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LUNAR ORBITER I

EXTENDED-MISSION OPERATIONS

Prepared by
THE BOEING COMPANY
Seattle, Wash.
for Langley Research Center

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TABLE OF CONTENTS

*The section number 4 signifies only that this report is the fourth in a series of numbered volumes submitted by the contractor on the Lunar Orbiter I Project. Publication of the complete series by NASA is not necessarily contemplated.

ILLUSTRATIONS

TABLES

LUNAR ORBITER I

EXTENDED MISSION OPERATIONS

4.1 INTRODUCTION

The extended mission for Lunar Orbiter I began on September 17, 1966. The flight plan was to track the spacecraft for two orbits per day for the first 30 days and then for two orbits every third day until the end of the extended mission unless undue conflicts arise with the following photographic missions. The extended mission for Lunar Orbiter I would continue for up to 1 year from the August 10, 1966, launch. However, the flight plan was changed on October 29 when Lunar Orbiter I was maneuvered to impact on the farside of the Moon to ensure noninterference with Lunar Orbiter II.

The spacecraft was tracked as planned until Earth occultation just prior to lunar impact which occurred during the occultation period. Tracking and environmental data was received as specified in the flight plan. In addition, experiments were conducted to further determine the characteristics of the Moon and the spacecraft subsystems.

4.2 SUMMARY

On September 14, 1966, Lunar Orbiter I completed its primary mission with the readout of the final photographic frame. A brief period of experimental effort was expended conducting experiments in addition to star and antenna mapping (Refer to Section 4.6).

The spacecraft met or exceeded specifications for all subsystems except those noted during the photographic mission. No additional anomalies were discovered during the extended-mission period. The thermal control anomaly was the only one that had a significant impact on extended-mission operations. Thermal control paint continued to degrade, allowing spacecraft temperatures to become excessive unless the spacecraft attitude was carefully controlled.

Periodic excessive temperatures were expected to reduce somewhat the spacecraft subsystem life. However, such was not the case. In fact, the battery degradation did not become significant until at least 40 orbits later than was predicted. Once degradation started, the rate was greater than expected.

The thermal control procedure employed during both the photographic mission and the extended mission was to maintain an off-sunline attitude large enough to maintain low temperatures and also small enough to supply adequate solar power. To minimize the number of attitude maneuvers required and conserve attitude control gas, the spacecraft remained in the inertial hold mode for extended periods and allowed to drift in accord with established gyro drift characteristics. Periodic updating of the attitude control reference was required to prevent drifting outside safe limits (refer to Section 4.3). The safe limits were adjusted several times during the extended mission within the region of 25 to 55 degrees off the sunline.

Initially, it was expected that the spacecraft would orbit the Moon for up to 1 year. However, the extended-mission phase was terminated on October 29, 42 days after it started, when the velocity control system was activated to impart a velocity change that placed the spacecraft on a farside lunar-impact trajectory. It is estimated that the impact time was at 13:29 GMT, shortly after Earthset. Just prior to impact, the spacecraft passed through a penumbra eclipse. Although power levels were reduced, the power system restored itself after coming out of the eclipse without any apparent degradation. This mission termination, prior to the launch of Lunar Orbiter II, eliminated any telemetry or tracking anomalies that might occur when two spacecraft operate on the same transponder frequencies.

4.3 SPACECRAFT OPERATION

The initial operating plan called for the spacecraft to operate on the Sunline and to drift about the roll axis. However, the thermal paint degraded more rapidly than expected, and it was therefore necessary to fly the spacecraft in an off-Sunline attitude to maintain thermal control. An optimum off-Sunline attitude was selected to reduce spacecraft temperatures and remain within the constraints of the power subsystem. Spacecraft control for this off-Sunline attitude was accomplished in accordance with the following stored program command sequence:

1) Execute Acquire Sun;
2) Wait 81 minutes;
3) Execute Pitch Plus 25 degrees;
4) Wait 25.5 hours

This command sequence allows for slewing the spacecraft to the Sunline for establishing a zero reference in the pitch and yaw axes. The 81-minute interval ensures that whenever Sun occultation occurs, there is sufficient time for the spacecraft to acquire the Sunline before again being pitched off 25 degrees. A maximum Sun occultation period is approximately 52 minutes; thus, a minimum Sun acquisition period of 29 minutes remains, in the 81-minute wait period, which is available when Sun occultation occurs immediately before the spacecraft is to acquire the Sunline. The spacecraft, after acquiring the Sunline, is pitched plus 25 degrees off the Sunline. During the 25.5-hour Wait time period, the spacecraft drifts while it is being flown in the Inertial Hold mode using a ±2.0 dead zone. The spacecraft will drift about the pitch and yaw axis to a position of approximately 42 degrees off the Sunline. The magnitude of drift in roll is approximately 12 degrees per day.

The above command sequence was to be repeated after completion of each 25.5-hour Wait time period. However, to reduce temperature extremes and still maintain an adequate output from the power subsystem, the magnitude of the Pitch Plus command was changed to 30 degrees. Also, the stored program sequence was modified to obtain a more precise control of events by initiating each command sequence by Compare Time rather than Wait Time commands. These changes were implemented on October 1.

The above modified command sequence was used until October 12. Then additional spacecraft cooling was provided by reducing the time that the spacecraft remained on the Sunline. Thus, the 81-minute Wait Time was reduced to 20.48 minutes and the Compare Time values were selected to ensure that the Acquire Sun command was executed when the spacecraft was in the Sunlight.

On October 15 it was apparent that additional spacecraft cooling was required. This cooling was provided on October 16 by pitching the spacecraft to plus 40 degrees off-Sunline. Also, the stored command sequence was modified to reflect a pitch plus 40-degree angle, and the Wait time period following the Acquire Sun command was extended to allow sufficient time for the spacecraft to reorient to the Sunline.

While conducting tests between October 17 and 20, the power subsystem load current became greater than normal. This unexplained increased load could not be sustained and so the spacecraft was pitched from plus 40 to 30 degrees off the Sunline to obtain more power from the solar array panels. The reason and corrective measures for this problem are explained in Section 4.7.2 of this document. After correcting this problem, the load current dropped to a normal value. It was then possible on October 24 to again increase the pitch angle off the Sunline to 40 degrees.

4.4 SPACECRAFT PERFORMANCE

All spacecraft subsystems at the start of this extended mission were operating within design tolerances. The battery was discharging to 23.8 volts just prior to Sunrise, and there was 2.73 pounds of gas in the nitrogen tank. At the end of this extended mission, the spacecraft subsystems were still operating within design tolerances. The battery was discharging to 21.6 volts just prior to Sunrise and there was 0.80 pound of gas remaining in the nitrogen tank. Spacecraft performance was generally monitored just before Sunset, and Sunrise during each tracking orbit, and graphs were prepared to show spacecraft subsystem trends. These graphs are shown in Figures 4-1 through 4-9.

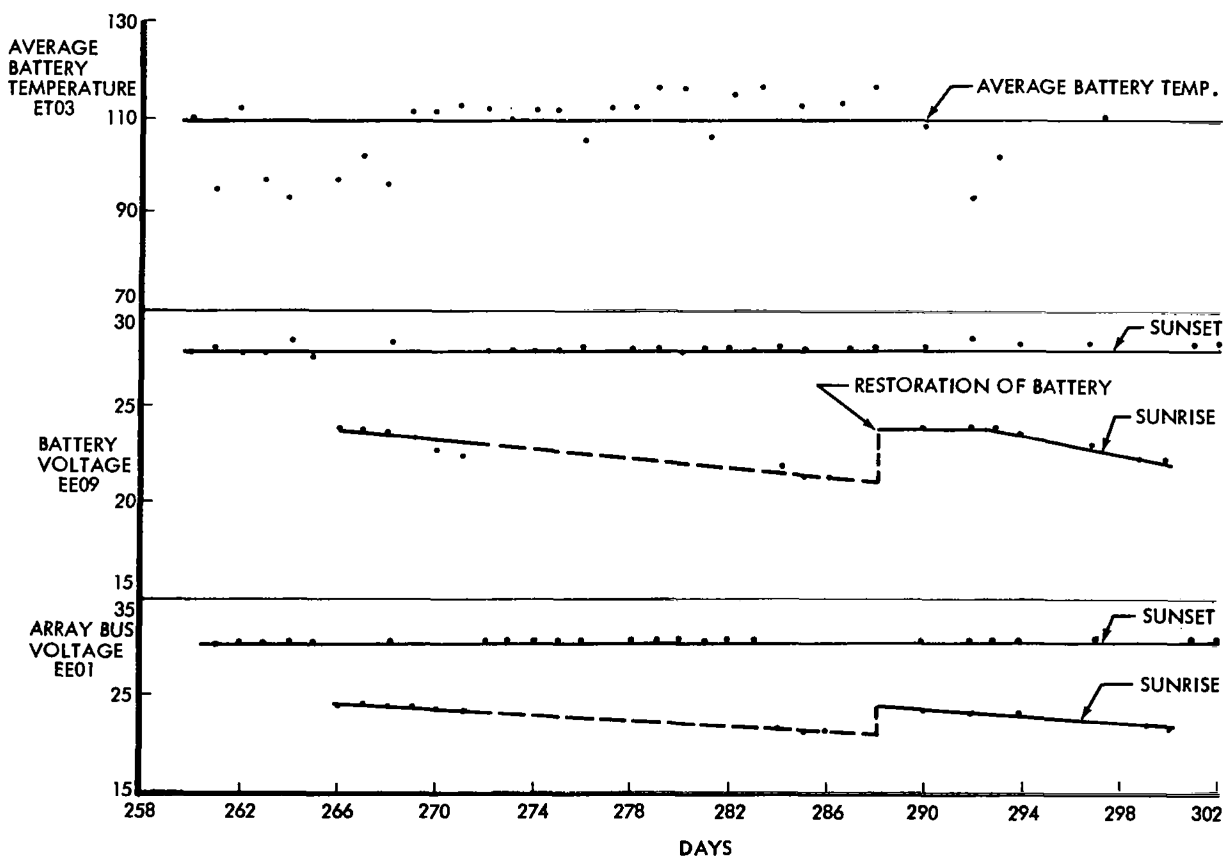


Figure 4-1: Battery Characteristics

4.4.1 POWER SUBSYSTEM

The power subsystem performance was monitored very closely during each extended-mission tracking orbit. Figure 4-1 shows no solar panel degradation in power output, and a constant 30.56-volt output while the spacecraft was in the Sunlight. The power system's load characteristics may been seen in Figure 4-5. The battery module experienced an average temperature of 109°F with monitored highs of 129°F, which is 4°F above the acceptable upper limit. Battery degradation was apparent when the battery discharge voltage, prior to Sunrise, slowly decayed from an orbit low of 23.8 volts on September 17 to a low of 21.6 volts on October 13.

