NASA CONTRACTOR REPORT

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LUNAR ORBITER I

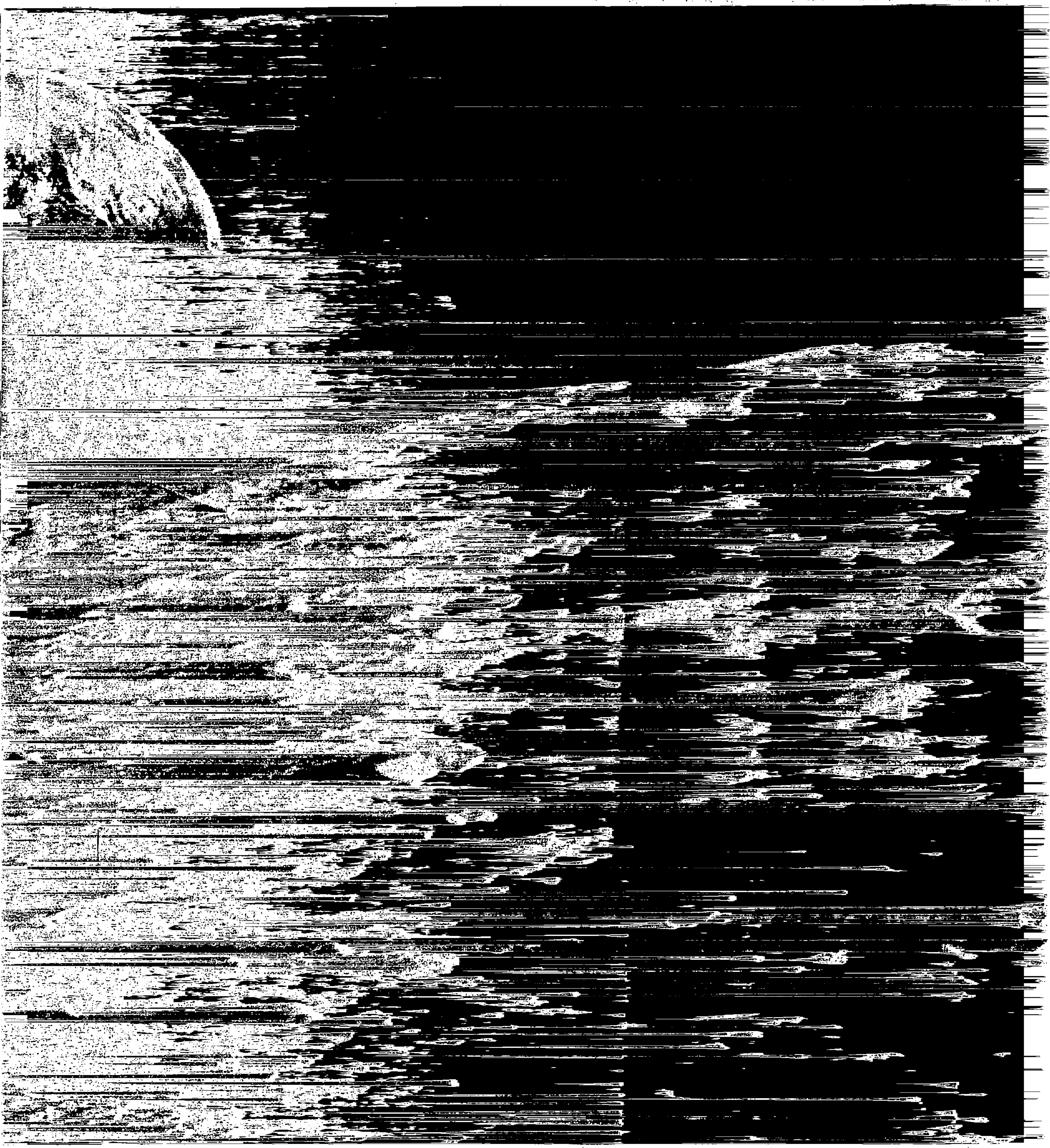
PHOTOGRAPHIC MISSION SUMMARY

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

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • APRIL 1967

First Earth photograph from lunar vicinity taken by Lunar Orbiter I on August 23,1966 - 16:36:23 GMT

GMT



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Prepared under Contract No. NAS 1-3800 by
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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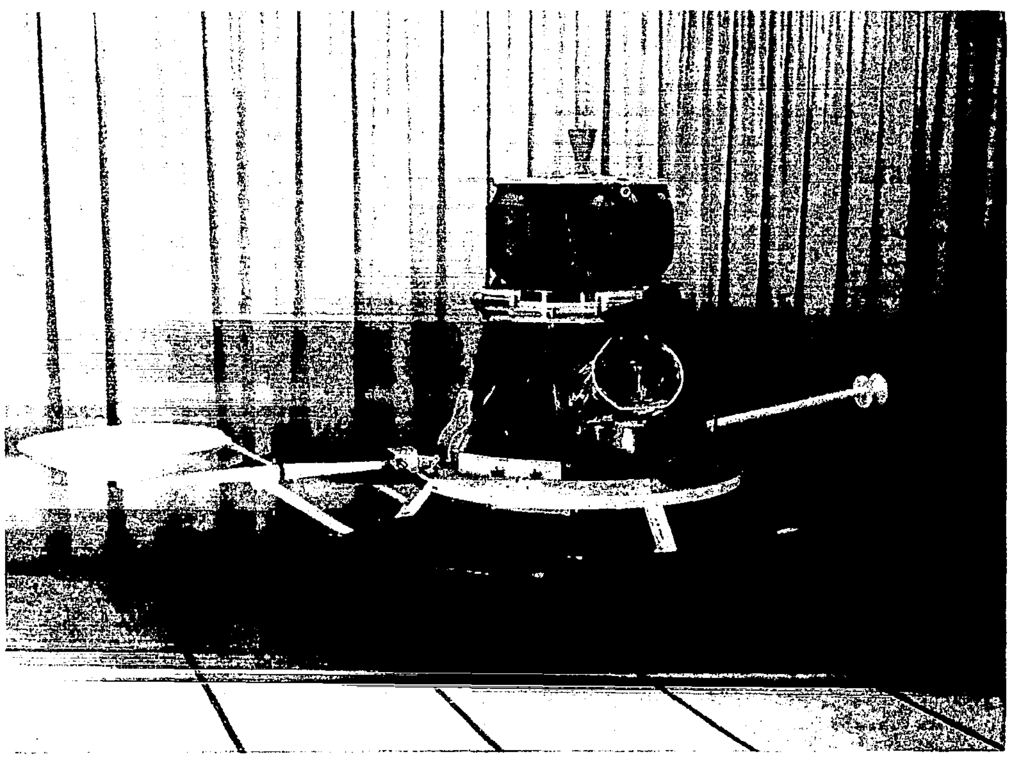
1.0 SUMMARY - LUNAR ORBITER MISSION I

The first Lunar Orbiter spacecraft was successfully launched from Pad 13 at the Air Force Eastern Test Range by an Atlas-Agena launch vehicle at 19:26 GMT on August 10, 1966. Tracking data from the Cape Kennedy and Grand Bahama tracking stations were used to control and guide the launch vehicle during Atlas powered flight. The Agena-spacecraft combination was maneuvered into a 100-nauticle-mile-altitude Earth orbit by the preset onboard Agena computer. In addition, the Agena computer determined the maneuver and engine burn period required to place the spacecraft on the cislunar trajectory 40 minutes after launch. Tracking data from the down range stations were used to monitor the entire boost trajectory. During the next 15 minutes, the spacecraft deployment sequences were completed and the Sun acquired.

Twenty-eight hours after launch, the single midcourse maneuver was executed using the Moon and Sun as references to establish the spacecraft attitude. Ninety-two hours after launch, the spacecraft injected into a lunar orbit with an initial orbit perilune of 189 km. On August 18th, 4 days and 23 hours after injection into lunar orbit, a series of 20 photos were taken of Mare Smythii (approximately 90° E longitude and the lunar equator). The velocity control engine was ignited a third time on August 21st to reduce perilune altitude to 56 kilometers. Twenty-nine hours later the first primary site was photographed with a 16-exposure sequence. Photo sites I-2 through I-5 were photographed during the next 3 days.

Readout and evaluation of these early photos taken of Site I-0 (Mare Smythii) showed that the moderate-resolution photos were satisfactory while the high-resolution photos contained smeared images caused by electrical transients tripping the focal-plane shutter prematurely. On August 26th, a velocity change of only 5.4 meters per second was commanded and executed to reduce the perilune altitude by 8 kilometers in an effort to improve the quality of the high-resolution photos. The remaining photo sites (I-6 through I-9.2) were photographed between August 26th and 28th and showed no improvement in the high-resolution-photo quality. In addition, photos were taken of the farside of the Moon, areas of scientific interest on the frontside, the Earth as seen from the vicinity of the Moon, and possible sites for future Lunar Orbiter missions.

Film processing was completed on August 30th and the complete readout of all photos initiated. Readout of the last of the 211 dual-exposure photos and completion of the photographic mission occurred on September 14th.



The Lunar Orbiter Spacecraft

1.1 PROGRAM DESCRIPTION

The Lunar Orbiter program was formalized by Contract NASI-3800 on May 7, 1964, as one of the lunar and planetary programs directed by the NASA headquarters Office of Space Sciences and Applications. It is managed by the Langley Research Center, Hampton, Virginia, with The Boeing Company as the prime contractor. Lunar Orbiter is the third of a family of unmanned photographic spacecraft, each contributing complementary data. These data-gathering programs are designed to enhance the success of a manned lunar landing and return. The three successful Ranger flights provided a series of decreasing area photographs with increasing resolution (approaching 1 foot) as each spacecraft approached and impacted the Moon. The Surveyor provides detailed information on lunar surface characteristics (with resolution in millimeters) in the immediate area of each successful soft landing. In turn the Lunar Orbiter's mission is to photograph large areas at a resolution level adequate to provide information for selection and verification of safe landing sites for manned Apollo vehicles.

The Lunar Orbiter system was designed to photograph specific target sites within an area of interest bounded by ±10 degrees latitude and ±60 degrees longitude. Sites of interest within the primary region can be classified as:

1) Single-site search and examination;

2) Large-area search;

3) Spot photos.

Designated areas of scientific interest and landmarks for Apollo navigation outside of the primary area may also be photographed.

Each of the five missions (during the 1966 to 1967 period) should provide topographic information of at least 8,000 square kilometers at nominal 1-meter resolution and approximately 40,000 square kilometers at nominal 8-meter resolution. This coverage can be obtained by single photographs or 4, 8, or 16 exposure sequences in either of two automatic sequencing modes. (Nominal 2 or 8 seconds between exposures.)

Additional program objectives include the collection of selenodetic data which can be used to improve the definition of the lunar gravitational field, and the size and shape of the Moon. Radiation intensity and micrometeoroid impact measurements are also to be obtained to further define the lunar environment.

At the completion of each photographic mission (approximately 30 to 35 days after launch), the spacecraft may remain in lunar orbit for an extended period and additional tracking data, environmental monitoring, and scientific experiments conducted.

1.1.1 PROJECT DESCRIPTION

Successful accomplishment of Lunar Orbiter program objectives requires the integrated and cooperative efforts of government agencies, private contractors, numerous subcontractors, and the worldwide data collection system of the NASA Deep Space Network. The functional relationship and responsibilities of these organizations is shown in Figure 1.1-1.

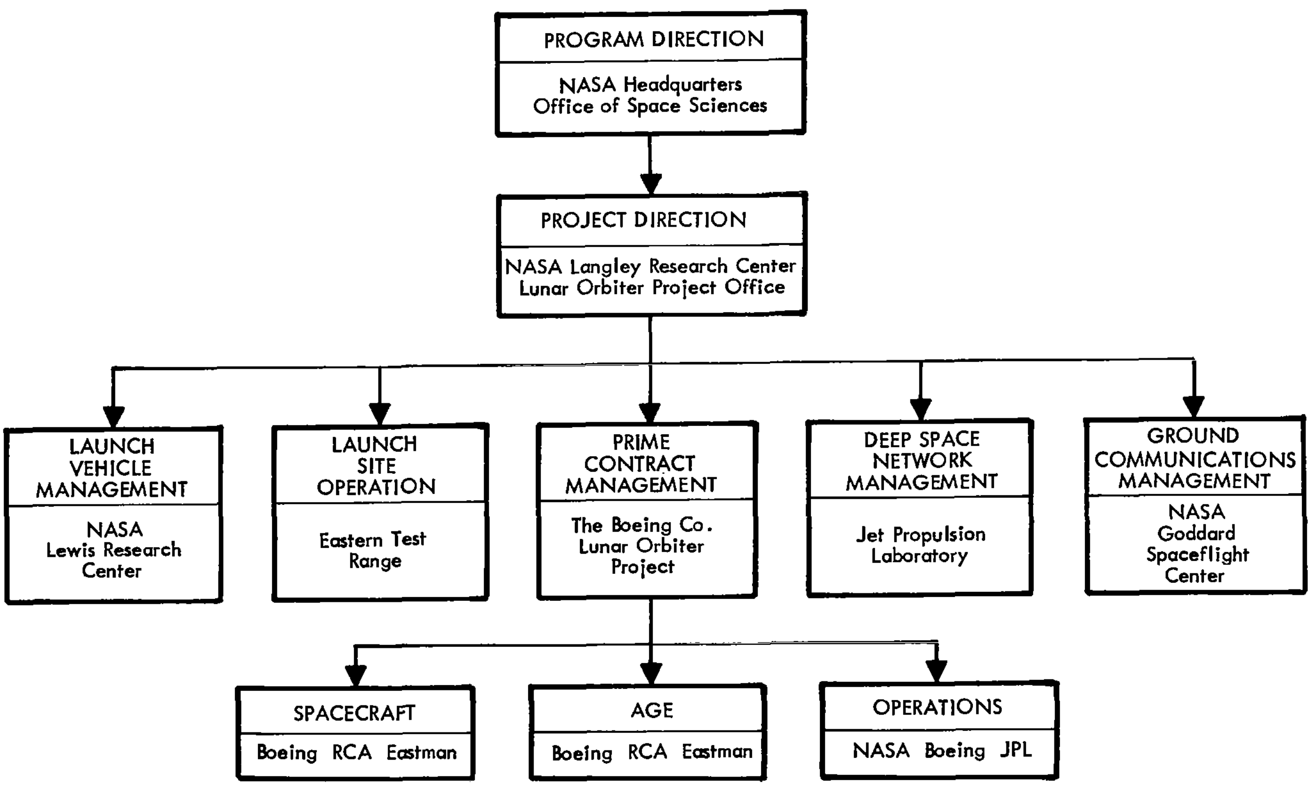
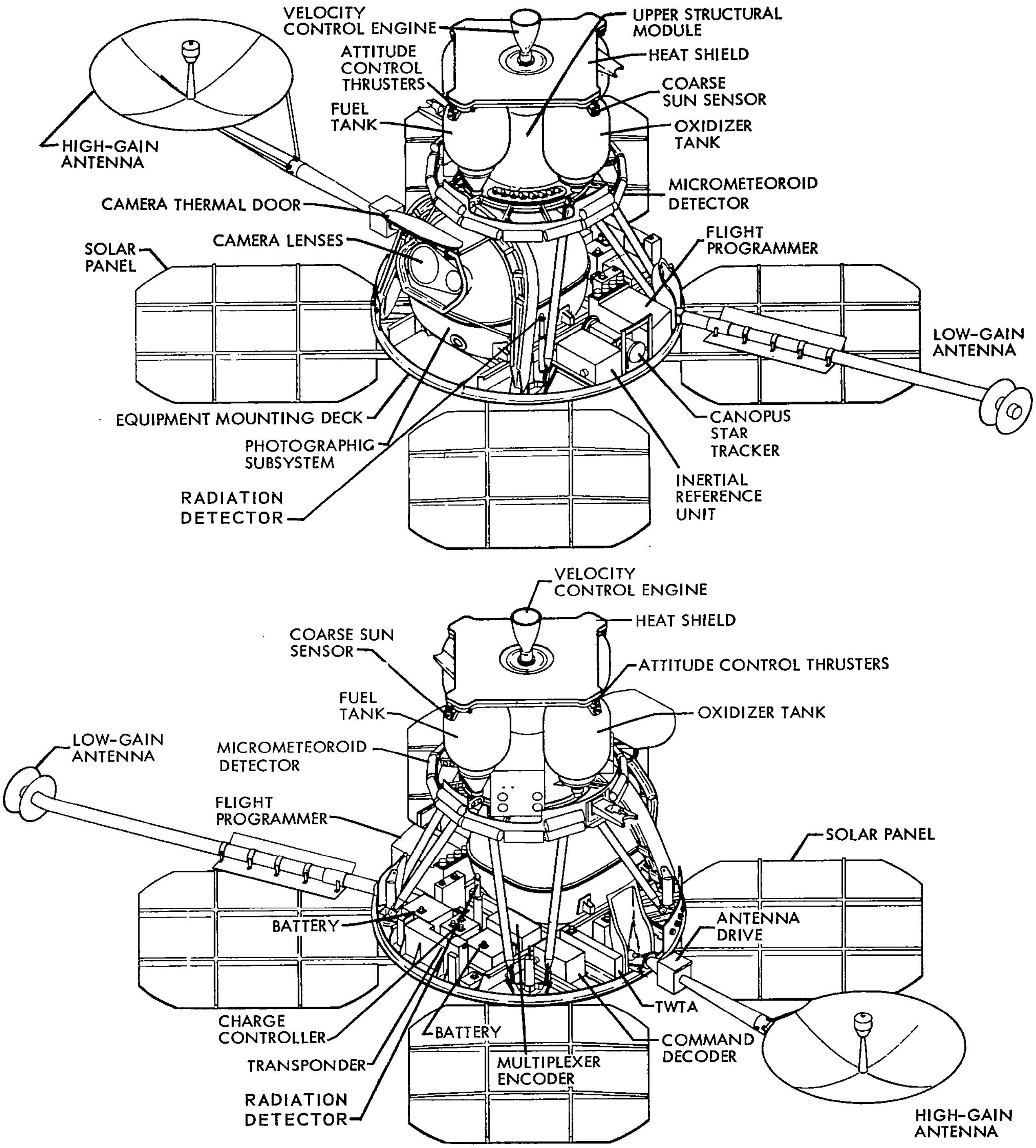


