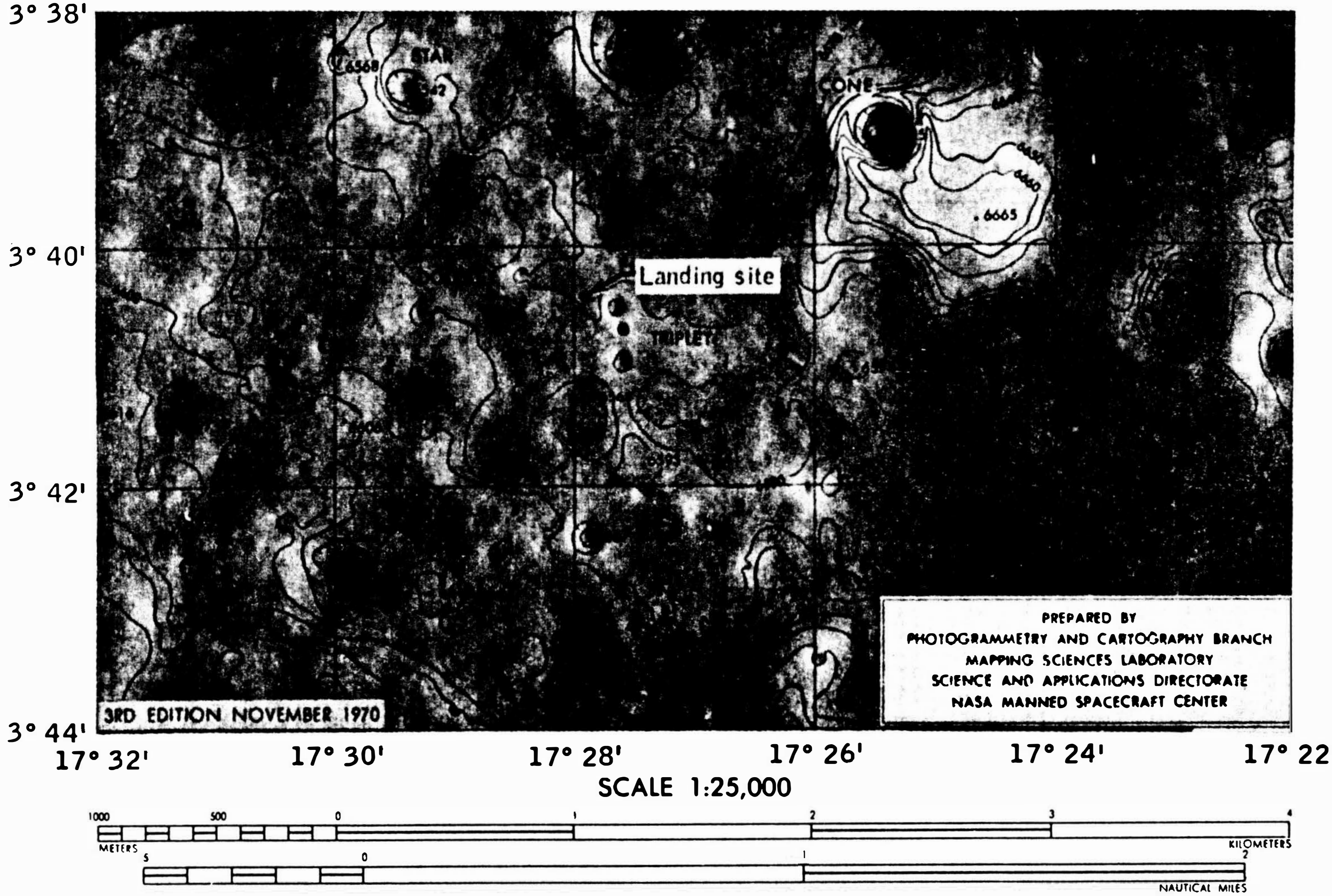

Figure 6-2.- Lunar module landing site on lunar topographic photomap of Fra Mauro.

6.2.3 Lunar Ascent and Rendezvous

Lift-off from the lunar surface occurred at 141:45:40, during the 31st lunar revolution of the command and service modules. After 432.1 seconds of firing time, the ascent engine was automatically shut down with velocity residuals of minus 0.8, plus 0.3, and plus 0.5 ft/sec in the X, Y, and Z axes, respectively. These were trimmed to minus 0.1, minus 0.5, and plus 0.5 ft/sec in the X, Y, and Z axes, respectively. Comparison of the primary guidance, abort guidance, and the powered flight processor data showed good agreement throughout the ascent as can be seen in the following table of insertion parameters.

Data source	Horizontal velocity, ft/sec	Radial velocity, ft/sec	Altitude, ft
Primary guidance and navigation system	5544	30	60 311
Powered flight processor	5544	29	60 345
Abort guidance system	5542	29	60 309

To accomplish a direct rendezvous with the command module, a reaction control system vernier adjustment maneuver of 10.3 ft/sec was performed approximately 4 minutes after ascent engine cutoff. The maneuver was necessary because the lunar module ascent program is targeted to achieve an insertion velocity and not a specific position vector. Direct rendezvous was nominal and docking occurred 1 hour 47 minutes 10 seconds after lunar lift-off.

The lunar module rendezvous navigation was accomplished throughout the rendezvous phase and all solutions agreed closely with the ground solution. The command module which was performing backup rendezvous navigation was not able to obtain acceptable VHF ranging data until after the terminal phase initiation maneuver. The VHF anomaly is discussed in section 14.1.4. Figure 14-7 is a comparison of the relative range as measured by lunar module rendezvous radar and command module VHF, and determined from command module state vectors and the best-estimate trajectory propagations. The VHF mark taken at 142:05:15 and incorporated into the command module computer's state vector for the lunar module caused an 8.8-mile relative range error.

Several sextant marks were taken after this error was introduced. Because the computer weighs the VHF marks more heavily than the sextant marks, the additional sextant marks did not reduce the error significantly. The ranging problem apparently cleared up after the terminal phase initiation maneuver and the VHF was used satisfactorily for the midcourse corrections. Table 6-VI provides a summary of the rendezvous maneuver solutions.

TABLE 6-VI.- RENDEZVOUS SOLUTIONS

Maneuver	Computed velocity change, ft/sec		
	Network	Lunar module	Command and service module
Terminal phase initiation	V_x = 63.0 V_y = 1.0 V_z = 67.0 V_t = 92.0	V_x = 62.1 V_y = 0.1 V_z = 63.1 V_t = 88.5	V_x = -67.4 V_y = 0.5 V_z = -69.2 V_t = 96.6
First midcourse correction	No ground solution.	V_x = -0.9 V_y = 0.2 V_z = 0.6 V_t = 1.1	V_x = 1.3 V_y = -0.1 V_z = -1.1 V_t = 1.7
Second midcourse correction	No ground solution.	V_x = -0.1 V_y = 0.1 V_z = -1.4 V_t = 1.6	V_x = 0.6 V_y = -0.2 V_z = -2.2 V_t = 2.3

6.2.4 Lunar Module Deorbit

Two hours after docking, the command and service modules and lunar module were oriented to the lunar module deorbit attitude, undocked, and the command and service modules then separated from the lunar module. The lunar module was deorbited on this mission, similar to Apollo 12. The deorbit was performed to eliminate the lunar module as an orbital debris hazard for future missions and to provide an impact that could be used as a calibrated impulse for the seismographic equipment. The reaction control system of the lunar module was used to perform the 75-second deorbit firing 1 hour 24 minutes 19.9 seconds after the command and service modules had separated from the lunar module. The lunar module impacted the lunar surface at 3 degrees 25 minutes 12 seconds south latitude and 19 degrees 40 minutes 1 second west longitude with a velocity of about 5500 feet per second. This point was 36 miles from the Apollo 14 landing site, 62 miles from the Apollo 12 landing site, and 7 miles from the prelaunch target point.

6.3 TRANSEARTH AND ENTRY TRAJECTORIES

A nominal transearth injection maneuver was performed at about 148 hours 36 minutes. Seventeen hours after transearth injection, the third and final midcourse correction was performed.

Fifteen minutes prior to entering the earth's atmosphere, the command module was separated from the service module. The command module was then oriented to blunt-end-forward for earth entry. Entry was nominal and the spacecraft landed in the Pacific Ocean less than one mile from the prelaunch target point.

6.4 SERVICE MODULE ENTRY

The service module should have entered the earth's atmosphere and its debris landed in the Pacific Ocean approximately 650 miles southwest of the command module landing point. No radar coverage was planned nor were there any sightings reported for confirmation.

7.0 COMMAND AND SERVICE MODULE PERFORMANCE

7.1 STRUCTURAL AND MECHANICAL SYSTEMS

Structural loads on the spacecraft during all phases of the mission were within design limits. The predicted and calculated loads at lift-off, in the region of maximum dynamic pressure, at the end of first stage boost, and during staging were similar to those of previous missions. Command module accelerometer data prior to S-IC center engine cutoff indicate a sustained 5-hertz longitudinal oscillation with an amplitude of 0.17g, which is similar to that measured during previous flights. Oscillations during the S-II boost phase had a maximum measured amplitude of less than 0.06g at a frequency of 9 hertz. The amplitudes of both oscillations were within acceptable structural design limits.

Six attempts were required to dock the command and service module with the lunar module following translunar injection. The measured rates and indicated reaction control system thruster activity during the five unsuccessful docking attempts show that capture should have occurred each time. The mechanism was actuated and inspected in the command module following docking. This investigation indicated that the probe mechanical components were functioning normally. Subsequent undocking and docking while in lunar orbit were normal. The probe was returned for postflight analysis. The docking anomaly is discussed in detail in section 14.1.1.

7.2 ELECTRICAL POWER

7.2.1 Power Distribution

The electrical power distribution system performed normally except for two discrepancies. Prior to entry, when the bus-tie motor switches were operated to put the entry batteries on the main busses, battery C was not placed on main bus B. This anomaly was discovered by the data review after the flight. Postflight continuity checks revealed that the circuit breaker tying battery C to main bus B was inoperative. This anomaly is described in section 14.1.7.

The second discrepancy occurred during entry. Procedures call for main bus deactivation, at 800 feet altitude, by opening the bus tie motor switches. The crew reported that operation of the proper switches did not remove power from the buses. The buses were manually deactivated, after landing, by opening the in-line circuit breakers on Panel 275 (a normal procedure). Review of data indicated and postflight tests confirmed that the motor switch which tied battery A to main bus A was inoperative. This anomaly is described in section 14.1.6.

7.2.2 Fuel Cells

The fuel cells were activated 48 hours prior to launch, conditioned for 4 hours, and configured with fuel cell 2 on the line supplying a 20-ampere load as required in the countdown procedure. Fuel cells 1 and 3 remained on open circuit until 5 hours prior to launch. At launch, fuel cell 1 was on main bus A with fuel cell 2, and fuel cell 3 was on main bus B. This configuration was maintained throughout the flight. Initially, the load variance was approximately 5 amperes, but it stabilized to 3 or 4 amperes early in the flight. This is normal and typical of other flights.

All fuel cell parameters remained within normal operating limits and agreed with predicted flight values. As expected, the fuel cell 1 condenser-exit temperature exhibited a periodic fluctuation about every 6 minutes throughout the flight. This zero-gravity phenomenon was similar to that observed on all other flights and has no effect on fuel cell performance (ref. 6).

The fuel cells supplied 435 kW-h of energy at an average current of 23 amperes per fuel cell and a mean bus voltage of 29 volts during the mission.

7.2.3 Batteries

The command and service module entry and pyrotechnic batteries performed normally. Entry batteries A and B were both charged once at the launch site and five times during flight with nominal charging performance. Load sharing and voltage delivery were satisfactory during each of the service propulsion firings. The batteries were essentially fully charged at entry.

7.3 CRYOGENIC STORAGE

Cryogenics were satisfactorily supplied to the fuel cells and to the environmental control system throughout the mission. The configuration changes made as a result of the Apollo 13 oxygen tank failure are described in appendix A. A supplemental report giving details of system performance will be issued at a later date (appendix E).

During preflight checkout of the oxygen system, the single-seat check valve for tank 2 was found to have failed in the open position and was replaced with an in-line double-seat valve. During flight, this valve allowed gas leakage into tank 2 from tank 3. The purpose of this valve is primarily to isolate tank 2 from the remainder of the system should tank 2 fail. Thus, it was qualified at a reverse differential pressure of 60 psid. This is significantly higher than that normally experienced during a flight. Tests have been conducted to characterize the nature of the check valve leakage at low pressure differential and show that this situation is not detrimental to operation under abnormal as well as normal conditions.

Two flow tests on the oxygen system were conducted during flight. One was to demonstrate the capability of the system to support additional flow requirements for extravehicular activities. The other was to determine the heater temperature while operating with the oxygen density less than 20 percent. The intent of these two tests was met and favorable results were obtained although test procedures were modified because of time constraints. The oxygen system is capable of supporting the anticipated requirements for Apollo 15 and subsequent missions. The low-density flow test indicated that the oxygen system can provide required flow rates at low densities and the data obtained provides for a more accurate assessment of heater operating temperature.

Consumable quantities in the cryogenic storage system are discussed in section 7.10.3.

7.4 COMMUNICATIONS EQUIPMENT

The communications system satisfactorily supported the mission except for the following described conditions.

The high-gain antenna failed to acquire and track properly at various times during the mission. The problems occurred during the acquisition of signal rather than after acquisition. In this regard, the problem is different from those experienced during Apollo 12 and 13 where the high-gain antenna lost lock or failed to track after acquisition. This is discussed in further detail in section 14.1.2.

From just prior to lunar lift-off through terminal phase initiation, the VHF system performance was marginal. Voice communications were weak and noisy, and the VHF ranging performance was erratic and erroneous. The voice communications problem is not related to the VHF problems experienced on previous missions where they were determined to be procedural errors. Switching antennas in the command and service module and elimination of the ranging signal did not clear up the problems. The problems are believed to have been caused by equipment malfunction, but the source has not been isolated to a particular component of the total system. Section 14.1.4 contains a detailed discussion of this anomaly.

7.5 INSTRUMENTATION

The instrumentation system functioned normally throughout the mission except for the loss of the reaction control system quad B oxidizer manifold pressure measurement during separation of the command and service module from the launch vehicle. The most probable cause of the failure was a break of the signal or power leads initiated by the pyrotechnic shock associated with the spacecraft/launch vehicle adapter panel separation. Since this is the only failure of four measurements of this type on each of eight flights, the pyrotechnic shock is not considered a problem for normal elements of the instrumentation circuit. Further, redundant measurements are available to permit determination of the required data. Consequently, no corrective action is required.

7.6 GUIDANCE, NAVIGATION, AND CONTROL

Attitude control was nominal throughout the mission including all periods of passive thermal control, cislunar navigation, as well as photography and landmark tracking from lunar orbit. The stability of the inertial measurement unit error parameters was excellent. The only anomaly in the guidance, navigation and control systems was failure of the entry monitor system 0.05g light to illuminate. This is discussed in section 14.1.5.

Because of inclement weather, the lift-off was delayed for the first time in the Apollo program. This required the flight azimuth to be changed from 72 degrees to 75.56 degrees and the platform to be realigned accordingly. A comparison of command and service module and S-IVB navigation data indicated satisfactory performance during the launch phase. Insertion errors were plus 7.02, plus 61.02, and minus 7.50 ft/sec in the X, Y, and Z axes, respectively. These errors were comparable to those observed on other Apollo launches. The only significant error was in the Y-axis velocity caused by a prelaunch azimuth alignment error of 0.14 degree due to one-sigma gyrocompassing inaccuracies. Table 7-I is a summary of preflight inertial measurement unit error parameters after its installation in the command module. An update to the inertial parameters was performed at approximately 29 hours. The three accelerometer biases were updated to minus 0.32, plus 0.12 and minus 0.13 cm/sec^2, and the X-gyro null bias drift was updated to plus 0.4 meru (milli earth-rate units).

The first platform realignment was performed after insertion and agreed with the predicted alignment errors due to prelaunch azimuth errors. Table 7-II is a summary of significant parameters during each of the platform realignments.

TABLE 7-I.- INERTIAL COMPONENT PREFLIGHT HISTORY - COMMAND MODULE

Error	Sample mean	Standard deviation	No. of samples	Countdown value	Flight load	Inflight performance
Accelerometers						
X - Scale factor error, ppm	-444	58	8	-500	-370	-
Bias, cm/sec^2	-0.23	0.13	8	-0.31	-0.23	-0.34
Y - Scale factor error, ppm	-441	49	8	-505	-500	-
Bias, cm/sec^2	0.05	0.07	8	0.13	0.04	0.09
Z - Scale factor error, ppm	-278	49	8	-320	-310	-
Bias, cm/sec^2	-0.29	0.07	8	-0.18	-0.29	-0.18
Gyroscopes						
X - Null bias drift, meru	0.9	0.6	8	1.8	2.5	[a]0.0
Acceleration drift, spin reference axis, meru/g	3.0	2.0	8	4.9	1.0	-
Acceleration drift, input axis, meru/g	1.7	1.5	8	-1.6	0.0	-
Y - Null bias drift, meru	-3.4	0.8	8	-4.2	-3.4	1.7
Acceleration drift, spin reference axis, meru/g	3.2	1.5	8	3.8	3.0	-
Acceleration drift, input axis, meru/g	-9.9	4.5	16	-9.7	-5.0	-
Z - Null bias drift, meru	1.6	0.9	8	2.5	1.6	0.0
Acceleration drift, spin reference axis, meru/g	-3.1	1.0	8	-2.4	-3.9	-
Acceleration drift, input axis, meru/g	43.8	6.4	8	54.1	40.0	-

[a]Inflight performance average before update was minus 2.0.

Spacecraft dynamics during separation from the S-IVB were very small. Spacecraft dynamics during each docking attempt were small and comparable to those seen on previous Apollo missions. Figure 7-1 is a time history of significant control system parameters during each docking attempt.

Performance during each of the seven service propulsion system maneuvers was nominal. Trimming of residual velocity errors was performed only after the circularization and transearth injection maneuvers. Table 7-III is a summary of significant control system parameters for each of the maneuvers. The second midcourse correction was accomplished with a minimum-impulse service propulsion system maneuver in order to

TABLE 7-II.- COMMAND AND SERVICE MODULE PLATFORM ALIGNMENT SUMMARY

TABLE 7-II.- COMMAND AND SERVICE MODULE PLATFORM ALIGNMENT SUMMARY

Time, hr:min	Program option*	Star used	Gyro torquing angle, deg			Star angle difference, deg	Gyro drift, meru			Comments
			X	Y	Z		X	Y	Z	
00:58	3	22 Regulus, 24 Gienah	0.085	0.010	0.166	0.00				Launch orientation
6:40	3	17 Regor, 14 Canopus	0.127	-0.060	-0.011	0.00	-1.4	+0.7	-0.1	Launch orientation
14:13	3	31 Arcturus, 35 Rasalhague	0.271	-0.127	-0.036	0.01	-2.5	1.2	-0.3	Passive thermal control orientation
29:20	3	20 Dnoces, 23 Denebola	0.449	-0.130	0.082	0.01	-2.0	0.6	0.4	Passive thermal control orientation
40:11	3	1 Alpheratz, 40 Altair	-0.039	-0.221	0.046	0.00	0.2	1.4	0.3	Passive thermal control orientation
53:11	3	20 Dnoces, 23 Denebola	0.006	-0.129	0.052	0.00	-0.0	0.7	0.3	Passive thermal control orientation
59:41	3	13 Capella, 3 Navi	-0.073	-0.093	0.033	0.00	0.8	1.1	0.4	Passive thermal control orientation
76:52	3	23 Denebola, 32 Alphecca	0.056	-0.262	0.038	0.00	-0.2	1.0	0.1	Passive thermal control orientation
79:39	3	27 Alkaid, 35 Rasalhague	-0.007	-0.045	0.010	0.00	0.2	1.1	0.2	Passive thermal control orientation
84:09	3	30 Menkent, 35 Rasalhague	0.001	-0.055	0.002	0.01	-0.2	1.2	-0.5	Landing site orientation
86:10	3	16 Procyon, 17 Regor	-0.050	-0.070	-0.045	0.01	1.7	2.3	-1.5	Landing site orientation
88:05	3	16 Procyon, 20 Dnoces	-0.031	0.002	0.027	0.01	1.1	0.1	0.9	Landing site orientation
101:24	3	17 Regor, 30 Menkent	0.073	-0.229	0.000	0.00	-0.4	1.1	0.0	Landing site orientation
105:09	3	40 Altair, 42 Peacock	0.030	-0.038	0.028	0.01	-0.6	0.7	0.2	Landing site orientation
109:12	3	34 Atria, 37 Nunki	-0.012	-0.043	0.003	0.01	0.2	0.7	0.0	Landing site orientation
117:08	3	22 Regulus, 27 Alkaid	0.021	-0.105	0.055	0.02	-0.2	0.9	0.5	Landing site orientation
119:27	3	12 Rigel, 21 Alphard	-0.027	-0.065	0.018	0.00	1.3	1.9	0.5	Launch orientation
131:19	3	10 Mirfak, 12 Rigel	-0.036	-0.157	0.091	0.01	0.3	1.2	0.7	Launch orientation
137:18	3	6 Acamar, 14 Canopus	-0.002	-0.166	-0.005	0.00	0.0	1.8	-0.1	Launch orientation
140:53	3	31 Arcturus, 30 Menkent	0.079	-0.006	-0.001	0.00	-1.3	0.1	-0.0	Launch orientation
146:58	3	24 Gienah, 31 Arcturus	0.018	-0.091	0.050	0.00	-0.2	1.0	0.5	Launch orientation
150:17	3	4 Achernar, 34 Atria	0.037	-0.106	-0.043	0.01	-0.7	2.1	0.9	Transearth injection orientation
163:49	3	11 Aldebaran, 16 Procyon	0.046	-0.174	0.017	0.00	-0.2	0.8	0.1	Passive thermal control orientation
186:34	3	25 Acrux, 42 Peacock	0.040	-0.460	0.076	0.00	-0.1	1.3	0.1	Passive thermal control orientation
192:14	3	41 Dabih, 34 Atria	-0.038	-0.104	-0.003	0.01	0.1	1.2	0.0	Passive thermal control orientation
196:58	3	17 Regor, 40 Altair	-0.009	-0.109	0.038	0.01	0.1	1.5	0.5	Passive thermal control orientation
208:11	3	25 Acrux, 33 Antares	0.071	-0.161	0.026	0.01	-0.4	1.0	0.2	Passive thermal control orientation
212:59	3	16 Procyon, 23 Denebola	-0.049	-0.010	0.014	0.01	0.7	0.1	0.2	Passive thermal control orientation
213:11	1	23 Denebola, 16 Procyon	0.021	0.002	-0.036	0.01	-1.0	-1.0	-1.6	Entry orientation
214:39	3	30 Menkent, 37 Nunki	0.039	-0.040	-0.069	0.00	-1.8	1.8	-3.2	Entry orientation

*1 - Preferred; 2 - Nominal; 3 - REFSMMAT; 4 - Landing site.

TABLE 7-III.- GUIDANCE AND CONTROL MANEUVER SUMMARY

Parameter	Maneuver							
	First midcourse correction	Second midcourse correction	Lunar orbit insertion	Descent orbit insertion	Lunar orbit circularization	First plane change	Transearth injection	Third midcourse correction[a]
Time								
Ignition, hr:min:sec	30:36:07.91	76:58:11.98	81:56:40.70	86:10:52.97	105:11:46.11	117:29:33.17	148:36:02.3	165:34:56.69
Cutoff, hr:min:sec	30:36:18.10	76:58:12.63	82:02:51.54	86:11:13.78	105:11:50.13	117:29:51.67	148:38:31.53	--
Duration, min:sec	0:10.19	0:00.65	6:10.84	0:20.81	0:04.02	0:18.5	2:29.23	0:03.00
Velocity gained, ft/sec[b] (desired/actual)								
X	+11.0/+10.9	-1.8/-1.9	+1957.9/+1958.2	+185.3/+185.7	-76.8/-74.9	-74.5/-74.4	-3284.7/-3285.4	-0.5/-0.7
Y	+63.1/+63.3	+0.3/+0.2	-2301.0/-2301.2	-51.4/-52.5	-11.1/-10.6	+188.1/+188.0	+236.3/+236.6	+0.2/0
Z	+30.9/+30.9	+3.3/+3.4	+80.0/+79.9	-73.0/-73.2	-9.6/-9.3	-310.1/-310.9	-1061.3/-1061.8	+0.1/0
Velocity residual, ft/sec[c]								
X	+0.3	+0.3	+0.3	+0.6	-1.0	+0.6	+.1	+0.2
Y	0	0	0	+0.2	0	+0.4	+.8	+0.2
Z	-0.1	0	0	0	+0.5	+0.2	-.3	+0.1
Entry monitor system	+0.3	+0.5	-0.3	-0.2	+0.4	+1.2	+2.5	0
Engine gimbal position, deg								
Initial								
Pitch	+1.00	+0.87	+0.87	+1.50	-0.75	-0.88	-0.66	
Yaw	-0.18	-0.24	-0.26	-0.60	+0.24	+0.20	+0.12	
Maximum excursion								
Pitch	+0.32	+0.05	+0.49	+0.27	-1.92	-2.14	-2.10	
Yaw	-0.47	-0.09	+0.49	-0.30	+1.61	+1.53	+1.27	N/A
Steady-state								
Pitch	+1.00	N/A	+1.14	+1.59	-0.71	-0.68	-0.53	
Yaw	-0.18		-0.26	-0.60	+0.12	+0.16	-0.26	
Cutoff								
Pitch	+1.00	+0.92	+1.63	+1.72	-0.71	-0.62	-0.62	
Yaw	-0.26	-0.35	-0.65	-0.60	+0.07	-0.05	-1.62	
Maximum rate excursion, deg/sec								
Pitch	-0.12	0	-0.16	+0.28	+1.23	+1.42	+1.32	
Yaw	-0.12	0	+0.16	+0.20	-0.68	-1.12	-1.32	N/A
Roll	±0.08	0	+0.20	+0.12	-0.59	-0.72	-1.86	
Maximum attitude error, deg								
Pitch	-0.15	0	+0.16	-0.16	-0.31	+0.25	+0.24	
Yaw	-0.22	-0.04	-0.15	-0.08	-0.14	-0.26	-0.31	N/A
Roll	-1.31	0	+5.00	-0.60	-0.84	-3.78	±5.00	

[a]This maneuver was performed using reaction control system.

[b]Inertial coordinates

[c]Body coordinates (+ indicates underburn)

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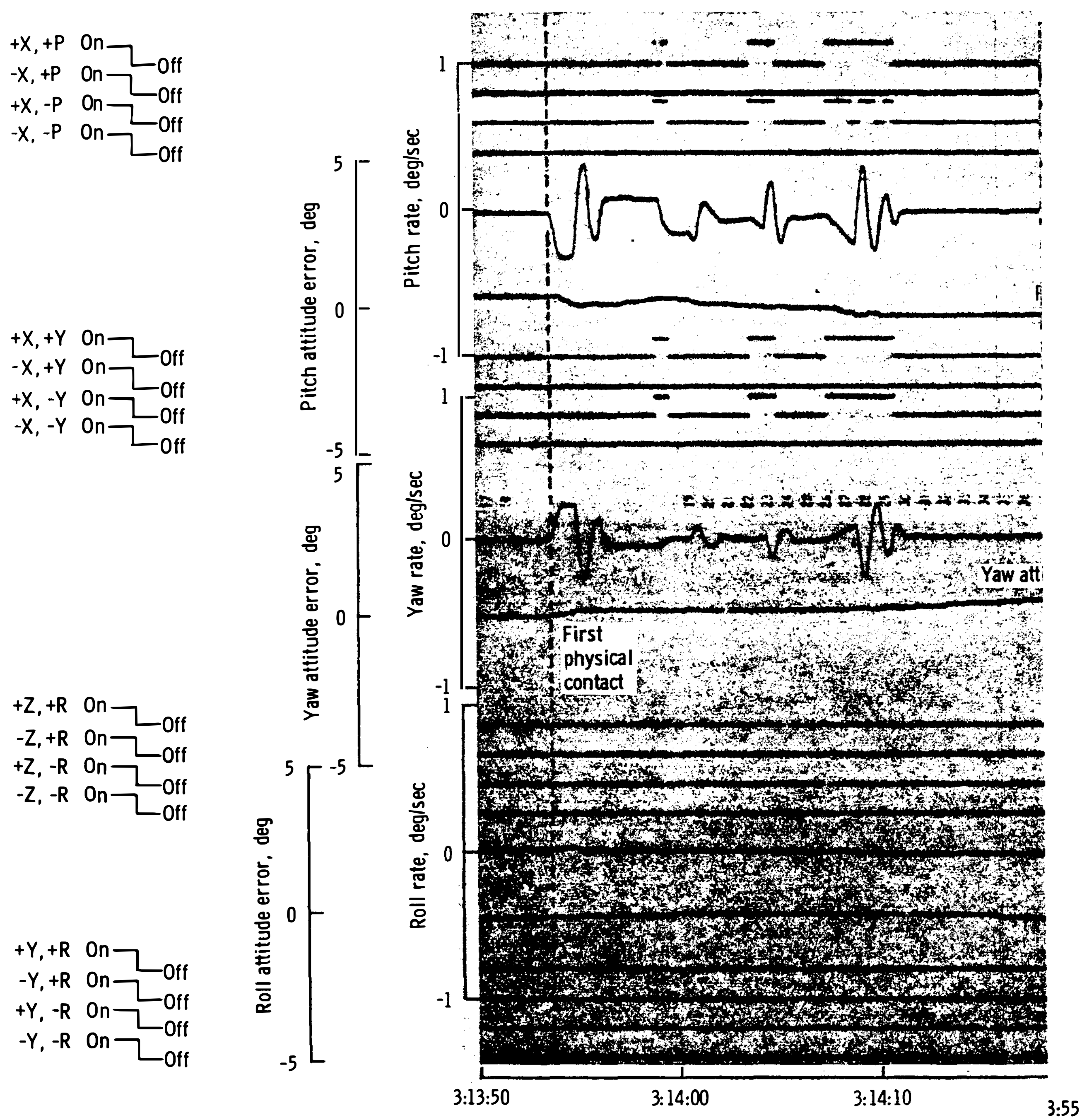
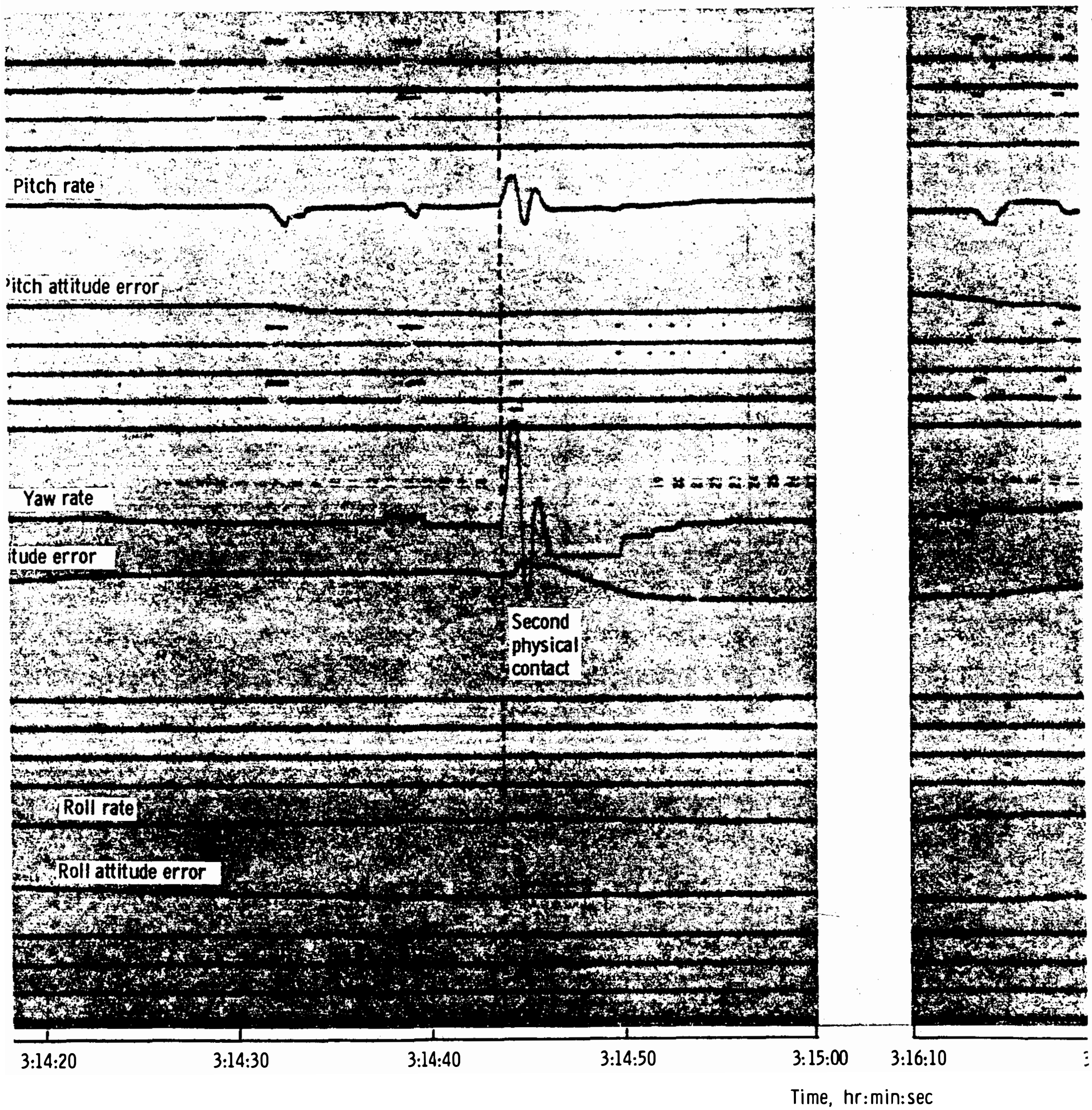


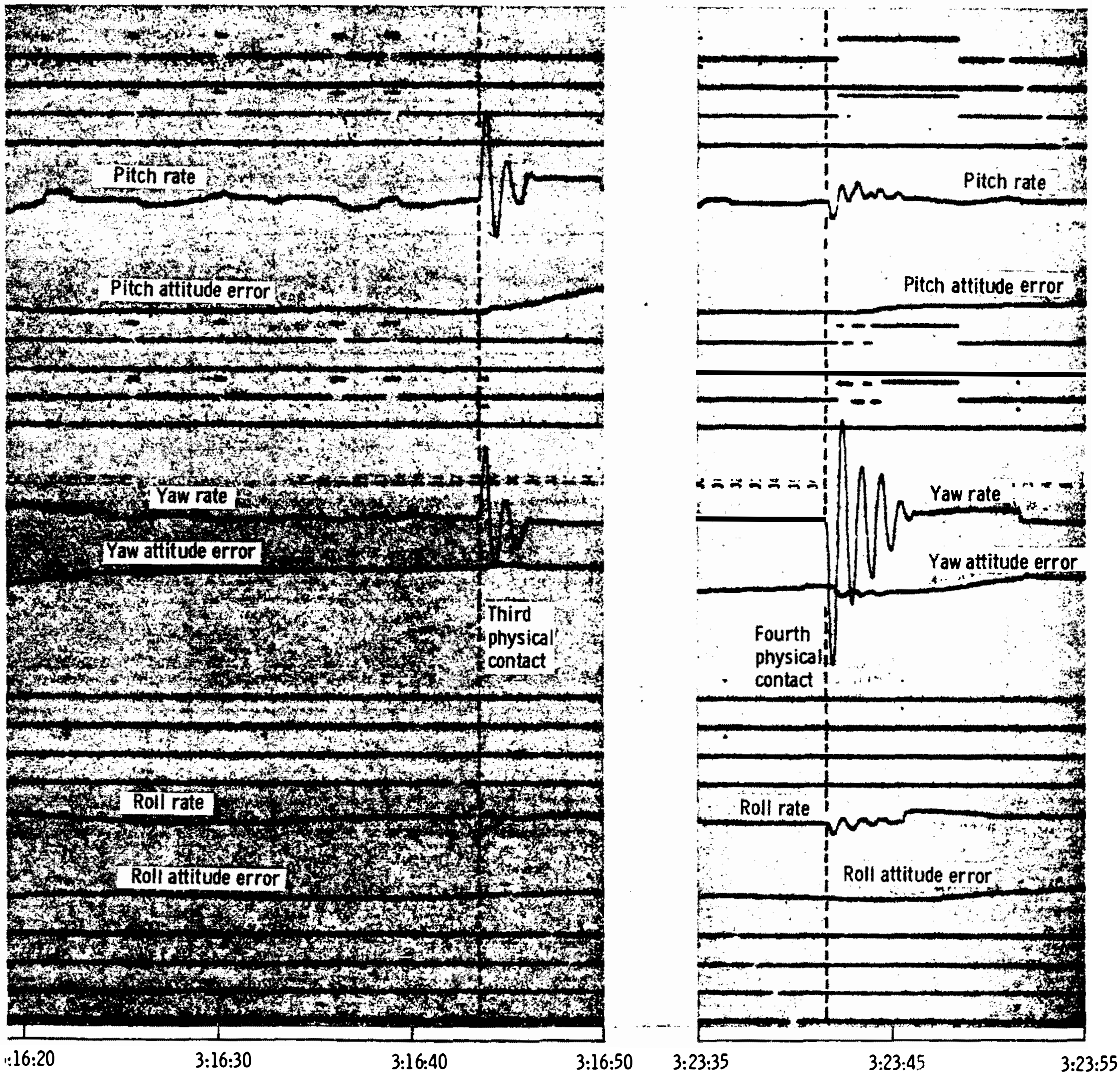
Figure 7-1. - Time history of control system parameters during mu

7-8 B

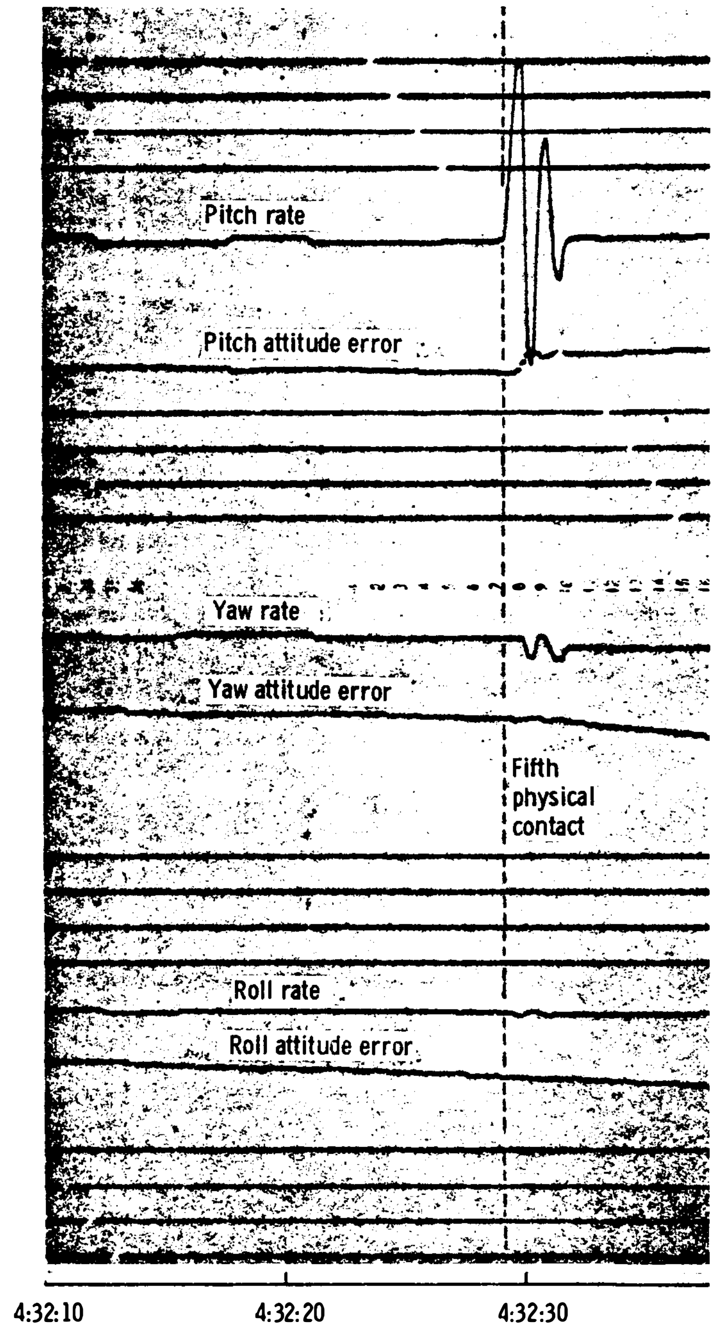


ltiple docking attempts.

7-8 C



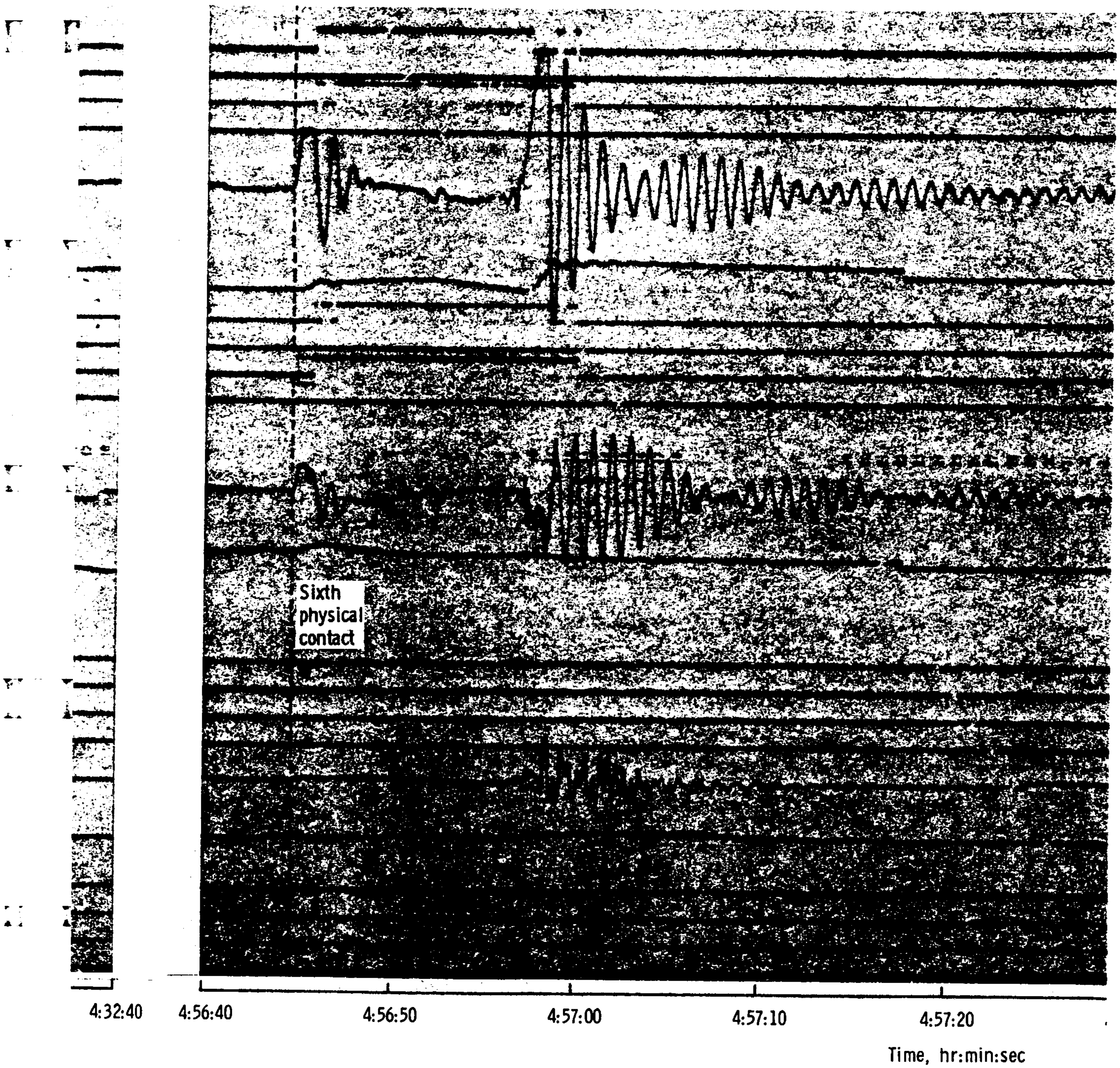
Pitch rate
Pitch attitude error
Yaw rate
Yaw attitude error
Third physical contact
Roll rate
Roll attitude error
:16:20
3:16:30
3:16:40
3:16:50
Pitch rate
Pitch attitude error
Yaw rate
Yaw attitude error
Fourth physical contact
Roll rate
Roll attitude error
3:23:35
3:23:45
3:23:55



Pitch rate
Pitch attitude error
Yaw rate
Yaw attitude error
Fifth physical contact
Roll rate
Roll attitude error
4:32:10
4:32:20
4:32:30

7-9 A

7-


Sixth physical contact
4:32:40
4:56:40
4:56:50
4:57:00
4:57:10
4:57:20
Time, hr:min:sec

9 B

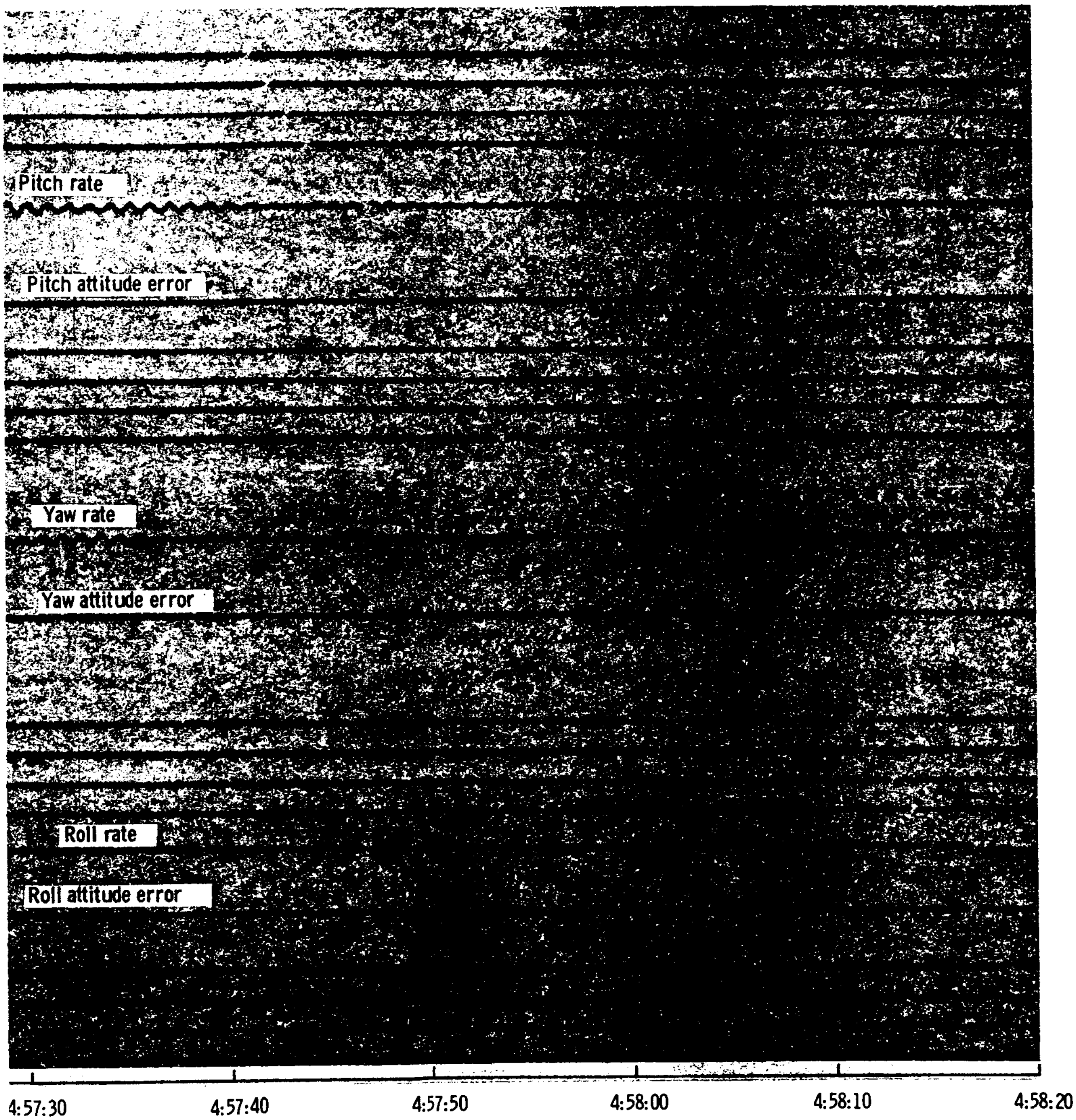


Figure 7-1. - Time history of control system parameters during multiple docking attempts.

conserve reaction control propellant. This was the first service propulsion system minimum-impulse maneuver performed during a lunar mission. The third midcourse correction was performed with the reaction control system.

During the translunar phase, a series of star-horizon measurements were taken to establish the precise location of the earth horizon. This was done in preparation for a cislunar navigation exercise to be performed during the transearth phase.

The command and service module combination was separated from the lunar module after the descent orbit insertion maneuver. Command and service module circularization and plane-change maneuvers were then performed, and the Command Module Pilot accomplished a series of photographic and landmark tracking operations. For the first time, rate-aided optics were available to assist the crew in making optical sightings.

The sextant and VHF ranging data were used to track the lunar module after the vernier adjustment maneuver following ascent from the lunar surface. Unacceptable VHF ranging data were received in the interval between lunar module insertion and the terminal phase initiation maneuver; however, the data received during the final phase of rendezvous were good. For a detailed discussion of rendezvous, see section 6.2.3. For a discussion of the VHF ranging anomaly, see section 14.1.4.

Only one midcourse correction was required on the return trip to meet the entry interface conditions. Cislunar navigation was performed during the transearth phase to simulate returning to earth with no communications. Accuracy of the onboard navigation techniques was demonstrated but the crew commented that the computer/crew operational interface could be improved by incorporating a recycle feature in the cislunar navigational sighting program.

The command module was separated from the service module at 215:32:42 and the normal pitch-down disturbance was observed. The entry monitor system 0.05g light did not illuminate within the allowed 3 seconds after the predicted time for 0.05g. The crew started the system manually according to the checklist. Refer to section 14.1.5 for further discussion of this anomaly. Table 7-IV is a summary of entry monitor system null-bias tests performed during the mission. Accelerometer stability and performance were excellent.

The primary guidance system guided the command module to a landing at 27 degrees 0 minutes 45 seconds south latitude and 172 degrees 39 minutes 30 seconds west longitude, which is 0.62 mile from the targeted landing point.

TABLE 7-IV.- RESULTS OF ENTRY MONITOR SYSTEM NULL BIAS TESTS

Test*	1	2	3	4	5	6	7	8	9
Time	01:50:00	09:34:50	29:11:20	58:28:00	75:59:00	79:45:00	84:31:00	118:20:00	165:15:00
Entry monitor system reading at start of test, ft/sec	-100	-100	-100	-100	-100	-100	-100	-100	-100
Entry monitor system reading at end of test, ft/sec	-99.5	-99.4	-99.6	-98.9	-98.4	-98.5	-99.4	-98.5	-99.0
Differential velocity bias, ft/sec**	+0.5	+0.6	+0.4	+1.1	+1.6	+1.5	+0.6	+1.5	+1.0
Null bias, ft/sec^2	+0.005	+0.006	+0.004	+0.011	+0.016	+0.015	+0.006	+0.015	+0.010

*Each test duration is 100 seconds.
**Count up is positive bias.

7.7 REACTION CONTROL SYSTEMS

7.7.1 Service Module

Performance of the service module reaction control was normal throughout the mission. All telemetry parameters stayed within nominal limits throughout the mission with the exception of the quad B oxidizer manifold pressure. This measurement was lost when the command and service module separated from the S-IVB. The quad B helium and fuel manifold pressures were used to verify proper system operation. Total propellant consumption for the mission was 102 pounds less than predicted; however, propellant consumption during transposition, docking and extraction was about 60 pounds more than planned because of the additional maneuvering associated with the docking difficulties. The propellant margin deficiency was recovered prior to lunar orbit insertion, and nominal margins existed during the remainder of the mission. Consumables information is contained in section 7.10.2.

7.7.2 Command Module

The command module reaction control systems performed satisfactorily. Both systems 1 and 2 were activated during the command module/service module separation sequence. Shortly after separation, system 2 was disabled and system 1 was used for the remainder of entry. All telemetry data indicated nominal system performance throughout the mission. Consumables information is contained in section 7.10.2.

7.8 SERVICE PROPULSION SYSTEM

Service propulsion system performance was satisfactory based on the steady-state performance during all firings. The steady-state pressure data, gaging system data, and velocities gained indicated essentially nominal performance. The engine transient performance during all starts and shutdowns was satisfactory. Nothing in the flight data or postflight analysis indicated combustion instability or the cause of the slight hum or buzzing noise reported by the pilot (ref. 9.13).

The propellant utilization and gaging system provided near-ideal propellant utilization. The unbalance at the end of the transearth injection firing was reported by the crew to be 40 lbm, decrease, which agrees well with telemetry values.

During the Apollo 9, 10, 11, and 12 missions, the service propulsion system mixture ratio was less than expected, based on static firing data. The predicted flight mixture ratio for this mission was based on previous flight data to more closely simulate the expected mixture ratio. To achieve the predicted mixture ratio at the end of the mission, the majority of the mission would have to be flown with the propellant utilization valve in the increase position. Consequently, the propellant utilization valve was in the increase position at launch.

Figure 7-2 shows the variance in fuel and oxidizer remaining at any instant during the lunar orbit insertion and transearth injection firings, as computed from the telemetry data, and the propellant utilization valve movements made by the crew. The preflight expected values and propellant utilization movements are also shown. The service propulsion system propellant usage for the mission is discussed in section 7.10.1.

7.9 ENVIRONMENTAL CONTROL AND CREW STATION

The environmental control system performed satisfactorily and provided a comfortable environment for the crew and adequate thermal control of the spacecraft equipment. The crew station equipment also satisfactorily supported the flight.

The environmental control system was used in conjunction with the cryogenic oxygen system to demonstrate the capability of providing oxygen at high flow rates such as those that will be required during extravehicular operations on future missions. A modified hatch overboard dump nozzle with a calibrated orifice was used to obtain the desired flow rate. The emergency cabin pressure regulator maintained the cabin pressure at

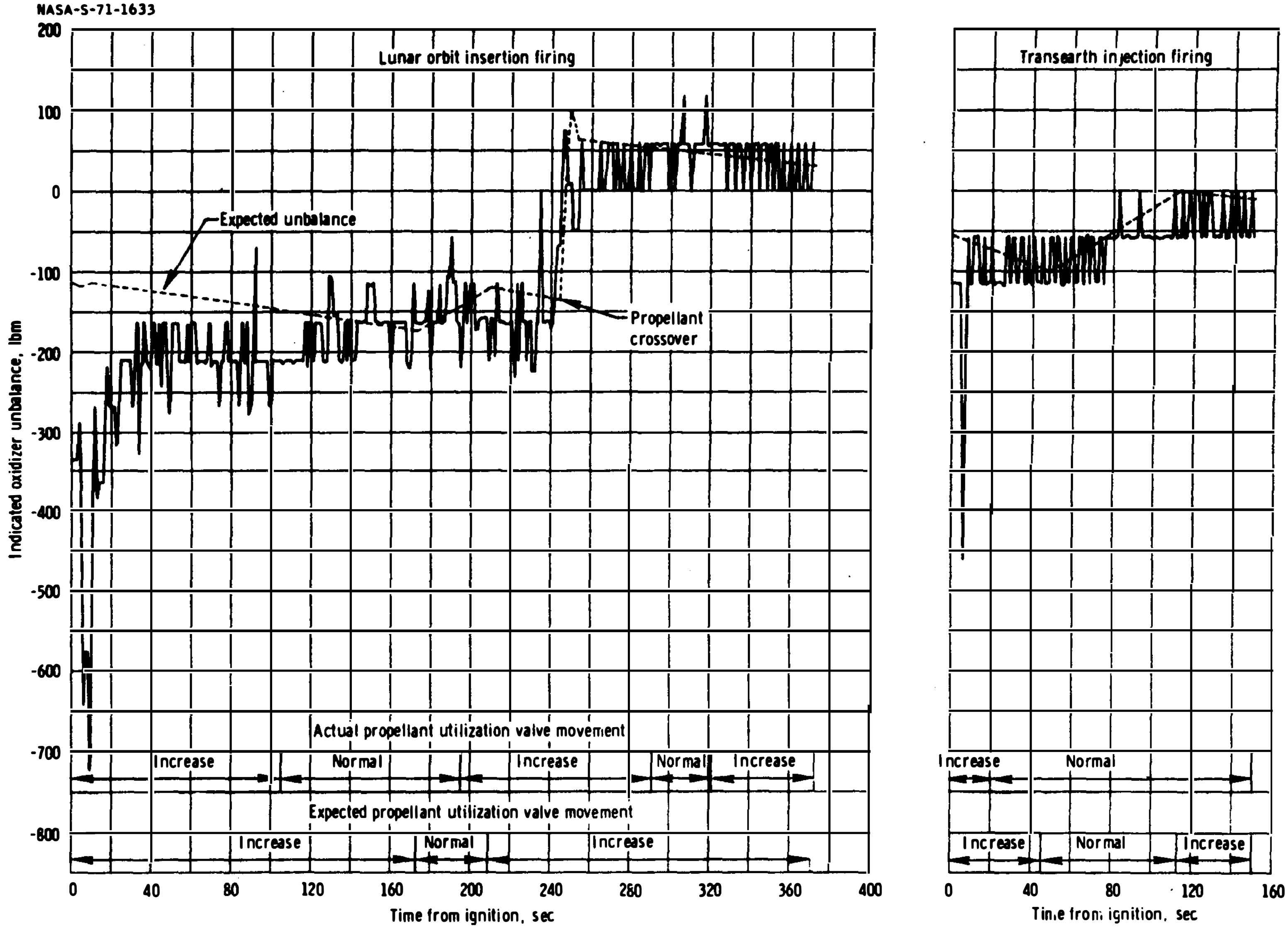


Figure 7-2.- Oxidizer unbalance during lunar orbit insertion and transearth injection firings.

approximately 4.45 psia. The test, scheduled to last 2-1/2 hours, was terminated after 70 minutes when the 100-psi oxygen manifold pressure decayed to about 10 psi. This was caused by opening of the urine overboard dump valve which caused an oxygen demand in excess of that which the oxygen restrictors were capable of providing. However, sufficient data were obtained during the test to determine the high-flow capability of the cryogenic oxygen system. (Also see section 7.3.)

Inflight cabin pressure decay measurements were made for the first time during most of the crew sleep periods to determine more precisely the cabin leakage during flight. Preliminary estimates indicate that the flight leakage was approximately 0.03 lb/hr. This leak rate is within design limits.

Partial repressurization of the oxygen storage bottles was required three times in addition to the normal repressurizations during the mission. This problem is discussed in section 14.1.8.

The crew reported several instances of urine dump nozzle blockage. Apparently the dump nozzle was clogged with frozen urine particles. The blockage cleared in all instances when the spacecraft was oriented so that the nozzle was in the sun. This anomaly is discussed further in section 14.1.3.

Intermittent communications dropouts were experienced by the Commander at 29 hours. The problem was corrected when the Commander's constant wear garment electrical adapter was replaced. The anomaly is discussed further in section 14.3.4.

A vacuum cleaner assembly and cabin fan filter, used for the first time, along with the normal decontamination procedures eliminated practically all of the objectionable dust such as that present after the Apollo 12 lunar docking. The fans were operated for approximately 4 hours after lunar docking.

Sodium nitrate was added to the water buffer ampules to reduce system corrosion. This addition also allowed a reduction in the concentration of chlorine in the chlorine ampules. No objectionable taste was noted in the water. The crew reported some difficulty in inserting the buffer ampules into the injector. The ampules and injector are being tested to establish the cause of the problem. The crew also indicated that the food preparation unit leaked slightly after dispensing hot water. This problem is discussed further in section 14.1.7.

7.10 CONSUMABLES

The command and service module consumable usage during the Apollo 14 mission was well within the red line limits and, in all systems, differed no more than 5 percent from the predicted limits.

7.10.1 Service Propulsion Propellant

Service propulsion propellant loadings and consumption values are listed in the following table. The loadings were calculated from gaging system readings and measured densities prior to lift-off.

Condition	Propellant, lb		
	Fuel	Oxidizer	Total
Loaded	15 695.2	25 061	40 756.2
Consumed	14 953.2	23 900	38 853.2
Remaining at command module/ service module separation	742	1 161	1 903
Usable at command module/ service module separation	596	866	1 462

7.10.2 Reaction Control System Propellants

<u>Service module</u>.- The propellant utilization and loading data for the service module reaction control system were as shown in the following table. Consumption was calculated from telemetered helium tank pressure histories and were based on pressure, volume, and temperature relationships.

Condition	Propellant, lb		
	Fuel	Oxidizer	Total
Loaded			
Quad A	110.1	225.3	335.4
Quad B	109.9	225.2	335.1
Quad C	110.4	226.5	336.9
Quad D	109.7	223.5	333.2
Total	440.1	900.5	1340.6
[a]Usable loaded			1233
Consumed	250	476	726
Remaining at command module/ service module separation			507

[a]Usable loaded propellant is the amount loaded minus the amount trapped and with corrections made for gaging errors.

Command module.- The loading and utilization of command module reaction control system propellant was as follows. Consumption was calculated from pressure, volume and temperature relationships.

Condition	Propellant, lb		
	Fuel	Oxidizer	Total
Loaded			
System 1	44.3	78.6	122.9
System 2	44.5	78.1	122.6
Total	88.8	156.7	245.5
[a]Usable loaded			210.0
Consumed			
System 1			[b]41
System 2			4
Total			45

[a]Usable loaded propellant is the amount loaded minus the amount trapped and with corrections made for gaging errors.
[b]Estimated quantity based on helium source pressure profile during entry.

7.10.3 Cryogenics

The total cryogenic hydrogen and oxygen quantities available at lift-off and consumed were as follows. Consumption values were based on quantity data transmitted by telemetry.

Condition	Hydrogen, lb		Oxygen, lb	
	Actual	Planned	Actual	Planned
Available at lift-off				
Tank 1	26.97		320.2	
Tank 2	26.55		318.9	
Tank 3	-		197.2	
Total	53.52	[a]53.52	836.3	[a]836.3
Consumed				
Tank 1	19.12		119.3	
Tank 2	19.14		113.8	
Tank 3	-		163.4	
Total	38.26	38.62	396.5	412.1
Remaining at command module/ service module separation				
Tank 1	7.85	7.87	200.9	204.2
Tank 2	7.41	7.03	205.1	195.2
Tank 3	-	-	33.8	24.8
Total	15.26	14.90	439.8	424.2

[a]Updated to lift-off values.

7.10.4 Water

The water quantities loaded, produced, and expelled during the mission are shown in the following table.

Condition	Quantity, lb
Loaded (at lift-off)	
Potable water tank	28.5
Waste water tank	32.4
Produced inflight	
Fuel cells	342.3
Lithium hydroxide reaction	21.0
Metabolic	21.0
Dumped overboard	
Waste tank dumping	236.9
Urine and flushing	133.2
Evaporated up to command module/ service module separation	9.0
Remaining onboard at command module/ service module separation	
Potable water tank	29.7
Waste water tank	36.4

8.0 LUNAR MODULE PERFORMANCE

8.1 STRUCTURAL AND MECHANICAL SYSTEMS

Lunar module structural loads were within design values for all phases of the mission. The structural assessment was based on guidance and control data, cabin pressure measurements, command module acceleration data, photographs, and crew comments.

Based on measured command module accelerations and on simulations using actual launch wind data, lunar module loads were determined to be within structural limits during earth launch and translunar injection. The sequence films from the onboard camera showed no evidence of structural oscillations during lunar touchdown, and crew comments agree with this assessment.

Landing on the lunar surface occurred with estimated landing velocities of 3.1 ft/sec vertical, 1.7 ft/sec in the plus-Y footpad direction, and 1.7 ft/sec in the plus-Z footpad direction. The spacecraft rates and attitude at touchdown are shown in figure 8-1. The minus-Y footpad apparently touched first, followed by the minus-Z footpad approximately 0.4 second later. The plus-Y and plus-Z footpads followed within 2 seconds and the vehicle came to rest with attitudes of 1.8 degrees pitch down, 6.9 degrees roll to the right and 1.4 degrees yaw to the left of west. Very little, if any, of the vehicle attitude was due to landing gear stroking. The final rest attitude of approximately 7 degrees was due almost entirely to local undulations at the landing point (fig. 8-2). From a time history of the descent engine chamber pressure, it appears that descent engine shutdown was initiated after first footpad contact but before plus-Y footpad contact. The chamber pressure was in a state of decay at 108:15:11, and all vehicle motion had ceased 1.6 seconds later.

Flight data from the guidance and propulsion systems were used in performing engineering simulations of the touchdown phase. As in Apollo 11 and Apollo 12, these simulations and photographs indicate that landing gear stroking was minimal if it occurred at all. Photographs also indicate no significant damage to the landing gear thermal insulation.

Sixteen-millimeter films taken from the command module prior to lunar-orbit docking support a visual observation by the crew that a strip of material about 4 feet long was hanging from the ascent stage base heat shield area. The base heat shield area is designed to protect the ascent stage from the pressure and thermal environment resulting from ascent engine plume impingement during staging. The absence

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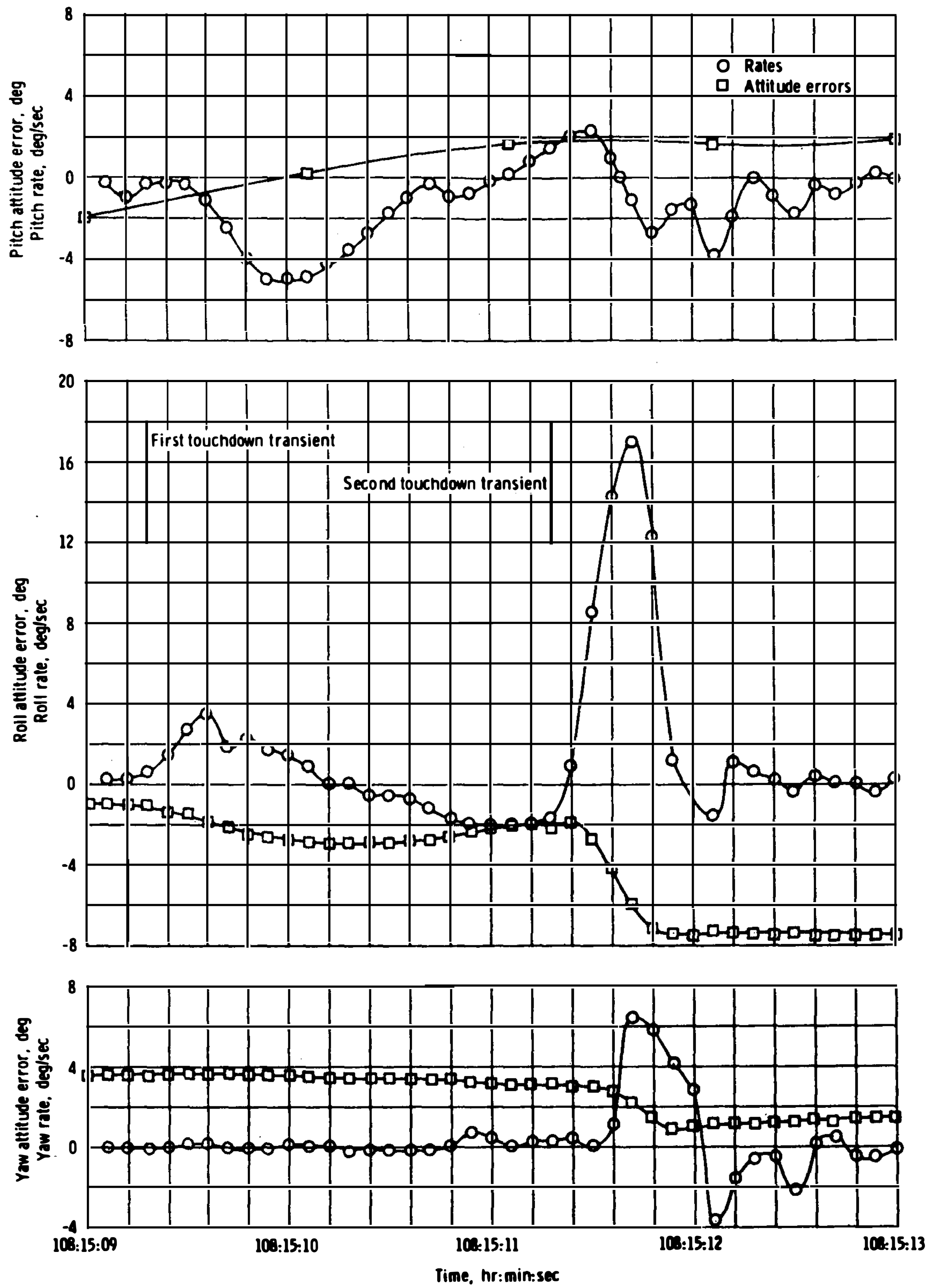

Figure 8-1.- Attitude errors and rates during lunar landing sequence.