On or about September 30 it became obvious that the battery end of discharge voltage was falling rapidly and that within a short time the battery voltage would fall below 20 volts, or 1.0 volt per cell. On the advice of Dr. Harvey Seiger of Gulton Industries, the manufacturer of the battery cells, and on the basis of results obtained by RCA during multicycle tests on the Lunar Orbiter battery, it was decided to discharge the battery below its normal level during one orbit. This was accomplished by pitching the spacecraft approximately 98 degrees off-Sun prior to Sunrise of final Orbit 383 October 15. The average battery discharge current at night was 5.0 amps, and with an orbital nighttime period of 48 minutes, the normal discharge capacity was 4.0 amp-hours. By remaining off-Sun for 18 minutes after Sunrise it was hoped to discharge the battery an additional 1.2 amp-hours, providing the battery voltage did not fall below 20.0 volts. Figure 4-2 shows the battery voltage versus time.

When the battery voltage reached 19.7 volts, 12 minutes after Sunrise, the spacecraft was commanded to pitch to +36 degrees so that the array was again illuminated; however, before there was sufficient array current to recharge the battery, the battery voltage had fallen to 19.52 volts and telemetry data was terminated by Earthset. Data received at Earthrise indicated that everything was operating normally and a decision was made to change the stored pitch maneuver from 30 to 40 degrees immediately to reduce spacecraft, and battery, temperatures. An attempt to overstore this new pitch command resulted in a real-time pitch maneuver being commanded 23 times at 1-minute intervals, due to an undetected bit error in the command message. This series of incorrect commands caused the spacecraft to lose solar power, resulting in excessive power drain from the battery. By the time this was recognized, and an emergency command was sent to the spacecraft to acquire the Sun, the battery voltage had dropped to 18.0 volts.

Before the Sun was acquired in response to this command, the battery voltage dropped to 15.2 volts and the spacecraft transponder ceased transmitting. The spacecraft remained silent until about 8 minutes after the next Sunrise.

The total depth of battery discharge is unknown because insufficient power was available for a period of 70 minutes to operate the spacecraft transponder and telemetry data was not dependable for another 10 to 15 minutes. Before the Sun was reacquired and battery charging resumed, the battery discharge was estimated to be 7.3 amp-hours. During this period the flight programmer was affected and the programmer memory had to be restored.

As a result of this excessive discharge and loss of charging time, the battery did not fully recharge during Orbit 384 and in the next discharge period the battery voltage fell to 15.8 volts, resulting in a bus voltage of 15.2 volts,

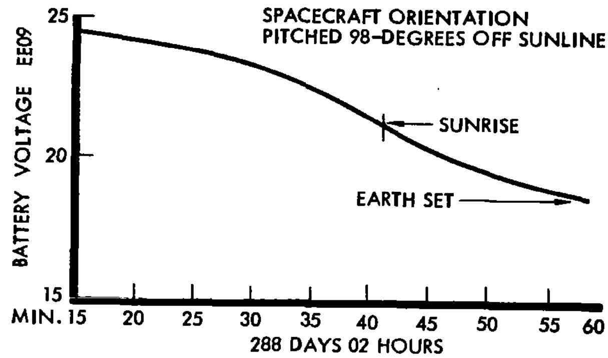


Figure 4-2: Battery Discharge Voltage, Orbit 382

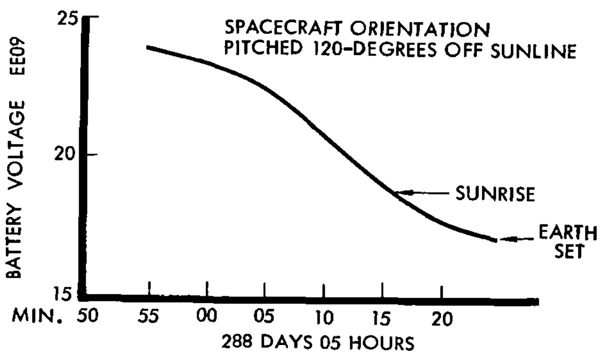


Figure 4-3: Battery Discharge Voltage, Orbit 383

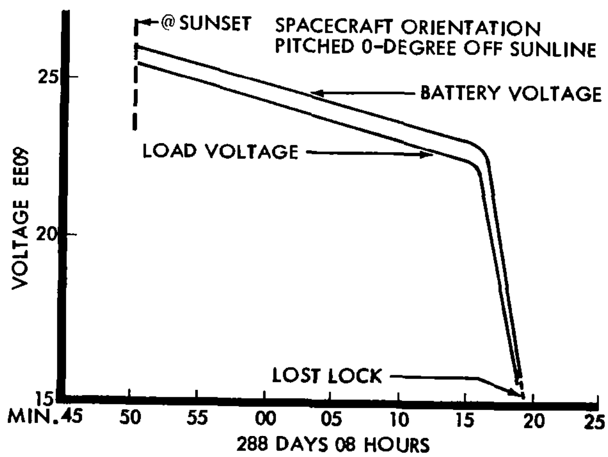


Figure 4-4: Battery Discharge Voltage, Orbit 384

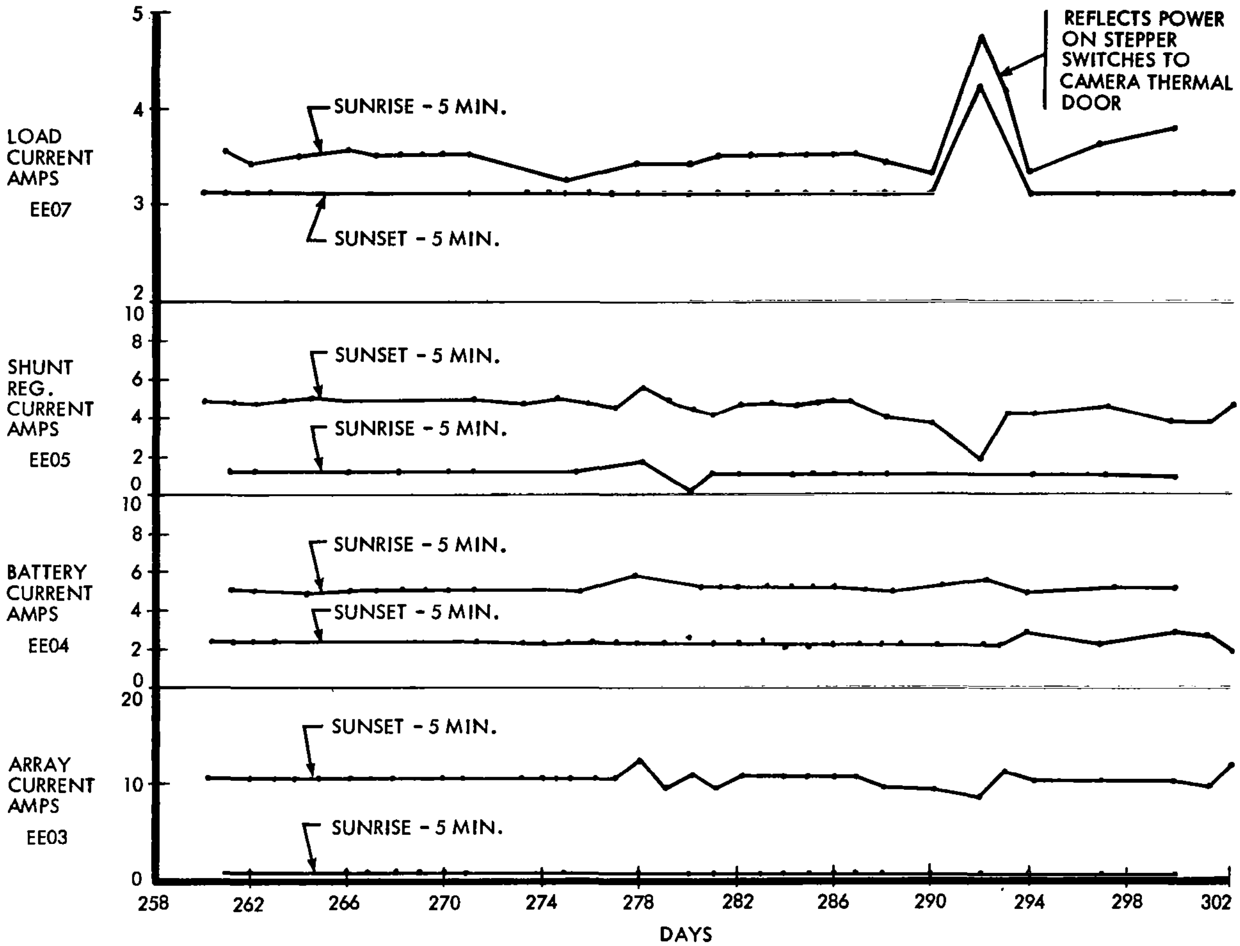


Figure 4-5: Power System Characteristics

at which time, 09:19 GMT, lock was lost (refer to Figure 4-4). No further data was available until 18:22 GMT when it was found that the bus voltage was 30.56 volts with the battery at 27.8 volts and being charged at 2.28 amps, tapered charge. The subsequent battery end of discharge voltage was 23.8 volts, as shown in Figure 4-1, indicating an improvement in battery performance.

Although the first intentional deep discharge was probably insufficient to appreciably affect battery performance, the subsequent unintentional excessive discharge apparently improved the battery discharge characteristics.

On October 29, the last day of this extended mission, the battery discharge voltage again dropped to an orbit low of 21.6 volts. This indicated that the batteries were rapidly losing their ability to hold a charge. At this time it was observed from the data that the batteries had cycled 547 times, or once per orbit, from Mission I's initial injection into lunar orbit until impact of the spacecraft.

4.4.2 ATTITUDE CONTROL SUBSYSTEM

This system performed without encountering major difficulties. The inertial reference unit's pitch, yaw, and roll gyro currents remained within operating tolerances and did not develop any degrading trends. However, the current readings increased with increases in spacecraft temperatures. It is shown in Figure 4-6 that gyro temperatures exceeded the upper operating limit of 69% and reached a telemetry saturation value of 100%. Although the gyro currents reflected this increase, they still remained within tolerance.