Figure 1.1-1: Lunar Orbiter Project Organization



NOTE: SHOWN WITH THERMAL BARRIER REMOVED

Figure 1.1-2: Lunar Orbiter Spacecraft

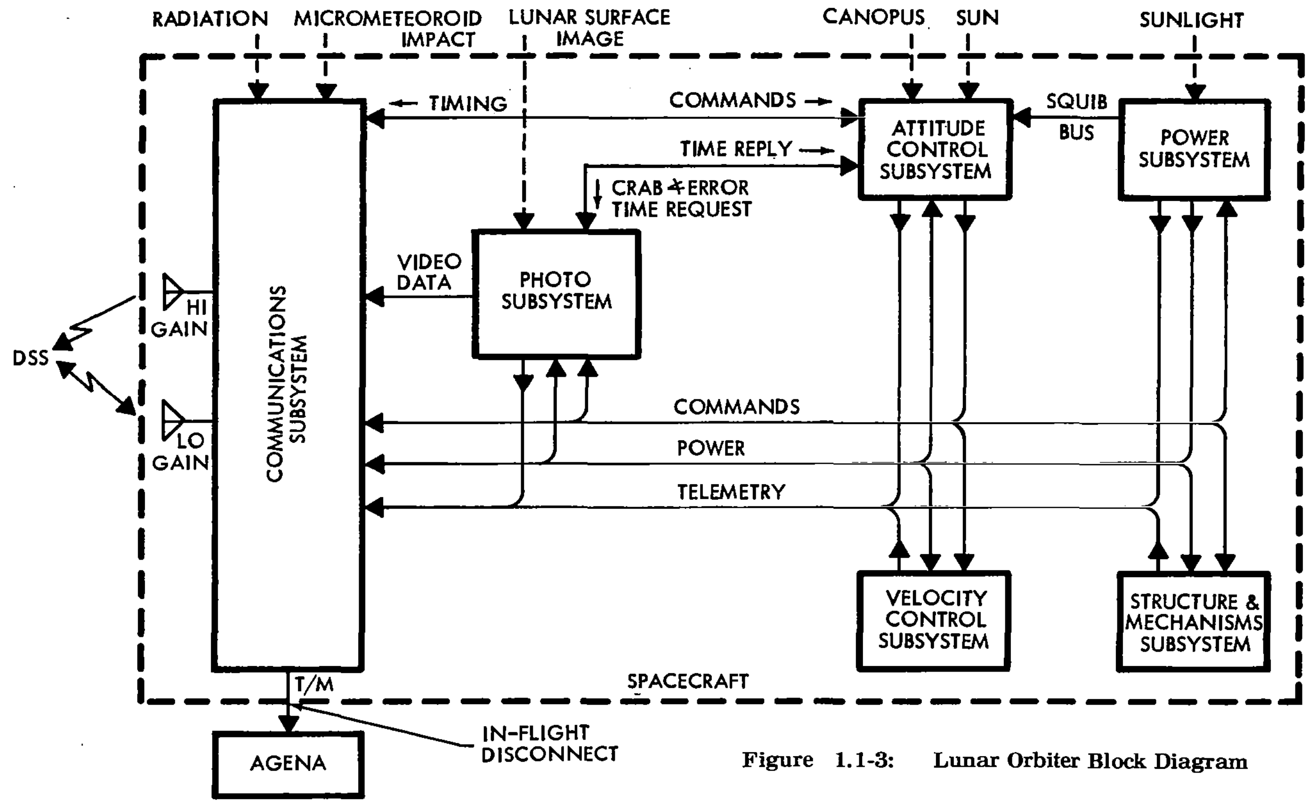


Figure 1.1-3: Lunar Orbiter Block Diagram

As the prime contractor, Boeing is responsible to the Lunar Orbiter Project Office of the NASA Langley Research Center for the overall project management and implementation necessary for the complete operating system. Boeing is also responsible for maintaining - - with and through the NASA-Langley Research Center - - effective working relationships with all participating government agencies.

The NASA Lewis Research Center supports the Lunar Orbiter program by providing the Atlas-Agena launch vehicle and associated services that are necessary to: (1) ensure compatibility of the spacecraft with the launch vehicle; and (2) launch and boost the spacecraft into the proper cislunar trajectory.

The Deep Space Network is managed by the Jet Propulsion Laboratory. This network, consisting of the Space Flight Operations Facility (SFOF) and the Deep Space Stations (DSS), provides two-way communications with the spacecraft, data collection, and data processing. Facilities are provided for operational control which interface with Lunar Orbiter mission-peculiar equipment. Support is also provided in terms of personnel, equipment calibration, and housekeeping services.

Eastern Test Range provides facilities, equipment, and support required to test, check out, assemble, launch, and track the spacecraft and launch vehicle. The ETR also controls the Atlas launch vehicle trajectory and monitors Agena performance through cislunar injection, separation, and retro fire to ensure orbital separation. Appropriate instrumentation facilities, communications, and data recorders are provided at downrange and instrumentation ships to ensure the availability of data for boost trajectory control, acquisition by the Deep Space Network, and postmission analysis.

Goddard Spaceflight Center is the responsible agency for the worldwide network of communication lines necessary to ensure prompt distribution of information between the several tracking stations and the Space Flight Operations Facility during the mission and mission training periods.

1.1.2 SPACECRAFT DESCRIPTION

The 380-kilogram (853-pound) Lunar Orbiter spacecraft is 2.05 meters (6.83 feet) high, spans 5.30 meters (17.5 feet) from the tip of the rotatable high-gain dish antenna to the tip of the low-gain antenna, and measures 3.96 meters (12 feet) across the solar panels. Figure 1.1-2 shows the spacecraft in the flight configuration with all elements fully deployed (the mylar thermal barrier is removed to provide visibility). Major components are attached to the largest of three deck structures which are connected by a tubular truss network. Thermal control is maintained by controlling emitted internal energy and absorbed solar energy through the use of a special paint covering the bottom side of the deck structure. The entire spacecraft periphery above the large equipment-mounting deck is covered with a highly reflective aluminum-coated mylar shroud, providing an adiabatic thermal barrier. The tank deck is designed to withstand radiant energy from the velocity control engine to minimize heat losses in addition to its structural functions. Three-axis stabilization is provided by using the Sun and Canopus as primary angular references and by a three-axis inertial system when the vehicle is required to operate off celestial references during maneuvers or when the Sun and Canopus are occulted by the Moon.

The spacecraft subsystems (as shown in Figure 1.1-3) have been tailored around a highly versatile photo laboratory, containing two cameras, a film supply, film processor, a processing web supply, an optical electronic readout system, an image motion compensation system (to prevent image smear induced by spacecraft velocity), and the control electronics necessary to program the photographic sequences and other operations within the photo subsystem. Operational flexibility of this photo

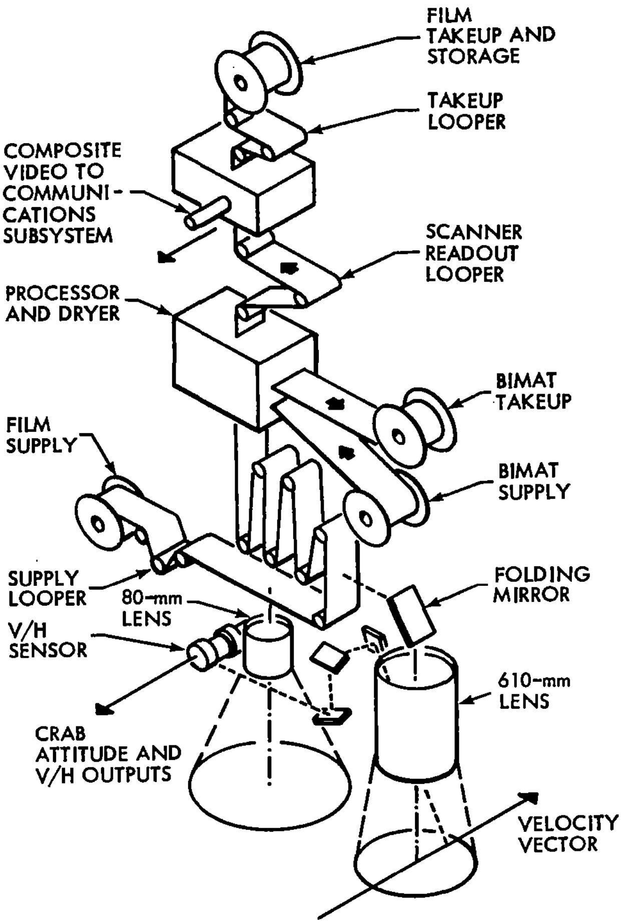


Figure 1.1-4: Photo Subsystem

subsystem was emphasized by providing in-flight capability to adjust key system parameters (e.g., number of frames per sequence, time interval between frames, shutter speed, line scan tube focus) by remote control from the ground.

The influence of constraints and requirements peculiar to successful operation in lunar orbit are apparent in the specific design selected.

1) A three-axis stabilized vehicle and control system were selected to accommodate the precise pointing accuracies required for photography and for accurate spacecraft velocity-vector corrections during midcourse, lunar orbit injection, and orbit-transfer maneuvers.

2) The spacecraft is occulted by the Moon during each orbit with predictable loss of communication from Earth. Since spacecraft operations must continue behind the Moon, an on-board command system was provided with a 128-word memory to support up to 16 hours of automatic operation. It can, however, be interrupted at virtually any time during radio communication to vary the stored sequences or introduce real-time commands.

To afford this operational flexibility, the selected programmer design was essentially a digital data processing system containing register, precision clock, and comparators, to permit combining 65 spacecraft control functions into programming sequences best suited to the spacecraft operations required during any particular phase of the mission.

3) The communications system high-gain dish antenna was provided with a ±360-degree rotation capability about the boom axis to accommodate pointing errors introduced by the Moon's rotation about the Earth.

4) Two radiation detectors were provided to indicate the radiation dosage levels in the critical unexposed film storage areas. One detector measured the exposure seen by the unexposed film remaining in the shielded supply spool, the second, the integrated radiation exposure seen by the undeveloped film in the camera storage looper.

The data from these detectors will allow the selection of alternate mission plans in the advent of solar flare activity.

1.1.2.1 PHOTO SUBSYSTEM

The Lunar Orbiter photo subsystem exposes and processes film and converts the information contained on the film to an electrical signal for transmission to Earth. The complete system, shown schematically in Figure 1.1-4, is contained in a pressurized temperature-controlled container.

The camera system features a dual-lens optical system that simultaneously produce two images on the 70-mm SO-243 Film. High-resolution photographs are obtained by using a 610-mm (24 inch) focal-length Pacific Optical Paxoramic lens at the same time that moderate-resolution photographs are obtained by an 80-mm (3-inch) focal-length Schneider Xenotar lens. Both of these lenses operate at a fixed aperture of f/5.6 with controllable shutter speeds of 0.04, 0.02, and 0.01 second. A double-curtained focal-plane shutter is used with the high-resolution lens and a between-the-lens shutter is used with the moderate-resolution lens. Volume limitations within the photo system container necessitated the use of a mirror in the optical path of the 610-mm lens. This mirror results in the reversal of all high-resolution photographs on the spacecraft film (from left to right across the flight path) with respect to the moderate-resolution photographs.

An auxiliary optical system, which operates through the high-resolution lens system, samples the image of the lunar terrain and determines a velocity-to-height (V/H) ratio. This output is converted to an image motion compensation signal (IMC), which moves each camera platen to compensate for image motion at the film plane. The V/H ratio also controls the spacing of shutter operations to provide the commanded overlap.

Each exposure command produces a medium-resolution and a high-resolution picture. The physical arrangement of the lens system prevents the placing of these two photographs on adjacent areas in the spacecraft film. Figure 1.1-5 identifies the picture format on the spacecraft film. The overlay of high- and moderate- resolution photos for single- and multiple-frame sequence in the fast and slow mode is shown in Figure 1.1-6. The time of each exposure is exposed on the film in digital code by 20 timing lights.

The latent image (exposed) film is developed, fixed, and dried by the processor-dryer. The Eastman Kodak "Bimat" system processes the spacecraft film at a rate of 2.4 inches per minute and requires 3.4 minutes to fully process the latent image. The processing is accomplished by temporarily laminating the emulsion side of the Bimat film against the SO-243 film emulsion as it travels around the processor drum. After leaving the processor drum, the film passes over a heated drum where any moisture in the film is removed and collected by a desiccant. Thereafter, the film can be read out and moreover is no longer susceptible to radiation damage. After processing the last photograph, the Bimat web may be cut by a hot wire upon command. The film can then be moved in either direction by command after Bimat clear.

The photographic data is converted by the readout system into an electrical form that can be directly transmitted to the ground receiving stations. This is accomplished in the readout system by scanning the film with a high-intensity beam of light. The variations in light intensity, produced by variations in the density of the film, are detected by a photomultiplier tube and converted to an analog electrical voltage illustrated in Figure 1.1-7. The readout system electronics adds timing and synchronization pulses to the composite video signal. Thus, it is possible to transmit continuous variations in film tone or density rather than the discrete steps associated with a digital system. The electrical signals are fed to a video amplifier and in turn passed to the modulation selector; transmission is via a traveling-wave-tube (TWT) amplifier and high-gain antenna.

The high-intensity light beam for film density readout is focused to a 6.5-micron-diameter spot on the spacecraft film by a lens system. The spot sweeps 2.67 mm in the long dimension of the spacecraft film. This sequence is repeated 286 times for each millimeter of film scanned. When the scanned sequence across the 70-mm spacecraft film is completed, a film advance mechanism advances the film 2.54 mm (0.1 inch) and the mechanical scanning process proceeds in the opposite direction across the film. Readout of the photographic data for a length of spacecraft film equal to a single high- and moderate-resolution picture requires approximately 43 minutes. The process is shown in Figure 1.1-8.

A film storage buffer (looper) is provided between the camera and the processor dryer because of the different film travel rates during exposure and processing. Another looper is provided between the processor-dryer and the readout unit because the direction of film movement during readout is opposite to the direction of processing. The Bimat web can move in only one direction, thus priority readout is limited to four spacecraft frames by the capacity of the readout looper. After Bimat cut and Bimat clear, the film can move in either direction throughout the photo subsystem. Thereafter, readout periods are limited only by the photo subsystem temperatures, power availability, and DSIF view periods.

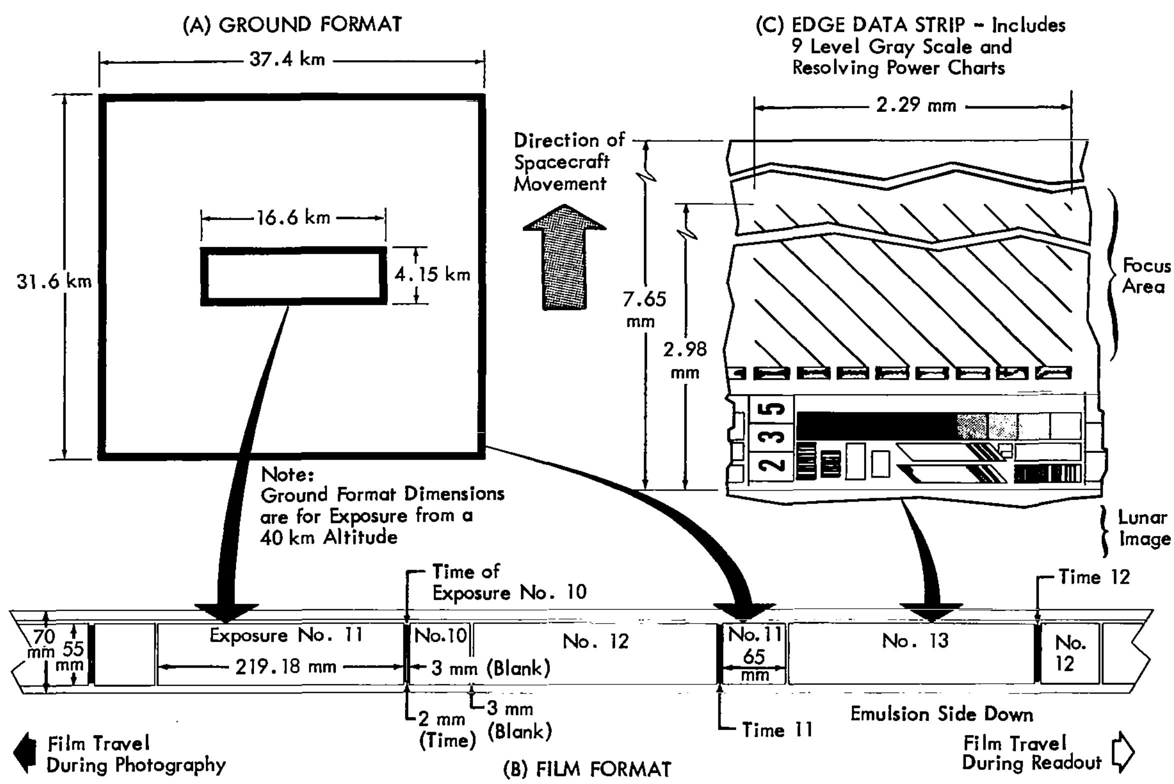
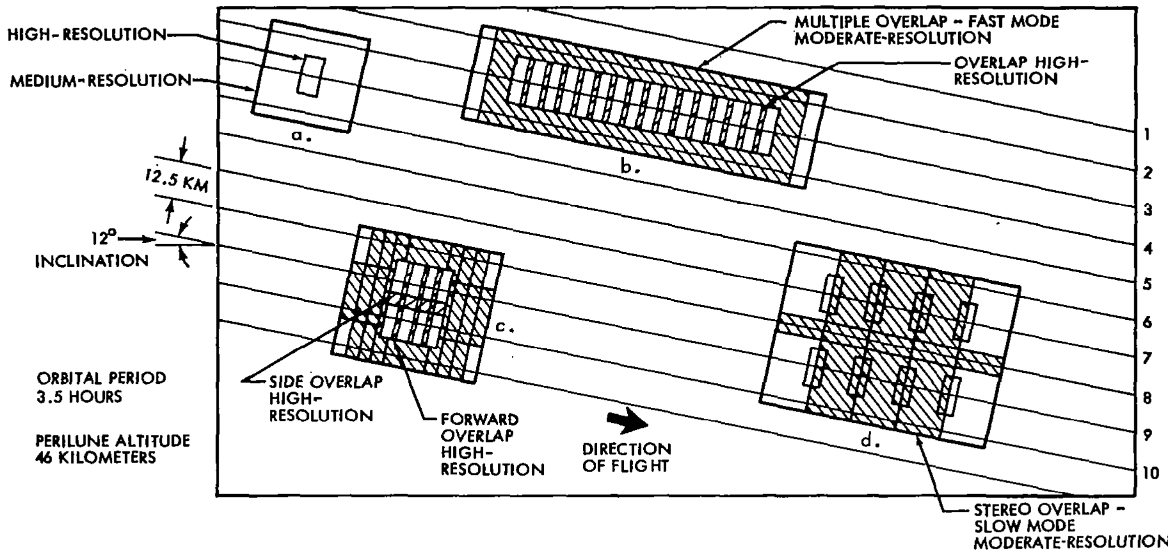