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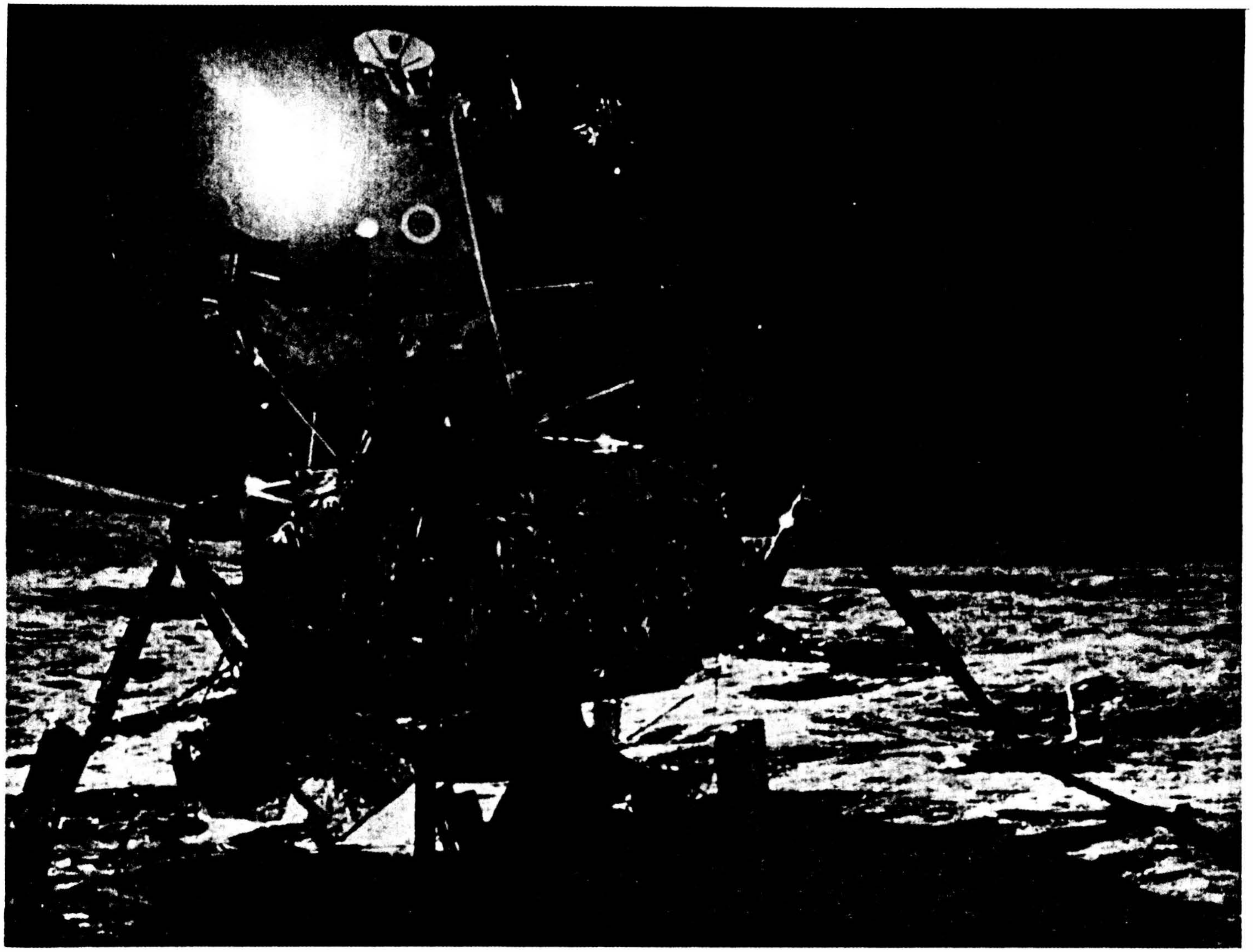

Figure 8-2.- Lunar module landing site.

of abnormal thermal responses in the ascent stage indicates that the heat shield was fully effective. Similar conditions have occurred during qualification tests whereby one or more layers of the heat shield material have become unattached. In these cases, the thermal effectiveness of the heat shield was not reduced.

8.2 ELECTRICAL POWER

The electrical power distribution system and battery performance was satisfactory with one exception, the ascent battery 5 open-circuit voltage decayed from 37.0 volts at launch to 36.7 volts at housekeeping, but with no effect on operational performance. All power switchovers were accomplished as required, and parallel operation of the descent and ascent batteries was within acceptable limits. The dc bus voltage was maintained above 29.0 volts, and maximum observed current was 73 amperes during powered descent initiation.

The battery energy usage throughout the lunar module flight is given in section 8.11.6. The ascent battery 5 open-circuit low voltage is discussed in section 14.2.1.

8.3 COMMUNICATIONS EQUIPMENT

S-band steerable antenna operation prior to lunar landing was intermittent. Although antenna operation during revolution 13 was nominal, acquisition and/or tracking problems were experienced during revolutions 11 and 12. Acquisition was attempted but a signal was not acquired during the first 3 minutes after ground acquisition of signal on revolution 14. Because of this, the omnidirectional antennas were used for lunar landing. The steerable antenna was used for the ascent and rendezvous phase and the antenna performed normally. The problems with the steerable antenna are discussed in section 14.2.3.

Prior to the first extravehicular period, difficulty was experienced when configuring the communication system for extravehicular activity because of an open audio-center circuit breaker. Extravehicular communications were normal after the circuit breaker was closed.

During the latter part of the first extravehicular period, the television resolution decreased. The symptoms of the problem were indicative of an overheated focus coil current regulator. This condition, while not causing a complete failure of the camera, resulted in defocusing of the electron readout beam in the television tube and, consequently, a degradation of resolution. The high-temperature condition was caused by operating the camera for about 1 hour and 20 minutes while it was within the thermal environment of the closed modular equipment stowage assembly. The camera was turned off between the extravehicular periods to allow cooling. Picture resolution during the second extravehicular activity was satisfactory.

The VHF system performance was poor from prior to lunar lift-off through terminal phase initiation. This problem is discussed in detail in sections 7.4 and 14.1.4.

8.4 RADAR

The landing radar self-test was performed at 105 hours 44 minutes, and the radar was turned on for the powered descent about 2 hours later. Four minutes fifty seconds prior to powered descent initiation, the radar changed from high- to low-scale. At that time, the orbital altitude of the lunar module was about 10.9 miles[a]. This condition prevented acquisition of ranging signals at slant ranges greater than 3500 feet, and velocity signals at altitudes greater than about 4600 feet. The radar was returned to high-scale by recycling the circuit breaker. A detailed discussion of this problem is given in section 14.2.4. Range and velocity performance from a slant range of about 25 000 feet to touchdown is shown in figure 14-22. There were no zero Doppler dropouts and no evidence of radar lockup resulting from particles scattered by the engine exhaust plume during lunar landing.

Rendezvous radar performance was nominal in all respects, including self-tests, checkout, rendezvous and lunar surface tracking, and temperature.

8.5 INSTRUMENTATION

The instrumentation system performed normally throughout the flight with the exception of three of the four ascent helium tank pressure measurements (two primary and two redundant). Coincident with propulsion system pressurization, these measurements exhibited negative shifts of up to 4 percent. The largest shifts were in the redundant measurements. These transducer shifts were caused by the shock induced by the pyrotechnically operated isolation valves. Since these measurements are used to monitor for leaks prior to propulsion system pressurization, a

[a]Referenced to landing site elevation.

shift in these measurements at the time of system pressurization will not affect future missions. (See appendix A, section A.2.3, for a description of changes made subsequent to Apollo 13.)

8.6 GUIDANCE, NAVIGATION, AND CONTROL

At approximately 102 hours, the primary guidance system was turned on, the computer digital clock was initialized, and the platform was aligned to the command module platform. Table 8-I is a summary of the primary guidance platform alignment data. The abort guidance system was turned on at 102 hours 40 minutes and the attitude reference aligned to the lunar module platform. Table 8-II is a summary of inertial measurement unit component errors measured prior to launch and in flight. The abort guidance system was aligned to the primary guidance system six times, but data were available for only five, and are shown in table 8-III. Also shown in table 8-III are data from the independent alignment of the abort system performed in preparation for lunar lift-off. The abort guidance system had been aligned to the gravity vector and an azimuth angle supplied by the ground. Twenty-seven minutes later, just before lift-off, the abort system compared well with the primary system which had been inertially aligned to the predicted local vertical orientation for lift-off.

The performance of the abort sensor assembly of the abort guidance system was not as good as on previous missions but was within allowable limits. The accelerometers exhibited stable performance, but the Z-axis gyro drift rate change of 1.2 degrees per hour from the prelaunch value was about 30 percent greater than the expected shift. The expected and the actual shifts between preflight values and the first inflight calibration, and shifts between subsequent inflight calibrations are shown in table 8-IV.

Table 8-V is a sequence of events prior to and during the powered descent to the lunar surface. A command to abort using the descent engine was detected at a computer input channel at 104:16:07 (but was not observed at other telemetry points) although the crew had not depressed the abort switch on the panel. The crew executed a procedure using the engine stop switch and the abort switch which isolated the failure to the abort switch. Subsequently, the command reappeared three more times; each time, the command was removed by tapping on the panel near the abort switch. (For a discussion of the probable cause of this failure, see section 14.2.2.)

If the abort command is present after starting the powered descent programs, the computer automatically switches to the abort programs and the lunar module is guided to an abort orbit. To avoid the possibility

TABLE 8-I.- LUNAR MODULE PLATFORM ALIGNMENT SUMMARY

Time, hr:min	Type alignment	Alignment mode		Telescope detent[c]/star used	Star angle difference, deg	Gyro torquing angle, deg			Gyro drift, meru		
		Option[a]	Technique[b]			X	Y	Z	X	Y	Z
102:58		Docked alignment				0.009	0.029	-0.052	-0.5	-1.5	-2.8
105:09	P52	3	NA	2/22; 2/16	0.04	0.030	-0.038	0.028	-	-	-
105:27	P52	3	NA	-- --	-	0.097	0.062	0.013	-1.5	2.0	-0.6
109:17	P57	3	1	NA NA	0.03	-0.016	0.015	-0.113	-	-	-
109:46	P57	3	2	2/31; 6/00	0.02	-0.041	0.003	-0.054	1.0	-0.1	-1.4
110:05	P57	3	2	2/26; 6/00	-0.07	0.018	0.047	-0.121	-	-	-
129:56	P57	4	3	-- --	0.01	0.044	0.069	-0.46	-	-	-
141:53	P57	4	3	-- --	0.02	0.119	0.135	-0.349	-0.7	-0.8	-1.9

[a]1 - Preferred; 2 - Nominal; 3 - REFSMMAT; 4 - Landing site.

[b]0 - Anytime; 1 - REFSMMAT plus g; 2 - Two bodies; 3 - One body plus g.

[c]1 - Left front; 2 - Front; 3 - Right front; 4 - Right rear; 5 - Rear; 6 - Left rear.

Star names:

00 Pollux
16 Procyon
22 Regulus
26 Spica
31 Arcturus

TABLE 8-II.- INERTIAL COMPONENT HISTORY - LUNAR MODULE

(a) Accelerometers

Error	Sample mean	Standard deviation	Number of samples	Countdown value	Flight load	Inflight performance		
						Power-up to landing	Surface power-up to lift-off	Lift-off to rendezvous
X - Scale factor error, ppm	-895	36	6	-922	-950	-	-	-
Bias, cm/sec^2	1.27	0.05	6	1.26	1.30	1.27	1.38	1.36
Y - Scale factor error, ppm	-1678	79	9	-1772	-1860	-	-	-
Bias, cm/sec^2	1.63	0.03	9	1.61	1.65	1.62	1.74	1.71
Z - Scale factor error, ppm	-637	25	6	-643	-670	-	-	-
Bias, cm/sec^2	1.39	0.02	6	1.41	1.39	1.35	1.46	1.45

(b) Gyroscopes

Error	Sample mean	Standard deviation	Number of samples	Countdown value	Flight load	Inflight performance
X - Null bias drift, meru	0.8	0.4	6	0.0	0.9	-1.9
Acceleration drift, spin reference axis, meru/g	0.2	0.8	6	1.1	0	-
Acceleration drift, input axis, meru/g	4.0	2.8	6	2.9	3.0	-
Y - Null bias drift, meru	-2.8	0.6	6	-3.6	-2.7	0.3
Acceleration drift, spin reference axis, meru/g	3.0	1.3	6	4.5	3.0	-
Acceleration drift, input axis, meru/g	-9.6	4.0	12	-7.5	-12.0	-
Z - Null bias drift, meru	-1.1	0.9	6	-1.1	-0.3	-0.5
Acceleration drift, spin reference axis, meru/g	4.5	0.4	6	4.5	5.0	-
Acceleration drift, input axis, meru/g	5.8	1.4	6	7.2	6.0	-

TABLE 8-III.- GUIDANCE SYSTEMS ALIGNMENT COMPARISON

Time of alignment	Primary minus abort system Alignment error (degrees)		
	X	Y	Z
103:54:44.99	0.000	0.003	0.014
104:04:45.9	0.061	0.030	0.002
104:34:45.2	0.000	0.007	0.003
109:28:36	-0.002	0.034	0.000
141:15:25.2	0.000	0.002	0.001
[a]141:45:29.2	0.010	0.003	0.018

[a]Systems aligned independently. Actual time of abort guidance system alignment was 141:18:35.2.

TABLE 8-IV.- ABORT GUIDANCE SYSTEM CALIBRATION COMPARISONS

Calibrations	Three-sigma capability estimate	Actual gyro drift rate, deg/hr		
		X axis	Y axis	Z axis
First inflight minus pre-installation	±0.91	0.08	-0.07	-1.2
Second inflight minus first inflight	±0.63	-0.01	0.23	0.26
First surface minus second inflight	±0.56	-0.02	-0.08	-0.43
Second surface minus first surface	±0.55	0.0	-0.08	-0.21

TABLE 8-V.- SEQUENCE OF EVENTS DURING POWERED DESCENT

Elapsed time from lift-off, hr:min:sec	Time from ignition, min:sec	Event
107:51:18.66	-11:07.86	Landing radar on
107:52:46.66	-9:39.86	False data good indications from landing radar
107:57:34.66	-4:51.86	Landing radar switched to low scale
107:58:13.80	-4:12.72	Start loading abort bit work-around routine
108:02:19.12	-0:07.40	Ullage on
108:02:26.52	0:00.00	Ignition
108:02:53.80	+0:27.28	Manual throttle-up to full throttle position
108:04:49.80	+2:23.28	Manual target update (N69)
108:08:47.68	+6:21.16	Throttle down
108:08:50.66	+6:24.14	Landing radar to high scale (circuit breaker cycle)
108:09:10.66	+6:44.14	Landing radar velocity data good
108:09:12.66	+6:46.14	Landing radar range data good
108:09:35.80	+7:09.28	Enable altitude updates
108:11:09.80	+8:43.28	Select approach phase program (P64)
108:11:10.42	+8:43.90	Start pitch over
108:11:51.60	+9:25.08	Landing radar redesignation enable
108:11:52.66	+9:26.14	Landing radar antenna to position 2
108:13:07.86	+10:41.34	Select attitude hold mode
108:13:09.80	+10:43.28	Select landing phase program (P66)
108:15:09.30	+12:42.78	Left pad touchdown
108:15:11.13	+12:44.61	Engine shutdown (decreasing thrust chamber pressure)
108:15:11.40	+12:44.88	Right, forward,and aft pad touchdown

of an unwanted abort, a work-around procedure was developed by ground personnel and was relayed to the crew for manual entry into the lunar module computer. Part one of the four-part procedure was entered into the computer just after the final attitude maneuver for powered descent. The remainder was accomplished after the increase to the full-throttle position. Part one consisted of loading the abort stage program number into the mode register in the erasable memory which is used to monitor the program number displayed to the crew. This did not cause the active program to change, but it did inhibit the computer from checking the abort command status bit. At the same time, it inhibited the automatic command to full-throttle position, automatic guidance steering, and it affected the processing of the landing radar data. Therefore, in order to reestablish the desired configuration for descent, the increase to full-throttle position was accomplished manually and then the second, third, and fourth parts of the procedure were entered into the computer. In order, they accomplished:

a. Setting a status bit to inform the descent program that throttle-up had occurred and to re-enable guidance steering

b. Resetting a status bit which disabled the abort programs

c. Replacing the active program number back into the mode register so that landing radar data would be processed properly after landing radar lock-on

The abort capability of the primary guidance system was lost by use of this procedure. Therefore, it would have been necessary to use the abort guidance system if an abort situation had arisen.

Prior to powered descent maneuver ignition, the landing radar scale factor switched to low, which prevented acquisition of data through the first 400 seconds of descent. (For further discussion, refer to section 14.2.4.) The crew cycled the radar circuit breaker, which reset scaling to the high scale, and landing radar lock-on occurred at 22 486 feet. Figure 14-22 is a plot of slant range as measured by landing radar and as computed from primary guidance system state vectors. Figure 8-3 is a plot of altitudes computed by the abort and primary guidance systems and shows a 3400-foot update to the abort guidance system at the 12 000-foot altitude.

Throttle oscillations that had been noted on previous flights were not detected during the descent although some oscillation in the automatic throttle command was detected after descent engine manual shutdown. The reaction control system propellant consumption during the braking phase and approach phase programs was approximately half that seen on previous missions. Further discussion of these two areas will be provided in a supplement to this report.

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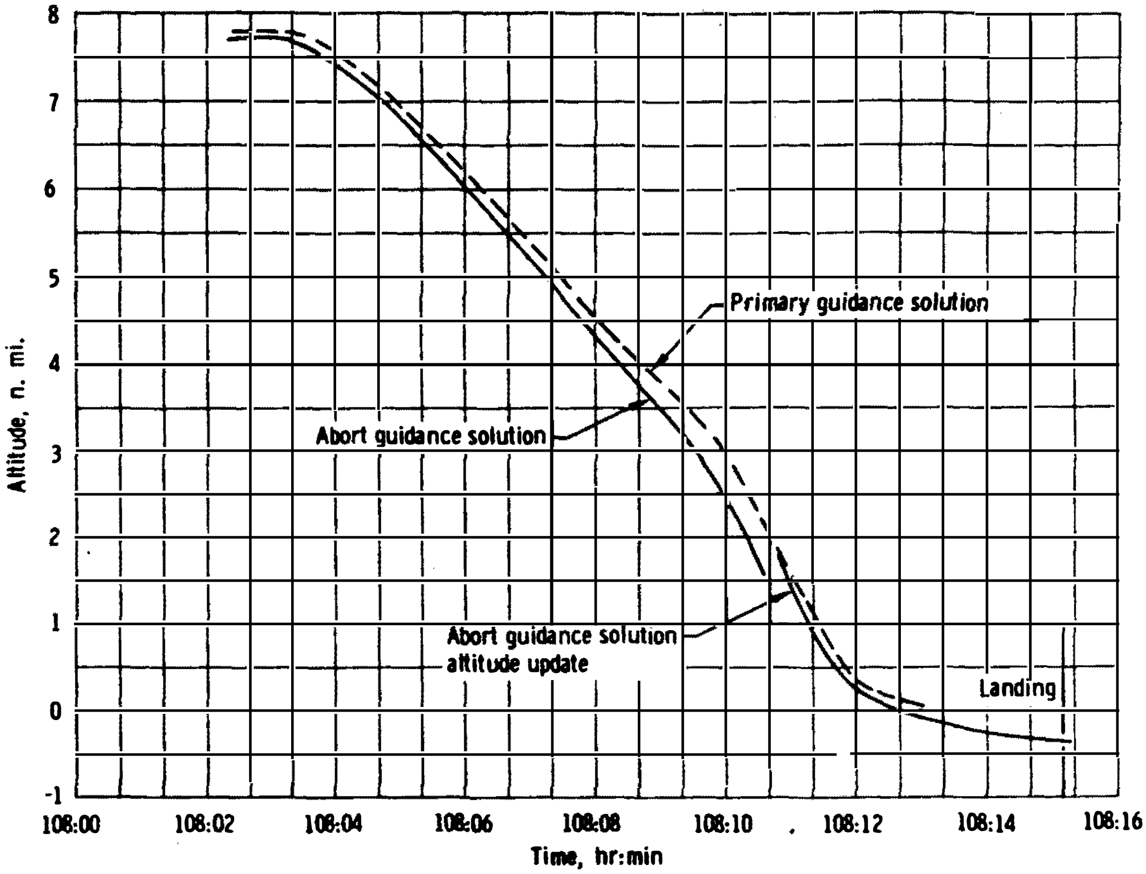


Figure 8-3.- Comparison of altitudes computed by abort and primary guidance systems during descent.

While on the lunar surface, a test was performed to compute gravity using primary guidance system accelerometer data. The value of gravity was determined to be 162.65 cm/sec^2.

Performance during the ascent from the lunar surface was nominal. The primary and abort systems and the powered flight processor data compared well throughout ascent. The ascent program in the onboard computer does not include targeting for a specific cutoff position vector; therefore, a vernier adjustment maneuver of 10.3 ft/sec was performed to satisfy the phasing conditions for a direct rendezvous with the command and service module.

Performance throughout rendezvous, docking, and the deorbit maneuver was also nominal. The velocity change imparted to the lunar module at jettison was minus 1.94, minus 0.05, and minus 0.10 ft/sec in the X, Y, and Z axes, respectively.

The abort guidance system functioned properly until the braking phase of the rendezvous with the command and service module when a failure caused the system to be down-moded to the standby mode and resulted in the loss of this system for the remainder of the mission. Another anomaly reported was a crack in the glass window of the address register on the data entry and display assembly. These anomalies are discussed in sections 14.2.5 and 14.2.6, respectively.

8.7 DESCENT PROPULSION

The descent propulsion system operation was satisfactory. The engine transients and throttle response were normal.

8.7.1 Inflight Performance

The duration of the powered descent firing was 764.6 seconds. A manual throttle-up to the full throttle position was accomplished approximately 26 seconds after the engine-on command. The throttle-down to 57 percent occurred 381 seconds after ignition, about 14 seconds earlier than predicted but within expected tolerances. Three seconds of the 14 are attributed to the landing site offset to correct for the downrange error in actual trajectory, and the remaining 11 seconds to a thrust increase of approximately 80 pounds at the full-throttle position.

8.7.2 System Pressurization

During the period from lift-off to 104 hours, the oxidizer tank ullage pressure decayed from 111 to 66 psia and the fuel tank ullage pressure decreased from 138 to 111 psia. These decays resulted from helium absorption into the propellants and were within the expected range.

The supercritical helium system performed as anticipated. The system pressure rise rates were 8.0 psi/hour on the ground and 6.2 psi/hour during translunar coast, which compare favorably with the preflight predicted values of 8.1 psi/hour and 6.6 psi/hr, respectively. During powered descent, the supercritical helium system pressure profile was well within the nominal ±3-sigma pressure band, even though the pressure at ignition was about 50 psi lower than anticipated.

8.7.3 Gaging System Performance

The gaging system performance was satisfactory throughout the mission. The low-level quantity light came on approximately 711 seconds after ignition, and was most probably triggered by the point sensor in oxidizer tank 2. Engine cutoff occurred 53 seconds after the low-level signal, indicating a remaining firing-time-to-depletion of 68 seconds. Using probe data to calculate remaining firing time gave approximately 70 seconds remaining. This is within the accuracy associated with the propellant quantity gaging system.

The new propellant slosh baffles installed on Apollo 14 appear to be effective. The propellant slosh levels present on Apollo 11 and 12 were not observed in the special high-sample-rate gaging system data of this mission.

8.8 ASCENT PROPULSION

The ascent propulsion system duty cycle consisted of two firings — the lunar ascent and the terminal phase initiation. Performance of the system for both firings was satisfactory. Table 8-VI is a summary of

TABLE 8-VI.- STEADY-STATE PERFORMANCE DURING ASCENT

Parameter	10 seconds after ignition		400 seconds after ignition	
	Predicted[a]	Measured[b]	Predicted[a]	Measured[b]
Regulator outlet pressure, psia	184	182	184	181
Oxidizer bulk temperature, °F	70.0	69.4	69.0	69.4
Fuel bulk temperature, °F	70.0	69.8	69.8	69.4
Oxidizer interface pressure, psia	170.5	168	169.7	167
Fuel interface pressure, psia	170.4	169	169.7	167
Engine chamber pressure, psia	123.4	121	123.2	120
Mixture ratio	1.607	-	1.598	-
Thrust, lb	3502.	-	3468.	-
Specific impulse, sec	310.3	-	309.9	-

[a]Preflight prediction based on acceptance test data and assuming nominal system performance.
[b]Actual flight data with no adjustments.

actual and predicted performance during the ascent maneuver. The duration of engine firing for lunar ascent was approximately 432 seconds, and for terminal phase initiation, 3 to 4 seconds. A more precise estimate of the terminal phase initiation firing time is not available because the firing occurred behind the moon and no telemetry data were received. System pressures were as expected both before and after the terminal phase initiation maneuver and crew reports indicate that the maneuver was nominal.

No oscillations were noted during flight in either helium regulator outlet pressure measurement. Oscillations in the outlet pressure of 6 to 19 psi have been noted in previous flight data. Also, oscillations of a similar nature and approximately twice that magnitude were noted during preflight checkout of the ascent propulsion system class I secondary helium regulator. However, during flight, control is maintained, normally, by the class I primary regulator.

8.9 ENVIRONMENTAL CONTROL AND CREW STATION

Performance of the environmental control system was satisfactory throughout the mission. Glycol pump noise, a nuisance experienced on previous missions, was reduced below the annoyance level by a muffler on the pump system. Although the water separator speed was higher than expected much of the time, the separator removed water adequately and there were no problems with water condensation or cabin humidity.

Because of water in the suit loop on Apollo 12 (ref. 1), a flow restrictor had been installed in the primary lithium hydroxide cartridges to reduce the gas flow in the suit loop and, thereby, reduce water separator speed below 3600 rpm. (Separator speed is a function of the water mass to be separated and the gas flow.) However, the water separator speed was above 3600 rpm while the suit was operated in the cabin mode (helmets and gloves removed). The high speed when in the cabin mode resulted from low moisture inputs from the crew (approximately 0.14 lb/hr) and a high gas flow caused by low back pressure which, in turn, developed from a low pressure drop across the suit.

During preparations for the first extravehicular activity, the transfer hose on the urine collection transfer assembly was kinked. The kink was eliminated by moving the hose to a different position.

The crew repeatedly had trouble getting the lunar module forward window shades to remain in their retainers. The shades had been processed to reduce the curl and prevent cracking, a problem experienced on previous flights. In reducing the curl, the diameter of the rolled shades was increased so that the shades would not fit securely in the retainers. For

Apollo 15, the shades will be fabricated to permit them to be rolled small enough to be held securely by the retainers.

The interim stowage assembly could not be secured at all times because the straps could not be drawn tight enough to hold. This problem resulted from stretch in the fabric and in the sewing tolerances. In the future, more emphasis will be placed upon manufacturing fit checks and crew compartment fit checks to assure that the problem does not recur.

8.10 EXTRAVEHICULAR MOBILITY UNIT

Performance of the extravehicular mobility unit was very good during the entire lunar stay. Oxygen, feedwater, and power consumption (section 8.11.7) allowed each extravehicular period to be extended approximately 30 minutes with no depletion of contingency reserves. Comfortable temperatures were maintained using the diverter valve in the minimum position throughout most of both extravehicular activities.

Preparations for the first extravehicular activity proceeded on schedule with few exceptions. The delay in starting the first extravehicular activity occurred while the portable life support system power was on, resulting in battery power being the limiting consumable in determining the extravehicular stay time.

Oxygen consumption of the Lunar Module Pilot during the first extravehicular activity was one-third higher than that of the Commander. Telemetry data during the Lunar Module Pilot's suit integrity check indicated a pressure decay rate of approximately 0.27 psi/min; a rate of 0.30 psi/min is allowable. In preparation for the second extravehicular activity, special attention was given to cleaning and relubricating the Lunar Module Pilot's pressure garment assembly neck and wrist ring seals in an effort to lower the extravehicular mobility unit leak rate. A 0.22 psi/min pressure decay rate was reported by the Lunar Module Pilot prior to the second extravehicular activity. Postflight unmanned leak rate tests on the Lunar Module Pilot's pressure garment assembly show no significant increase in leakage.

Just prior to lunar module cabin depressurization for the second extravehicular activity, the Lunar Module Pilot reported a continuous force in his right extravehicular glove wrist pulling to the left and down. A more detailed discussion is given in section 14.3.2. The extravehicular activity started and was completed without any reported difficulty with the glove.

8.11 CONSUMABLES

On the Apollo 14 mission, all lunar module consumables remained well within red line limits and were close to predicted values.

8.11.1 Descent Propulsion System

Propellant.- The quantities of descent propulsion system propellant loading in the following table were calculated from readings and measured densities prior to lift-off.

Condition	Actual quantity, lb		
	Fuel	Oxidizer	Total
Loaded	7072.8	11 344.4	18 417.2
Consumed	6812.8	10 810.4	17 623.2
Remaining at engine cutoff			
Total	260.0	534.0	794.0
Usable	228.0	400.0	628.0

Supercritical helium.- The quantities of supercritical helium were determined by computation utilizing pressure measurements and the known volume of the tank.

Condition	Quantity, lb	
	Actual	Predicted
Loaded	48.5	
Consumed	42.8	39.2 [a](40.8)
Remaining at touchdown	5.7	9.3 [a](7.7)

[a]Adjusted prediction to account for longer-than-planned firing duration.

8.11.2 Ascent Propulsion System

Propellant.- Ascent propulsion system total propellant usage was within approximately 1 percent of the predicted value. The loadings in the following table were determined from measured densities prior to launch and from weights of off-loaded propellants.

Condition	Actual quantity, lb			Predicted quantity, lb
	Fuel	Oxidizer	Total	
Loaded	2007.0	3218.2	5225.2	
Total consumed	1879.0	3014.0	4893.0	4956.0
Remaining at lunar module jettison	128.0	204.2	332.2	265.8

Helium.- The quantities of ascent propulsion system helium were determined by pressure measurements and the known volume of the tank.

Condition	Actual quantity, lb
Loaded	13.4
Consumed	8.8
Remaining at lunar module impact	4.6

8.11.3 Reaction Control System Propellant

The reaction control system propellant consumption was calculated from telemetered helium tank pressure histories using the relationships between pressure, volume, and temperature.

Condition	Actual, lb			Predicted, lb
	Fuel	Oxidizer	Total	
Loaded				
System A	108	209		
System B	108	209		
Total	216	418	634	633
Consumed to				
Docking			260	283
Impact			378	393
Remaining at lunar impact			256	240

8.11.4 Oxygen

The oxygen tank was not loaded to the nominal 2730 psia used for previous missions because of a possible hydrogen embrittlement problem with the descent stage oxygen tank. Launch pressure for the tank was an indicated 2361 psia.

Condition	Actual quantity, lb	Predicted quantity, lb
Loaded (at lift-off)		
Descent stage	42.3	
Ascent stage		
Tank 1	2.4	
Tank 2	2.4	
Total	47.1	
Consumed		
Descent stage	24.9	23.9
Ascent stage		
Tank 1	(a)	1.1
Tank 2	0	0
Total		25.0
Remaining in descent stage at lunar lift-off	17.4	18.4
Remaining at docking		
Tank 1	(a)	1.3
Tank 2	2.4	2.4
Total		3.7

[a]Consumables data are not available because the tank 1 pressure transducer malfunctioned before launch.

8.11.5 Water

In the following table, the actual quantities loaded and consumed are based on telemetered data.

Condition	Actual quantity, lb	Predicted quantity, lb
Loaded (at lift-off)		
Descent stage	255.5	
Ascent stage		
Tank 1	42.5	
Tank 2	42.5	
Total	340.5	
Consumed		
Descent stage (lunar lift-off)	200.9	190.9
Ascent stage (docking)		
Tank 1	6.0	6.2
Tank 2	5.8	6.2
Total	212.7	203.3
Ascent stage (impact)		
Tank 1	14.4	-
Tank 2	14.9	-
[a]Total	230.2	-
Remaining in descent stage at lunar lift-off	54.6	59.1
Remaining in ascent stage at impact		
Tank 1	28.1	-
Tank 2	27.6	-
Total	55.7	-

[a]Consumed during flight, both stages.

8.11.6 Electrical Power

The total battery energy usage is given in the following table. Preflight predictions versus actual usage were within 3 percent.

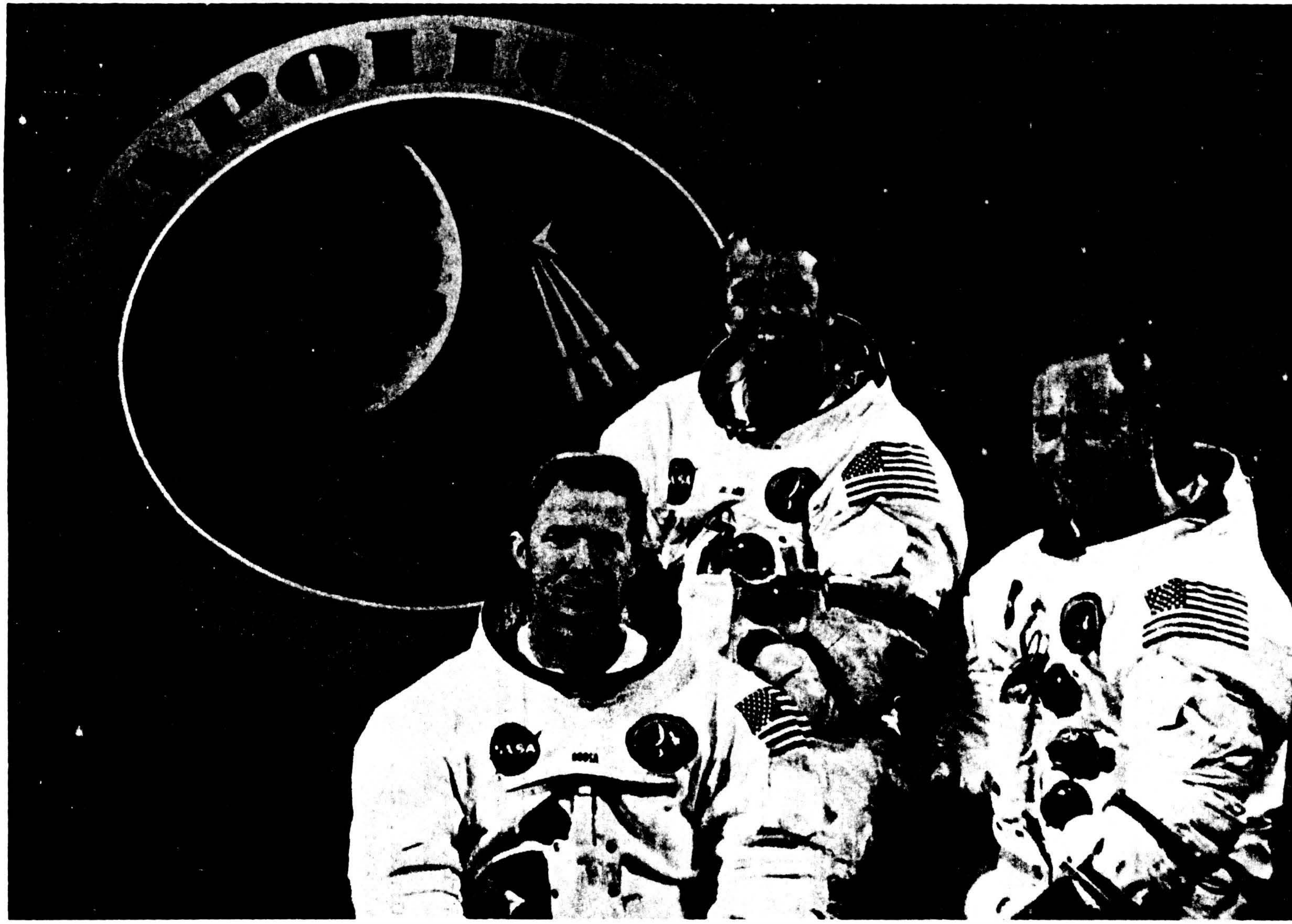
Batteries	Available power, A-h	Electrical power consumed, A-h	
		Actual	Predicted
Descent	1600	1191	1220
Ascent	592	128	125

8.11.7 Extravehicular Mobility Unit

Oxygen, feedwater and power consumption of the extravehicular mobility unit for both extravehicular periods are shown in the following table.

Condition	Commander		Lunar Module Pilot	
	Actual	Predicted	Actual	Predicted
First extravehicular activity				
Time, min	288	255	288	255
Oxygen, lb				
Loaded	1.31	1.31	1.31	1.31
Consumed	0.70	0.97	1.02	0.97
Remaining	0.61	0.34	0.29	0.34
Feedwater, lb				
Loaded	8.59	8.55	8.66	8.55
Consumed	4.85	7.08	5.71	7.08
Remaining	3.74	1.47	2.95	1.47
Power, W-h				
Initial charge	282	282	282	282
Consumed	228	223	237	223
Remaining	54	59	45	59
Second extravehicular activity				
Time, min	275	255	275	255
Oxygen, lb				
Loaded	1.26	1.31	1.26	1.31
Consumed	0.86	1.02	0.96	1.02
Remaining	0.40	0.29	0.30	0.29
Feedwater, lb				
Loaded	8.80	8.55	8.80	8.55
Consumed	[a]6.43	7.55	[a]7.13	7.55
Remaining	[a]2.37	1.0	[a]1.67	1.0
Power, W-h				
Initial charge	282	282	282	282
Consumed	225	225	222	225
Remaining	57	57	60	57

[a]Estimate based on extravehicular mobility unit source heat predictions because portable life support system feedwater weight was not taken following the second extravehicular activity.



Apollo 14 flight crew

Commander Alan B. Shepard, Jr. (center), Command Module Pilot Stuart A. Roosa (left), and Lunar Module Pilot Edgar D. Mitchell

9.0 PILOT'S REPORT

The Apollo 14 mission expanded the techniques and overcame some of the operational limitations of previous lunar landing missions. Specific differences included performing onboard cislunar navigation to simulate a return to earth with no communications, using the service propulsion system for the descent orbit maneuver, landing in the lunar highlands, extending the lunar surface excursion time and making a lunar-orbit rendezvous during the first revolution of the spacecraft. The detailed flight plan, executed in its entirety, was used as a reference for the activities of the pilots during the mission (fig. 9-1, at end of section).

9.1 TRAINING

The formal training for this crew was conducted over a time span of 20 months in general accordance with the schedules used for previous missions. The training equipment and methods were concluded to be excellent and are recommended for subsequent crews essentially unchanged. Although none of the crew members had completed actual flight experience in the Apollo program, each of the pilots felt that he was completely ready for all phases of the flight.

9.2 LAUNCH

The countdown proceeded on schedule with no problems encountered in the area of crew integration or ingress. The general condition of the crew station and displays was excellent. The crew was kept well informed of the nature of the launch delay and was apprised of launch azimuth change procedures; accordingly, that phase went smoothly. The Commander noted no visible moisture on windows 2 and 3 either prelaunch or during atmospheric flight. The proprioceptive cues reported by earlier crews were essentially unchanged during the launch of Apollo 14. No communication difficulties were noted during the launch. A very slight longitudinal oscillation occurred during second stage flight starting at 8 minutes 40 seconds and continuing through shutdown. The launch profiles flown during preflight training on the dynamic crew procedures simulator and the command module simulator were more than adequate for crew preparation.

9.3 EARTH ORBIT

This crew had placed special emphasis on suited training periods in the command module simulator for this particular phase. The spacecraft system checks and unstowage of equipment were performed slowly and precisely coincident with the process of familiarization with the weightless state. No anomalies or difficulties were noted.

The Command Module Pilot noted that, although he had heard the optics cover jettison, there was no debris, and a finite period of several minutes of dark-adaption was required to permit viewing of stars through the telescope. The extension of the docking probe is mentioned here only to indicate that it was extended on schedule, per the checklist, with no problems noted from either audio or visual cues.

9.4 TRANSLUNAR INJECTION

The delay in launch produced off-nominal monitoring parameters with the second S-IVB firing. These updates were forwarded smoothly and in a timely fashion so that all preparations for the injection were normal. Attitude control of the S-IVB was excellent and right on schedule. The ignition was on time, positive, and without roughness. The guidance parameters comparison between the command module computer and the instrumentation unit was very close. A very light vibration or buzz was noted toward the end of the powered phase, and is mentioned only to inform future crews as to a resonance reference point. The state vector conditions at cutoff were excellent and the tanks vented on schedule. The Commander and Command Module Pilot changed couch positions in accordance with the flight plan.

9.5 TRANSLUNAR FLIGHT

9.5.1 Transposition and Docking

The physical separation from the S-IVB closed two propellant isolation valves on the service module reaction control system. These were immediately reset with no problems. The entry monitor system was not used as a reference during any portion of the transposition and docking maneuver. The plus-X thrusting on separation and the initial thrusting to set up a closing velocity were performed using the event timer.

Several attempts were required before docking was successfully achieved. [Editor's note: Six contacts were made and these are referred to as six "docking attempts" in other sections of the report. The pilots considered the first two contacts to be one attempt.] The first attempt was made at a closing velocity of approximately 0.1 to 0.2 ft/sec. At contact, the capture latches did not lock with the drogue. Plus-X thrust was used to drive the probe back into the drogue, but again, capture was not achieved. All switches and circuit breakers were verified by the checklist and another docking attempt was made with a closing velocity of approximately 1.0 ft/sec. The latches again failed to capture on this pass. The procedures were verified with Houston and the docking probe switch was placed to extend, then back to retract (the talkbacks were verified gray in both positions). On the third attempt, plus-X thrust was held for approximately 4 seconds after drogue contact, but the latches failed to capture. Three prominent scratches, approximately 2 inches long and spaced 120 degrees around the drogue, were noted at this time and Houston was informed. The scratches started near the hole in the drogue and extended radially outward. The docking probe switch was placed to extend-release for 5 seconds, then back to retract; the talkbacks were verified gray in both positions. Another attempt was made using normal procedures, and again, no capture was achieved. On the fifth and final attempt, the probe was aligned in the drogue and held with plus-X thrust. The primary 1 retract switch was actuated, and approximately 4 to 5 seconds later, the talkbacks went barberpole, then gray, and the docking ring latches were actuated by the lunar module docking ring. The post-docking procedures were performed using the normal crew checklist and the locking of all twelve latches was verified.

Immediately upon lunar module ejection, a maneuver was started to view the S-IVB. As soon as the S-IVB was in sight, Houston was notified. An S-IVB yaw maneuver was then commanded in preparation for the auxiliary propulsion system evasive maneuver. Both the auxiliary propulsion system evasive maneuver and the propellant dump of the S-IVB were visually monitored. The S-IVB was stable when last viewed by the crew.

The probe and drogue were removed during the first day for examination and checkout using the crew checklist and procedures provided by the Mission Control Center. The probe functioned properly at that time.

9.5.2 Translunar Coast

A clock update was performed at approximately 55 hours to compensate for a weather hold of approximately 40 minutes during the launch countdown. This procedure was an aid to the Command Module Pilot while in lunar orbit because it eliminated the need for numerous updates to the Command Module Pilot's solo book.

9.5.3 Midcourse Correction

Two midcourse corrections were performed during the translunar coast phase. The first midcourse correction was performed at the second option point and placed the spacecraft on a hybrid trajectory. The maneuver was performed under control of the guidance and control system with residuals of plus 0.2, zero, and minus 0.1 ft/sec. The second midcourse correction was performed at the fourth option point and was targeted for a velocity change of 4.8 ft/sec. It was a service propulsion system maneuver performed under control of the guidance and control system. The residuals were plus 0.3, zero, and minus 0.1 ft/sec.

9.6 LUNAR ORBIT INSERTION

Residuals resulting from the lunar orbit insertion maneuver were plus 0.3, zero, and zero ft/sec. The firing time was within 1 second of the pad value[a]. The only unexpected item noted during this maneuver was the operation of the propellant utilization and gaging system. The preflight briefings on the system indicated that, at crossover, the unbalance meter would oscillate and then settle out in the 100 to 150 increase position. At crossover, during the actual maneuver, the unbalance meter went from its decrease position smoothly up to approximately zero. It was controlled about the zero point using the increase and normal positions of the switch.

9.7 DESCENT ORBIT INSERTION

On Apollo 14, for the first time, the descent orbit insertion maneuver was made with the service propulsion system. The command module computer indicated a 10.4- by 58.8-mile orbit after the maneuver. The Network indicated a 9.3- by 59.0-mile orbit. The firing time observed by the crew was 20.6 seconds. Pad firing time was 20.8 seconds. The maneuver was controlled by the guidance and control system with command module computer shutdown. Immediately after the descent orbit insertion maneuver, the spacecraft was oriented to an attitude from which an abort maneuver could have been performed if required, and shortly after acquisition of signal, Houston gave a "go" to stay in the low orbit. Pad firing time was the crew monitoring shutdown criteria. This technique virtually eliminated the possibility of an unacceptable overspeed.

[a]Pad values are the voice-updated parameter values used to perform a maneuver.

9.8 LUNAR MODULE CHECKOUT

The checkout of the lunar module was conducted in two phases — the first during translunar coast and the second on the day of the descent. Pressure readings, prior to entering the lunar module, indicated that the lunar module had a low leakage rate. Power transfer to the lunar module occurred at 61:41:11. The only anomaly was a slightly low voltage reading on battery 5. There were about five or six very small screws and washers floating around upon ingress. During this period, 16-mm motion pictures were made of a command module waste water dump. Some additional housekeeping and equipment transfer served to reduce the workload on descent day. Power was transferred back to the command module at 62:20:42.

The second lunar module checkout was accomplished on the same day as powered descent initiation. Two checklists, one for each pilot, were used to speed up the activation process. The Commander and the Lunar Module Pilot both suited in the command and service module prior to intravehicular transfer, but all equipment had been located the night before to assure that this would be a timely and successful process. An electrode problem with the Lunar Module Pilot's biosensors made this period full with no extra time available. The window heaters were used to clear some condensation found after ingress. The probe and drogue were installed and checked with no problem. Prior to reaction control system pressurization, the system A main shutoff valve clicked during recycle, indicating that it was probably closed at that time.

The remainder of the activation proceeded without incident until separation. Subsequent to separation, the checkout of the lunar module systems continued with only two additional problems becoming evident.

a. The S-band antenna behavior was erratic at various times when in the "auto" track mode. On two occasions, the S-band antenna circuit breaker opened without apparent reason, but functioned properly upon being reset. On at least two other occasions, the ground signal was lost unexpectedly. The antenna drove to the mechanical stop, at which time the breaker opened (as expected). An unusually loud noise associated with the antenna was noted. It was subsequently found, by observing the antenna shadow on the lunar surface, that the noise was coincident with an oscillation in both pitch and yaw. Upon one occasion, the antenna pitch position indicator dial was observed to be full-scale up, with the antenna functioning properly. This anomaly corrected itself a short time later and did not recur.

b. The other major problem, which occurred before powered descent initiation, was observed by the Mission Control Center. The crew was advised of an abort discrete being set in the lunar module guidance computer with the abort button reset. The crew did not participate significantly in solving this problem except to follow the instructions given by the Mission Control Center. The remainder of the lunar module checkout was nominal up to the point of powered descent initiation.

9.9 POWERED DESCENT

The primary guidance computer was used to select the descent program for an initial ignition algorithm check[a] about 50 minutes prior to actual ignition. The computer was also targeted for a no-ignition abort at this time. Final systems checks and switch settings were then made and the abort guidance system was initialized to the ground state vector (which had been uplinked 30 minutes prior to ignition). The anomalies present at this time included the computer abort bit problem and the S-band steerable antenna malfunction. To assure continuous communications, a decision was made to use omnidirectional antennas during powered descent.

The descent program was reselected in the primary computer at ignition minus 10 minutes and a final attitude trim was completed about 5 minutes later. The first computer entry, to inhibit the abort command, was made just after final trim. The remaining entries were made after ignition. Both the ullage and the ignition were automatic and occurred on time. The engine was throttled-up manually by the Commander 26 seconds after ignition. The throttle was returned to the idle position after the computer entries had been completed, at about 1 minute 25 seconds into the firing. The computer guidance was initialized, by manual keyboard entry, about 42 seconds after ignition. A landing point target update of 2800 feet downrange was entered manually about 2 minutes 15 seconds after ignition. The steering equations and torque-to-inertia ratio of the lunar module simulator are nearly identical to those for the actual vehicle. Therefore, the pilot's preflight training was completely adequate for the actual vehicle response exhibited during the descent phase.

The throttle recovery point occurred about 12 seconds prior to the predicted time. The altitude and velocity lights of the computer display continuously indicated that landing radar data were invalid to an altitude well below the nominal update level. A call was received from the Mission Control Center to "cycle the landing radar circuit breaker." This allowed a valid update. The lights extinguished and the computer entry was made to enable this function at an altitude of about 21 000 feet. The Commander did not evaluate manual control after throttle recovery, as planned, because the time required for the landing

[a]Verification of computer performance.

radar update precluded such action. The abort guidance system followed the primary system very closely during the period prior to landing radar update. There was, therefore, only a single altitude update to the abort system. This update was made at an altitude of 12 000 feet. There was no abnormal divergence of the abort guidance system through the remainder of the landing phase.

The landing program of the primary computer was entered 8 minutes 44 seconds after ignition and at an altitude of about 8000 feet. The vehicle pitched down, as expected, and the lunar surface was readily visible. The target landing point was recognized immediately by the Commander without reference to the computer landing point designator. The unique terrain pattern contributed to this successful recognition, but the determining factor was the high fidelity of the simulator visual display and the training time associated with the device. The first comparison of the landing point designator showed zero errors in cross range and down range. A redesignation of the target point 350 feet to the south was made at an altitude of about 2700 feet to allow a landing on what had appeared to be smoother terrain in the preflight studies of charts and maps. Several cross references between the target and the landing point designator were made until an altitude of about 2000 feet was reached, and good agreement was noted. At some altitude less than 1500 feet, two things became apparent — first, that the redesignated (south) landing point was too rough and, second, that the automatic landing was to occur short of the target.

The manual descent program was initiated at an altitude of 360 feet at a range of approximately 2200 feet short of the desired target. The lunar module was controlled to zero descent rate at an altitude of about 170 feet above the terrain. Translation maneuvers forward and to the right were made to aim for the point originally targeted. Although this area appeared to be gradually sloping, it was, in general, smoother than the ridge south of the target. The fact that no dust was noted during the translation was reassuring because it helped corroborate the primary computer altitude. Velocity on the cross pointer was about 40 ft/sec forward at manual takeover and this was gradually reduced to near-zero over the landing point. A cross velocity of about 6 ft/sec north was also initiated and gradually reduced to zero over the landing point. The cross pointers (primary guidance) were steady and their indications were in good agreement with visual reference to the ground. Control of the vehicle in primary guidance attitude-hold mode and rate-of-descent mode was excellent at all times. The use of the lunar landing training vehicle and the lunar module simulator had more than adequately equipped the pilot for his task. It was relatively easy to pick out an exact landing spot and fly to it with precise control.

Blowing surface dust was first noted at an altitude of 110 feet, but this was not a detrimental factor. The dust appeared to be less than

6 inches in depth and rocks were readily visible through it. A final descent from 100 feet was made at a descent rate of 3 ft/sec, with a deliberate forward velocity of about 1 ft/sec and, essentially, zero cross range velocity. The forward velocity was maintained until touchdown to preclude backing into any small craters. To provide a soft landing, a delay of about 2 seconds was allowed between acquisition of the contact lights and activation of the engine stop button. Touchdown occurred at shutdown with some small dust-blowing action continuing during engine thrust tailoff or decay. The landing forces were extremely light and the vehicle came to rest within 1 degree of zero in pitch and yaw attitudes, and with a 7-degree right roll attitude (northeast tilt). (Refer to figure 8-2.)

Some lineations were evident in the area of thrust impingement on the surface along the final track and in the landing area. As might be expected, these areas are generally coincident with those in which blowing surface dust was noted at low altitudes. The area in the vicinity of the descent engine after touchdown appeared to have been cratered only to a depth of about 6 inches and, as photographs show, only in a small, well-defined area.

There were no spurious thruster firings after touchdown. The lunar dump valves were recycled with no anomalies noted and the descent engine propellant vents were initiated. Although the primary guidance computer was targeted with a lift-off time of 108:24:31, this early lift-off time was not required. The lunar "stay" was forwarded by the Mission Control Center and the computer was set to idle at 108:21:13.

The S-band communications were maintained on the forward omnidirectional antenna during the descent, switched to aft at pitchdown, and then switched to the steerable antenna, in "slew" mode, after the lunar stay was approved.

9.10 LUNAR SURFACE ACTIVITY

9.10.1 Cabin Activity

Operations.- Subsequent to lunar module touchdown, lunar surface activities progressed in accordance with the checklist. On the checklist is an item requesting a description of the lunar surface to the Mission Control Center. Although important from a scientific point of view, this task proved to be most useful in allowing the crew to acclimate themselves to the lunar environment and, in conjunction with Mission Control, to determine more precisely the location of the lunar module. In subsequent extravehicular work, it will be important that the crewmen have a precise knowledge of their starting point on the traverse map.

The preparation for the first extravehicular period was nominal at all times except for a communications problem which became evident during switchover to portable life support system communications. This problem subsequently proved to be the result of cockpit error, which points again to the necessity of having checklists that leave no latitude for misinterpretation. The cue cards utilized during all of the extravehicular preparations and the post-extravehicular activity were quite adequate except for the one entry. However, the cue cards need to be attached more securely to the instrument panel to prevent their being dislodged by inadvertent contact.

Very little sleep was obtained. This resulted primarily from being uncomfortable in the suits, but was also due, in a lesser degree, to the tilt of the cabin. The tilt was especially noticeable during the sleep periods and made sleep difficult because the crew was uneasy in this awkward position. It is the crew's feeling that an unsuited sleep period would greatly contribute to sufficient crew sleep for the longer missions.

In general, the lunar module cabin provided an adequate base of operations during lunar surface activities in spite of the small area and the 7-degree tilt. However, it is felt that, were the lunar module to land on terrain inclined more than about 10 to 12 degrees, some difficulty would be experienced in moving about the cabin.

Equipment.- On the lunar surface, the alignment optical telescope was satisfactorily used to align the platform. Reflections in the alignment optical telescope appeared to come from the lunar module rendezvous radar antenna and the lunar module upper surfaces. These reflections eliminate the less-bright stars as candidates for use. During alignment optical telescope sighting, the radar antenna had drifted from its parked position into the field of view of the telescope. The antenna was repositioned before continuing with the alignments.

A difficulty was experienced with the interim stowage assembly in the lunar module cabin. Its retaining brackets did not hold satisfactorily. The interim stowage assembly was continually slipping out of the aft, upper restraint and interfering with cabin activity. There was no adequate place to stow used urine bags; consequently, they were in the way until such time that they could be placed in jettison bags for disposal. The disposable containers and jettison bags which were stowed in the 16-mm camera compartment on the left-hand side fell out while the camera was being removed, creating a short delay during hard-suit operations.

Even though extravehicular preparations and post-extravehicular procedures were quite adequate, meticulous effort is required to properly stow a large number of lunar surface samples. Although there is adequate stowage space when samples are properly handled, it is impossible to estimate the number, size and shape of the samples prior to flight. Thus, much time is required to sort, weigh and stow all of the material in the lunar module cabin in accordance with stowage area weight constraints. Marking of weigh bags as they are sorted and stowed is important.

Two hours after landing on the lunar surface, the rendezvous radar satisfactorily performed the command and service module tracking exercise.

9.10.2 Egress/Ingress

During cabin depressurization, a cabin pressure of less than 0.1 psia was required before the cabin door could be opened easily. The first person out is crowded as he egresses because the hatch cannot be fully opened to the Lunar Module Pilot's side with the other crewman standing behind it. The first person to egress must remember, or be coached, to lean to his left during egress in order to avoid the hatch seal. However, the hatch opening is adequate. During egress and ingress the crew must also remember to maintain horizontal clearance in order not to scrape the portable life support system and remote control unit on the upper and lower hatch seals. These techniques require practice but are worth the effort to assure integrity of the seal.

On previous missions, dust carried into the cabin during ingress was a problem. However, it did not seem to be a problem on Apollo 14, perhaps because there was less dust on the lunar surface, or perhaps, being aware of the problem made the crew more meticulous in contamination control than they would have been otherwise. Care was taken to remove the dust from the pressure garment assembly and other equipment before entry into the cabin. The brush that was used for pressure garment assembly cleaning was adequate. The technique of stomping the boots against the lunar module ladder seemed to help to some extent.

During egress and ingress, stability and mobility while on the lunar module ladder is adequate even when grasping the ladder with one hand. This leaves the other hand free to carry equipment. However, one should maneuver slowly and deliberately in order to assure stability when negotiating the lunar module ladder with one hand. No difficulty was experienced in passing equipment from the man on the surface to the man on the ladder. The lunar equipment conveyor and equipment transfer bag worked more easily than in one-g simulations.

9.10.3 Lunar Surface Operations

Mobility.- Mobility on the lunar surface is excellent. Each crewman employs a technique for travel that is most suitable for that individual. The step-and-hop gait appears to require a minimum of effort. The 1/6g simulations in the KC-135 aircraft were adequate to give one a feel of the lunar surface gravitational field. The zero-g experienced on the way to the moon aided considerably in conditioning for good mobility during operations in 1/6g. There was very little tendency to over-control or use too much force when using tools or walking on the lunar surface.

Visibility.- Visibility on the lunar surface is very good when looking cross-sun. Looking up-sun, the surface features are obscured when direct sunlight is on the visor, although the sunshades on the lunar extravehicular visor assembly helped in reducing the sun glare. Looking down-sun, visibility is acceptable; however, horizontal terrain features are washed out in zero phase, and vertical features have reduced visibility. A factor in reducing down-sun visibility is that features are in the line of sight of their shadows, thus reducing contrast. A crewman's shadow appears to have a heiligenschein around it. The visibility on the lunar surface also distorts judgment of distance. There is a definite tendency to underestimate distance to terrain features. An adequate range finder is essential.

Navigation.- Navigation appears to have been the most difficult problem encountered during lunar surface activities. Unexpected terrain features, as compared to relief maps, were the source of navigational problems. The ridges and valleys had an average change in elevation of approximately 10 to 15 feet. The landmarks that were clearly apparent on the navigational maps were not at all apparent on the surface. Even when the crewmen climbed to a ridge, the landmark often was not clearly in sight. Interpretation of the photography contributes to the navigation problem because photographs of small craters make them appear much smaller than they do to the eye. On the contrary, boulders reflect light so that in the orbital photographs they appear much larger than they do in the natural state. Boulders 2 or 3 feet in size sometimes appear in the orbital photography, but craters of that size are completely indiscernible.

Dust.- Dust on the lunar surface seemed to be less of a problem than had been anticipated. The dust clings to soft, porous materials and is easily removed from metals. The pressure garments were impregnated with dust; however, most of the surface dust could be removed. The little dust that accumulated on the modular equipment transporter could easily be removed by brushing. The lunar map collected dust and required brushing or rubbing with a glove to make the map usable.

Timeline.- Operations on the lunar surface required a much longer time than had been anticipated. The planned activities require 25 to 30 percent more time than would be required under one-g conditions. Scheduling additional activities, in the event that certain portions of the extravehicular activity have to be cancelled, is advisable.

9.10.4 Lunar Module Interfaces

Modular equipment stowage assembly.- The release handle was pulled and the assembly dropped to a height suitable for operations on the lunar surface. The modular equipment stowage assembly was manually adjusted to a higher position to remove the modular equipment transporter and readjusted to a lower position for subsequent operations. The height adjustments were made without difficulty. The thermal blankets were more difficult to take off than had been anticipated. Similarly, the thermal blankets which protected the modular equipment transporter supported its weight and manual removal of the blankets was required during modular equipment transporter deployment.

As on previous flights, all cables used on the lunar surface had sufficient set to prevent them from lying flat when deployed on the lunar surface. Both crewmen became entangled in the cables from time to time. The cables emanating from the modular equipment stowage assembly area should either be buried or routed through restraining clips to keep them from being underfoot during work around the modular equipment stowage assembly.

Scientific equipment bay.- Both the doors and the pallets were removed easily from the scientific equipment bay by utilizing the booms. The pallets could have been removed manually if required. However, the height of the pallets was at the limit for easy manual deployment on level terrain.

The offloading of the Apollo lunar surface experiment package was somewhat hindered by a small crater 8 to 10 feet to the rear of the lunar module. However, sufficient working area was available in which to place a pallet and conduct fueling operations.

Since the landing gear did not stroke significantly during the landing, a jump of about 3 feet was required from the footpad to the lowest rung of the ladder. This provided no appreciable difficulty; however, a firm landing which would stroke the landing gear a few inches would facilitate a manual offloading operation as well as egress and ingress.

9.10.5 Lunar Surface Crew Equipment

Extravehicular mobility unit.- Both extravehicular mobility units performed well during both of the extravehicular activities. There was sufficient cooling in the minimum position for normal activity. Both crewmen were required to go to intermediate, or between minimum and intermediate, for various periods of time during the climb to Cone Crater and the high-speed return from Cone Crater to Weird Crater. However, other than during these periods, minimum cooling was used predominantly.

The Lunar Module Pilot's pressure garment assembly evidenced a higher-than-usual leak rate for the first extravehicular activity, dropping 0.25 psi during the 1-minute check. The suit showed no drop during preflight checkout.

The Commander's urine collection transfer assembly hose had a kink in it which prevented proper transfer of the urine to the collection bags. Before both extravehicular activities it was necessary to unzip the suit and straighten this kink out. In one instance the suit was removed to the waist to facilitate access. The only other minor problem with the pressure garment assembly concerned the Lunar Module Pilot's right glove. The glove developed an anomalous condition before the second extravehicular activity which caused it to assume a natural position to the left and down.

It should be noted that the wrist-ring and neck-ring seals on both pressure garment assemblies were lubricated between extravehicular activities. At that time, there was very little evidence of grit or dirt on the seals. Lubricating the seals between extravehicular activities is a procedure that should be continued on subsequent missions.

Modular equipment transporter.- The modular equipment transporter deployed satisfactorily from the lunar module except as previously noted. The spring tension on the retaining clips was sufficient to hold all the equipment on the modular equipment transporter during lunar surface activities. However, with the transporter unloaded, the retaining springs have sufficient tension to lift it clear of the lunar surface when placing equipment in stowage locations. This was not noticed after the transporter was fully loaded.

The wheels did not kick up or stir up as much dust as expected before the flight. Very little dust accumulated on the modular equipment transporter.

The modular equipment transporter was stable, easily pulled, and proved to be a very handy device for both extravehicular activities. Only at maximum speeds did the transporter evidence any instability and, then, only because of rough terrain. This instability was easy to control by hand motion on the triangular-shaped tongue.

Hand tool carrier.- The hand tool carrier mated to the modular equipment transporter well, and was adequately retained by the hand tool carrier retaining clip. All stowage areas except the deep pocket were acceptable. This pocket was very difficult to reach when standing adjacent to the modular equipment transporter. It is too deep for one to easily retrieve small items. With this exception, the hand tool carrier performed satisfactorily.

Cameras.- All cameras carried in the lunar module worked well. Only two anomalies were noted. On the Commander's camera, the screw which retains the handle and the remote control unit clip worked loose several times and had to be retightened. The second anomaly concerned a 16-mm magazine which jammed and produced only 30 feet of usable film.

The television camera performed satisfactorily. It seems to be a useful tool for lunar surface exploration. A remotely operated camera with adjustment of focus, zoom, and lens setting controlled from the ground would be very useful in making available lunar surface time presently required for these tasks.

S-band erectable antenna.- The S-band antenna was easily offloaded from the lunar module and presented no problems in deployment except that the netting which forms the dish caught on the feed horn and had to be released manually. The antenna obstructs the work area immediately around the modular equipment stowage assembly. A longer cable would allow deployment at a greater distance from the lunar module. Although the deployment and erection of the S-band antenna is a one-man job, the antenna is more easily aligned with the two crewmen cooperating.

Lunar surface scientific equipment.- Offloading of the Apollo lunar surface experiments subpackages was normal, and all operations were adequate except for the operation of the dome removal tool. It required several attempts to lock the dome removal tool onto the dome. During the traverse to the Apollo lunar surface equipment package deployment site, the pallets on either end of the mast oscillated vertically and the mast flexed, making the assembly difficult to carry and to hold in the hands. However, the arrangement is acceptable for traverse up to approximately 150 yards.

There was some difficulty in finding a suitable site for Apollo lunar surface experiments package deployment because of undulations in the terrain. It was necessary to spend several moments considering the constraints that had been placed on Apollo lunar surface experiments package deployment and matching those to the site in order that the experiments could be properly deployed. After the site had been selected, the lunar dust presented some problems for the remainder of the Apollo lunar surface experiments package deployment. The suprathermal ion detector experiment sub-pallet had dust piled up against it and into the hidden Boyd bolt, which must be reached blind with the hand tool. Several minutes were wasted before the suprathermal ion detector experiment was successfully released from the sub-pallet. Subsequent to that, the suprathermal ion detector experiment was carried to its deployment site and additional difficulty was experienced in handling the three components of this experiment simultaneously. The suprathermal ion detector experiment was not sufficiently stable to prevent it from turning over several times during deployment.

No problems were experienced during removal of the mortar pack. During deployment, however, the footpads rotated out of the proper position, and the package had to be picked up and the pads rotated to a position in which they would rest properly against the surface.

The thumper deployed as expected, but the lunar regolith was so loose that the center geophone was pulled out during deployment of the last half of the thumper cable. This was confirmed during return along the line. Only 13 of the 21 thumper cartridges were fired and the first several of these required an extraordinary amount of force to fire them (section 14.4.1). The problem seemed to clear up for the last several initiators and the equipment operated precisely as expected.

Laser ranging retro-reflector experiment.- The laser reflector was deployed and leveled in the normal fashion and in the prescribed location. The dust cover was removed, the level rechecked, and the unit photographed.

Solar wind composition experiment.- No difficulty was experienced in erection of the solar wind composition experiment. The only anomaly occurred during the retrieval of the apparatus, at which time it rolled up only about half way and had to be manually rolled the remainder of the distance.

Lunar portable magnetometer experiment.- This piece of equipment performed quite satisfactorily. The only difficulty experienced was the reeling in of the cables. The set in the cable prevented a successful rewind; consequently, the cable was allowed to protrude in loops from the reel during the remainder of the traverse (section 14.4.3).

Geology.- The geology hand tools are good and, if time had permitted, they would have all been used. As in previous missions, the hammer was used by striking with the flat of the hammer rather than the small end. The only discrepancy associated with the geology tools was the use of the geology sample bags. It was difficult to find rocks small enough to fit into the small sample bags. Furthermore, they are hard to roll up. The tabs which should facilitate rolling up the bags become entangled, making it difficult to remove them from the dispenser.

9.10.6 Lunar Surface Science

Geology.- The appearance of the lunar surface was much as expected. A loose gray mantle of material covered the entire surface to an undetermined depth; however, core tubes driven into the surface would not penetrate more than 1-1/2 tube lengths and, in most cases, considerably less than that. A "rain drop" pattern over most of the regolith was observed and is clearly shown in photographs. Also observed, in certain sections of the traverse, were small lineations in the regolith material, which can be seen in certain photographs.

There was evidence of cratering and recratering on all of the area that was traversed. There was no surface evidence of multiple layers. Even in the craters, the loose gray mantle covered the entire surface, except where rocks protruded through, and concealed any evidence of stratigraphy. In the trench dug by the crew, however, evidence of three different layers was found. In one or two places on the flank of Cone Crater the crewmen's boots dug through the upper layer exposing a white layer about 3 inches from the surface. It is interesting to note that very few rocks are entirely on the lunar surface; most are buried or partially buried. Nearly all rocks of any size have soil fillets around them. The small rocks are generally coated with dirt, but some of the larger rocks are not. Many of the larger rock surfaces are soft and crumbly. However, when one uses the hammer and breaks through this, it is found that they are hard underneath.

Subtle variations in rocks are not easily discernible, primarily because of the dust. It must be remembered that the crew selected candidate samples after having observed the rocks from at least 5 or 6 feet away in order to prevent disturbing the soil around them. Features which are obvious in a hand-held specimen are not discernable at initial viewing distance. Furthermore, once the rock has been sampled, good utilization of time precludes examining the rock except to note its more prominent features. The point is that only the characteristics of a rock that are discernible at the initial viewing distance enter into the decision to sample. Sampling strategy should allow for this limitation when a wide variety of samples is desired.

The crew did observe, however, the evidence of breccia in some of the rock; and, on a few occasions, crystalline structure was evident. In most cases, the crystals were small. Only on two occasions was glass seen on the lunar surface at Fra Mauro. In one small crater there seemed to be glass-like spatter on the bottom. In the traverse to the rim of Cone Crater, one 3-foot rock was observed to be well coated with "glass".

The population of rocks in the Fra Mauro area was surprisingly low, much less than 0.5 percent of the total area. Predominantly, the rocks in evidence were 3 to 5 centimeters or smaller and, being covered with dirt, were in many cases indistinguishable from irregularities in the surface or from clumps of soil. As the crew progressed to the crest of Cone Crater, boulders became more prominent. In the boulder field, on the southeast edge of Cone, the boulder population reached, perhaps, 3 to 5 percent of the entire surface, with many boulders undoubtedly being concealed just below the surface. Rays were not discernible on the edge of the craters, possibly because of the low population and also because the nearest horizon was seldom more than 150 feet away.

<u>Soil mechanics</u>.- Footprints on the lunar surface were not more than 1/2 inch to 3/4 inch deep except in the rims of craters, where, at times, they were 3/4 inch to 1-1/2 inches deep. The modular equipment transporter tracks were seldom more than 1/2 inch deep. The penetrometer was easily pushed into the lunar surface almost to the limit of the penetrometer rod. During the trenching operation, the trench walls would not remain intact and started crumbling shortly after the trench was initiated. When obtaining one core tube sample, the soil did not compact and spilled from the tube upon withdrawal.