4.4.3 COMMUNICATIONS SUBSYSTEM

No communications subsystem anomalies were observed during the extended mission. This subsystem continued to perform within its operating limits except for the period of time that the battery power output dropped below the minimum operating limits. Also, the spacecraft drifted through an antenna null position and the DSIF lost lock with the spacecraft for an extended period of flight. The power and temperature characteristics of the transponder may be seen in Figure 4-8.

4.4.4 THERMAL CHARACTERISTICS

Spacecraft temperatures remained within operating limits when the spacecraft was pitched 40 degrees off the Sun-

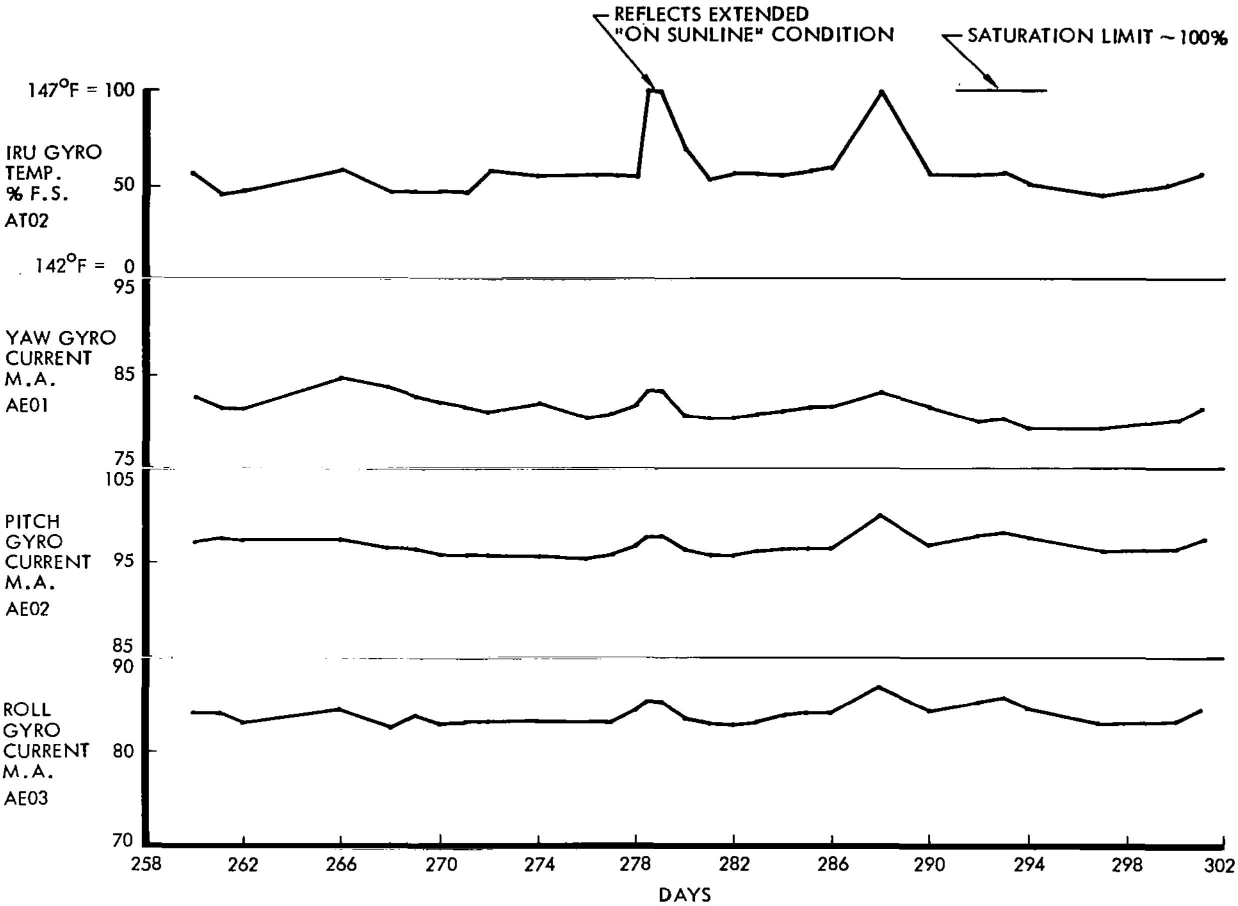


Figure 4-6: Gyro Electrical Characteristics

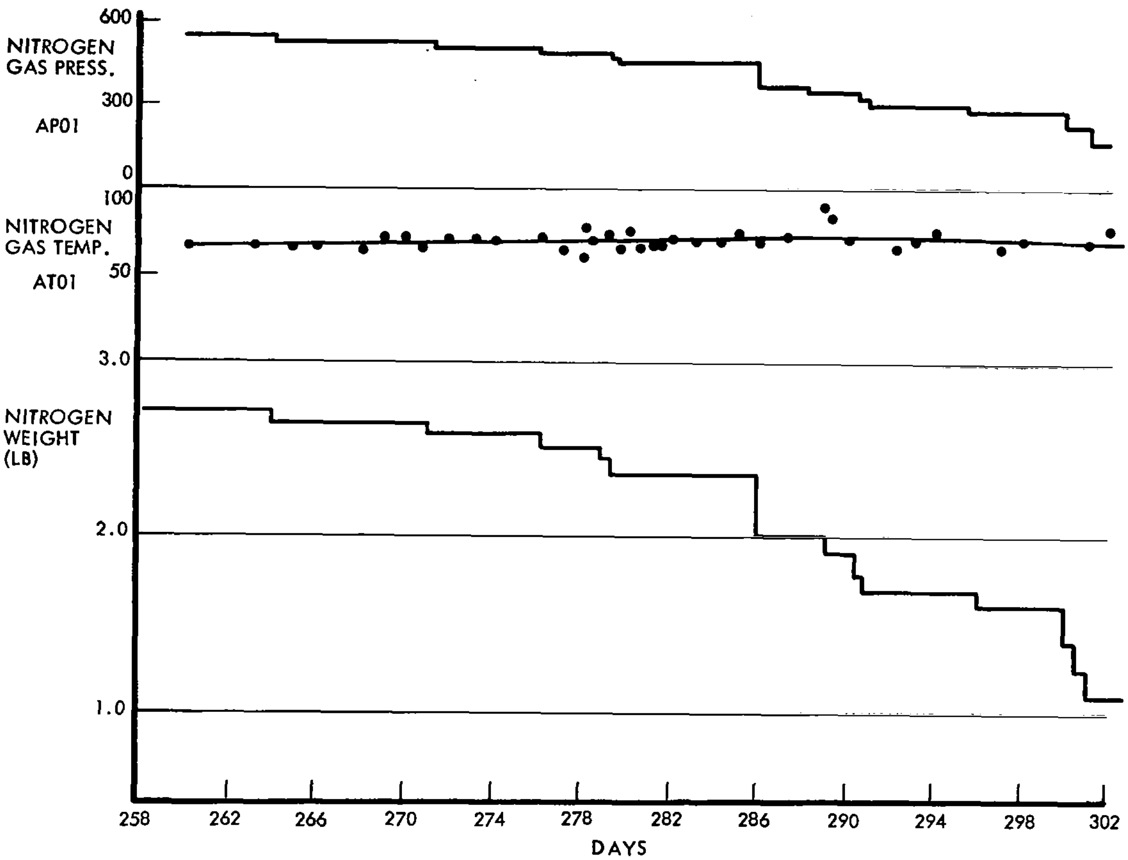
line. However, during the period of spacecraft flight on the Sunline, the cold-plate temperature transducers frequently reached saturation. As shown in Figure 4-8, the cold-plate temperature transducers frequently reached and exceeded the upper limit of 108.1°F and remained above this saturation limit until Sunset occurred. During the most severe flight periods, the temperatures of ST01, ST02, and ST03 would exceed 116°F and could not be monitored. Also, the spacecraft paint characteristics continued to degrade as predicted.

4.4.5 RADIATION OBSERVATIONS

The spacecraft radiation detectors did not record any major solar flares or micrometeoroid hits during the extended mission. Figure 4-9 shows that the radiation level in the No. 1 cassette increased linearly from 10.50 rads to 12.25 rads and the radiation level in the No. 2 loopers increased linearly from 138.5 rads to 141.5 rads. Although the radiation increases were linear, the plots of Figure 4-9 are shown as a step-function which conforms to the method of radiation sampling.

4.4.6 FLIGHT PROGRAMMER

The flight programmer performed within operating limits as programmed. However, on October 15, this system became temporarily inoperative when the power supply voltage was below the programmer operating limits. See Paragraph 4.4.1 for a description of the low-voltage anomaly. This low-voltage condition caused the flight programmer to sequence to the Halt mode. Also, both the fixed-memory address locations and the programmer-addressed location were set to zero. Then a real-time command was transmitted and executed to cause the programmer to leave the Halt mode and return to normal operation, but the zero state of the memory-addressed locations caused the programmer to cycle until it sequenced to an infinite-jump location. Subsequently, the flight programmer memory was updated and it operated normally from that time on until lunar impact of the spacecraft.



$$WT_{GN_2} = \frac{P\,V}{12\,Z\,R\,T}$$

WHERE:

P = PRESSURE IN PSIA ~ AP01 + 60 (60 PSIA ADDED TO COMPENSATE FOR CALIBRATION ERROR)

V = VOLUME IN CU. IN. (1600 CU. IN.)

Z = COMPRESSION CONSTANT (0.995)

R = CONSTANT (55.2)

T = TEMP. IN °R (AT01 + 460)

NOTE: N_2 WEIGHTS MENTIONED IN THE TEXT WERE BASED ON A CALIBRATION THAT DID NOT INCLUDE 60 PSIA BEING ADDED TO AP01

Figure 4-7: Nigrogen Usage Trends

4.4.7 VELOCITY CONTROL SUBSYSTEM

The minimum nitrogen gas usage was 0.11 pound per week when the spacecraft was configured in a ±2-degree deadband. However, when attitude maneuvers were performed, the gas consumption greatly increased. Figure 4-7 indicates the gas usage for the extended mission. The average weekly consumption for the extended mission was 0.275 pound. Therefore, it was apparent that spacecraft operating life was severly reduced when attitude maneuvers were performed, especially when the ±0.2-degree deadband was employed.