Figure 1.1-5: Film Format



a. SINGLE EXPOSURE AT THE 46-KILOMETER ALTITUDE; HIGH-RESOLUTION COVERAGE IS 16.6 KM X 4.15 KM MODERATE-RESOLUTION COVERAGE IS 37.4 KM X 31.6 KM.

b. DESIGN MISSION TARGET SITE COVERAGE, 16 CONSECUTIVE EXPOSURES DURING ONE ORBITAL PASS OVER THE TARGET SITE. THE INTERVAL BETWEEN EXPOSURES (APPROXIMATELY 2.2 SECONDS) IS TIMED TO PROVIDE 5% OVERLAP OF THE HIGH-RESOLUTION FRAMES WITH APPROXIMATELY 87% OVERLAP ON MODERATE-RESOLUTION FRAMES.

c. EIGHT TYPICAL FRAMES FROM A SITE EXAMINATION TYPE MISSION - CONTIGUOUS HIGH-RESOLUTION COVERAGE IS PROVIDED BY RAPID EXPOSURE RATE AS IN (b) TO GIVE HIGH-RESOLUTION FORWARD OVERLAP AND BY PHOTOGRAPHING ON CONSECUTIVE ORBITS (9 & 10) TO GIVE HIGH-RESOLUTION SIDE OVERLAP.

d. EIGHT TYPICAL EXPOSURES FROM A SITE SEARCH TYPE MISSION-STEREO MODERATE-RESOLUTION COVERAGE IS PROVIDED BY INCREASING THE TIME INTERVAL BETWEEN EXPOSURES (APPROXIMATELY 8.8 SECONDS) TO OBTAIN 50% MODERATE-RESOLUTION FORWARD OVERLAP AND BY PHOTOGRAPHING ON ALTERNATE ORBITS (7 & 9) TO OBTAIN MODERATE RESOLUTION SIDE OVERLAP

Figure 1.1-6: Photographic Modes

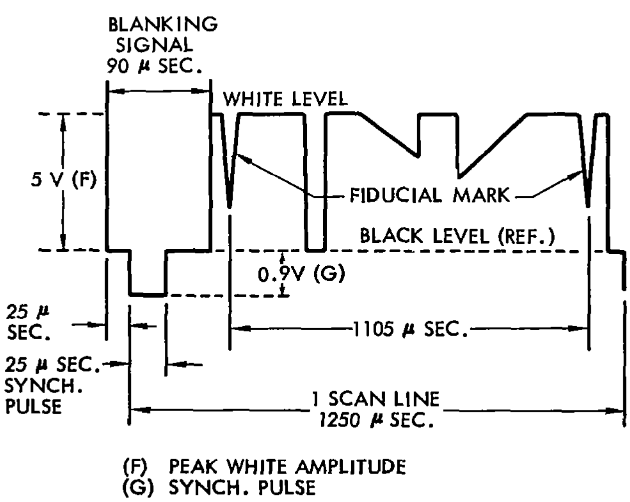


Figure 1.1-7: Video Signal Waveform

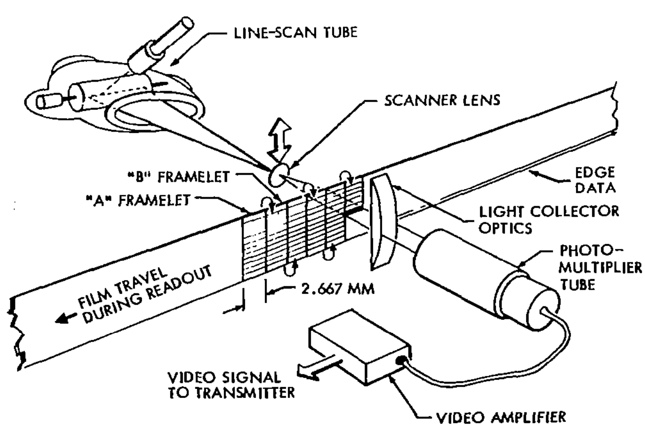


Figure 1.1-8: Readout

1.1.2.2 ELECTRICAL POWER SUBSYSTEM

All of the electrical power required and used by the spacecraft is generated by the solar cells mounted on the four solar panels. Solar energy is converted into electrical energy to supply spacecraft loads, power subsystem losses, and charge the hermetically sealed nickel-cadmium battery as shown in Figure 1.1-9 Excess electrical energy is dissipated through heat dissipation elements. The shunt regulator also limits the output of the solar array to a maximum of 31 volts. Auxiliary regulators provide closely regulated 20-volt d.c. outputs for the temperature sensors and the telemetry converter. Charge controller electronics protect the battery from over-voltage, and over-temperature conditions by regulating the charging current. The 12-ampere-hour battery (packaged in two 10-cell modules) provides electrical power at all times when there is insufficient output from the solar array. During Sun occultation periods, the electrical load of the spacecraft is supplied by the battery.

Each of the four solar panels has 2,714 individual solar cells mounted in a 12.25-square-foot area. The N-on-P silicon solar cells on each solar panel are connected into five diode-isolated circuits. Individual circuits are connected in series-parallel combinations.

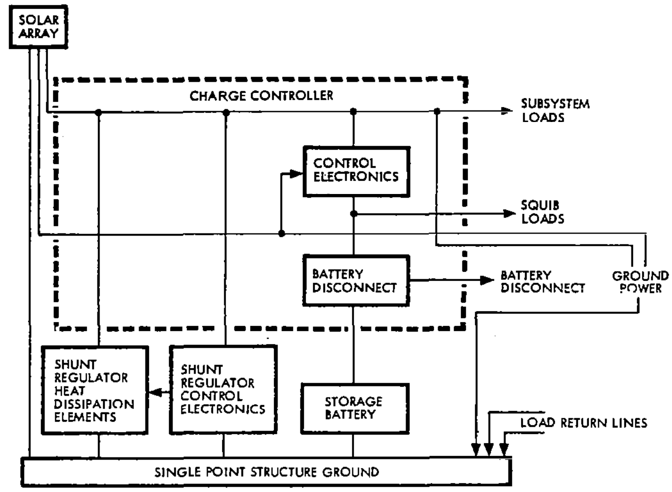


Figure 1.1-9: Power Subsystem Block Diagram

1.1.2.3 COMMUNICATIONS SUBSYSTEM

The Lunar Orbiter I communications system is an S-band system capable of transmitting telemetry data, video data, doppler and ranging information, and receive and decode command messages and interrogations.

Major components of the communication subsystem are the transponder, command decoder, multiplexer encoder, modulation selector, telemetry sensors, traveling-wave-tube amplifier, and two antennas.

The transponder is a coherent receiver-transmitter that provides demodulation of the receiver carrier and modulation of the transmitter carrier. Associated with the transponder are a directional coupler and diplexer that enable the low-gain antenna to receive and transmit simultaneously and provide for diversion fo approximately 30 mw of power to drive the TWTA. The transponder consists of an automatic phase tracking receiver with a nominal receiving frequency of 2116.38 MHz, a narrow-band phase detector for the command subcarriers, a wide-band phase detector for the range code, a phase modulator, and a 0.5-watt transmitter with a nominal frequency of 2298.33 MHz. In the two-way phase lock mode the transmitted frequency is coherently locked to the received frequency in the ratio of 240 to 221.

The command decoder is the command data interface between the transponder receiver and the flight programmer. Spacecraft commands in the form of digital messages modulated by the DSIF transmitter on the command subcarriers are decoded by the command decoder. To verify that the commands have been properly decoded, the decoded command is temporarily stored in a shift register, and retransmitted to the DSIF by the telemetry system. After validating the proper decoding of the command, appropriate signals are transmitted to the spacecraft to shift the stored command into the flight programmer for execution at the proper time. The command decoder also contains the unique binary address of the spacecraft. This feature makes possible the transmission of commands to each spacecraft when more than one spacecraft orbits the Moon simultaneously.

The PCM multiplexer encoder is the central device that puts performance telemetry data into the desired format for transmission. The analog section of the multiplexer sequentially samples 77 inputs at the rate of one sample per frame; and, in addition, samples one channel at eight times per frame. The output of these 85 data samples is converted from analog to a digital word form by an 8-stage counter. The output of the counter, an 8-bit binary word together with a 9th (complimentary to 8th bit) bit, is supplied to the digital multiplexer. In addition to the analog signals, the multiplexer also combines the 20-bit flight programmer words, the 133 one-bit discretes, and the 4-bit spacecraft identification code into 9-bit parallel output words.

The modulation selector mixes and conditions the performance information received from the subsystems for transmission to the Earth. The photo video base band information and the 50-bit-per-second performance telemetry base band information from the encoder are mixed and input into the transponder for transmission. The selector receives control signals from the flight programmer to determine the modulation mode to be used, as shown in Figure 1.1-10.

The telemetry system consists of signal monitors within the various spacecraft subsystems. The normal telemetry data channels include such information as temperatures, pressures, voltages, currents, and error signals. Special instrumentation includes 20 micrometeoroid detectors located on the tank deck periphery. These half-cylinder-shaped detectors are pressurized with helium gas. A rupture of the shell by a micrometeoroid releases the gas pressure, thus activating a microswitch that provides the input signal to the telemetry system. Radiation dosage measurement, in the form of two scintillation counter dosimeters and the associated logic, are mounted in the photo subsystem area. The total radiation dosage detected by the system is measured by accumulating the current in the integrating circuit portion of the signal conditioning circuit. This instrumentation provides radiation data for two purposes: (1) to indicate the level of radiation to which the unexposed film was subjected in sufficient time so that action can be taken to recover as much of the unused film as possible before radiation damage would result; (2) the data obtained at this point can be

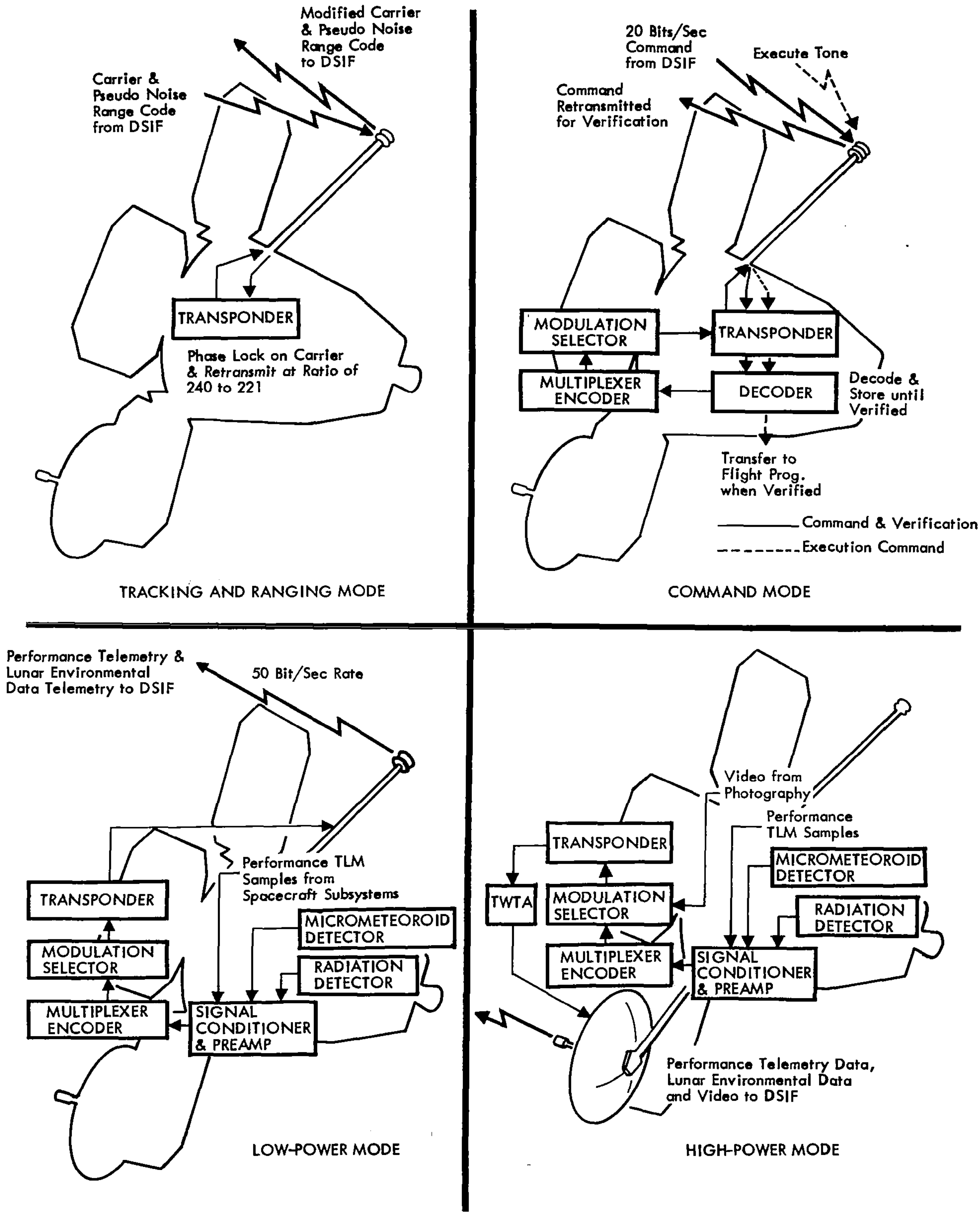


Figure 1.1-10: Communications Modes

extrapolated to the exterior of the spacecraft, thus providing information on the radiation levels of the lunar environment. To prevent damage, the less sensitive of radiation detectors was not turned on until the spacecraft had passed through the Van Allen belt.

The traveling-wave-tube amplifier (TWTA) consists of a traveling wave tube, a bandpass filter, and the required power supplies. This equipment is used only to transmit the wide-band video data and telemetry (Mode 2) during photo readout. It has a power output of 10 watts at a frequency of 2295 MHz from an input rf signal level of approximately 20 to 30 mw. All of the necessary controls and sequencing for warmup of the traveling wave tube are self-contained.

The spacecraft employs two antennas, one of which provides a strongly directional pattern and the other is as nearly omni-directional as practical. The low-gain antenna is a biconical-discone slot-fed antenna mounted at the end of an 82-inch boom. This antenna radiates an elliptically polarized wave and is fed by the boom, which acts both as a low-loss coaxial transmission line and as a physical support. The antenna is oriented to put the maximum radiation approximately in the ecliptic plane. All spacecraft reception is via this antenna.

The high-gain antenna is a 36-inch parabolic reflector that provides at least 20.5 db of gain within ±5 degrees of the antenna axis. The radiated output is right-hand circularly polarized. The antenna dish is mounted on a boom and is rotatable about the boom axis. This boom is adjustable prior to launch to a nominal position for the orbital relationships between the spacecraft, the Moon, and the Earth during the period of the photographic mission. The antenna position controller rotates the antenna boom upon command in increments of 1 degree in either direction. Thus, it is possible, during the photo video readout period, to orient the antenna to give maximum signal strength at the receiving site. This antenna is normally used for the transmission of video data (high-power mode).

1.1.2.4 ATTITUDE CONTROL SUBSYSTEM

Execution of all spacecraft events and maneuvers is controlled by or through the attitude control subsystem (ACS), Figure 1.1-11. Exacting requirements have been placed on this subsystem to precisely position the spacecraft for picture taking, velocity changes, or orbit transfers.

The operating conditions of the attitude control system are divided into the following basic modes.

Celestial Hold The basic references in this mode are the Sun and Canopus; the gyro systems operate as rate sensors. This mode is used during normal cruise operations and as the initial conditions for all commanded attitude changes.

Inertial Hold The basic references in this mode are the three gyros operating as attitude-angle sensors. This mode is used during all attitude and velocity change maneuvers, and whenever the celestial reference system is occulted.

Maneuver Mode - In this mode the spacecraft acquires the commanded angular rate about a single axis. The remaining two gyros may be held in the inertial hold mode.

Engine On, Inertial Hold - This mode is similar to the previously defined inertial hold mode except that the attitude of the spacecraft during the velocity change is accomplished by feedback control to the engine actuators.

Limit Cycling - The spacecraft is commanded to maintain a position within ±0.2 degree for all photographic and velocity control maneuvers or whenever commanded. (The normal dead band is ±2 degrees.)

The on-board digital programmer directs the spacecraft activities by either stored program command or real-time command. The programmer contains a 2,688-bit magnetic core memory and 600 microelectronic logic circuits. The unit provides spacecraft time, performs computations and comparisons, and controls 120 spacecraft functions through real-time, stored, and automatic program modes. The information stored in the 128-word memory is completely accessible at all times through appropriate programming instructions. A capability of providing up to 16 hours of stored information and instructions for the spacecraft is inherent in the flight programmer design. This feature provides a high degree of reliability of executing commands without redundant equipment.

The inertial reference unit (IRU) maintains the spacecraft attitude. Three gyros provide appropriate rate or angular deviation information to maintain proper attitude and position control. A linear accelerometer provides velocity change information to the flight programmer during any firing of the velocity control engine. Velocity changes of up to 3,000 feet per second (in increments of 0.1 foot per second) and attitude maneuvers of 360 degrees (in increments of 0.011 degree) are attainable by the attitude control subsystem from these inputs.

Sun sensors are located in five positions about the spacecraft to provide spherical coverage and ensure Sun acquisition and lockon and the resulting alignment of the solar panels. Yaw and pitch error signals are also generated as inputs to the attitude control system when any angular deviation from the spacecraft-Sunline exists. A celestial reference line for the spacecraft roll axis is established by identifying the celestial body that the star tracker acquires, locks on, and tracks. Under normal conditions the star, Canopus, is used for this purpose; however, any known celestial body of suitable brightness about the spacecraft roll axis can be used to satisfy this function.

The closed-loop electronics (CLE) provides the switching and electronic controls for the reaction control thrusters and positioning of the velocity control engine actuators. Attitude maneuver and control is maintained by the controlled ejection of nitrogen gas through the cold-gas thrusters. Two 0.05-pound thrusters are used for pitch maneuver control and two for yaw control. Four 0.028-pound thrusters in two couples are available for roll control. The pitch and yaw thrusters are operated singly; however, the roll thrusters are operated in pairs. The minimum duration of a thruster operation is 11 milliseconds. All of these thrusters are mounted on the periphery of the engine deck. During a velocity control maneuver, gimbaling of the velocity control engine is used to maintain stable orientation of the spacecraft.