9.11 ASCENT, RENDEZVOUS, AND DOCKING

Although the ingress at the conclusion of the second extravehicular period was approximately 2 hours ahead of the timeline, an hour of this pad was used up in stowing samples and equipment preparatory to lift-off. The remaining hour assured adequate time for crew relaxation and an early start on pre-ascent procedures. There were no deviations from the checklist, although a standby procedure was available in the event of subsequent communications problems. Lift-off occurred on time. As in previous missions, debris from the interstage area was evident at staging. In addition, at docking, the Command Module Pilot reported a tear in ascent stage insulation on the bottom right side of the lunar module ascent stage (section 8.1).

Ascent was completely nominal with auto ignition and cutoff. Both guidance systems performed well. The Mission Control Center voiced up an adjustment maneuver which was performed at 141:56:49.4 using the reaction control system. The adjustment delta velocity was monitored with both guidance systems.

9.11.1 Rendezvous

Following the adjustment firing, a manual maneuver was made to the tracking attitude and rendezvous navigation procedures were initiated. For the backup charts, an elapsed time of 4 minutes 3 seconds was available (from the beginning of the adjustment maneuver until the required terminal phase initiation minus 30 minutes rendezvous radar mark). This proved to be insufficient time to complete the required procedures comfortably. The backup charts should be revised to permit ample time to obtain this first mark. The guidance systems were updated independently using their respective insertion state vectors as initial conditions. Nineteen marks were obtained with the primary guidance system. The abort guidance system updates were commenced at terminal phase initiation minus 27 minutes and continued to terminal phase initiation minus 7 minutes at which time the maneuver solution was compared. Eight marks were entered into the abort guidance system. The solutions from both lunar module guidance systems compared extremely well, agreeing on line-of-sight angles within 0.3 degree and on total delta velocity within 1.6 ft/sec. Because of VHF difficulties (section 14.1.4), the command module computer was updated with sextant marks only, prior to terminal phase initiation and produced a maneuver solution of minus 67.4, plus 0.5, minus 69.2 (uncorrected) compared with the primary guidance navigation system solution of plus 62.1, plus 0.1, plus 63.1. Using a two-out-of-three vote, the primary guidance navigation system solution was selected for the maneuver, and the corresponding rotated vector was entered into the abort guidance system. The ascent propulsion system terminal phase initiation maneuver was executed without incident. As anticipated, the guided ascent propulsion system shutdown resulted in a slight underburn.

Subsequent to terminal phase initiation, both lunar module navigation solutions were reinitialized and tracking was resumed. Simultaneously, the command module VHF tracking was found to be operating and both sextant and VHF marks were entered into the command module computer. The first midcourse solution in the primary guidance navigation system was used. The abort guidance system solution for the first midcourse correction was in excess of 5 ft/sec; consequently, this solution was discarded and abort guidance system navigation was continued without reinitialization. At the second midcourse correction, the primary guidance navigation system solution was used, and the abort guidance system solution was within 2 ft/sec.

The lunar module remained active during braking and the rendezvous was completed without incident. After passing through the final braking gate, the lunar module began station keeping on the command and service module. The Command Module Pilot executed a 360-degree pitch maneuver. No anomalies were observed during the inspection of the command and service modules. Consequently, the Commander proceeded with the predocking maneuver consisting of a 90-degree pitch down and right yaw to bring the lunar module docking target into the Command Module Pilot's field of view. At this point in the mission, the abort guidance displays were blank and the flight director attitude indicator, driven by the abort guidance system, was still indicating 150 degrees pitch and zero yaw. Efforts to restore the abort guidance system to operation were unsuccessful (section 14.2.5). Docking with the command and service module active was completed uneventfully, despite earlier concern about the docking mechanism.

The transfer of crew and equipment to the command and service module proceeded on schedule but with some concern regarding the time remaining to complete assigned tasks. The time allotted proved to be adequate but not ample. The procedures for contamination control in the command module were quite satisfactory, and particles were not observed in the command module subsequent to hatch opening.

9.12 COMMAND AND SERVICE MODULE LUNAR ORBIT ACTIVITIES

9.12.1 Circularization and Plane Change Maneuvers

Two service propulsion system firings were made during the command and service module solo phase. The circularization maneuver, which placed the command and service module in approximately a 60-nautical-mile circular orbit, was a 4-second firing performed after separating from the lunar module. The maneuver was controlled by the guidance and control system and resulted in a 2.0 ft/sec overspeed, which was trimmed to 1.0 ft/sec. Subsequent to this maneuver, a change to the constants in the command module computer short firing logic was uplinked by the Mission Control Center. The plane change maneuver was nominal with an 18-second firing controlled by the guidance and control system.

9.12.2 Landmark Tracking

All tracking, with the exception of the lunar module on revolution 17, was done using the telescope with the 16-mm data acquisition camera mounted on the sextant. Fourteen landmarks were tracked by the command and service module, two of these near perigee while in the 60- by 8-nautical-mile orbit. The low-altitude landmark tracking was accomplished with no significant difficulties. Acquisition of the target was no problem and the manual optics drive provided constant tracking of the landmark through nadir.

Landmark DE-2 was not tracked satisfactorily. The high sun angle at the time of tracking prevented acquisition of the landmark. Another landmark in the area of DE-2 was tracked and identified from the 16-mm photographs. All of the other landmarks were tracked quite easily. With the exception of DE-2, all of the graphics for the landmark targets were very satisfactory.

The lunar module, on the surface, was tracked on revolution 17. The sun reflecting from the lunar module as well as the long shadow of the lunar module made identification positive. Acquisition of the lunar module was accomplished by using the site map in the lunar graphics book and identification of surface features in the landing area. Also, on revolution 29, between scheduled landmarks, the lunar module was again acquired by manual optics. At that time, the sun could be seen reflecting off the Apollo lunar surface experiment package station.

9.12.3 Bootstrap Photography

The lunar topographic camera was used on revolution 4 to obtain pictures of the proposed Descartes landing site from the low orbit. Approximately one-third of the way into the photography pass, a loud noise developed in the camera. The camera counter continued to count and the photography pass was completed. One entire magazine was exposed. Subsequent troubleshooting established that the shutter was not operating properly (section 14.3.1). The only other pictures taken with the lunar topographic camera were of the lunar module landing on the surface.

The flight plan was changed so that three photography passes on the Descartes site were made using the 500-mm lens on the 70-mm Hasselblad camera mounted on a bracket in window 4 (fig. 9-2). The Descartes site was tracked manually with the crew optical alignment sight and the camera manually operated to expose a frame every 5 seconds. The ground supplied inertial angles and times to start the camera and the spacecraft maneuver. The spacecraft was maneuvered in minimum impulse to keep the crew optical alignment sight on the target. These same procedures were also used on revolution 34 to photograph the area near Lansburg B where the Apollo 13 S-IVB impacted.

A vertical stereo strip was obtained on revolution 26 using the 70-mm Hasselblad and 80-mm lens. This vertical stereo strip encompassed almost the entire ground track from terminator to terminator. A crew optical alignment sight maneuver was accomplished at the end of the strip for camera calibration.

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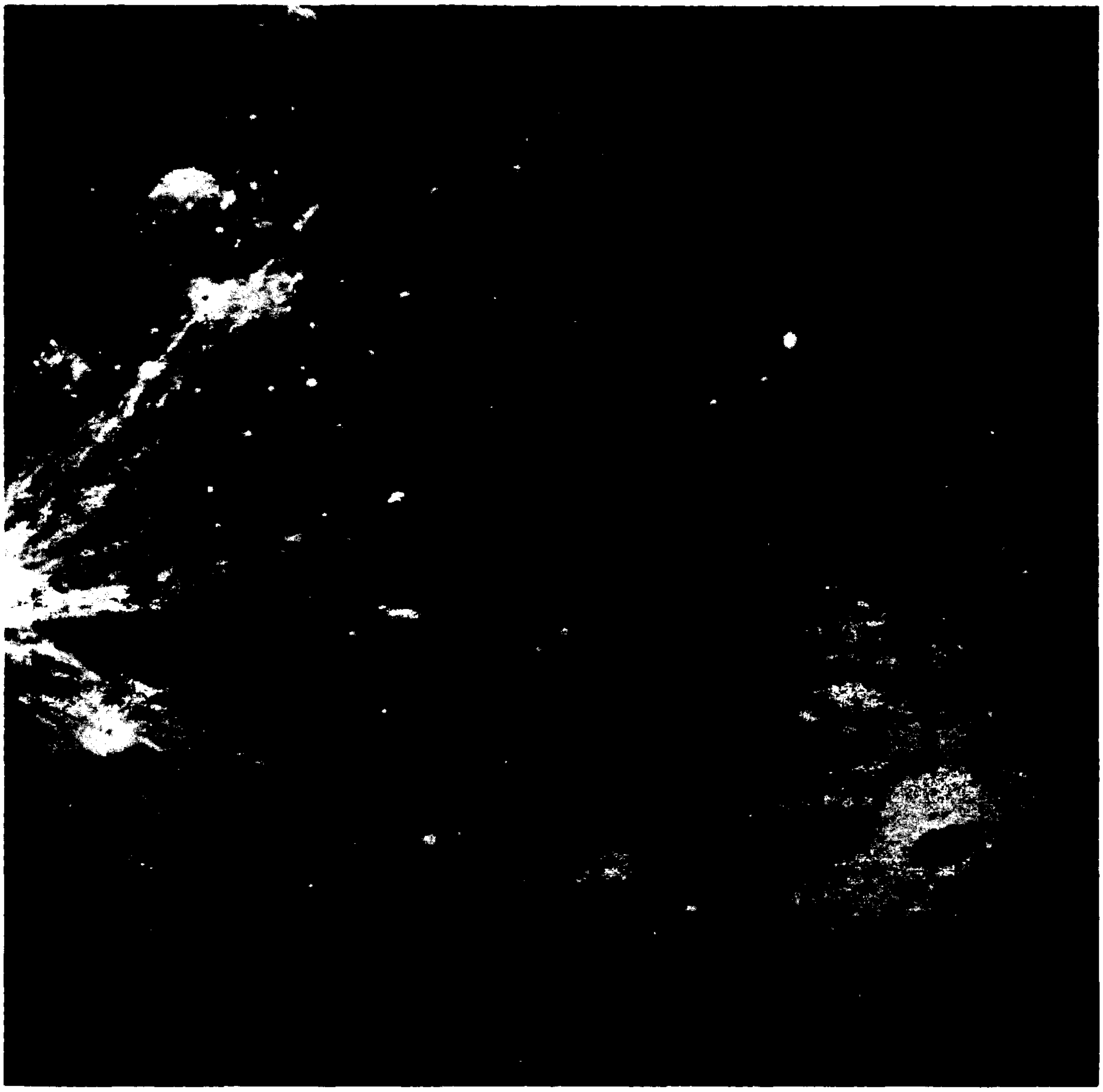

Figure 9-2.- Lunar surface features in Descartes landing site area.

9.12.4 Orbital Science Hand-Held Photography

Approximately half the planned targets for orbital science hand-held photography were deleted because of the flight plan change to use crew optical alignment sight tracking of the Descartes site. There were three stereo strips taken with the 500-mm lens using the hand-held mode (fig. 9-3). The ring sight was used to improve the sighting accuracy. Utilization of the camera in this mode was quite acceptable as long as

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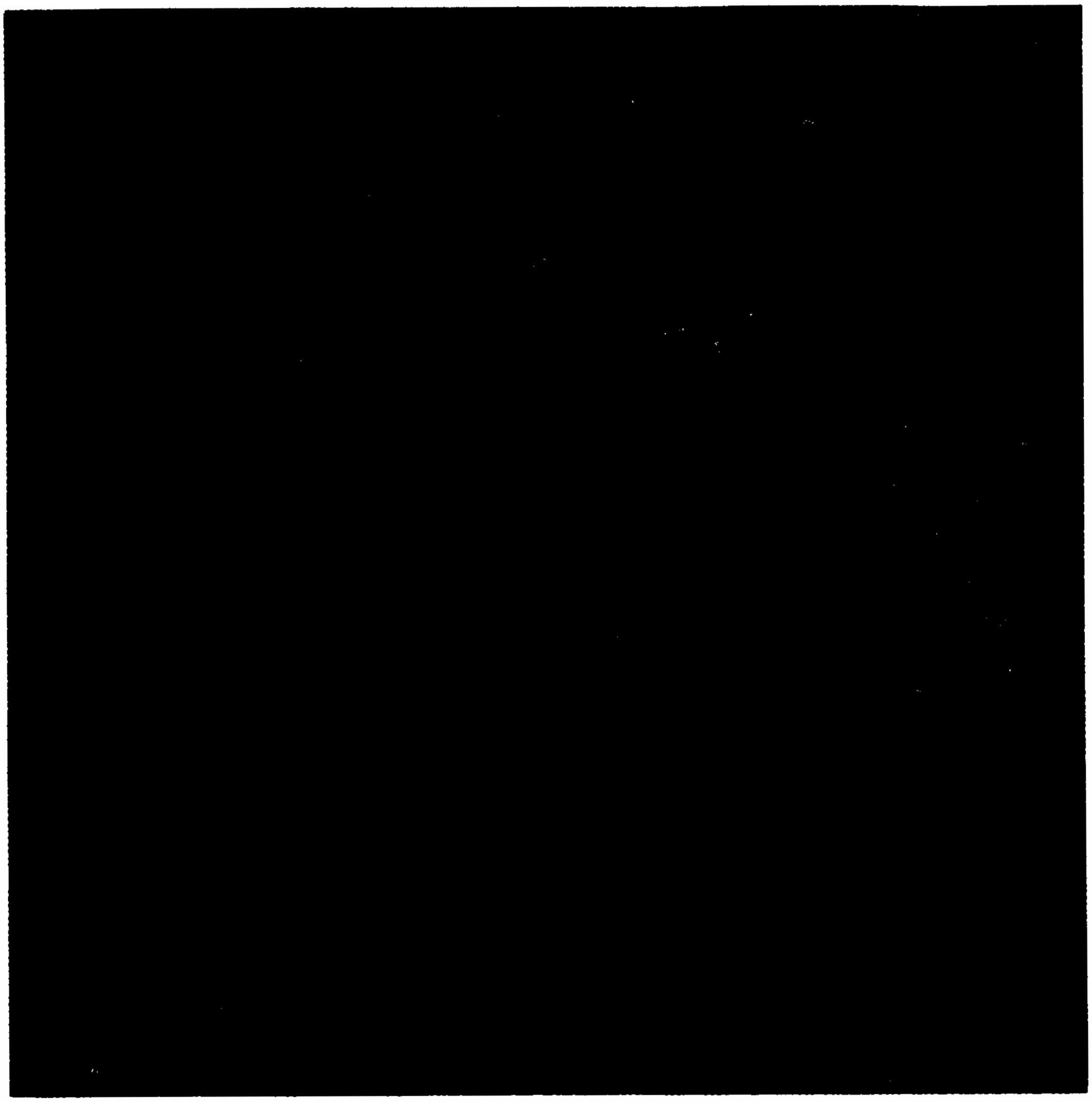

a. Western portion of King crater with smaller crater in left foreground having an 0.8-mile diameter and located 32.4 miles from center of King crater.

Figure 9-3.- Selected stereo strip photographs from lunar orbit.

the spacecraft attitude was satisfactory for target acquisition. During this flight, all hand-held photography was taken at the spacecraft attitude dictated by other requirements. On a few of the targets, the attitude made it difficult to satisfactorily acquire the target at the proper time out of any window.

During the hand-held photography and also during the crew optical alignment sight tracking, a variable intervalometer would certainly have been an asset. A single-lens reflex camera would greatly simplify the pointing task. Having orbital science targets listed in the flight plan, at times they are available, is certainly more preferable than just listing them as targets of opportunity. This is true of both photographic and visual targets.

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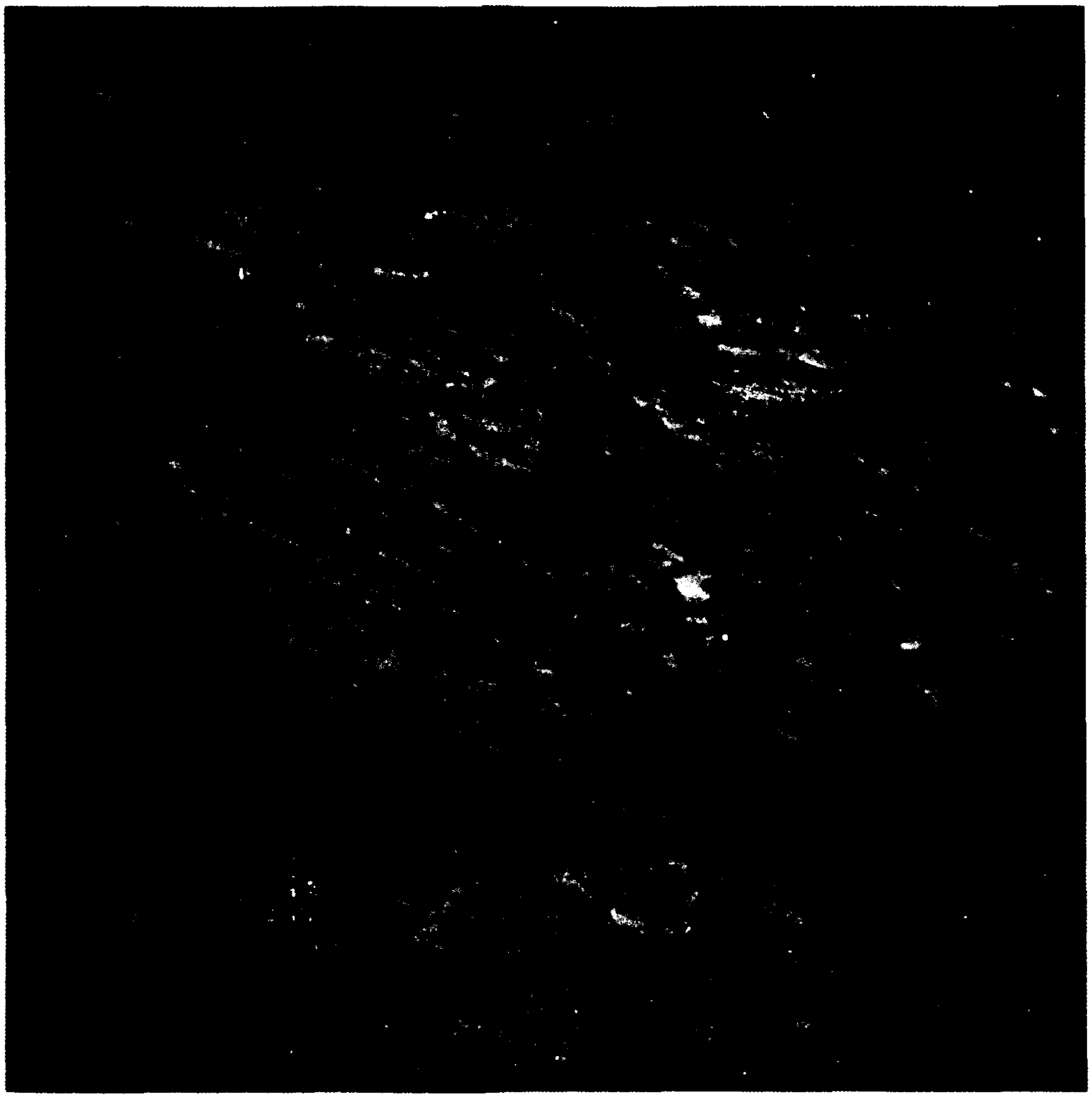
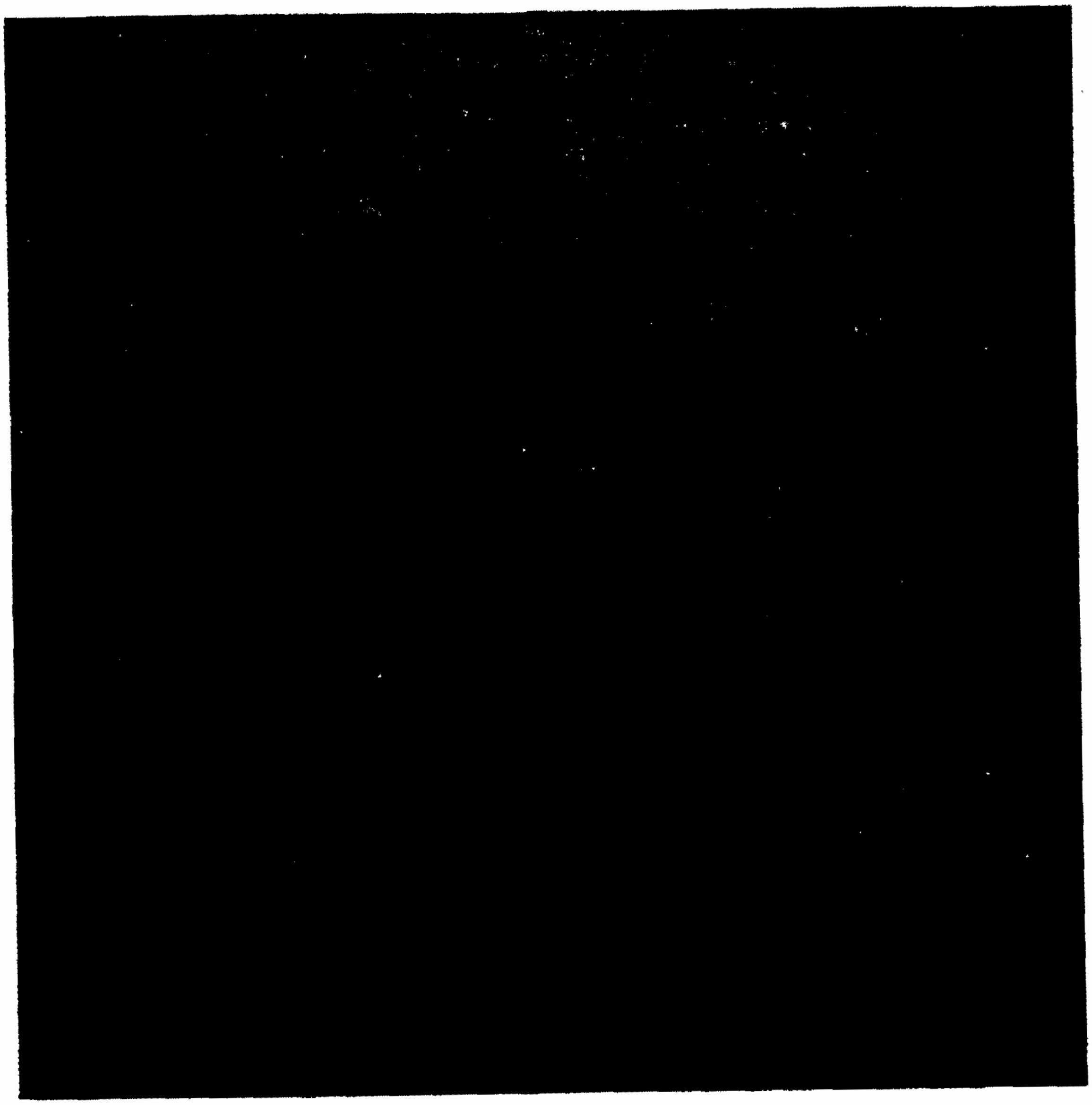


b. Central portion of 41-mile diameter King crater.

Figure 9-3.- Continued.

NASA-S-71-1655



c. Eastern portion of King crater photographed from 178 miles away.

Figure 9-3.- Concluded.

9.12.5 Zero-Phase Observations

The camera configuration was changed from that listed in the flight plan because the telemetry cable was not long enough to reach the camera mounted in the hatch window. This configuration was not checked prior to the flight because the bracket arrived late and no bracket was available for the simulator. A mark was given over the intercom and/or the air-to-ground loop on the first and last camera actuation of each pass. It was noted that the camera operated close to zero phase on each target. Eight separate areas were listed for zero-phase observations but only six of these were observed. The other two were cancelled as a result of a flight plan change. Four of the targets were on the back side of the moon and two were on the front side. There was a significant difference in the ability to observe the targets at zero phase between the back-side and front-side targets. The two significant parameters are albedo and structural relief, or contrast. Because of the lack of contrast in relief on the back side, the targets were difficult or, in some cases, impossible to observe at zero phase. Two views of a back-side target, one at zero phase and one at low phase, are shown in figure 9-4. The two front-side targets were craters located in a mare surface. The structural relief between the flat surface and the crater rim made the targets more visible at zero phase.

9.12.6 Dim-Light Photography

The window shade for the right-hand rendezvous window was easy to install and appeared to fit properly. In addition to using the window shade, the flood lights near the right-hand rendezvous window were taped. The green shutter actuation light on the camera was taped and, in general, all spacecraft lights were turned off for the dim-light photography.

All of the procedures were completed as listed in the flight plan. The only discrepancy noted was on the earth dark-side photography. There was considerable scattered light in the sextant when it was pointed at the dark portion of the earth. There was also a double image of the earth's crescent in the sextant.

9.12.7 Communications

Communications between the command and service module and the Manned Space Flight Network were marginal many times while in lunar orbit. The high-gain antenna pointing angles were very critical; a very small adjustment of the angles was the difference between having a good communication lockup or no acquisition at all (section 14.1.2).

The separate communications loop for the command and service module should be activated soon after command module/lunar module separation. The time between separation and touchdown is an extremely busy time for the lunar module and any prolonged communication with the command and service module is difficult, if not impossible. VHF communications with the lunar module were good at the time of separation and through touchdown. On rendezvous, the VHF communications from lift-off to shortly

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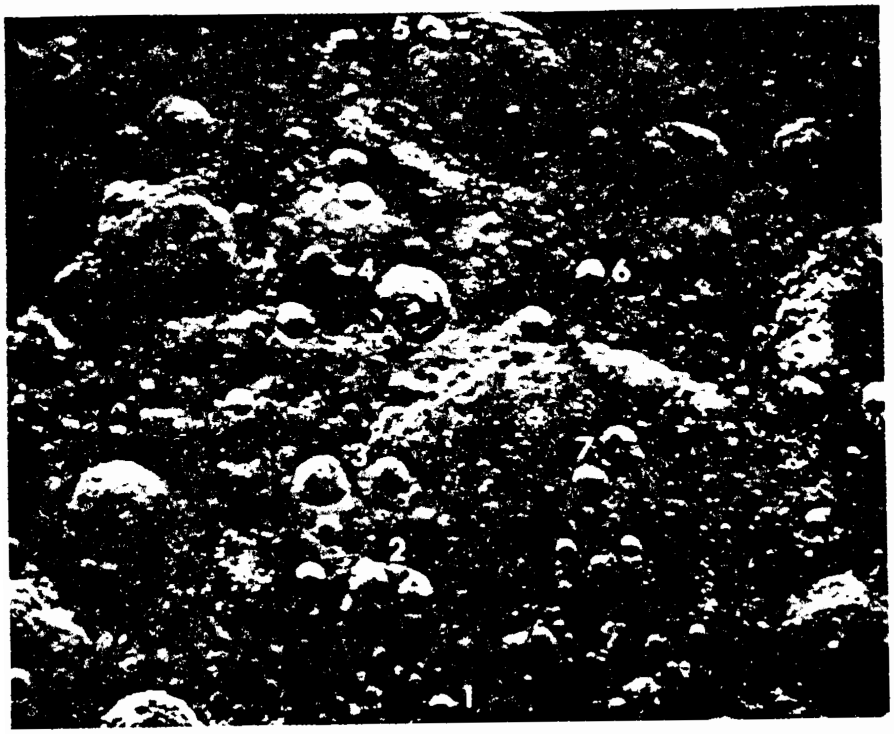
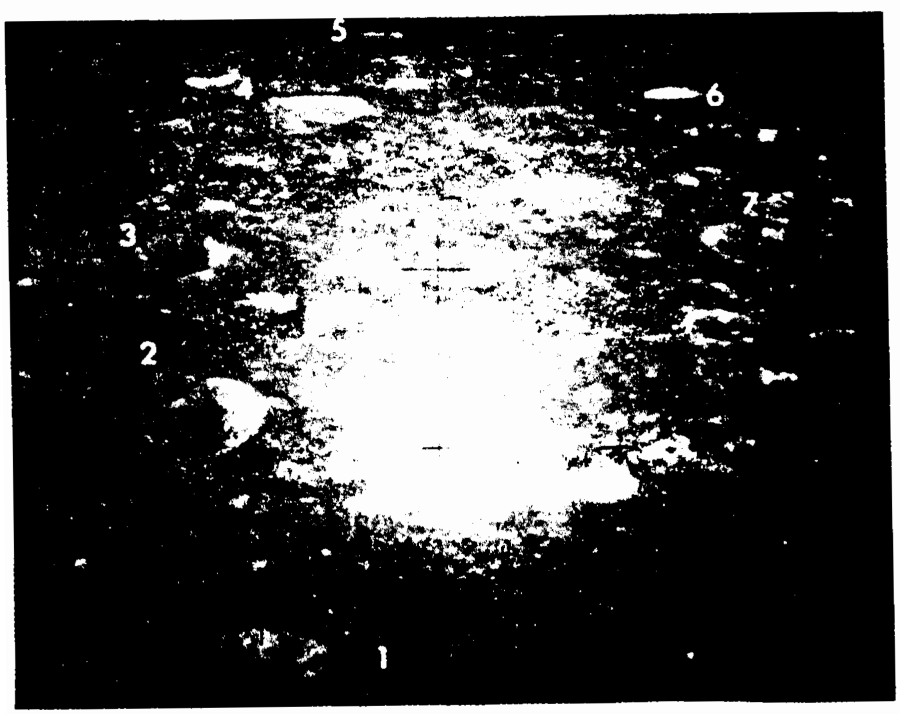


(a) High overhead view with no zero phase washout.

Note: Recognizable landmarks are identified with like numbers on each photograph.



(b) Low elevation showing zero phase washout.

Figure 9-4.- Comparison of visibility of lunar surface details looking west to east in the Pasteur crater area.

before terminal phase initiation were marginal. Also, the VHF ranging would not lock up or, when it did, a false range was indicated most of the time. Both antennas were tried, the squelch was adjusted, and ranging was turned off temporarily. However, none of these procedures improved the situation to any great degree (section 14.1.4). After terminal phase initiation the voice communications and VHF ranging were satisfactory.

9.13 TRANSEARTH INJECTION

The transearth injection maneuver was essentially nominal in all aspects. The only item worthy of comment occurred about 20 seconds prior to the end of the maneuver. There was a slight hum or buzz in the service propulsion system that continued through shutdown. Everything was steady, however, and it was not a matter of great concern. The residuals were plus 0.6, plus 0.8, and minus 0.1 ft/sec. These were trimmed to plus 0.1, plus 0.8, and minus 0.3 ft/sec. The firing time was within 1 second of the pad value.

9.14 TRANSEARTH COAST

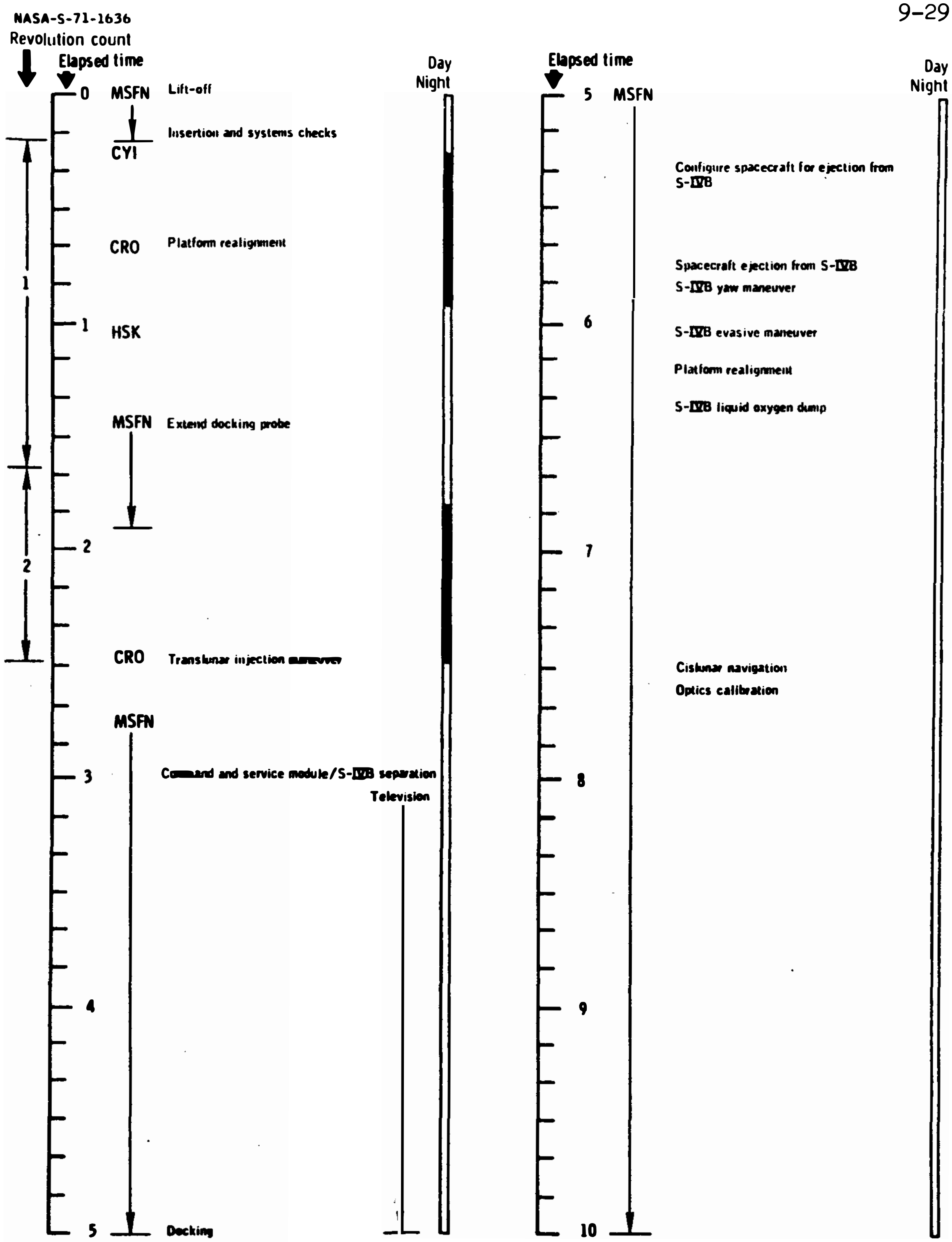
The only midcourse correction during the transearth coast phase was one reaction control system maneuver performed approximately 17 hours after transearth injection. The total delta velocity was 0.7 ft/sec. During the transearth coast phase, a schedule of no-communications navigational sightings was completed. The state vector from the transearth injection maneuver was not updated except by navigational sightings. The state vector was downlinked to the Network prior to the one midcourse correction. The midcourse correction was then incorporated and uplinked to the spacecraft. An updated Network state vector was maintained in the lunar module slot at all times. Just prior to entry, the onboard state vector compared quite well with the vector obtained by Network tracking. In addition to the navigational sightings for the onboard state vector, additional sightings were performed to obtain data on stars outside of the present constraint limits. The updates obtained on the constraint stars were not incorporated into the state vector. The cislunar navigational sighting program would be improved if a recycle feature were incorporated. Recalling the program for each mark is a drawback to expeditious navigational sightings.

The rest of the transearth coast was like that of previous lunar missions with two exceptions—inflight demonstrations were performed to evaluate the effects of zero-gravity on physical processes, and a command and service module oxygen flow-rate test was performed. Even though the metal composites demonstration was started during translunar coast, there was not sufficient time while out of the passive thermal control mode to complete all of the 18 samples. The other three demonstrations were completed.

9.15 ENTRY AND LANDING

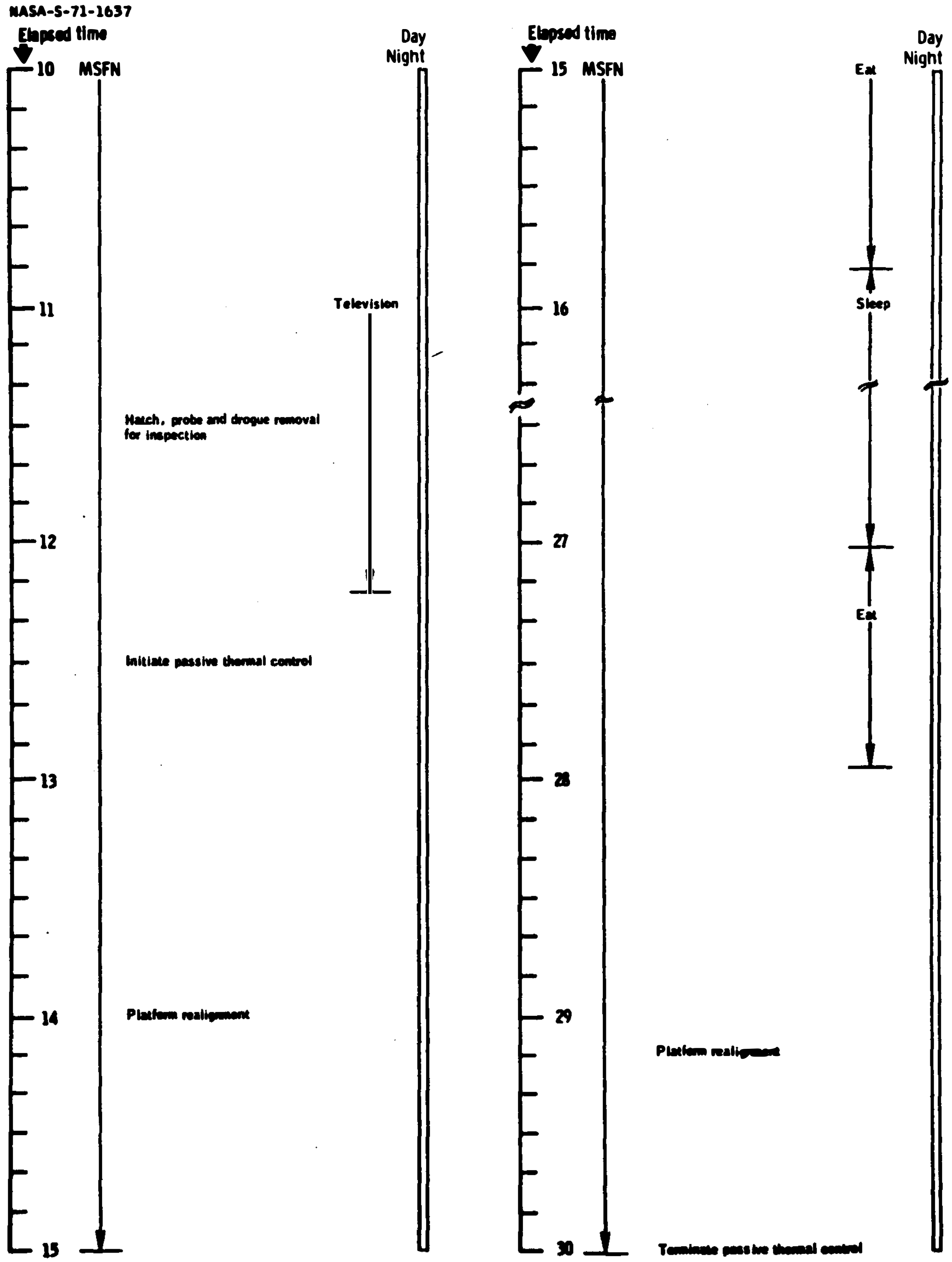
A change to the nominal entry stowage was the addition of the docking probe. The docking probe was tied down for entry at the foot of the Lunar Module Pilot's couch using procedures voiced by the Mission Control Center. Three discrepancies were noted during entry. The entry monitor system was started manually at 0.05g time plus 3 seconds. The 0.05g light never illuminated (section 14.1.5). The steam pressure was late in reaching the peg. However, the cabin pressure was used as a backup. The time of steam pressure pegging was approximately 5 to 10 seconds late and occurred at an altitude below 90 000 feet. [Editor's note: The crew checklist gives a specific time at which the steam pressure gage should peg high relative to the illumination of the 0.05g light as an indication of the 90 000-foot altitude; however, the steam pressure measurement is only an approximate indication. The crew interpreted the checklist literally.] Also, power was still on at least one of the main buses after the main bus tie switches were turned off at 800 feet. The main buses were not completely powered down until the circuit breakers on panel 275 were pulled after landing (section 14.1.6).

The landing impact was milder than anticipated. The parachutes were jettisoned and the spacecraft remained in the stable I attitude. Recovery personnel arrived at the spacecraft before the completion of the 10-minute waiting period required prior to initiating inflation of the uprighting bags for a stable I landing. One parachute became entangled on the spacecraft and was cut loose by the recovery team. The carbon dioxide bottle on the Lunar Module Pilot's life preserver was loose and the vest would not inflate when the lever was pulled. The bottle was tightened, and then the life preserver inflated properly.



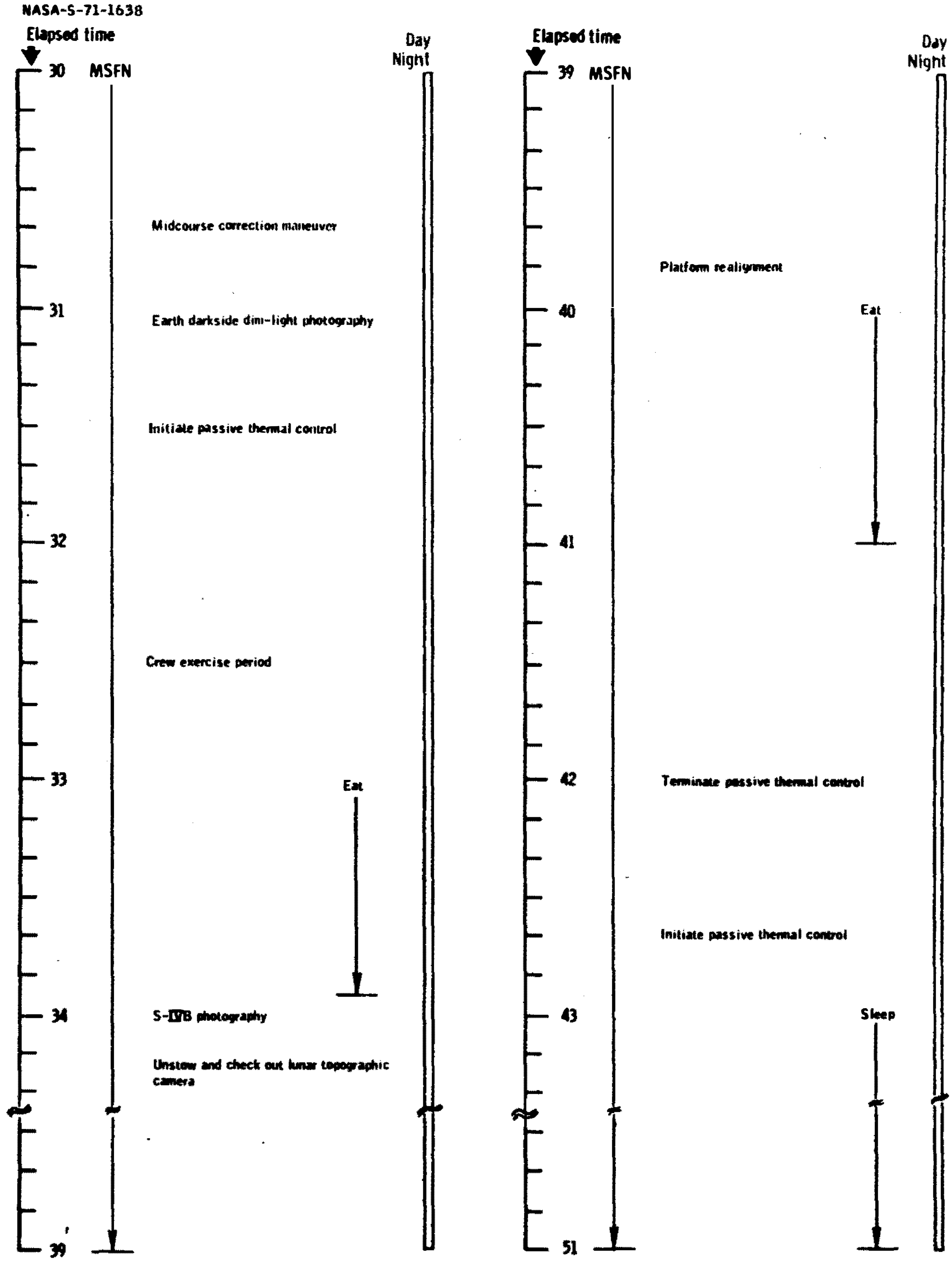
(a) 0 to 10 hours.

Figure 9-1.- Flight plan activities.



(b) 10 to 30 hours.

Figure 9-1.- Continued.



(c) 30 to 51 hours.

Figure 9-1.- Continued.

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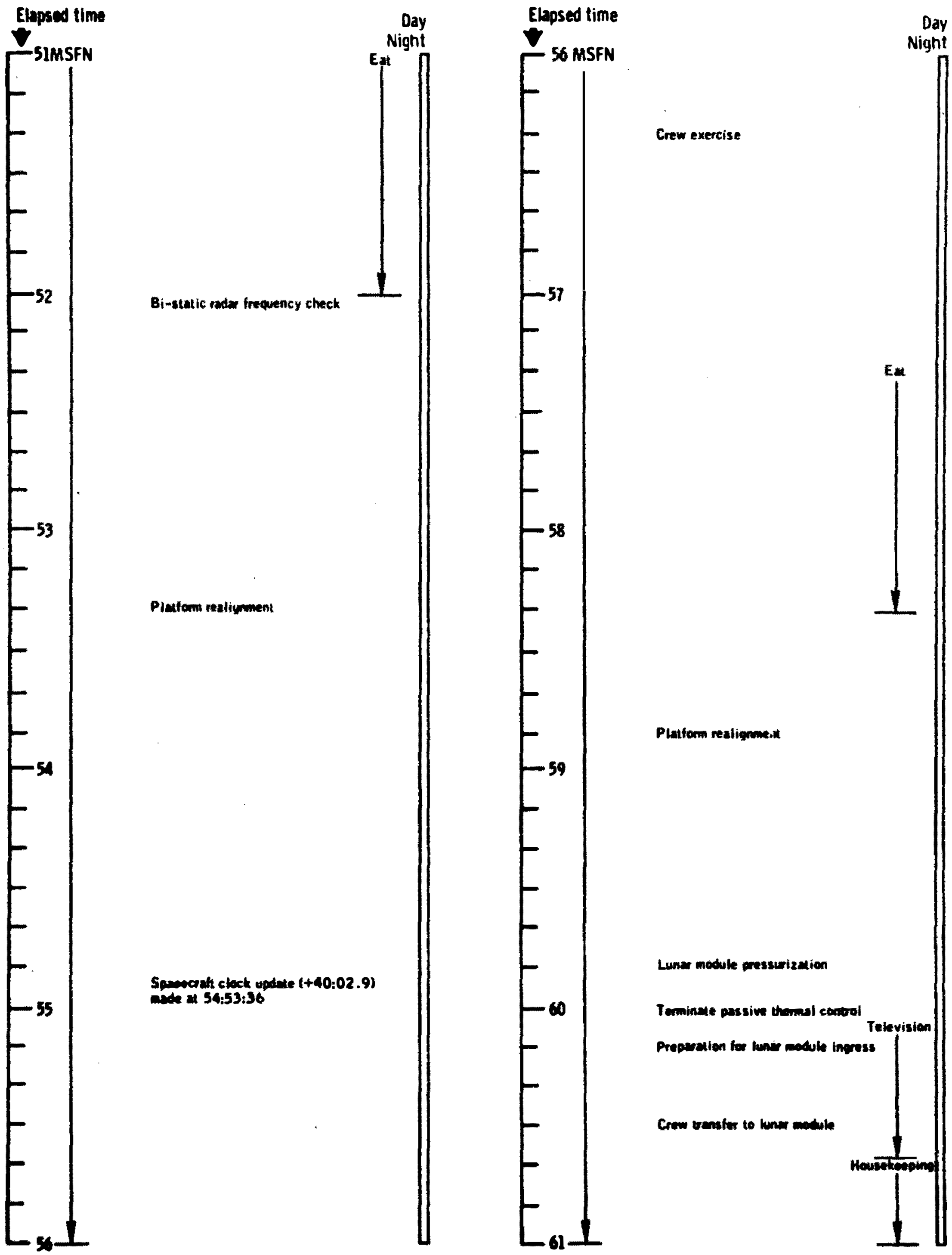
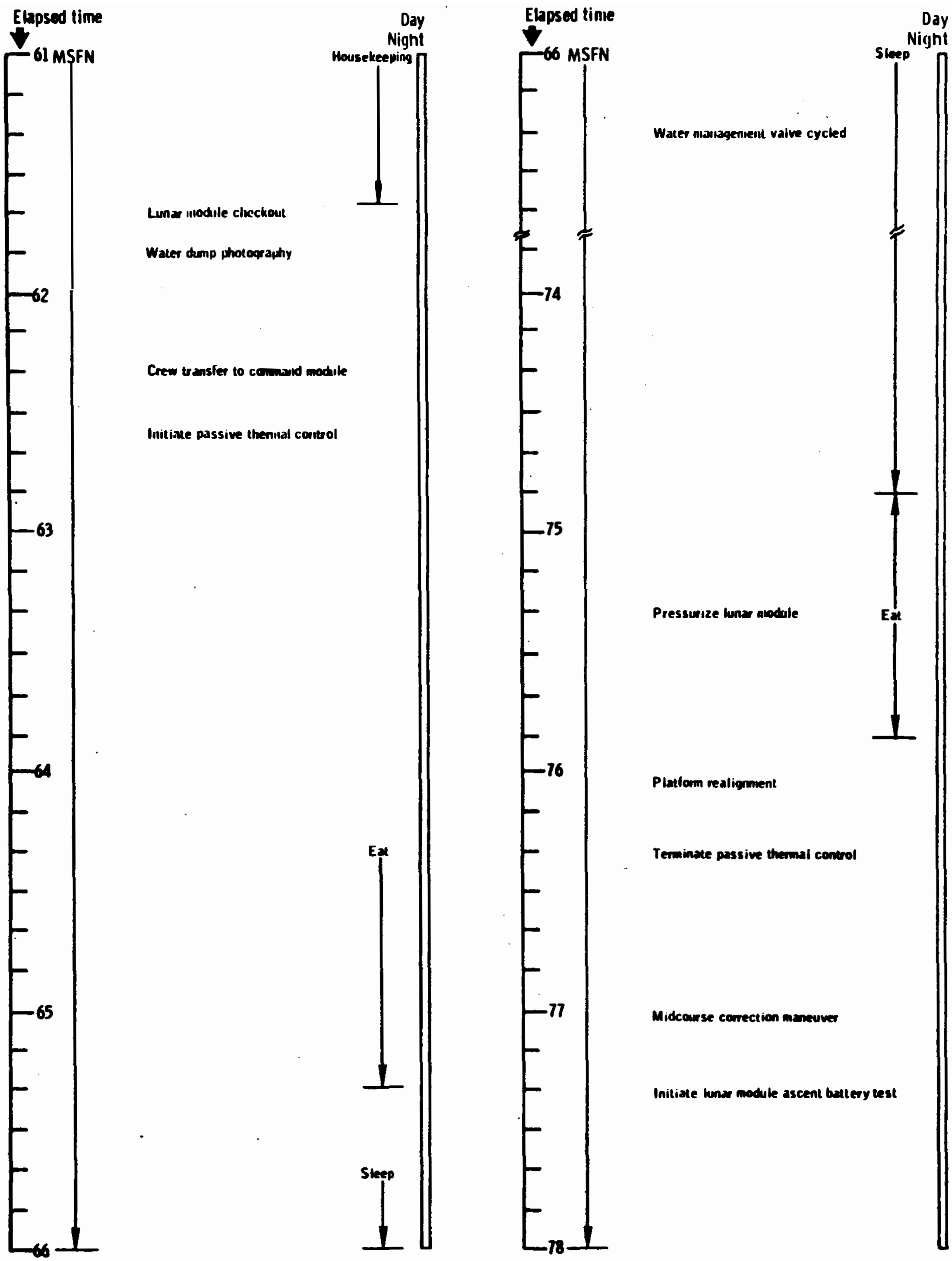


(d) 51 to 61 hours.

Figure 9-1.- Continued.

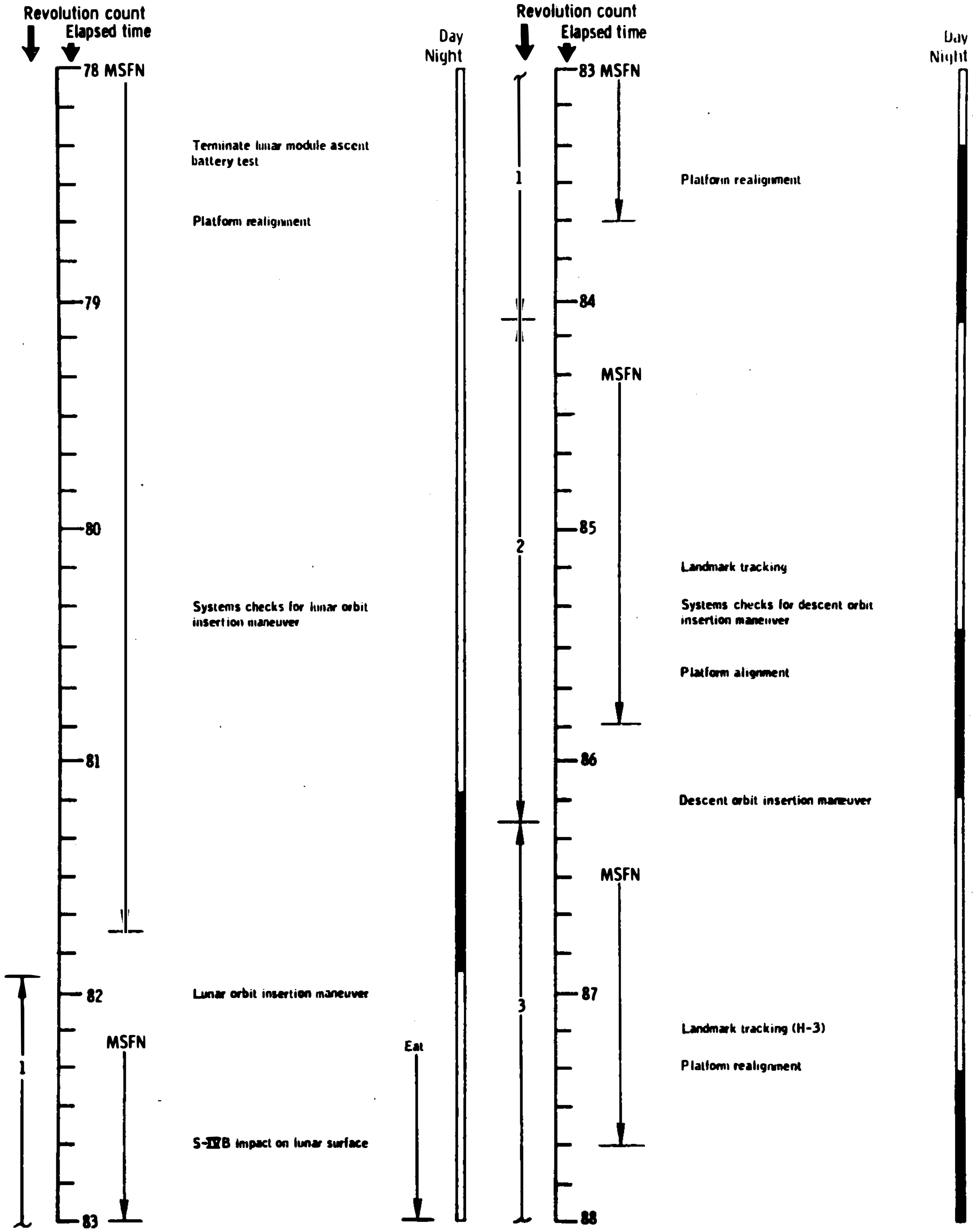
NASA-S-71-1640



(e) 61 to 78 hours.

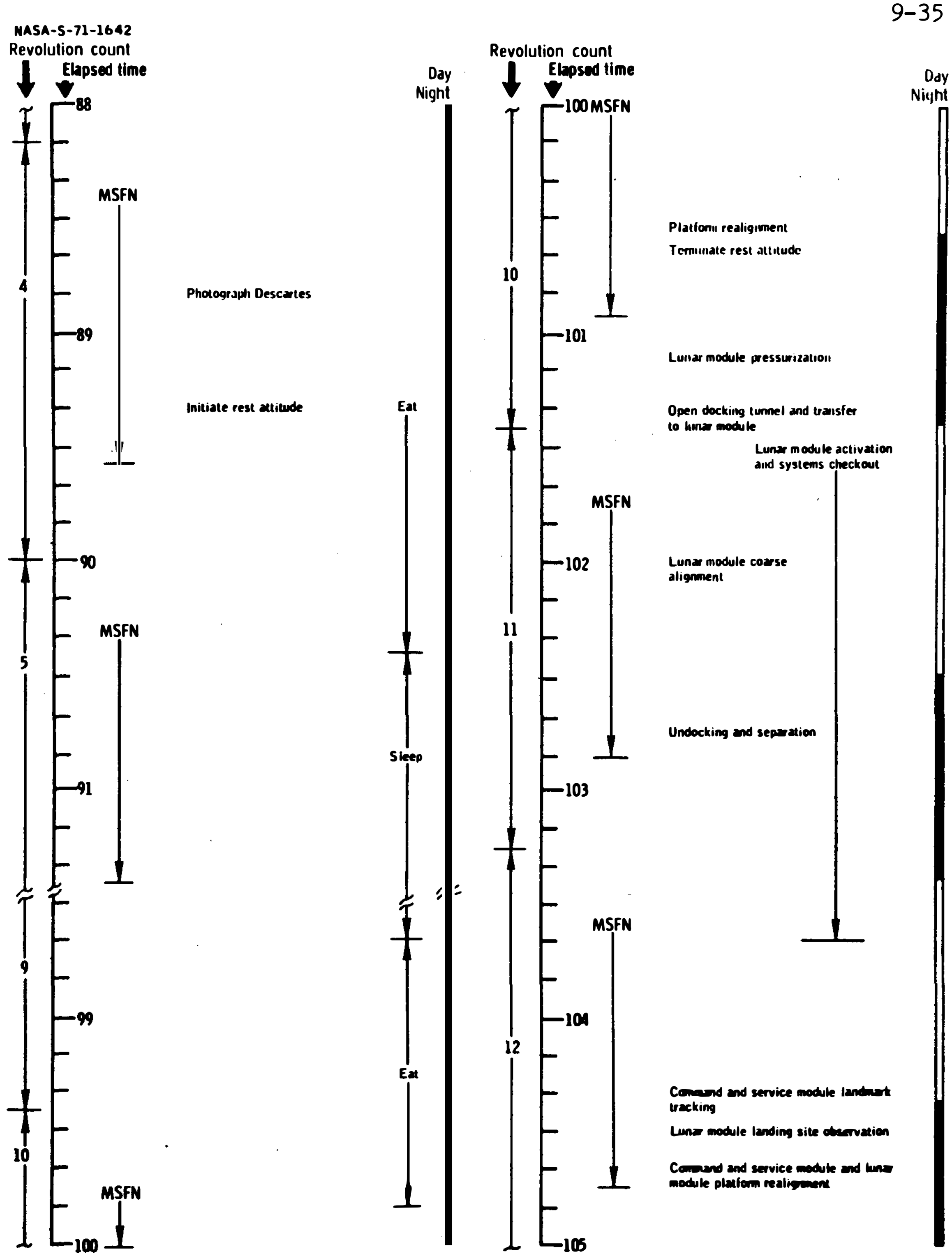
Figure 9-1.- Continued.

NASA-S-71-1641



(f) 78 to 88 hours.

Figure 9-1.- Continued.



(g) 88 to 105 hours.

Figure 9-1.- Continued.

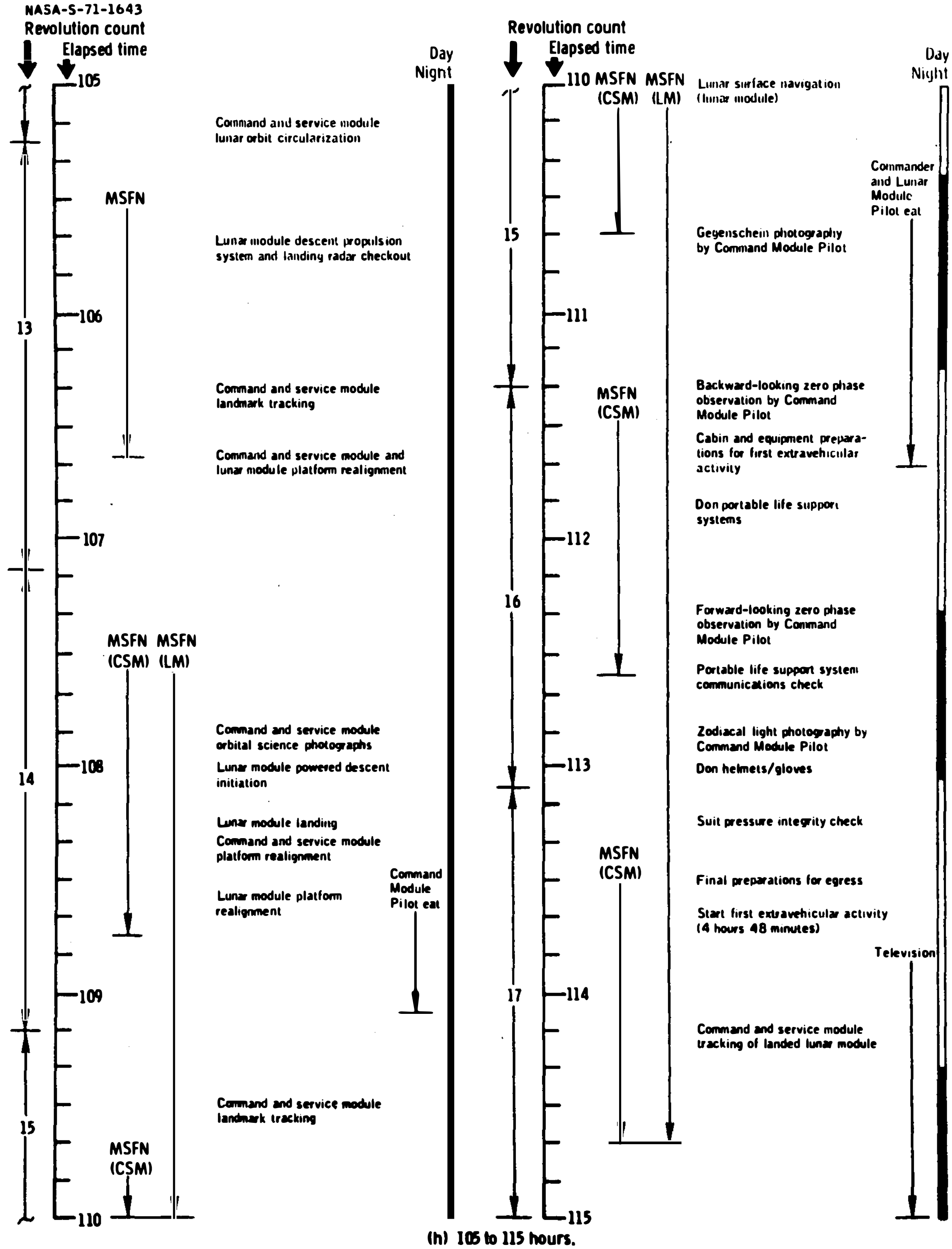
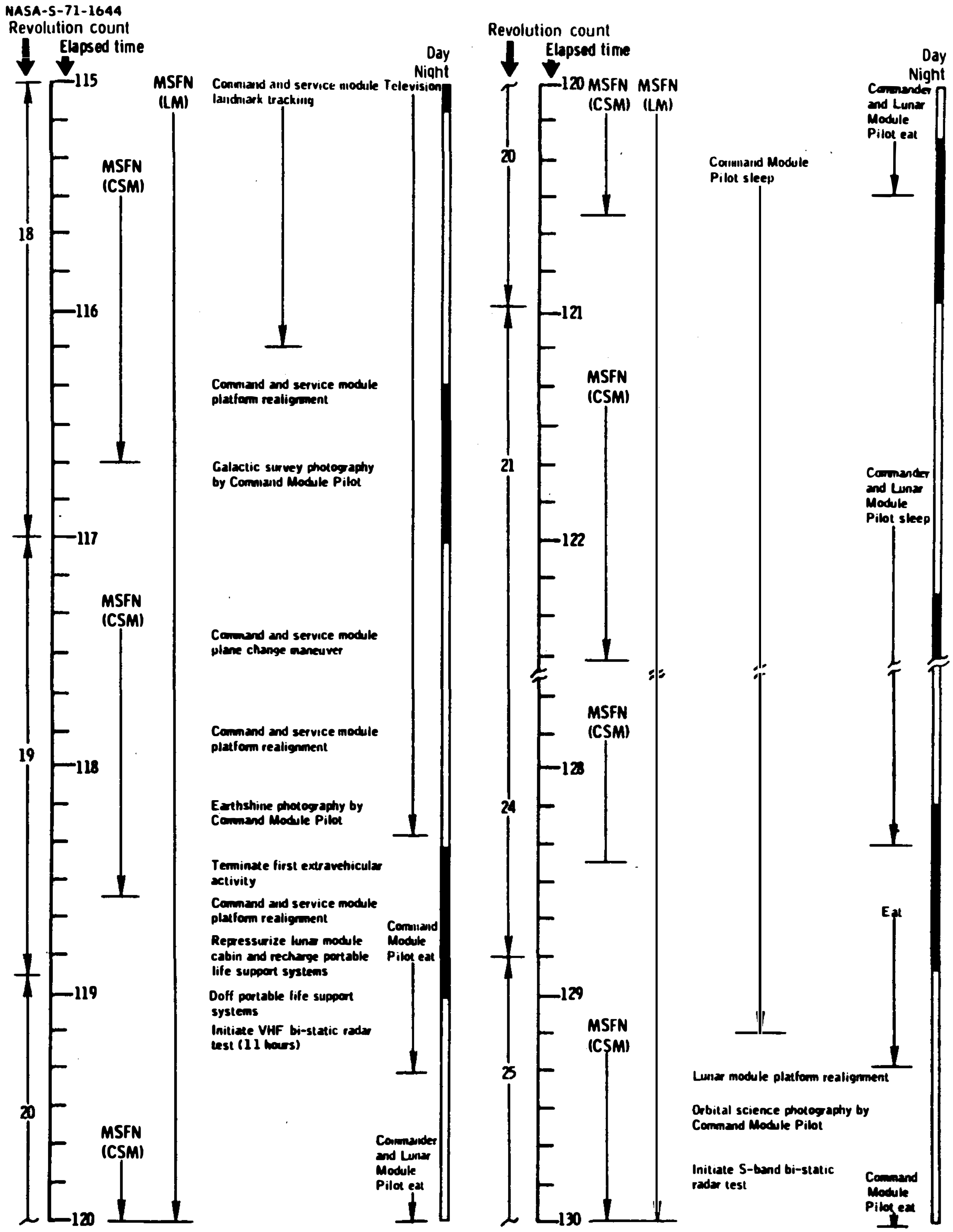
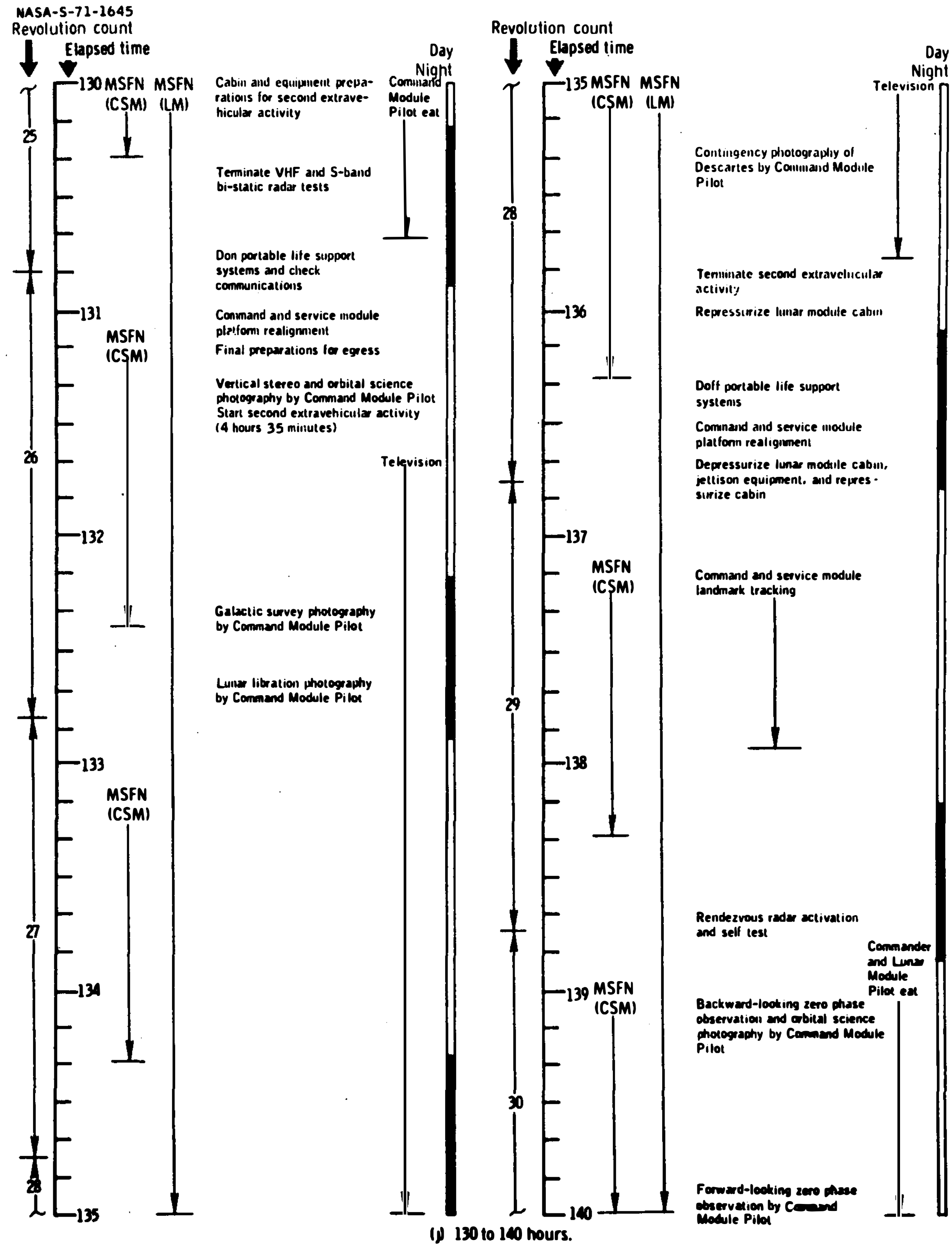


Figure 9-1.- Continued.