The propulsion fuel and oxidizer tank pressures remained constant for the extended mission until the final engine burns required for lunar impact. During the first burn of the lunar impact sequence, propellant tank pressures decayed from approximately 226 psia to 195 psia. This burn lasted 94.4 seconds and imparted a velocity change of 168.85 mps, at which time the propellants were apparently exhausted. The propellant valves remained open for an additional 23 seconds in a "dribble mode" of operation which imparted an additional 6.4 mps. The propellant valves were opened a second time for a period of 150 seconds. No pressure change was observed and

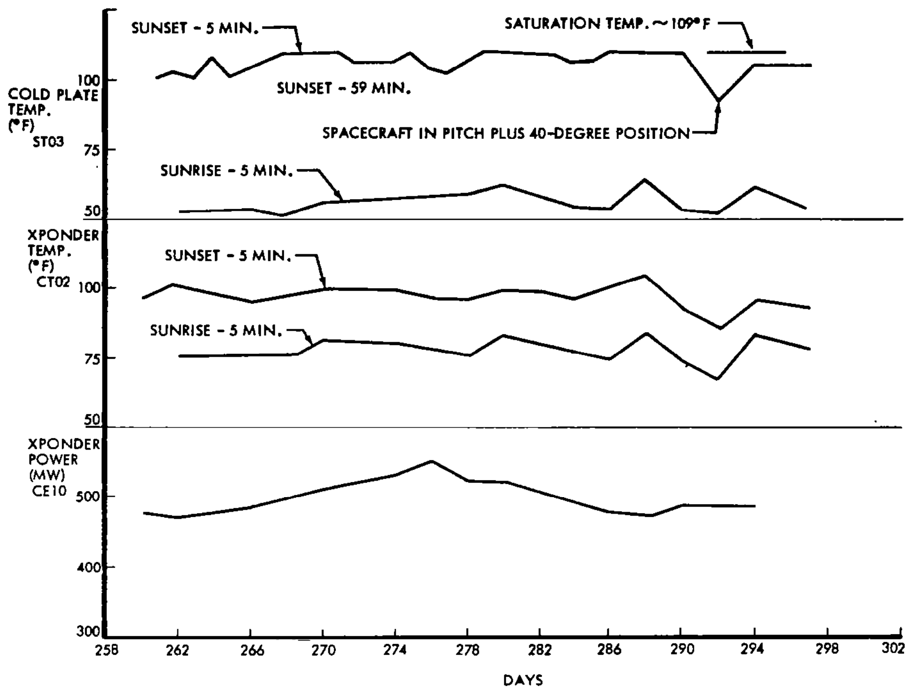


Figure 4-8: Transponder Data

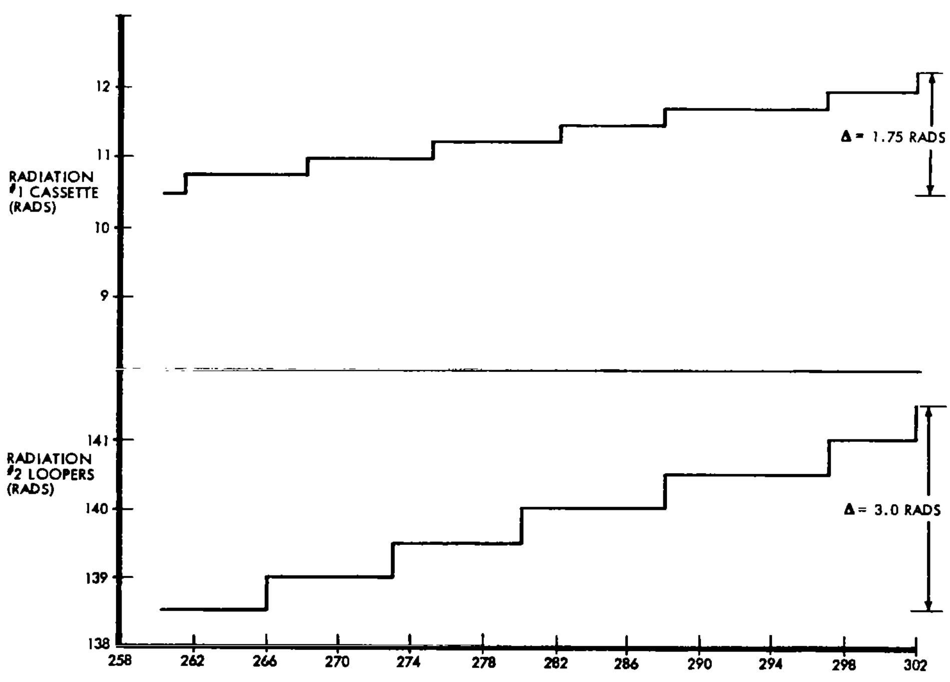
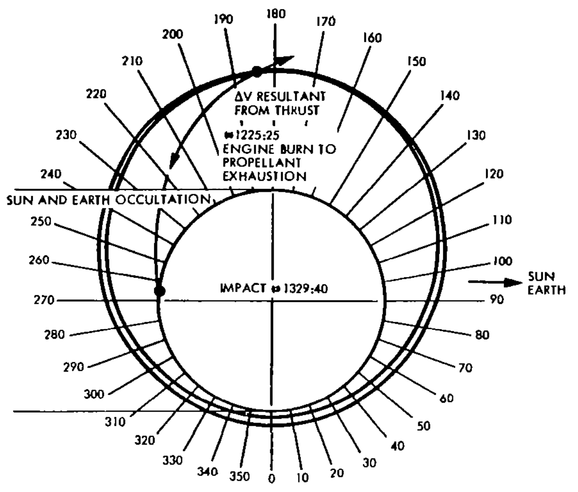


Figure 4-9: Spacecraft Radiation Levels



NOTE: THRUST OCCURRED WITH NO ATTITUDE MANEUVER (I.E., SPACECRAFT WAS ON SUNLINE)

Figure 4-10: Impact Trajectory

	BURN DURATION (SECONDS)	V (M/SEC)	IMPACT TIME	LATITUDE (DEGREES)	LONGITUDE (DEGREES)	APPROACH ANGLE (DEGREES)
1)	86.8	156	13:31:18.7	5.81	166.28	28.7
2)	92.9 (NOMINAL)	167	13:30:21.0	6.38	163.70	29.9
3)	99.0	179	13:29:26.8	6.90	161.25	31.0
ACTUAL		182	13:29:06.3	6.35	160.71	31.0

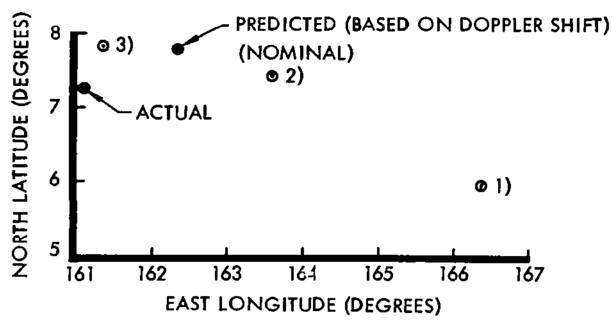


Figure 4-11: Impact Dispersions

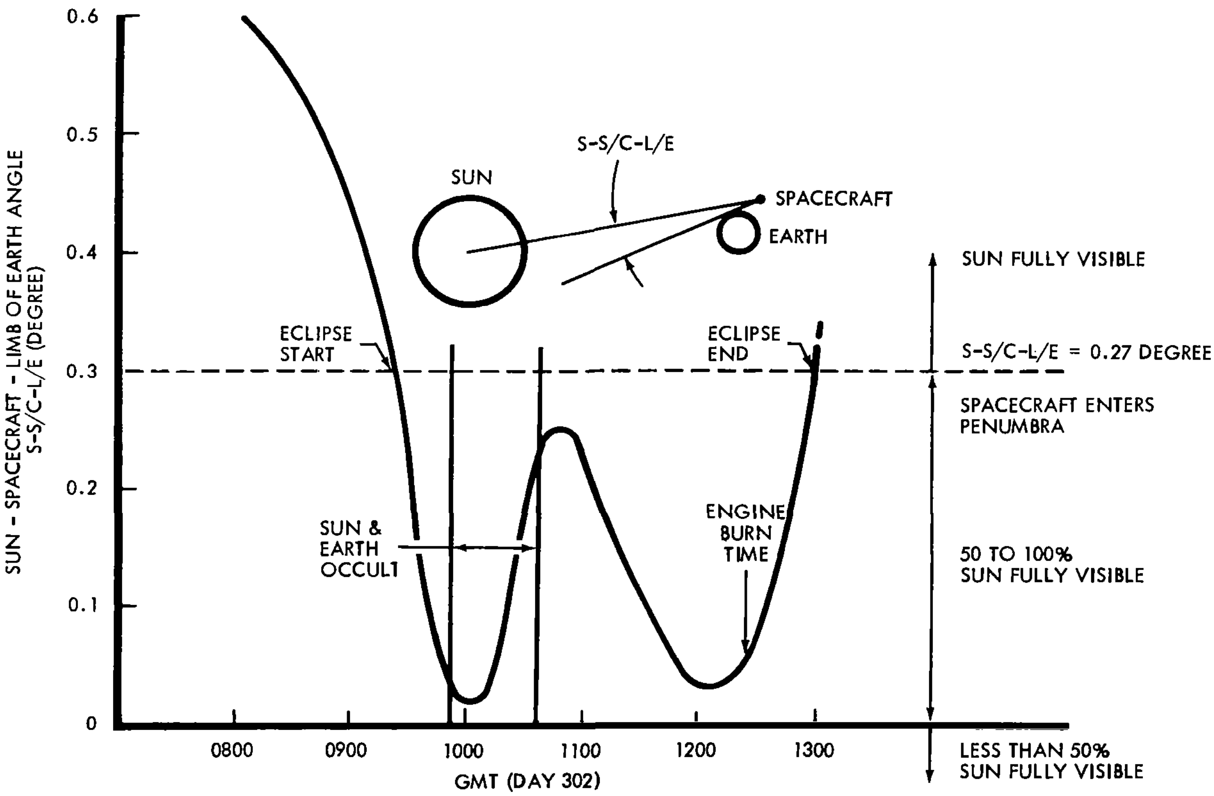


Figure 4-12: Sun-Spacecraft-Near Limb of Earth Angle during Penumbra Eclipse

no actual engine burn occurred but an additional 2.77 mps were imparted to the spacecraft. The trajectory to lunar impact and the impact dispersions are shown in Figures 4-10 and 4-11 respectively.

Gimbal actuators behaved in a normal manner during the first burn with excursions on the order of ±0.1 degree. During the second and final burn, the actuators cycled radically and at times went "hard over" to the "stall" position. With the actuators "stalled" and the engine valves actuated, their load current at times reached 10.25 amps. This load plus other spacecraft loads resulted in intermittent peak loads of 13.5 amps. Additional details regarding the velocity control subsystem are presented in Volume V.