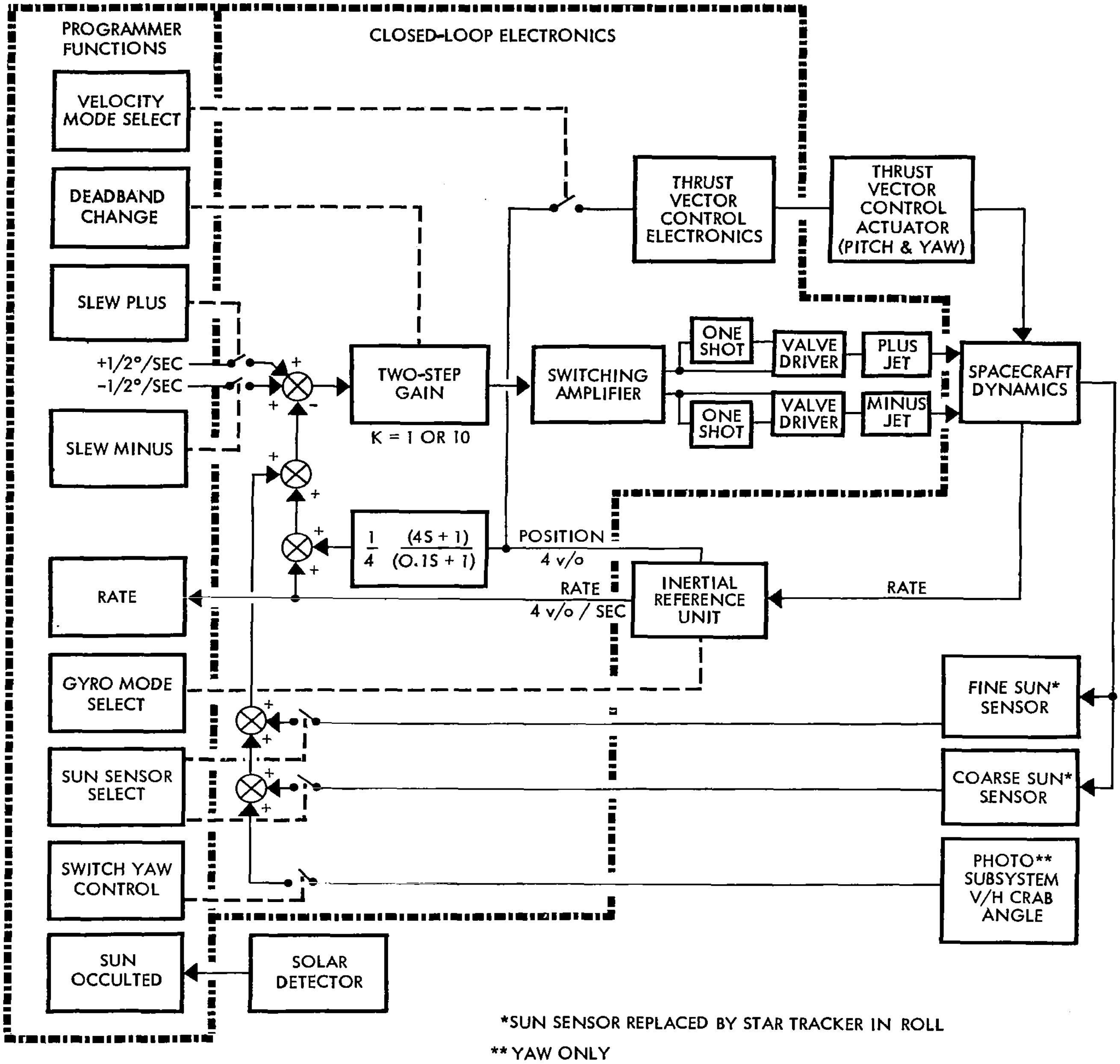


Figure 1.1-11: ACS Functional Block Diagram

1.1.2.5 VELOCITY CONTROL SUBSYSTEM

The velocity control subsystem provides the velocity change capability required for midcourse correction, lunar orbit injection, and orbit adjustment as required. The spacecraft includes a 100-pound-thrust gimbaled liquid rocket engine. The propulsion system uses a bipropellant liquid rocket engine that employs nitrogen tetroxide as the oxidizer and Aerozine-50 (a 50-50 mixture by weight of hydrazine and unsymmetrical dimethylhydrazine, UDMH). The propellants are expelled from the tanks by pressurized nitrogen acting against teflon expulsion bladders. The propellants are hypergolic so no ignition system is required. The radiation-cooled engine was developed by Marquardt.

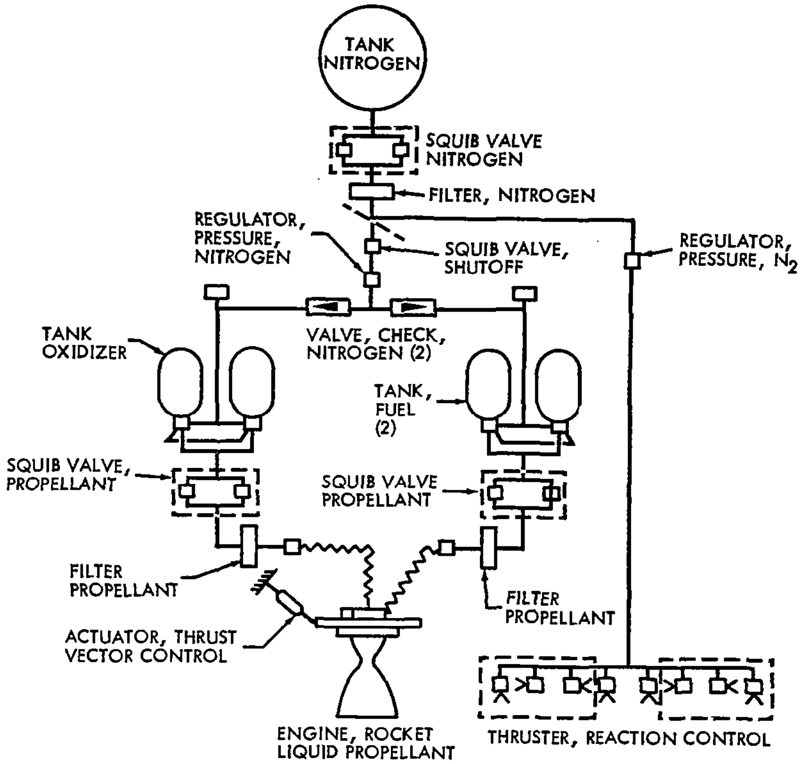


Figure 1.1-12: Velocity and Reaction Control Subsystem

The engine is mounted on two axis gimbals with electrical-mechanical actuators providing thrust directional control during engine operations. A central nitrogen storage tank provides gas required to expel the propellents in the velocity control system and the gas for the attitude control thrusters. However, separate regulators are employed. Figure 1.1-12 identifies the components of the subsystem and shows how they are connected. The 262.5 pounds of usable propellants provide a nominal velocity change capability of 1025 meters per second at a oxidizer-to-fuel ratio of 2.0.

1.1.2.6 STRUCTURES AND MECHANISMS

The Lunar Orbiter spacecraft structure includes three decks and their supporting structure. The equipment mounting deck includes a structural ring around the perimeter of a stiffened plate. Mounted on this deck are the photo subsystem and the majority of the spacecraft electrical components. The tank deck is a machined ring v-shaped in cross section closed out with a flat sheet. Fuel, oxidizer, and nitrogen tanks are mounted on this deck. The 20 micrometeoroid detectors are located on the periphery of the ring. The engine deck is a beam-stiffened plate that supports the velocity control engine, its control actuators, the reaction control thrusters, and the heat shield which protects the propellant tanks during engine operation.

Prior to deployment, the low-and high-gain antennas are positioned and locked along the edges of these three decks. The four solar panels are mounted directly under the equipment mounting deck. Electrically fired squibs unlock the antennas and the solar panels at the appropriate time to permit them to be deployed into the flight attitude.

Thermal control of the spacecraft is passively maintained. An insolated thermal barrier, highly reflective on both the interior and exterior surfaces, encloses the spacecraft structure except for the Sun-oriented equipment mounting deck and the insulated heat shield on the engine deck. The objective is to maintain spacecraft temperature within the thermal barriers within a nominal range of 35 to 85°F. The equipment mounting deck exterior surface is painted with a silicone-based paint which has a zinc-oxide pigment selected to achieve the desired heat balance. This paint has the properties of high emissivity in the infrared region (for dissipation of spacecraft heat) and low absorption at the wave lengths which contain most of the Sun's emitted heat.

A camera thermal door protects the photo subsystem lenses from heat loss except during photographic periods. Immediately prior to each photographic sequence, the door is opened to permit photography.

1.1.3 LAUNCH VEHICLE DESCRIPTION

The Atlas-Agena is a two and a half stage vehicle. All engines of the SLV-3 Atlas are ignited and stabilized prior to commitment to launch. The Agena engine is ignited twice, first to accelerate the Agena-Lunar Orbiter combination to the velocity required to achieve a circular Earth orbit, and secondly to accelerate the spacecraft to the required injection velocity for the cislunar trajectory. The SLV-3 profile and general configuration are shown in Figure 1.1-13.

Two interconnected subsystems are used for Atlas guidance and control - - the flight control (autopilot) and radio guidance subsystems. Basic units of the flight con-

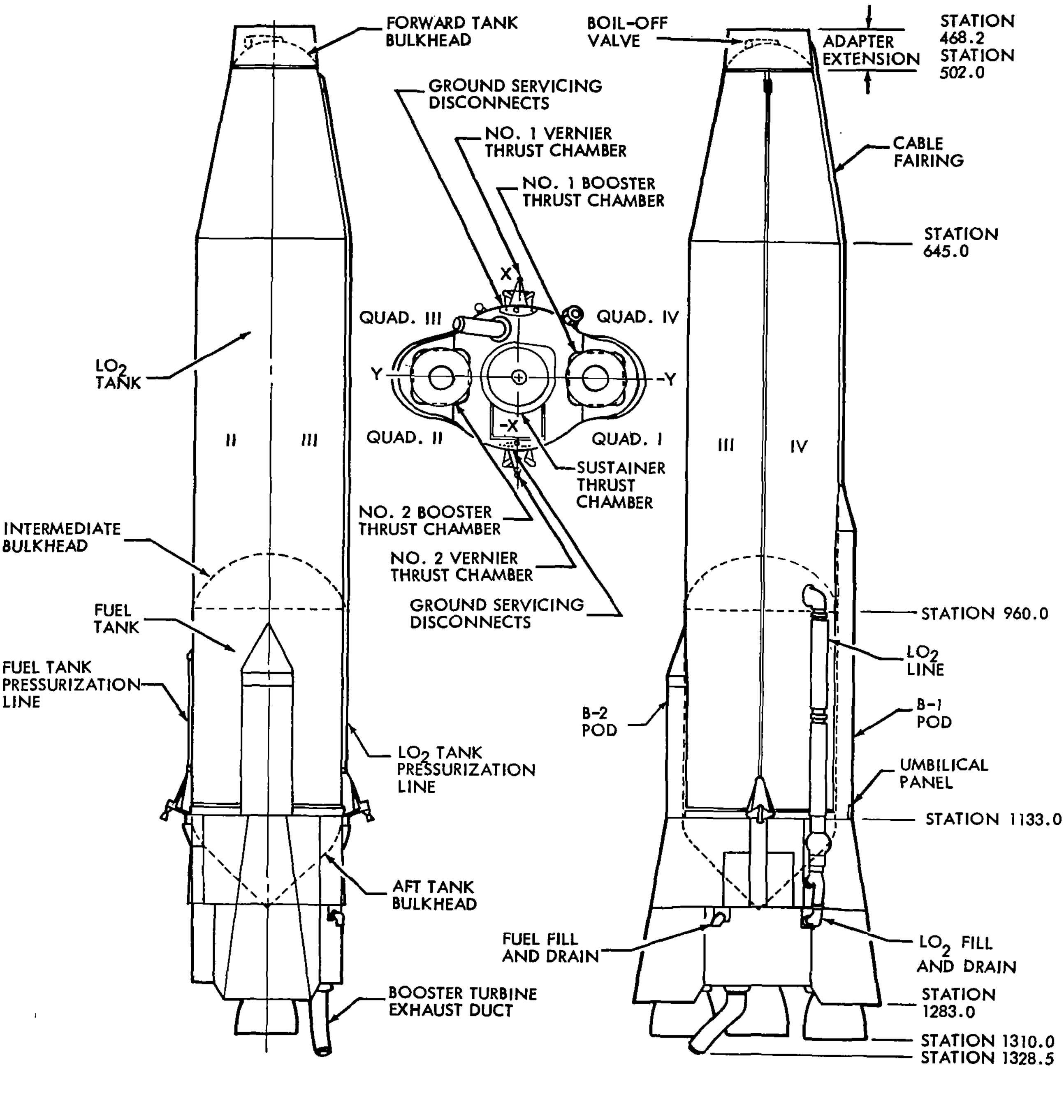


Figure 1.1-13: Atlas SLV-3 Configuration

trol subsystem are the flight programmer, gyro package, servo control electronics, and hydraulic controllers.

The main ground elements of the radio guidance subsystem are the monopulse X-band position radar, continuous-wave X-band doppler radar (used to measure velocity), and a Burroughs computer. The airborne unit is a General Electric Mod III-G guidance package which includes a rate beacon, pulse command beacon, and decoder. The radio guidance subsystem interfaces with the flight control (autopilot) subsystem to complete the entire guidance and control loop.

The upper stage is an Agena space booster with the spacecraft adapter included. It is adapted for use in the Lunar Orbiter mission by inclusion of optional and "program peculiar" equipment as depicted in Figure 1.1-14.

The Agena Type V telemetry system includes an E-slot VHF antenna, a 10-watt transmitter, and individual voltage-controlled oscillators for channels 5 through 18 and channel F. Channels 12 and 13 are used to provide vibrational data in the spacecraft during the launch phase. Channel F contains the complete spacecraft telemetry bit stream during the launch phase.

1.1.4 OPERATIONAL GROUND SYSTEMS

1.1.4.1 AIR FORCE EASTERN TEST RANGE (AFETR)

The AFETR provides receiving, inspection, assembly, and checkout for the space vehicle; office and storage space to meet individual user's requirements; and the launch complex.

Data acquisition and tracking, from space vehicle liftoff through the Agena yaw-around and retromaneuver, are provided by ETR instrumentation facilities. A summary of ETR stations, instrumentation available at each, and general use category are compiled in Table 1.1-1.

Radar tracking data provides trajectory information for determining the spacecraft position and maneuver commands. These data are gathered by the various tracking stations, transmitted to Cape Kennedy, and fed into computers, which compute trajectories and velocity.

AFETR stations at Antigua, Grand Turk, and GBI are connected to the mainland by a submarine coaxial cable, which may be used for voice, teletype, or instrumentation data transmission. Twelve individual circuits are available for transmission either uprange or downrange; each of these channels will pass a 0.3- to 3.0-kc signal. Three of the channels may be coupled to provide a single 1- to 40.8-kc wide-band data (TLM) channel, which will transmit uprange only. In the voice mode only, any or all of the 12 channels may be placed in a special mode that permits two information signals to be put on each channel, thus providing a maximum capability of 24 voice channels.

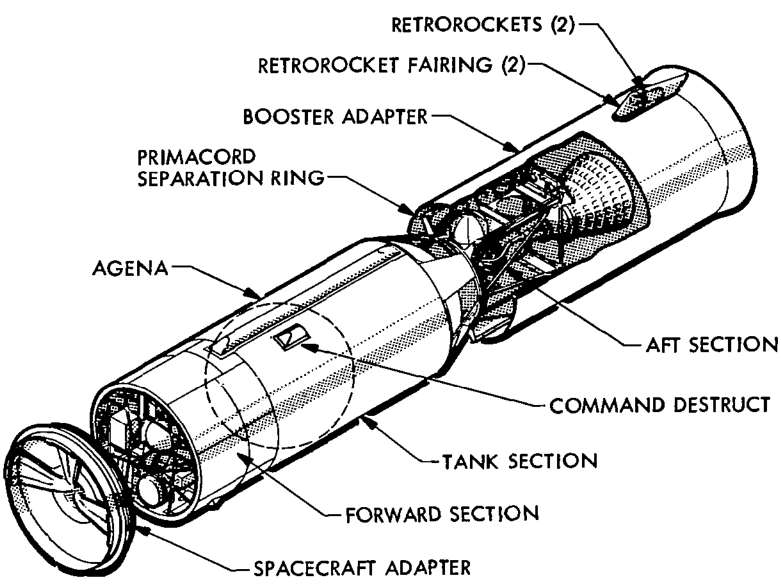


Figure 1.1-14: Agena Basic Configuration

	PHOTOGRAPHIC CAMERAS						TELEMETRY		RADAR				RANGE SAFETY				
	BALLISTIC	CINETHEODOLITE	FIXED CAMERA	PAD CAMERA	IGOR	ROTI	PAM/FM/FM(VHF)	FM/PM (S BAND)	FPS-16 C BAND	TPQ-18	A/N-FPQ-6	MPS-25	SKY SCREEN	ELSS E	FPS-8 L BAND	IBM 7094 COMPUTER	COMMAND DESTRUCT
CAPE KENNEDY STATION 1	●	●	●	●	●		● (2)	●	●				●	●	●	●	●
WILLIAMS POINT					●												
COCOA BEACH		●				●											
PATRICK AFB STATION 0		●			●						●						
MELBOURNE BEACH						●											
VERO BEACH						●											
GRAND BAHAMA STATION 3	●						● (2)	●	●								●
MERRITT ISLAND STATION 19											●						
GRAND TURK STATION 7										●							●
ANTIGUA STATION 91							● (2)	●			●						●
ASCENSION STATION 12							●	●	● OR	●							
BERMUDA									●								
PRETORIA STATION 13							●	●	●			●					
SHIPS							●	●	●								
AIRCRAFT							●	●									

Table 1.1-1: Instrumentation Summary (AFETR)

Two ground systems, the Mod III radio guidance and tracking, and the Mod III guidance computer, provide guidance signals to the Mod III airborne equipment in the Atlas during portions of Atlas-powered flight.

Single-sideband high-frequency radio links Pretoria (South Africa), Ascension, and Antigua with Cape Kennedy. Each radio link can handle voice and teletype or high-bit rate data.

The Mod III radio guidance and tracking subsystem includes position measuring, rate measuring, and flight data recording equipment. An X-band tracking radar in conjunction with the airborne pulse beacon provides space vehicle position data to the ground computer. Position data consists of azimuth and elevation angles in digital form and slant range as a time delay between a ranging pulse and a reference pulse. Radial and lateral velocity vectors of the space vehicle are provided to the computer by the X-band rate measuring radar and the airborne rate beacon. The flight data recording equipment provides the means for recording the performance of the ground-based Mod III radio guidance and tracking equipment.