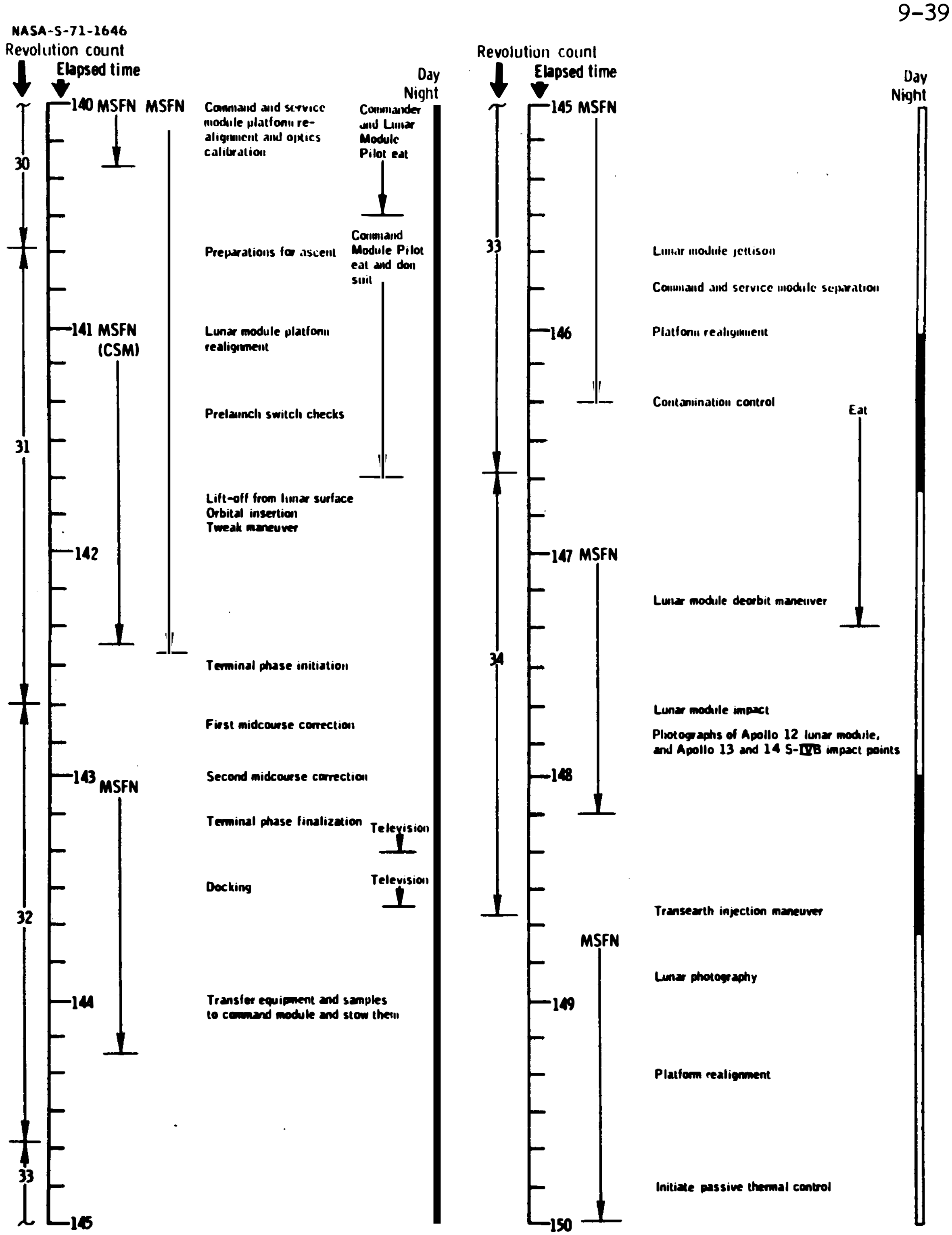
(i) 115 to 130 hours.

Figure 9-1.- Continued.



(j) 130 to 140 hours.

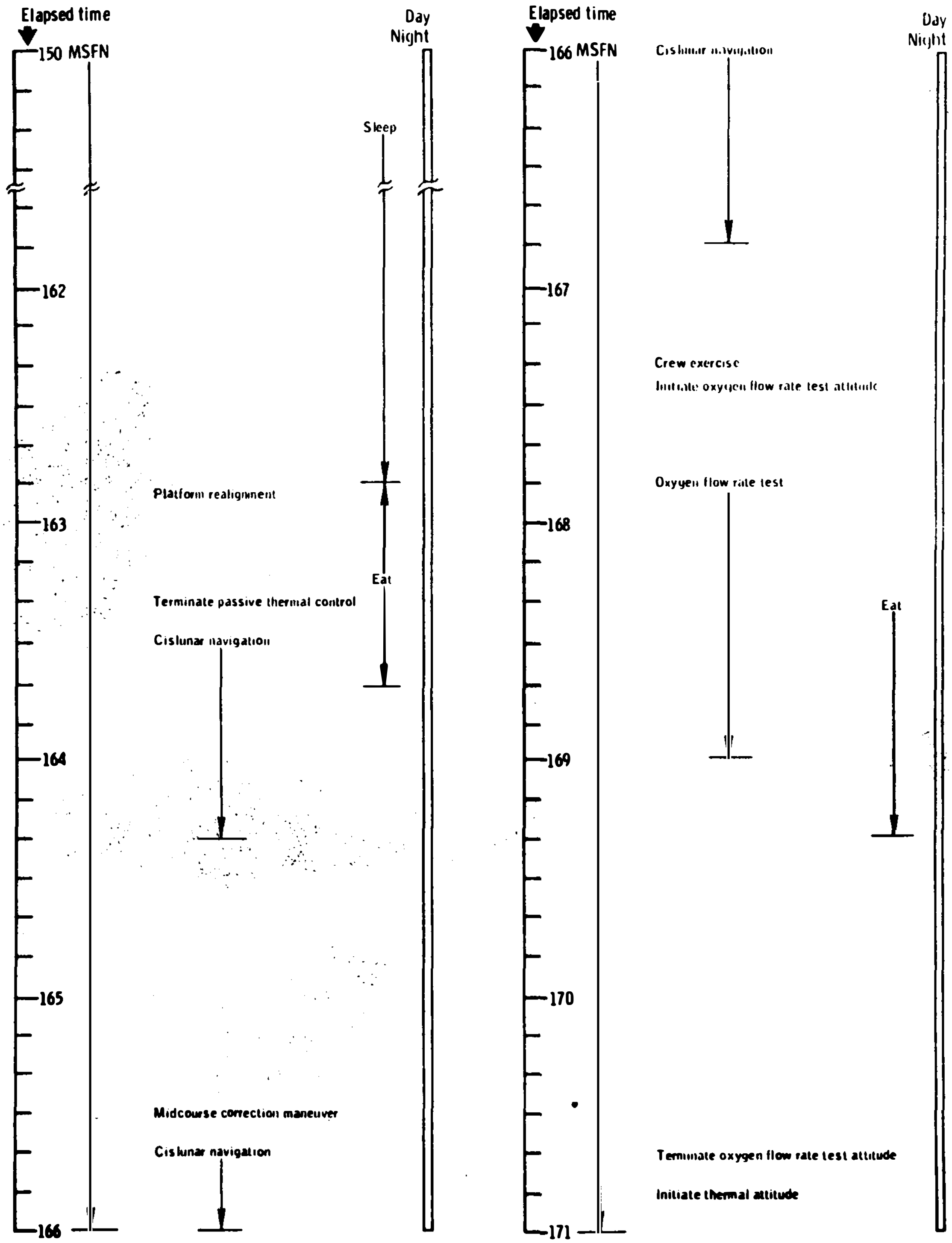
Figure 9-1.- Continued.



(k) 140 to 150 hours.

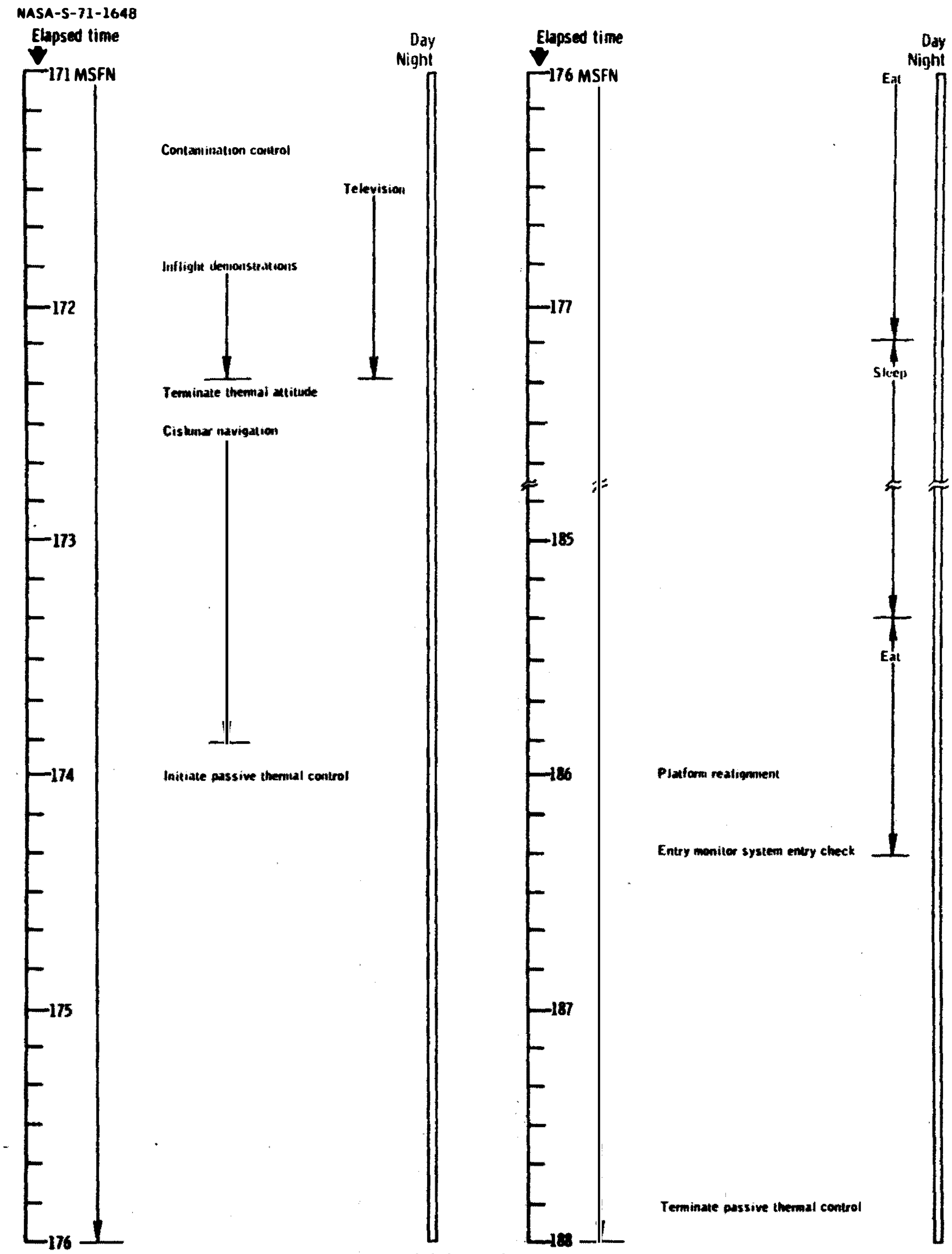
Figure 9-1.- Continued.

NASA-S-71-1647



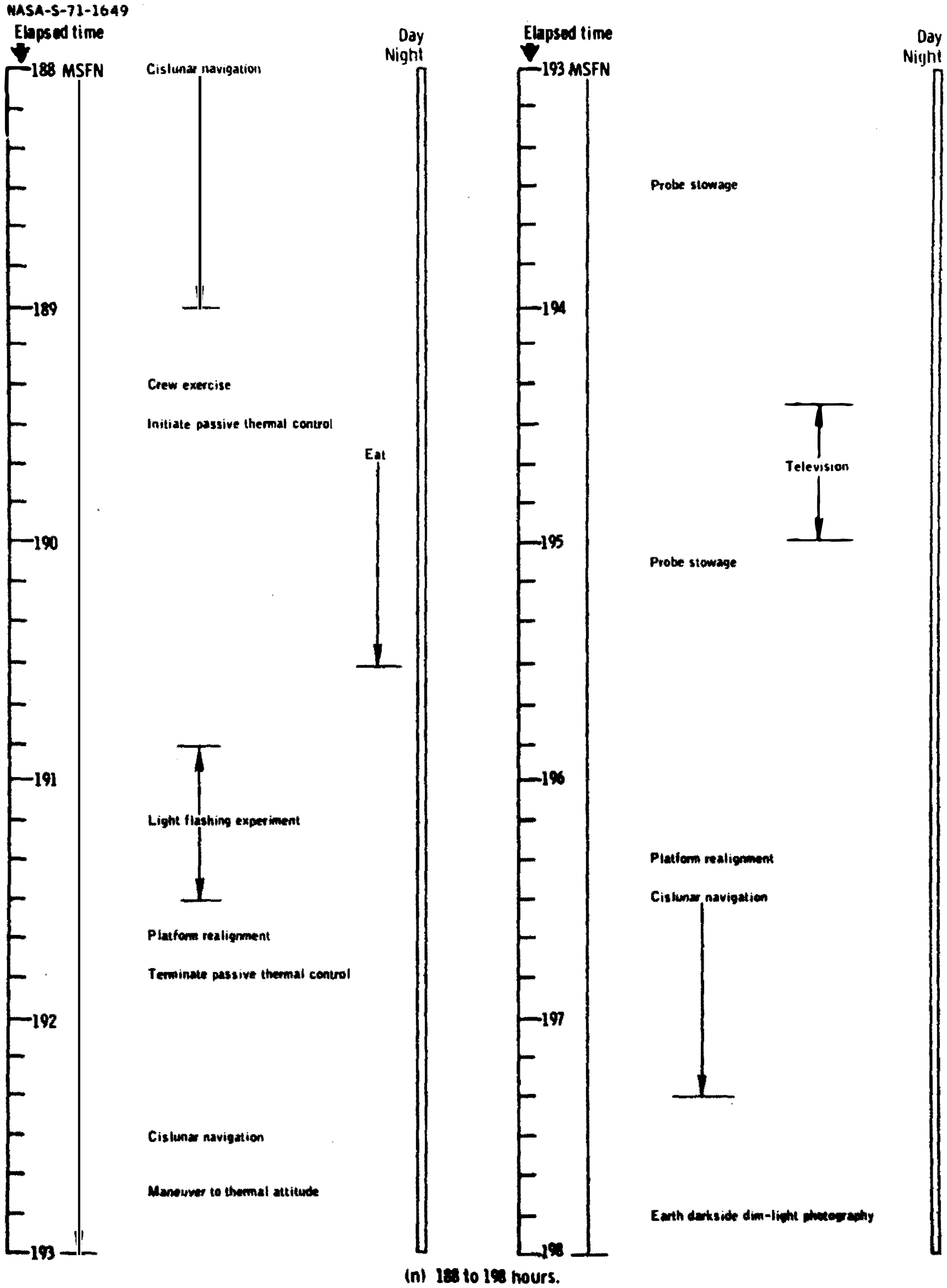
(l) 150 to 171 hours.

Figure 9-1.- Continued.



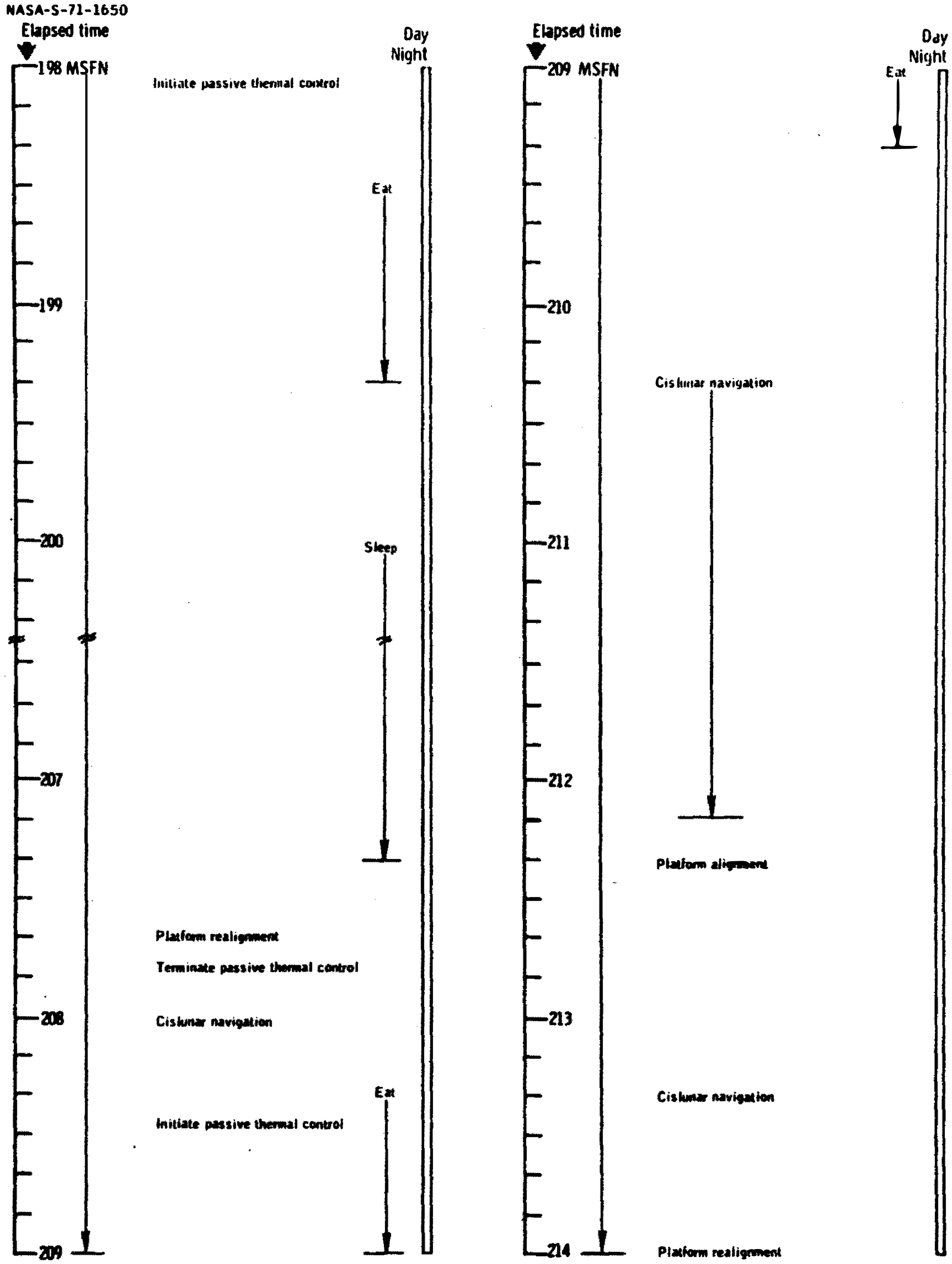
(m) 171 to 188 hours.

Figure 9-1.- Continued.



(n) 188 to 198 hours.

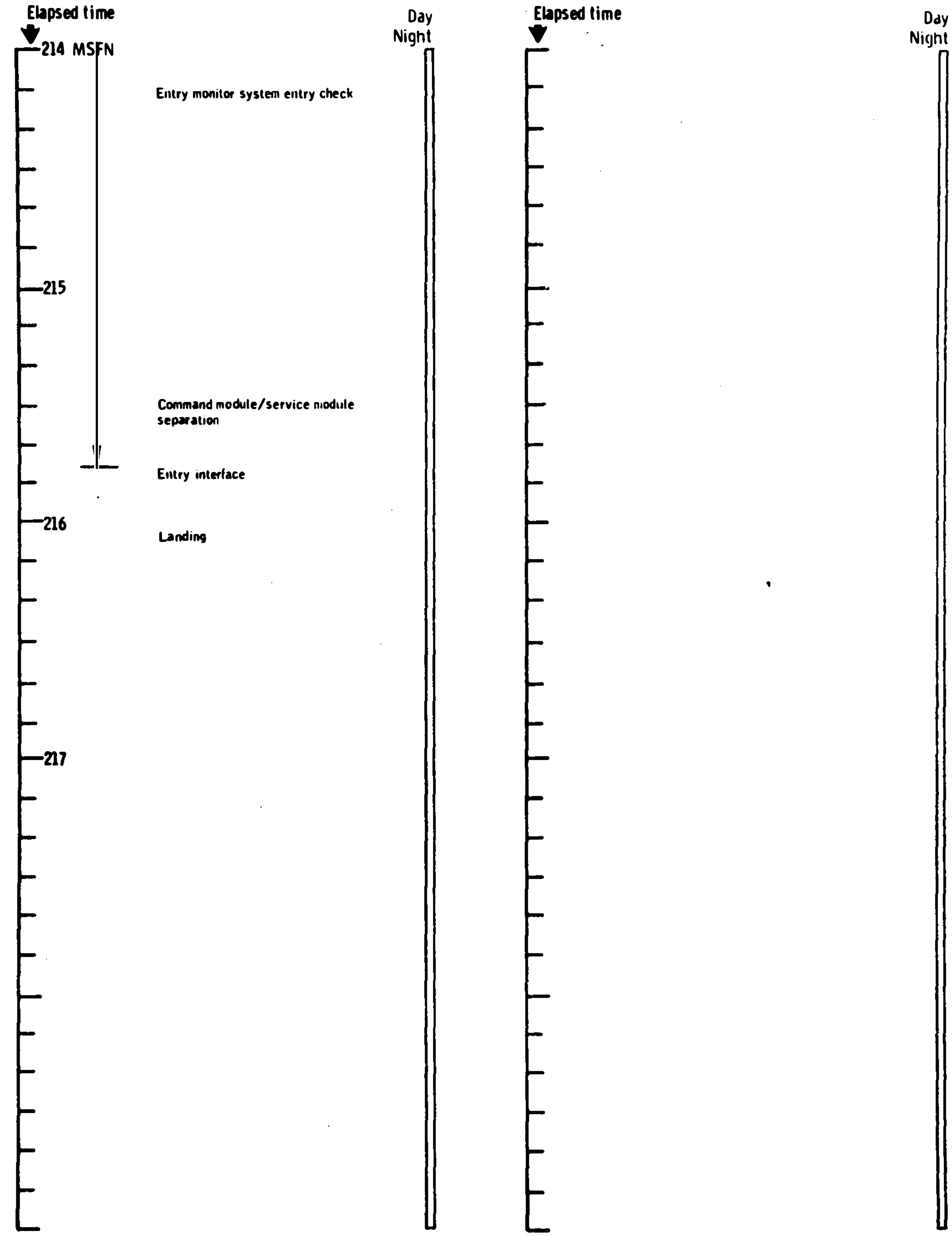
Figure 9-1.- Continued.



(o) 198 to 214 hours.

Figure 9-1.- Continued.

NASA-S-71-1651



(p) 214 to 217 hours.

Figure 9-1.- Concluded.

10.0 BIOMEDICAL EVALUATION

This section is a summary of the Apollo 14 medical findings based on a preliminary analysis of the biomedical data. A comprehensive evaluation will be published in a separate report. The three crewmen accumulated a total of 650 man-hours of space flight experience.

The crewmen remained in excellent health throughout the mission and their performance was excellent despite an alteration of their normal work/rest cycle. All physiological parameters obtained from the crew remained within the expected ranges during the flight. No adverse effects which could be attributed to the lunar surface exposure have been observed.

10.1 BIOMEDICAL INSTRUMENTATION AND PHYSIOLOGICAL DATA

Problems with the Commander's biomedical instrumentation harness began prior to lift-off when the sternal electrocardiogram signal became unreadable 3 minutes after spacecraft ingress. A waiver was made to the launch mission rule requiring a readable electrocardiogram on all crewmen. During the first orbit, the Commander's sternal electrocardiogram signal returned to normal.

At about 57 1/2 hours, the Commander noted that his lower sternal sensor had leaked electrode paste around the sealing tape. This situation was corrected by applying fresh electrode paste and tape.

When the Commander transferred to the portable life support system in preparation for the extravehicular activity, his electrocardiogram was so noisy on two occasions that the cardiotachometer outputs in the Mission Control Center were unusable and manual counting of the heart rate for metabolic rate assessment became necessary. A good electrocardiogram signal on the Commander was reacquired after completion of the extravehicular activity and return to the lunar module. The threads on the top connector of the signal conditioner were accidentally stripped. However, the electrocardiogram signal was restored for the remainder of the flight by tightening this connector.

The quality of the Lunar Module Pilot's electrocardiogram was excellent from spacecraft ingress until approximately three days into the mission. At that time, intermittent noise transmissions typical of a loose sensor were received. The lower sternal sensor was reserviced with fresh paste and tape. This happened two additional times. No attempt was made to correct the situation on the last occurrence.

The Lunar Module Pilot also lost his impedance pneumogram after the eighth day of flight. Postflight examination showed that the signal conditioner had failed.

Physiological measurements were within expected ranges throughout the mission. The average crew heart rates for work and sleep in the command module and lunar module are listed in the following table.

Activity	Average heart rates, beats/min		
	Commander	Command Module Pilot	Lunar Module Pilot
Command module:			
Work	57	66	62
Sleep	52	46	50
Lunar module:			
Work	77	--	76
Sleep	70	--	--

Figure 10-1 presents the crew heart rates after translunar injection during the multiple unsuccessful docking attempts and the final hard dock.

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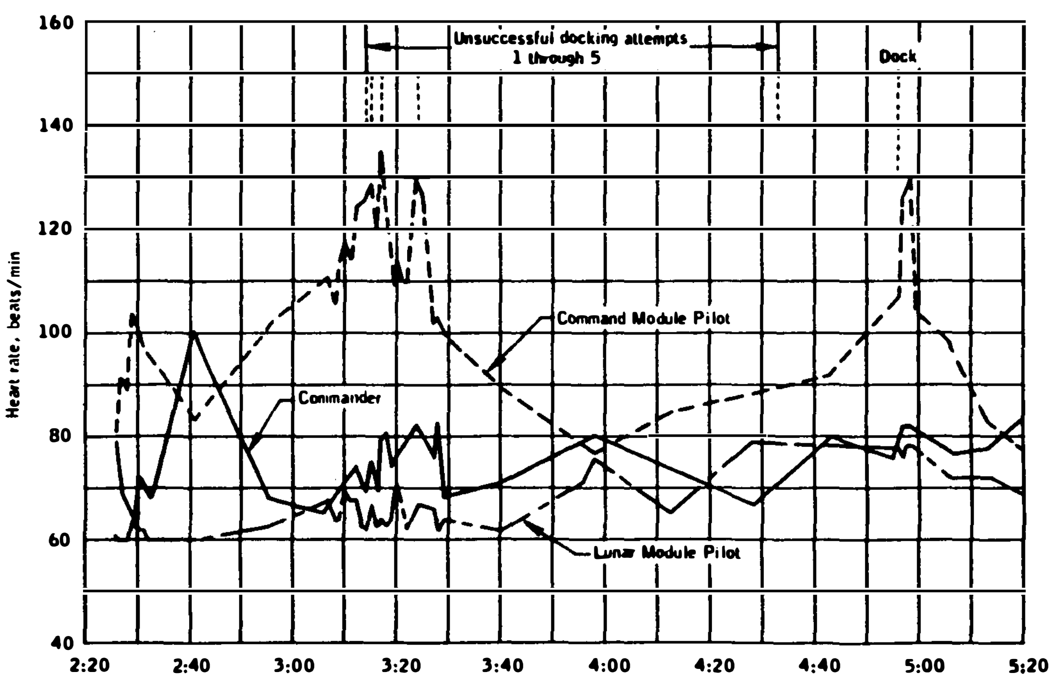

Figure 10-1.- Crew heart rates during multiple docking attempts.

During powered descent and ascent, the Commander's heart-rate averages ranged from 60 to 107 beats per minute during descent and from 69 to 83 beats per minute during ascent, as shown in figures 10-2 and 10-3, respectively. These heart-rate averages for descent and ascent were the lowest observed on a lunar landing mission.

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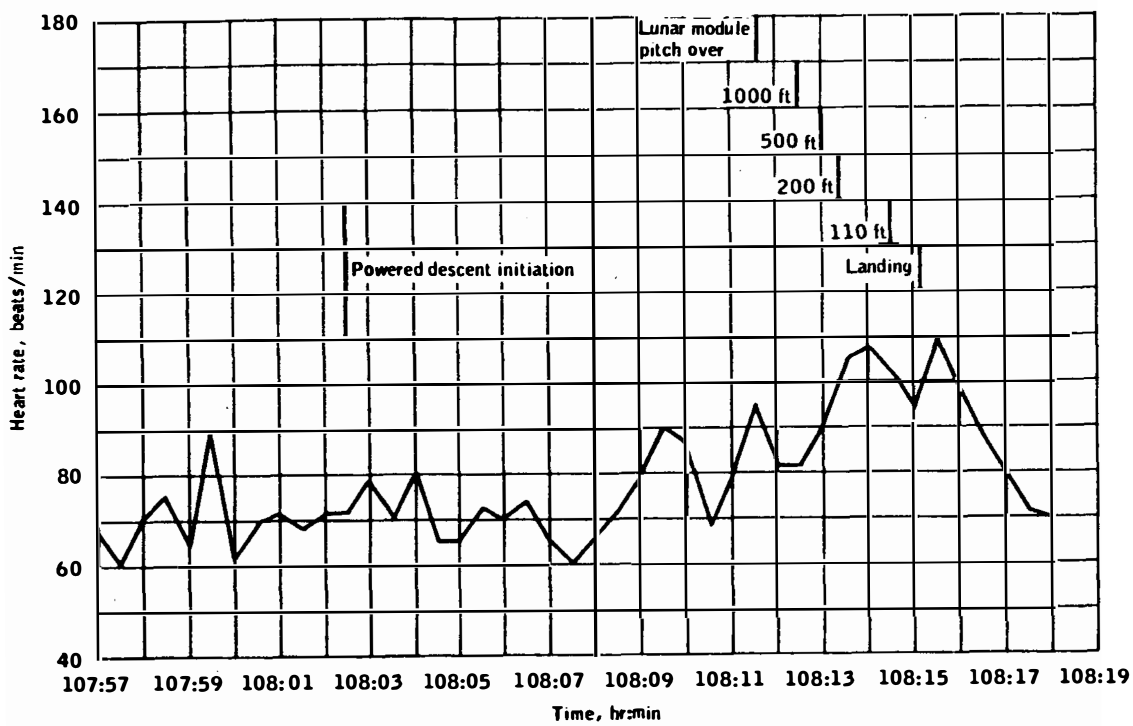

Figure 10-2.- Heart rates of the Commander during lunar descent.

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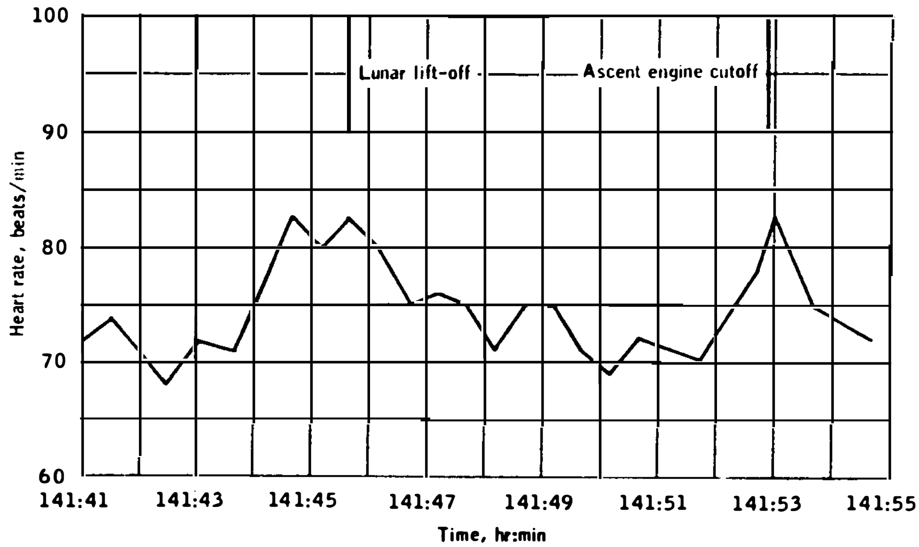

Figure 10-3.- Heart rates of the Commander during lunar ascent.

Heart rates during the two extravehicular activity periods are shown in figures 10-4 and 10-5. The Commander's average heart rates were 81 and 99 beats per minute for the first and second periods, respectively; and the Lunar Module Pilot's average heart rates were 91 and 95 beats per minute. The metabolic rates and the accumulated metabolic production of each crewman during the extravehicular activity periods are presented in tables 10-I and 10-II. A summary of the metabolic production during the two extravehicular periods is presented in the following table.

Crewmen	Metabolic production			
	First period		Second period	
	Btu/hr	Total, Btu	Btu/hr	Total, Btu
Commander	800	3840	910	4156
Lunar Module Pilot	930	4464	1000	4567

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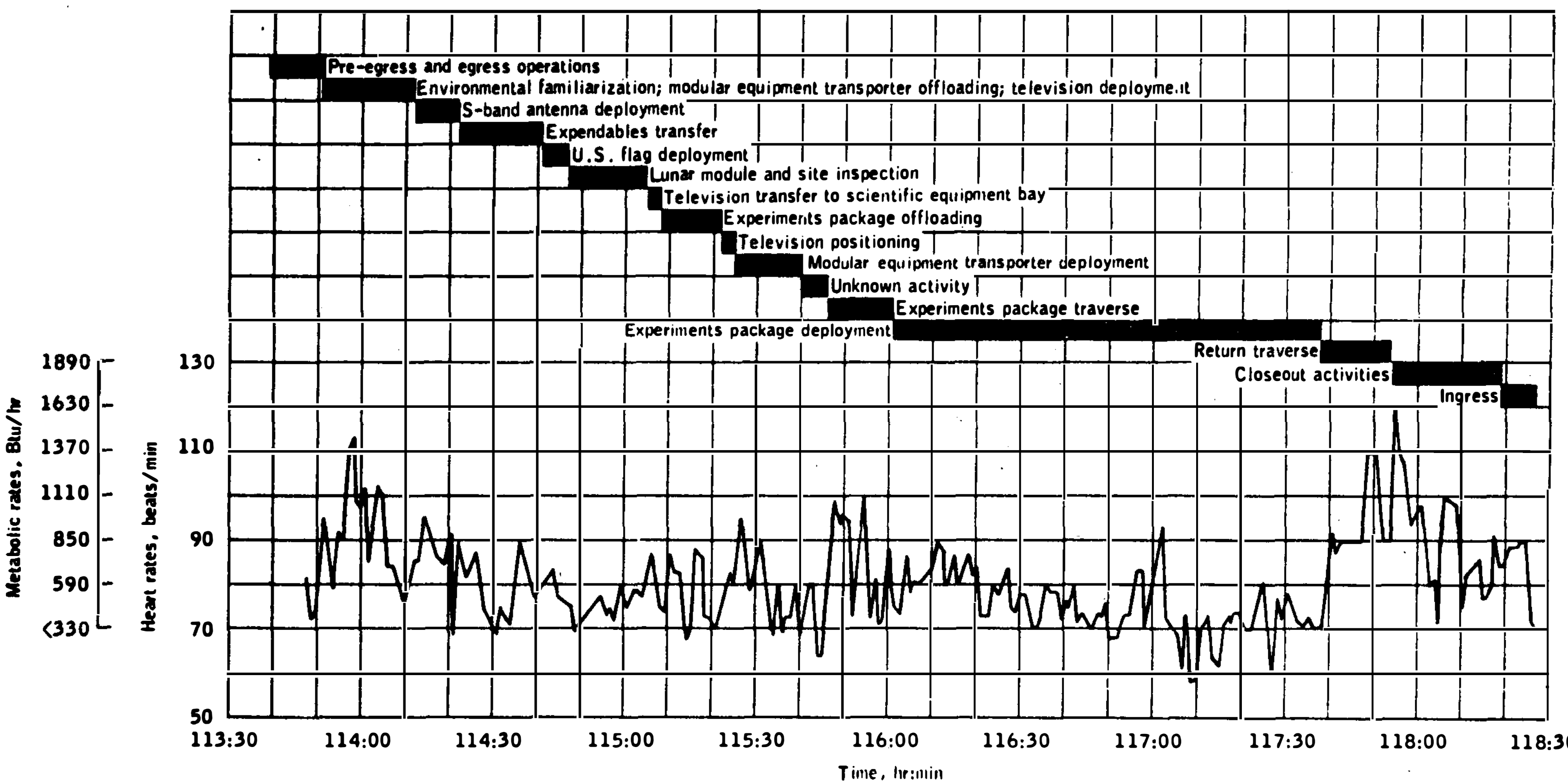

(a) Commander.

Figure 10-4.- Heart rates during first extravehicular activity.

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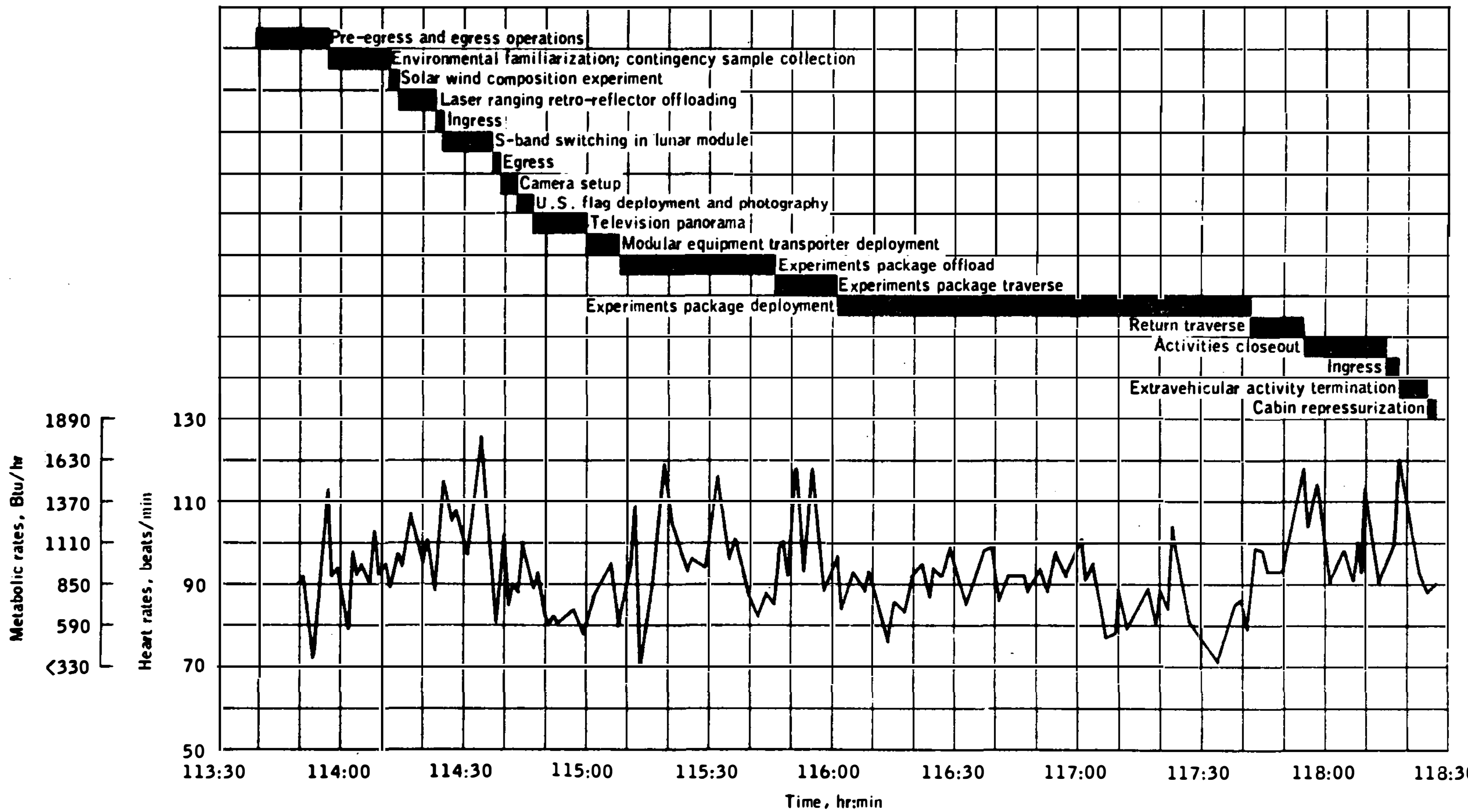
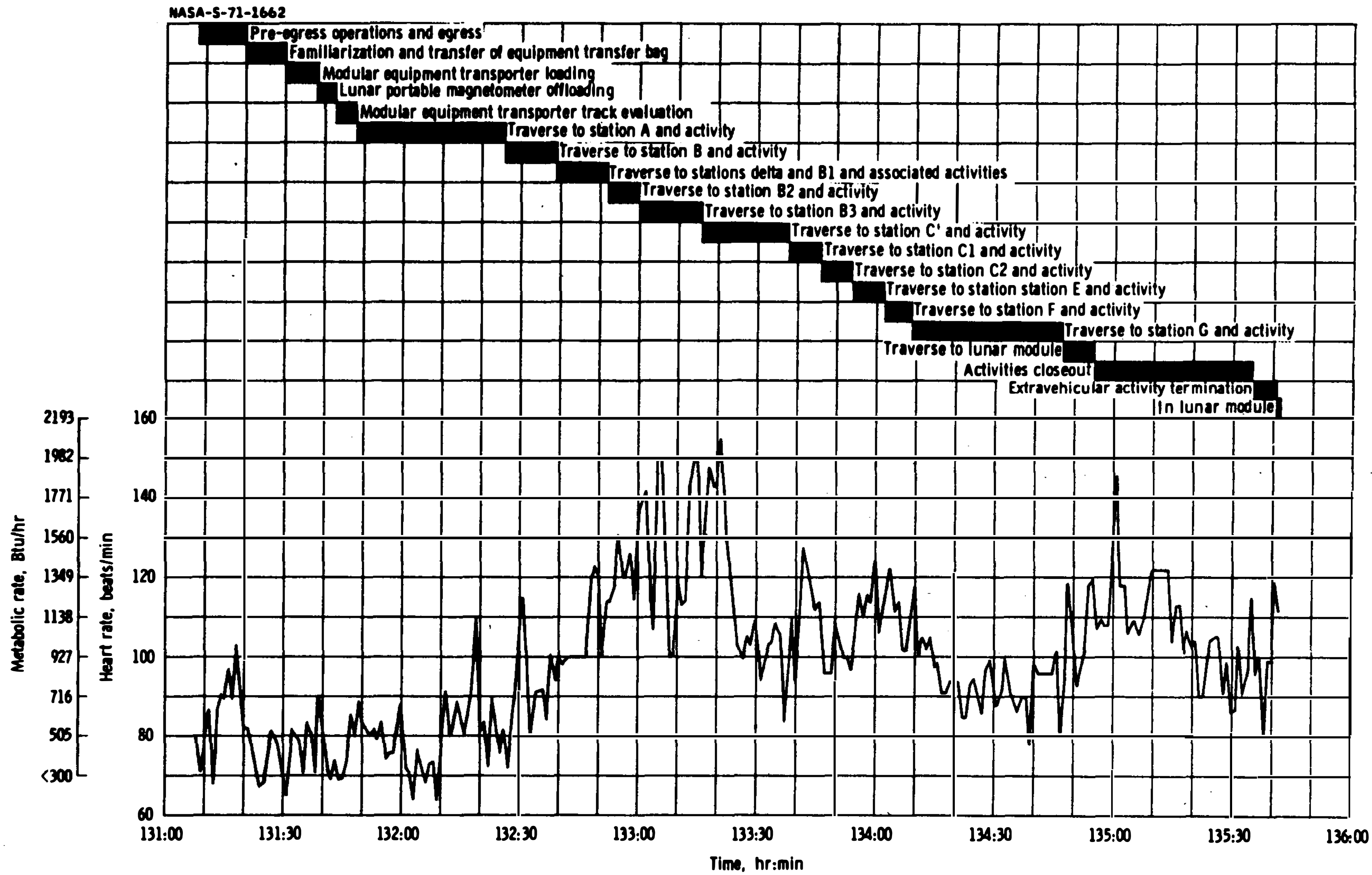


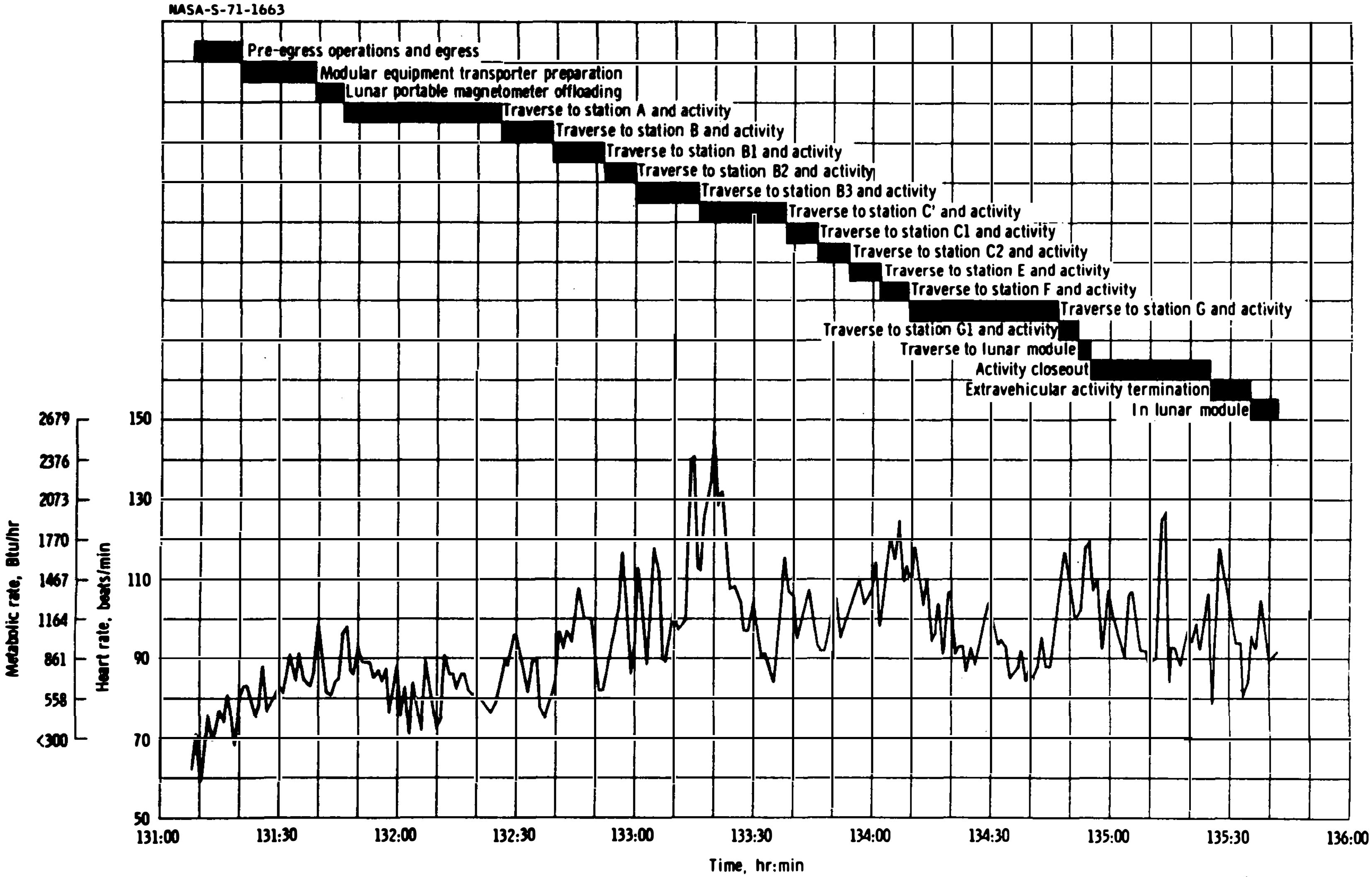
(b) Lunar Module Pilot.

Figure 10-4.- Concluded.



(a) Commander.

Figure 10-5.- Heart rates during second extravehicular activities.



(b) Lunar Module Pilot.

Figure 10-5.- Concluded.

TABLE 10-I.- METABOLIC ASSESSMENT OF THE FIRST EXTRAVEHICULAR ACTIVITY PERIOD

Surface activity[a]	Start time, hr:min	Duration, min	Average metabolic rate, Btu/hr	Metabolic production, Btu	Cumulative metabolic production[b] Btu
Commander					
Cabin depressurization	113:39	8	(b)	(b)	(b)
Egress	113:47	4	712	47	47
Environmental familiarization, modular equipment transporter unloading, and television deployment	113:51	21	1201	420	467
S-band antenna deployment	114:12	10	1052	175	642
Transferal of expendables	114:22	19	717	227	869
United States flag deployment and photography	114:41	6	726	73	942
Lunar module and site inspection	114:47	18	587	176	1118
Television transfer to scientific equipment bay	115:05	3	868	43	1161
Experiment package offloading	115:08	13	690	149	1310
Unknown activity	115:21	1	651	11	1321
Television positioning	115:22	3	840	42	1363
Modular equipment transporter loading	115:25	15	733	183	1546
Unknown activity	115:40	6	581	58	1604
Traverse to experiment package deployment site	115:46	15	984	246	1850
Unknown activity	116:01	3	677	34	1884
Experiment package system interconnect, passive seismic off-loading, laser ranging retro-reflector deployment	116:04	26	794	344	2228
Charged particle lunar environment experiment deployment	116:30	5	496	41	2269
Deployment of experiment package antenna, passive seismic experiment, and laser ranging retro-reflector; and sample collection	116:35	63	517	543	2812
Return traverse	117:38	16	1273	339	3151
Unknown activity	117:54	6	1735	174	3325
Sample collection	118:00	3	1165	58	3383
Extravehicular activity closeout	118:03	16	1029	274	3657
Ingress	118:19	4	1098	73	3730
Cabin repressurization	118:23	4	793	53	3783
Total	4:48	288	[c]800	[d]3783	3783
Lunar Module Pilot					
Cabin depressurization	113:39	8	(b)	(b)	(b)
Pre-egress operations	113:47	8	711	95	95
Egress	113:55	2	1582	53	148
Environmental familiarization, contingency sample collection	113:57	15	901	225	373
Deployment of solar wind composition experiment	114:12	2	1045	35	408
Laser ranging retro-reflector unloading	114:14	9	1061	159	567
Ingress	114:23	2	1265	42	609
S-band antenna switching	114:25	12	1195	239	848
Egress	114:37	2	889	30	878
Camera setup	114:39	4	883	59	937
United States flag deployment and photography	114:43	4	948	63	1000
Traverse to television	114:47	3	747	37	1037
Television panorama	114:50	10	620	103	1140
Modular equipment transporter deployment	115:00	8	746	99	1239
Experiment package offloading	115:08	38	1038	657	1896
Traverse to experiment package deployment site	115:46	15	1098	275	2171
Unknown activity	116:01	2	786	26	2197
Experiment package system interconnect, thumper and geophone unloading	116:03	23	786	301	2498
Mortar offload	116:26	3	972	49	2547
Unknown activity	116:29	5	778	65	2612
Suprathermal ion detector experiment unloading and deployment	116:34	11	905	156	2768
Penetrometer activity	116:45	2	795	26	2794
Geophone deployment	116:47	15	941	235	3029
Thumper activity	117:02	32	707	377	3406
Unknown activity	117:34	3	634	32	3438
Mortar pack arming	117:37	4	695	46	3484
Unknown activity	117:41	1	721	12	3496
Return traverse	117:42	12	1041	208	3704
Extravehicular activity closeout	117:54	21	1111	389	4093
Ingress	118:15	3	1231	62	4155
Extravehicular activity termination	118:18	5	1248	104	4259
Cabin repressurization	118:23	4	915	61	4320
Total	4:48	288	[c]930	[d]4320	4320

[a]Refer to figure 3-1 for lunar surface activity sites.
[b]An 8 minute loss of the biomedical data signal occurred at the beginning of the extravehicular activity period.
[c]Average value.
[d]The total metabolic production for the entire 4 hour 48 minute period, including metabolic production during the first 8 minutes, is 3840 and 4464 Btu for the Commander and Lunar Module Pilot, respectively.

TABLE 10-II.- METABOLIC ASSESSMENT OF THE SECOND EXTRAVEHICULAR PERIOD

Surface activity[a]	Starting time, hr:min	Duration, min	Average metabolic rate, Btu/hr	Metabolic production, Btu	Cumulative metabolic production, Btu
Commander					
Cabin depressurization	131:08	5	486	88	88
Egress	131:13	7	750	40	128
Familiarization and transferal of equipment transfer bag	131:20	8	423	56	184
Modular equipment transporter loading	131:28	10	410	68	252
Lunar portable magnetometer offloading	131:38	5	465	39	291
Evaluation of modular equipment transporter track	131:43	5	423	35	326
Lunar module to A traverse	131:48	6	562	56	382
Station A activity	131:54	32	509	271	653
A to B traverse	132:26	8	761	101	754
Station B activity	132:34	5	772	64	818
B to Delta traverse[b]	132:39	3	844	42	860
Station Delta activity	132:42	3	928	46	906
Delta to B1 traverse	132:45	3	1068	53	959
Station B1 activity	132:48	4	1228	82	1041
B1 to B2 traverse	132:52	5	1362	113	1154
Station B2 activity	132:57	3	1455	73	1227
B2 to B3 traverse	133:00	14	1492	348	1575
Station B3 activity	133:14	2	1655	55	1630
B3 to C' traverse	133:16	6	1810	181	1811
Station C' activity	133:22	16	1020	272	2083
C' to C1 traverse	133:38	2	970	32	2115
Station C1 activity	133:40	6	1272	127	2242
C1 to C2 traverse	133:46	6	945	95	2337
Station C2 activity	133:52	2	896	30	2367
C2 to E traverse	133:54	6	1244	124	2491
Station E activity	134:00	2	1128	38	2529
E to F traverse	134:02	4	1281	85	2614
Station F activity	134:06	3	940	47	2661
F to O traverse	134:09	2	1118	37	2698
Station G activity	134:11	36	779	467	3165
G to G1 traverse	134:47	2	1065	35	3200
Station G1 activity	134:49	3	935	47	3247
G1 to lunar module	134:52	3	1209	60	3307
Extravehicular activity closeout	134:55	40	1108	739	4046
Extravehicular activity termination	135:35	6	903	90	4136
Post-extravehicular activity operations and cabin repressurization	135:41	2	1180	20	4156
Total	4:35	275	[c]910	4156	4156
Lunar Module Pilot					
Cabin depressurization	131:08	12	410	82	82
Egress	131:20	1	633	11	93
Modular equipment transporter preparation	131:21	18	633	190	283
Lunar portable magnetometer offloading	131:39	5	756	63	346
Lunar portable magnetometer operation	131:44	2	921	31	377
Lunar module to A traverse	131:46	8	829	111	488
Station A activity	131:54	32	606	323	811
A to B traverse	132:26	8	840	112	923
Station B activity	132:34	5	555	46	969
B to Delta traverse	132:39	3	893	45	1014
Station Delta activity	132:42	2	1013	34	1048
Delta to B1 traverse	132:44	4	1272	85	1133
Station B1 activity	132:48	4	824	55	1188
B1 to B2 traverse	132:52	5	1154	96	1284
Station B2 activity	132:57	3	1336	67	1351
B2 to B3 traverse	133:00	14	1251	292	1643
Station B3 activity	133:14	2	1973	66	1709
B3 to C' traverse	133:16	6	2064	206	1917
Station C' activity	133:22	16	1142	304	2237
C' to C1 traverse	133:38	2	1283	43	2257
Station C1 activity	133:40	6	1160	116	2373
C1 to C2 traverse	133:46	6	1057	106	2479
Station C2 activity	133:52	2	1177	39	2518
C2 to E traverse	133:54	6	1337	134	2652
Station E activity	134:00	2	1341	45	2697
E to F traverse	134:02	4	1463	97	2794
Station F activity	134:06	3	1640	82	2876
F to G traverse	134:09	2	1551	52	2928
Station G activity	134:11	36	993	596	3524
G to G1 traverse	134:47	2	1504	50	3574
Station G1 activity	134:49	3	1260	63	3637
G1 to lunar module	134:52	3	1558	78	3715
Unknown activity	134:55	2	1415	47	3762
Extravehicular activity closeout	134:57	28	1082	904	4267
Extravehicular activity termination	135:25	10	1102	184	4451
Post-extravehicular activity operations and cabin repressurization	135:35	8	996	116	4567
Total	4:35	275	[c]1000	4567	4567

[a]Refer to figure 3-1 for lunar surface activity sites.
[b]Station Delta location is about 380 feet past Station B.
[c]Average value.

10.2 MEDICAL OBSERVATIONS

10.2.1 Adaptation to Weightlessness

Adaptation to the weightless state was readily accomplished. Shortly after orbital insertion, each crewman experienced the typical fullness-of-the-head sensation that has been reported by previous flight crews. No nausea, vomiting, vertigo, or disorientation occurred during the mission, and the crew did not observe distortion of facial features, such as rounding of the face due to lack of gravity, as reported by some previous crewmen.

During the first two days of flight, the crew reported discomfort and soreness of the lower back muscles as has been noted on previous missions. The discomfort was sufficient in magnitude to interfere with sleep during the first day of the mission, and was attributed to changes in posture during weightlessness. Inflight exercise provided relief.

10.2.2 Visual Phenomenon

Each crewman reported seeing the streaks, points, and flashes of light that have been noted by previous Apollo crews. The frequency of the light flashes averaged about once every 2 minutes for each crewman. The visual phenomenon was observed with the eyes both open and closed, and the crew was more aware of the phenomenon immediately upon awakening than upon retiring. In a special observation period set aside during the transearth coast phase, the Command Module Pilot determined that dark adaptation was not a prerequisite for seeing the phenomenon if the level of spacecraft illumination was low. Furthermore, several of the light flashes were apparently seen by two of the crewmen simultaneously. Coincidence of light flashes for two crewmen, if a true coincidence, would substantiate that the flashes originated from an external radiation source and would indicate that they were generated by extremely-high-energy particles, presumably of cosmic origin. Low-energy highly-ionizing particles would not have the range through tissue to have reached both crewmen.

10.2.3 Medications

No medications other than nose drops, to relieve nasal stuffiness caused by spacecraft atmosphere, were used during the mission. On the third day of flight, the Commander and the Lunar Module Pilot used one drop in each nostril. Relief was prompt and lasted for approximately 12 hours. The Command Module Pilot used the nose drops 3 hours prior to entry.

On this mission, the nasal spray bottles in the inflight medical kit were replaced by dropper bottles because previous crews had reported difficulties in obtaining medication from spray bottles in zero-g. The crew reported no problems associated with the dropper bottle.

10.2.4 Sleep

The shift of the crew's normal terrestrial sleep cycle during the first four days of flight was the largest experienced so far in the Apollo series. The displacement ranged from 7 hours on the first mission day to 11-1/2 hours on the fourth. The crew reported some difficulty sleeping in the zero-g environment, particularly during the first two sleep periods. They attributed the problem principally to a lack of kinesthetic sensations and to muscle soreness in the legs and lower back. Throughout the mission, sleep was intermittent; i.e., never more than 2 to 3 hours of deep and continuous sleep.

The lunar module crewmen received little, if any, sleep between their two extravehicular activity periods. The lack of an adequate place to rest the head, discomfort of the pressure suit, and the 7-degree starboard list of the lunar module caused by the lunar terrain were believed responsible for this insomnia. The crewmen looked out the window several times during the sleep period for reassurance that the lunar module was not starting to tip over.

Following transearth injection, the crew slept better than they had previously. The lunar module crewmen required one additional sleep period to make up the sleep deficit that was incurred while on the lunar surface.

The crewmen reported during postflight discussions that they were definitely operating on their physiological reserves because of inadequate sleep. This lack of sleep caused them some concern; however, all tasks were performed satisfactorily.

10.2.5 Radiation

The Lunar Module Pilot's personal radiation dosimeter failed to integrate the dosage properly after the first 24 hours of flight. To ensure that each lunar module crewman had a functional dosimeter while on the lunar surface, the Command Module Pilot transferred his unit to the Lunar Module Pilot on the fourth day of the mission. The final readings from the personal radiation dosimeters yielded net integrated (uncorrected) values of 640 and 630 millirads for the Commander and the Command Module Pilot, respectively. No value can be determined for the Lunar Module

Pilot. The total radiation dose for each crewman was approximately 1.15 rads to the skin and 0.6 rad at a 5-centimeter tissue depth. These doses are the largest observed on any Apollo mission; however, they are well below the threshold of detectable medical effects. The magnitudes of the radiation doses were apparently the result of two factors: (1) The trans-lunar injection trajectory lay closer to the plane of the geomagnetic equator than that of previous flights and, therefore, the spacecraft traveled through the heart of the trapped radiation belts. (2) The space radiation background was greater than previously experienced. Whole-body gamma spectroscopy was also performed postflight on the crew and indicated no cosmic ray induced radioactivity.

10.2.6 Water

The crew reported that the taste of the drinking water in both the command module and the lunar module was excellent. All eight scheduled inflight chlorinations of the command module water system were accomplished. Preflight testing of the lunar module potable water system showed that the iodine level in both water tanks was adequate for bacterial protection throughout the flight.

10.2.7 Food

The inflight food was similar to that of previous Apollo missions. Six new foods were included in the menu:

a. Lobster bisque (freeze dehydrated)

b. Peach ambrosia (freeze dehydrated)

c. Beef jerky (ready-to-eat bite-sized)

d. Diced peaches (thermostabilized)

e. Mixed fruit (thermostabilized)

f. Pudding (thermostabilized)

The latter three items were packaged in aluminum cans with easy-open, full-panel, pull-out lids. The crew did not report any difficulties either with removing the pull-out lids or eating the food contained in these cans with a spoon.

Prior to the mission, each crewman evaluated the available food items and selected his individual flight menu. These menus provided approximately 2100 calories per man per day. During most of the flight, the crew maintained a food consumption log. The Commander and the Lunar Module Pilot ate all the food planned for each meal, but the Command Module Pilot was satisfied with less.

Recovery-day physical examinations revealed that the Commander and the Lunar Module Pilot had maintained their approximate preflight weight, while the Command Module Pilot lost nearly 10 pounds. The Command Module Pilot stated that he would have preferred a greater quantity of food items requiring little or no preparation time.

10.3 PHYSICAL EXAMINATIONS

Each crewman received a comprehensive physical examination at 27, 15, and 6 days prior to launch, with brief examinations conducted daily during the last 5 days before launch.

Shortly after landing, a comprehensive physical examination showed that the crew was in good health. Both the Commander and the Command Module Pilot had a small amount of clear, bubbly fluid in the left middle-ear cavity and slight reddening of the eardrums. These findings disappeared in 24 hours without treatment. The Lunar Module Pilot had moderate eyelid irritation in addition to slight redness of the eardrums. All crewmen showed a mild temporary reaction to the micropore tape covering their biomedical sensors. This reaction subsided within 24 hours.

10.4 FLIGHT CREW HEALTH STABILIZATION

During previous Apollo missions, crew illnesses were responsible for numerous medical and operational difficulties. Three days before the Apollo 7 launch, the crew developed an upper respiratory infection which subsided before lift-off, but recurred inflight. Early on the Apollo 8 mission, one crewman developed symptoms of a 24-hour viral gastroenteritis which was epidemic in the Cape Kennedy area around launch time. About two days prior to the Apollo 9 flight, the crew developed common colds which necessitated a delay of the launch for three days. Nine days before the Apollo 13 launch, the backup Lunar Module Pilot developed German measles (rubella) and inadvertently exposed the prime Command Module Pilot. The day before launch, the prime Command Module Pilot was replaced by his backup counterpart because laboratory tests indicated that the prime crewman was not immune to this highly communicable disease with an incubation period of approximately two weeks.

In an attempt to protect the prime and backup flight crew members from exposure to communicable disease during the critical prelaunch and flight periods, such as experienced on previous flight, a flight crew health stabilization program was implemented. This program consisted of the following phases:

a. Identification, examination, and immunization of all primary contacts (personnel who required direct contact with the prime or backup crew during the last three weeks prior to flight).

b. Health and epidemiological surveillance of the crew members and the primary contacts, their families, and the community.

c. Certain modifications to facilities used for training and housing the crew, such as the installation of biological filters in all air conditioning systems.

d. Housing of both the prime and backup crew members in the crew quarters at the Kennedy Space Center from 21 days before flight until launch.

The flight crew health stabilization program was a complete success. No illnesses occurred during the preflight period in any of the prime or backup crew members. This result is of particular significance because the incidence of infectious disease within the local community was near a seasonal high during the prelaunch period.

10.5 QUARANTINE

No change in quarantine procedures were made on this mission, except as follows:

a. Two mobile quarantine facilities were used.

b. Two helicopter transfers of the crew and support personnel were performed.

The new procedures were implemented to return the crew to the Lunar Receiving Laboratory five days earlier than on previous lunar landing missions.

The crew and 14 medical support personnel were isolated behind the microbiological barrier in the Lunar Receiving Laboratory at Houston, Texas, on February 12, 1971. Daily medical examinations and periodic laboratory examinations showed no signs of illness related to lunar material exposure. No significant trends were noted in any biochemical, immunological, or hematological parameters in either the crew or the medical support personnel. On February 27, 1971, after 20 days of isolation within the Lunar Receiving Laboratory, the flight crew and the medical support personnel were released from quarantine. Quarantine for the spacecraft and samples of lunar material was terminated April 4, 1971.

11.0 MISSION SUPPORT PERFORMANCE

11.1 FLIGHT CONTROL

Flight control performance was satisfactory in providing timely operational support. Some problems were encountered and most are discussed in other sections of the report. Only those problems that are of particular concern to flight control operations or are not reported elsewhere are reported in this section.

All launch vehicle instrument unit analog data were lost just prior to lift-off. A faulty multiplexer within the instrument unit that processes the analog flight control data had failed. The flight controllers were able to recover most of the analog data from the S-IVB VHF downlink; however, because of its limited range, an early loss of data was experienced at 4 hours 27 minutes.

All launch vehicle digital computer data were lost at 3 hours and 5 minutes after launch. The vehicle, however, executed a normal propulsive vent about 29 minutes later indicating that the computer was operating properly. As a result of the loss of digital computer data, commands to the S-IVB had to be transmitted without verification of proper execution. The crew provided visual attitude information for the evasive maneuver.

High-gain antenna lockup problems were noted during revolution 12 lunar orbit operations. Because of this problem, a data storage equipment dump could not be accomplished to obtain data from the revolution 12 low-altitude landmark tracking operation. These data were to be used for powered descent targeting.

During revolution 12, the planned voice updates fell behind the timeline because of problems with the lunar module steerable antenna. Consequently, the powered descent was performed using the spacecraft forward and aft omnidirectional antennas and the 210-foot ground receiving antenna. Receiving of communications and high-bit-rate data were satisfactory except for some small losses when switching to the aft antenna late in the descent phase.

An abort command was set in the lunar module guidance computer and the indication was observed by Flight Control during lunar module activation, about 4 hours prior to scheduled powered descent initiation. A procedure was uplinked to the crew which reset the abort command and led to the conclusion that the abort switch had malfunctioned. Subsequently, the abort command reappeared three times and, each time, the command was reset by tapping on the panel near the abort switch. A procedure to inhibit the primary guidance system from going into an abort program was developed in the interval prior to powered descent, and was uplinked to the crew for manual entry into the computer. The first part of the four-part procedure was entered just prior to powered descent initiation and the other parts after throttle-up of the descent engine. Had an abort been required, it would have been accomplished using the abort guidance system and would have allowed reestablishment of the primary guidance system by keyboard entry after the abort.

A delay of approximately 50 minutes occurred in the first extravehicular activity because of the lack of satisfactory communications. The crew were receiving ground communications but the Mission Control Center was not receiving crew communications. The problem was corrected by resetting the Commander's audio circuit breaker which was not engaged.

The color television camera resolution gradually degraded during the latter portions of the first extravehicular activity. The degradation was caused by overheating resulting from 1.5 hours of operation while in the modular equipment stowage assembly prior to its deployment. The camera was turned off between the extravehicular periods for cooling, instead of leaving it operating as required by the flight plan. The camera picture resolution was satisfactory during the second extravehicular activity.