4.4.8 PHOTO SUBSYSTEM

During the extended mission, the photo subsystem temperature and pressures, which were monitored, remained relatively constant and always within the acceptable operating range. The observed data was as follows:

Transducer	Average Value	Variation
Nitrogen Bottle Press.	2216 psi	± 1.3%
Subsystem Press.	1.477 psi	± 4.7%
Environmental Temp.	68.7°F	± 10%

4.5 PENUMBRA ECLIPSE

The following sequence of events occurred during penumbra eclipse. The spacecraft entered the eclipse on October 29, Day 302, 09:20 GMT. At 09:50 Earthset and Sunset occurred. Referring to Figure 4-12, it may be seen that at 10:10 the minimum "Sun-Spacecraft- near limb of Earth" angle of 0.01 degree occurred. At 10:37, Earthrise and Sunrise occurred, at which time DSS-12 acquired a two-way lock with a received signal strength of -136.5 dbm. At 10:38, DSS-41 acquired a three-way lock with 12:26. The spacecraft's transponder was turned off at 12:30 and back on again at 12:36. At 12:52 the spacecraft was commanded and performed a successful clock switchover. This was followed by a successful "accelerometer on" and "accelerometer off" command. At 12:55 the spacecraft was out of the penumbra eclipse.

Due to Earth occultation, shortly after emerging from the eclipse, DSS-12 lost lock at 13:15:57 and DSS-41 lost lock at 13:15:20. At this time the spacecraft was on a lunar impact trajectory, as illustrated in Figure 4-12. It is estimated that the time of impact was 13:29:40.

The spacecraft power profile during the eclipse is shown in Figure 4-13. The trace of the "Sun-Spacecraft-near limb of Earth" angle as it passed through the eclipse is shown in Figure 4-12. The spacecraft was pitched off

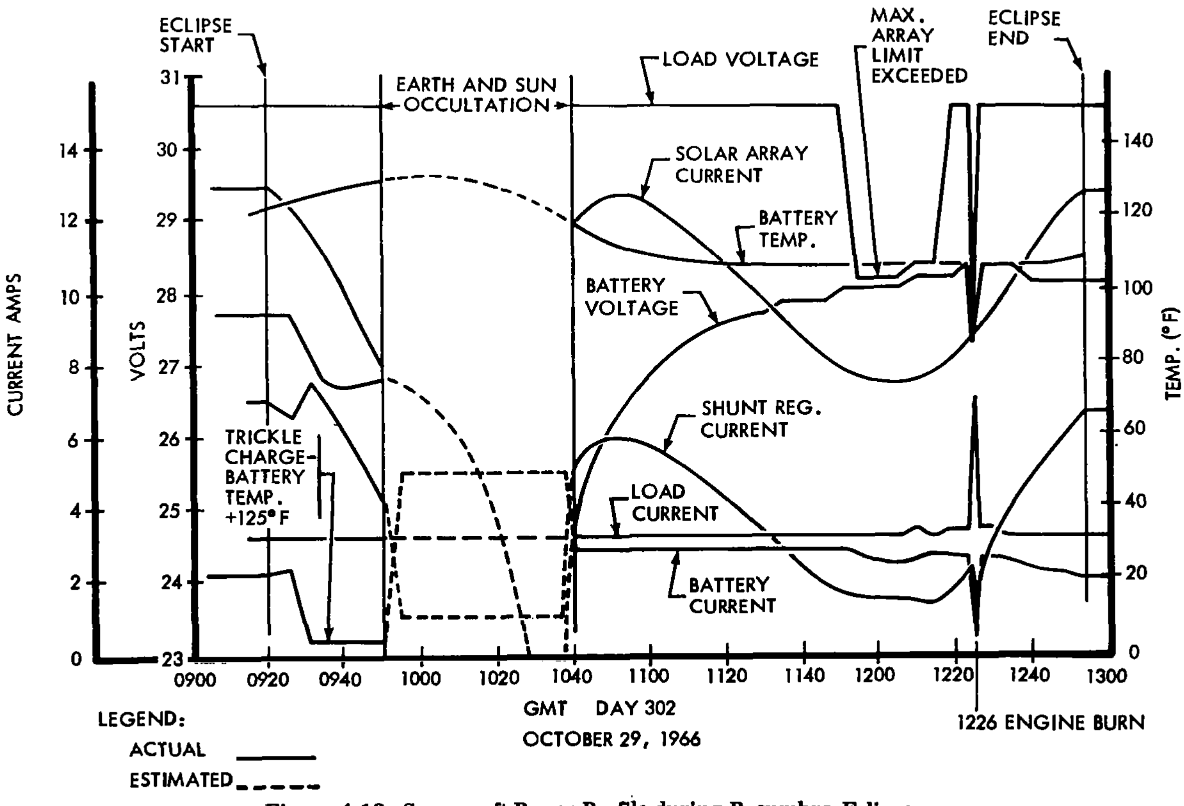


Figure 4-13: Spacecraft Power Profile during Pneumbra Eclipse

the Sunline plus 35 degrees when it entered the solar eclipse. From the time that the spacecraft entered the solar eclipse until telemetry was terminated by Earth occultation, the solar array current dropped from 12.67 amps to 7.93 amps. The battery temperature increased from 125°F and then reached an estimated high of 129°F following Sunset. When the battery temperature reached 125°F, the battery current dropped to a trickle charge value, as designed. The battery voltage dropped 1 volt and the shunt regulator current dropped about 3 amps. The cold-plate temperature saturated at 108°F prior to Sunset. Earth and Sun occultation occurred simultaneously, which prevented monitoring the battery performance during the Sun occultation period. It is estimated that the battery voltage dropped to about 20 volts prior to Sunrise. Shortly after Sunrise the array current began to drop again. This time the battery temperature was lower and the battery current did not drop to the trickle charge level. However, the solar array power limits were exceeded and the solar array voltage dropped to near the battery voltage level of 28 volts. The solar array output current began to increase, as may be seen in Figure 4-13, and then the solar array voltage returned to its normal level of 30.56 volts. At the end of the eclipse, the power system performed normally and apparently survived the eclipse without any degradation.

Table 4-1 lists the recorded telemetry data from the power subsystem during the penumbra eclipse.

4.6 PRE-EXTENDED MISSION TESTING

Several experiments were performed with the Lunar Orbiter I spacecraft following completion of the photographic mission and prior to the start of extended-mission operations on September 17. A complete description of these tests is available in Boeing Document D2-100719-1, Early Post-Photo Mission Tests with Lunar Orbiter I. The name and results of these tests are as follows:

1) Second Readout of Goldstone Test Film - - Substantial reduction in light output from the photo subsystem was evident; however, focus required no adjustment changes during the mission.

2) Low-Gain-Antenna Glint Test - - This test confirmed that the low-gain antenna was the principal source of reflected light that caused Canopus star tracker difficulties in Sunlight.

3) Bright-Object-Sensor Test - - Bright-object shutter closure occurred within the required limit when viewing the Moon.

4) Northern Hemisphere Star Map - - Canopus star tracker data were obtained on Vega, Capella, Betelgeuse, and Sirius, and generally confirmed a prior computations.

5) Extended-Mission Mode Test (Wide Deadband, Coarse Sun Sensors) - - This test confirmed that the spacecraft must be oriented off the Sun to prevent serious overheating.

6) Exploratory Investigations - Cyclic Doppler Residuals Near Perilune - - This test showed that power reflections off the lunar surface had no significant effect on doppler residuals near perilune.

7) Downlink Modulation Index Test - - Measurements of the normal received signal characteristics were made. No abnormal downlink transmission conditions have since been reported.

8) Command Threshold Test- -Starting at a ground transmitter power of 2 kw, the power was reduced while monitoring the spacecraft AGC. Dummy commands were transmitted and, if verified, power was further reduced until the command threshold was found. This occurred at an AGC between -126.1 and -127.3 dbm.

4.7 EXPERIMENTS

4.7.1 STANFORD UNIVERSITY LUNAR RF REFLECTION EXPERIMENT

On October 8 there was a deviation in the normal flight plan to enable Stanford University to conduct an experiment that involved receiving rf signals from the spacecraft after they were reflected off the Moon. While the spacecraft was being prepared to support the Stanford experiment, DSS-42 collimated it's antenna using the RF signal from the spacecraft. The preparation for the experiment included determining the spacecraft's roll position. This was done by rotating the high-gain antenna to a predetermined position, then rotating the spacecraft to the point where the rf signal strength was the highest. This oriented the spacecraft's position in roll. The antenna was then rotated to face the Moon, and the spacecraft was pitched to the proper position. The high-gain antenna was then turned on and RF from the high-gain antenna was reflected off the Moon for a period of 1 hour and 26 minutes. The spacecraft performed as commanded and the experimenters received what appeared to be satisfactory data. While performing this test the spacecraft experienced higher than normal temperatures as a result of being on the Sunline for an extended period. Also, an additional 0.15 pound of gas was consumed.

On October 12 the Stanford University experimenters received the rf signal from the spacecraft with a special receiver that determined the signal's phase shift from the reflected rf signal off the Moon. This data was received by the experimenters from DSS-12 when the spacecraft was in the zenith position during Orbit 365. The experiment was conducted without having to deviate from the standard operation plan. The final segment of the Stanford University lunar reflection experiment was completed on October 19.

4.7.2 V/H SENSOR MAPPING EXPERIMENT

On October 16 a V/H sensor mapping experiment was attempted, but was unsuccessful because mission control personnel overlooked the fact that the photo subsystem had been inhibited by a "solar eclipse on" command early in the extended mission. Also, when the camera thermal door was commanded to open and to close, no "camera thermal door control off" command was sent to terminate the sequence. Since this left power on the stepping motor for an extended period of time, an excessive load current developed. This excessive load condition was discovered on October 19 when the V/H sensor test was again attempted. However, due to improper procedures used during the first test, this test was also unsuccessful. Power was then removed from the stepping motor, allowing the spacecraft's load current to return to normal.