The Mod III guidance computer is a Burroughs A-1 computer. It accepts space vehicle position and velocity data from the X-band position and rate radars and, in accordance with guidance equations and target coordinates, computes the steering and discrete commands that are transmitted to the Atlas portion of the space vehicle via the X-band radar.

1.1.4.2 SPACE FLIGHT OPERATIONS FACILITY (SFOF)

The SFOF at Pasadena, California, is the command and control center for Lunar Orbiter operations during the entire mission. Flight control was delegated to the Lewis Research Center at Cape Kennedy from launch to cislunar injection. After spacecraft separation, operational control was handed over the SFOF and DSN for initial acquisition by the Deep Space Station at Woomera, Australia. Mission control was supported by a staff of mission advisors and by spacecraft performance analysis and command (SPAC) and flight path analysis and command (FPAC) specialist teams. The communications, data processing, display, and support capabilities required by these technical groups to perform analysis, evaluation, and interpret spacecraft data are provided within the facilities of the SFOF. Additional equipment and software procedures peculiar to the Lunar Orbiter program are integrated into the existing facilities to provide the overall data support required.

Mission-independent functions in the SFOF required to support the Lunar Orbiter mission are divided into four major areas: (1) data processing systems (DPS), including the computers, telemetry processing equipment, data processing control and display equipment, and programming equipment. The IBM 7044 and 7094 computers work together to process the tracking and telemetry data in the proper format for recording and displaying. (2) Communications equipment, including incoming and outgoing voice and teletype lines, high-speed data lines, and the internal communications systems within the SFOF. External connections from the communications center are used to contact the AFETR and the Deep Space Stations that support the mission. (3) Data display systems supporting the mission and including such equipment as plotters, high-speed data printers, visual displays, monitor panels, and status boards. (4) Reproduction and distribution indexing of all of the data received and generated within the SFOF. A team of spacecraft and mission specialists supports the operation on a 24-hour basis. In addition to their function of evaluating the performance data in real time to verify the operational status of the spacecraft, these teams coordinate their efforts at determining the commands to be sent to the spacecraft and the times in which they must be executed to satisfactorily accomplish the mission objectives.

1.1.4.3 DEEP SPACE STATIONS (DSS)

The DSS provide command, control, tracking, and data acquisition to support the Lunar Orbiter mission. Continuous communications coverage for the spacecraft is provided during the entire mission by the stations at Goldstone, California (DSIF-12); Woomera, Australia (DSIF-41); and Madrid, Spain (DSIF-61). The station at Cape Kennedy, Florida (DSIF-71) supports the program during the checkout, launch, and boost phase, and the station in Johannesburg, South Africa (DSIF-51) provides tracking data during the boost phase. The equipments at each of the primary sites that were used to support the Lunar Orbiter program are shown in Table 1.1-2.

ANTENNA CHARACTERISTICS

TRACKING:

TYPE	-	85-FOOT PARABOLA
FEED	-	SIMULTANEOUS - LOBING CASSEGRAIN
BEAMWIDTH	-	0.36-DEGREE RECEIVE; 0.45-DEGREE TRANSMIT
POLARIZATION	-	RIGHT-HAND CIRCULAR
GAIN	-	51-db TRANSMIT; 53-db RECEIVE

ACQUISITION:

TYPE	-	2 BY 2 FOOT APERTURE
FEED	-	SIMULTANEOUS - LOBING WAVEGUIDE HORN
BEAMWIDTH	-	16 DEGREES
POLARIZATION	-	RIGHT-HAND CIRCULAR
GAIN	-	20-db TRANSMIT; 21-db RECEIVE

TRANSMITTER CHARACTERISTICS

TUNING RANGE	-	2110 TO 2120 MHz
FREQUENCY	-	2116.38 MHz
POWER	-	200 w TO 10 kw TO ANTENNA
STANDARD	-	RUBIDIUM THROUGH SYNTHESIZER
MODULATION	-	PHASE MODULATION (PM)

RECEIVER CHARACTERISTICS

SIGNAL CHARACTERISTICS:

TELEMETRY	-	PCM/PSK/PM (DIFFERENTIAL DIPHASE)
RANGING	-	PM (PSEUDONOISE RANGE CODE)
VIDEO	-	AM/PM (VESTIGIAL SIDEBAND)
FREQUENCY	-	2298.33 MHz
DOPPLER	-	ONE-WAY
	-	TWO-WAY COHERENT
	-	THREE-WAY TWO-STATION NON-COHERENT
	-	THREE-WAY TWO-STATION COHERENT

RECORDERS

TELEMETRY	-	2 FR-1400, 7-CHANNEL MAGNETIC TAPES
TRACKING	-	2-5 LEVEL TELETYPE PAPER TAPES
VIDEO	-	FR-900 MAGNETIC TAPES

TABLE 1.1-2: DSN Equipment Summary

In addition to the equipment normally installed at the DSS, the Lunar Orbiter program provides additional equipment to interface in the areas of command preparation for the spacecraft; receiving, decoding, and decommunicating the telemetry data; and receiving and reconstructing the photo video data. Supporting computer programs were developed for the SDS920 computer at each site to properly format and transmit the telemetry data to the SFOF via the communications system and also to process the Lunar Orbiter-peculiar commands received from the SFOF for transmission to the spacecraft.

The block diagram of the data flow through the DSS is shown in Figure 1.1-15. Sufficient data is displayed and available for the Lunar Orbiter technical personnel to evaluate the spacecraft performance data and to generate any necessary commands for transmission to the spacecraft and assume control in the event that communications between the DSS and the SFOF were interrupted. Tracking data from each of the Deep Space Stations can be provided to the other stations to facilitate acquisition and orderly transfer of tracking responsibility. The spacecraft transponder can be tracked by two stations simultaneously, thus providing more accurate tracking data. Spacecraft tracking and ranging is accomplished by using the existing DSS equipment.

Command transmission capability is also provided by the existing DSS equipment. Lunar Orbiter mission-dependent equipment was also developed and installed at the sites for the transmission and reception of data to and from the spacecraft.

The spacecraft performance telemetry data is demodulated by Lunar Orbiter equipment and converted into a format for inputting to the SDS 920 computer at each site, where the bit stream is further formatted into the proper mode for transmission to the SFOF by either the high-speed data lines or by teletype wire communications. The spacecraft commands received from the SFOF are verified by an automatic routine by the SDS 920 computer. They are relayed to the Lunar Orbiter-peculiar equipment where they are properly formatted for transmission to the spacecraft by the DSS transmitter. The decoded commands received by the spacecraft are transmitted to the DSS in the telemetry bit stream. After verification of the proper command coding, the execute tone is transmitted, completing the command sequence. Verification is normally performed at both the DSS and SFOF.

1.1.4.4 PHOTO RECONSTRUCTION AND REASSEMBLY

The photo video amplifier output is used to modulate the 310-kc subcarrier of the spacecraft S-band transponder output by employing vestigial sideband and suppressed subcarrier techniques to improve the video and telemetry signals. At the DSS, the demodulated signals are recorded on a video tape recorder at the output of the 10-MHz I.F. A video demodulator separates the video and telemetry data and inputs the video signal to the ground reconstruction system. Figure 1.1-16 shows the entire photo process in schematic form.

Ground reconstruction electronics (GRE) convert the video signal to variations of light intensity on a kinescope tube. A 35-mm recording camera records these signals on SO-349 film as a positive photographic image. A magnification of 7.2 produces an image 0.72 by 16.76 inches on the GRE film for each framelet read out by the spacecraft. The original spacecraft photograph can be reassembled by manual or automatic methods by placing the framelets side by side.

Manual reassembly required the accurate trimming of the GRE film along the fiducial marks before the film could be reassembled. After completing the reassembly the positive transparency was copied photographically. This time-consuming operation was used to support the mission advisors and mission control requirements and for early public releases.

Automatic reassembly required the use of the reassembly printer at Eastman Kodak, Rochester, N. Y.. The operation of this printer is shown schematically in Figure 1.1-17. A reassembled subframe consists of 14 GRE framelets printed on 9.5-inch film. The resultant 9-by 14-inch image is reduced by a factor of 0.8927 with respect to the GRE film.

In the process of reassembling an entire picture (which requires more than one 9.5-inch frame), the last two framelets of one frame are printed as the first two of the next frame, for registration purposes. Therefore, three 9.5-inch subframes are required for the 80-mm frame. Similarily, a high-resolution frame requires eight subframes. The total magnification factor of the com-

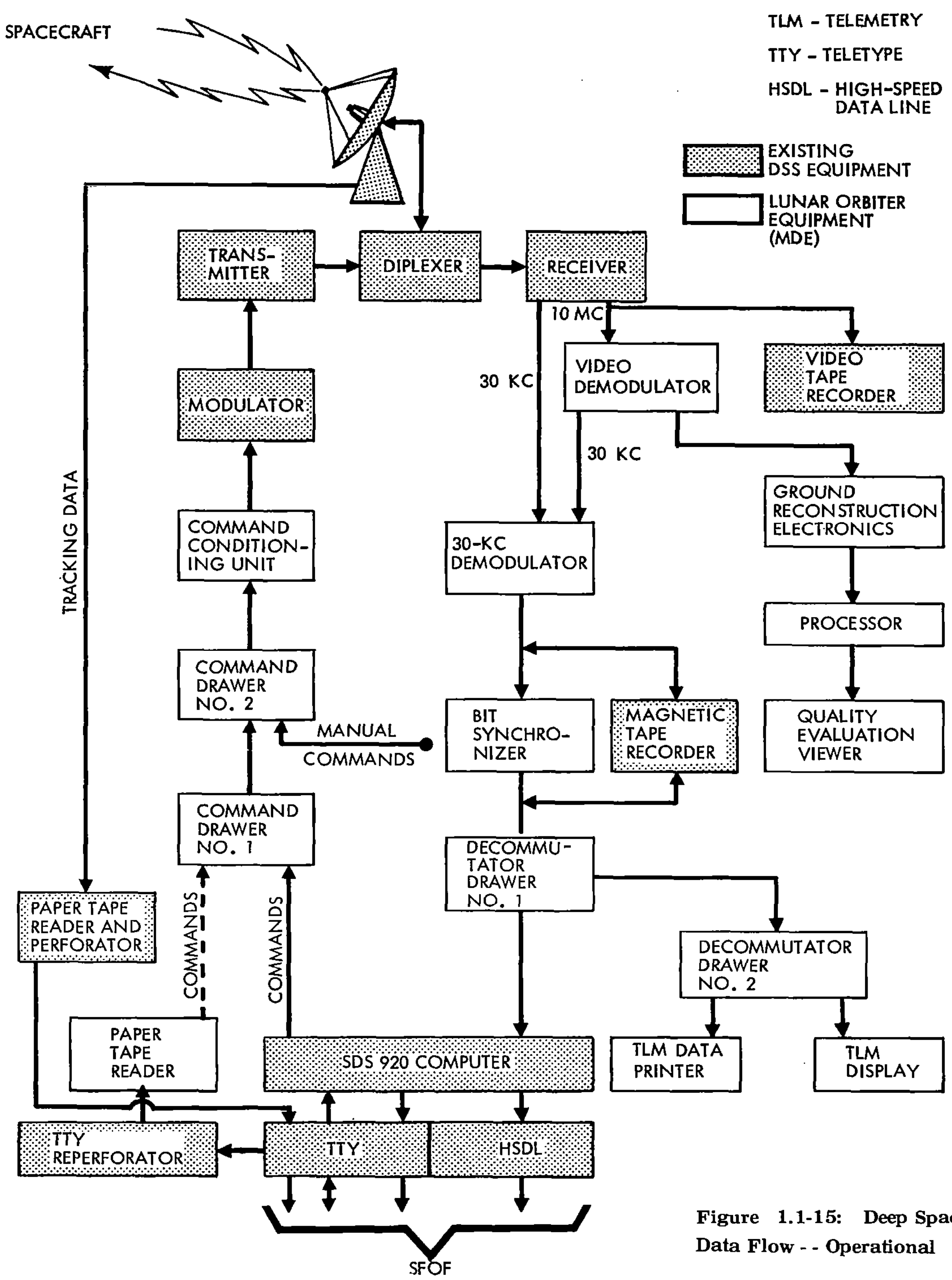


Figure 1.1-15: Deep Space Station Data Flow - - Operational

bined GRS reassembly operation is 7.20 (0.893) = 6.43, with respect to the original 70-mm spacecraft film. The reassembly printer adds a data block at the bottom of each reassembled photograph. This data positively identifies the photograph with respect to exposure number, readout sequence, and reassembly identification. Supporting data required to evaluate, position, and interpret the photos is documented separately. An appropriate note is added to the titling data of the moderate-resolution reassembled photographs to compensate for the mirror image resulting from the folding mirror.

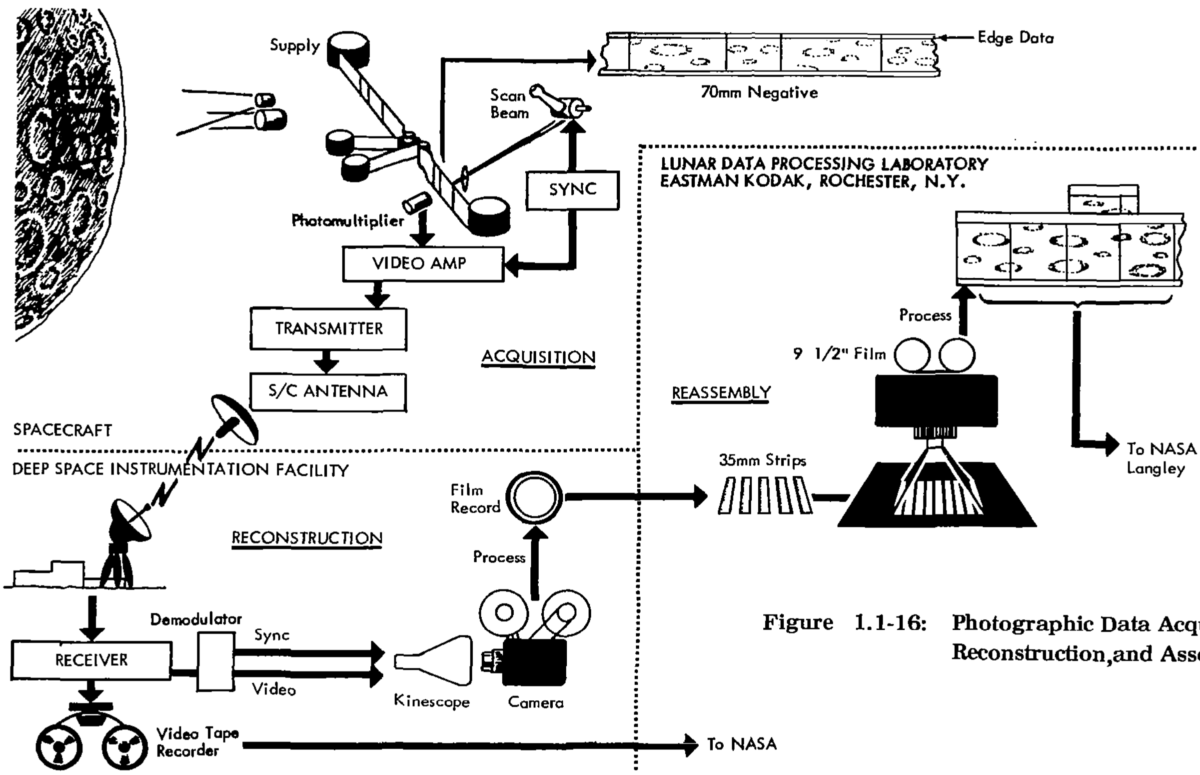


Figure 1.1-16: Photographic Data Acquisition, Reconstruction,and Assembly

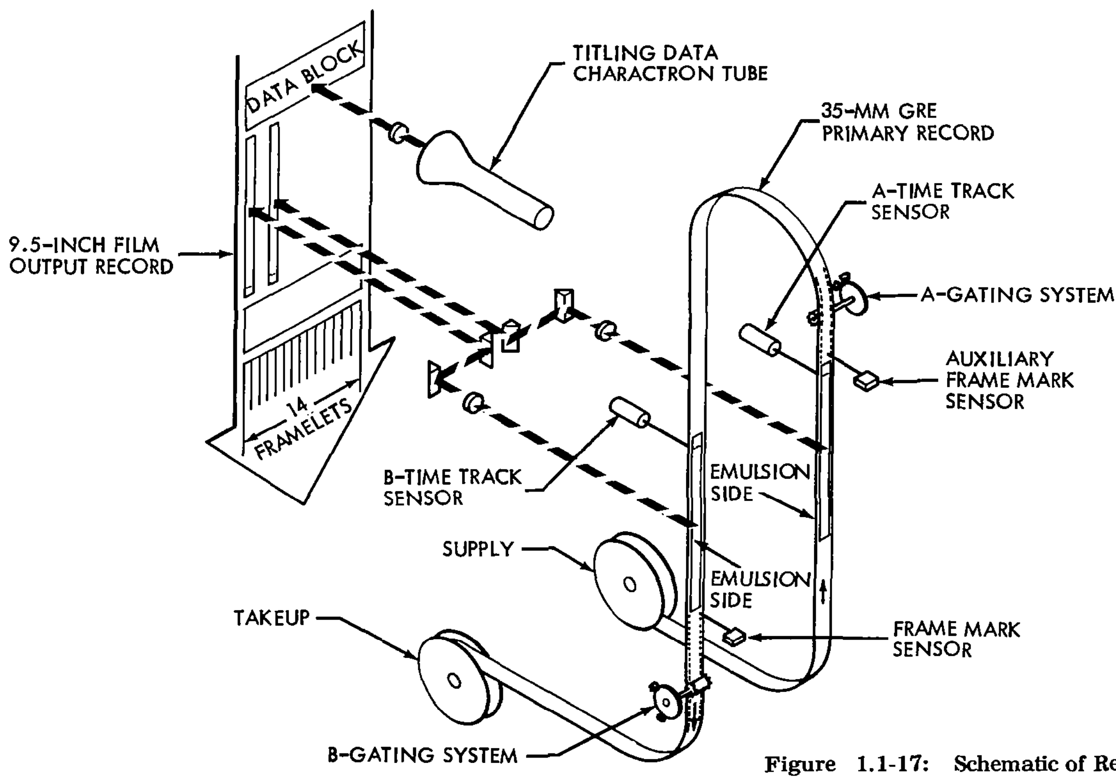


Figure 1.1-17: Schematic of Reassembly Printer

1.2 SYSTEM DEVELOPMENT AND READINESS

The Lunar Orbiter Statement of Work L-3270 to NASA Contract NASI-3800 dated March 18, 1964, established the following requirement: "The Contractor shall plan, integrate, design (including analysis and tradeoff studies), fabricate, qualify, environmentally test, establish and implement reliability and quality assurance programs for all necessary spacecraft and required equipment. Also, the Contractor shall provide launch support, flight operations support, acquire and deliver all required photographs and supporting data, and documentation for the Lunar Orbiter Project. All required work based on five launches shall be carried out in such a manner and sequence as to support a first launch capability no later than fiscal year 1966, and subsequent launches to be scheduled approximately every 75 days. The delivery of the spacecraft shall be such that two are available at ETR for each scheduled launch, except the last."