Three problems developed during the Apollo 14 mission that, had the crew not been present, would have prevented the achievement of the mission objectives. These problems involved the docking probe (section 7.1), the landing radar (section 8.4) and the lunar module guidance computer, described above. In each case, the crew provided ground personnel with vital information and data for failure analysis and development of alternate procedures. The crew performed the necessary activities and the required work-around procedures that allowed the mission to be completed as planned.

11.2 NETWORK

The Mission Control Center and the Manned Space Flight Network provided excellent support. There were only two significant problems. A defective transfer switch component caused a power outage at the Goddard Space Flight Center during lunar orbit. The power loss resulted in a 4 1/2-minute data loss. On lunar revolution 12, a power amplifier failure occurred at the Goldstone station. The problem was corrected by switching to a redundant system. The Network Controller's Mission Report for Apollo 14, dated March 19, 1971, published by the Manned Spacecraft Center, Flight Support Division, contains a summary of all Manned Space Flight Network problems which occurred during the mission.

11.3 RECOVERY OPERATIONS

The Department of Defense provided recovery support commensurate with mission planning for Apollo 14. Ship support for the primary landing area in the Pacific Ocean was provided by the helicopter carrier USS New Orleans. Active air support consisted of five SH-3A helicopters from the New Orleans and two HC-130 rescue aircraft staged from Pago Pago, Samoa. Two of the helicopters, designated "Swim 1" and "Swim 2", carried underwater demolition team personnel and the required recovery equipment. The third helicopter, designated "Recovery", carried the decontamination swimmer and the flight surgeon, and was utilized for the retrieval of the flight crew. The fourth helicopter, designated "Photo", served as a photographic platform for both motion-picture photography and live television coverage. The fifth helicopter, designated "Relay", served as a communications-relay aircraft. The ship-based aircraft were initially positioned relative to the target point; they departed station to commence recovery operations after the command module had been visually acquired. The two HC-130 aircraft, designated "Samoa Rescue 1" and "Samoa Rescue 2", were positioned to track the command module after it had exited from S-band blackout, as well as provide pararescue capability had the command module landed uprange or downrange of the target point. All recovery forces dedicated for Apollo 14 support are listed in table 11-I. Figure 11-1 illustrates the recovery force positions prior to predicted S-band acquisition time.

11.3.1 Command Module Location and Retrieval

The New Orleans' position was established using a navigation satellite (SRN-9) fix obtained at 2118 G.m.t. The ship's position at the time of command module landing was determined to be 26 degrees 59 minutes 30 seconds south latitude and 172 degrees 41 minutes west longitude. The command module landing point was calculated by recovery forces to be 27 degrees 0 minutes 45 seconds south latitude and 172 degrees 39 minutes 30 seconds west longitude.

The first electronic contact reported by the recovery forces was an S-band contact by Samoa Rescue 1. Radar contact was then reported by the New Orleans. A visual sighting was reported by the communications-relay helicopter and then by the New Orleans, Recovery, Swim 1 and Swim 2. Shortly thereafter, voice transmissions from the command module were received by the New Orleans.

The command module landed February 9, 1971, at 2105 G.m.t. and remained in the stable I flotation attitude. The VHF recovery beacon was activated shortly after landing, and beacon contact was reported by Recovery at 2107 G.m.t. The crew then turned off the beacon as they knew the recovery forces had visual contact.

TABLE 11-I.- APOLLO 14 RECOVERY SUPPORT

Type	Number	Ship name/ aircraft staging base	Area supported
		Ships	
ATF LCU	1 1	USS Paiute	Launch site area
DD	1	USS Hawkins	Launch abort area and West Atlantic earth-orbital recovery zone
LSD	1	USS Spiegel Grove	Deep-space secondary landing areas on the Atlantic Ocean line
DD	1	USS Carpenter	Mid-Pacific earth-orbital recovery zone
LPH	1	USS New Orleans	Deep-space secondary landing areas on the mid-Pacific line and the primary end-of-mission landing area
		Aircraft	
HH-53C	3	Patrick Air Force Base	Launch site area
HC-130	[a]1	McCoy Air Force Base	Launch abort area, West Atlantic recovery zone, contingency landing area
HC-130	[a]1	Pease Air Force Base	Launch abort area, West Atlantic recovery zone
HC-130	[a]1	Lajes Field, Azores	Launch abort area, earth orbital contingency landing area
HC-130	[a]1	Ascension Island	Atlantic Ocean line and contingency landing area
HC-130	[a]2	Hickam Air Force Base	Mid-Pacific earth orbital recovery zone, deep-space secondary landing area and primary end-of-mission landing area
SH-3A	5	USS New Orleans	Deep-space secondary landing area and primary end-of-mission landing area

[a]Plus one backup

NASA-S-71-1664

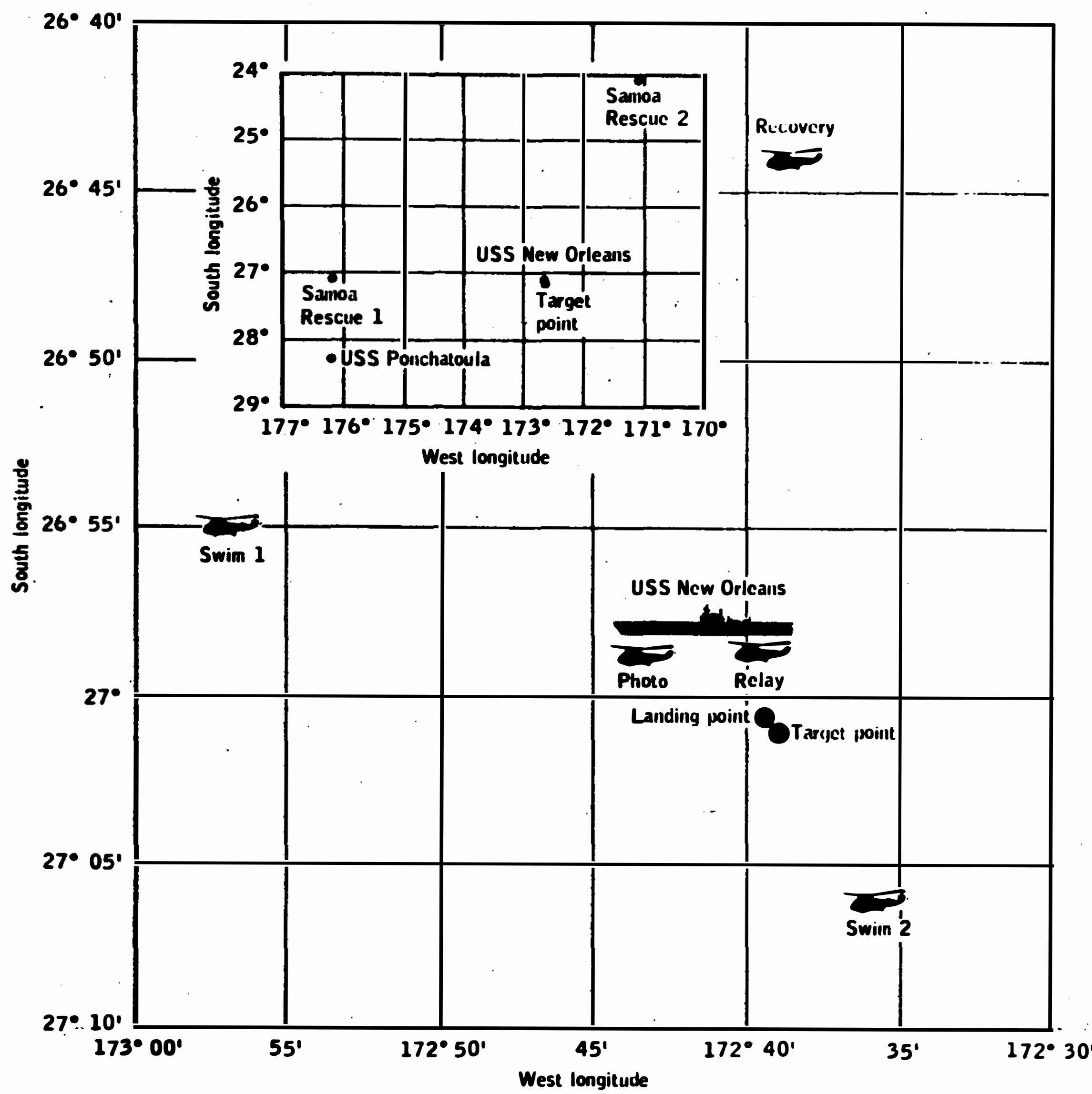


Figure 11-1.- End-of-mission recovery support.

After confirming that the command module and the crew were in good condition, Swim 2 attempted to retrieve the main parachutes, and swimmers were deployed to the command module to install the flotation collar. Recovery forces were unable to retrieve any of the main parachutes, but did retrieve two drogue parachute covers and one sabot. The decontamination swimmer was deployed to pass flight suits and respirators to the crew and assist them from the command module into the life raft. The flight crew were onboard the recovery helicopter 7 minutes after they had egressed the command module and were aboard the New Orleans 5 minutes later. Command module retrieval took place at 27 degrees 2 minutes south latitude and 172 degrees 4 minutes west longitude at 2309 G.m.t.

The flight crew remained aboard the New Orleans in the mobile quarantine facility until they were flown to Pago Pago, Samoa, where they transferred to a second mobile quarantine facility aboard a C-141 aircraft. The crew was flown to Ellington Air Force Base, with a stop at Norton Air Force Base, California, where the aircraft was refueled.

After arrival of the New Orleans at Hawaii, the command module was offloaded and taken to Hickam Air Force Base for deactivation. Upon completion of deactivation, the command module was transferred to Ellington Air Force Base via a C-133 aircraft, arriving on February 22, 1971.

The following is a chronological listing of events during the recovery and quarantine operations.

Event	Time G.m.t.	Time relative to landing days:hr:min
	Feb. 9, 1971	
S-band contact by Samoa Rescue 1	2055	-0:00:10
Radar contact by New Orleans	2056	-0:00:09
Visual contact by "Relay" helicopter	2100	-0:00:05
Voice contact with flight crew	2101	-0:00:04
Command module landing	2105	0:00:00
Swimmers deployed to command module	2112	0:00:07
Flotation collar installed and inflated	2120	0:00:15
Decontamination swimmer deployed	2127	0:00:22
Hatch opened for crew egress	2140	0:00:35
Flight crew in egress raft	2141	0:00:36
Flight crew aboard helicopter	2148	0:00:43
Flight crew aboard New Orleans	2153	0:00:48
Flight crew in mobile quarantine facility	2203	0:00:58
Command module aboard New Orleans	2309	0:02:04
	Feb. 11, 1971	
First sample flight departed ship	0355	1:05:00
Flight crew departed ship	1746	1:18:51
First sample flight arrived Houston (via Samoa and Hawaii)	2057	1:22:02
	Feb. 12, 1971	
Flight crew arrived Houston	0934	2:10:39
Flight crew arrived at Lunar Receiving Laboratory	1135	2:12:40
	Feb. 17, 1971	
Mobile quarantine facility and command module offloaded in Hawaii	2130	7:22:35
	Feb. 18, 1971	
Mobile quarantine facility arrived Houston	0740	8:08:45
	Feb. 19, 1971	
Reaction control system deactivation completed	2300	10:00:05
	Feb. 22, 1971	
Command module arrived Houston	2145	12:22:50
Command module delivered to Lunar Receiving Laboratory	2330	13:00:35

11.3.2 Postrecovery Inspection

The docking probe was removed from the command module and secured in the mobile quarantine facility for return to Houston. Otherwise, all aspects of the command module postrecovery operations, the mobile quarantine facility operations and lunar sample return operations were normal with the exception of the following discrepancies noted during command module inspection.

a. There was an apparent chip (1-inch wide, 3-inches long, and 1/2-inch deep) in the minus Z quadrant of the heat shield adjacent to the small heat sensor, about 30-inches inboard from the lip of the heat shield. However, the heat shield can be considered to be in normal post-reentry condition.

b. There was a film layer on all windows ranging from approximately 10-percent coverage on the left side window to 100-percent on the right side window.

c. The backup method was used to obtain the water samples because the direct oxygen valve had been left slightly open, causing the primary pressurization system to lose pressure.

d. The chlorine content of the potable water was not analyzed on the ship because of lack of time.

e. The Commander's radiation dosimeter was broken and no reading was obtained. The other two dosimeters were left aboard the command module.

12.0 ASSESSMENT OF MISSION OBJECTIVES

The four primary objectives (ref. 7) assigned to the Apollo 14 mission were as follows:

a. Perform selenological inspection, survey, and sampling of materials in a preselected region of the Fra Mauro formation.

b. Deploy and activate the Apollo lunar surface experiments package.

c. Develop man's capability to work in the lunar environment.

d. Obtain photographs of candidate exploration sites.

Eleven detailed objectives (derived from primary objectives) and sixteen experiments (listed in table 12-I and described in ref. 8) were assigned to the mission. All detailed objectives, with the following exceptions, were successfully completed:

a. Photographs of a candidate exploration site

b. Visibility at high sun angles

c. Command and service module orbital science photography

d. Transearth lunar photography

On the basis of preflight planning data, these four objectives were only partially satisfied.

Two detailed objectives were added and were performed during translunar coast: S-IVB photography and command and service module water-dump photography. The S-IVB could not be identified on the film during postflight analysis and, although some particles were seen on photographs of the water dump, there was no indication of the "snow storm" described by the crew.

In addition to the spacecraft and lunar surface objectives, the following two launch vehicle objectives were assigned and completed:

a. Impact the expended S-IVB/instrumentation unit on the lunar surface under nominal flight profile conditions.

b. Make a postflight determination of the S-IVB/instrumentation unit point of impact within 5 kilometers and the time of impact within one second.

TABLE 12.I.- DETAILED OBJECTIVES AND EXPERIMENTS

Description	Completed
Detailed objectives	
Contingency sample collection	Yes
Photographs of a candidate exploration site	Partial
Visibility at high sun angles[a]	Partial
Modular equipment transporter evaluation	Yes
Selenodetic reference point update	Yes
Command and service module orbital science photography	Partial
Assessment of extravehicular activity operation limits	Yes
Command and service module oxygen flow rate	Yes
Transearth lunar photography	Partial
Thermal coating degradation	Yes
Dim-light photography	Yes
Experiments	
Apollo lunar surface experiments package:	
M-515 Lunar dust detector	Yes
S-031 Lunar passive seismology	Yes
S-033 Lunar active seismology	Yes
S-036 Suprathermal ion detector	Yes
S-058 Cold cathode gauge	Yes
S-038 Charged particle lunar environment	Yes
S-059 Lunar geology investigation	Yes
S-078 Laser ranging retro-reflector	Yes
S-200 Soil mechanics	Yes
S-198 Portable magnetometer	Yes
S-170 Bistatic radar	Yes
S-080 Solar wind composition	Yes
S-178 Gegenschein from lunar orbit	Yes
S-164 S-band transponder	Yes
S-176 Apollo window meteroid	Yes
M-078 Bone mineral measurement	Yes

[a]Preliminary analysis indicates that sufficient data were collected to verify that the visibility analytical model can be used for Apollo planning purposes.

The impact of the S-IVB was detected by the Apollo 12 passive seismic experiment. The impact of the spent lunar module ascent stage was detected by both the Apollo 12 and Apollo 14 passive seismic experiments.

12.1 PARTIALLY COMPLETED OBJECTIVES

12.1.1 Photographs of a Candidate Exploration Site

Four photographic passes to obtain Descartes landing data were scheduled: one high-resolution sequence with the lunar topographic camera at low altitude, two high-resolution sequences with the lunar topographic camera at high altitude and one stereo strip with the Hasselblad electric data camera at high altitude. On the low altitude (revolution 4) lunar topographic camera pass, the camera malfunctioned and, although 192 frames were obtained of an area east of Descartes, no usable photography was obtained of Descartes. On the subsequent high-altitude photographic passes, the electric Hasselblad camera with the 500-mm lens was used instead of the lunar topographic camera. Excellent Descartes photography was obtained during three orbits, but the resolution was considerably lower than that possible with the lunar topographic camera. Another problem was encountered during the stereo strip photographic pass. Because the command and service module S-band high-gain antenna did not operate properly, no usable high-bit-rate telemetry, and consequently, no camera shutter-open data were obtained for postflight data reduction.

12.1.2 Visibility at High Sun Angles

Four sets of zero-phase observations by the Command Module Pilot were scheduled in order to obtain data on lunar surface visibility at high sun elevation angles. The last set, scheduled for revolution 30, was deleted to provide another opportunity to photograph the Descartes area. Good data were obtained from the first three sets.

12.1.3 Command and Service Module Orbital Science Photography

All objectives were completed with the exception of those that specified use of the lunar topographic camera. The Apollo 13 S-IVB impact crater area was photographed using the electric Hasselblad 70-mm camera with the 500-mm lens as a substitute for the inoperable lunar topographic camera.

12.1.4 Transearth Lunar Photography

Excellent photography of the lunar surface with the electric Hasselblad data camera using the 80-mm lens was obtained. No lunar topographic camera photography was obtained because of the camera malfunction.

12.2 INFLIGHT DEMONSTRATIONS

In addition to detailed objectives and experiments, four zero-gravity inflight demonstrations were conducted. They were performed on a noninterference basis at the crew's option. The four inflight demonstrations and responsible NASA centers were:

a. Electrophoretic separation - Marshall Space Flight Center

b. Heat flow and convection - Marshall Space Flight Center

c. Liquid transfer - Lewis Research Center

d. Composite casting - Marshall Space Flight Center.

12.3 APPROVED OPERATIONAL TESTS

The Manned Spacecraft Center participated in two of eight approved operational tests. Operational tests are not required to meet the objectives of the mission, do not affect the nominal timeline, and add no payload weight. The two operational tests were: lunar gravity measurement (using the lunar module primary guidance system) and a hydrogen maser test (a Network and unified S-band investigation sponsored by the Goddard Spaceflight Center). Both tests were completed, and the results of the hydrogen maser test are given in reference 9.

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

a. Chapel Bell (classified Department of Defense test)

b. Radar Skin Tracking

c. Ionospheric Disturbance from Missiles

d. Acoustic Measurement of Missile Exhaust Noise

e. Army Acoustic Test

f. Long-Focal-Length Optical System.

13.0 LAUNCH PHASE SUMMARY

13.1 WEATHER CONDITIONS

Cumulus clouds existed in the launch complex area with tops at 15 000 feet 20 minutes prior to the scheduled launch and with tops at 18 000 feet 10 minutes later. During this time, the ground-based electric field meters clearly showed fluctuating fields characteristic of mildly disturbed weather conditions. Since the mission rules do not allow a launch through cumulus clouds with tops in excess of 10 000 feet, a 40-minute hold was required before a permissible weather situtation existed. At launch, the cloud bases were at 4000 feet with tops to 10 000 feet. The launch under these conditions did not enhance the cloud electric fields enough to produce a lightning discharge, thus providing further confidence in the present launch mission rules.

13.2 ATMOSPHERIC ELECTRICITY EXPERIMENTS

As a result of the lightning strikes experienced during the Apollo 12 launch, several experiments were performed during the launch of Apollo 13 and Apollo 14 to study the effects of the space vehicle on the atmospheric electrical field during launch. Initially, it was hoped that the effects could be related simply to the electrical-field-enhancement factor of the vehicle. However, the results of the Apollo 13 measurements showed that the space vehicle produced a much stronger electrical field disturbance than had been expected and also produced some low-frequency radio noise. This disturbance may have been caused by a buildup of electrostatic charges in the exhaust cloud, charge buildup on the vehicle, or a combination of both of these sources. To define the origin and the carriers of the charge, additional experiments were performed during the Apollo 14 launch to study the electric field phenomena in more detail, to measure radio noise, and to measure the temperature of the Saturn V exhaust plume, which is an important parameter in calculating the electrical breakdown characteristics of the exhaust. The preliminary findings of these experiments are given here. When analyses of data have been completed, a supplemental report will be issued (appendix E).

13.2.1 Electrical Field Measurements

Atmospheric electrical field measurements were made by the New Mexico Institute of Mining and Technology and the Stanford Research Institute at the locations shown in figure 13-1. In addition, a field measuring instrument (field mill) was installed on the launch umbilical

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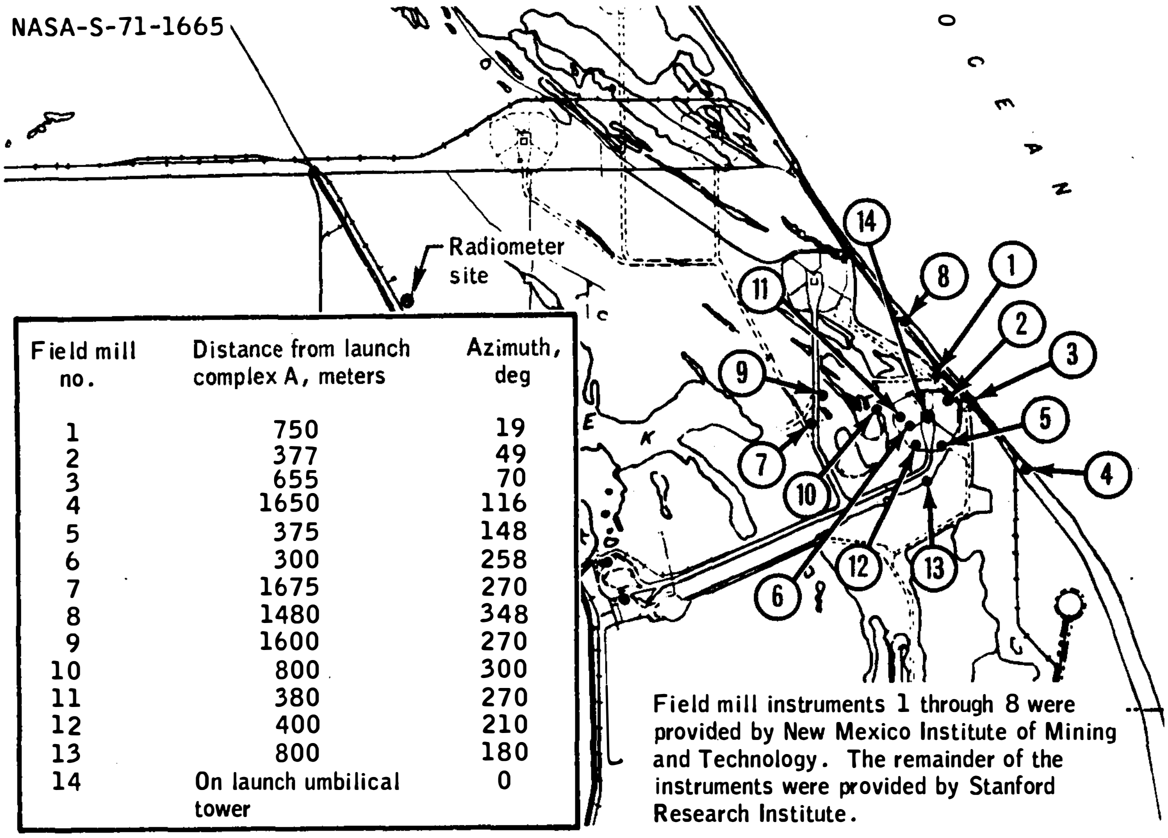

Field mill no.	Distance from launch complex A, meters	Azimuth, deg
1	750	19
2	377	49
3	655	70
4	1650	116
5	375	148
6	300	258
7	1675	270
8	1480	348
9	1600	270
10	800	300
11	380	270
12	400	210
13	800	180
14	On launch umbilical tower	0

Figure 13-1.- Field mill locations at the launch site.

tower to detect any charge buildup on the vehicle during ignition and the initial seconds after lift-off. Accurate timing signals, which were not available on Apollo 13, were provided to most of the field measurement equipment locations on Apollo 14. Time-lapse photographs of the launch cloud were also taken to aid in the interpretation of the data. Like Apollo 13, the Apollo 14 launch produced a significant electrical disturbance in the field mill records (fig. 13-2). Although the data are still being analyzed, some preliminary observations can be made.

Prior to the Apollo 13 launch, the field mills indicated stable fine-weather fields of 100 to 200 volts per meter. Before the Apollo 14 launch, however, the fields were fluctuating several hundred volts per meter, positive and negative. This behavior was entirely consistent with the difference in weather conditions — good conditions for Apollo 13 but mild disturbances for Apollo 14.

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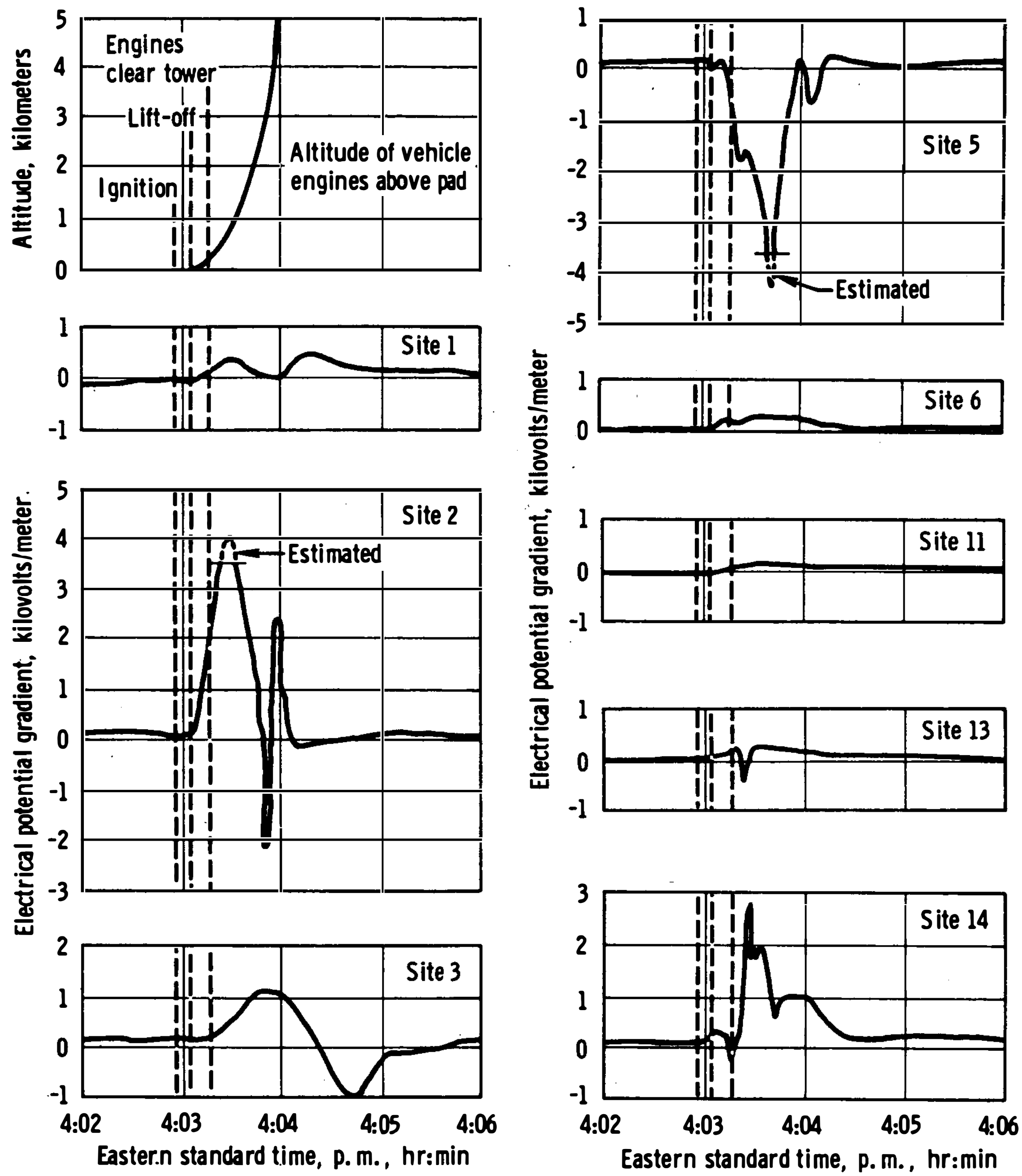

Note: Location of sites can
be seen on figure 13-1.

Figure 13-2.- Potential gradient data during launch.

During the Apollo 13 launch, the instruments at sites west of the launch complex registered a smooth positive field increase, succeeded by a less pronounced negative excursion. For Apollo 14, the negative excursion was not evident; however, the field variations occurred at approximately equivalent times for both launches. The positive excursion was approximately five times greater for Apollo 13 than for Apollo 14, and reached maximum when the space vehicle was at altitudes greater than 1000 meters. This observation, coupled with the fact that the maximum electric fields were observed downwind on both launches makes it unlikely that the space vehicle charge was the dominant factor but, rather, that the positively charged clouds were the dominant sources of the electric fields.

During lift-off, the swiftly moving exhaust clouds are channeled both north and south through the flame trough. The principal cloud which moved through the north end of the flame trough was composed largely of condensed spray water and contained a positive charge of approximately 50 millicoulombs and a field of approximately 4000 volts/meter (Site 2 of fig. 13-2). The cloud that exhausted to the south had much less water and contained about a 5-millicoulomb negative charge. The cloud also appeared to contain solid particulate matter which rapidly fell out.

The field mill on the launch umbilical tower indicated a small positive value (<400 volts/meter) a few seconds after lift-off. Model measurements using a 1/144-scale model of the launch umbilical tower and the Apollo/Saturn vehicle indicated that, in this configuration, the launch umbilical tower field and the vehicle potential are related by volts/field = 20 meters. Thus, the vehicle potential is less than 8000 volts (400 × 20). A comparison of the launch umbilical tower record with the data from the other sites indicates that the charge on the vehicle appears to be less than 1 millicoulomb.

13.2.2 Radio Noise Measurements

Narrow-band radio receivers operating at frequencies of 1.5, 6, 27, 51, and 120 kHz were located at camera pad 5 (field mill site 11) together with a broadband detector. As in the case of Apollo 13, signals were detected at several different frequencies, but the time behavior of different frequency components was not the same during the two launches.

The loop-antenna data (fig. 13-3) indicate a large increase in noise on the 1.5-kHz and 6-kHz channels 3 seconds after engine ignition, while the noise on the 51-kHz channel did not begin until 2 seconds after lift-off (about 11 seconds after ignition). Initially, it appeared that the 1.5- and 6-kHz data might not represent radiated electromagnetic noise, rather, microphonic noise generated by some component of the system such as the loop antenna preamplifier. Preliminary data from the electric

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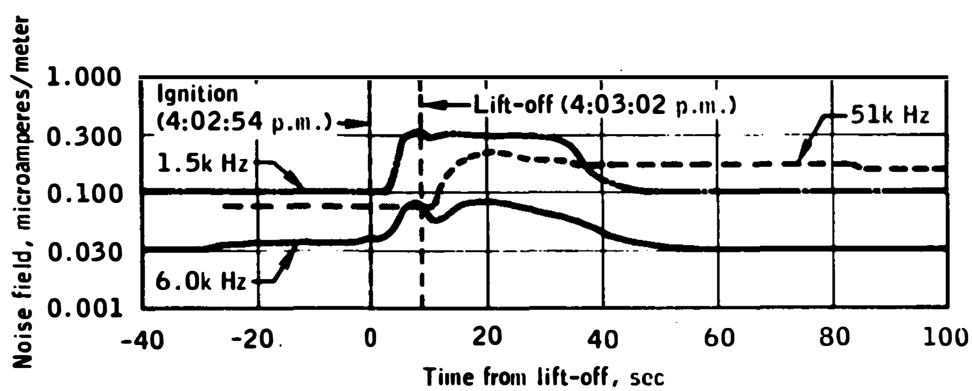

Figure 13-3.- Noise recorded by loop antenna system.

dipole antenna at camera pad 5, however, indicate the same general behavior, and as the two antenna systems use separate amplifiers, it appears that the data are valid. An abrupt cessation of the 1.5- and 6-kHz noise by both systems prior to the loss of the 51-kHz noise is not understood and further studies of the noise data are presently being made.

13.2.3 Plume Temperature Measurements

The temperature characteristics of the Saturn V exhaust plume were studied from a site about 5 miles west of the launch complex using a two-channel radiometer system operating at 1.26 and 1.68 microns. The radiometers viewed a narrow horizontal section of the exhaust plume which, in turn, provided temperature as a function of distance down the plume as the vehicle ascended vertically. Figure 13-4 shows the measured plume temperature as a function of distance behind the vehicle. These results are now being used to improve the theoretical calculations of the electrical characteristics of the exhaust plume. It appears that the plume may be a reasonable electrical conductor over a length of some 200 feet. This result is consistent with the low value of vehicle potential when the vehicle is passing the launch umbilical tower field meter since, at that time, the vehicle is probably still effectively connected electrically to earth. (Reference 10 contains additional information concerning plume temperature measurements.)

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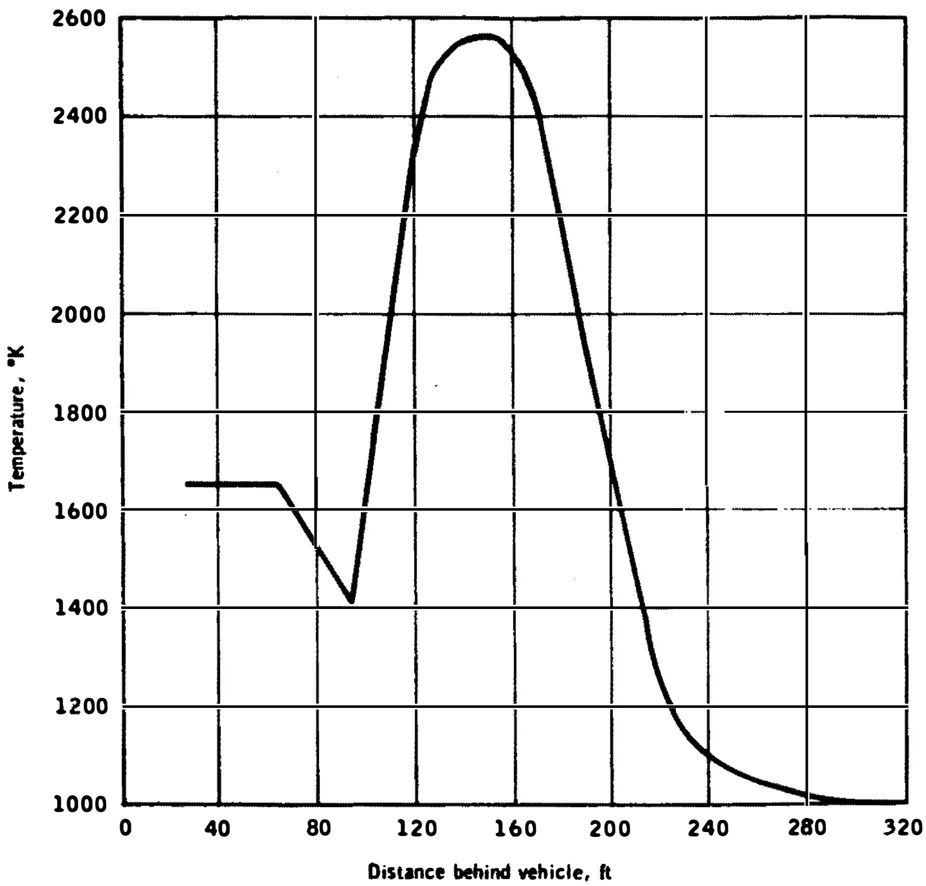

Figure 13-4.- Exhaust plume temperature characteristics.

13.3 LAUNCH VEHICLE SUMMARY

The seventh manned Saturn V Apollo space vehicle, AS-509, was launched on an azimuth 90 degrees east of north. A roll maneuver was initiated at 12.8 seconds that placed the vehicle on a flight azimuth of 75.558 degrees east of north. The trajectory parameters from launch to translunar injection were close to nominal with translunar injection achieved 4.9 seconds earlier than nominal.

All S-IC propulsion systems performed satisfactorily. Total propellant consumption rate was 0.42 percent higher than predicted with the consumed mixture ratio 0.94 percent higher than predicted. Specific impulse was 0.23 percent higher than predicted.

The S-II propulsion system performed satisfactorily. Total propellant flow rate was 0.12 percent below predicted and specific impulse was 0.19 percent below predicted. Propellant mixture ratio was 0.18 percent above predicted. The pneumatically actuated engine-mixture-ratio control valves operated satisfactorily. Engine start tank conditions were marginal at S-II engine start command because of the lower start tank relief valve settings caused by warmer-than-usual start tank temperatures. These warmer temperatures were a result of the hold prior to launch.

The S-IVB stage engine operated satisfactorily throughout the operational phase of first and second firings and had normal shutdowns. The S-IVB first firing time was 4.1 seconds less than predicted. The restart at the full-open propellant utilization valve position was successful. S-IVB second firing time was 5.5 seconds less than predicted. The total propellant consumption rate was 1.38 percent higher than predicted for the first firing and 1.47 percent higher for the second firing. Specific impulses for each were proportionally higher.

The structural loads experienced were below design values. The maximum dynamic pressure period bending moment at the S-IC liquid oxygen tank was 45 percent of the design value. The thrust cutoff transients were similar to those of previous flights. The S-II stage center engine liquid oxygen feedline accumulator successfully inhibited the 14- to 16-hertz longitudinal oscillations experienced on previous flights. During the maximum dynamic pressure region of flight, the launch vehicle experienced winds that were less than 95-percentile January winds.

The S-IVB/instrument unit lunar impact was accomplished successfully. At 82:37:52.2 elapsed time from lift-off, the S-IVB/instrument unit impacted the lunar surface at approximately 8 degrees 5 minutes 35 seconds south latitude and 26 degrees 1 minute 23 seconds west longitude, approximately 150 miles from the target of 1 degree 35 minutes 46 seconds south latitude and 33 degrees 15 minutes west longitude. Impact velocity was 8343 ft/sec.

The ground systems, supporting countdown and launch, performed satisfactorily. System component failures and malfunctions requiring corrective action were corrected during countdown without causing unscheduled holds. Propellant tanking was accomplished satisfactorily. Damage to the pad, launch umbilical tower, and support equipment was minor.

14.0 ANOMALY SUMMARY

This section contains a discussion of the significant anomalies that occurred during the Apollo 14 mission. The discussion of these items is divided into four major areas: command and service modules; lunar module; government-furnished equipment; and Apollo lunar surface experiments package.

14.1 COMMAND AND SERVICE MODULES

14.1.1 Failure to Achieve Docking Probe Capture Latch Engagement

Six docking attempts were required to successfully achieve capture latch engagement during the transposition and docking event. Subsequent inflight examination of the probe showed normal operation of the mechanism. The lunar orbit undocking and docking were completely normal. Data analysis of film, accelerometers, and reaction control system thruster activity indicates that probe-to-drogue contact conditions were normal for all docking attempts, and capture should have been achieved for the five unsuccessful attempts (table 14-I). The capture-latch assembly must not have been in the locked configuration during the first five attempts based on the following:

a. The probe status talkback displays functioned properly before and after the unsuccessful attempts, thus indicating proper switch operation and power to the talkback circuits. The talkback displays always indicated that the capture latches were in the cocked position during the unsuccessful attempts (fig. 14-1). (Note that no electrical power is required to capture because the system is cocked prior to flight and the capture operation is strictly mechanical and triggered by the drogue.)

b. Each of the six marks on the drogue resulted from separate contacts by the probe head (fig. 14-2). Although three of the marks are approximately 120 degrees apart, a docking impact with locked capture latches should result in three double marks (to match the latch hooks) 120 degrees apart, and within one inch of the drogue apex or socket. Although the drogue marks could indicate that the individual capture-latch hooks were difficult to depress, such marks are not abnormal for impact velocities greater than 0.25 feet per second.

Since the latches were not locked, the anomaly was apparently caused by failure of the capture-latch plunger (fig. 14-1) to reach the forward or locked position. Motion of the plunger could have been restricted by

TABLE 14-I.- RELATED DATA AND FILM INVESTIGATION RESULTS

Docking attempt	Contact, hr:min:sec	Estimated velocity, ft/sec	Contact position, clock-oriented	[a]Socket contact time, seconds	+X thrusting after contact, seconds	Comments
1A	3:13:53.7	0.1	11:00	1.55	None	a. No thruster activity b. Contact moderately close to apex
1B	3:14:01.5	[b]0.14 max	9:00	1.65	None	Contact close to apex
1C	3:14:04.45	[b]0.14 max	4:30	1.4	0.55	Contact close to apex
1D	3:14:09.0	[b]0.29 max	4:00	2.35	1.95	Contact close to apex
2	3:14:43.7	0.4 to 0.5	8:30	1.7	None	Contact close to apex
3	3:16:43.4	0.4	7:00	2.45	None	Contact close to apex
4	3:23:41.7	0.4 to 0.5	3:00	6.5	6.2	Contact close to apex
5	4:32:29.3	0.25	6:00	2.9	None	Contact close to apex
6	4:56:44.9	0.2	7:00	In and hard docked	14.3	a. Contact moderately close to apex b. Retract cycle began 6.9 seconds after contact c. Initial latch triggered 11.8 seconds after contact

[a]The maximum capture-latch response time is 80 milliseconds.

[b]Estimated velocity from X-thruster activity time. These are maximums with some velocity being used to null out small separation velocity. Other velocities were estimated by film interpretation.

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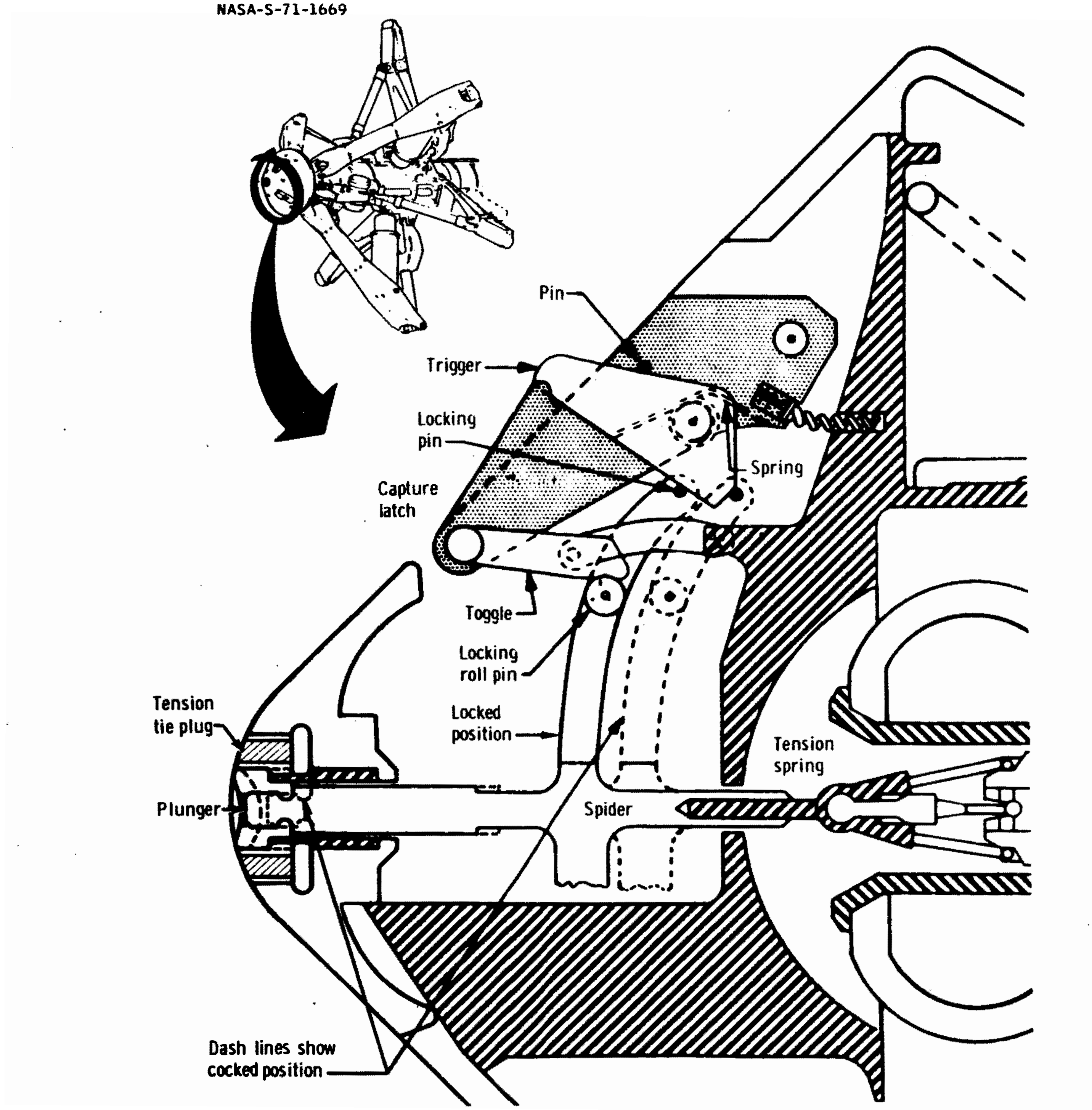

Figure 14-1.- Cross section of probe head and capture-latch assembly.

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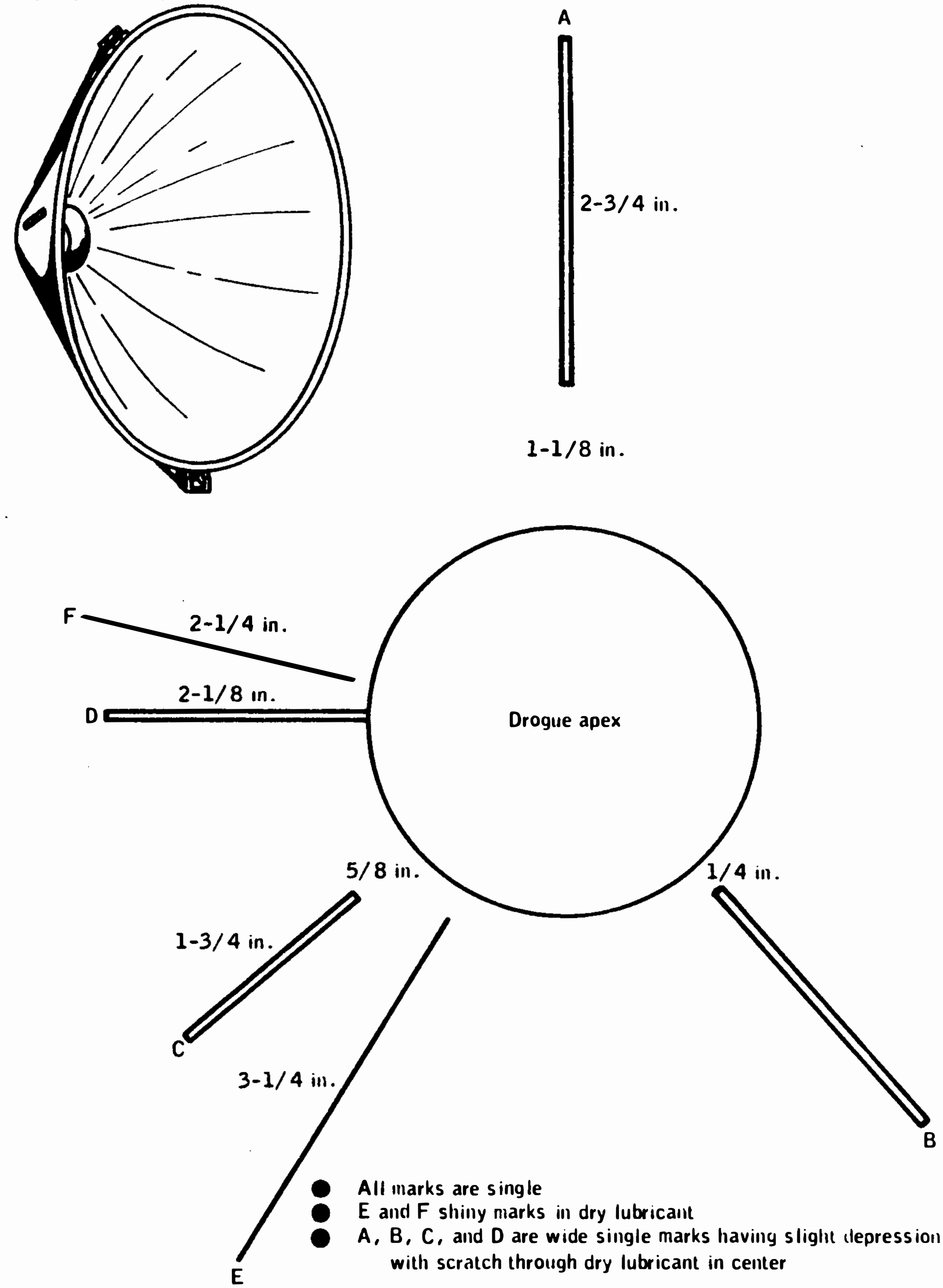

Figure 14-2.- Location of marks on drogue assembly.

contamination or dimensional changes due to temperature. Internal damage to the capture-latch mechanism can be ruled out because the system functioned properly in all subsequent operations following the sixth docking attempt and during postflight testing.

Analyses were performed to investigate tolerances and thermal effects on mating parts and surfaces associated with the operation of the capture latches. The results indicate that neither temperature nor tolerances could have caused the problem. In addition, a thermal analysis shows that neither the latches nor the spider could have been jammed by ice.

Tests using qualification probes to determine capture-latch response measurements were made and showed no aging degradation of the system. Tension tie tests produced clearly sheared pins; however, in one test, a sheared portion of the pin did leave the tension tie with some velocity and landed outside the ring itself.

No contamination, corrosion, significant debris, or foreign materials were found, and the mechanism worked normally during all functional tests. The loads and response times compared with the specifications and with the probe preflight data. Motor torque values and actuator assembly torque values (static drag and capture-latch release) compare favorably with preflight values.

During the inspection, small scratches and resulting burrs were found on the tension tie plug wall adjacent to the plunger. The scratches are being analyzed. An anomaly report will be issued under separate cover when the investigation has been completed.

The most probable cause of the problem was contamination or debris which later became dislodged. A cover will be provided to protect the probe tip from foreign material entering the mechanism prior to flight.

This anomaly is open.

14.1.2 High-Gain Antenna Tracking Problems

During translunar coast and lunar orbit operations, occasional problems were encountered in acquiring good high-gain antenna tracking with either the primary or secondary electronics. The specific times of high-gain antenna acquisition and tracking problems were:

a. 76:45:00 to 76:55:00
b. 92:16:00 to 93:22:00
c. 97:58:00 to 98:04:02
d. 99:52:00.

An instrumentation problem with the antenna readout occurred for about 3 hours early in the mission when an error of about 30 degrees existed. Subsequently, the readings were normal. A mechanical interference in the instrument servos is the most likely cause. The instrument servos are an independent loop which drive the antenna pitch and yaw meters in the command module. No corrective action is planned since the servos do not affect the antenna performance for any modes of operation.

The ground data signatures which show the first acquisition and tracking problems are illustrated in figure 14-3. The antenna started tracking a point approximately 5 to 8 degrees off the earth pointing angle at 76:45:00 elapsed time and continued tracking with low uplink and downlink signal levels for 10 minutes at which time a good narrow beam lock-up was achieved.

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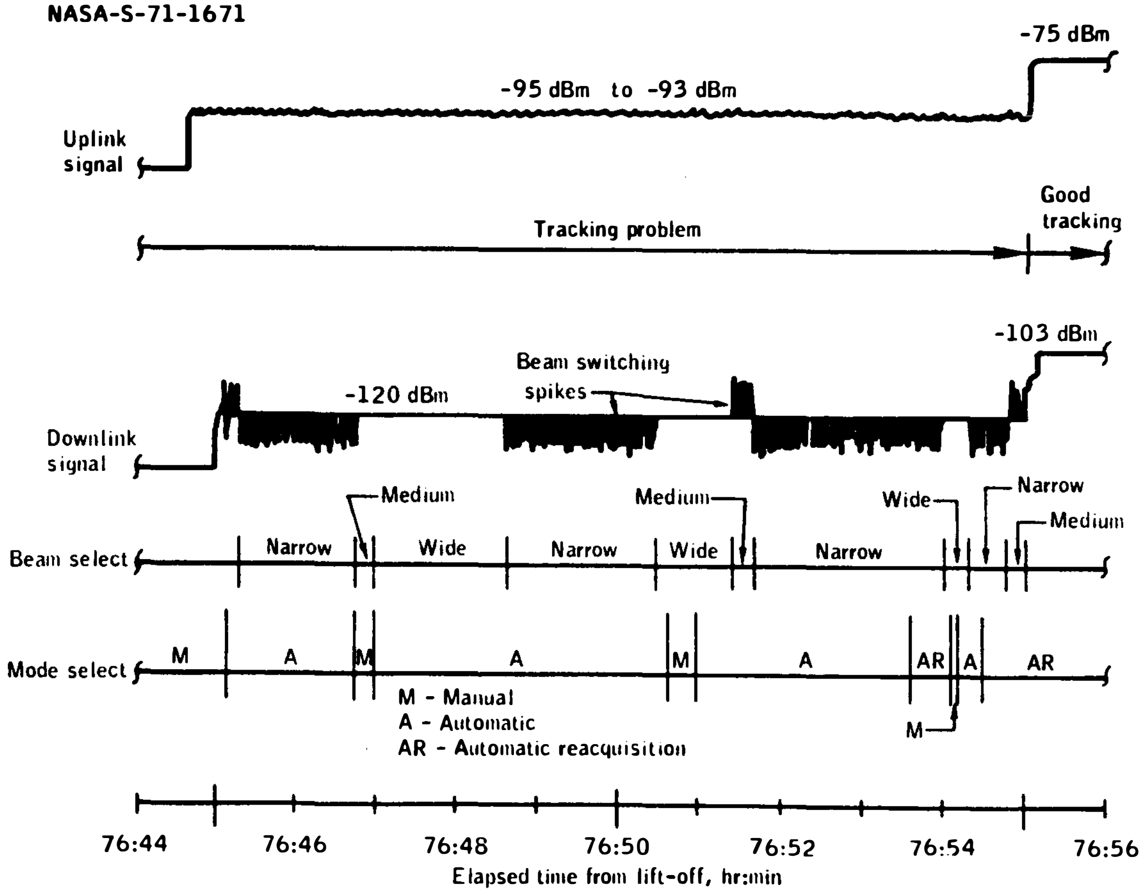

Figure 14-3.- Data from first period of anomalous operation.

The low signals correlate with antenna pattern and gain data for a 5- to 8-degree boresight shift in the wide-beam mode. The direction of the spikes observed on the downlink data in figure 14-3 are consistent with switching between the wide and narrow beams. Conditions for a normal alignment and a misalignment of the wide and narrow beams are shown in figure 14-4. A 5- to 8-degree shift in the wide-beam mode boresight

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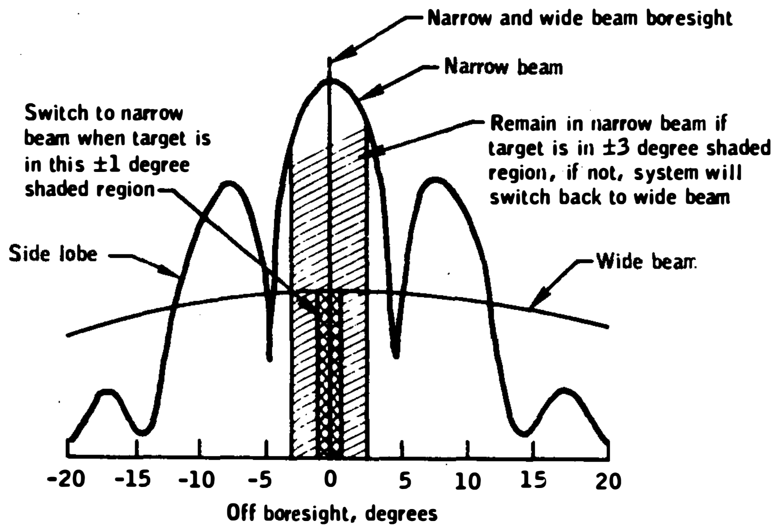
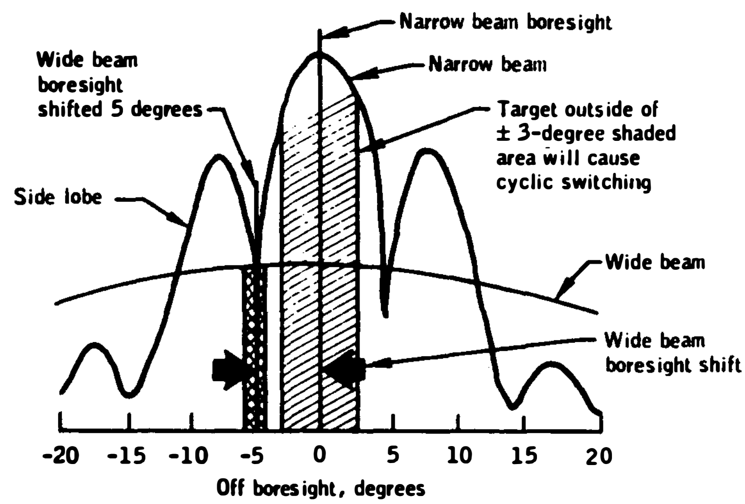


(a) Normal wide beam/narrow beam antenna alignment patterns.



(b) Alignment conditions indicated by Apollo 14 data.

Figure 14-4.- Antenna narrow and wide beam boresight relationship.

Figure 14-4.- Antenna narrow- and wide-beam boresight relationship.

will not allow narrow-beam lock since continuous switching between the wide beam and narrow beam will occur with the target outside the ±3-degree limit of the narrow-beam boresight. These large error signals will initiate cyclic switching between the wide-beam and narrow-beam modes.

The acquisition and tracking problems for the other time periods were similar. As a result of the 5- to 8-degree boresight shift of the wide beam, the antenna at times would lock-up on the first side lobe instead of the main lobe (fig. 14-4). Since the antenna array is not symmetrical, the boresight error in the wide-beam mode is a function of the target approach path.

A number of problems could have caused the electrical shift of the wide beam; however, they effectively reduce to an interruption of one of the four wide-beam elements which supply signals to the wide-beam comparator. The most likely cause is that a connector to one of the coaxial cables which are used to connect the wide-beam antennas to the comparator assembly of the strip lines was faulty.

In support of this cause, five bad coaxial center conductors have been found. Also, a coaxial connector was disconnected on the antenna and the effect in the beam occurred. There are two causes of the problem with the center conductor, both of which occur during cable-to-connector assembly (fig. 14-5). The sleeve is assembled to the cable, a Lexan insulator is then slipped over the center conductor, and the pin is inserted over the center conductor and soldered. If the wire gets too hot during soldering, the Lexan grows and no longer fits loosely through the hole in the outer body. When this occurs and the outer body is screwed onto the sleeve, the wire can be twisted and the center conductor may fail.

Another possible failure occurs when too much solder is used or the wire is not centered in the pin. These conditions will bind the pin to the outer body insulation and, during assembly, the wire is twisted to failure. These conditions are being corrected by reworking all connectors and applying proper inspection and control procedures during the rework.

Failures on previous flights, in addition to the one on this mission, were of the type that appear under certain thermal conditions. The malfunction conditions of each of the failures were isolated to different components of the antenna. All of these defects were of a type which could escape the test screening process. Another possibility is that the shock which an antenna experiences during the spacecraft-lunar module adapter separation when the pyrotechnics fire might have caused defects in the circuitry which could open up under certain thermal conditions

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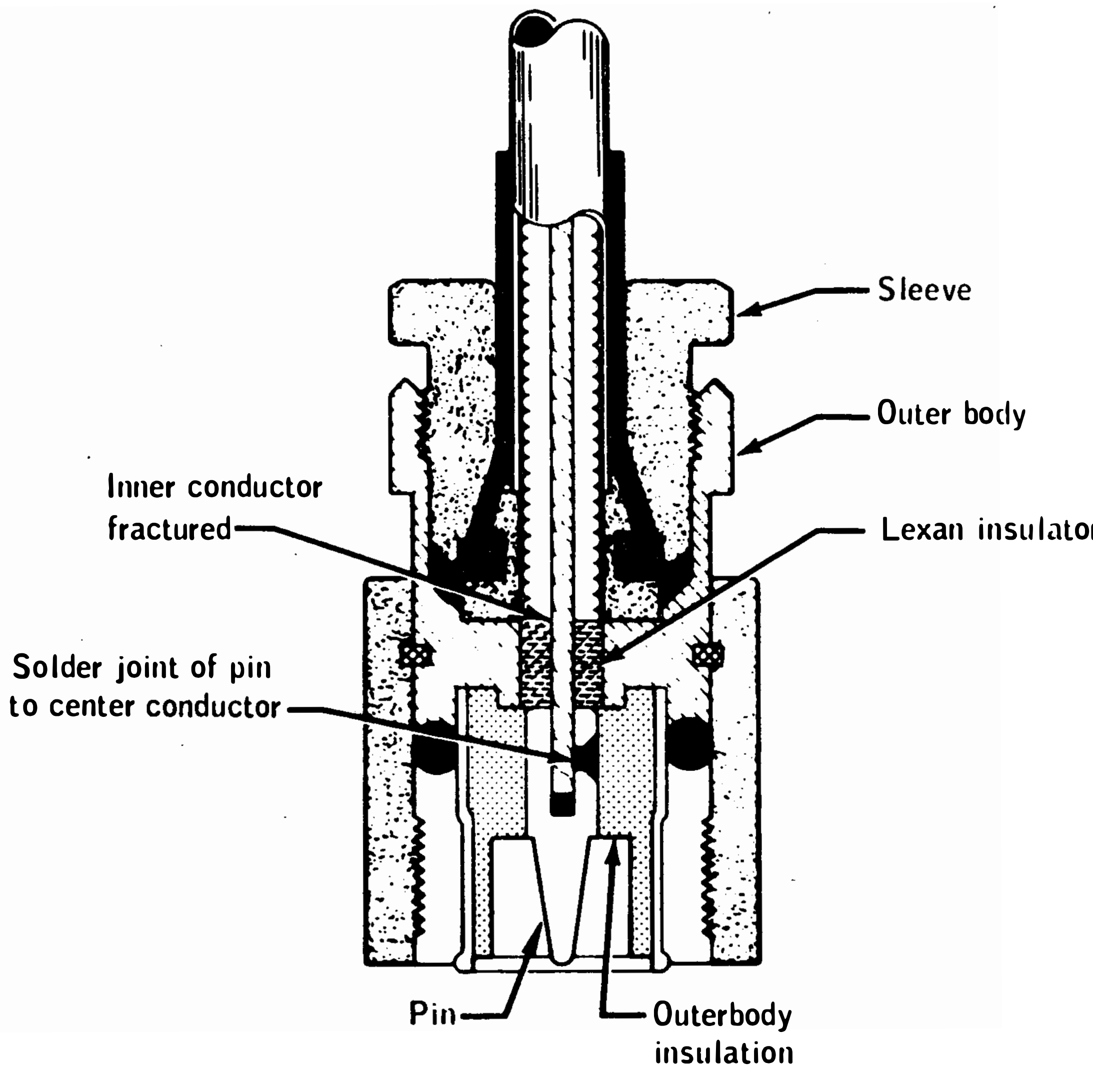

Note: A slip fit is required between the pin and insulator so that the pin does not rotate when turning outer body during assembly of the threaded sleeve.

Figure 14-5.- Coaxial cable failures.

during the mission. The original qualification tests considered that the shock environment would be low.

To further investigate the effects of the spacecraft-lunar module adapter pyrotechnic shock on an antenna, a shock test has been conducted. The results show that the antenna experiences about an order-of-magnitude greater shock than had been originally anticipated. However, thermal testing of the antenna has shown no detrimental effects because of the shock. To better screen out defects which potentially could affect the functioning of the antenna, a thermal acceptance test will be performed on all future flight antennas while radiating and under operating conditions.

This anomaly is closed.

14.1.3 Urine Nozzle Blockage

After transposition and docking and prior to initiating passive thermal control, the crew indicated that the urine nozzle (fig. 14-6)

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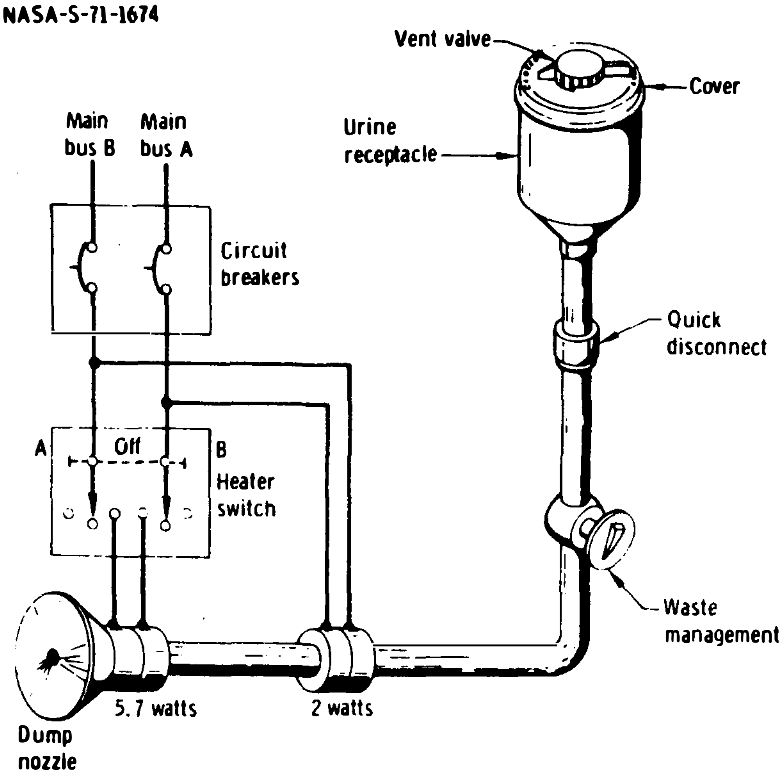

Figure 14-6.- Urine receptacle and nozzle.

was obstructed. The same condition occurred several other times during the mission and, in each case, the dump nozzle had not been exposed to sunlight for prolonged periods.

The heaters and circuitry were checked and found to be normal. The system design has been previously verified under some, but not all, likely thermal conditions while dumping urine. Although the history of previous missions has shown no indications of freezing, the dumps during this flight may have coincided with a colder nozzle condition than on any previous flight. Also, the purge-and-dry procedure used during this mission was different from that used in previous missions in that the urine receiver was rinsed with water after each use and the system was purged with oxygen for longer times than in past missions. These changes may have contributed to the freezing. A test is planned to determine the contribution of the procedures to the freezing.

If freezing occurs in the future, thawing can be accomplished very quickly by orienting the spacecraft so that the nozzle is in sunlight. This was demonstrated several times during this flight. The auxiliary hatch nozzle and the water overboard dump nozzle provide backup capabilities. No hardware change is in order, but procedural changes may be necessary that would either restrict the times when urine may be dumped or modify the purging techniques.

This anomaly is closed.

14.1.4 Degraded VHF Communications

The VHF link between the command and service module and lunar module was degraded from prior to lunar lift-off through terminal phase initiation. The received signal strength measured in the lunar module was lower than predicted during the periods when VHF performance was degraded. VHF voice was poor and, 11 minutes prior to lunar lift-off, noise was received in the lunar module through the VHF system. Therefore, the system was disabled. When the system was again enabled about 4-1/2 minutes before lunar lift-off, the noise had disappeared.

Prior to lunar descent, the VHF ranging and rendezvous radar range measurements correlated closely. However, during the time period preceding terminal phase initiation, the VHF ranging system indicated erroneous measurements. During this same time period, numerous range tracking dropouts also occurred. The range measurements were in error by 5 to 15 miles when compared with the rendezvous radar range measurements (fig. 14-7). The VHF ranging data presented in the figure results from a number of different acquisitions. After terminal phase initiation, the

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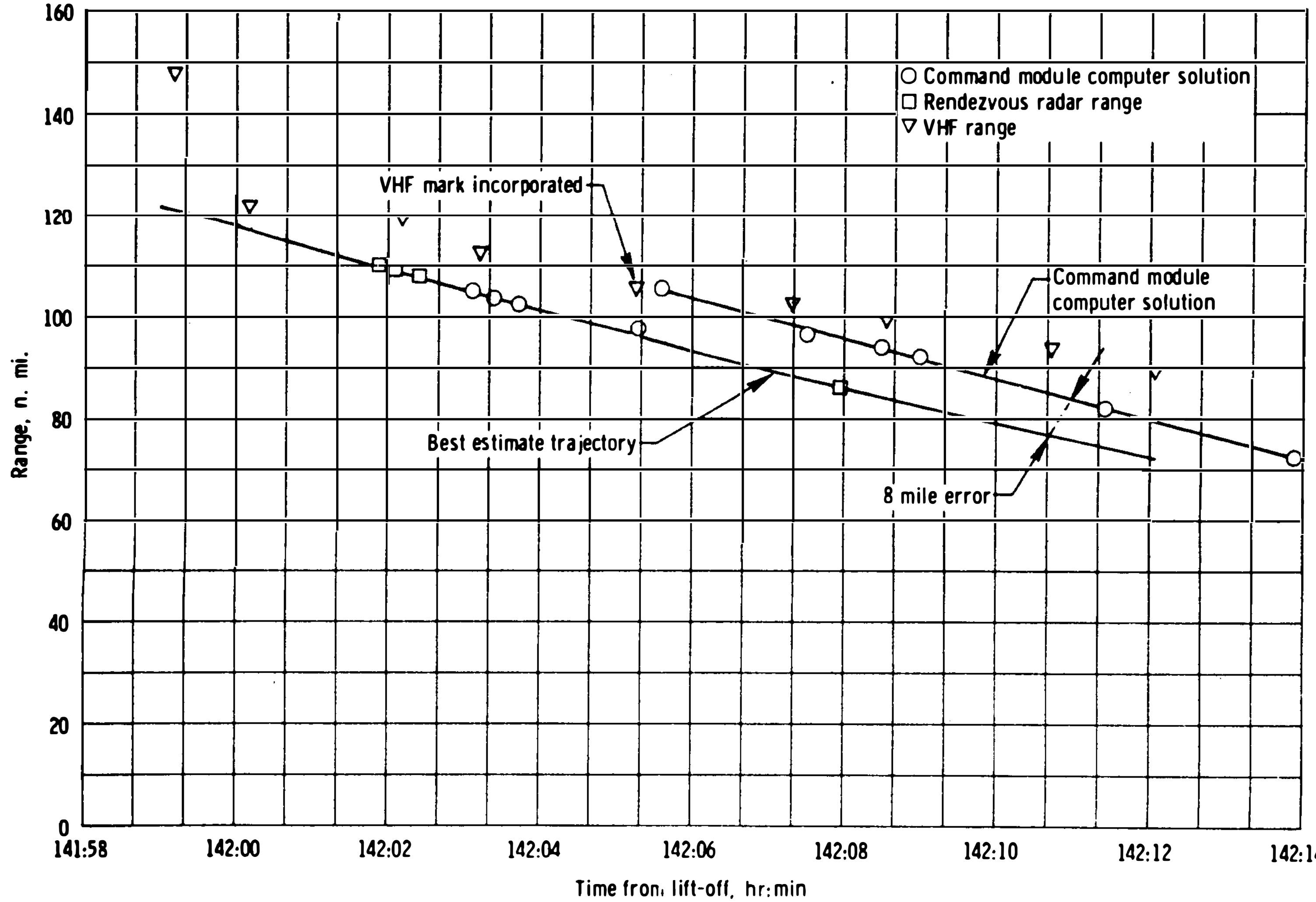

Figure 14-7.- Relative range comparisons during rendezvous.

signal strength, as indicated by the lunar module receiver automatic gain control voltage measurement, was adequate and VHF ranging operation was normal.