Day	GMT (Hr, Min)	Array Current (Amps)	Load Voltage	Battery Voltage	Battery Current (Amps)	Shunt Reg. Current (Amps)	Load Current (Amps)	Battery Temp. #1 (°F)	Solar Panel Temp. #2 (°F)
302	09 15	12.67	30.56	27.68	2.089	6.92	3.11	120.9	90.9
	09 20	START ECLIPSE							
		12.61	30.56	27.68	2.089	6.86	3.11	122.5	90.9
	25	12.25	30.56	27.68	2.121	6.50	3.11	123.9	90.9
	30	11.53	30.56	27.04	.276	7.52	3.11	125.5	84.4
	35	10.69	30.56	26.88	.276	6.74	3.11	126.8	77.9
	40	9.79	30.56	26.88	.276	5.90	3.11	127.8	68.4
	45	8.77	30.56	26.72	.276	4.92	3.17	128.8	56.1
	50	7.93	30.56	26.77	.276	4.12	3.17	129.4	47.0
	09 5017	EARTH OCCULT							
	10 3600	EARTH RISE							
	10 37	2-WAY LOCK							
	10 3703	DSS-12 2-WAY LOCK							
	10 3926	11.71	30.56	24.89	2.846	5.05	3.30	118.5	-87.3
	10 45	12.37	30.56	25.76	2.846	5.78	3.24	115.4	35.2
	10 50	12.55	30.56	26.24	2.846	5.78	3.29	112.9	111.0
	10 55	12.49	30.56	26.56	2.846	5.90	3.24	111.2	150.4
	11 00	12.31	30.56	26.72	2.846	5.78	3.17	109.4	177.4
	05	11.95	30.56	27.04	2.840	5.41	3.17	108.4	185.3
	10	11.59	30.56	27.28	2.846	5.11	3.17	107.6	177.4
	15	11.23	30.56	27.36	2.846	4.74	3.17	106.9	165.6
	20	10.81	30.56	27.52	2.846	4.31	3.17	106.6	150.4
	25	10.27	30.56	27.52	2.846	3.87	3.17	106.6	132.1
	30	9.67	30.56	27.68	2.846	3.43	3.17	106.2	117.9
	35	9.25	30.56	27.84	2.846	2.92	3.11	106.2	100.8
	40	8.71	30.56	27.84	2.844	2.33	3.17	106.2	84.4
	45	8.29	30.56	27.84	2.846	2.00	3.17	106.2	74.7
	50	7.87	30.56	28.00	2.846	1.60	3.11	106.2	62.2
	55	7.64	28.16	28.00	2.690	1.46	3.17	106.2	50.0
	12 00	7.46	28.16	28.00	2.501	1.46	3.17	105.9	44.1
	05	7.40	28.16	28.00	2.478	1.46	3.17	105.9	38.1
	10	7.46	28.32	28.16	2.501	1.46	3.24	105.9	32.3
	15	7.69	28.32	28.16	2.721	1.46	3.17	105.5	32.3
	20	7.99	30.56	28.16	2.752	1.67	3.24	105.5	32.3
	25	8.47	30.56	28.32	2.596	2.40	3.24	105.5	38.1
	ENGINE BURN	1.65	27.52	27.36	.406	1.39	7.16	105.9	41.0
	30	9.07	30.56	28.32	2.752	2.79	3.24	105.5	44.1
	32	RF TURN OFF							
	36	RF TURN ON							
	3630	10.09	30.56	24.32	2.470	4.06	3.17	105.9	53.1
	40	10.69	30.56	28.16	2.407	4.68	3.17	105.9	59.2
302	12 45	11.53	30.56	28.16	2.375	5.41	3.17	106.2	71.6
	50	12.25	30.56	28.16	2.658	5.78	3.17	106.9	81.1
	54	END OF ECLIPSE							
	55	12.67	30.56	28.16	2.343	6.68	3.17	108.4	90.9
	13 00	12.67	30.56	28.16	2.312	6.68	3.17	109.1	90.9

Table 4-1: Power Subsystem Data Recorded during Penumbra Eclipse

4.7.3 RANGING TESTS

Ranging tests were performed during each tracking period for the entire extended mission.

4.8 SIMULATION TRAINING

Simulation training exercises in preparation for Mission B were performed from October 27 to 29 using Lunar Orbiter I. A summary of the exercises performed and results follow.

4.8.1 STAR MAP

A star map was made and several of the major intensity stars were located. The star map was started following one of the Sun occultations. Sunrise occurred just prior to the completion of the map, and glint from the Sun on the star tracker prevented the tracker from locating any other stars.

4.8.2 ROLL ORIENTATION

It was desired to determine the orientation of the spacecraft in the roll position; this was accomplished by making an antenna map and noting the roll position where the signal strength from the spacecraft was the highest.

4.8.3 V/H SENSOR TEST

A V/H sensor test was attempted. The spacecraft was initially rotated to a position where Canopus was located and then a three-axis maneuver was performed. The attempt to open the camera thermal door appeared to fail. The spacecraft was then returned to its initial orientation.

4.8.4 PHOTOGRAPHING SEQUENCE

During an attempt to simulate the picture taking it was found that the film was stuck to the supply reel. This problem was solved by commanding the spacecraft into the readout mode for 20 minutes. This placed about 5 inches of film in the readout looper. Then the photo system was placed in the Camera Take Pictures mode (fast 16 Frame), and when the Camera On command was given the film was freed.

The following frames from Sites I-6 and 7 were read out:

Site	Frame	Resolution	Readout DSS
I-6	145	High	-41
I-6	145	Medium	-41
I-6	146	High	-41
I-6	144	Medium	-41
I-6	143	Medium	-41
I-6	144	High	-41
I-6	142	Medium	-41
I-6	143	High	-41
I-6	141	Medium	-41
I-6	142	High	-41
I-7	167	High	-61
I-7	165	Medium	-61
I-7	166	High	-61
I-7	164	Medium	-61
I-7	165	High	-61

4.8.5 ENGINE BURN

As previously mentioned, an engine burn was performed to ensure the spacecraft would impact the Moon.

4.8.6 RF SWITCHING

Following the engine burn, the transponder was commanded off by a real-time command and was commanded on by the emergency real-time command method. This latter method allows execution of the command even if verification is not received from the spacecraft.

4.8.7 SPACECRAFT CLOCK SWITCHOVER

A spacecraft clock switchover was successfully performed.

4.8.8 ACCELEROMETER SWITCHING

A pair of commands that were sent to the spacecraft turned the accelerometer off and on.

NOTE

The rf switching test and the spacecraft clock switchover test were performed to confirm the design capability between the command mission dependent equipment and the spacecraft as it had not been necessary to use the emergency command mode or clock switchover during Mission I. It was also desirable to determine if the accelerometer could be turned off and on by real-time commands.

4.9 FAILURES

4.9.1 There were no spacecraft or ground equipment failures.

4.9.2 PROCEDURE ERRORS

4.9.2.1 On October 15, 23 pitch-plus-40 degree real-time commands were transmitted to the spacecraft when it was intended that they be stored program commands. This error resulted in an excessive power drain on the battery, which in turn caused the spacecraft subsystems to become inoperative due to insufficient power during two successive orbits. This event is described in more detail in Section 4.4.1 of this document.

4.9.2.2 On October 16 an improper command sequence, intended to accomplish the V/H sensor test, was transmitted to the spacecraft. This improper sequence prevented the experiment from being conducted.

4.9.2.3 On October 24, while performing a pitch-plus-40 degree maneuver, SPC's were transmitted to the spacecraft. This caused the current maneuver to terminate prematurely at an angle of only 33 degrees. These new commands should have been sent to the spacecraft during either a Compare Time (COT) or Wait Time (WAT)

DATE	DAY	TIME	PERIOD (min)	PERILUNE (km)	APOLUNE (km)
9-14	257	06:00	206.26	48.78	1813.7
9-17	260	12:25	206.31	57.67	1805.6
9-19	262	12:50	206.32	53.62	1809.9
9-23	266	17:40	206.32	49.86	1813.6
9-26	269	07:40	206.34	54.08	1809.8
9-28	271	09:00	206.38	54.81	1808.8
9-29	272	22:22	206.39	53.14	1810.8
10- 4	277	02:01	206.4	39.01	1823.3
		19:12	206.3		1825.8
10- 5	278	05:31	206.3	35.11	1827.0
		22:42	206.2	33.45	1828.6
10- 6	279	02:10	206.8	32.69	1833.8
10- 7	280	02:16	206.3	31.04	1835.2
		05:42	206.9	31.00	1835.2
		19:28	206.5	31.46	1834.0
		21:11	206.5	31.71	1834.6
10- 8	281	02:21	206.8	32.03	1833.9
		19:34	206.7	34.39	1831.4
10- 9	282	02:27	206.7	35.59	1830.0
		19:39	206.6	39.70	1825.8
10-10	283	02:32	206.5	41.56	1823.9
		19:45	206.4	46.60	1818.7
10-12	285	17:00	206.3	52.96	1910.9
10-13	286	21:00	206.31	58.13	1805.2
10-15	288	00:30	206.33	60.67	1802.9
10-15	288	16:00	206.37	60.66	1803.3
10-17	290	04:30	206.32	58.36	1805.1
10-19	292	15:50	206.30	53.24	1809.9
10-21	294	17:50	206.32	53.58	1810.0
10-24	297	16:41	206.35	56.99	1807.1

Table 4-2: Orbit Summary

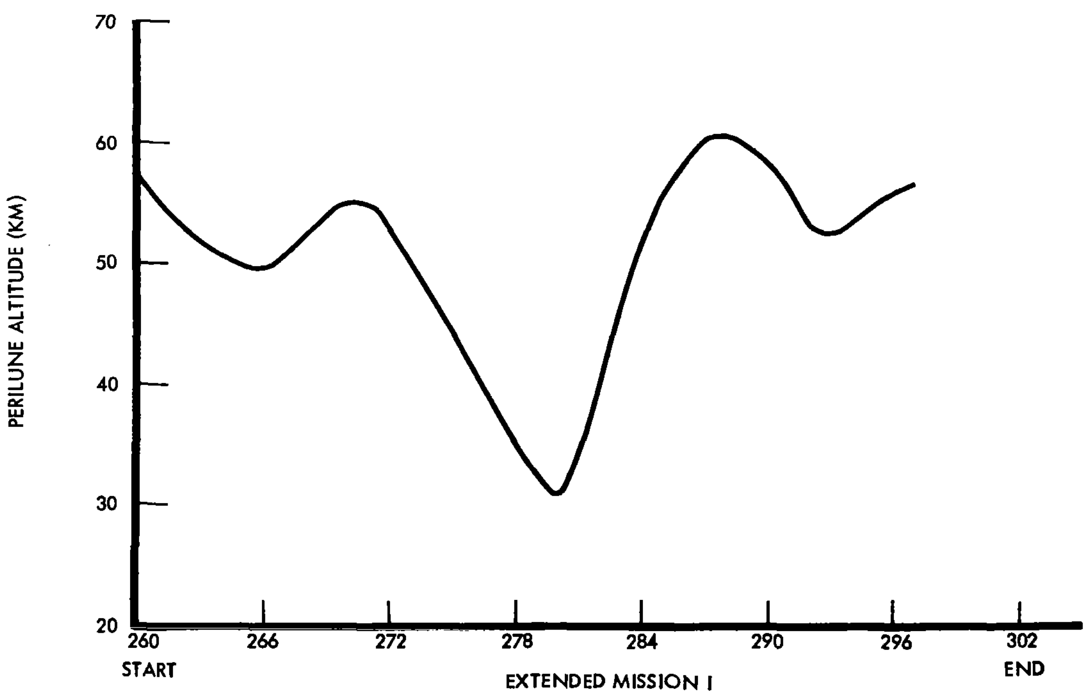


Figure 4-14: Perilune Altitude

period. The above errors can be attributed to insufficient time for operation planning and inadequate manning during high-activity periods.