The first flight of the Lunar Orbiter I spacecraft occurred 28 months and 15 days after the start of the program. Appropriate milestones and major accomplishments in the program, from "contract go-ahead" to the launch countdown of Lunar Orbiter I, are shown in the following chart, Figure 1.2-1. To provide continuity by subject matter, and also to show the interrelation of paralled operations, the milestone dates are charted by appropriate function.

Project Management

A highly integrated management team with direct lines of communication reporting to the program manager was formed. Managers for Engineering, Launch and Flight Operations, Business Management, Materiel, Reliability, Quality Assurance and Factory Operations were established.

Personnel planning for the program was developed to retain certain key engineers and supervisors by moving them from design and system engineering functions into test and operations functions as the program progressed. As each spacecraft took form and moved into test, a team of engineers and technicians, including NASA personnel, was assigned to provide a continuity of experience, knowledge, and responsibility for all test operations through Launch and Flight Operations.

Because of the tight schedule, all design, development, and production activity required rapid and efficient transmittal of data, commitment and direction. The Prime Contractor established close liaison with subcontractors and suppliers during the complete program activity. Very close communication was established with NASA Langley personnel who consulted with and assisted the Boeing team at all levels of the organization.

A PERT network was developed to control the Lunar Orbiter program schedules. Subnets were developed for the following areas: Spacecraft design, AGE, Launch and Flight Operations, Logistics, documentation, and test. Frequent (often daily) schedule reviews were held to implement the various procurement, manufacturing, and testing activities. Periodically a major rescheduling of master program events was made to adjust for changes experienced during the course of the program.

Design

Preparation of design and test specifications for the Spacecraft, and ground equipment, based on a comprehensive list of flight and ground environmental criteria, was initiated by the contractor and approved by NASA early in the program. Parallel efforts were expended to develop quality and reliability requirements as well as an integrated test program plan. Engineering requirements and procurement specifications for the components and subsystems were prepared, the suppliers selected, and development tests initiated.

A comprehensive review program was established to give assurance to NASA and Boeing management that program progress was satisfactory. These reviews were categorized as follows: preliminary and critical design reviews; periodic program reviews; and progressive Spacecraft test reviews.

Design reviews were held at both the component and subsystem level on all deliverable hardware. Specifications were reviewed to assure consistency with mission requirements. Critical design reviews were held after preliminary design reviews and before fabrication. Each review measured the actual design against the design requirements. Program reviews were held quarterly to report program progress to the NASA. During these reviews design progress, program schedules, operational planning and reliability progress were examined.

The progressive spacecraft test reviews examined the fabrication history of the spacecraft, inspection and test records, configuration status, and spacecraft test progress. These reviews culminated in the pre-shipment review in Seattle, and the delivery and flight readiness reviews at ETR. The history of the first flight spacecraft tests and reviews is shown in Figure 1.2-2.

During the design phase, a failure mode, effect, and criticality analysis was conducted in two major steps; at the spacecraft and subsystem level and at the component level. Each effect of component failure was reflected as a cause of system failure and therefore, component failure modes were related to possible mission failure and criticality analysis was used in several applications as system development progressed;

1) To focus attention on design areas requiring additional analysis,
2) To identify critical failure areas,
3) To establish the need for redundancy,
4) To aid in definition of test and checkout procedures,
5) To provide a basis for flight operation system analysis, training of spacecraft system flight controllers and preparation of mission rules for off nominal system operation during flight.

Manufacturing Operation

The many highly specialized skills required for the spacecraft manufacturing operations were accomplished by careful selection of personnel and approximately 30 thousand hours of training, including clean room practices based on procedures developed by JPL.

The manufacturing personnel were assigned to the program early enough to become completely familiar with the drawings and processes necessary to fabricate the parts, assemblies, and test equipment. Process controls were carefully monitored and clear room techniques enforced. Special manufacturing shops within The Boeing Company were utilized as required but were closely monitored by program personnel to insure compliance with program peculiar procedures. The suppliers manufac-

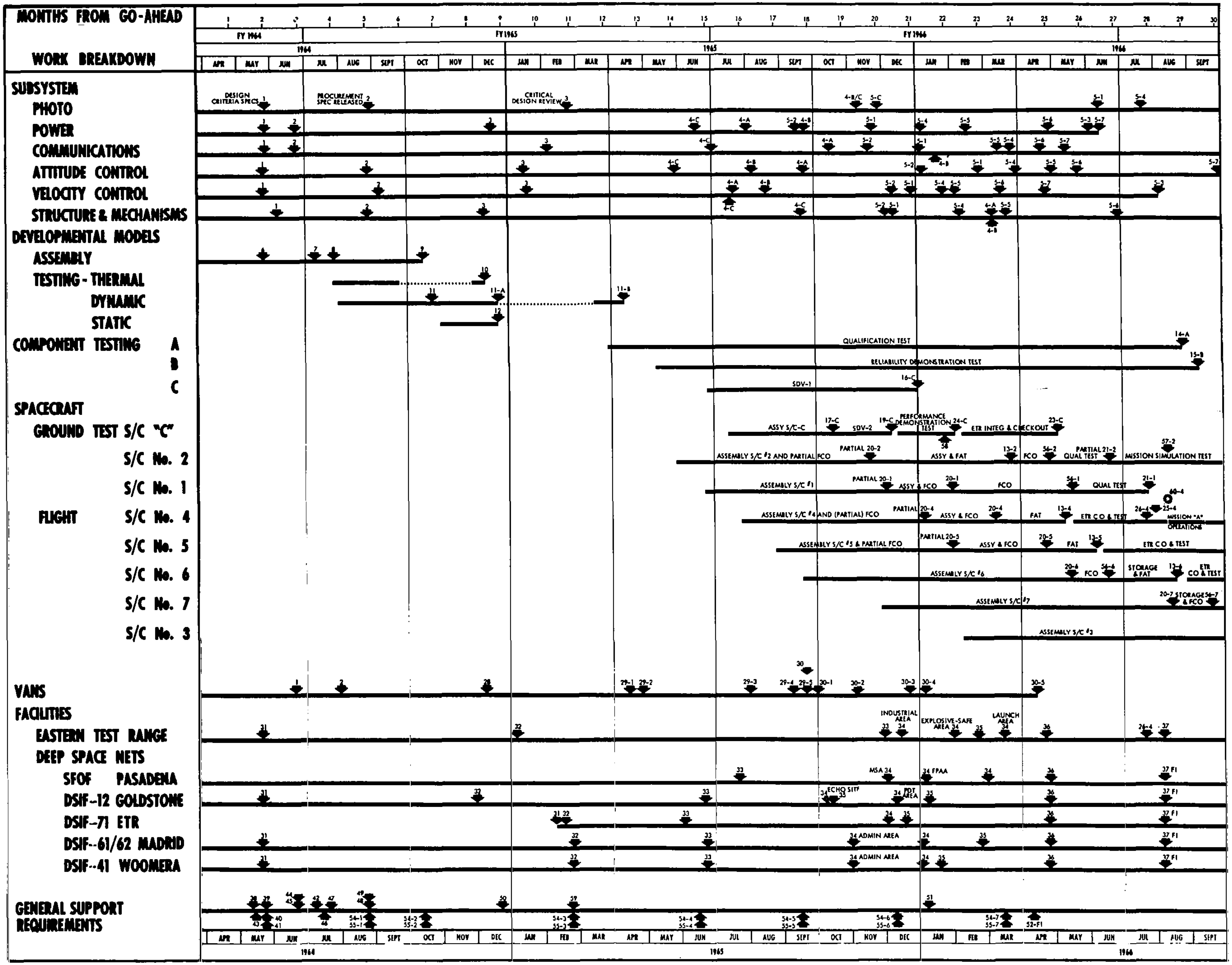


Figure 1.2-1: Lunar Orbiter Development Summary

1. DESIGN CRITERIA SPECS SUBMITTED TO NASA
2. PROCUREMENT SPECS RELEASED
3. CRITICAL DESIGN REVIEW
4. COMPONENT SETS A-B-C FLIGHT ACCEPTANCE TEST START
5. COMPONENT SETS - AVAILABLE FOR S/C ASSEMBLY
6. ENVIRONMENTAL CRITERIA SPEC TO NASA FOR APPROVAL
7. THERMAL DEVELOPMENTAL MODEL COMPLETE
8. DYNAMIC DEVELOPMENT MODEL COMPLETE
9. CLASS II MOCKUP COMPLETE
10. THERMAL DEVELOPMENT TEST COMPLETE
11. DYNAMIC DEVELOPMENT TEST COMPLETE
    II-A RETEST NEW CONFIGURATION
    II-B PHOTO S/S DYNAMIC TEST COMPLETE
12. STATIC TEST COMPLETE
13. S/C FLIGHT ACCEPTANCE TEST COMPLETE
14. QUALIFICATION TEST COMPLETE - COMPONENT SET A
15. RELIABILITY TEST COMPLETE - COMPONENT SET B
16. SYSTEMS DESIGN VERIFICATION TEST NO. 1 COMPLETE - COMPONENT SET C
17. SPACECRAFT ASSEMBLY COMPLETE
    AVAILABLE TO START SDV 2 TESTING
19. SYSTEMS DESIGN VERIFICATION TEST NO. 2 COMPLETE
20. SPACECRAFT ASSEMBLY COMPLETE
21. FCO AND QUALIFICATION TEST COMPLETE
22. MISSION SIMULATION TEST COMPLETE
23. ETR INTEGRATION AND CHECKOUT COMPLETE
24. GOLDSTONE PERFORMANCE DEMONSTRATION TEST COMPLETE
25. CONTRACTUAL DELIVERY
26. ETR CHECKOUT AND TEST COMPLETE (HANGAR "S" COMPLETION ONLY)
28. BOEING VAN DESIGN COMPLETE
29. VAN ON DOCK SEATTLE
30. VAN AVAILABLE FOR S/C TESTING
31. SUBMIT NON/SEVERABLE FACILITIES REQUIREMENTS
32. NASA PROVIDE "AS BUILT" DRAWINGS
33. FINAL FACILITY COORDINATION
34. BENEFICIAL OCCUPANCY DATA (BOD)
35. ALL AGE ON-DOCK
36. COMPATIBILITY CHECKS COMPLETE
37. MISSION SIMULATION EXERCISES COMPLETE
38. PRELIMINARY PERT SUBMITTED TO NASA
39. RELIABILITY PROGRAM PLAN TO NASA FOR APPROVAL
40. ETR PROGRAM REQUIREMENTS DOCUMENT TO NASA
41. FACILITIES SPECS TO NASA
42. QUALITY ASSURANCE PROGRAM PLAN TO NASA FOR APPROVAL
43. FIRST MONTHLY FINANCIAL PROGRESS REPORT
44. FIRST QUALIFICATION STATUS LIST TO NASA
45. FORMAL PERT SUBMITTED TO NASA
46. TECHNICAL PROGRESS REPORT (FIRST QUARTER)
47. FAMILIARIZATION MANUAL TO NASA FOR APPROVAL
48. LOGISTICS SUPPORT PLAN COMPLETE
49. FAMILIARIZATION FILM
50. DATA REDUCTION AND COLLECTION PLAN TO NASA FOR APPROVAL
51. FLIGHT OPERATIONS PROCEDURES TO NASA
52. COUNTDOWNS TO NASA FOR APPROVAL
54. INTEGRATION MEETINGS
55. TECHNICAL SUMMARY AND CONTRACTOR MGMT. STATUS MEETING
56. FUNCTIONAL CHECKOUT COMPLETE
57. FIRST MST CYCLE COMPLETE
58. PHASE I PDT COMPLETE
59. SPACECRAFT DESIGN STATUS REVIEW
60. LAUNCH DATE

NOTE:

A THE FIRST NUMBER ADJACENT TO MILESTONE DENOTES FUNCTION COMPLETION (AS 5 - COMPONENT SET MFG. COMPLETE - AVAILABLE FOR ASSEMBLY)

B THE SECOND NUMBER OR LETTER (AS 5-1 OR 5-A) DENOTES S/C UNIT NUMBER

C MILESTONES 4 AND 5 DENOTES REQUIREMENT FOR FOR LAST COMPONENT IN S/S

D FAT - FLIGHT ACCEPTANCE TEST

E S/S - SUBSYSTEM

F S/C - SPACECRAFT

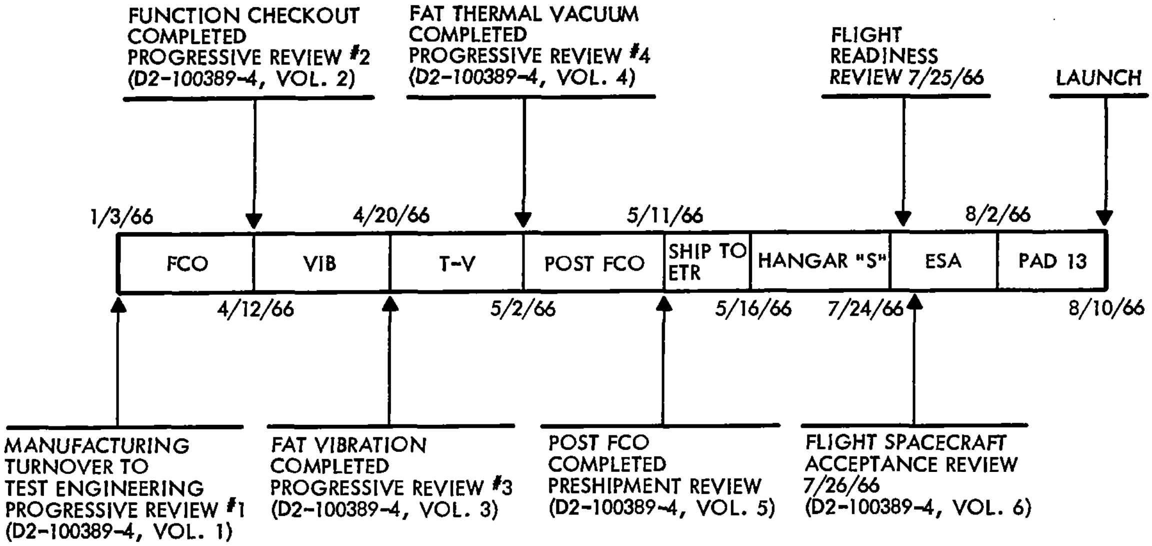


Figure 1. 2-2 Lunar Orbiter I Test and Review Schedule

turing operations were also monitored to assure conformity of quality. The turnover of manufacturing personnel throughout the program was maintained at an absolute minimum.

The selection, application and qualification of parts and materials was one of the important steps to achieve trouble-free hardware. Major considerations in the qualification of all parts were as follows:

1) Parts and materials selection was controlled through the use of an Approved Parts and Materials List.

2) All parts and materials were controlled by appropriate specifications.

3) All parts were subjected to electrical, mechanical, and environmental tests prior to acceptance and in addition, 100% preconditioning (burn in) was used as a goal.

4) All parts and material applications and processes were reviewed and approved for use in the operating environment.

5) All part failures recorded during acceptance screening or during testing of hardware were thoroughly analyzed for appropriate action or part substitution.

The scope of the review and control of qualification data was beyond that usually followed by industry. Special management emphasis was required to achieve a suitable working relationship with vendor and subcontractors to employ a satisfactory parts and materials qualification program. Usually qualification data was forwarded to Boeing for review by parts specialists and NASA representatives. In some instances, qualification data was reviewed at the supplier plants where the supplier parts and reliability groups were used for assessment of qualification data. Boeing monitored these functions through frequent review of this data at the suppliers' plants. Two methods of part and material qualification of some parts by means of satisfactory qualification of the components. The latter category was carefully considered and accepted in certain cases for Lunar Orbiter. 89% of the parts and materials were qualified prior to the first launch.

Test

The Lunar Orbiter test program was conducted in three major phases, namely:

a) Development tests.

b) Component tests.

c) Spacecraft tests.

The development tests (i.e. thermal, structure, mechanical, electrical) generated engineering knowledge and resolved technical problems which verified or improved design concepts. The degree and depth of development testing depended on prior use of the individual component or subsystem.

Component tests included a flight acceptance test (FAT) which consisted of functional tests to assure compliance with performance specifications at flight level environments. After completing these tests, components were subjected to qualification tests (set A) and reliability demonstration tests RDT (set B). The qualification tests were conducted to determine adequacy of the design and fabrication plus the effects of variation on hardware and environment. The RDT was designed to achieve a confidence level from the operational reliability of components. The RDT basically consisted of two simulated 30-day mission cycles, however, adjustments were made to some RDT tests, during the program. The mission cycle consisted of flight profile modes of operation and environmental exposure

System design verification tests (SDV-1) at the subsystem level demonstrated physical and operational compatibility between subsystem components.

Three ground test spacecraft were used during the test program to checkout the spacecraft before flight. One spacecraft was used to conduct:

1) Spacecraft level System Design Verification tests (SDV-2), which demonstrated physical and operational compatibility between subsystems,

2) Performance Demonstration Tests (PDT) at Goldstone, California, which demonstrated compatibility of the spacecraft-to-ground data link and the ground-to-spacecraft communication link, and

3) Spacecraft ETR tests which verified that the spacecraft, launch vehicle, and supporting ground equipment were compatible.

The second ground test spacecraft was used to conduct integrated spacecraft level qualification tests. This proved that the design and fabrication procedures were adequate to allow for expected variations in individual hardware items and environments and that these variations would not compromise the performance of the spacecraft. Included were vibration, thermal vacuum, shock and acoustics.

The third ground test spacecraft was used to conduct

1) A flight acceptance test (FAT)

2) A quick pump down test (QPD)

3) An electromagnetic compatibility test (EMC), and

4) A mission simulation test (MST)

The QPD test verified satisfactory spacecraft operation during the rapid depressurization phase experienced during the boost portion of the spacecraft flight profile. This test included use of the spacecraft, Agena adapter and nose fairing. The EMC tests verified that the end product did not produce electro-interference beyond specified limits and was not susceptible below specified limits. The MST verified spacecraft operation under realistic flight-environmental conditions, and provided personnel flight mission training.