These problems would be expected if the signal strength were low. The signal strength was determined by measuring the automatic gain control voltage in the lunar module VHF receiver. The measurement range was -97.5 to -32 dBm. Figure 14-8 shows the predicted signal strengths and those measured during the mission at the lunar module receiver.

The maximum predicted values assume that direct and multipath signals add. For the minimum predicted, the multipath signal is assumed to subtract from the direct signal. The antenna pattern model used consisted of gain values in 2-degree increments and did not include all the peaks that are known to occur because of antenna polarization differences between the lunar module and command and service module. Line-of-sight to the command module passing through one of these peaks would explain the pulses shown in figure 14-8(a).

Figure 14-8(b) shows that the signal strength should have been on scale subsequent to about 10 minutes after insertion. Figure 14-8(c) shows that the measured signal strength was below that expected for the right-forward antenna, the one which the checklist called out to be used, from insertion to docking and above that predicted for the right-aft antenna for this same time period. This indicates that the proper antenna was selected, but some condition existed which decreased the signal strength to the lunar module receiver.

The lower-than-normal RF link performance was a two-way problem (voice was poor in both directions); therefore, certain parts of the VHF system are prime candidates for the cause of the problem. Figure 14-9 is a block diagram of the VHF communications system as configured during the rendezvous phase of the mission. Also shown are those areas in which a malfunction could have affected the two-way RF link performance. A single malfunction in any other area would have affected one-way performance only.

The VHF ranging problems resulted from lower-than-normal signal strength together with the existing range rate. The ranging equipment is designed to operate with signal strengths greater than -105 dBm. The lunar module received signal strength data are essentially qualitative, since most of the inflight data during the problem period were off-scale low. Unfortunately, the scale selection was not chosen for failure analysis. A spot check of relative vehicle attitudes, as evidenced by normal performance of the rendezvous radar and by sextant sightings, indicates that the attitudes were proper. The crew also indicated that they followed the checklist for VHF antenna selection.

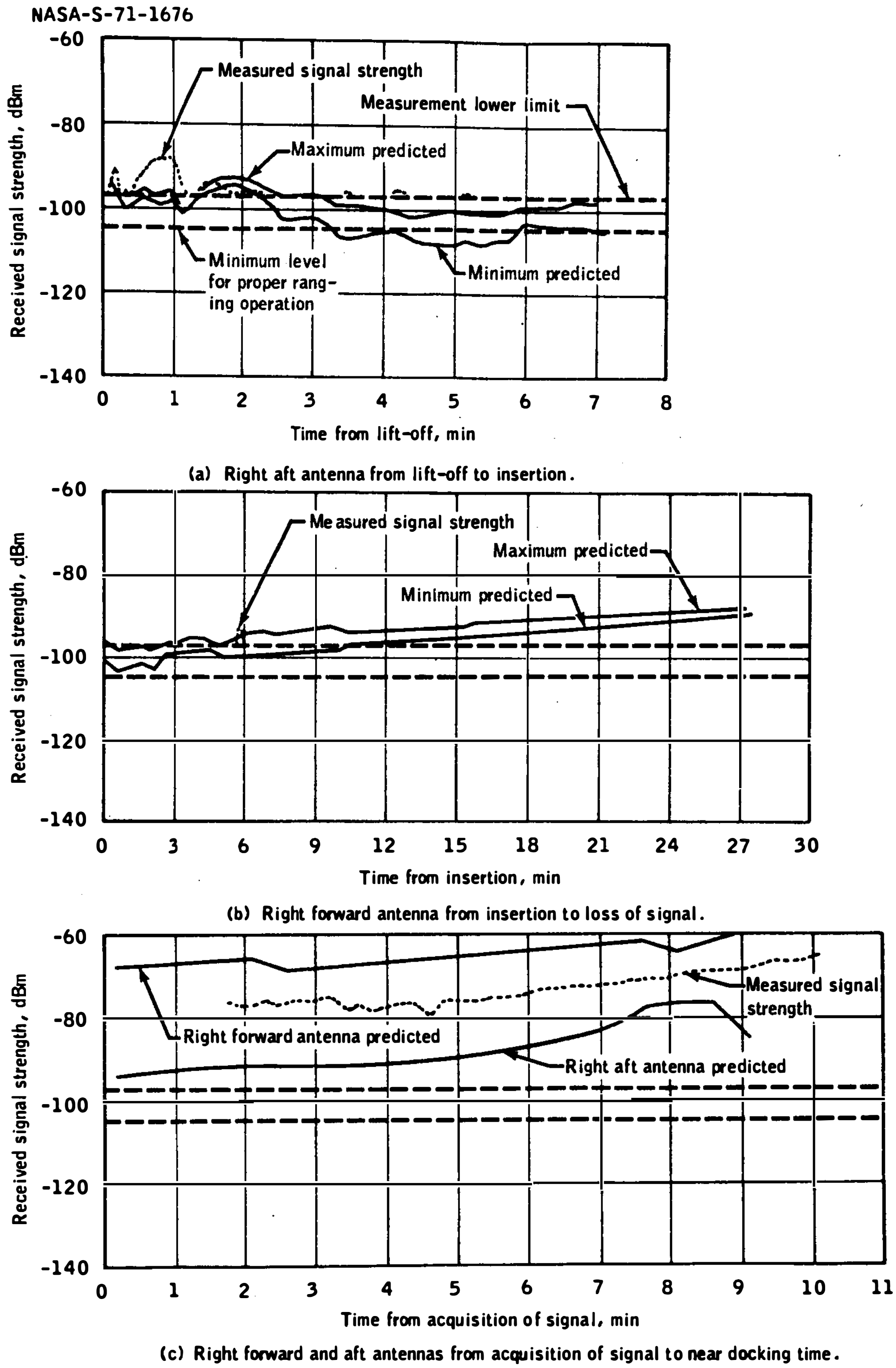

Figure 14-8.- Received signal strength from omnidirectional antennas.

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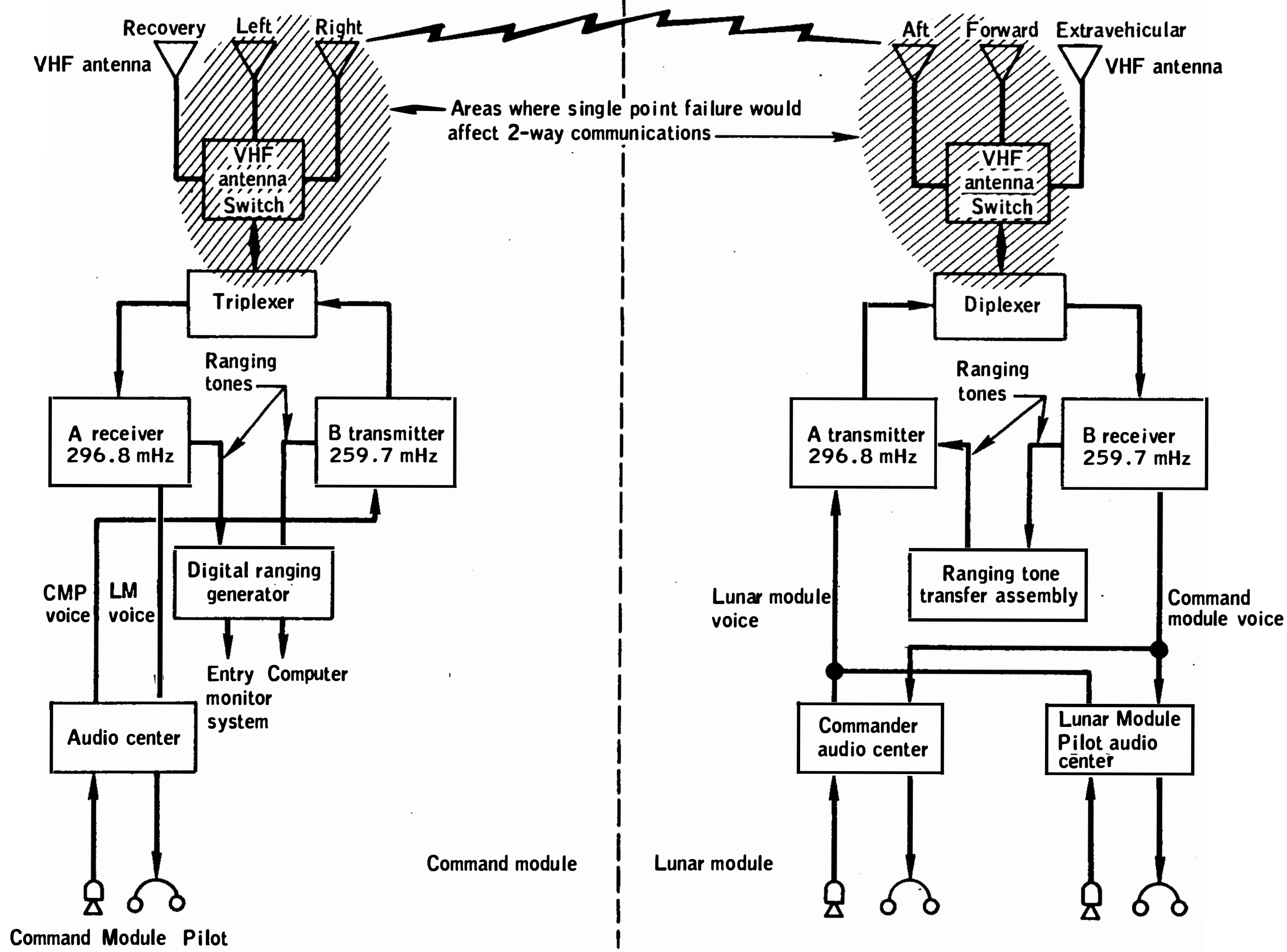

Figure 14-9.- Block diagram of VHF communications systems.

A flight test was performed to verify that the VHF ranging problem was associated with the low VHF signal strength and was not related to the VHF ranging elements. The Apollo 14 range and range rate were duplicated and the results showed that, for signal strengths below about -105 dBm, errors in indicated range similar to those experienced on Apollo 14 will be generated.

The procedures for test and checkout of the lunar module and command module elements of the VHF system have been reassessed and found to be sufficient, and additional inspection or testing is not practical or necessary. The only action that will be taken is to add instrumentation on both the lunar module and the command and service module to provide more insight into the nature of the problem if it occurs on subsequent flights. Therefore, for subsequent vehicles, receiver automatic gain control measurements will be added to both the lunar module and the command and service module. Measurement scale factors will be selected to give on-scale data at the low signal strength range. The lunar module data storage and electronics assembly (tape recorder) was retained for subsequent postflight evaluation of voice quality associated with the automatic gain control measurements.

Crew training will be expanded to include realistic simulations of weak signal strengths and the effects of ranging on voice quality. The effects of the modes selected and operational techniques such as voice level and microphone position become important near the range limits of the system.

This anomaly is closed.

14.1.5 Entry Monitor System 0.05g Light

The entry monitor system 0.05g light did not illuminate within 3 seconds after an 0.05g condition was sensed by the primary guidance system. The crew then manually switched to the backup position.

The entry monitor system is designed to start automatically when 0.05g is sensed by the system accelerometer. When this sensing occurs, the 0.05g light should come on, the scroll should begin to drive (although barely perceptible) and the range-to-go counter should begin to count down. The crew reported the light failure but was unable to verify whether the scroll or counter responded before the switch was manually changed to the backup mode. The crew also reported that the neutral density filter was covering the 0.05g light and that there were sunlight reflections in the cabin.

Analysis of the range counter data reported by the crew indicates a landing point about 5 nautical miles short; whereas, if the entry monitor system had not started when 0.05g was sensed and had started 3 seconds later, the indicated landing point would have been on the order of 20 nautical miles long.

Postflight tests conducted on the system show that the lamp driver circuit and the redundant lamp filaments were operating properly. Analysis of the range counter data and postflight tests indicate that the failure of the crew to see the light was caused by having the filter positioned in front of the light. Reflected light from the sun and the ionization layer would make it very difficult to see the light. Further, a clear glass filter is used in the simulator; whereas, the spacecraft filter is silvered.

The corrective action is to replace the filter in the simulator with a flight unit. Also, a flight procedural change will be made to position the filter so that it will not obscure the light.

This anomaly is closed.

14.1.6 Inability to Disconnect Main Bus A

During entry, when the main bus tie switches (motor-driven switches) were placed in the off position at 800 feet, main bus A should have deenergized; however, the bus remained on until after landing when the battery bus-tie circuit breakers were opened. Postflight testing showed that the main motor switch contacts were closed (fig. 14-10). Also, the

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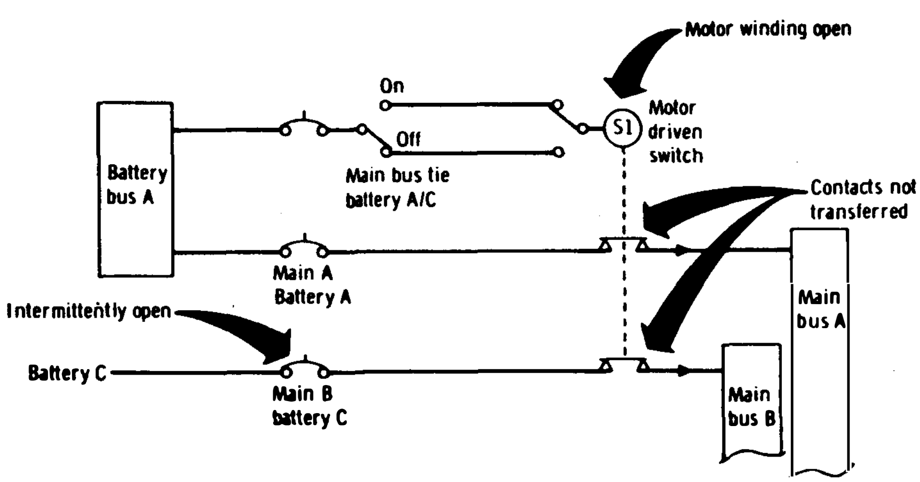

Figure 14-10.- Bus-tie circuitry.

internal switches which control the drive motor were shorted together and the motor windings were open. These conditions indicate that the motor switch stalled.

Main bus B should have been powered because of this failure, but was not. Postflight testing showed that this occurred because the main bus B circuit breaker for battery C was intermittent. This problem is discussed in section 14.1.7.

A similar motor switch failure was experienced during tests of the Apollo 15 command and service module at the launch site. Also, a second similar motor switch on the Apollo 15 vehicle required 100 milliseconds to transfer; whereas, normal transfer time is 50 milliseconds. A motor current signature was taken for one switch cycle of the slow-operating switch and compared to a similar signature taken prior to delivery. It showed that contact resistance between the brushes and commutator had degraded and become extremely erratic. Torque measurements of the failed motor switch without the motors were normal. This isolates the problem to the motors of the switch assembly.

A black track of deposits from the brushes was found on the Apollo 14 commutator, as well as on both of the commutators from the Apollo 15 motors. One motor had failed, and the other was running slow. Normally, a commutator should show some discoloration along the brush track, but a buildup of brush material along the track is abnormal. As a result of the track buildup, the resistance between the brushes and commutator became higher. The higher resistance drops the voltage into the armature causing the motor to run slower. (Switch transfer, open to closed, or vice versa, requires 11 revolutions of the motor.) The increased resistance at the brushes generates more heat than normal. A visual inspection of the Apollo 14 motor brush assembly showed high heating of the brushes had occurred, and this was concentrated at the brush-commutator interface. The condition was evident by the melting pattern of a thin nylon dish which retains the brush in the brush holder.

An analysis is being made to determine the deposit buildup on the commutator. Either the brush composition is in error, or a contamination exists in the brush composition. X-ray refraction analysis shows the same elements throughout the brush. The percentage of each of the substances will be determined and compared to the specification analysis of the brush.

Inspection of the commutator outside of the track shows a clean copper surface comparable to other machined surfaces within the motor. It can be inferred from this that there are no problems associated with the age/life of the lubricants from the bearings or with outgassing from organic materials which might deposit on the commutators. The switch assemblies are hermetically sealed and under a 15-psi pressure of nitrogen and helium gas.

Each motor is operated continuously for 4 to 8 hours to seat the brushes. The motors are then disassembled, inspected, and cleaned. Procedures for cleaning the motor assembly are not explicit as to materials or techniques to be used. This could be the cause of the problem. A further study of this aspect is being made. An anomaly report will be issued upon completion of the investigation.

There are 36 motor-driven switch assemblies in the spacecraft. Some of the switches are normally not used in flight. Some are used once or, at most, several times. The increased resistance of brush to the commutator as a result of deposits is gradual from all indications. A check of the switch operation time can be related to the deposit buildup on the commutator. Consequently, a check of the switch response time can indicate the dependability of the switch to perform one or several additional switch transfers in flight. This will be done for Apollo 15 on each of the switches. Work-around procedures have been developed if any of the motor switches are questionable as a result of the timing test.

This anomaly is open.

14.1.7 Intermittent Circuit Breaker

The motor switch failure discussed in section 14.1.6 should have resulted in main buses A and B being energized after the motor switch was commanded open (fig. 14-10). Postflight continuity checks, however, showed that there was an open circuit between battery C and main bus B and that the main bus B circuit breaker for battery C was intermittent.

Disassembly and inspection of the circuit breaker showed that the contacts are cratered (fig. 14-11). The crater contains a white substance which held the contacts apart when the circuit breaker was actuated.

The white substance will be analyzed to determine its composition and source. Circuit breakers which have been used in similar applications in Apollo 14 will also be examined. An anomaly report will be issued under separate cover when the analysis has been completed.

This anomaly is open.

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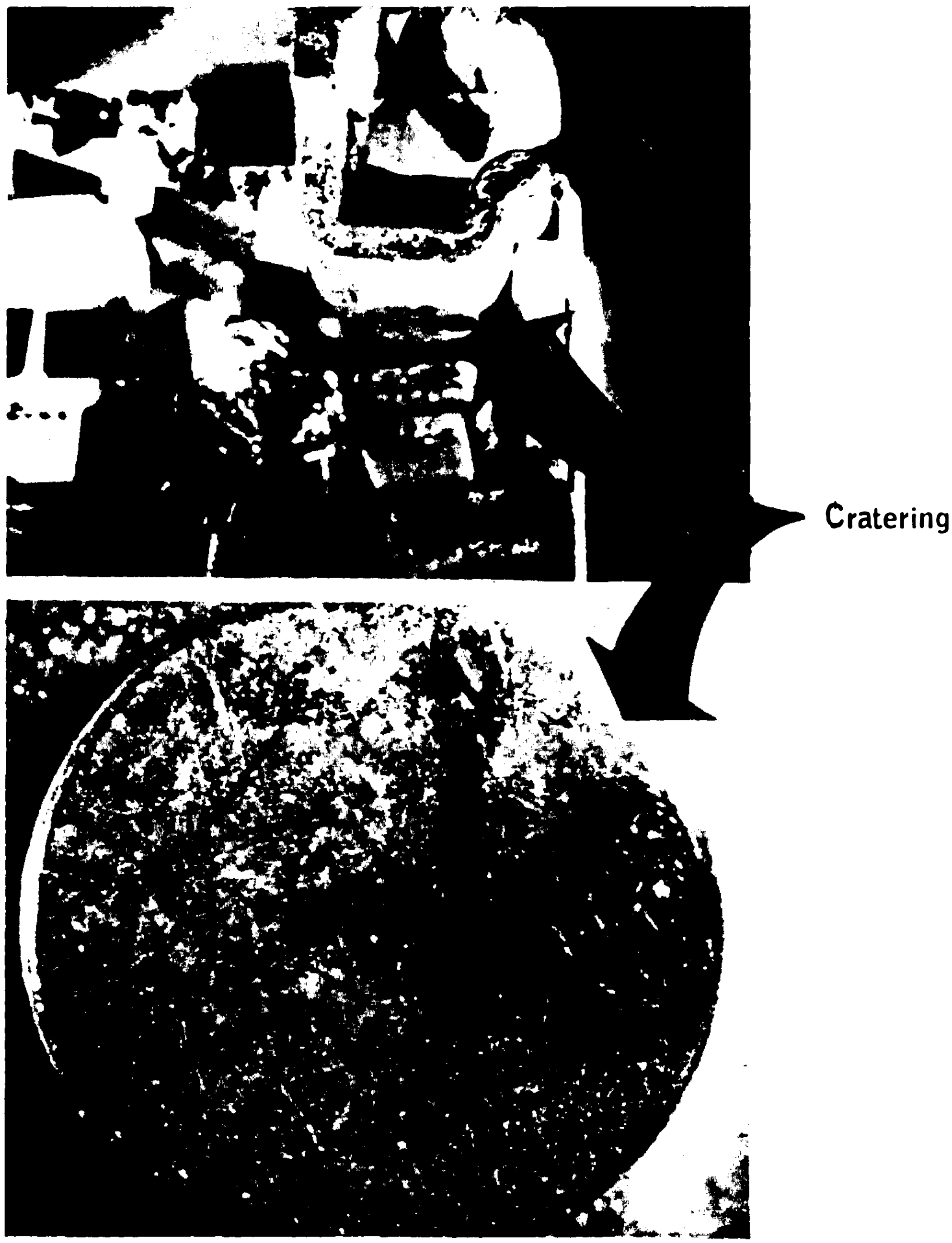

Figure 14-11.- Circuit breaker contact.

14.1.8 Food Preparation Unit Leakage

The crew reported that a bubble of water collected on the stem of the food preparation unit after hot water was dispensed, indicating a slight leak. This problem also occurred on Apollo 12.

Tests of both the Apollo 12 and Apollo 14 units showed no leakage when room temperature water was dispensed through the hot water valve; however, at an elevated water temperature of approximately 150° F, a slight leakage appeared after valve actuation. Disassembly of the Apollo 12 dispenser showed damage in two valve O-rings, apparently as a result of the considerable particle contamination found in the hot water valve. Most of the contamination was identified as material related to component fabrication and valve assembly and probably remained in the valve because of incomplete cleaning procedures. Since the particles were found only in the hot water valve, the contamination apparently originated entirely within that assembly and was not supplied from other parts of the water system.

Postflight, when the hot water valve was cycled several times, the outflow was considerably less than the specified 1 ounce per cycle. Disassembly of the valve will be performed and an anomaly report will be issued under separate cover upon completion of the investigation. The Apollo 15 unit has been checked during altitude chamber tests with hot water and no leakage was noted.

This anomaly is open.

14.1.9 Rapid Repressurization System Leakage

Repressurization of the three storage bottles in the rapid repressurization system (fig. 14-12) was required three times in addition to the normal repressurizations during the mission. The system required repressurization once in lunar orbit and twice during the transearth coast phase. Just prior to the first of the transearth coast repressurizations, the system had been used (face mask checks) and refilled (fig. 14-13). In this instance, the fill valve was closed before the system was fully recharged. Calculations from the surge tank pressure data indicate that the repressurization package was at approximately 510 psi at 199 hours 48 minutes and was only recharged to about 715 psi (fig. 14-13). The cabin indication of the repressurization package pressure would have indicated a higher pressure because of the temperature rise of the compressed gas. The crew noted a value of about 700 psi (due to temperature stabilization) at approximately 211 hours and recharged the system again.

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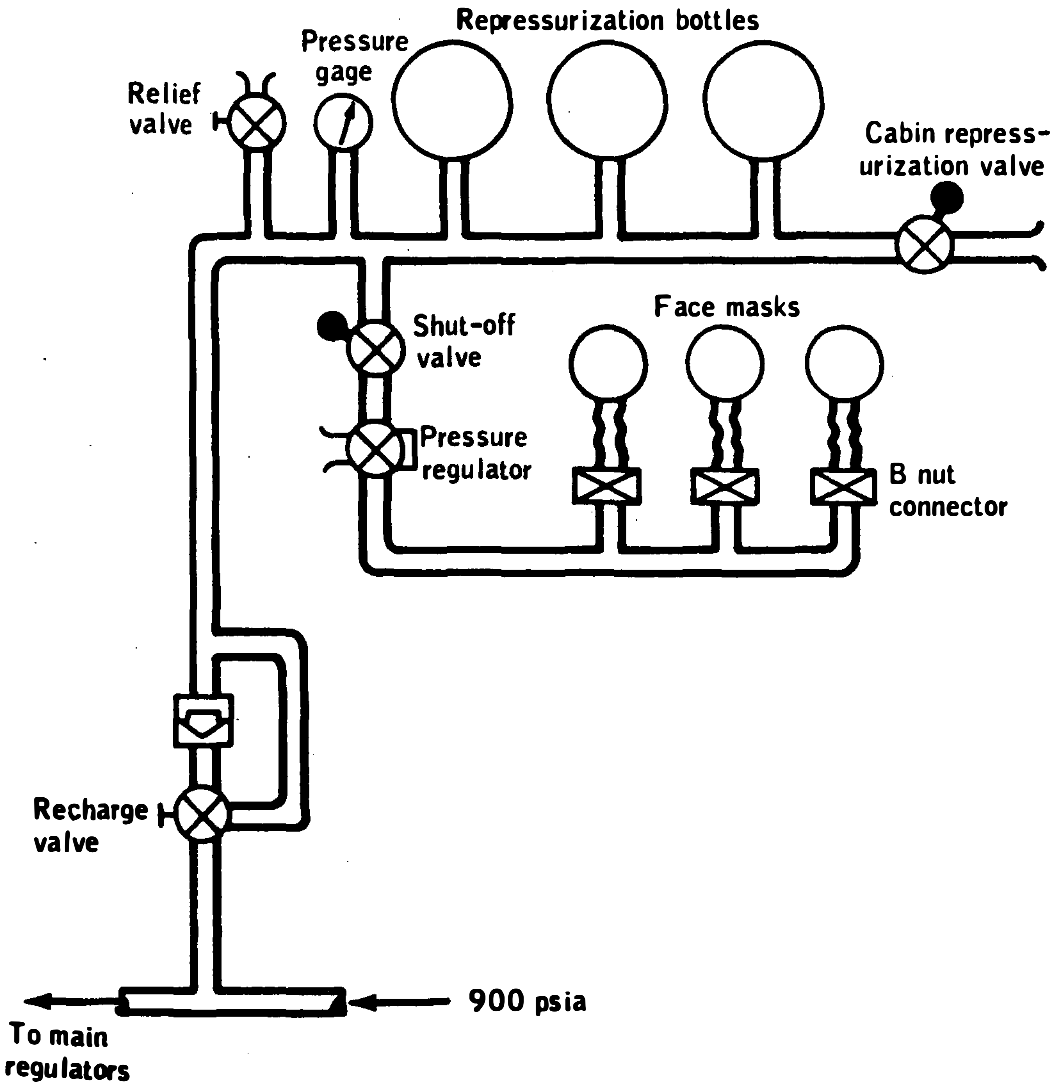

Figure 14-12.- Rapid repressurization system.

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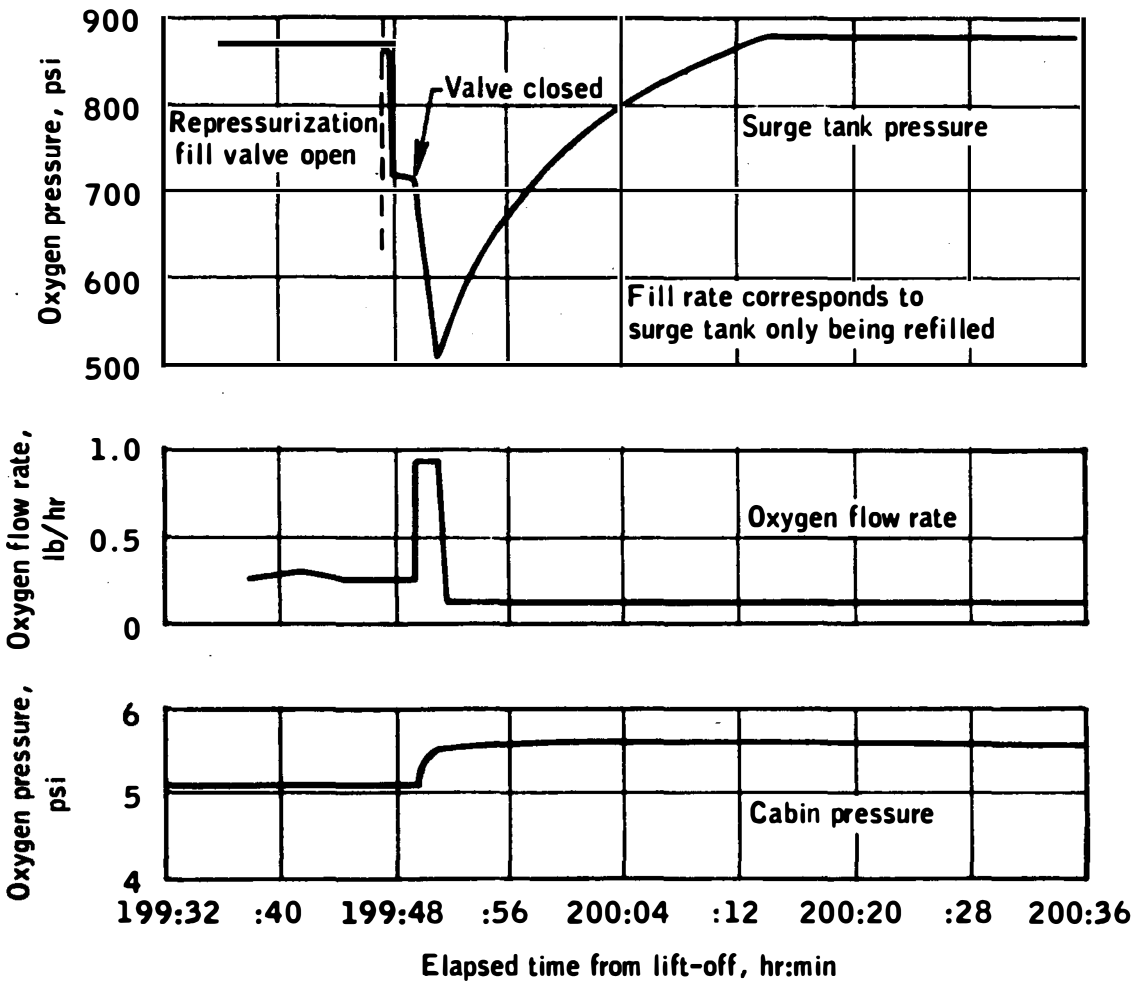

Figure 14-13.- Rapid repressurization package data.

Data are not available from the lunar orbit repressurization as the spacecraft was on the back side of the moon during the operation. However, the general procedure used during the transearth coast phase would only partially recharge the system.

Postflight checks of the 900-psi system showed that the leakage rate was about 40 standard cc/min as compared with the preflight value of 14 standard cc/min. This change in leakage rate is not considered abnormal. A leakage rate of this magnitude would lower the system pressure about 100 psi every 3 days. Therefore, the lunar orbit recharging of the system probably resulted from normal leakage.

Future crews will be briefed on the recharging techniques for other than normal rechargings to insure that the system is fully recharged.

This anomaly is closed.

14.2 LUNAR MODULE

14.2.1 Ascent Battery 5 Low Voltage

At 62 hours, the ascent battery 5 open-circuit voltage had decreased from a lift-off value of 37.0 volts to 36.7 volts instead of remaining at a constant level (fig. 14-14(a)). Figure 14-14(b) shows characteristic open-circuit voltages for a fully charged battery (peroxide level of all cells) and all cells operating on the monoxide level of the silver plate. Note that one cell at the monoxide level and the remaining 19 at the peroxide level would have caused the observed open-circuit voltage of 36.7 volts. Any one of the following conditions could have caused the voltage drop.

a. Battery cell short

b. Cell short-to-case through an electrolyte path

c. External battery load.

A single-cell short could be caused by inclusion of conductive foreign material in the cell-plate pack at the time of manufacture or excessive braze material at the brazed joint between the plate tab and plate grid, either of which could pierce the cellophane plate separator during the launch powered-flight phase, providing a conductive path between positive and negative plates (fig. 14-15).

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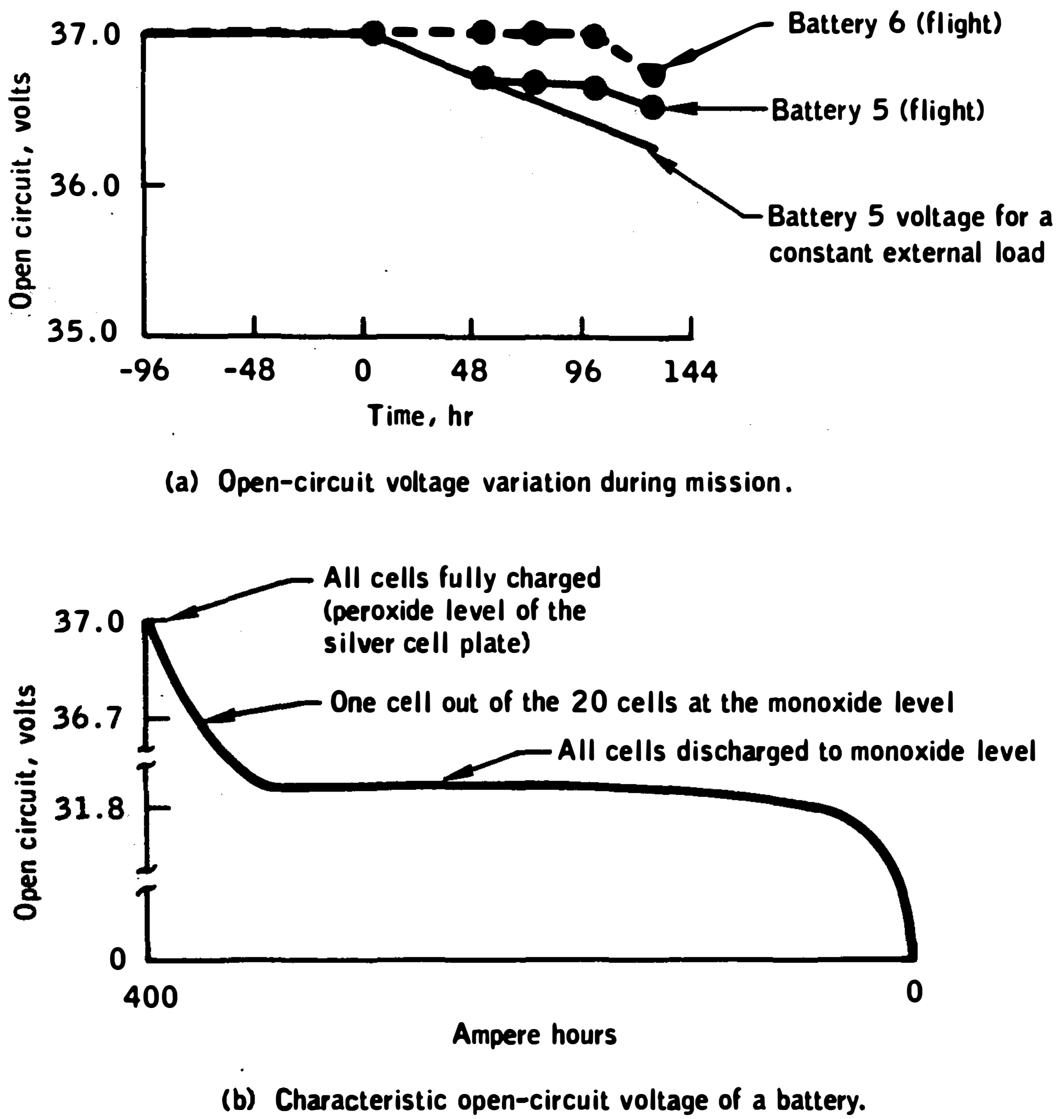

Figure 14-14.- Ascent battery voltage characteristics.

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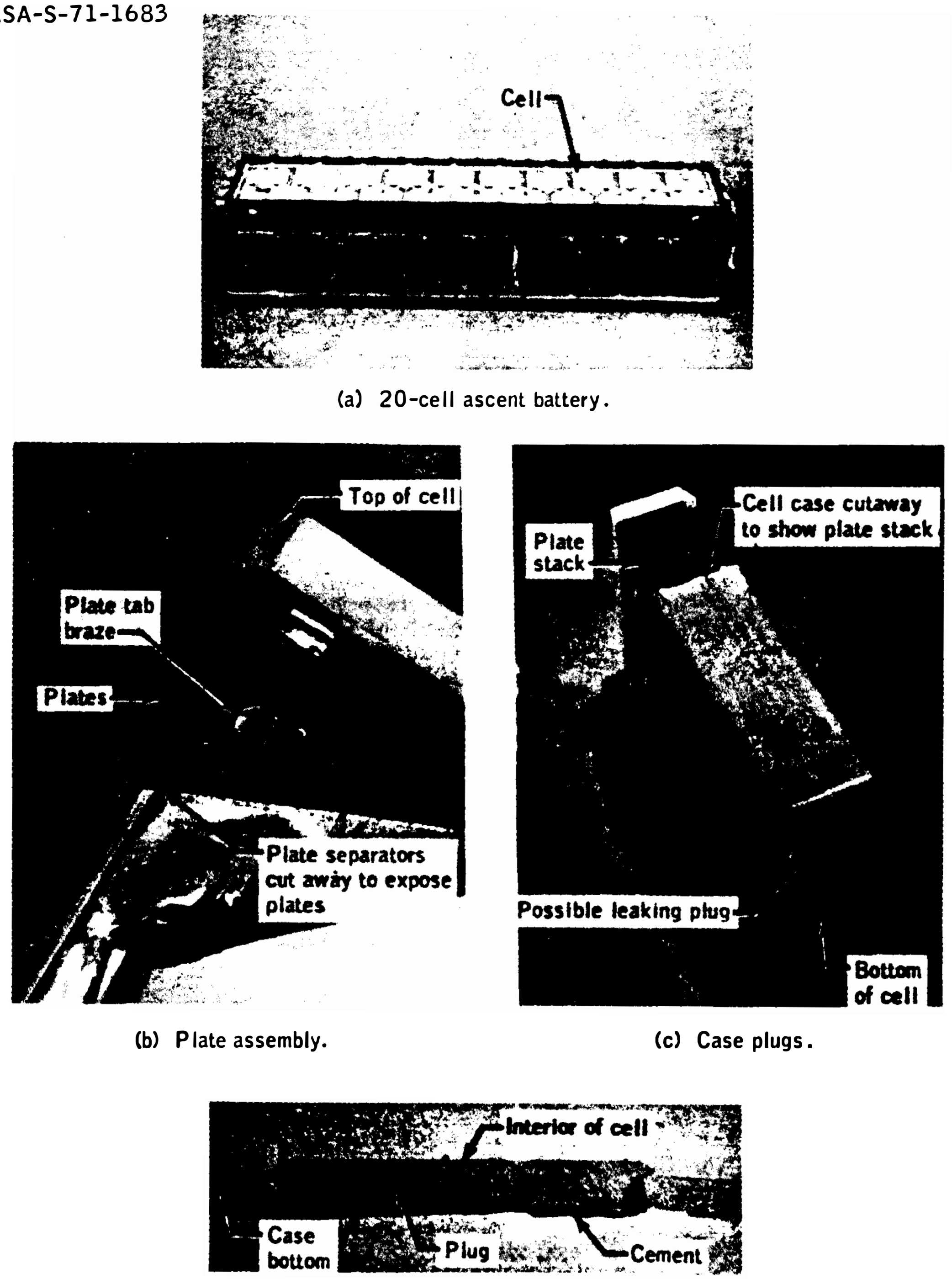

(a) 20-cell ascent battery.

(b) Plate assembly.

(c) Case plugs.

(d) Cross section of plug.

Figure 14-15.- Ascent battery cell structure.

During battery activation, one of the descent batteries had a cell short to the case through an electrolyte path around a cell plug joint (fig. 14-15). The cell plug was not properly sealed to the bottom of the plastic cell case. If this condition existed in ascent battery 5 in flight, it could have decreased the battery open-circuit voltage.

An external battery load could have existed from lift-off to 62 hours on the circuit shown in figure 14-16 in which typical types of high resistance shorts are also shown. For this condition, the current drain would have occurred on all cells. Figure 14-14 shows the time history of the

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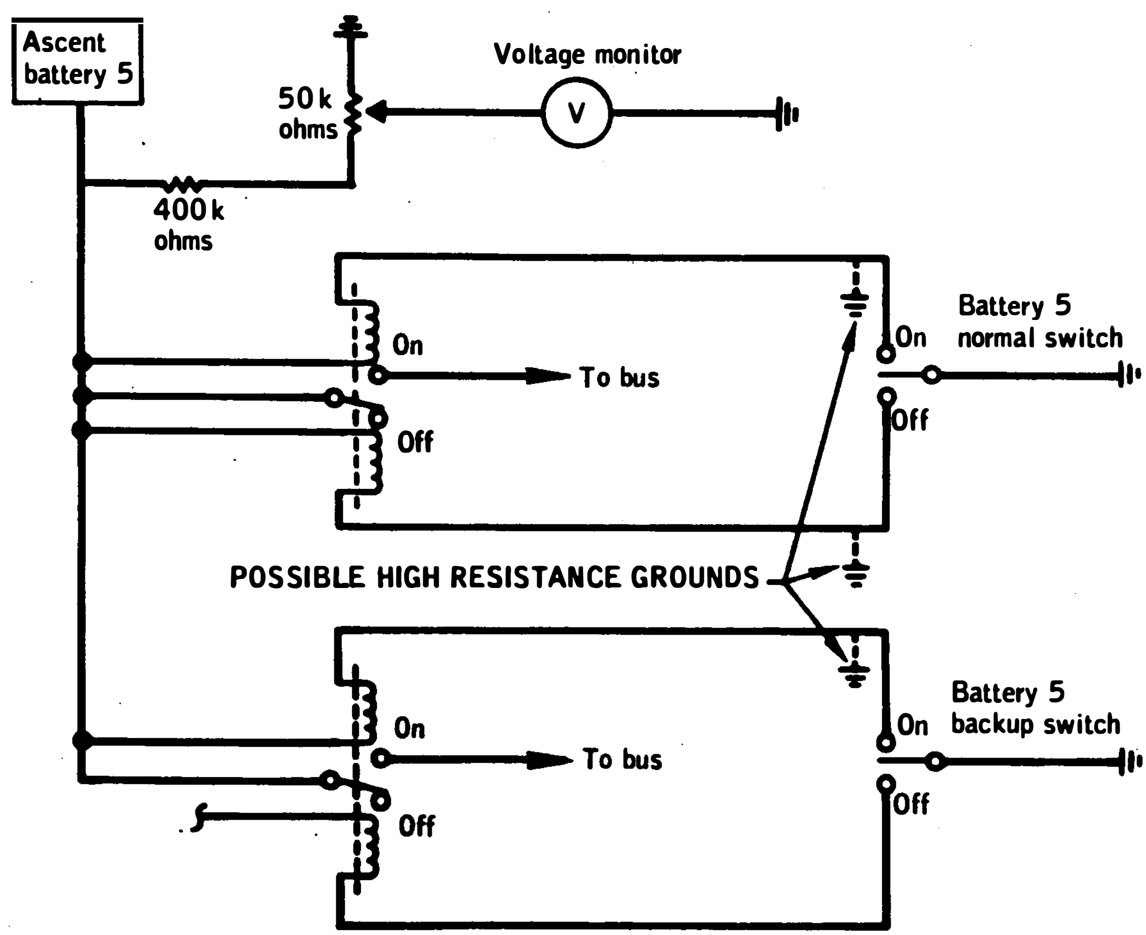

Figure 14-16.- Ascent battery 5 configured for open-circuit loads.

open-circuit bus voltage for battery 5. For a constant external load, the battery 5 open-circuit bus voltage would have been lower than the flight data at 141 hours. Therefore, the external load would have had to change with time.

To reduce the possibility of recurrence, corrective action has been taken for each of the possible causes. Stricter inspection and improved procedures have been incorporated for installation of the plugs. Particcular attention will be given to the assembly of the cell plates on future units. In addition, a test has been added at the launch site to measure lunar module parasitic loads prior to battery installation to insure that no abnormal loads are present.

This anomaly is closed.

14.2.2 Abort Signal Set In Computer

Prior to descent, the primary guidance computer received an abort command four different times. The computer would have reacted if the descent program had been initiated. The failure was isolated to one

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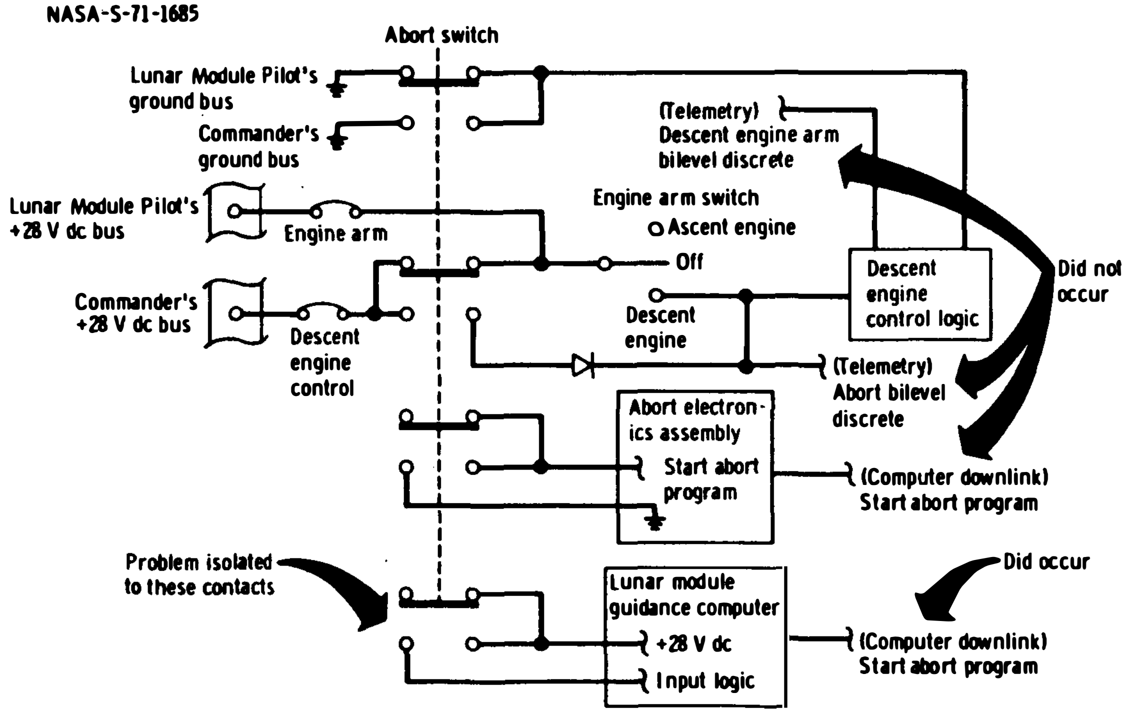

Figure 14-17.- Abort switch logic.

set of contacts of the abort switch (fig. 14-17) because the abort command appeared only on the lunar module primary guidance computer downlink (telemetry) and not on the abort guidance computer downlink (telemetry) or the telemetry bilevel discretes associated with the descent engine control logic. Recycling the switch or tapping the panel removed the signal from the computer. To prevent an unwanted abort during powered descent, a computer program was developed and verified within 2 hours, and in time to be manually inserted into the lunar module computer prior to powered descent initiation. The program would have allowed the lunar module computer to ignore the abort command, had it appeared during powered descent.

The most probable cause of the abort command was metallic contamination within the hermetically sealed abort-switch module (fig. 14-18). The failure of an internal switch component would not likely have caused the abort indication because such a failure would not have been intermittent. X-rays and dissection of similar switches have shown metallic contamination in several switches of the size which could have caused the flight failure. The metallic contamination appears to come from the internal switch parts, particularly one of the three studs which hold the contact components. The stud is, in effect, riveted by heat and pressure (fig. 14-18). This type of switch is used in eight different locations, which are:

a. Abort switch
b. Abort stage switch
c. Engine stop switches (2)
d. Master alarm switches (2)
e. Plus X translation switch
f. Engine start switch.

Corrective action consists of replacing all switches of this type with switches screened by x-ray and vibration. Since the screening is not fool-proof, circuit modifications were made to eliminate single-point failures of this type. These modifications are:

a. The abort stage switch descent-engine override function was removed from the abort-stage circuit breaker and placed on the logic power switch contact. This involved relocating one wire from one switch terminal to another.

b. Each of the two engine stop switches were rewired so that two series contacts are required to close in order to stop the engine. Formerly, the two sets of contacts in each stop switch were connected in parallel so that closure of either would shut down the engine.

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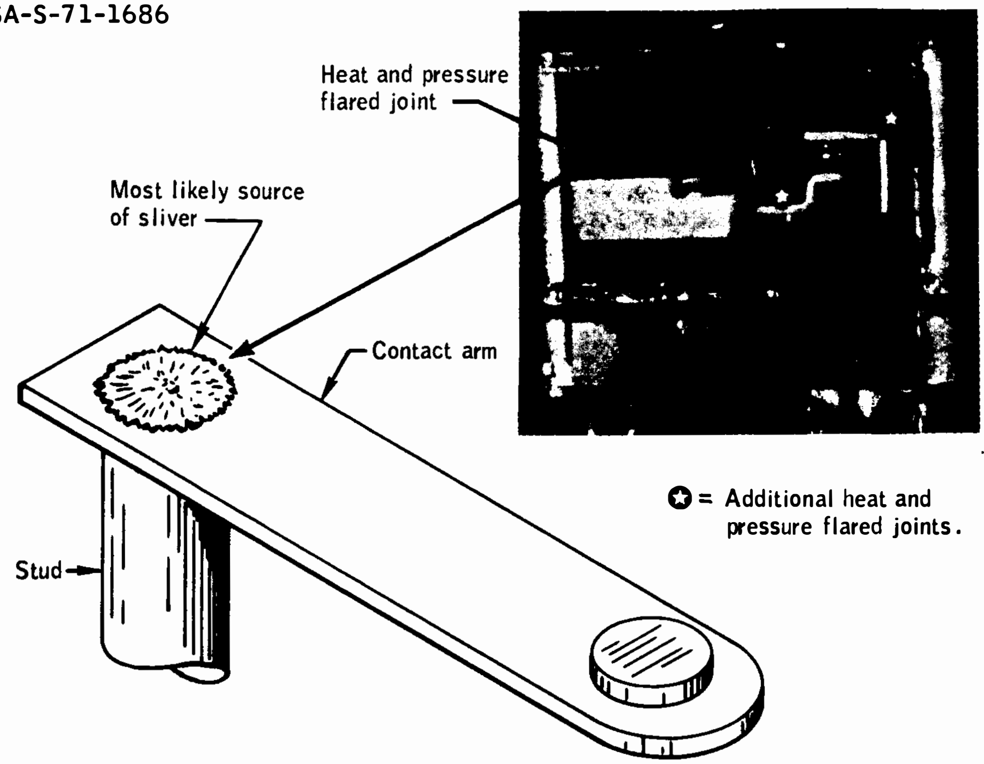
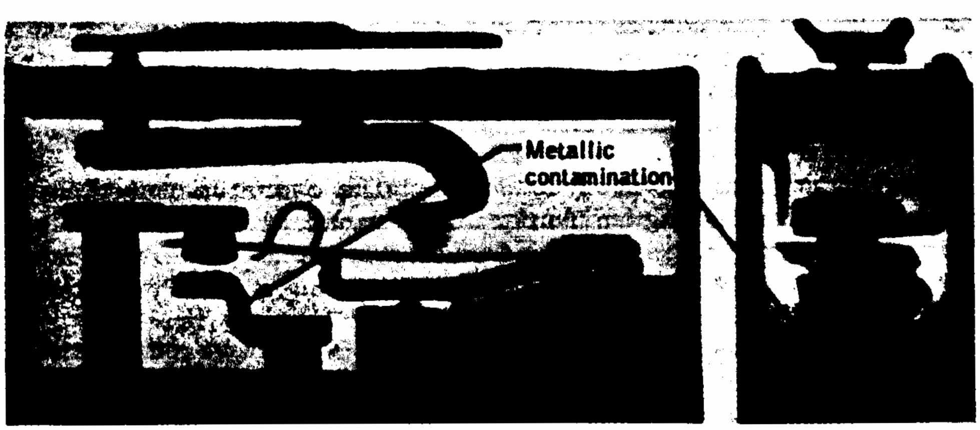


Metal contamination up to 0.030-inch long slivers found in several switches

(a) Simplified sketch of internal switch parts.



(b) X-rays of switch showing metallic contamination.

Figure 14-18.- Abort switch contamination.

c. The plus-X translation switch was rewired so that two series contact closures are required to fire the plus-X reaction control system thrusters. This removed the four-thruster translation capability, leaving only a two-thruster translation capability.

d. The engine-start switch and circuitry were not changed because of this problem since inadvertent closure would only give the manual start command, and the engine arm command is also required to fire the engine. However, because of a switch failure in another spacecraft during ground tests, the switch was rewired so that a series-parallel combination of four switch contacts are used for the function. That failure was caused by nonmetallic contamination (rust) preventing switch contact closure. This contamination is undetectable by x-rays.

e. The two master alarm switches were not rewired since inadvertent contact closure would only reset the master alarm, and this would not affect the mission or crew safety.

f. The abort and abort stage switch circuitry to the computer was not modified. Instead, the primary guidance computer software was modified to allow the crew to lock out the computer abort and abort stage program. If the crew exercises this option, any required abort would have to be performed using the abort guidance system.

This anomaly is closed.

14.2.3 Intermittent Steerable Antenna Operation

Prior to the descent phase of the mission, the S-band steerable antenna operation was intermittent. There were nine instances of unscheduled interruption of antenna tracking. Three of these have been explained. One was caused by the crew switching to an omnidirectional antenna because of an erroneous reading of the pitch position indicator at full scale of 255 degrees when the antenna was actually at 122 degrees. Another occurred because the antenna was in the manual slew mode and not in automatic-track. After undocking, the lunar module attitude was changed and, as a result, the antenna was pointed away from the earth resulting in a loss of signal. The third interruption which has been explained was caused by a failure in the ground station power amplifier resulting in a temporary loss of uplink signal.

The remaining unexplained tracking interruptions (fig. 14-19) have similar characteristics. Five tracking interruptions occurred during Goldstone coverage and figure 14-20 is a plot of ground-station-received signal strengths at these times. During the Madrid ground station coverage of revolution 32, another incident was noted with the same type of

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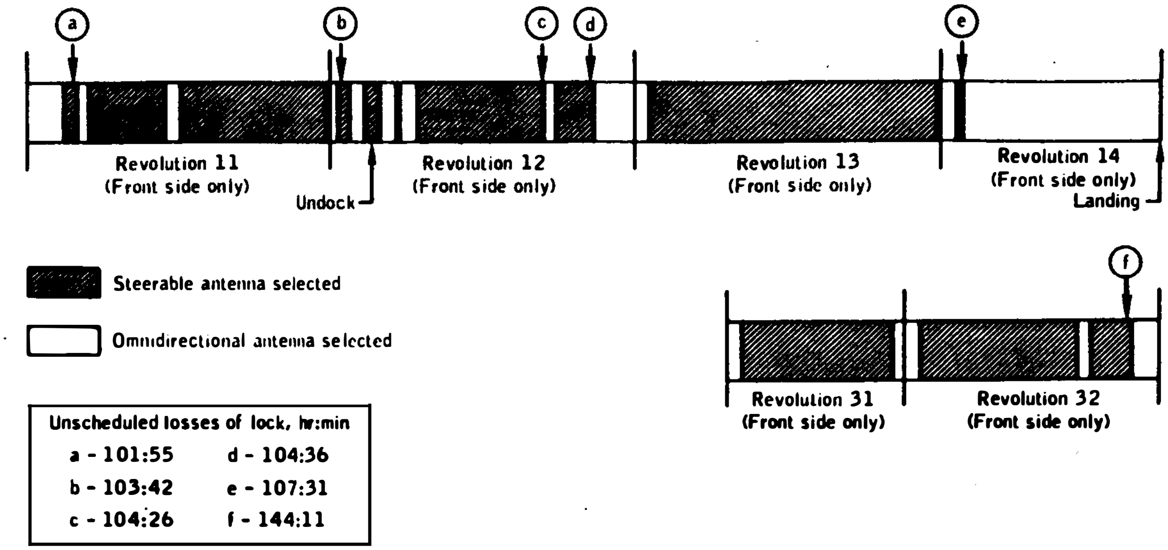

Figure 14-19.- S-band steerable antenna operation.

antenna response. It indicates that the antenna began to experience a mechanical oscillation of approximately 2 to 3 hertz, which became increasingly larger in amplitude until the antenna lost lock. When antenna oscillations exceed plus or minus 5 degrees, excessive motor drive current causes the 28-volt dc circuit breaker to open and the antenna ceases to track. The crew reset this circuit breaker several times. The antenna was also reported to be noisy, indicating the continual driving that would have occurred during the oscillations. The oscillations occurred randomly at other times during the problem period, but damped out and did not cause tracking interruptions.

The two most probable causes of these oscillations are an unwanted variation in the uplink signal or a condition of instability in the antenna/S-band transceiver tracking loop system. The conditions which can cause the first item are vehicle blockage, reflections from the spacecraft structure, multipath signal reflections from the lunar surface, noise transients induced on the uplink signal, or incidental amplitude modulation on the carrier at the critical antenna lobing frequency (50 to 100 hertz or odd harmonics).

Look-angle data indicate that the antenna was not pointed at or near the vehicle structure during the time periods when antenna lock was lost.

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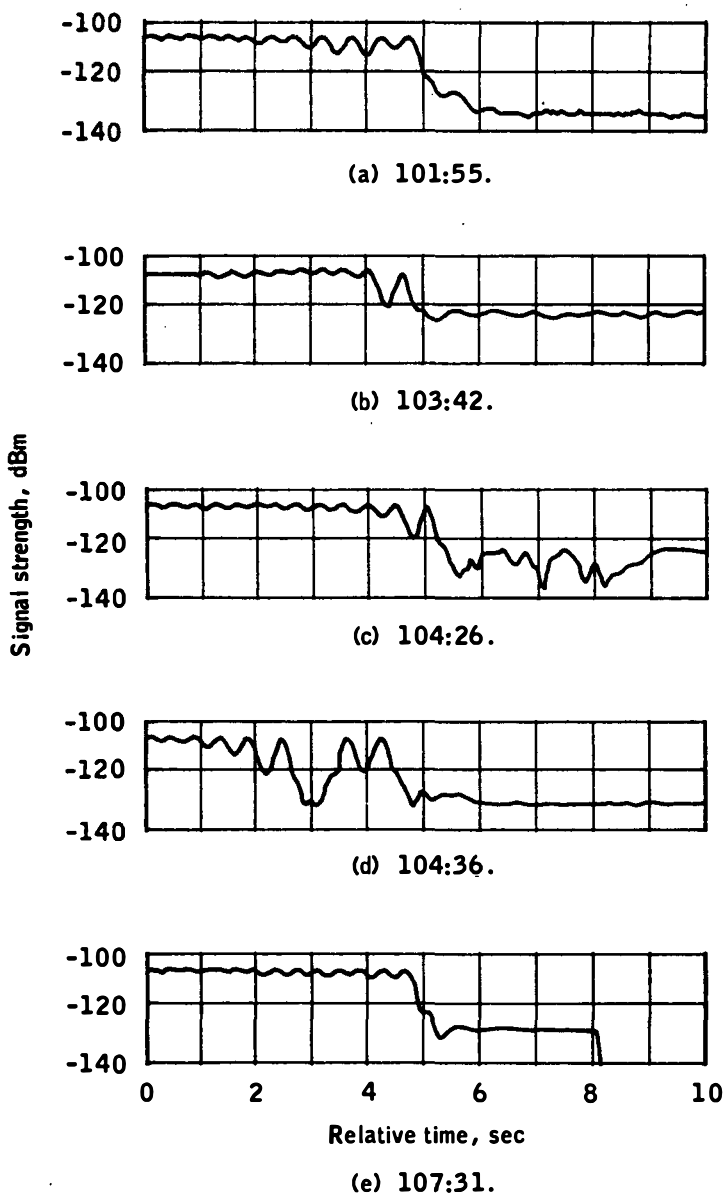

Figure 14-20.- Signal strength oscillations associated with five unexplained losses of lock.

Multipath normally occurs when the spacecraft is near the lunar horizon. However, antenna loss-of-lock did not occur at these times.

Noise transients on the uplink are held to a minimum because the ground station power amplifier operates in saturation. Also, the verification receiver which monitors the uplink signal at the ground station displayed normal output during the problem time periods. Although the incidental amplitude modulation has not been recently measured at Goldstone and Madrid, production sub-carrier oscillators have been checked. These tests showed that the incidental amplitude modulation at the critical frequencies was not detectable (less than 0.1 percent). A test was also performed which showed that the steerable antenna response to incidental amplitude modulation became worse with the addition of voice on the sub-carrier and the presence of pulse repetition ranging. However, there is no correlation between either of these and losses of antenna lock. The most probable causes for tracking loop instability are high loop gain, low gimbal friction, and low received signal strength resulting in low signal-to-noise ratio in the tracking loop. Both uplink and downlink signal strengths indicated that the RF levels were nominal and were within the antenna's capability to track.

The loop gain as measured during the acceptance test of the Apollo 14 equipment indicated a lower-than-nominal value indicating that the stability should have been greater than nominal.

There are no likely failures in the antenna that would cause a gain change sufficient to produce instability without complete loss of the antenna. There are many component failures in the transceiver which might produce the right amount of gain change for oscillations. However, these failures would also affect the receiver automatic gain control reading which appeared normal throughout the problem time.

The gimbal friction on the Apollo 14 antenna was measured during ground tests and found to be higher than nominal. This would increase the antenna stability. For gimbal friction to cause the problem, a variation in friction which characteristically changed from normal to low, or no friction, at short intervals and at random times consistent with the antenna oscillations would have had to occur.

There was no obvious variation in uplink signal and no obvious change in the antenna/transceiver tracking loop which would cause the antenna to oscillate. There must have been some intermittent condition that existed in the spacecraft/ground station system, which has not yet been identified. The investigation is continuing and an anomaly report will be issued when the investigation is completed.

An additional problem occurred one time during revolution 11 when the antenna pitch-position indicator stuck at the full-scale reading of 255 degrees. However, it became operative again and continued to function properly. This may have been caused by a failure in the position-sensing circuits in the antenna or in the meter itself. This meter hung up twice during acceptance testing. A malfunction was found, corrected, and a retest was successful. The indicator is used only as a gross indication of antenna movement. Consequently, no further action will be taken.

This anomaly is open.

14.2.4 Landing Radar Acquisition

Two conditions occurred during the landing radar operation which were not expected; however, they were not abnormal. The first condition occurred approximately 6 minutes after initial actuation of the landing radar. The system switched to the low-range scale, forcing the trackers into the narrow-band mode of operation. This was corrected by recycling the main power circuit breaker which switched the radar to high scale. Figure 14-21 shows the radar scale switching logic. The radar then locked on and "velocity-data-good" and "range-data-good" indications were transferred to the computer. The "velocity-data-good" signal is generated when the Doppler trackers lock on and the "range-data-good" signal is generated when the range tracker also locks on.

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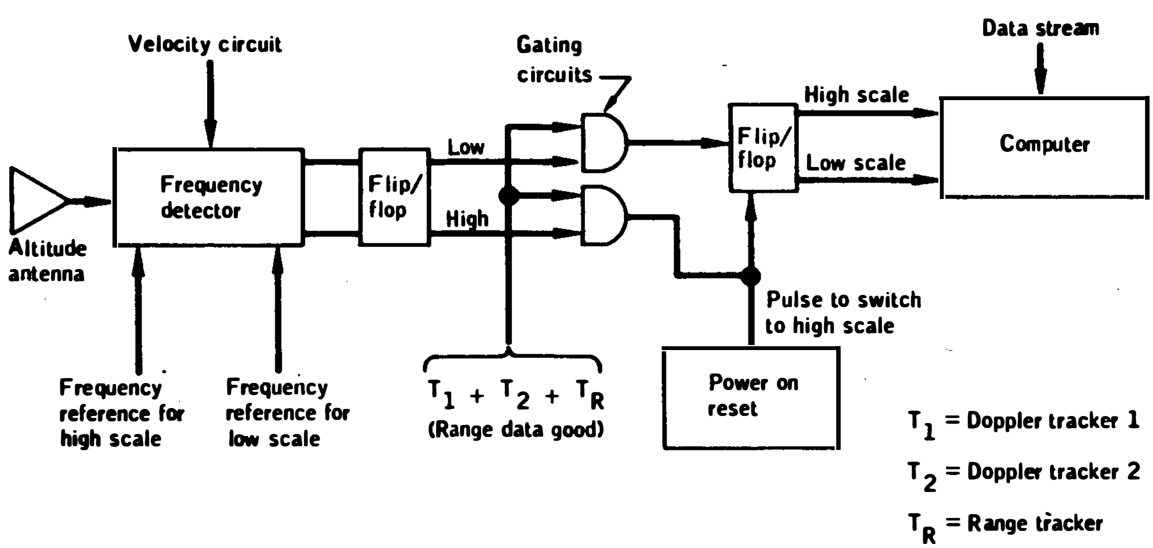

Figure 14-21.- Landing radar scale switching logic.

The second condition which was not expected occurred after the circuit breaker was recycled. At this time the initial slant range reading was approximately 13 000 feet greater than that calculated from the operational trajectory. Several seconds later, the indicated slant range jumped from 32 000 to 25 000 feet. Subsequently, the landing radar readings compared favorably with the operational trajectory (fig. 14-22).

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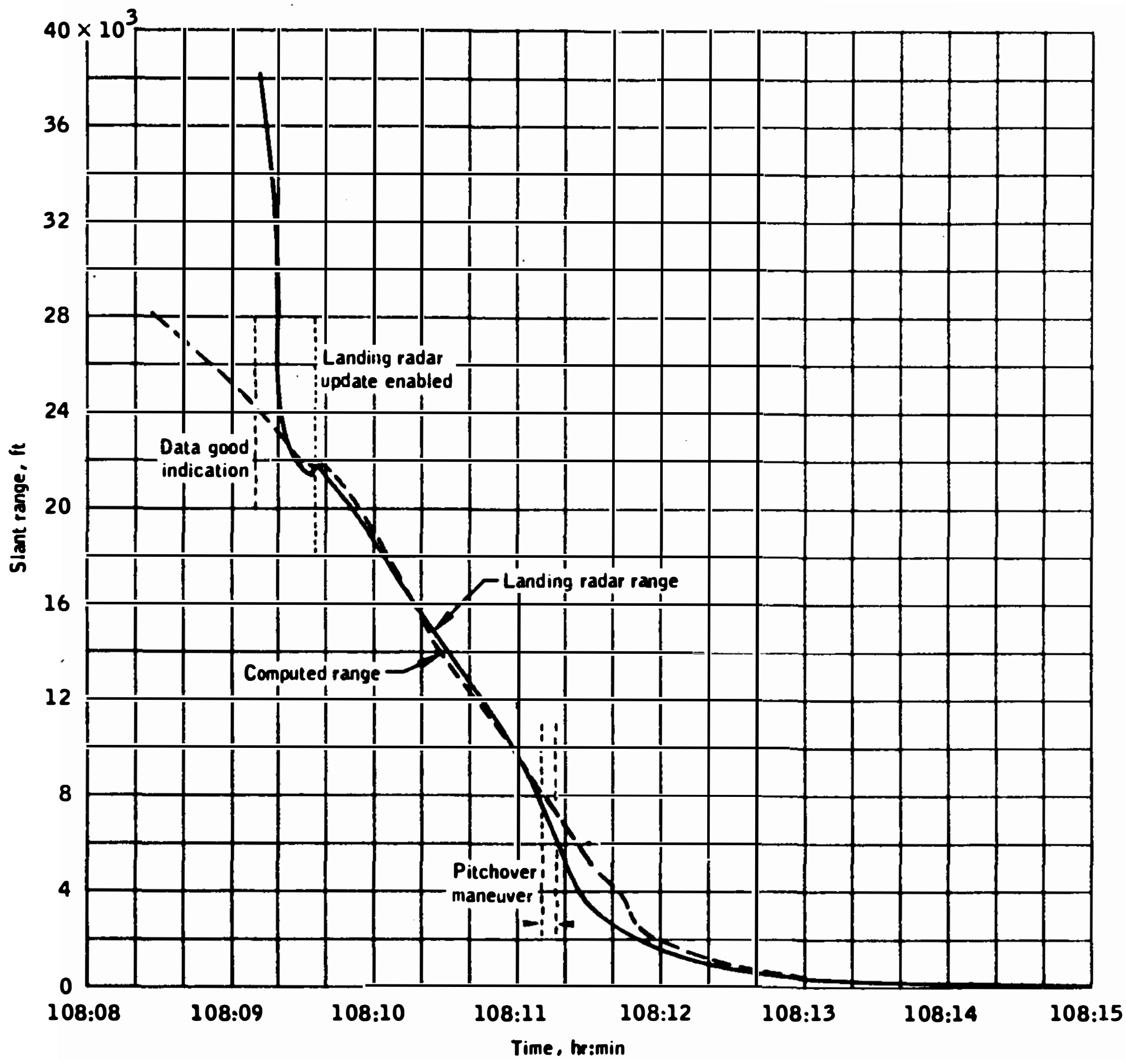

Figure 14-22.- Comparison of measured and computed slant range during powered descent.