4.10 ORBIT SUMMARY

The changes in the orbit are given in Table 4-2 and a trace of the perilune altitude is shown in Figure 4-14.

4.11 ORBIT AND TRACKING DATA

Tracking data summaries, orbit determination results, and spacecraft maneuvers between September 17 and October 17 are discussed herein.

4.11.1 TRACKING DATA EDITING

Tracking data editing was accomplished by use of two programs, TDPX and ODGX.

4.11.1.1 TDPX In general, the disposition of the data file on disk was controlled by initialization from a master file tape (B1). This tape was updated during each day's allotted computer time. At the beginning of each run, all rejected data was cleared. Furthermore, since Mode 4 operation was frequently necessary, a "CHANGE SELECT DISK" card was used. Finally, a "CHANGE TIME SORT" option was included to allow processing of past data played into the computer via punched cards, punched paper tapes, and log tapes.

4.11.1.2 ODGX Three types of data were available during the extended-mission phase; they were angle, doppler, and ranging unit data. However, because only small arcs were traversed between observations in orbital phase, angle data were not considered useful and, therefore, were ignored. Doppler and ranging data were processed and placed on the ODP file for use in orbit determination.

Table 4-3 contains a transmitter and frequency summary necessary for processing these data. Tables 4-4 and 4-5 provide time synchronization and ranging correction information respectively which is used to eliminate biases in the data. Table 4-6 contains a list of the master file tapes and the copies that were sent to Langley and retained in the Boeing library at JPL.

4.11.2 ORBIT DETERMINATION

Orbit determination was accomplished by operating the ODPL program to process available data in the interval of interest. These following ground rules were generally followed: (1) a data arc consisting of two orbits of data was used; (2) only CC3 doppler data were used; (3) only the state vector was considered in the solution; and (4) a diagonal a priori covariance matrix having position of 5.0 kilometers and velocity of 0.5 meter per second was used in the solutions until stabilization occurred.

Two exceptions to the above rules are noteworthy: (1) during the brief periods in which two-station tracking coverage existed, C3 data were processed and the doppler bias was estimated along with the state vector, (2) in OD 7010 a special test to evaluate ranging unit data was conducted. Ranging data were used alone and in combination with CC3 data to investigate the effectiveness of such a data type. The results were encouraging - - the fit of the data was excellent, yielding a small sum of squares and standard deviation on the data. In all other orbit determination solutions, ranging unit residuals were calculated, but the data were not used in the solutions.

Throughout all the determinations, data weights of 0.1 cycle and 15 meters were used for doppler data and range units, respectively. Furthermore, the spherical harmonic coefficients released by NASA on September 4, 1966, were used as the lunar model. These gravitational coefficients, the Keplerian state vectors resulting from orbit determinations, and the data used in each are summarized in Table 4-7.

4.11.3 MANEUVERS

Roll maneuvers do not have any perceptible effect on the orbit. However, due to the velocity at which the antenna rotates around the spacecraft roll axis, changes are caused in doppler data. Pitch and yaw maneuvers will, however, cause extremely minor perturbations in the lunar orbit. These are normally ignored in operational orbit determinations but may be of some significance ultimately in the selenodesy studies. A record of all attitude maneuvers is found in Table 4-8.

DSIF	DAY	TRANSMITTER ON (GMT)	FREQUENCY CHANGE AT (GMT)	FREQUENCY 2204XXXX.X
-61	260	12:23:00	12:23:32	5680.0
			12:42:02	5700.0
			13:00:32	5640.0
			15:49:32	5700.0
-61	261	12:37:00	12:38:32	5670.0
			16:05:32	5690.0
-61	261	12:54:00	12:55:02	5670.0
			16:24:02	5680.0
-41	263	09:41:00	09:44:22	5690.0
-61	263	16:40:00	16:49:02	5670.0
-41	264	09:57:00	10:00:02	5680.0
-61	264	16:59:00	17:11:02	5660.0
-12	265	00:41:00	00:47:02	5650.0
-61	265	15:26:00	15:28:02	5670.0
			17:19:02	5650.0
-61	266	17:40:00	17:45:02	5640.0
			20:58:02	5660.0
-61	267	17:44:00	17:48:02	5630.0
			21:24:02	5650.0
-61	268	18:00:00	18:06:02	5620.0
			21:29:02	5640.0
-41	269	07:58:00	07:54:02	5610.0
			11:20:02	5630.0
41	270	08:03:00	08:06:02	5600.0
			11:33:02	5620.0
41	271	08:15:00	08:17:00	5590.0
			11:45:00	5610.0
-61	272	22:20:00	22:21:02	5610.0
			01:42:02	5630.0
-61	273	22:27:00	22:28:02	5600.0
	274		01:54:02	5630.0
12	274		02:43:02	5630.0
-61	274	22:33:00	22:34:02	5600.0
	275		02:02:02	5620.0
-61	275	22:38:00	22:39:02	5600.0
	276		02:04:02	5620.0
-41	276	15:48:00	15:50:00	5620.0
			19:18:00	5640.0
-41	277	15:55:00	15:57:00	5620.0
			19:23:00	5640.0
-12	277		15:55:00	5650.0
-12	278	14:28:00	14:31:00	5640.0
			16:02:00	5650.0

Table 4-3: Transmitter And Frequency Summary

DSIF	DAY	TRANSMITTER ON (GMT)	FREQUENCY CHANGE AT (GMT)	FREQUENCY 2204XXXX.X
41	278	16:02:00	16:03:00	5620.0
			19:30:00	5640.0
-41	279	16:07:00	16:10:00	5620.0
			19:35:00	5640.0
-42	279	20:15:00	Frequencies used and times of each were not recorded by the station.	
12	280	09:44:00	09:54:00	5630.0
			12:53:00	5650.0
-41	280	16:24:00	16:27:00	5630.0
			19:43:00	5640.0
-12	281	12:54:00	12:58:00	5650.0
			16:25:00	5670.0
-61	282	06:08:00	06:15:00	5660.0
			09:38:00	5670.0
-61	283	06:17:00	06:24:00	5660.0
			09:48:00	5680.0
-12	284	13:30:00	13:31:00	5660.0
			16:50:00	5680.0
-12	285	16:58:00	17:01:00	5680.0
			20:27:00	5700.0
-41	285	20:28:00	20:29:00	5660.0
			23:56:00	5680.0
-41	286	20:40:00	20:41:00	5660.0
	287		00:08:00	5680.0
-41	288	00:21:00	00:23:00	5680.0
			00:52:00	5580.0
			03:50:00	5600.0
			07:25:00	5610.0
-12	288	15:54:00	15:56:00	5570.0
			17:41:00	5580.0
			21:09:00	5560.0
-41	290	04:19:00	04:19:00	5580.0
			07:47:00	5560.0
-61	292	15:42:00	15:43:00	5560.0
			18:41:00	5580.0
-12	292	22:10:00	22:13:00	5560.0
			01:40:00	5570.0

Table 4-3: Transmitter And Frequency Summary (Cont.)

DAY	TIME (GMT)	DSIF	TIMING BIAS
257	06:30	-61	7.075 msec
257	22:30	-41	-1.840 msec
273	00:00	-61	-1.000 msec

All biases are measured with respect to DSIF-12.

Table 4-4: Station Time Synchronizations

MADRID RANGING

DAY	GMT Hr. Min. - Hr. Min.	DELAY	Temp. (°F)	NO. OF POINTS	DAY Or NIGHT
257	08 : 21 - 10 : 49	310.		38	D
	12 : 17 - 12 : 18			1	D
	16 : 18 - 16 : 35			13	
	17 : 12 - 17 : 38			5	D
258	08 : 41 - 14 : 24	317.		92	D
	12 : 04 - 14 : 24	317.		94	D
259	08 : 46 - 11 : 17	318.		98	D
	12 : 09 - 14 : 43			73	D
	15 : 38 - 15: 42			3	D
260	13 : 37 - 14 : 55	287.		43	D
	16 : 05 - 18 : 20			67	D
261	12 : 40 - 15 : 08	295.		96	D
	16 : 09 - 18 : 34			90	D
262	12 : 57 - 15 : 20	300.		53	D
	16 : 24 - 18 : 48	300.		22	D
263	16 : 24 - 19 : 03	317.		22	D
	Wrong data condition code this pass (may have some questionable data).				
264	No ranging--signal strength down -155.				
265	15 : 38 - 18 : 49	318.		10	D
266	18 : 02 - 23 : 26	312.		54	D
267	17 : 50 - 23 : 45	319.		68	D
268-269	18 : 53 - 00 : 02	319.		35	N
273	01 : 58 - 03 : 15	320.	94.	2	N
273-274	22 : 32 - 04 : 31	320.	96.	30	N
274-275	22 : 41 - 04 : 41	313.	95.	31	N
275-276	23 : 02 - 04 : 49	306.	89.	26	N
282	07 : 01 - 12 : 23	219.6	86.	10	D
283	06 : 31 - 12 : 36	319.0	87	8	D

GOLDSTONE RANGING

DAY	GMT Hr. Min. - Hr. Min.	DELAY	Temp. (°F)	NO. OF POINTS	DAY OR NIGHT
257	20 : 43 - 23 : 30			34	D
258	15 : 45 - 16 : 25	289.		24	D
	18 : 47 - 19 : 42			34	D
260	No Ranging	312.			
265	No Ranging	304.			
274	No Ranging - Three Way				
278	14 : 46 : 42			1	D
281	13 : 00 - 19 : 16	296.	85	49	D
284	13: 48 - 19 : 34	296.4	86	54	D
285	No Ranging	293.0		0	
288	No Ranging				