Interface Integration

During the early stage of the program a technical coordination procedure was developed to insure integrated interfaces between the Spacecraft, booster, booster adapter, nose fairing, launch complex and range. Integration of these interfacing elements during the design phase was effected by means of interface meetings between NASA-Lewis, NASA-Langley, KSC-ULO, JPL, Lockheed Missiles and Space Company, The Boeing Company, and General Dynamics-Convair. Beginning in May of 1964, and extending to May of 1966, over 20 interface meetings were held to establish:

1) Operational requirements and restraints.

2) Tracking, Communications, and Control requirements.

3) Flight Operations and Communications.

4) Design Requirements and Restraints covering:

   a) Configuration definition of interface hardware,

   b) Mass and stiffness properties,

   c) Mechanical interface requirements such as alignment, matchmate, and tolerances,

   d) Electrical interface requirements,

   e) Electrical requirements,

   f) Environmental requirements

   g) Clearance requirements,

   h) Reliability

5) Quality Assurance and Test requirements.

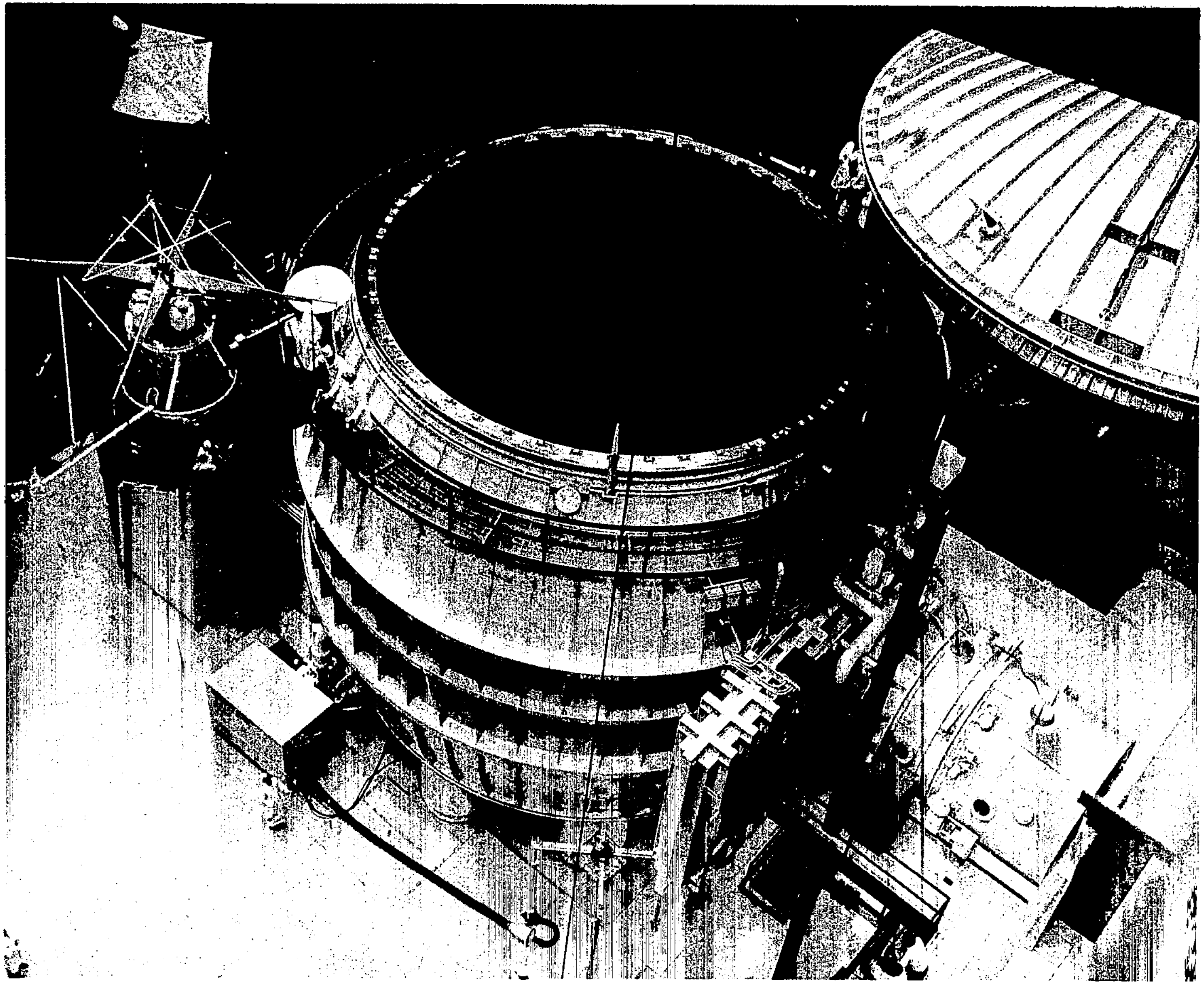
6) Launch Base Requirements and Restraints.

As the Lunar Orbiter program progressed into the test phase, NASA arranged coordination meetings between the contractor and participating government agencies to work out the launch complex acquisition requirements. A series of coordinating documents were prepared to assist in the dissemination of necessary information between cooperating agencies at ETR.

By use of the interface meetings and proper documentation reflecting the interfacing requirements, the mating of interfacing entities, launch complex acquisition, and installation of mission peculiar facilities and equipment were accomplished within the allotted schedule and quality requirements for a successful launching of the Lunar Orbiter.

Software

Many of the members of the flight operations team participated in generation of technical requirements for the operational software. These requirements were based on the information that the subsystem analysts anticipated would be required to evaluate the performance and status of their subsystems during the flight. These same people assisted the applied mathematics and computer personnel in transcribing the engineering requirements into mathematical equations and exact computer programs. As the operational software computer programs were further developed and made ready for checkout and test, the analysts, mathematicians, and computer programmers again combined their efforts to verify that the requirements were satisfactorily implemented by the appropriate programs. Thus, as the mission was flown, the personnel of the flight operations team were thoroughly familiar with all phases of the operational software, its limitations, and its adaptability to change during the flight.



Lunar Orbiter Being Transfered to 39-Foot Boeing Space Chamber (Kent, Washington)

1.3 MISSION CONDUCT

The Lunar Orbiter mission is characterized by the ambition of its objectives: i.e., the operational precision required to attain a specified lunar orbit, the extensive coverage with simultaneous high-and moderate-resolution photographs of the lunar surface, and the sustained effort to collect scientific data on the lunar environment. The accomplishment of these objectives requires the effort of many individuals, contractors, and government agencies who must jointly plan the integration of their resources so that the desired results are obtained efficiently and on time.

The ground rules for Lunar Orbiter I's mission defined that it should be a terrain-sampling mission to examine the Surveyor I site, promising future Surveyor sites, and concentrate on the Apollo zone (±5° lunar latitude and ±45° longitude). Exact locations of the sites for Lunar Orbiter I were selected by iterating between the geological considerations, the ground rules and constraints, and the bounds defined by the orbital design. Ten sites were initially chosen within the zone of interest. After the successful Surveyor I soft landing, the mission plan was modified to increase the coverage at Site I-9 to include the Surveyor landing area.

During the mission design period, there was no indication or requirement for the photo subsystem operational constraint ("film set") exposures to photograph any specific lunar sites. The mission time line analysis documentation scheduled the taking of the film-set photos near orbit perilune. As the program developed, it became more desirable and important to use these photos to obtain additional information on potential mission B site candidates as well as other terrain features from Apollo considerations and scientific interests. Sufficient planning was accomplished so that not only were many of these photos obtained, but also specific sites selected during the conduct of the mission. These sites included the farside of the Moon and the historic first photo of the Earth and the Moon's limb.

The location of the primary photographic sites for Lunar Orbiter I within the Apollo zone are shown in Figure 1.3-1. One additional primary site (I-0) was located out side the Apollo zone at 90° east longitude and the lunar equator.

The Lunar Orbiter mission started with the arrival of the spacecraft at ETR, where it was assembled, tested, and readied for launch. The early planning included the dissemination of information to the launch agency for proper programming of the Atlas-Agena system for the projected launch days. The activities at AFETR of the Atlas, Agena, and Lunar Orbiter spacecraft were integrated so that all systems were properly checked out to support the scheduled launch date. Lunar illumination requirements, Earth-Moon geometry, and Sun-Moon relationships required that these plans be geared to utilize the available launch windows.

Control of the launch was delegated to the Lewis Research Center, supported by the downrange stations and appropriate instrumentation ships located in the Atlantic and Indian Oceans.

Upon acquisition of the spacecraft by the Deep Space Network tracking stations, control of the Lunar Orbiter mission was passed from Cape Kennedy to the Space Flight Operations Facility at Pasadena, California.

The performance of the overall mission management, mission design, launch operations, flight operations, and the logistic system are discussed in the following sections.

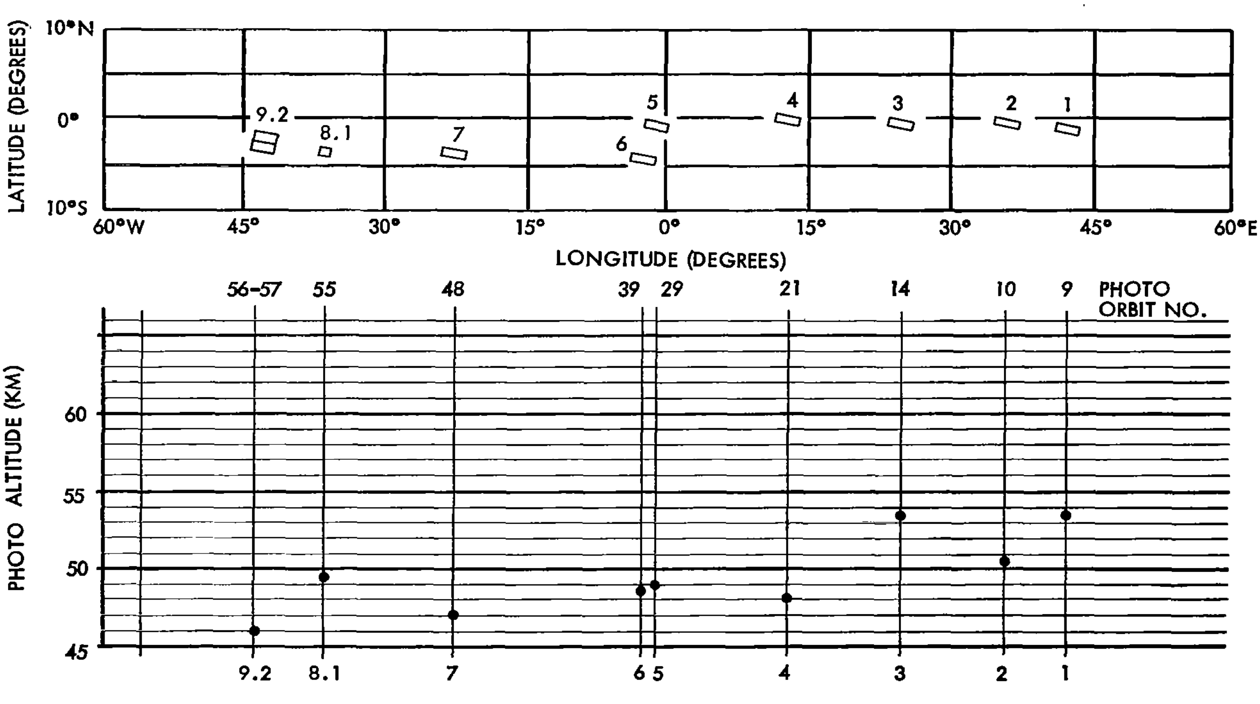


Figure 1.3-1: Site Location and Photo Altitude

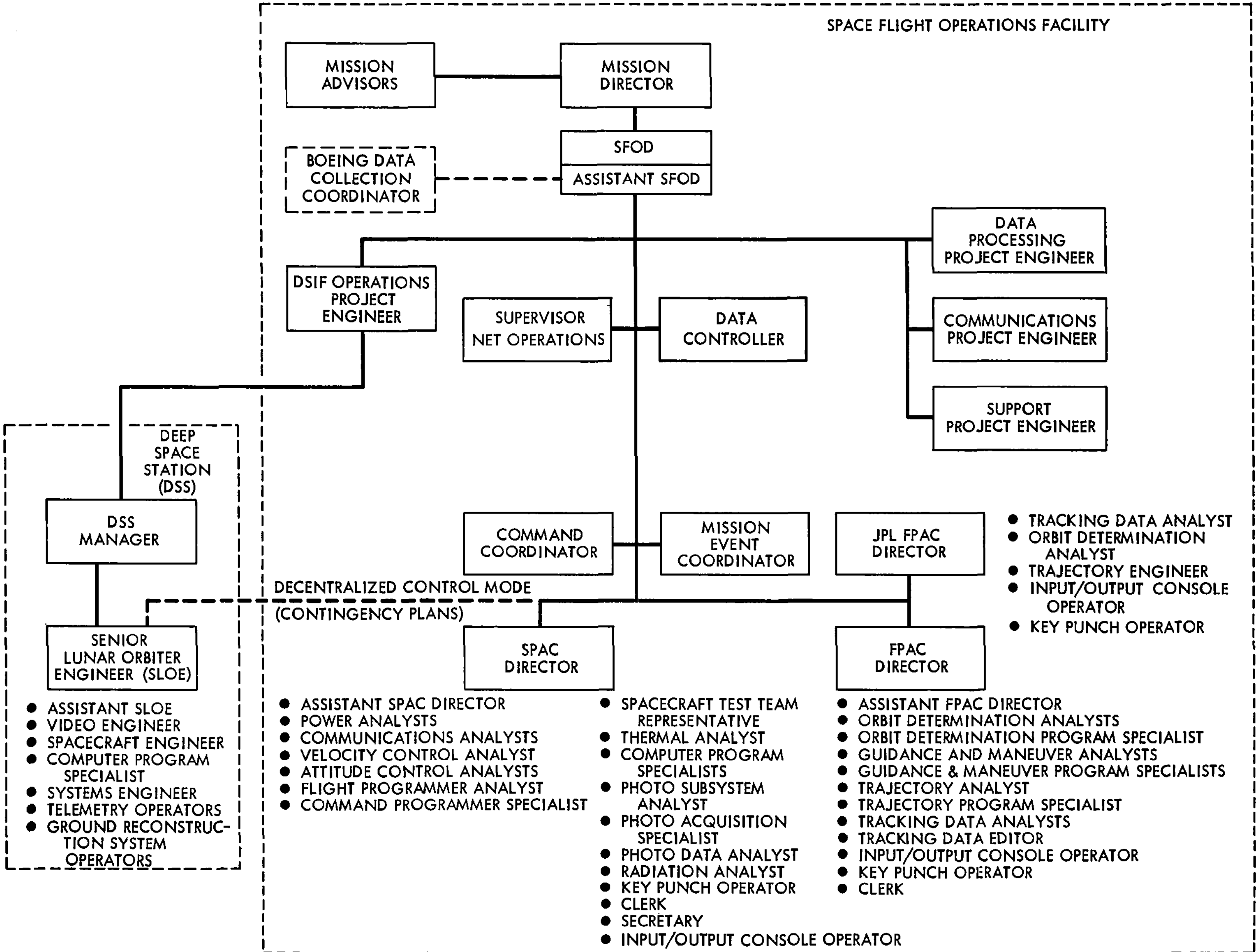


Figure 1.3-2: Operational Organization

1.3.1 MISSION MANAGEMENT

Operation and control of Lunar Orbiter I required the integrated services of a large number of specialists stationed at the SFOF facility in Pasadena, as well as at the worldwide Deep Space Stations. Figure 1.3-2 shows the relationship and indicates responsibilities of this team of specialists composed of representatives from the Lunar Orbiter project office, supporting government agencies, Jet Propulsion Laboratory, The Boeing Company, and the Deep Space Stations.

The Langley Research Center exercised management control of the mission through the mission director. Two primary deputies were employed: the first, the launch operations director located at Cape Kennedy; the second, the space flight operations director located at the SFOF.

During all spacecraft-preparation and mission-simulation periods prior to the beginning of the final countdown, the mission director was kept apprised of the status of all preparations of the launch and flight control of the spacecraft by his appropriate deputies. Based on this information, the mission director initiated the start of the final countdown from the mission control center at Cape Kennedy. Once the countdown started, the launch operations director directed the progress of the countdown on the launch pad, while the space flight operations director directed the countdown of the Deep Space Network. From the time that these countdowns were synchronized, all decisions (other than Eastern Test Range safety factors) regarding the countdown were made by the mission director, based on recommendations from the launch operations director and/or the space flight operations director.

After liftoff, the performance of the launch vehicle and spacecraft was monitored in the launch mission control center at ETR by the mission director. Telemetry data was used by the launch team and was relayed in real time to the SFOF through the Cape Kennedy DSS. This dissemination of spacecraft performance data to the launch team and the operations team enabled the efficient and orderly transfer of control from Cape Kennedy to the SFOF.

After the spacecraft had been acquired by the Deep Space Network, flight control of the spacecraft was assumed by the space flight operations director. Thereafter, the mission director moved to the SFOF and continued control of the mission. Spacecraft operations were delegated to the space flight operations director.

Flight control of the mission was centralized at the SFOF for the remainder of the mission. All commands to the spacecraft were coordinated by the SPAC and FPAC team of subsystem specialists and submitted to the space flight operations director for approval prior to being transmitted to the DSIF site for retransmission to the spacecraft. As a backup capability, each prime DSIF was supplied with a contingency capability (including, predetermined commands and process tapes) to permit local assumption of the basic flight control functions in the event of communications failures.

1.3.2 MISSION DESIGN

The Lunar Orbiter spacecraft was designed "around" its photographic payload to ensure the maximum probability of success of the photographic mission. In similar fashion, the mission design maximized the probability of quality photography by placing the spacecraft over the mission target(s) in the proper attitude, altitude, and within the established photographic lighting limitations. These photographic requirements, when combined with other factors, resulted in a list of considerations that were integrated into the mission design. Some of the more significant items are:

1) Launch from AFETR using an Atlas-Agena vehicle and subject to:
   a) Range azimuth constraints,
   b) Atlas guidance system constraints,
   c) Agena guidance system constraints,
   d) Booster system performance,
   e) DSN and downrange tracking capabilities,
   f) Range safety considerations;
2) Lunar photography within the Sun illumination band (50-to 75-degree phase angle);
3) Photo data transmission only during spacecraft Sun and Earth visibility;
4) Film processing only during solar illumination;
5) Contiguous high-resolution coverage, of lunar surface on successive orbit passes;
6) Photo altitude consistent with V/H capability and resolution requirements;
7) Spacecraft maneuver requirements within the velocity change capability of the spacecraft;
8) Limited Sun occultation time;
9) A maximum of consecutive launch days per month with a minimum of 2-hours-per-day launch window;
10) A minimum of 3 days in Lunar Orbit, for tracking, prior to initial photography;
11) "Sunrise" photography;
12) Descending node photography.
13) Two-station visibility during deboost and part of the lunar orbit before first lunar occultation.