The scale switching occurred at a slant range of 63 000 feet with a beam 4 velocity of 3000 ft/sec at an incidence angle of 35.4 degrees. Operating the landing radar under these conditions exceeds the maximum range measurement design limit (fig. 14-23). Under these conditions, the receiver is sweeping with maximum gain and the system will be sensitive to any received noise. A test was performed with a radar operating under the Apollo 14 conditions (two range-rate beams locked up and the range beam unlocked). By inserting low-level noise for a fraction of a second into the receiver, range scale switching occurred.

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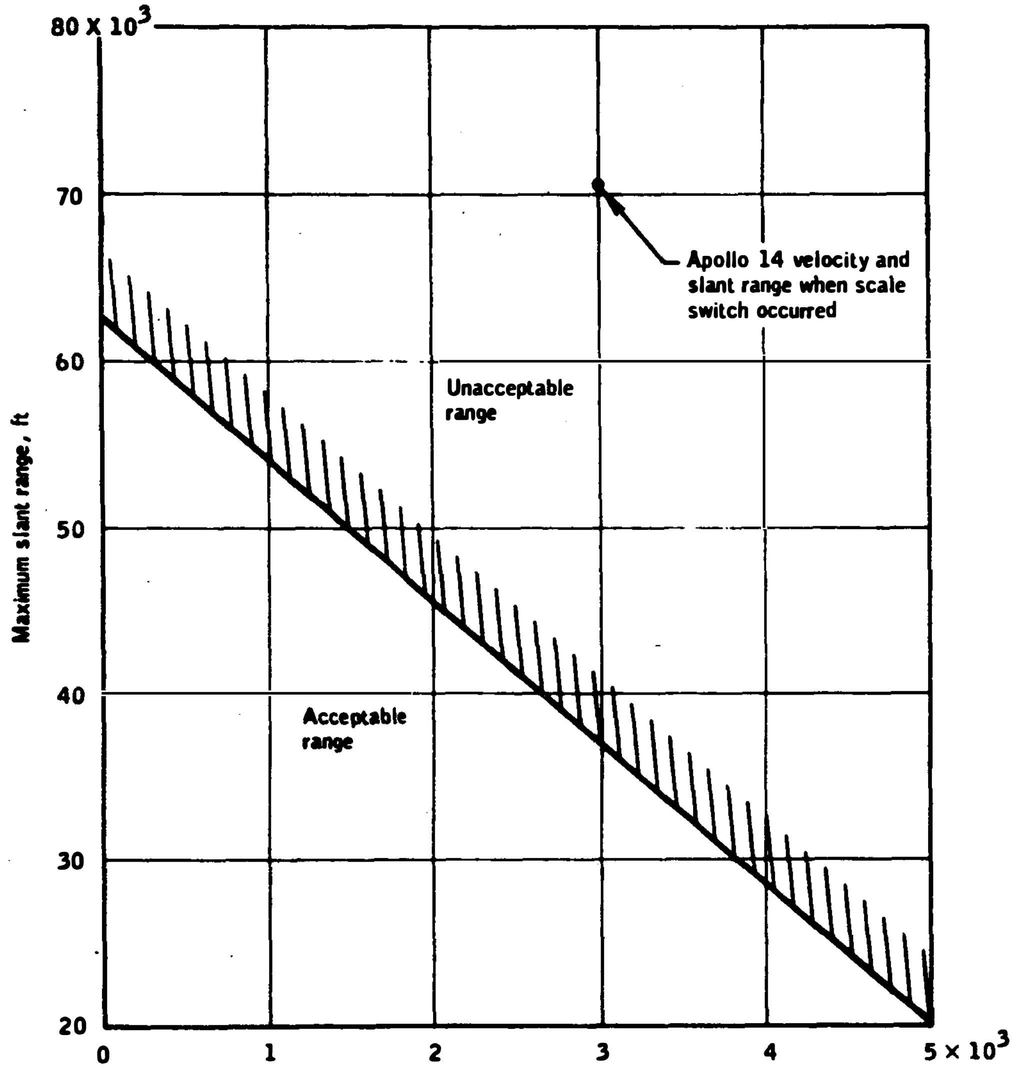

Figure 14-23.- Landing radar range measurement design limitation as a function of vehicle velocity component along range beam.

The high slant range indicated at lock-on by the landing radar was most likely caused by the radar locking onto energy returned into the antenna side lobe. Based on the preflight terrain profile and the preflight operational trajectory, side lobe lock-on can be expected. Checklist procedures exist to correct a sustained side lobe lock-on. Once the radar is locked on the main lobe, side lobe lock-on cannot occur.

On future spacecraft, a wiring modification will be made to enable holding the system in high scale while in antenna position 1. Low scale will only be enabled in position 2. Position 2 of the antenna is automatically selected by the computer at high gate (7500 feet altitude). The manual selection of antenna positions 1 and 2 will also control high scale and enable low scale switching, respectively.

This anomaly is closed.

14.2.5 Loss of the Abort Guidance System

The abort guidance system failed during the braking phase of rendezvous. Telemetry data were suddenly lost at 143:58:16; however, there was no indication of an abort guidance system warning light or master alarm. The crew was unable to access the data entry and display assembly and depressing any of the pushbuttons had no effect. The status switch was cycled from operate to standby to operate with no effect. Cycling the 28-volt circuit breakers likewise had no effect. The system remained inoperative for the remainder of the mission.

The system was determined to have been in the standby mode after the failure by comparing expected and actual bus current changes that were observed at the time of the failure and the subsequent cycling of the circuit breakers. Further evidence of the system having been in standby was the absence of the warning light and master alarm at the time of the failure. If standby power in the electronics assembly were not maintained, clock pulses to the abort sensor assembly would have been lost and the warning light would have illuminated and the master alarm sounded. A warning light and a master alarm would also have occurred if the failure had been in the abort guidance status switch or the associated external wiring. These conditions isolate the failure to the power supply section or the sequencer of the abort electronics assembly (fig. 14-24).

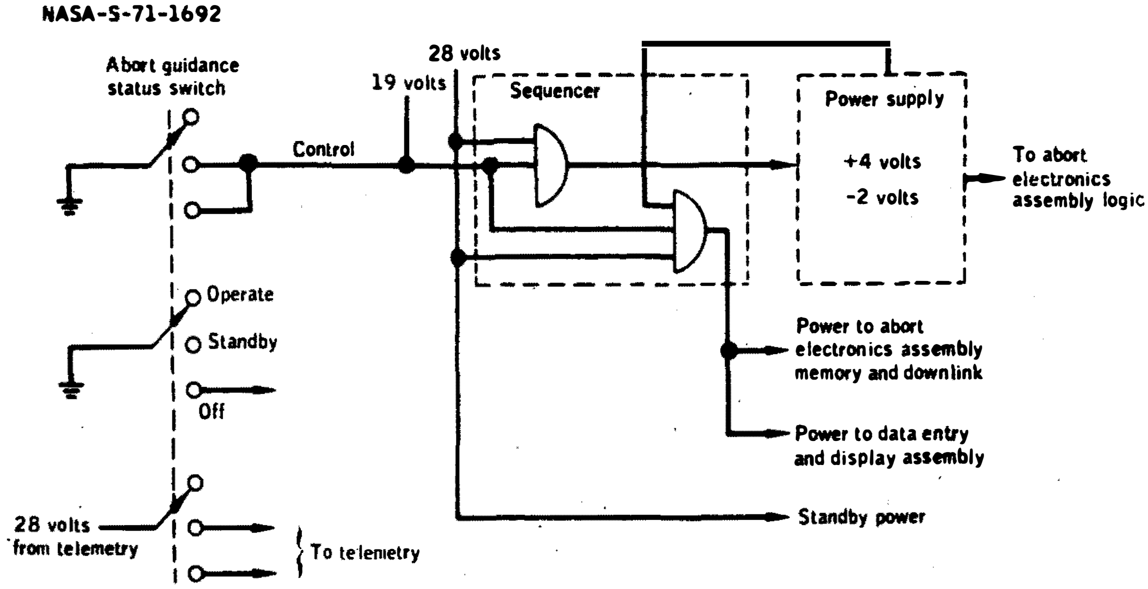

Figure 14-24.- Partial abort guidance system functional diagram.

The failure has been isolated to one of seven modules in the plus 4-volt logic power supply, one module in the sequencer, or one of 27 interconnections between the modules. There are a total of 27 component part types; twelve resistor, two capacitor, four transistor, four diode, four transformer, and one saturable reactor that could have caused the failure.

A complete failure history review of the component part types revealed no evidence of a generic part problem. A power dissipation analysis and a thermal analysis of maximum case temperature for each of the suspect parts showed adequate design margins.

Manufacturing procedures were reviewed and found to be satisfactory. Finally, a review was conducted of the testing that is performed at the component level, module level, and power supply level. Test procedures were found to be adequate for detection of failed units and not so severe that they would expose the units to unacceptable or hazardous test conditions.

A component or solder joint failure could have been due to either an abnormal thermal stress or a non-generic deficiency or quality defect that was unable to withstand a normal environment. An abnormal thermal stress could have been caused by improper installation of the equipment on the cold rails. If this occurred, the first component which should fail is in the particular power supply to which the failure was isolated.

In any event, the methods and techniques used to verify system performance show no apparent areas which require improvement. Further stress analysis of components and solder joints shows that the design is adequate. The methods, techniques and procedures used in installation of the equipment on the cold rails are also adequate, providing these procedures are followed. Consequently, no corrective action is in order.

This anomaly is closed.

14.2.6 Cracked Glass on Data Entry and Display Assembly

The crew reported a crack in the glass across the address register of the data entry and display assembly. Figure 14-25 shows the assembly and the location of the crack. Figure 14-26 is an enlarged drawing of the glass and associated electroluminescent segments.

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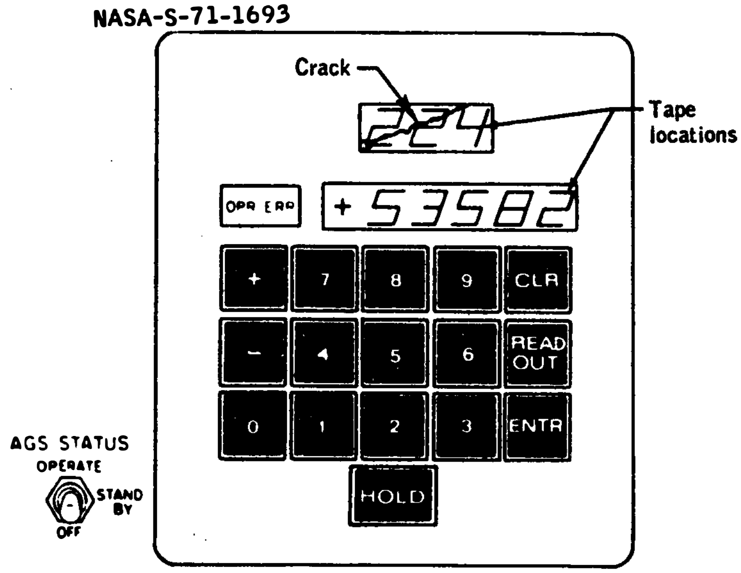

Figure 14-25.- Locations of crack and tape on data entry and display assembly.

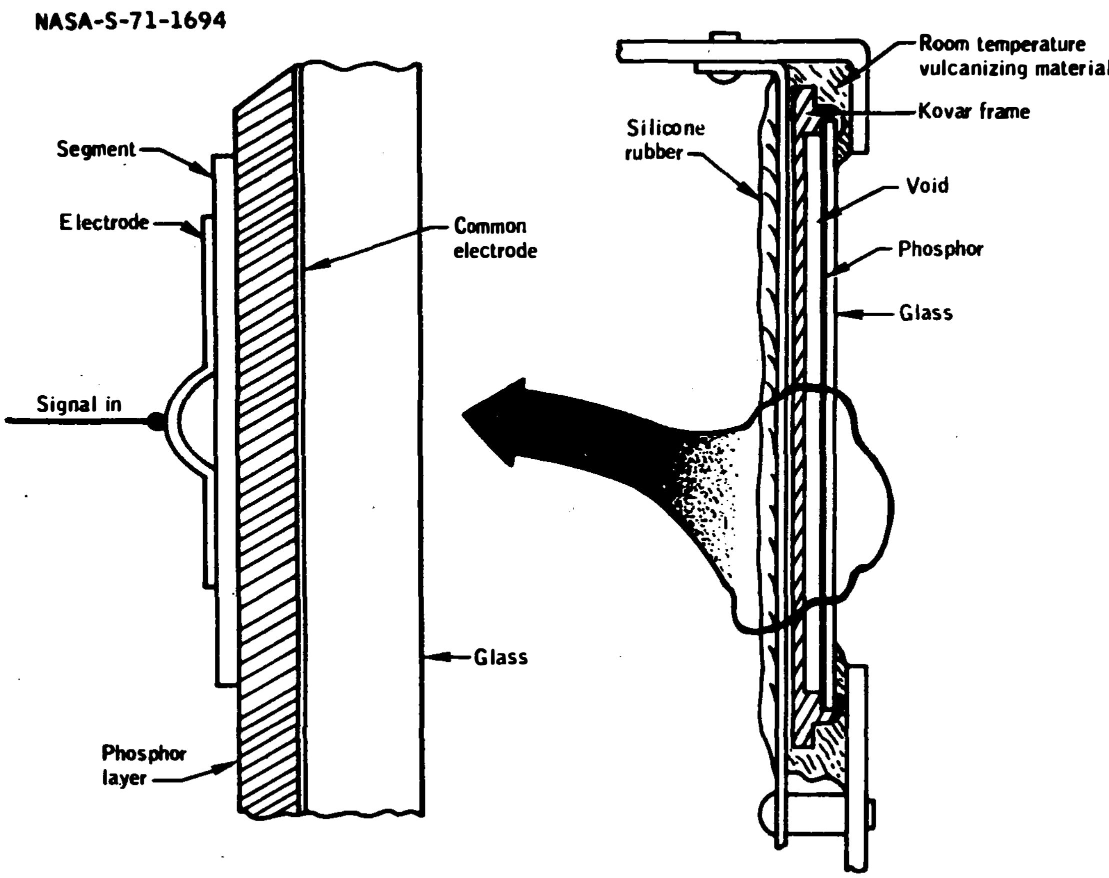

Figure 14-26.- Cross section of data entry and display assembly glass.

The cause of the crack is unknown. Glass cracks have not occurred since a revision was made to the procedure used to mount the glass to the faceplate of the data entry and display assembly. The assembly is qualified for an environment in excess of the flight conditions. Therefore, either excessive internal stresses (under normal conditions) were built into the glass, or the mounting was improper (not as designed), or the glass was inadvertently hit.

Corrective action consists of applying a clear plastic tape prior to flight on the glass of the electroluminescent windows above the keyboard (fig. 14-25), like that previously used on the mission timer windows. The tape is to prevent dislodging of any glass particles if cracks occur in the future, as well as help prevent moisture from penetrating the electroluminescent segments should a crack occur. The presence of moisture would cause the digit segments to turn dark in about 2 hours if voltage were applied to a cracked unit, making the assembly unreadable.

This anomaly is closed.

14.3 GOVERNMENT FURNISHED EQUIPMENT

14.3.1 Noisy Lunar Topographic Camera Operation

The lunar topographic camera exhibited noisy operation from the time of the Descartes site photography pass at about 90 hours. In both the operate and standby modes with power on the camera, the shutter operation was continuous.

The developed film indicates that the camera was functioning properly at the time of camera checkout at about 34 hours. On the fourth lunar revolution, good imagery of the lunar surface was obtained on 192 frames, starting at Theophilus Crater and ending about 40 seconds before passing the Descartes site. The rest of the film consists of multiple-exposed and fully over-exposed film.

Postflight tests with the flight camera showed satisfactory operation in all simulated environments (pressure, thermal, and vibration) at one-g. An intermittent failure was found in a transistor in the shutter control circuit (fig. 14-27). The transistor was contaminated with a

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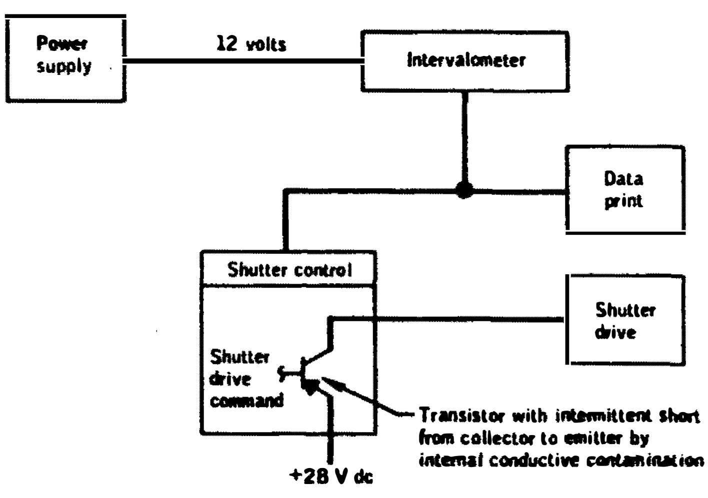

Figure 14-27.- Lunar topographic camera shutter control.

loose piece of aluminum 0.130 inch by 0.008 inch, which was foreign to the transistor material. With a shorted transistor, 28 volts is applied continuously to the shutter drive circuit, causing continuous shutter operation, independent of the intervalometer and independent of the single, auto, or standby mode selections. The sprocket holes in the 1/200 slot in the shutter curtain were torn as a result of the prolonged, continuous, high-speed shutter operation (fig. 14-28).

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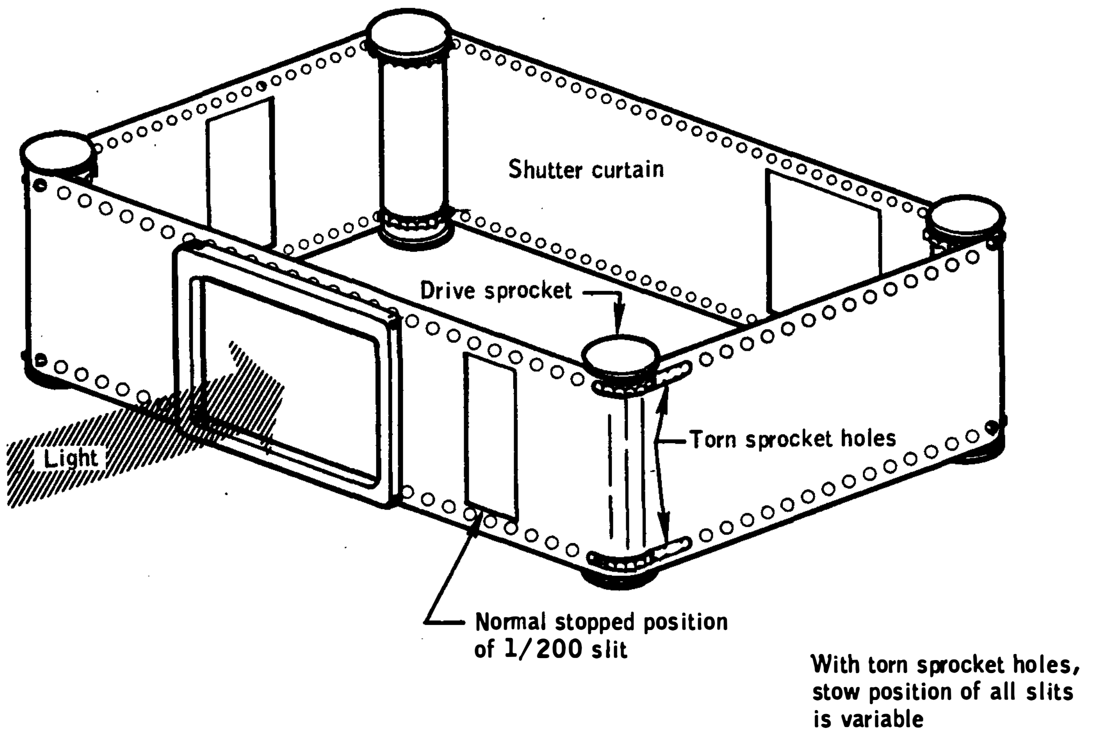

Figure 14-28.- Lunar topographic camera film track.

The transistor had been passed by normal high reliability screening and by premission and postmission system acceptance tests operating under vibration, thermal, pressure, and humidity conditions; none of which detected the piece of aluminum. Additional screening being considered for future applications includes the use of N-ray and acoustic inspection. An occurrence of this nature is rare, but it is even rarer for such a condition to pass the high reliability screening.

The anomaly occurred only after a period of operation at zero-g in flight, and when the case of the transistor itself was tapped postflight.

This anomaly is closed.

14.3.2 Extravehicular Glove Control

After suit pressurization for the second extravehicular activity, the Lunar Module Pilot reported that his right glove had pulled his hand to the left and down and that he had not had this trouble during the first extravehicular activity period. The condition was a nuisance throughout the second extravehicular activity period. Initial indications from the Lunar Module Pilot were that a cable had broken in the glove (fig. 14-29).

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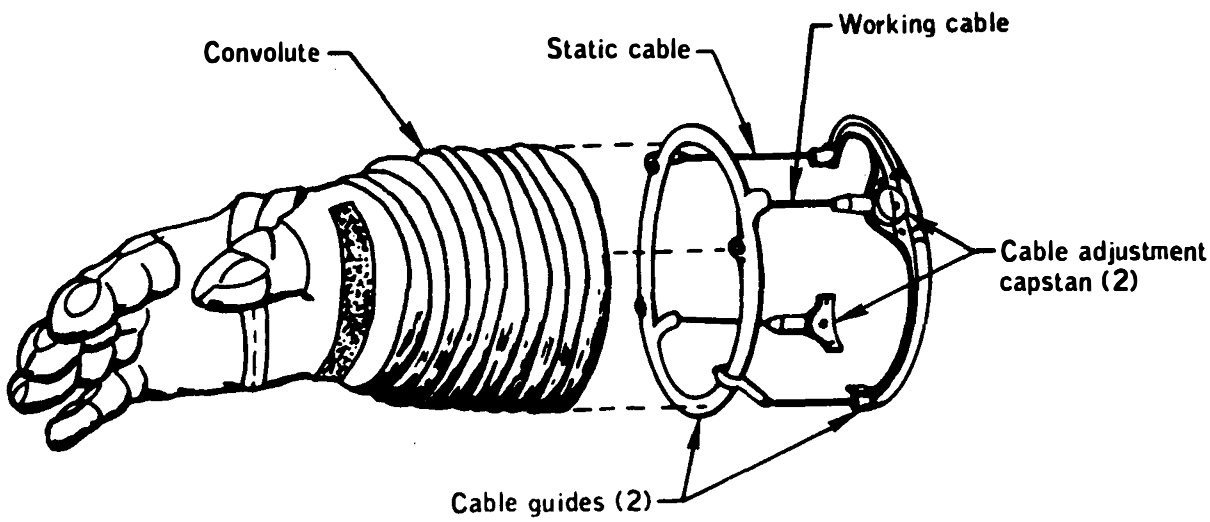

Figure 14-29.- Extravehicular glove wrist control.

A detailed examination of the returned glove, together with chamber tests, have shown that there are no broken cables and that there is free operation of the glove wrist-control cable system. However, with the Lunar Module Pilot in the pressurized flight suit, the glove took the position which was reported during the mission.

The wrist control assembly provides a free-moving structural interface between the glove and the wrist disconnect so as to assure convolute action for wrist movement in the pressurized state. The design inherently allows the glove to take various neutral positions.

This anomaly is closed.

14.3.3 Intervalometer Cycling

During intervalometer operation, the Command Module Pilot heard one double cycle from the intervalometer. Photography indicated that double cycling occurred 13 times out of 283 exposures.

Postflight testing with the flight intervalometer and camera has indicated that the double cycling was caused by a random response of the intervalometer to the camera motor current. The camera motor used on the Apollo 14 cameras was a new motor having slightly higher current characteristics. Preflight testing of the equipment indicated compatibility of the units and no double cycling.

Double cycling does not result in detrimental effects to the camera or the intervalometer. No loss of photographic data occurs as a result of double cycling. Modifications to the intervalometer to make it less sensitive to the random pulses of the camera motor will be made, if practical. On Apollo 15, the intervalometer will only provide Hasselblad backup to the scientific instrument module cameras.

This anomaly is closed.

14.3.4 Intermittent Voice Communications

At approximately 29 hours, Mission Control had difficulty in communicating with the Commander. The Commander replaced his constant wear garment electrical adapter (fig. 14-30) with a spare unit, and satisfactory communications were reestablished.

Following release of the hardware from quarantine, all four constant wear garment electrical adapters were tested for continuity and resistance, and all units were satisfactory. The adapters were then

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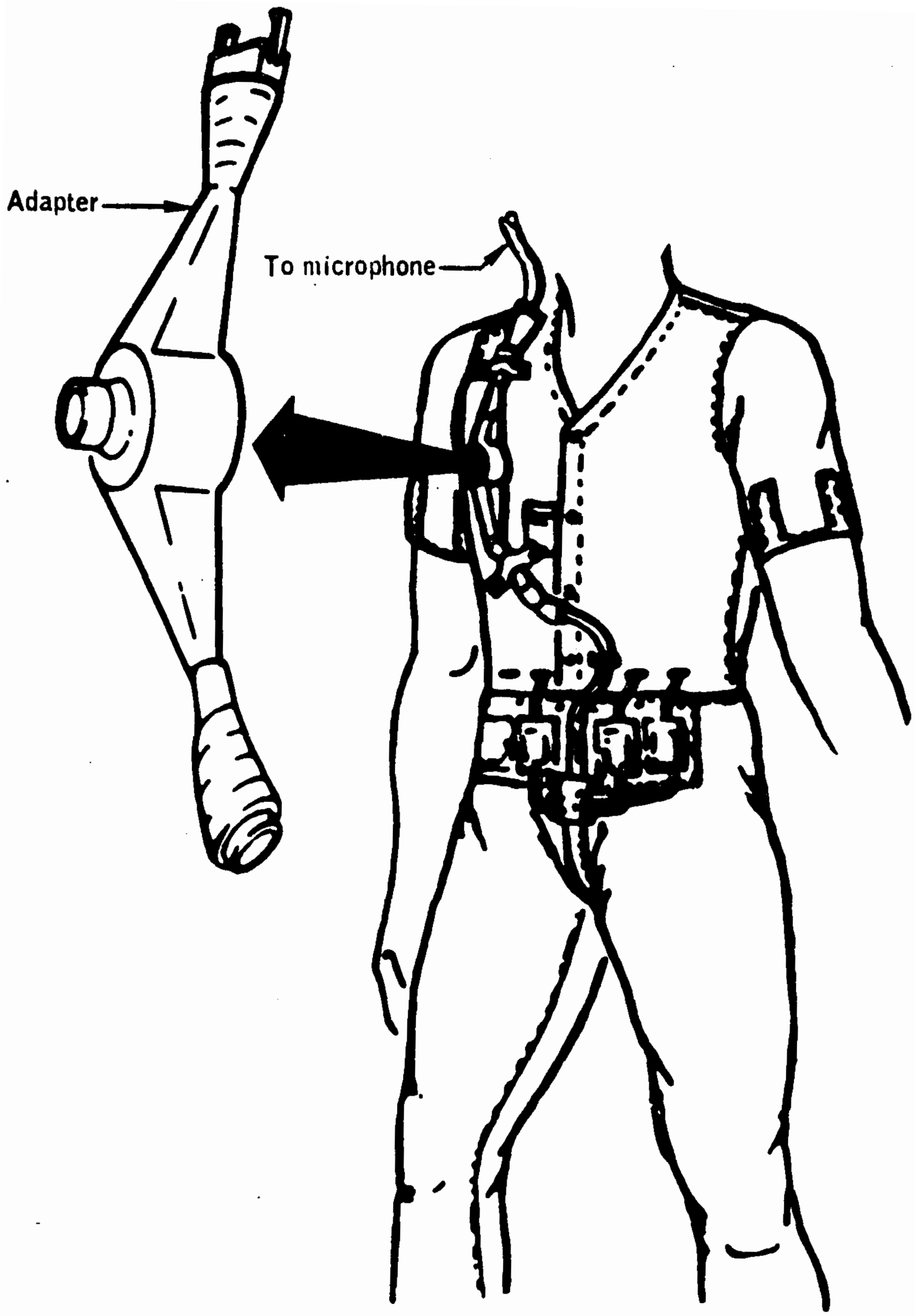

Figure 14-30.- Constant wear garment communications harness.

connected to a portable communications set which provided conditions similar to flight conditions. While connected, the adapters were subjected to twisting, bending, and pulling. None of the adapters showed any electrical intermittents.

The most likely cause of the problem was poor contact between connectors because of small contaminants or improper mating of a connector, which was corrected when the spare adapter was installed.

This anomaly is closed.

14.4 APOLLO LUNAR SURFACE EXPERIMENTS PACKAGE

14.4.1 Active Seismic Experiment Thumper Misfires

During the first extravehicular activity, the crew deployed the thumper and geophones and attempted to fire the initiators with the following results: 13 fired, 5 misfired, and 3 initiators were deliberately skipped to save time. In some instances, two attempts were made to fire each initiator. In addition, for the first four or five firings, it was necessary to squeeze the firing switch knob with both hands. Subsequently, the excessive stiffness seemed to be relieved and one-hand actuation was possible.

The most likely causes of the problem are associated with the detent portion of the selector switch (fig. 14-31) and dirt on the firing switch actuator bearing surface. The selector switch dial can reposition out of detent in the course of normal handling because of the lack of positive seating in the detent for each initiator position. For an initiator to be fired, the selector switch must provide contact to the proper unfired initiator position. Examination of the qualification unit has shown that the detent is positioned at the leading edge of the contact surface so that any movement toward the previous position will break the contact. Also, the lightening holes in the firing switch knob make it possible for dirt to get onto the Teflon bearing surfaces, temporarily increasing the force required to close the switch (fig. 14-31).

Corrective action for Apollo 16 consists of adding a positive detent mechanism, properly aligned with the selector switch contacts, and dust protection for the firing switch actuator assembly. The thumper is not carried on Apollo 15.

This anomaly is closed.

NASA-S-71-1699

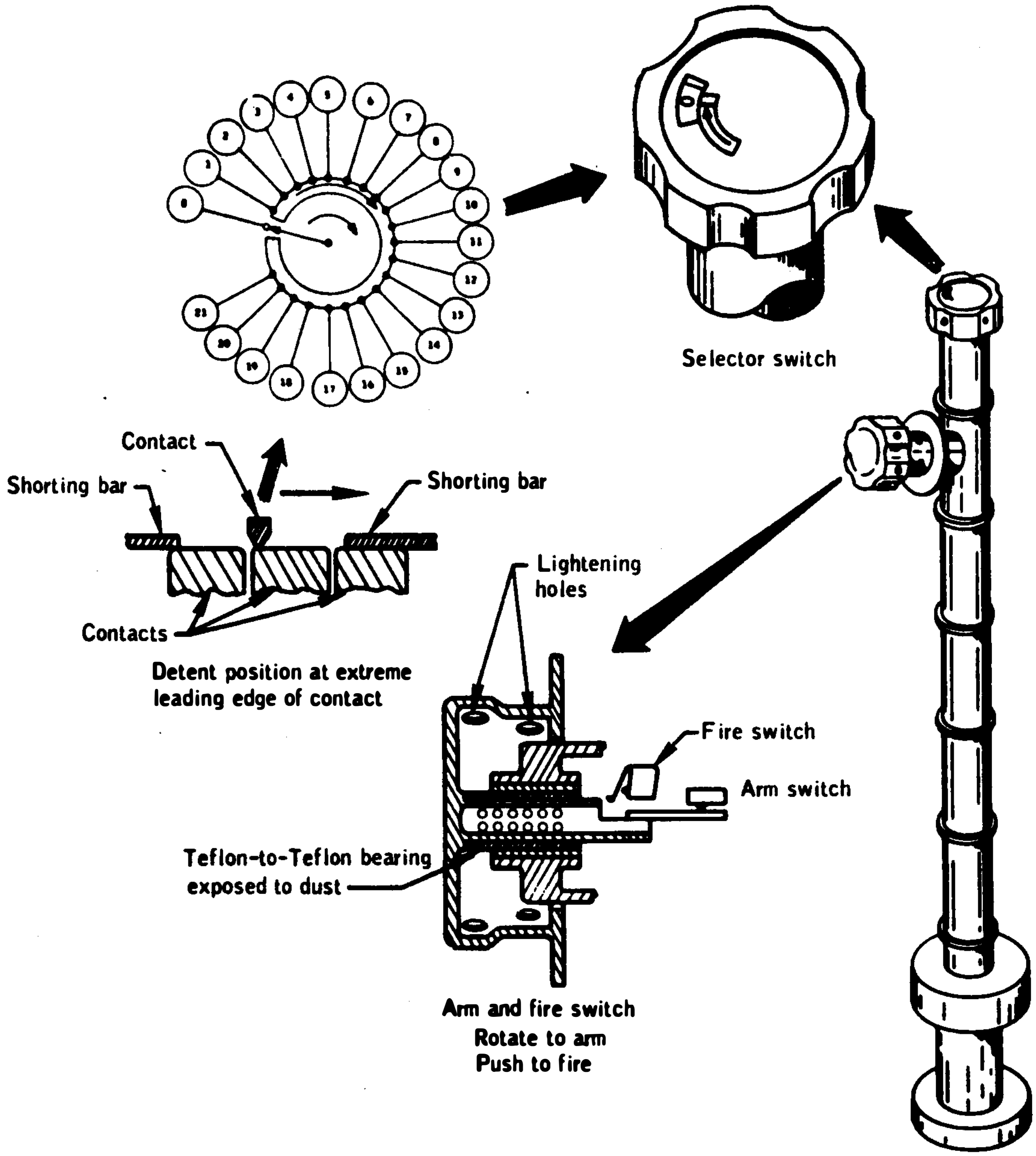

Figure 14-31.- Active seismic experiment.

14.4.2 Suprathermal Ion Detector Experiment Noisy Data

During initial turn-on of the Apollo lunar surface experiments, transmission of the suprathermal ion detector/cold cathode gage experiment operate-select command resulted in erratic data from the suprathermal ion detector experiment, the passive seismic experiment, and the charged particle lunar environment experiment. (Central station engineering parameters remained normal.) Subsequent commanding of the suprathermal ion detector/cold cathode gage experiments to the standby mode returned the other lunar surface experiment data to normal.

Several switching iterations of the central station and the experiment commands failed to clear the problem until the suprathermal ion detector experiment was commanded to the ×10 accumulation mode. Upon execution of this command, normal experiment data were received and the data have remained normal since that time. The suprathermal ion detector experiment dust cover and the cold cathode gage experiment dust seal had been removed at the time the data became normal.

The most probable cause was arcing or corona within the suprathermal ion detector equipment prior to dust cover removal. During ground tests under similar conditions, arcing or corona has resulted in the same type of data problems. Systems tests have indicated that the noise generated can also affect the passive seismic experiment and charged particle lunar environment experiment data; and that arcing or corona within the suprathermal ion detector experiment can result in spurious commands within the suprathermal ion detector experiment, causing removal of the dust protectors. However, no detrimental effects to the equipment have been experienced by this event.

Performance acceptance data from the Apollo 15 suprathermal ion detector/cold cathode gage experiments with the remaining lunar surface experiments have not indicated any abnormalities. The Apollo 15 unit will most likely exhibit the same characteristic arcing, with the dust covers intact and the high voltage on, as that of the Apollo 14 unit. However, operations prior to dust cover removal will be limited to the time required for operation verification prior to the last extravehicular activity.

This anomaly is closed.

14.4.3 Lunar Portable Magnetometer Cable Difficulties

The crew reported that it was difficult to rewind the lunar portable magnetometer cable. The cable is deployed and rewound at each location where the lunar portable magnetometer is used (fig. 14-32).

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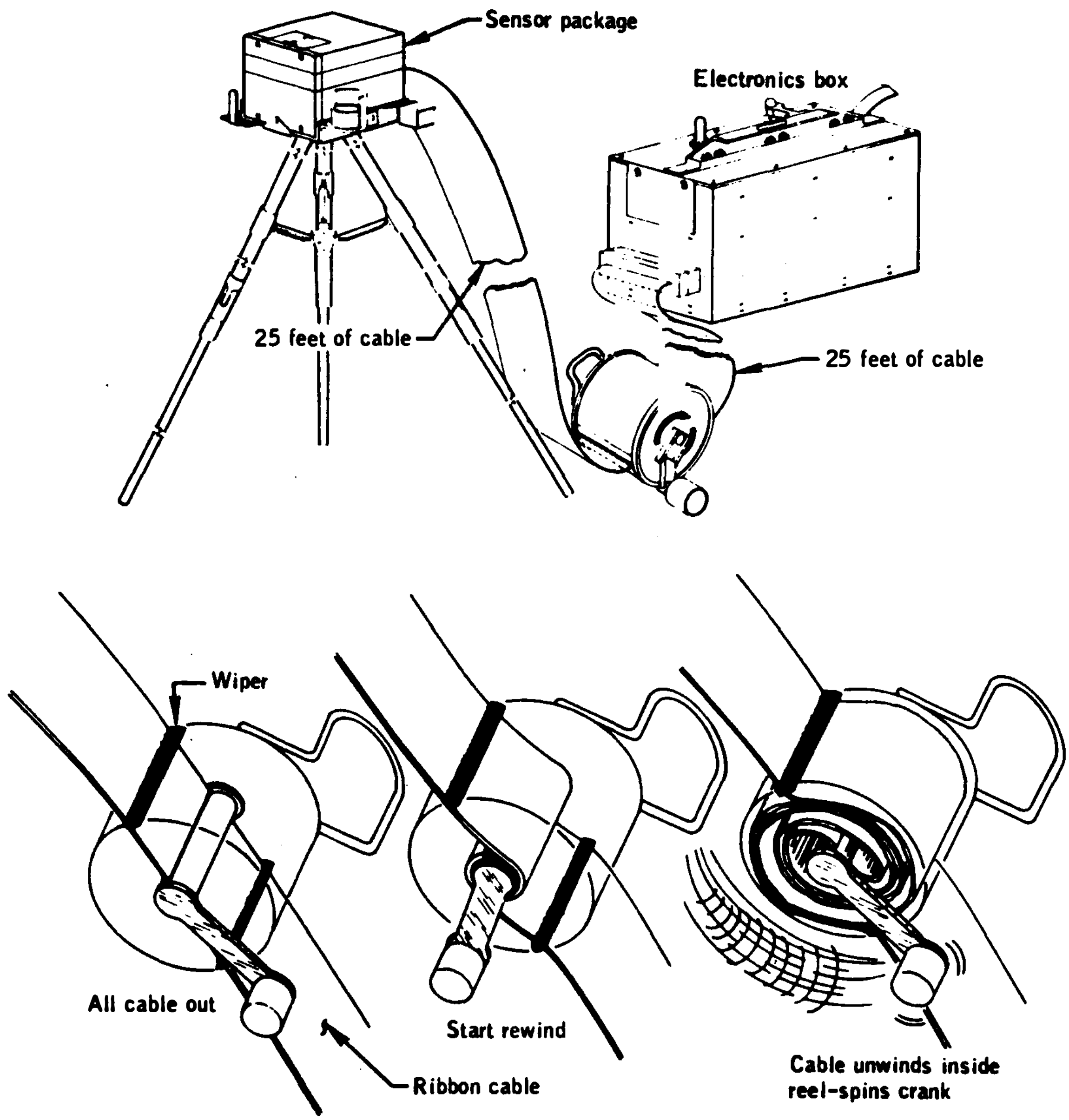

Figure 14-32.- Lunar portable magnetometer cable reel.

The lunar portable magnetometer ribbon cable snarls easily at 1/6g and is difficult and tedious to unsnarl. If it is necessary to remove the hand from the crank to unsnarl the cable during the first part of rewinding the cable, the cable will unwind within the reel and spin the reel handle (fig. 14-32). Free unwinding of the reel is required during deployment; however, it is desirable to be able to lock the reel against rotation at times during rewind of the cable. Rewinding was difficult because there was no provision to lock the reel during rewind, and gripping the reel and crank was difficult with the gloved hand.

Corrective action for Apollo 16 consists of adding a ratchet and pawl locking device for actuation with the gloved hand, and providing a better grip for the reel and crank. The lunar portable magnetometer is not carried on Apollo 15.

This anomaly is closed.

14.4.4 Central Station Twelve-Hour Timer Failure

The central station timer pulses did not occur after initial activation. Uplink command tests verified that the timer logic and the pulse switches were functioning satisfactorily, but that the mechanical section of the timer was not driving the switches. Timer functions started to occur and the 12-hour pulses were provided 13 times in succession, indicating that the timer battery and oscillator are satisfactory, but that the mechanical section is operating intermittently. The failure of the timer is associated with the mechanical design.

This anomaly is similar to the timer problem experienced on Apollo 12. The loss or erratic operation of the 12-hour timer output pulse has no adverse effect on experiments operations. The Apollo 15 central station has a new solid-state timer. The Apollo 14 central station will be turned off by ground command, as is planned for the Apollo 12 station.

This anomaly is closed.

14.4.5 Passive Seismic Experiment Y-Axis Leveling Intermittent

The horizontal Y-axis leveling motor of the gimbal leveling system operates intermittently (fig. 14-33). Although a command verification is received when commands are sent, power is not necessarily received by the motor. When there is an indication of power to the motor, the motor does operate. As a result, during the first lunar day, response to ground commands was normal except for 6 of the 22 commands when there was no response.

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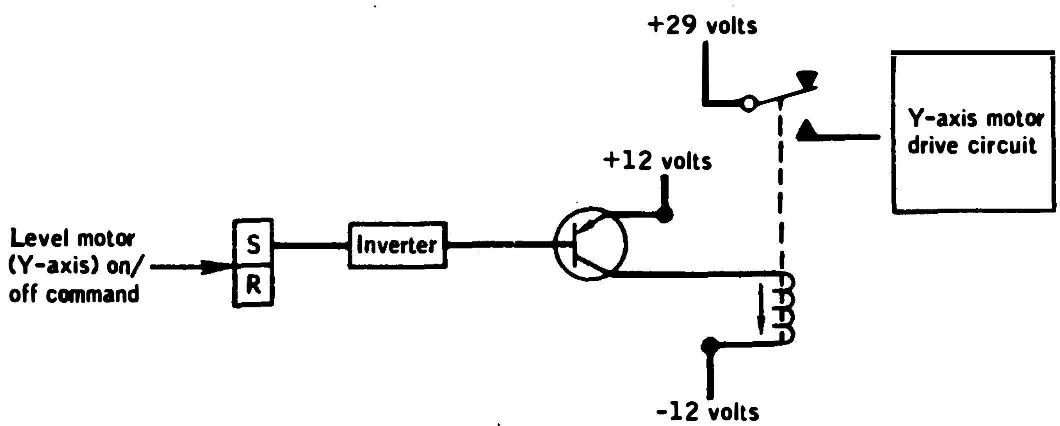

Figure 14-33.- Y-axis leveling motor circuitry.

Although no scientific data have been lost to date, intermittent problems have been encountered when leveling the Y-axis of the gimbal platform upon which are mounted the three orthogonal long-period seismometers. Occasionally, either there is no electro-mechanical response, or the response is delayed when the Y-axis motor is commanded on. Delay times vary. Thus far, leveling has been achieved by cycling on/off commands at varying time intervals.

The problem is caused by an intermittent component in the motor control circuit (fig. 14-33). There is no correlation between the occurrence of the problem and the temperature of the lunar surface, the central station electronics, or the experiment. Whenever there is an indication of power to the motor, the motor operates. When the motor operates, it operates properly and pulls the normal current.

If the problem becomes worse until Y-axis leveling cannot be achieved, an emergency operational mode can be implemented such as driving remaining axes to their stops in both directions in an attempt to free electro-mechanical components which may be sticking. Presently, however, the problem has not been sufficiently serious to warrant interruption of continuous scientific data to attempt such operations.

This anomaly is closed.

14.4.6 Passive Seismic Experiment Feedback Filter Failure

The long-period vertical (Z) seismometer was unstable when operated with the feedback filter in. The feedback filters for all three long-period axes (X, Y, and Z) were removed by command, and good data (undamped) now continue to be received. The filter-out mode provides feedback to the seismometer for all periods of operation with an instrument having a natural period of approximately 2.5 seconds. Although the response curves are peaked rather than flat, and critically damped, no seismic energy in the 0.5- to 15-second-period range is lost.

The filter-in mode provides a 1000-second time constant filter in the feedback loop for an instrument having a natural period of approximately 15 seconds with a critically damped, flat-response curve. On Apollo 14 long-period seismometers, the data during the filter-in mode have indicated a general characteristic of initial oscillations going on to saturation. The problem appears to be electrical rather than mechanical as experienced with the bent flexures of the Apollo 12 long-period vertical seismometer. Performance data during Apollo 14 acceptance testing have indicated no abnormalities.

Preliminary analysis of science data from Apollo 11, 12, and 14 indicates that the natural lunar seismic regime favors the range of 0.5- to 3.0-second periods. As a result it is quite probable that future passive seismic experiment units on the lunar surface will be operated in the filter-out mode in order to maximize the sensitivity at the apparently favored 2.0-second period. At present, both Apollo 12 and Apollo 14 long-period seismometers are being operated in the filter-out mode, producing satisfactory data.

This anomaly is closed.

14.4.7 Active Seismic Geophone 3 Electronic Circuit Erratic

The experiment was turned on in the listening mode on March 26, 1971, and geophone 3 data were spiking off-scale high (fig. 14-34). When the geophone channels were calibrated, the geophone 3 channel went off-scale high simultaneously with the start of the calibration pulse and stayed off-scale high for the remainder of the listening period. During the 1-second period when the calibration pulse was present, data from geophones 1 and 2 showed the normal 7-hertz ringing caused by the calibration pulse. However, geophone 3 data showed four negative-going spikes coincident with the first four negative half cycles of the ringing on the other two channels. The spikes decreased in duration from the first to the last, the last having an amplitude of 90 percent of

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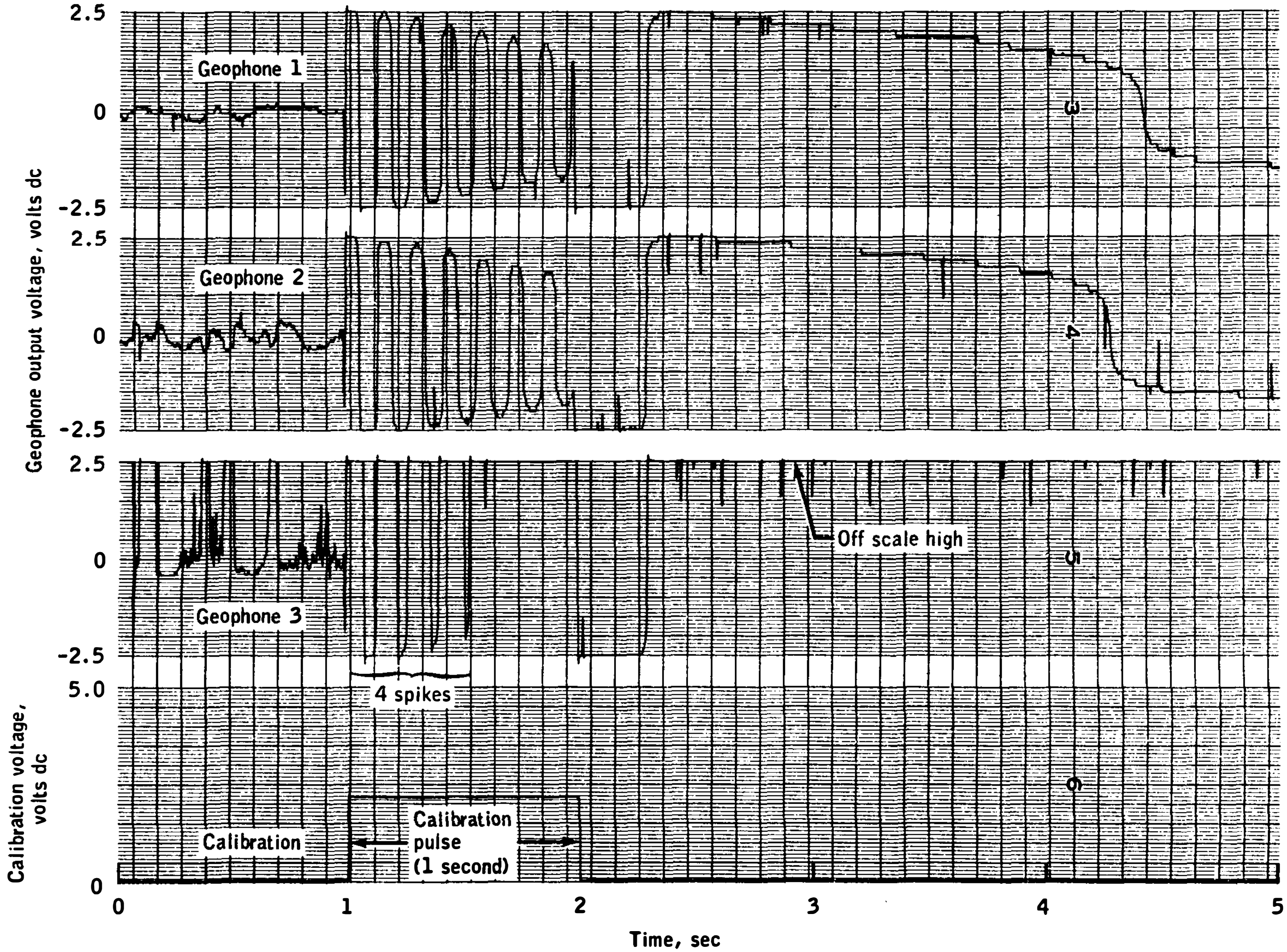

Figure 14-34.- Geophone calibration data.

full scale (plus 2.5 volts to minus 2.0 volts). During the time that this pulse was present, the signal on channel 2 changed from minus 2.2 volts to minus 2.35 volts, indicating that channel 3 was operating at an apparent gain of 30 times the channel 2 gain.

As shown in figure 14-35, each geophone channel consists of a geophone, an input preamplifier, a low-pass filter, and a logarithmic compressor amplifier. The output of the logarithmic compressor feeds the instrumentation system. The logarithmic compressor is basically an inverting amplifier with exponential negative feedback. Two diode-connected transistors between the output and input of the amplifier supply the feedback. The first diode is used for positive-going and the second for negative-going input signals. The diodes for all three geophone channels (two per channel) are physically located in an oven which controls their temperature at 105° C.

NASA-S-71-1703

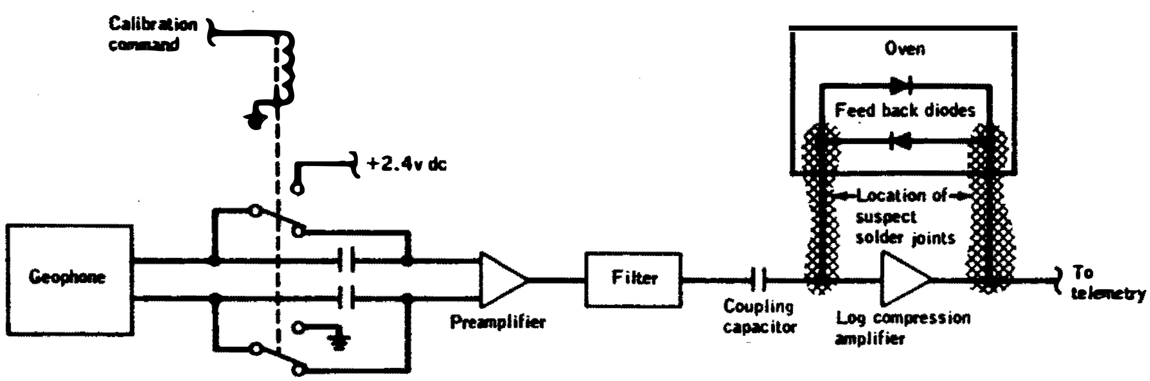

Figure 14-35.- Typical geophone channel.

It is believed that the failure is in the logarithmic compression amplifier because signals are coupled into it through an input coupling capacitor. This capacitor would block any offset voltages from the preceding stages which would be required to drive the output off-scale high. Analysis indicates that the most probable cause of the problem is an intermittent open circuit in the diode feedback path. This would allow the amplifier input transistor to saturate, driving the output off-scale high. When signals large enough to drive the input stage out of saturation were present, the output would then respond and the output signal would not be compressed.

The experiment electronics uses "cordwood" construction of the type which has caused solder cracks in other equipment. Two copper paths conduct the feedback diodes to the logarithmic compressor amplifier. A solder crack in either path would then result in the data characteristics.

There are 10 such solder joints for each geophone (fig. 14-36): four on the oven terminal board, four on the mother board, one on the top board of the log compressor module, and one on the bottom board of the log compressor module. The one most likely to fail first is on the top board of the log compressor module. Continuity at the joint recovers as long as the crack closes during the lunar day.

NASA-S-71-1704

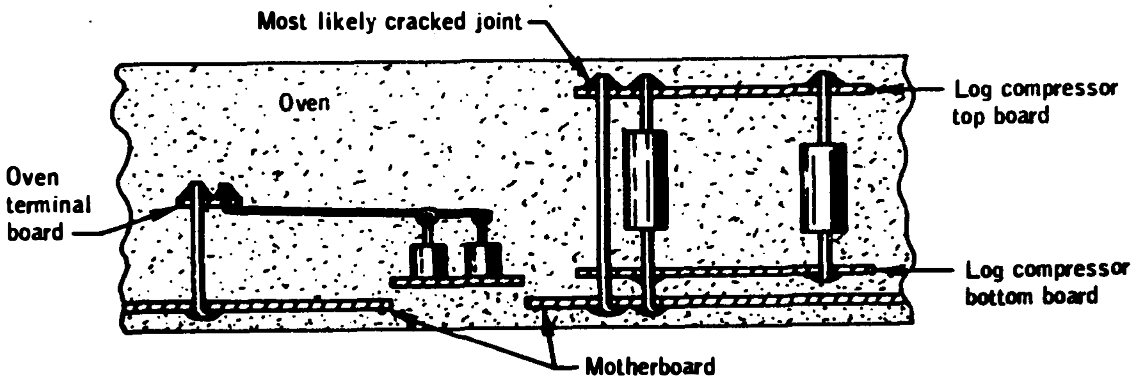

Figure 14-36.- Suspected cracked solder joints in amplifier.

The log compressor modules for geophones 1 and 2 are of the same type construction. Since these are located slightly further from the oven than the one for geophone 3, the maximum temperature may not be quite as high. As a result, it may take longer for them to crack, if at all.

Systems testing included operational thermal cycling tests over the temperature range for lunar day and night. However, cracked solder joints are a function of time as well as temperature, and apparently the ground test cycle did not allow enough time for a creep failure. This equipment was designed and built prior to the time when it was found that cordwood construction with soldered joints was unsatisfactory.

A breadboard of the logarithmic compressor has been constructed, and the diode feedback loop will be opened to duplicate the experiment data. The mechanical design of the logarithmic compressor will be reviewed to determine the changes that must be made to prevent solder cracks on Apollo 16. The active seismic experiment is not carried on Apollo 15.

Procedural changes under consideration include operation of the oven to maintain compressor module temperature because the solder joint which is most likely cracked is in compression (stronger) at the higher temperature.

This anomaly is open.

14.4.8 Intermittent Loss of Valid Data from Suprathermal Ion Detector Experiment Positive Analog-to-Digital Converter

The data in words 2, 3, 7, and 8 of the suprathermal ion detector experiment became erratic at 19:09 G.m.t. on April 5, 1971. This condition continued until 22:15 G.m.t. on April 6. The same erratic condition was also observed during operational support periods on April 7, 9, and 21. Only those measurements associated with the positive section of the log analog-to-digital converter were affected. There has been no loss of science data.

The affected measurements have a data characteristic wherein each parameter processed by the positive log analog-to-digital converter initially indicates full-scale output, followed by an erroneous data value. The erroneous data value correlates with the value of the preceding measurement in the serial data format processed by the negative analog-to-digital converter. The erroneous data value in some instances indicates one PCM count less than the negative analog-to-digital converter parameter.

An intermittent failure of the start reset pulse for the positive log analog-to-digital converter control logic (fig. 14-37) could cause the problem. Although the failure permits the positive converter initial output to fill the eight-bit binary counter and produce a full-scale reading; thereafter, when a start pulse for the positive converter should reset the eight-bit counter, it fails to do so, and the negative word which is still in the counter is read out as a positive word. The cause appears to be an intermittent component or wire connection in one of the associated modules. However, it does not appear to be a function of the temperature. The components have been passed by normal high reliability screening, and systems tests have included operational pressure, temperature, vibration, humidity, and accelerated lunar environment cycles. No failure of this

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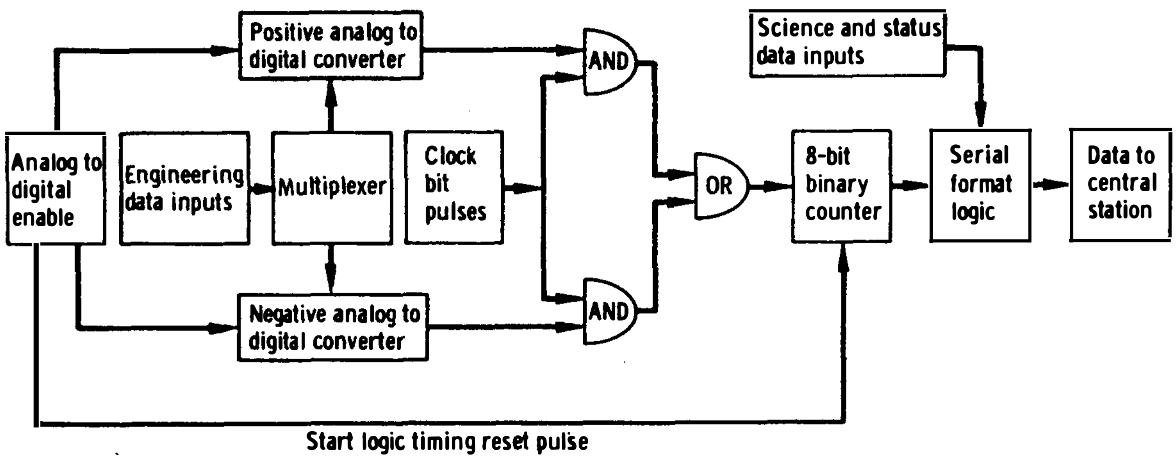

Figure 14-37.- Simplified data logic control.

type has been experienced with ground tests. No additional testing is considered warranted for Apollo 15, which will be the last mission for the experiment.

This anomaly is closed.

14.4.9 Charged Particle Lunar Environment Experiment Analyzer B Data Lost

The voltage measurement reading on the analyzer B power supply (fig. 14-38) became erratic on April 8, 1971, and the analyzer B science data were lost.

On April 10 and 16, the experiment was commanded on to normal (low-voltage) mode, and to increase (high-voltage) mode in a series of tests. The results indicate that the plus 28-volt input, the regulator, and the analyzer A power supply were functioning properly, and that the problem was in the analyzer B power supply.

The high-voltage power supply is a transistor oscillator. The resonant elements are a transformer primary winding and a capacitor connected in parallel between the transistor emitter and ground. A second transformer winding provides positive feedback to the transistor base, causing

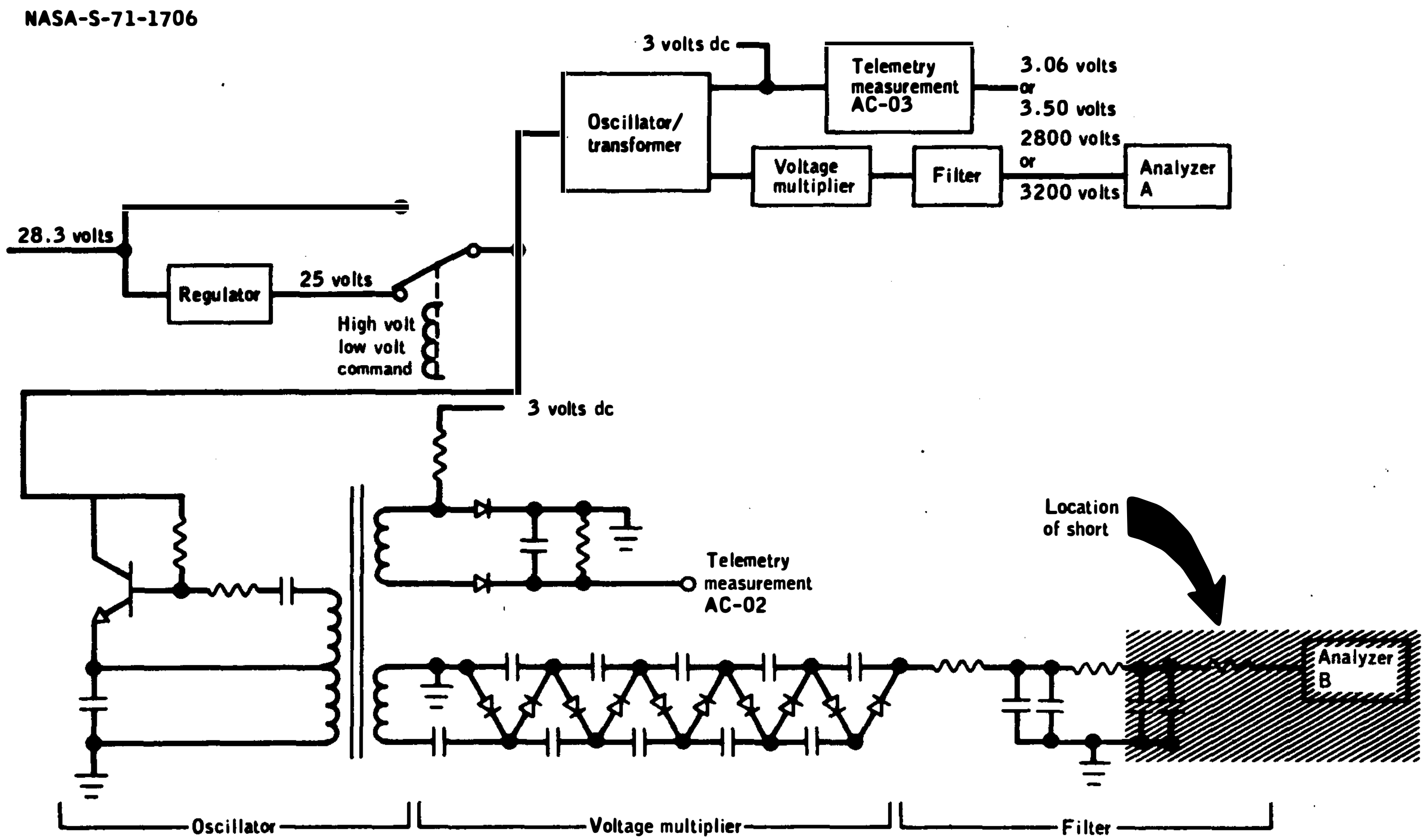

Figure 14-38.- Analyzer power supplies.

the circuit to oscillate. A third transformer winding supplies the input to a diode-capacitor voltage multiplier chain. The output of the voltage multiplier is then filtered and drives the charged particle analyzer. The output of the fourth transformer winding is rectified and filtered. The filtered voltage is then monitored by the instrumentation system and is proportional to the high voltage supplied to the analyzer.

Data indicated that after the failure occurred, the instrumentation output was between 2.00 and 2.25 volts dc. This could not occur if the oscillator were not still oscillating. The input to the voltage multiplier is also proportional to the instrumentation output. Shorts to ground can be postulated at various points in and downstream of the voltage multiplier, and the short circuit current can be reflected back into the transformer primary winding to determine how much the output voltage should be decreased. The decrease occurs because the transformer primary winding (the driving winding) has resistance (about 300 ohms), and any voltage dropped across this resistance is not available to drive the transformer.

These calculations show that the short circuit must be in one of the output filter capacitors in the high-voltage filter, in the interconnecting cable between the filter and analyzer, or in the analyzer. Short circuits in any other locations would result in a much lower instrumentation output voltage.

This is the last time the charged particle lunar environment experiment will be flown. If the failure propagates to the point where the malfunctioning power supply stops oscillating, the current taken by this supply would increase to about 0.1 ampere. If this is sufficient to damage the series voltage regulator used for low-voltage operation, the operating procedures will be modified to use low-voltage operation as little as possible to extend the voltage regulator's life.

This anomaly is closed.

15.0 CONCLUSIONS

The Apollo 14 mission was the third successful lunar landing and demonstrated excellent performance of all contributing elements, resulting in the collection of a wealth of scientific information. The following conclusions are drawn from the information in this report.

1. Cryogenic oxygen system hardware modifications and changes made as a result of the Apollo 13 failure satisfied, within safe limits, all system requirements for future missions, including extravehicular activity.

2. The advantages of manned spaceflight were again clearly demonstrated on this mission by the crew's ability to diagnose and work around hardware problems and malfunctions which otherwise might have resulted in mission termination.

3. Navigation was the most difficult lunar surface task because of problems in finding and recognizing small features, reduced visibility in the up-sun and down-sun directions, and the inability to judge distances.

4. Rendezvous within one orbit of lunar ascent was demonstrated for the first time in the Apollo program. This type of rendezvous reduces the time between lunar lift-off and docking by approximately 2 hours from that required on previous missions. The timeline activities, however, are greatly compressed.

5. On previous lunar missions, lunar surface dust adhering to equipment being returned to earth has created a problem in both spacecraft. The special dust control procedures and equipment used on this mission were effective in lowering the overall level of dust.

6. Onboard navigation without air-to-ground communications was successfully demonstrated during the transearth phase of the mission to be sufficiently accurate for use as a contingency mode of operation during future missions.

7. Launching through cumulus clouds with tops up to 10 000 feet was demonstrated to be a safe launch restriction for the prevention of triggered lightning. The cloud conditions at lift-off were at the limit of this restriction and no triggered lightning was recorded during the launch phase.

APPENDIX A - VEHICLE DESCRIPTION

The Apollo 14 space vehicle consisted of a block II configuration spacecraft and a Saturn V launch vehicle (AS-509). The assemblies comprising the spacecraft consisted of a launch escape system, command and service modules (CSM-110), a spacecraft/launch vehicle adapter, and a lunar module (LM-8). The changes made to the command and service modules, the lunar module, the extravehicular mobility unit, the lunar surface experiment equipment, and the launch vehicle since the Apollo 13 mission are presented. The changes made to the spacecraft systems are more numerous than for previous lunar landing missions primarily because of improvements made as a result of the Apollo 13 problems and preparations for more extensive extravehicular operations.

A.1 COMMAND AND SERVICE MODULE

A.1.1 Structural and Mechanical Systems

The major structural changes were installations in the service module to accommodate an additional cryogenic oxygen tank in sector 1 and an auxiliary battery in sector 4. These changes are discussed further in section A.1.3.

Structural changes were made in the spacecraft/launch vehicle adapter as follows. A door was installed at station 547 (305 deg) to provide access to quadrant 2 of the lunar module descent stage where Apollo lunar surface experiment subpackages 1 and 2 were stowed. Also, doublers were bonded on the adapter at station 547 (215 deg) in case a similar door had been required for contingency access to the lunar module cryogenic helium tank during prelaunch operations.

The interior of gussets 3 and 4, which contain the breech-plenum assemblies of the forward heat shield jettisoning system, were armored with a polyimide-impregnated fiberglass to prevent burn-through of the gussets and possible damage to adjacent equipment in the event of escaping gas from the breech assemblies.

A.1.2 Environmental Control System

The postlanding ventilation valves were modified to incorporate dry (non-lubricated) brake shoes to prevent possible sticking and a second shear pin was added to insure positive drive between the actuator shaft and cam.

To provide controlled venting for an oxygen tank flow test, the internal diameter of the auxiliary dump nozzle (located in the side hatch) was enlarged.

Sodium nitrate was added to the buffer ampules used in sterilizing the potable water. Addition of the sodium nitrate was to reduce system corrosion and enhance the sterilization qualities of the chlorine.

A vacuum cleaner with detachable bags was added to assist in removing lunar dust from suits and equipment prior to intravehicular transfer from the lunar module to the command module after lunar surface operations, and for cleanup in the command module.