WOOMERA RANGING

DAY	GMT Hr. Min. - Hr. Min.	DELAY	Temp. (°F)	NO. OF POINTS	DAY OR NIGHT
257	06 : 01 - 07 : 27	295.		22	D
257-258	22 : 10 - 00 : 45	286.		42	D
258	01 : 33 - 04 : 12	286.		103	D
	06 : 14 - 07 : 39	286.		41	D
258-259	23 : 26 - 00 : 55	293.		50	D
259	01 : 48 - 04 : 19	293.		95	D
	05 : 26 - 07 : 45	293.		78	D
263	09 : 53 - 12 : 05	278.		27	D
	Data condition code is 5, meaning bad angles by manual switch. Reset code to 1, should be good ranging data.				
264	10 : 08 - 12 : 11	335.		16	D
	Same comment on data condition code as above.				
265	07 : 57 - 13 : 53 ata Code 0	288.		51	D
270	08 : 09 - 14 : 03	290.	90.	50	D
271	09 : 03 - 14 : 20	286.	86.	46	D
275	15 : 54 - 21 : 14	288.	94.	40	N
277	16 : 00 - 22 : 09	284.	90.	47	N
278	16 : 12 - 16 : 15	279	80.	1	N
279	16 : 20 - 21 : 48	283.	90.	25	N
280	16 : 32 - 22 : 15	303.	92.	52	N
285-286	21 : 06 - 02 : 36	279.0	86.	60	N
286-287	20 : 51 - 02 : 41	281.0	86.	63	N
288	00 : 46 - 04 : 56	294.0	93.	15	D
290	04 : 26 - 06 : 11	317.3	84	11	D

Table 4-5: Ranging Summary

GMT TIME INTERVAL START TIME	GMT TIME INTERVAL STOP TIME	LANGLEY TAPE NUMBER	BOEING TAPE NUMBER
12:24, 2:60	04:29, 2:74	LT-28	
12:24, 2:60	02:45, 2:87	LT-29	
12:24, 2:60	10:05, 2:90		1334

Table 4-6: Master File Data Tapes

OD IDENTIFI-CATION NUMBER	EPOCH-GMT	STATE VECTOR (SELENOGRAPHIC TRUE-OF-DATE ORBITAL ELEMENTS)					
		a (km)	e	(deg)	(deg)	i (deg)	T_p (sec)
7000	66/09/14-06 : 00 : 00	2669.3543	0.33059480	206.40885	268.20995	12.218572	-5594.5992
7002	66/09/17-12 : 25 : 00	2669.7608	0.32737158	206.32569	226.10738	11.66	4431.6922
7004	66/09/19-12 : 50 : 00	2669.8596	0.32891369	209.89457	196.56347	12.44	5453.2084
7006	66/09/23-17 : 40 : 00	2669.8245	0.33031086	213.07467	139.82512	12.04	-2898.8714
7007	66/09/26-07 : 50 : 00	2670.0686	0.32879297	213.50984	105.93115	12.10	-1901.3643
7008	66/09/29-22 : 20 : 00	2670.0960	0.32915249	212.05859	60.833786	11.336929	66.491916
7009	66/09/28-09 : 03 : 00	2669.9061	0.32848192	213.41697	79.618555	11.915380	1994.7408
7010	66/09/28-09 : 03 : 00	2669.9582	0.32849278	Not Avail.	Not Avail.	Not Avail.	1995.2446
7010	66/09/28-09 : 03 : 00	2669.9156	0.32848090	213.44383	79.592185	11.915930	1994.8286
7011	66/09/30-22 : 20 : 00	2670.5387	0.32996645	210.69955	49.187890	10.589608	- 172.31723
7012	66/10/03-15 : 50 : 00	2670.2583	0.33380841	221.81496	3.5161623	11.013675	+ 451.63863
7013	66/10/04-16 : 00 : 00	2670.0529	0.33511672	222.34206	350.33222	12.270169	801.03837
7014	66/10/05-16 : 00 : 00	2670.3418	0.33641106	223.30184	336.88094	12.367491	546.20077
7015	66/10/07-10 : 00 : 00	2669.7456	0.33718412	224.57921	314.19142	12.615031	+3182.0529
7016	66/10/10-06 : 25 : 00	2669.8722	0.33449146	230.72813	273.26006	12.303504	+1877.2331
7017	66/10/12-17 : 00 : 00	2670.0328	0.32920019	234.33492	239.26937	12.375648	2333.6954
7018	66/10/13-21 : 00 : 00	2669.7649	0.32719865	234.79942	223.73534	12.308317	4113.8788
7019	66/10/15-00 : 30 : 00	2669.8978	0.32628080	235.13343	208.52326	12.074914	4094.8759
7020	66/10/15-16 : 00 : 00	2670.1355	0.32634532	235.77625	199.43294	12.134674	-1989.9489
7021	66/10/17-04 : 30 : 00	2669.8194	0.32713238	236.84906	178.75523	12.184619	5636.9245
7022	66/10/17-04 : 30 : 00	2669.8323	0.32712504	236.69861	178.89169	12.430417	5637.0172
7023	66/10/19-15 : 50 : 00	2669.6771	0.32901059	238.58058	145.3856	12.214208	-3573.2019
7024	66/10/21-17 : 50 : 00	2669.8934	0.32893560	237.67199	119.59021	12.396893	3123.0005

Table 4-7: Orbit Determination Data

OD IDENTIFI-CATION NUMBER	DSIF	DATA TYPE	START TIME	STOP TIME	NUMBER OF POINTS	STANDARD DEVIATIONS
7000	-41	CC3	9/14 06:00	9/14 07:30	178	0.0966
	-61	C3	9/14 06:05	9/14 07:29	150	0.104
		CC3	9/14 08:14	9/14 09:29	150	0.049
7002	-61	CC3	9/17 12:25	9/17 18:01	509	0.322
7004	-61	CC3	9/19 12:55	9/19 18:39	292	0.235
7006	-61	CC3	9/23 17:50	9/23 23:24	253	1.08
7007	-41	CC3	9/26 07:54	9/26 13:51	301	1.70
7008	-61	CC3	9/29 22:22	9/30 04:18	299	1.14
7009	-41	CC3	9/28	9/30 14:02	234	0.59
7010	-41	RU	9/28 09:20	9/28 13:20	31	10.3
7010	-41	RU	9/28 09:20	9/28 13:20	31	162.0
		CC3	9/28 09:03	9/28 14:02	234	0.99
7011	-12	C3	10/1 02:44	10/1 03:29	65	0.67
	-61	CC3	9/30 22:29	10/1 03:29	206	0.84
7012	-41	CC3	10/3 15:52	10/3 22:01	290	0.68
7013	-12	C3	10/4 16:00	10/4 16:58	59	0.081
	-41	CC3	10/4 16:00	10/4 18:43	147	0.06
7014	-41	CC3	10/5 16:05	10/5 20:29	220	0.28
	-12	C3	10/5 16:05	10/5 16:59	106	0.086
7015	-41	CC3	10/7 16:30	10/7 21:59	278	0.91
	-12	CC3	10/7 10:00	10/7 14:37	382	1.55
7016	-61	CC3	10/10 06:26	10/10 12:35	317	0.81
7017	-12	C3	10/12 20:32	10/12 23:11	122	0.24
	-41	CC3	10/12 20:33	10/13 02:38	289	0.26
7018	-41	CC3	10/13 21:00	10/14 02:44	282	0.262
7019	-41	CC3	10/15 00:30	10/15 06:23	291	0.27
7020	-12	CC3	10/15 16:00	10/15 22:44	286	0.326
7021	-41	CC3	10/17 04:30	10/17 10:04	267	0.497
7022	-41	CC3	10/17 04:30	10/17 09:29	234	0.335
7023	-12	CC3	10/19 22:16	10/20 04:05	218	1.05
	-61	CC3	10/19 15:50	10/19 20:50	212	1.27
7024	-61	CC3	10/21 17:51	10/21 21:49	55	0.065

Table 4-7: Orbit Determination Data (Cont.)

DAY	TIME (GMT)	MANEUVER
260	01:36	P+25
261	03:14	ASU
261	04:36	P+25
262	06:13	ASU
262	07:35	P+25
263	09:13	ASU
263	10:35	P+25
264	12:12	ASU
264	13:34	P+25
265	15:12	ASU
265	16:34	P+25
266	18:11	ASU
266	19:33	P+25
267	21:11	ASU
267	22:33	P+25
269	00:10	ASU
269	01:32	P+25
270	03:09	ASU
270	04:31	P+25
271	06:09	ASU
271	07:31	P+25
272	09:08	ASU
272	10:30	P+25
273	12:08	ASU
273	13:38	P+25
274	15:07	ASU
274	16:29	P+25
275	07:46	ASU
275	09:08	P+30
276	10:37	ASU
276	11:59	P+30
277	13:28	ASU
277	14:50	P+30
278	17:00	ASU
278	19:35	R+135
278	20:14	R+237
278	20:40	R-33.6
278	20:49	P+30
279	19:10	ASU
279	20:32	P+30
280	22:01	ASU
280	23:23	P+30
282	00:52	ASU
282	02:14	P+30
283	03:43	ASU
283	05:05	P+30
284	06:34	ASU
284	07:56	P+30
285	09:26	ASU
285	09:46	P+30
286	11:39	ASU
286	12:00	P+30
287	14:57	ASU
287	15:18	P+30
288	03:00	P-65
288	17:59	ASU
288	18:19	P+30
289	20:50	ASU
289	21:17	P+40
290	07:00	ASU
290	07:01	R+360
290	09:40	P+26.5
290	09:50	Y+7.5
290	09:59	P+40
290	23:41	ASU
291	00:08	P+40
291	03:13	ASU
291	03:40	P+40
292	22:25	ASU
292	22:52	R+16.3
292	23:01	P+34.9
293	02:10	P-35.9
293	02:25	R-53.5
293	03:16	P+38.0
294	08:03	ASU
294	08:30	P+40
295	11:05	ASU
295	11:33	P+40
296	13:56	ASU
296	14:24	P+40
297	16:48	ASU
297	17:15	P+40
298	19:39	ASU
298	20:06	P+40
299	22:30	ASU
299	22:57	P+40

R = Roll
P = Pitch
Y = Yaw
ASU = Acquire Sun
Numbers indicate the magnitude and direction of the maneuver.

Table 4-8: Summary Of Spacecraft Maneuvers

NASA-Langley, 1967 —— 31 CR-848

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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