A.1.3 Electrical Power System

The electrical power system was changed significantly after the Apollo 13 cryogenic oxygen subsystem failure. The major changes are as follows.

a. The internal construction of the cryogenic oxygen tanks was modified as described in the following table.

Previous block II vehicles	CSM-110 and subsequent vehicles
Each tank contained two destratification fans.	Fans were deleted.
Quantity gaging probe was made of aluminum.	Quantity gaging probe material was changed to stainless steel.
Heater consisted of two parallel-connected elements wound on a stainless steel tube.	Heater was changed to three parallel-connected elements with separate control of one element.
Filter was located in tank discharge.	Filter was relocated to external line.
Tank contained heater thermal switches to prevent heater element from overheating.	Heater thermal switches were removed.
Fan motor wiring was Teflon-insulated.	All wiring was magnesium oxide-insulated and sheathed with stainless steel.

b. A third cryogenic oxygen storage tank was installed in sector 1 of the service module. This tank supplied oxygen to the fuel cells and could be used simultaneously with the two tanks in sector 4. A new isolation valve was installed between tanks 2 and 3 to prevent the loss of oxygen from tank 3 in the event of damage to the plumbing for tanks 1 and 2. The closed isolation valve also would have prevented the flow of oxygen from tank 3 to the fuel cells. However, tank 3 could have supplied the environmental control system with the isolation valve closed while the auxiliary battery, mentioned in paragraph e, was the source of electrical power.

c. The tank 1 and 2 pressure switches remained wired in series as in the previous configuration; the tank 3 switch was wired in parallel and was independent of tanks 1 and 2.

d. The fuel cell shutoff valve used previously was an integral forging containing two check valves and three reactant shutoff valves. In the valve used for CSM-110, the two check valves remained in the integral forging; however, the reactant shutoff valves were removed and replaced by three valves relocated in line with the integral forging. These valves were the same type as those used in the service module reaction control helium system. The valve seals were changed to a type that provides a better seal under extreme cold. Figure A-1 illustrates the major changes to the system except for the internal tank changes.

e. An auxiliary battery, having a capacity of 400-ampere hours, was installed on the aft bulkhead in sector 4 of the service module to provide a source of electrical power in case of a cryogenic subsystem failure. Two control boxes, not used on previous flights, were added to accommodate the auxiliary battery. One box contained two motor switches which could disconnect fuel cell 2 from the service module and connect the auxiliary battery in its place. The second box contained an overload sensor for wire protection.

A.1.4 Instrumentation

Six new telemetry measurements associated with the high-gain antenna were added to indicate pitch, yaw, and beam-width, and whether the antenna was operating in the manual, automatic tracking, or reacquisition mode. This additional instrumentation provided data to support Flight Control management of the high-gain antenna.

Other instrumentation changes were as follows. The cabin pressure transducer was replaced with one which had been reworked, cleaned, and inspected for contaminants. In the past, loose nickel-plating particles had interfered with inflight measurements. Additional instrumentation was incorporated to monitor the auxiliary battery, the oxygen tank heater element temperatures, the oxygen tank 2 and 3 manifold pressure, and the tank 3 pressure.

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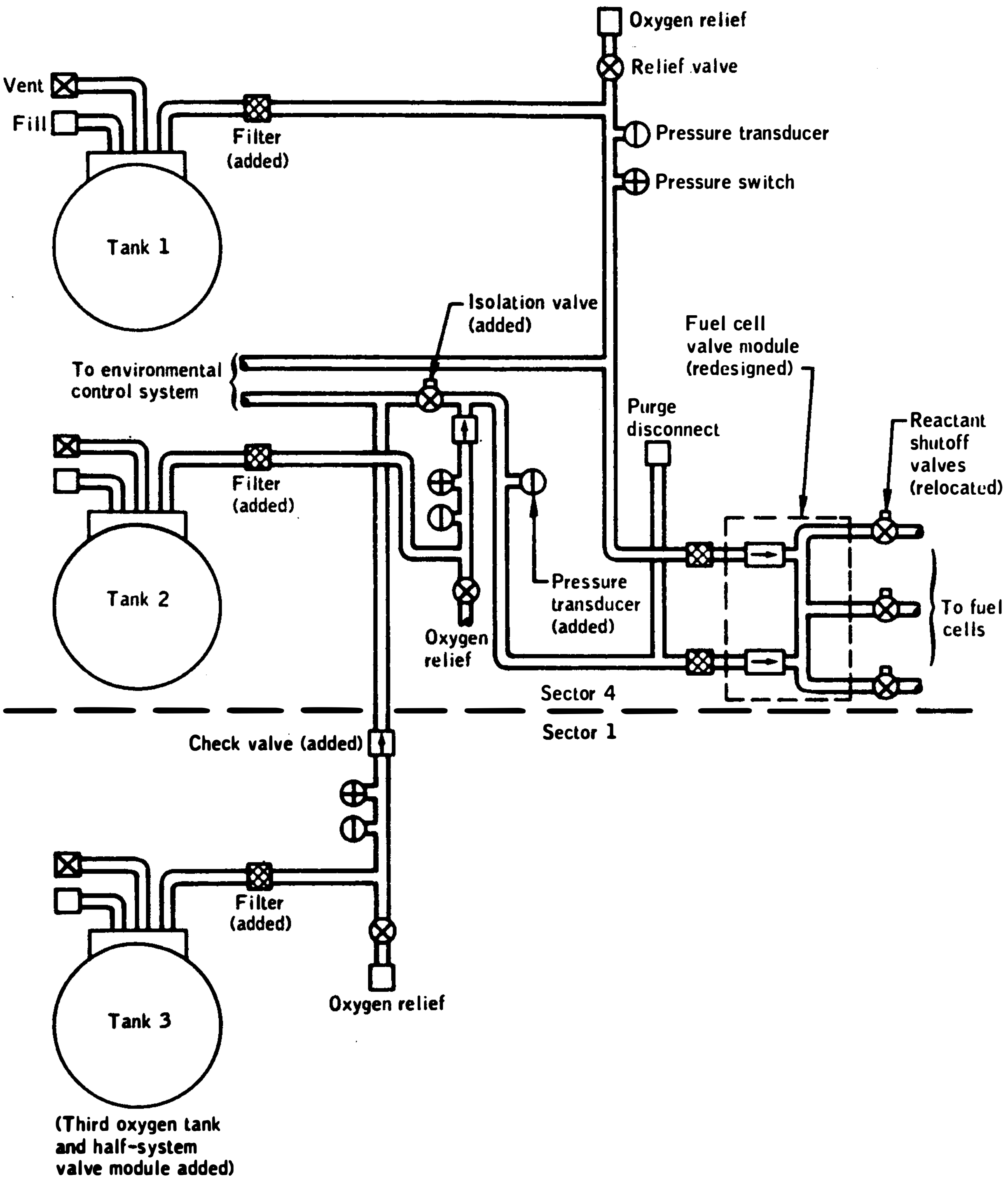

Figure A-1.- Cryogenic oxygen storage system.

A.1.5 Pyrotechnics

Fabrication and quality control procedures of two pyrotechnic devices used in the command and service module tension tie cutter and the command module forward heat shield jettisoning system were improved. Although no known inflight problem with the tension tie cutter has existed, a Skylab qualification test (performed under more severe vacuum and thermal conditions than for Apollo) revealed that it varied in performance. In the forward heat shield jettisoning system, the technique of assembling the breech to the plenum was improved to eliminate the possibility of damage to the O-ring during assembly. On Apollo 13, the propellant gas had leaked from the gusset 4 breech assembly, a hole was burned through the aluminum gusset cover plate, and the pilot parachute mortar cover was damaged. Structural modifications to gussets 3 and 4 are described in section A.1.1.

The docking ring separation system was modified by attaching the separation charge holder to the backup bars with bolts as well as the spring system used previously. This change was made to insure that the charge holder remained secure upon actuation of the pyrotechnic charge at command module/lunar module separation.

A.1.6 Crew Provisions

A contingency water storage system was added to provide drinking water in the event that water could not be obtained from the regular potable water tank. The system included five collapsible 1-gallon containers, fill hose, and dispenser valve. The containers were 6-inch plastic cubes covered with Beta cloth. The bags could also be used to store urine as a backup to the waste management system overboard dump nozzles. (The auxiliary dump nozzle in the side hatch was modified for an oxygen tank flow test and could not be used.)

A side hatch window camera bracket was added to provide the capability to photograph through the hatch window with the 70mm Hasselblad camera.

The intravehicular boot bladder was replaced with the type of bladder used in the extravehicular boot because it has superior wear qualities.

A.1.7 Displays and Controls

The following changes were made which affected crew station displays and controls. The alarm limit for cryogenic hydrogen and oxygen pressure was lowered from 220 psia to approximately 200 psia to eliminate nuisance alarms. The flag indicators on panel 3 for the hydrogen and oxygen reactant valves were changed to indicate closing of either shutoff valve instead of closure of both valves, and valve closure was added to the caution and warning matrix. Oxygen tank 2 and 3 manifold pressure was added to the caution and warning system. Circuitry and controls necessary to control and monitor oxygen tank 3 were added (heaters, pressure, and quantity). Switches were added to panel 278 to connect the auxiliary battery and activate the new isolation valve between oxygen tanks 2 and 3. Circuitry and controls (S19, S20 on panel 2; C/B on panel 226) for the cryogenic fan motors were deleted. The controls for the oxygen tank heaters were changed to permit the use of one, two, or three heater elements at a time depending upon the need for oxygen flow.

A.2 LUNAR MODULE

A.2.1 Structures and Mechanical Systems

Support structure was added to the descent stage for attachment of the laser ranging retro-reflector to the exterior of quadrant 1 and attachment of the lunar portable magnetometer to the exterior of quadrant 2 (see section A.4 for description of experiment equipment). A modular equipment transporter was attached to the modular equipment stowage assembly in quadrant 4. This system (fig. A-2) was provided to transport equipment and lunar samples, and to serve as a mobile workbench during extravehicular activities. The transporter was constructed of tubular aluminum, weighed 25 pounds, and was designed to carry a load of about 140 pounds, including about 30 pounds of lunar samples.

A.2.2 Electrical Power

Because of an anomaly which occurred on Apollo 13 in which the descent batteries experienced current transients and the crew noted a thumping noise and snowflakes venting from quadrant 4 of the lunar module, both the ascent and descent batteries were modified as follows:

a. The total battery container was potted and the potting on top of the battery cells was improved.

b. Manifolding from cell to cell and to the battery case vent was incorporated.

c. The outside and inside surfaces of the battery cover were reversed so that the ribs were on the exterior of the battery.

In addition, the ascent batteries were modified in the following manner:

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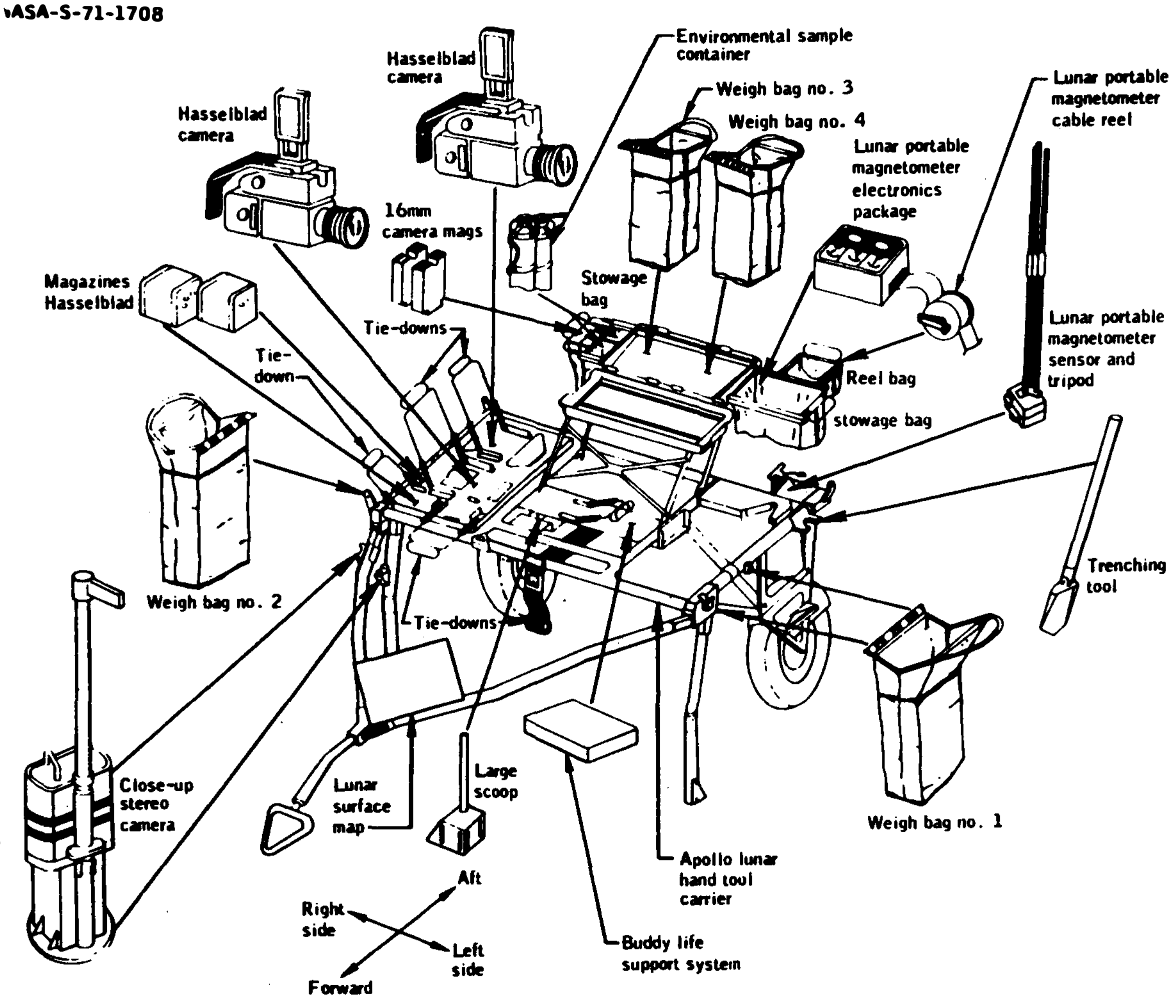

Figure A-2.- Modular equipment transporter and equipment.

a. The negative terminal was relocated to the opposite end of the battery.

b. The case vent valve was relocated to the same face as the positive terminal to allow purging the full length of the battery case.

c. The pigtail, purge port, and the manifold vent valve were relocated to the same face as the negative terminal.

A circuit breaker was added to the lunar module to bypass the command module/lunar module bus connect relay contacts for transferring power between vehicles after lunar ascent and docking. The command module/lunar module bus connect relay control circuit is interrupted at lunar module staging.

A.2.3 Instrumentation

Instrumentation changes in the ascent propulsion system included the installation of a pressure transducer in each of the two helium tanks in place of two tank temperature limit sensors which had been used for measuring structural temperature. The added pressure transducers, in conjunction with the primary pressure transducers already present, provided redundancy in monitoring for leaks. Two temperature measurements were added to the ascent water tank lines to monitor structural temperatures in place of the measurements deleted from the ascent propulsion system helium tanks.

A descent propulsion system fuel ball valve temperature measurement was added for postflight analysis purposes because of concern that damage could result from heat soak-back into propellant lines after powered descent.

A.2.4 Displays and Controls

In the ascent propulsion system, the inputs from the feedline interface pressure sensors to the caution and warning system were disabled. Because of the low pressure at these sensors prior to system pressurization, their inputs to the caution and warning system would have masked the low-pressure warning signal from the helium tanks at critical points in the mission.

Because of erratic indications given by the ascent propulsion system fuel low-level indicator during preflight checkout, the indicator was disabled to prevent master alarms.

The four reaction control system cluster temperature measurement inputs to the caution and warning system were inhibited to prevent nuisance alarms since it was determined that these measurements were no longer needed.

An incorrect indication of the ascent stage gaseous oxygen tank 1 pressure input to the caution and warning system was experienced during preflight checkout. Therefore, the input to the caution and warning system was disabled to prevent meaningless alarms.

A.2.5 Descent Propulsion

Anti-slosh baffles were installed inside the descent stage propellant tanks and the diameter of the outlet holes for the propellant quantity gaging system sensors was reduced from 5/8 inch to 0.2 inch to minimize premature low propellant level indications due to sloshing such as had been experienced on Apollo 11 and 12.

It was determined by test that the descent propulsion system fuel lunar dump valve would close under liquid flow conditions when installed in the normal flow direction and could not be reopened. It was further determined that, by reversing the valve and installing an orifice upstream of the valve, it would remain open under all expected liquid flow conditions. Because of a possible requirement to vent the propellant tanks and the cryogenic helium tank under zero-g conditions, the valve was reinstalled in the reverse flow direction.

The propellant quantity gaging system sensors were modified to include a metal split ring between the electronics package cover and the sensor flanges. This increased the clearance between the electronics package and cover to preclude the possibility of crushed wires due to improper clearance.

A.2.6 Ascent Propulsion

To improve the seal for the four-bolt flanged joint between the fill-and-drain lines and the main feed lines in the ascent propulsion system, O-rings were used in place of injected sealants. Teflon O-rings were used in the oxidizer lines, and butyl rubber O-rings were used in the fuel lines.

A.2.7 Environmental Control

A muffler was added in the line at the outlet of the water-glycol pump assembly to reduce the pump noise transmitted to the cabin through the water-glycol lines. The regulator band of the high-pressure oxygen assembly was shifted to increase the regulated pressure from approximately 950 psig to 990 psig, providing a higher recharge pressure for the portable life support system and, thus, increasing its operating time for extravehicular activities.

A.2.8 Crew Provisions

The flexible-type container assembly previously used for stowage in the left hand side of the lunar module cabin was replaced with a metal modularized container which was packed before being placed into the lunar module.

Return stowage capability was provided for two additional lunar rock sample bags.

A.3 EXTRAVEHICULAR MOBILITY UNIT

The thigh convolute of the pressure garment assembly was reinforced to decrease bladder abrasion which had been noted on training suits. Also, the crotch pulley and cable restraint system was reconfigured to provide for heavier loads.

The portable life support system was modified as follows. A carbon dioxide sensor was added and associated changes were made to provide telemetry of carbon dioxide partial pressure in the pressure garment assembly. In addition, an orifice was added to the feedwater transducer to prevent freezing of water trapped within the transducer housing, which would otherwise result in incorrect readings. The oxygen purge system was modified by the deletion of the oxygen heater system because the oxygen does not require preheating to be compatible with crew requirements.

A new piece of equipment, the buddy secondary life support system, was provided as a means of sharing cooling water from one portable life support system by both crewmen in the event that one cooling system became inoperative. The unit consists of a water umbilical, restraint hooks and tether line, and a water-flow divider assembly.

A.4 EXPERIMENT EQUIPMENT

Table A-I lists the experiment equipment carried on Apollo 14, identifies the stowage locations of the equipment in the lunar module, and references applicable Apollo mission reports if equipment has been described previously. Equipment not carried on previous missions is described in the following paragraphs. The two subpackages of the Apollo lunar surface experiments package are shown in figures A-3 and A-4.

A.4.1 Active Seismic Experiment

The active seismic experiment acquires information to help determine the physical properties of lunar surface and subsurface materials using artificially produced seismic waves.

The experiment equipment consists of three identical geophones, a thumper, a mortar package, a central electronics assembly, and interconnecting cabling. The geophones are electromagnetic devices which were deployed on the lunar surface to translate surface movement into electrical signals. The thumper is a device that was operated by one of

TABLE A-I.- APOLLO 14 EXPERIMENT EQUIPMENT

Experiment equipment	Experiment number	Stowage location in Apollo 14 lunar module	Previous missions on which carried	Mission report reference
Apollo lunar surface experiment package:				
(1) Fuel capsule for radioisotope thermoelectric generator		Stowed in cask assembly mounted on exterior of quadrant 2	Apollo 12 & 13	Apollo 12
(2) Subpackage 1:				
(a) Passive seismic experiment[a]	S-031	Scientific equipment bay - quadrant 2	Apollo 12 & 13	Apollo 12
(b) Active seismic experiment	S-033	Scientific equipment bay - quadrant 2		
(c) Charged particle lunar environment experiment	S-038	Scientific equipment bay - quadrant 2	Apollo 13	Apollo 13
(d) Central station for command control: Lunar dust detector	M-515	Scientific equipment bay - quadrant 2	Apollo 12 & 13	Apollo 12
(3) Subpackage 2:				
(a) Suprathermal ion detector experiment[a]	S-036	Scientific equipment bay - quadrant 2	Apollo 12	Apollo 12
(b) Cold cathode ion gauge	S-058	Scientific equipment bay - quadrant 2	Apollo 12 & 13	Apollo 12
Laser ranging retro-reflector experiment	S-078	Mounted on exterior of quadrant 1	Apollo 11	Apollo 11
Lunar portable magnetometer experiment	S-198	Mounted on exterior of quadrant 2	(b)	
Solar wind composition experiment	S-080	Modular equipment stowage assembly - quadrant 4	Apollo 11 & 12	Apollo 11
Lunar field geology:	S-059			Apollo 14:
(1) Tools and containers		Modular equipment stowage assembly - quadrant 4	Apollo 11, 12 & 13	Fig. A-2
(2) Cameras		Modular equipment stowage assembly and cabin	Apollo 11, 12 & 13	Fig. A-2
(3) Tool carrier		Apollo lunar surface experiment subpackage 2 - quadrant 2	Apollo 12 & 13	Fig. A-4
(4) Modular equipment transporter[c]		Modular equipment stowage assembly - quadrant 4		Fig. A-2
Lunar soil mechanics:	S-200			Apollo 14:
(1) Tools and containers		Modular equipment stowage assembly - quadrant 4	Apollo 11, 12 & 13	Fig. A-2
(2) Cameras		Modular equipment stowage assembly and cabin	Apollo 11, 12 & 13	Fig. A-2
(3) Modular equipment transporter[c]		Modular equipment stowage assembly - quadrant 4		Fig. A-2

[a]Modified from Apollo 12 configuration.

[b]Similar to experiment S-034 on Apollo 12, but different equipment used.

[c]See section A.2.1 for description.

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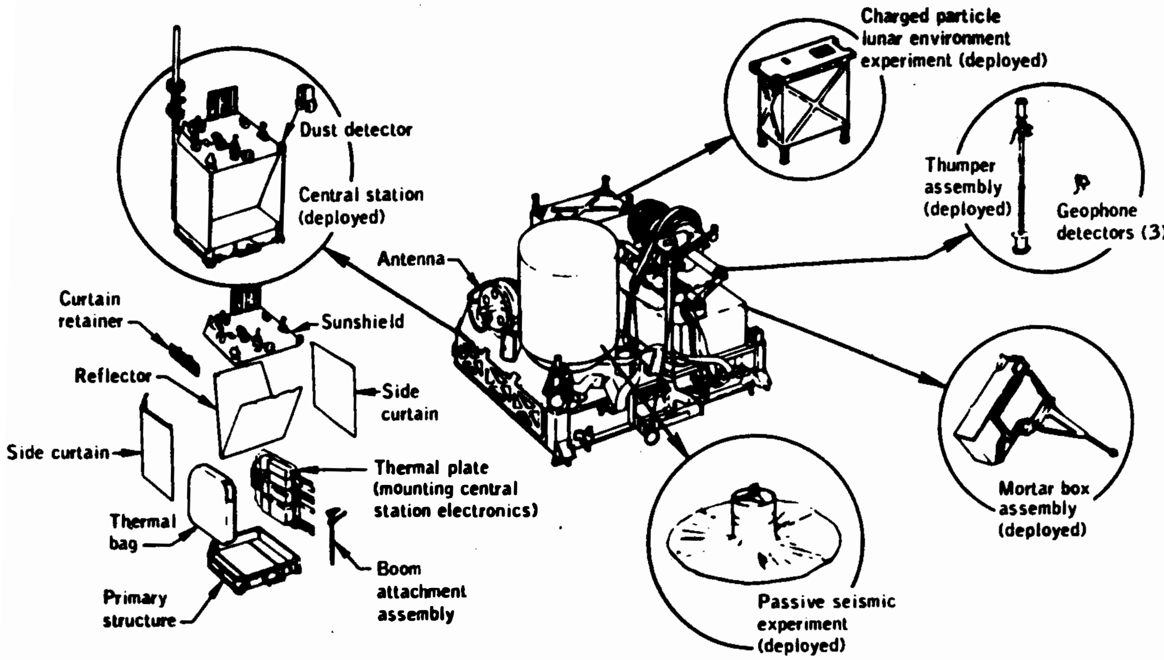

Figure A-3.- Experiment subpackage no. 1.

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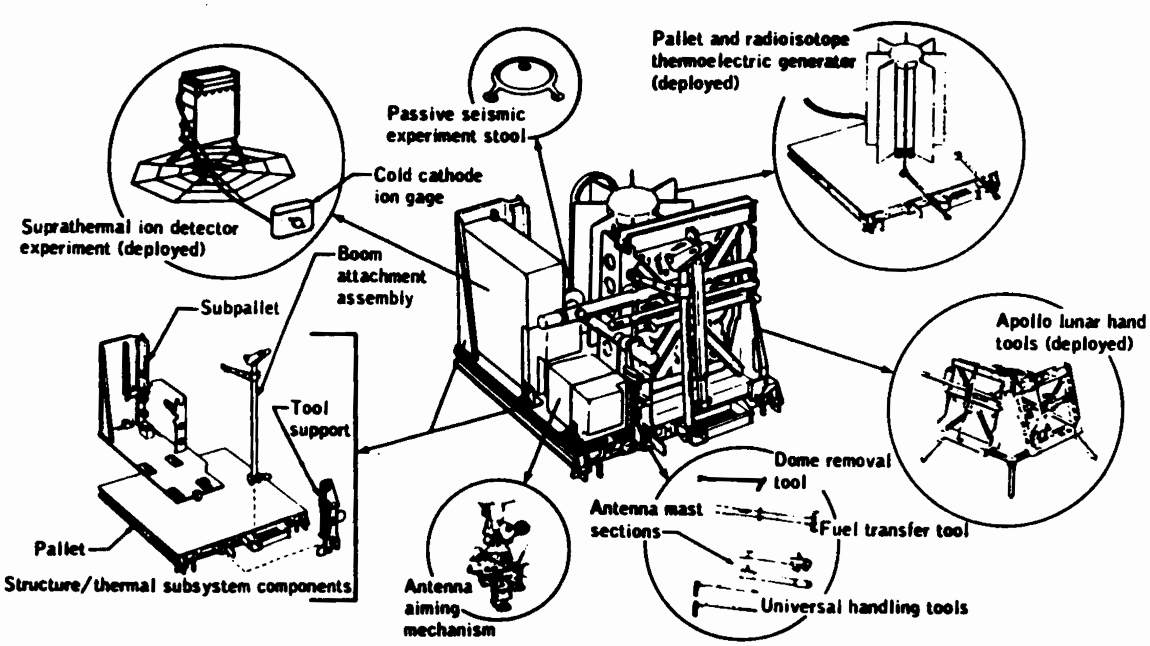

Figure A-4.- Experiment subpackage no. 2.

the crewmen to provide seismic signals. The signals were generated by holding the thumper against the lunar surface at various locations along the line of the geophones and firing explosive initiators located in the base of the thumper. The mortar package consists of a mortar box assembly and a grenade launch tube assembly. The mortar box electronics provide for the arming and firing of rocket motors which will launch four high-explosive grenades from the launch tube assembly upon remote command. The monitor package is designed to launch the grenades to distances of 5000, 3000, 1000, and 500 feet. Signals sensed by the geophones are transmitted to earth-based recorders.

A.4.2 Lunar Portable Magnetometer Experiment

The lunar portable magnetometer was used to measure the magnetic field at two locations along a traverse on the lunar surface. The measurements will be used to determine the location, strength and dimensions of the source, and, in turn, to study both local and whole-moon geological structure.

The experiment equipment consists of a sensor head containing three orthogonal single-axis fluxgate sensor assemblies, an electronics and data display package, and a tripod. The electronics package is powered by mercury cells. The package has an on-off switch and a switch to select high and low meter ranges (±100 gammas and ±50 gammas). The data display consists of three meters, one for each axis.

A.5 MASS PROPERTIES

Spacecraft mass properties for the Apollo 14 mission are summarized in table A-II. These data represent the conditions as determined from postflight analyses of expendable loadings and usage during the flight. Variations in command and service module and lunar module mass properties are determined for each significant mission phase from lift-off through landing. Expendables usage are based on reported real-time and postflight data as presented in other sections of this report. The weights and center-of-gravity of the individual modules (command, service, ascent stage, and descent stage) were measured prior to flight and inertia values calculated. All changes incorporated after the actual weighing were monitored, and the mass properties were updated.

TABLE A-II.- MASS PROPERTIES

Event	Weight, lb	Center of gravity, in.			Moment of inertia, slug-ft²			Product of inertia, slug-ft²		
		X	Y	Z	I_{XX}	I_{YY}	I_{ZZ}	I_{XY}	I_{XZ}	I_{YZ}
Command and service module/lunar module										
Lift-off	111 120.3	847.5	2.2	3.7	68 304	1 183 929	1 186 165	4058	9 610	3622
Earth orbit insertion	102 083.6	807.6	2.4	4.0	67 445	724 926	727 209	5759	11 665	3610
Transposition and docking										
Command & service modules	64 388.0	934.4	4.0	6.4	34 251	77 036	79 537	-1787	-370	3047
Lunar module	33 649.2	1236.7	-.2	-.3	22 533	24 350	24 949	-466	63	233
Total docked	98 037.2	1038.2	2.6	4.1	57 077	537 537	540 506	-8214	-9915	3412
[a]First midcourse correction	97 901.5	1038.3	2.6	4.1	56 969	537 197	540 171	-8232	-9900	3440
[a]Second midcourse correction	97 104.1	1038.9	2.6	4.0	56 547	535 756	539 024	-8223	-9847	3365
[a]Lunar orbit insertion	97 033.1	1039.0	2.6	4.0	56 499	535 582	538 872	-8231	-9834	3364
[a]Descent orbit insertion	71 768.8	1081.9	1.3	2.7	43 395	410 855	417 348	-5576	-5923	397
Separation	70 162.3	1086.4	1.3	2.7	43 872	402 639	408 496	-4684	-6279	290
[a]Command and service module circularization	35 996.3	945.0	2.2	5.8	19 725	57 161	62 490	-1981	547	84
[a]Command and service module plane change	35 610.4	945.2	2.2	5.8	19 494	57 032	62 244	-1963	528	91
Docking										
Command & service modules	34 125.5	946.5	1.9	6.0	18 662	56 594	61 218	-1872	482	69
Ascent stage	5 781.3	1165.2	4.6	-2.3	3 347	2 297	2 723	-117	-3	-352
Total after docking										
Ascent stage manned	39 906.8	978.2	2.3	4.8	22 090	109 973	114 958	-1341	-1444	-307
Ascent stage unmanned	39 903.9	976.3	1.9	4.9	21 910	105 741	110 695	-2009	-1038	-256
After ascent stage jettison	34 596.3	947.5	2.0	5.7	18 744	57 030	61 660	-1772	309	58
[a]Transearth injection	34 554.4	947.3	2.0	5.7	18 730	56 553	61 181	-1746	349	60
[a]Third midcourse correction	24 631.9	975.3	-1.6	7.4	13 592	41 585	41 392	142	-492	-458
Command and service module prior to separation	24 375.0	975.7	-1.6	7.5	13 386	41 344	41 190	138	-491	-399
After separation										
Service module	11 659.9	906.4	-3.1	9.4	7 459	12 908	13 280	-418	533	-359
Command module	12 715.1	1039.2	-.2	5.7	5 897	5 281	4 763	44	-373	-25
Entry	12 703.5	1039.2	-.2	5.6	5 890	5 274	4 762	44	-371	-24
Main parachute deployment	12 130.8	1037.6	-.1	5.8	5 686	4 874	4 403	44	-320	-21
Landing	11 481.2	1035.9	-.1	4.8	5 501	4 457	4 083	35	-297	-8
Lunar module										
Lunar module at earth launch	33 651.9	184.9	-.3	.0	22 538	24 925	25 034	177	434	374
Separation	34 125.9	186.0	-.3	.6	23 939	26 112	26 073	178	722	378
[a]Powered descent initiation	34 067.8	185.9	-.3	.7	23 904	26 018	25 965	175	719	371
Lunar landing	16 371.7	213.6	-.6	1.1	12 750	13 629	16 099	231	652	398
Lunar lift-off	10 779.8	243.9	.2	2.8	6 756	3 408	5 954	68	188	6
Orbit insertion	5 917.8	257.0	.3	5.0	3 417	2 908	2 144	61	104	5
Terminal phase initiation	5 880.1	256.8	.4	5.1	3 400	2 899	2 123	61	105	6
Docking	5 781.3	256.7	.4	5.2	3 347	2 878	2 055	61	105	8
Jettison	5 307.6	258.2	.2	1.7	3 126	2 771	2 056	64	129	3

[a]At ignition

APPENDIX B - SPACECRAFT HISTORIES

The history of command and service module (CSM 110) operations at the manufacturer's facility, Downey, California, is shown in figure B-1, and the operations at Kennedy Space Center, Florida, in figure B-2.

The history of the lunar module (LM-8) at the manufacturer's facility, Bethpage, New York, is shown in figure B-3, and the operations at Kennedy Space Center, Florida, in figure B-4.

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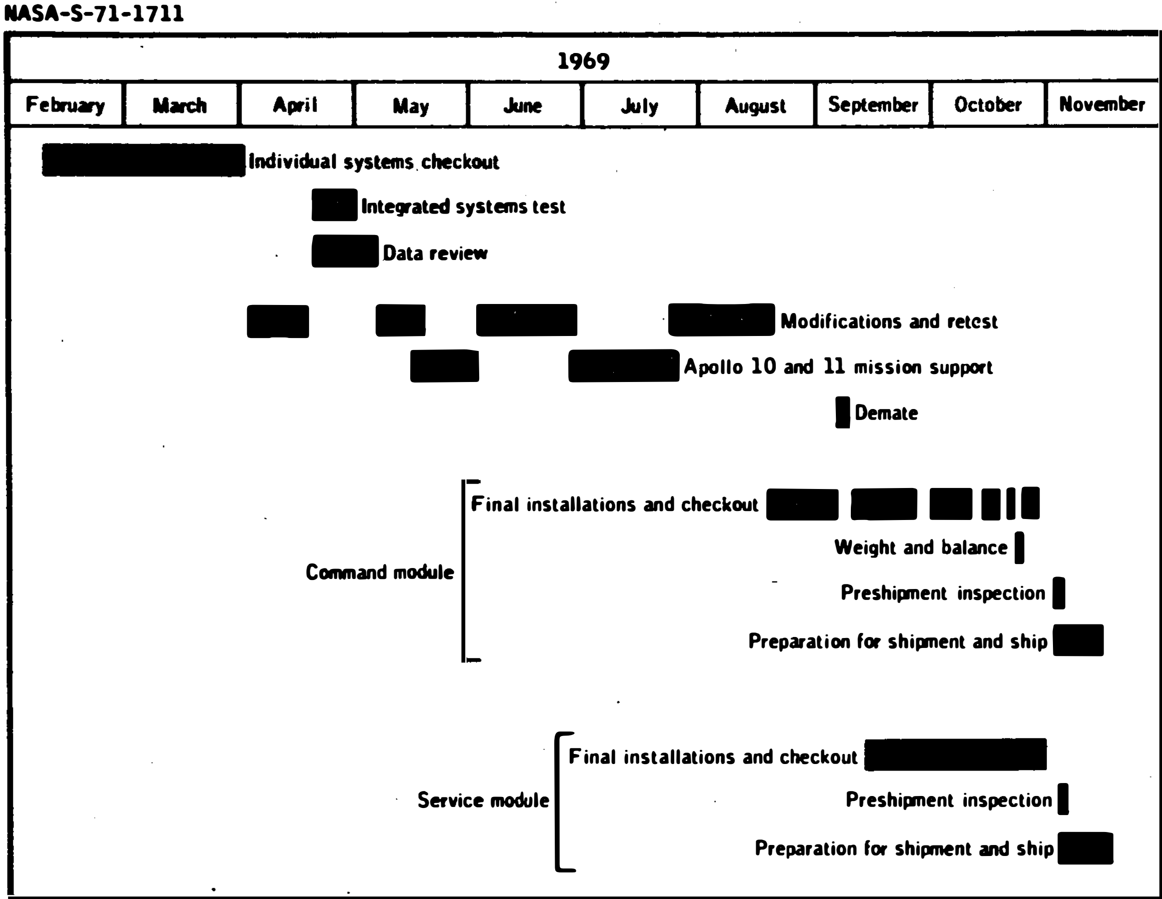

Figure B-1.- Checkout flow for Command and service modules at contractor's facility.

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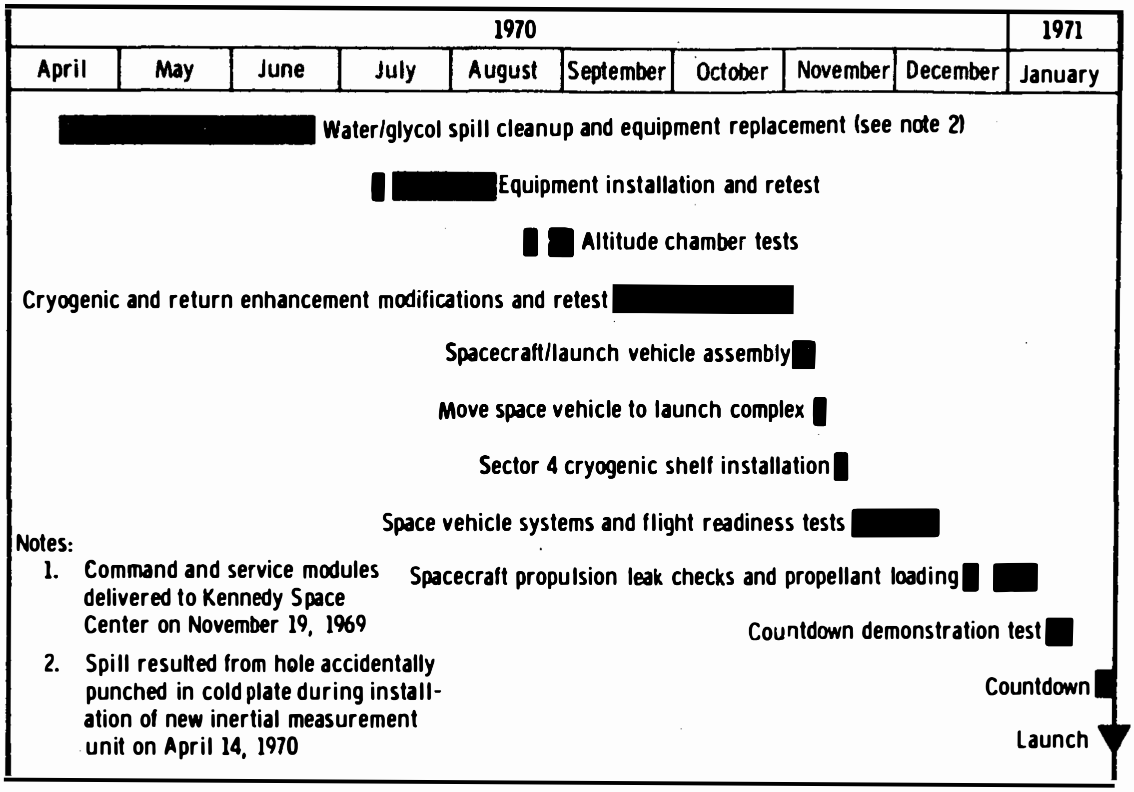

Figure B-2.- Command and service module checkout history at Kennedy Space Center.

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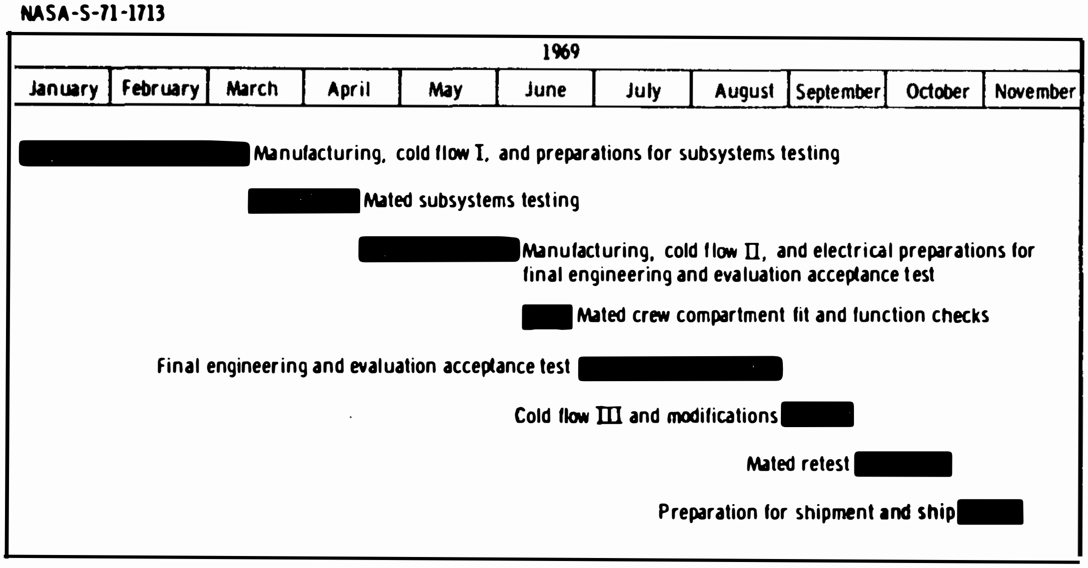

Figure B-3.- Checkout flow for lunar module at contractor's facility.

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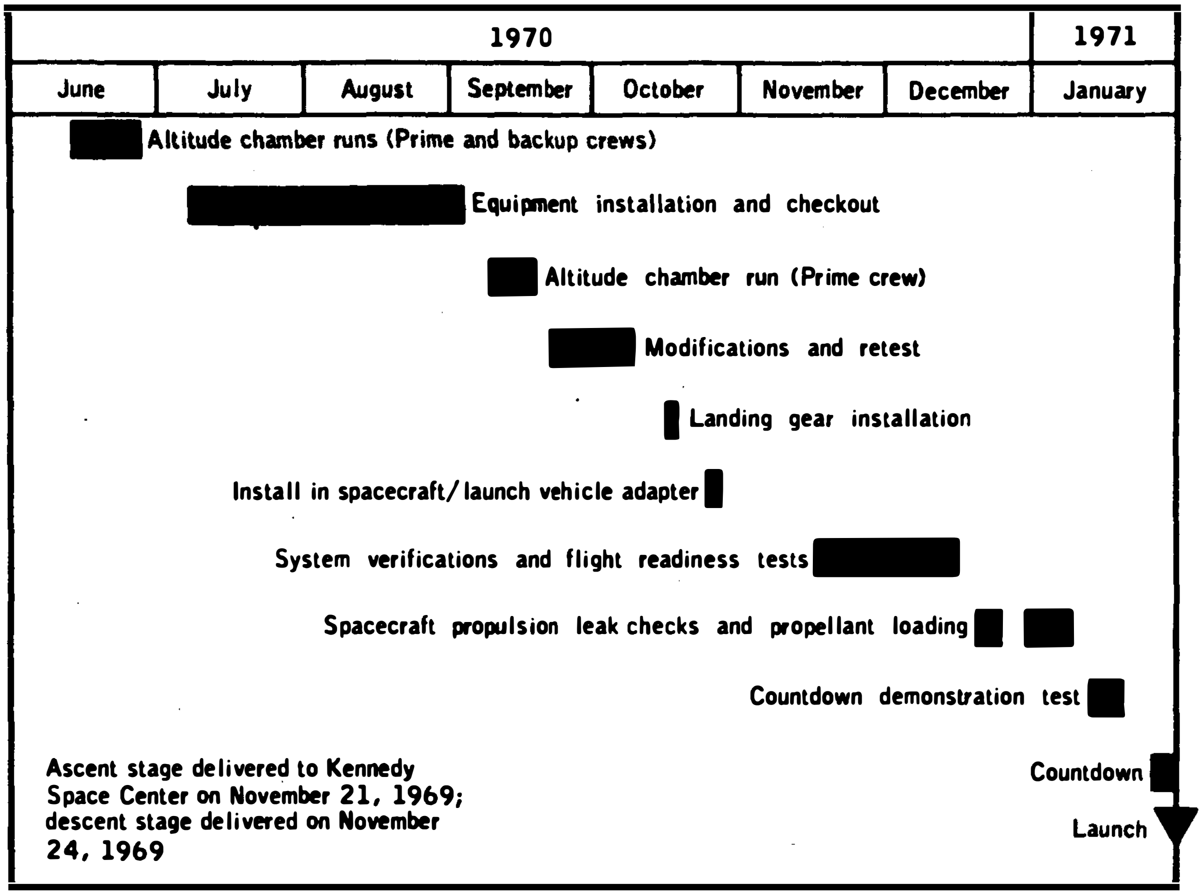

Figure B-4.- Lunar module checkout history at Kennedy Space Center.

APPENDIX C - POSTFLIGHT TESTING

The command module arrived at the Lunar Receiving Laboratory, Houston, Texas, on February 22, 1971, after reaction control system deactivation and pyrotechnic safing in Hawaii. At the end of the quarantine period, the crew equipment was removed and the command module was shipped to the contractor's facility in Downey, California, on April 8. Postflight testing and inspection of the command module for evaluation of the inflight performance and investigation of the flight irregularities were conducted at the contractor's and vendor's facilities and at the Manned Spacecraft Center in accordance with approved Apollo Spacecraft Hardware Utilization Requests (ASHUR's). The tests performed as a result of inflight problems are described in table C-I and discussed in the appropriate systems performance sections of this report. Tests being conducted for other purposes in accordance with other ASHUR's and the basic contract are not included.

TABLE C-I.- POSTFLIGHT TESTING SUMMARY

ASHUR no.	Purpose	Tests performed	Results
		Environmental Control	
110016	To investigate the high oxygen flow rate noted on several occasions.	Perform predelivery acceptance test on the urine receptacle assembly vent valve.	The leakage was slightly higher than allowed, but not significant enough to cause a problem with the valve in the closed position. An open vent valve produces the observed high flow.
110029	To determine the cause of difficulty in inserting water buffer ampules into the injector.	Perform inspection and fit and functional tests.	Insertion of one buffer ampule required excessive torque and a leak developed at a fold in the bag wall. Test not complete.
110030	To determine the cause of slight leakage of the oxygen repressurization package.	Perform leak test and failure analysis.	The leakage rate was within specification.
110040	To investigate the leak at the food preparation water port.	Perform functional and leakage tests.	The hot water port leaked initially in the test, then, no further leakage occurred. Test not complete.
110046	To investigate apparent freezing of the urine dump nozzle.	Perform continuity and resistance tests of the urine nozzle heater circuitry.	The electric circuitry resistance readings were normal.
		Structures	
110005	To determine the cause of the capture latch engagement problem during transposition docking.	Perform inspection, functional tests, and teardown of the docking probe.	Test not complete.
		Guidance and Navigation	
110026	To investigate the apparent failure of the entry monitor system .05g sensing function during entry.	Perform functional tests and failure analysis.	The entry monitor system functioned normally.
		Electrical Power	
110033	To determine the cause of power remaining on the main buses after the main bus switches were positioned off during entry.	Perform continuity and electrical tests to isolate cause.	Motor switch S1 failed. The main bus B-battery C circuit breaker was intermittent in the closed position. Foreign particles were found on the motor switch commutator. A hard deposit was found on a contact of the circuit breaker. Test not complete.
110045	To determine the cause of poor VHF voice communications between the lunar module and the command module.	Perform system test in command module and perform bench tests on VHF hardware.	Readings obtained in spacecraft test were normal. Test not complete.

TABLE C-I.- POSTFLIGHT TESTING SUMMARY - Concluded

ASHUR no.	Purpose	Tests performed	Results
Crew Equipment			
110006 110503	To determine the cause of the lunar topographic camera failure.	Duplicate camera failure and perform failure analysis. Perform functional test of the electrical power cable.	A failed transistor was found in the shutter control circuitry. An aluminum sliver was found in the transistor.
110009	To investigate the cause of the Lunar Module Pilot's personal radiation dosimeter not updating.	Perform response tests on the dosimeter at different dose rates.	The dosimeter was inoperative at the lowest dose rate due to loss of sensitivity. The dosimeter readings were within tolerance at other dose rates.
110010 110051	To investigate operational difficulties experienced with the Lunar Module Pilot's right extravehicular glove.	Inspect gloves for possible wrist cable damage. Perform pressure garment assembly evaluation of suited pressure with Lunar Module Pilot.	No wrist cable damage was found. The problem was duplicated in a test with the Lunar Module Pilot suited. Test not complete.
110017	To investigate the apparent high leak rate of the Lunar Module Pilot's pressure garment assembly.	Perform pressure garment assembly leak rate test.	The leak rate was nominal.
110019	To investigate loosening of the 70-mm camera handle on the lunar surface.	Examine fit of the handle to the camera and bracket.	Test not complete.
110020	To investigate occasional double cycling of the 70-mm camera intervalometer.	Perform functional tests and teardown analysis.	The intervalometer functioned properly, but was incompatible with camera motor characteristics.
110027	To investigate intermittent voice communications from the Commander.	Perform functional tests and failure analysis of constant wear garment electrical harnesses.	The electrical harnesses performed normally.

APPENDIX D - DATA AVAILABILITY

Tables D-I and D-II are summaries of the data made available for systems performance analyses and anomaly investigations. Table D-I lists the data from the command and service modules, and table D-II, the lunar module. For additional information regarding data availability, the status listing of all mission data in the Central Metric Data File, building 12, MSC, should be consulted.

TABLE D-I.- COMMAND AND SERVICE MODULE DATA AVAILABILITY

Time, hr:min From	Time, hr:min To	Range station	Bandpass plots or tabs	Bilevels	Computer words	Oscillo-graph records	Brush records	Special plots or tabs	Special programs
-04:00	00:30	ALDS	X						
00:00	00:10	MILA	X	X	X	X	X	X	
00:02	00:14	BDA	X	X		X		X	
00:48	03:15	MSFN	X	X	X				
01:28	01:44	GDS	X	X					
02:25	02:34	GDS	X	X	X	X		X	
02:49	03:49	GDS	X	X	X	X		X	
03:05	12:00	MSFN							X
03:14	06:21	MSFN	X	X	X				
03:47	04:47	GDS	X	X	X	X	X	X	
04:45	05:45	GDS	X	X	X	X		X	
05:43	06:45	GDS	X	X	X				
06:40	07:41	GDS	X	X	X				
07:18	10:36	MSFN	X	X	X				
07:40	08:39	GDS	X	X				X	
08:37	10:35	GDS	X	X					
10:36	14:35	MSFN	X	X	X				
10:50	13:46	HSK	X	X					
14:51	17:53	MSFN	X	X	X				
15:10	15:14	MAD	X	X					
16:07	16:20	MAD		X					
17:07	19:09	MAD						X	
18:07	22:49	MSFN	X	X	X				
19:08	23:09	MAD						X	
20:07	21:09	MAD				X			
22:49	26:56	MSFN	X	X	X				
23:08	24:09	MAD						X	
23:50	24:50	GDS						X	
27:04	30:59	MSFN	X	X	X				
29:37	30:37	GDS	X	X					
30:00	31:00	MSFN						X	X
30:00	30:37	GDS				X	X		
30:30	31:00	GDS	X	X	X	X	X	X	
31:01	34:51	MSFN	X	X	X				
34:00	35:28	GDS						X	
34:54	38:57	MSFN	X	X	X				
39:00	42:53	MSFN	X	X	X				
42:53	47:00	MSFN	X	X	X				
46:48	48:26	GDS				X			
49:21	51:19	GDS				X			
50:40	54:50	MSFN	X	X	X				
55:01	58:46	MSFN	X	X	X				
58:48	62:54	MSFN	X	X	X				
59:00	61:00	GDS					X		
59:00	61:00	MSFN					X	X	
60:57	61:19	GDS	X		X	X	X	X	
63:00	67:20	MSFN	X	X	X				
64:00	66:00	MSFN						X	
65:49	66:49	MAD		X					
67:28	69:18	MSFN	X	X	X				
67:49	69:49	MAD						X	
69:45	70:54	MSFN	X	X	X				
69:49	71:49	MAD						X	
70:55	75:04	MSFN	X	X	X				
71:49	72:49	MAD						X	
75:10	78:42	MSFN	X	X	X				
76:25	77:25	GDS	X	X					
76:40	77:00	GDS				X	X	X	X
76:57	77:02	GDS	X	X	X	X		X	
78:20	78:42	GDS		X					
79:40	82:51	MSFN	X	X	X				
81:15	82:04	GDS	X	X	X				
81:44	82:04	HSK	X	X	X	X	X	X	
82:02	82:20	HSK	X	X	X				

TABLE D-I.- COMMAND AND SERVICE MODULE DATA AVAILABILITY - Continued

Time, hr:min		Range station	Bandpass plots or tabs	Bilevels	Computer words	Oscillo-graph records	Brush records	Special plots or tabs	Special programs
From	To								
82:14	82:44	GDS	X	X					
82:39	83:43	GDS	X	X					
83:02	87:17	MSFN	X	X	X				
84:23	85:12	GDS		X					
85:10	86:09	HSK	X	X	X				
86:10	90:50	MSFN	X	X	X				
86:10	86:53	HSK	X	X	X		X		
88:25	89:35	MSFN					X	X	X
88:26	89:34	MAD	X	X	X			X	
89:42	90:23	MAD	X	X					
90:00	101:00	MSFN				X	X	X	
90:20	91:28	MAD		X					
91:00	94:59	MSFN	X	X	X				
94:10	95:18	MAD		X					
94:59	98:40	MSFN	X	X	X				
96:01	97:11	GDS		X					
97:55	98:20	GDS		X				X	
98:04	98:12	GDS					X	X	
98:19	99:05	GDS		X					
98:40	102:42	MSFN	X	X	X				
98:52	98:55	GDS					X	X	
99:49	100:59	GDS		X					
99:52	100:04	GDS						X	
102:00	102:54	GDS	X	X			X		
102:42	108:36	MSFN	X	X	X				
103:38	104:25	GDS	X	X	X	X		X	
104:23	104:47	GDS		X					
104:47	105:30	GDS	X	X	X	X		X	
105:31	106:47	GDS		X					
106:44	108:42	MSFN	X	X	X				
107:25	108:43	GDS		X					
108:42	110:42	MSFN	X	X	X				
108:42	109:30	HSK		X					
110:41	114:36	MSFN	X	X	X				
111:20	112:08	MAD			X				
114:54	118:37	MSFN	X	X	X				
116:32	118:32	MAD	X	X	X	X	X	X	
118:31	122:31	MSFN	X	X	X				
119:02	120:32	MAD			X				
120:02	120:32	MAD	X	X					
120:55	122:53	GDS			X				
122:31	126:28	MSFN	X	X	X				
123:15	124:49	GDS			X				
125:15	126:30	GDS			X				
126:28	129:38	MSFN	X	X	X				
127:15	128:25	GDS			X				
129:10	129:40	GDS			X				
129:26	130:40	GDS	X	X	X				
129:42	130:10	GDS		X					
131:00	132:00	MSFN					X	X	X
131:00	131:35	GDS			X				
131:12	135:58	MSFN	X	X	X				
131:33	132:34	GDS		X	X		X		X
133:29	134:24	GDS		X	X			X	
134:22	135:10	HSK			X				
135:08	135:12	HSK	X						
135:09	136:20	HSK			X				
136:19	138:46	MSFN	X	X	X				
136:20	138:14	HSK	X	X					
139:05	143:49	MSFN	X	X	X				
139:05	139:45	MAD			X			X	
141:40	142:18	MAD			X				X
142:10	143:00	MAD	X	X	X				X
142:14	146:05	MSFN	X	X	X				

TABLE D-I.- COMMAND AND SERVICE MODULE DATA AVAILABILITY - Concluded

Time, hr:min		Range station	Bandpass plots or tabs	Bilevels	Computer words	Oscillo-graph records	Brush records	Special plots or tabs	Special programs
From	To								
143:31	144:10	MAD	X	X	X	X		X	
144:12	145:08	GDS	X	X	X	X			
145:13	146:14	MAD	X	X	X				
146:05	150:54	MSFN	X	X	X				
146:56	147:55	GDS			X			X	
148:10	148:50	GDS	X	X	X	X		X	
151:14	154:52	MSFN	X	X	X				
154:56	158:57	MSFN	X	X	X				
159:08	162:56	MSFN	X	X	X				
162:40	164:00	MSFN							X
162:58	166:07	MSFN	X	X	X				
165:17	166:18	MAD	X	X	X	X	X	X	X
166:00	176:00	MSFN					X	X	X
166:18	167:18	MAD	X	X					
166:47	170:53	MSFN	X	X	X				
167:00	168:18	MAD	X	X					
167:23	168:03	MAD							X
168:18	169:19	MAD			X			X	
169:00	169:20	MAD	X	X					
169:17	170:08	MAD	X	X	X			X	
170:57	174:40	MSFN	X	X	X				
171:05	174:04	GDS			X		X		
174:01	175:59	GDS			X				
175:09	178:56	MSFN	X	X	X				
175:58	178:52	GDS			X				
179:05	182:52	MSFN	X	X	X				
179:50	184:00	HSK						X	
183:05	186:52	MSFN	X	X	X				
187:02	188:62	MSFN	X						
187:25	190:54	MSFN	X	X	X				
190:54	194:49	MSFN	X	X	X				
194:49	198:46	MSFN	X	X	X				
199:06	203:02	MSFN	X	X	X				
203:11	206:50	MSFN	X	X	X				
207:06	210:52	MSFN	X	X	X				
210:48	211:48	HSK	X	X					
211:11	214:49	MSFN	X	X	X				
214:17	215:06	CRO	X	X					
215:04	215:46	CRO	X	X	X				
215:08	215:43	MSFN	X	X	X				
215:08	215:44	ARIA					X		
215:31	215:51	HSK					X		
215:37	216:07	DSE	X	X	X	X	X	X	X

TABLE D-II.- LUNAR MODULE DATA AVAILABILITY

Time, hr:min		Range station	Bandpass plots or tabs	Bilevels	Computer words	Oscillo-graph records	Brush records	Special plots or tabs	Special programs
From	To								
-04:00	-02:00	ALDS	X						
61:50	62:15	HSK	X				X		
61:52	62:15	MSFN	X	X					
77:34	78:10	GDS	X				X		
101:45	102:50	GDS	X	X	X	X	X	X	X
101:46	102:42	MSFN	X	X					
102:42	106:44	MSFN	X	X	X				
103:38	104:25	GDS	X	X	X	X	X	X	X
104:14	108:51	MSFN	X	X	X				
104:23	104:47	GDS		X	X		X	X	X
105:31	106:07	GDS		X	X		X		X
106:05	106:47	GDS	X	X	X		X	X	X
106:44	108:42	MSFN	X	X	X				
107:25	107:45	GDS	X	X	X		X	X	X
107:42	108:43	GDS	X	X	X	X	X	X	X
108:42	110:15	MSFN	X	X					
108:43	109:00	GDS					X		
109:40	110:36	HSK	X	X	X			X	
110:34	111:34	HSK			X				
112:20	114:32	MSFN	X	X					
112:25	113:10	HSK						X	
113:02	115:03	MAD					X		
114:32	119:03	MSFN	X	X					
115:02	119:20	MAD					X		
119:21	122:45	MSFN	X	X					
120:15	122:53	GDS					X		
122:31	126:28	MSFN	X	X					
122:51	126:45	GDS					X		
126:28	129:38	MSFN	X	X	X				
126:43	129:40	GDS					X		
128:39	129:40	GDS	X	X	X			X	
129:24	129:36	GDS					X		
129:37	130:38	GDS			X		X		
130:35	131:35	GDS	X	X			X		
131:12	135:58	MSFN	X	X	X				
132:31	133:34	GDS	X	X			X		
133:29	135:17	GDS					X		
135:11	137:10	HSK	X	X			X		
136:19	138:46	MSFN	X	X	X				
137:08	138:07	HSK	X	X			X		
137:49	138:50	MAD					X	X	
138:50	139:50	MAD			X		X		
139:05	143:49	MSFN	X	X	X				
139:39	141:50	MAD					X		
140:39	140:50	MAD						X	
140:49	141:50	MAD	X	X	X				X
141:10	141:48	MAD				X	X	X	
141:45	141:50	MAD				X	X	X	
141:49	142:18	MAD	X	X	X	X	X	X	X
142:14	146:05	MSFN	X	X	X				
142:59	143:32	MAD	X	X	X	X	X	X	X
143:21	144:16	MAD	X	X	X	X	X	X	X
143:40	144:01	MAD					X	X	
144:58	145:15	MAD	X	X				X	
145:05	145:15	MAD					X		
145:12	146:14	MAD	X	X	X	X	X	X	
146:04	147:50	MSFN	X	X	X				
146:55	147:30	GDS	X	X	X	X	X	X	X
147:12	147:42	GDS	X	X	X	X	X	X	

APPENDIX E - MISSION REPORT SUPPLEMENTS

Table E-I contains a listing of all reports that supplement the Apollo 7 through Apollo 14 mission reports. The table indicates the present status of each report not yet completed and the publication date of those which have been published.

TABLE E-I.- MISSION REPORT SUPPLEMENTS

Supplement number	Title	Publication date/status
	Apollo 7	
1	Trajectory Reconstruction and Analysis	May 1969
2	Communication System Performance	June 1969
3	Guidance, Navigation, and Control System Performance Analysis	November 1969
4	Reaction Control System Performance	August 1969
5	Cancelled	
6	Entry Postflight Analysis	December 1969
	Apollo 8	
1	Trajectory Reconstruction and Analysis	December 1969
2	Guidance, Navigation, and Control System Performance Analysis	November 1969
3	Performance of Command and Service Module Reaction Control System	March 1970
4	Service Propulsion System Final Flight Evaluation	September 1970
5	Cancelled	
6	Analysis of Apollo 8 Photography and Visual Observations	December 1969
7	Entry Postflight Analysis	December 1969
	Apollo 9	
1	Trajectory Reconstruction and Analysis	November 1969
2	Command and Service Module Guidance, Navigation, and Control System Performance	November 1969
3	Lunar Module Abort Guidance System Performance Analysis	November 1969
4	Performance of Command and Service Module Reaction Control System	April 1970
5	Service Propulsion System Final Flight Evaluation	December 1969
6	Performance of Lunar Module Reaction Control System	August 1970
7	Ascent Propulsion System Final Flight Evaluation	December 1969
8	Descent Propulsion System Final Flight Evaluation	September 1970
9	Cancelled	
10	Stroking Test Analysis	December 1969
11	Communications System Performance	December 1969
12	Entry Postflight Analysis	December 1969

TABLE E-I.- MISSION REPORT SUPPLEMENTS - Continued

Supplement number	Title	Publication date/status
Apollo 10		
1	Trajectory Reconstruction and Analysis	March 1970
2	Guidance, Navigation, and Control System Performance Analysis	December 1969
3	Performance of Command and Service Module Reaction Control System	August 1970
4	Service Propulsion System Final Flight Evaluation	September 1970
5	Performance of Lunar Module Reaction Control System	August 1970
6	Ascent Propulsion System Final Flight Evaluation	January 1970
7	Descent Propulsion System Final Flight Evaluation	January 1970
8	Cancelled	
9	Analysis of Apollo 10 Photography and Visual Observations	In publication as SP-232
10	Entry Postflight Analysis	December 1969
11	Communications System Performance	December 1969
Apollo 11		
1	Trajectory Reconstruction and Analysis	May 1970
2	Guidance, Navigation, and Control System Performance Analysis	September 1970
3	Performance of Command and Service Module Reaction Control System	Review
4	Service Propulsion System Final Flight Evaluation	October 1970
5	Performance of Lunar Module Reaction Control System	Review
6	Ascent Propulsion System Final Flight Evaluation	September 1970
7	Descent Propulsion System Final Flight Evaluation	September 1970
8	Cancelled	
9	Apollo 11 Preliminary Science Report	December 1969
10	Communications System Performance	January 1970
11	Entry Postflight Analysis	April 1970

TABLE E-I.- MISSION REPORT SUPPLEMENTS - Concluded

Supplement number	Title	Publication date/status
Apollo 12		
1	Trajectory Reconstruction and Analysis	September 1970
2	Guidance, Navigation, and Control System Performance Analysis	September 1970
3	Service Propulsion System Final Flight Evaluation	Preparation
4	Ascent Propulsion System Final Flight Evaluation	Preparation
5	Descent Propulsion System Final Flight Evaluation	Preparation
6	Apollo 12 Preliminary Science Report	July 1970
7	Landing Site Selection Processes	Final review
Apollo 13		
1	Guidance, Navigation, and Control System Performance Analysis	September 1970
2	Descent Propulsion System Final Flight Evaluation	October 1970
3	Entry Postflight Analysis	Cancelled
Apollo 14		
1	Guidance, Navigation, and Control System Performance Analysis	Preparation
2	Cryogenic Storage System Performance Analysis	Preparation
3	Service Propulsion System Final Flight Evaluation	Preparation
4	Ascent Propulsion System Final Flight Evaluation	Preparation
5	Descent Propulsion System Final Flight Evaluation	Preparation
6	Apollo 14 Preliminary Science Report	Preparation
7	Analysis of Inflight Demonstrations	Preparation
8	Atmospheric Electricity Experiments on Apollo 13 and 14 Launches	Preparation

APPENDIX F - GLOSSARY

albedo	percentage of light reflected from a surface based upon the amount incident upon it
Brewster angle	the angle at which electromagnetic radiation is incident upon a nonmetallic surface for the reflected radiation to acquire maximum plane polarization
ejecta	material thrown out of a crater formed by impact or volcanic action
electrophoresis	movement of suspended particles in a fluid by electromotive force
foliation	Platy or leaf-like laminae of a rock
galactic light	total light emitted by stars in a given area of the sky
gegenschein	a faint glow seen from the earth along the sun-earth axis in the anti-solar direction
lunar libration region (L_4)	an area 60 degrees from the earth-moon axis in the direction of the moon's travel and on its orbital path
Moulton point	the earth's libration point (L_1) located on the sun-earth axis in the anti-solar direction
nadir	the point on the celestial sphere that is vertically downward from the observer
regolith	the surface layer of unsorted fragmented material that overlies consolidated bedrock
zero phase	the condition whereby the vector from a radiation source (sun) and the observer are colinear
zodiacal light	a faint wedge of light seen from the earth in the anti-solar direction extending upward from the horizon along the ecliptic. It is seen from tropical latitudes for a few hours after sunset or before sunrise.

REFERENCES

1. Manned Spacecraft Center: Apollo 12 Mission Report. MSC-01855. March 1970.

2. Manned Spacecraft Center: Apollo 12 Preliminary Science Report. NASA SP-235. July 1970.

3. Manned Spacecraft Center: Apollo 13 Mission Report. MSC-02680. September 1970.

4. Manned Spacecraft Center: Apollo 11 Preliminary Science Report. NASA SP-214. December 1969.

5. Marshall Space Flight Center: Saturn V Launch Vehicle Flight Evaluation Report AS-509 Apollo 14 Mission. MPR-SAT-FE-71-1. April 1971.

6. Manned Spacecraft Center: Apollo 10 and 11 Anomaly Report No. 1 - Fuel Cell Condenser Exit Temperature Oscillations. MSC-02426. April 1970.

7. NASA Headquarters: Apollo Flight Mission Assignments. OMSF M-D MA 500-11 (SE 010-000-1) October 1969.

8. Manned Spacecraft Center: Mission Requirements, H-1 Type Mission (Lunar Landing). SPD9-R-056. June 9, 1970.

9. Goddard Space Flight Center: Post Mission Analysis Report. S-832-71-175.

10. Manned Spacecraft Center: Radiometric Temperature Measurement of Apollo 14/Saturn V Exhaust. Lockheed Electronics Company (IG2061). Contract NAS9-10950. April 1971.

R-2

MISSION REPORT QUESTIONNAIRE

Mission Reports are prepared as an overall summary of specific Apollo flight results, with supplemental reports and separate anomaly reports providing the engineering detail in selected areas. Would you kindly complete this one-page questionnaire so that our evaluation and reporting service to our readership might be improved.

1. DO YOU THINK THE CONTENT OF THE MISSION REPORTS SHOULD BE:
☐ LESS DETAILED ☐ MORE DETAILED ☐ ABOUT THE SAME?

2. WOULD YOU SUGGEST ANY CHANGES TO THE PRESENT CONTENT?

3. YOUR COPY IS *(check more than one)*:
☐ READ COMPLETELY ☐ READ PARTIALLY ☐ SCANNED ☐ NOT READ OR SCANNED
☐ ROUTED TO OTHERS ☐ FILED FOR REFERENCE ☐ DISCARDED ☐ GIVEN TO SOMEONE ELSE

4. ON THE AVERAGE, HOW OFTEN DO YOU REFER LATER TO A MISSION REPORT?
☐ MORE THAN 5 TIMES ☐ FROM 2 TO 5 TIMES ☐ ONCE ☐ NEVER

5. REGARDING REPORT SUPPLEMENTS, YOU:
☐ USE THOSE YOU RECEIVE ☐ DO NOT RECEIVE ANY, BUT WOULD LIKE TO ☐ DO NOT NEED THEM

6. DO YOU WISH TO CONTINUE RECEIVING MISSION REPORTS?
☐ YES ☐ NO

7. FURTHER SUGGESTIONS OR COMMENTS:

NAME	ORGANIZATION	ADDRESS

Please fold this form in half with the address on the outside, staple, and mail the form to me. Thank you for taking the time to complete this form.

Donald D. Arabian, Chief
Test Division

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