Within the framework of the above constraints, nominal mission design was initiated as soon as the photo targets and possible launch months had been defined. Major outputs of this effort (for each possible launch month) were:

1) A targeting specification for the booster agency;
2) Tabulated trajectory data;
3) Nominal mission definition;
4) Mission error analysis;
5) Tracking and telemetry coverage plan;
6) Station viewing periods for tracking net;
7) Alternate missions studies;
8) Time line analysis (mission event sequence);
9) Detailed photo frame budget.

The first Lunar Orbiter Mission, then identified as Mission A, was defined on September 29, 1965. As approved by the Ad Hoc Surveyor/Orbiter Utilization Committee, the mission specification identified ten photo sites within the area of interest and an eleventh site at the eastern limb. Subsequently, on June 1, 1966, the specification was amended such that four of the 11 sites were changed. Three sites were shifted in location (one of which included the landing site of Surveyor I) and one was eliminated.

Planning was initiated for the months of June, July, and August, 1966, with September being added later.

Suitable launch periods were established, considering the operational constraints, for the 9th through the 13th of August. Detailed parametric data were defined for each of these possible launch days, along with required supporting planning and documentation. As actually flown, the August 10th mission was very close to nominal until the completion of the initial eastern limb photography. Following identification of a high-resolution-camera shutter problem, the subsequent photo mission was altered considerably as a result of actions taken in an effort to resolve this problem.

1.3.3 LAUNCH OPERATIONS

The Lunar Orbiter launch operations plan identified the requirements for the overall space vehicle program direction and coordination to complete preparation and launch. NASA, the range contractor, and launch vehicle and spacecraft contractors were required to ensure that all elements of the program were ready to support the scheduled launch. The NASA Langley Research Center had the overall responsibility for all Lunar Orbiter program activities at AFETR. A deputy for AFETR operations ensured the complete integration of resources, procedures, and personnel to achieve maximum utilization. The Unmanned Launch Operations Division of the Kennedy Space Center supported the Lunar Orbiter program in these areas.

1.3.3.1 Atlas-Agena Operations - The Unmanned Launch Operations Division was responsible for all activities associated with the lunar-vehicle assembly and checkout and space-vehicle operations. The Launch Operations Working Group (LOWG) was the primary coordinating agency for flight preparations at AFETR.

1.3.3.2 Vehicle and Space Support Operations (VSSO) - This group was responsible for providing spacecraft assembly and checkout facilities, coordinating spacecraft operations with launch vehicle and AFETR personnel, obtaining range support, ensuring that operations were conducted in accordance with existing regulations, and serving in a monitoring and controlling capacity to the deputy of AFETR operations.

1.3.3.3 NASA Test Support Agency (NTSA) - This agency was responsible for officially representing the Lunar Orbiter program to the AFETR, submitting all range documentation and requirements, and negotiating with the AFETR for operations support and requirements to ensure that all support was obtained in a timely manner.

1.3.3.4 Final Assembly and Checkout Operations

The Atlas and Agena boost vehicle and the Lunar Orbiter spacecraft each received acceptance quality tests at the individual contractor's plant prior to delivery to the AFETR. Upon arrival, each vehicle was prepared for launch as summarized in Figure 1.3-3.

The Atlas was given a receiving inspection prior to transportation to Launch Complex 13. The Agena vehicle completed a series of tests and operations to verify readiness before transfer to the launch area and assembly with the Atlas booster. Tests and operations of the Lunar Orbiter spacecraft were conducted in Hangar "S" and also in the Explosive-Safe Area(ESA).

As received in Hangar "S", the spacecraft had all ordnance and protective devices installed, and was completely assembled, except for the thermal barrier, flight batteries, photo subsystem, and propellants. After the completion of rf compatibility checks with DSS-71 and a low-pressure leak test of the gas and propellant systems, the photo subsystem was installed with test film in the supply reel. Dry-weight and balance measurements were made to establish a baseline for optical-alignment checks.

The spacecraft was then placed in a flight configuration except that the propellant, flight film, flight batteries, thermal shroud, Agena spacecraft adapter, and spacecraft nose fairing were not installed. Functional tests were made on all subsystems of the spacecraft using the checkout van and auxiliary test equipment. Mechanical systems of the spacecraft were checked for mechanical deployment and electrical alignment. Upon satisfactory completion of all of these tests, the photo system was removed from the spacecraft for final testing and loading of the flight film and Bimat. The protective covers were then installed over the spacecraft prior to transfer to the checkout van for transportation to the ESA.

Upon arrival in the ESA, the spacecraft was prepared for fueling. High-pressure leak tests were conducted on the nitrogen-gas system, the tanks were charged with nitrogen, and the spacecraft readied for fueling. After fueling, the photo system (with flight film installed) and the flight batteries were installed in the spacecraft. Optical alignment was verified and an abbreviated verification test performed using DSS-71.

The final preparation in the ESA for mating the spacecraft with the launch vehicle consisted of electrical connection of the spacecraft ordnance and performing a wet weight and balance check. This was followed by the installation of the spacecraft on the Agena adapter and installing the thermal barrier and nose fairing. A final system-verification test was conducted using the checkout van. Upon satisfactory completion of this test, the spacecraft was transported to the launch area for installation on the launch vehicle.

1.3.3.5 Launch Complex Operations

All assembly and test operations conducted at Launch Complex 13 were scheduled to ensure that the launch vehicle and spacecraft were completely assembled and tested in time to support the launch dates. The Atlas booster was the first vehicle element to arrive at the launch complex. Following booster erection, an integrated launch van-block house-vehicle system test was made. In addition, a booster propellant-tank test and a booster flight-acceptance composite test (B-FACT) were performed to demonstrate the Atlas systems flight readiness.

On satisfactory completion of the B-FACT, the Agena vehicle was transferred to the launch pad for assembly with the Atlas booster. Functional tests were performed on the Agena subsystems to determine launch pad control equipment ability to control and monitor the vehicle subsystem. The interface compatibility of the launch-vehicle system was determined prior to being mated with the spacecraft. On completion of the above sequences,

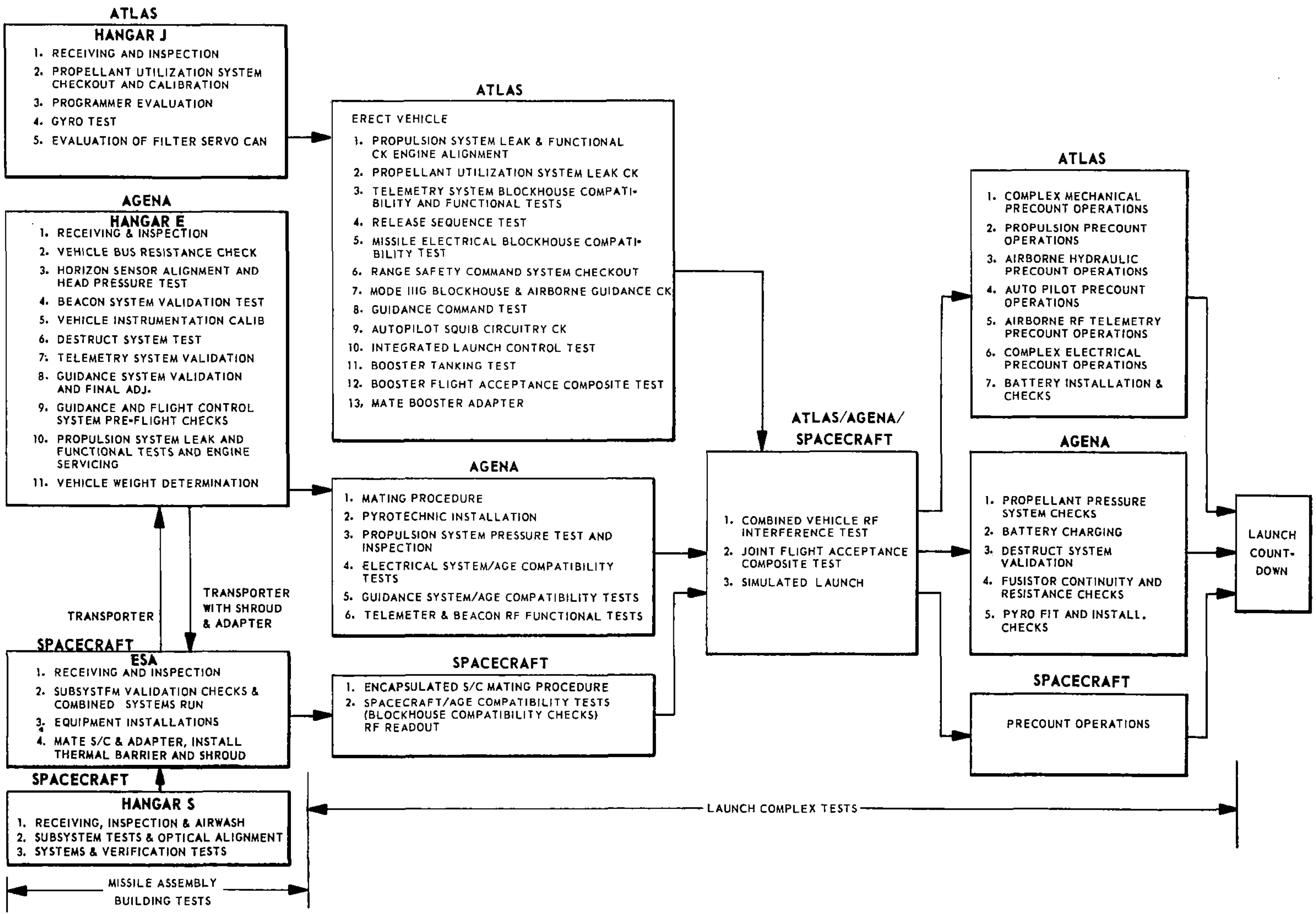
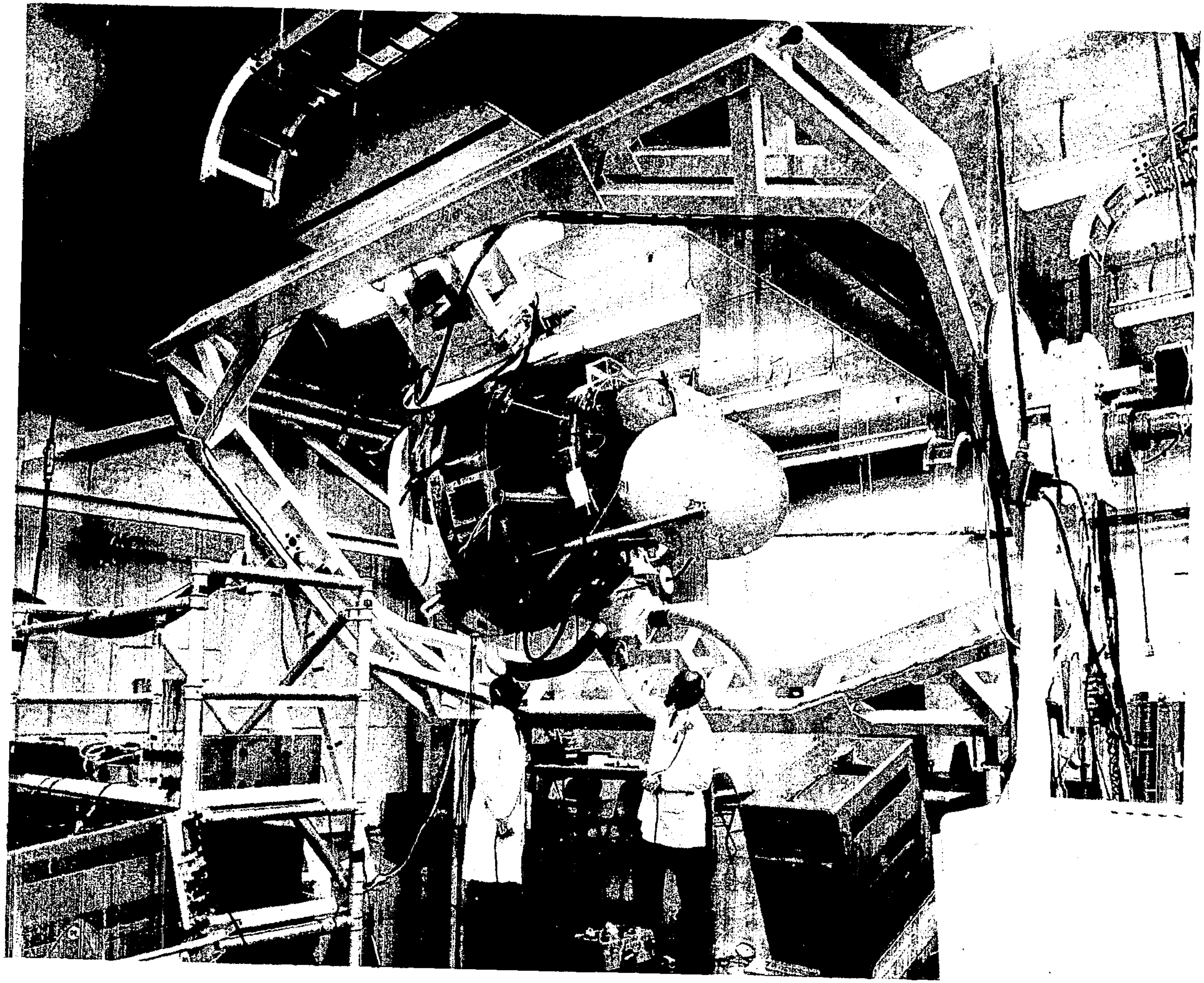


Figure 1.3-3: Launch Operations Flow Chart



Lunar Orbiter on Three-Axis Test Stand
(Seattle, Washington)

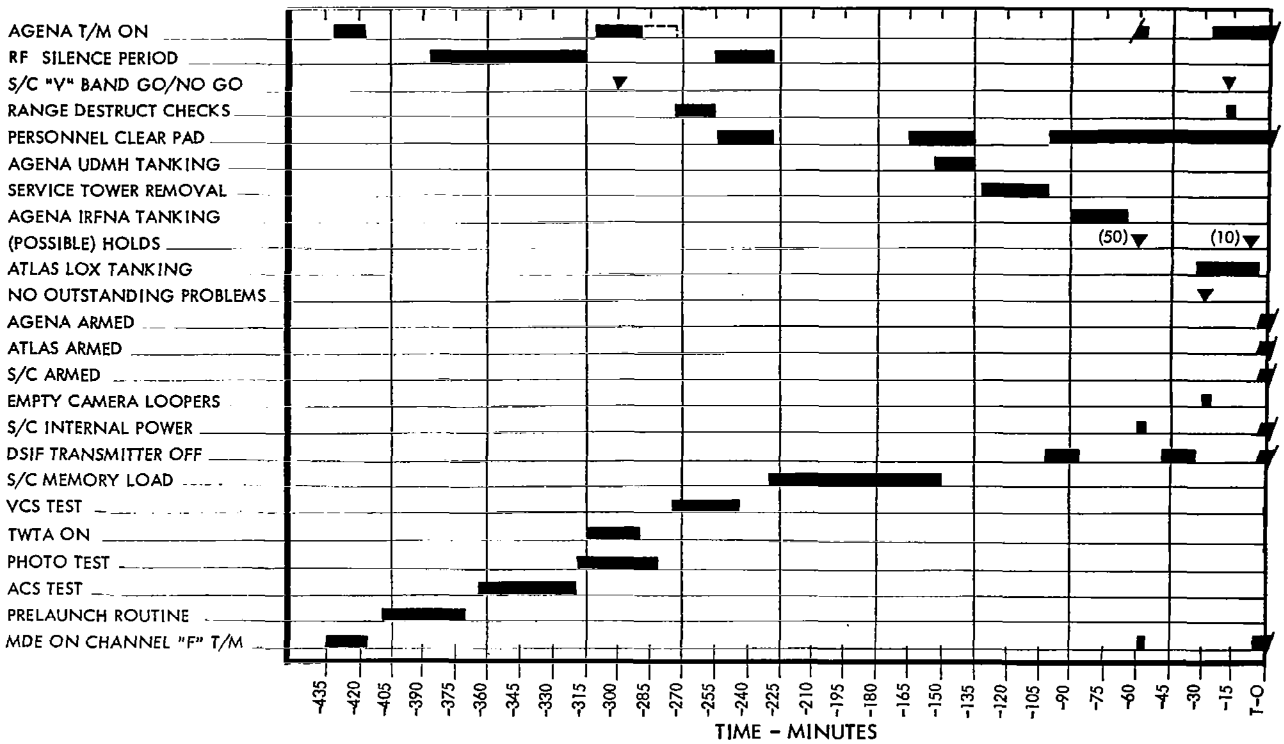


Figure 1.3-4: Master Countdown Time Sequence

the Lunar Orbiter spacecraft was transferred to the launch complex for installation on the Agena vehicle. After mechanical and electrical connections were made, functional tests were performed on the spacecraft to verify the operational status and compatibility with the DSS-71.

From this point, tests accomplished on the system were made with the launch vehicle and spacecraft combined to ensure and demonstrate the functional compatibility between the individual vehicles and between the space vehicle and the aerospace ground equipment (AGE). Radio-frequency interference tests (RFI) verified the compatibility of the space vehicle (Atlas, Agena, and spacecraft) and the ground devices used during the countdown. During these radiation tests, the service tower was removed to eliminate any possible interference or creation of multipaths.

Joint flight acceptance composite tests (J-FACT) were an integrated check of the vehicle systems, the launch complex, and all applicable range stations. Operations from T-O through Agena retromaneuver were simulated to verify flight operations, commands, and sequences. All operating systems were on internal power to ensure proper operation of all vehicle systems. Umbilical ejection and autopilot programmer operations were also performed. All standard safety precautions were employed to prevent the generation of inadvertent cutoffs or range safety destruct signals. All of the vehicle systems were radiating and the applicable airborne systems were interrogated by the range with the service tower in place. Closed-loop tests of the GE-Burroughs Mod III-G guidance system were performed both before and after the umbilical ejection.

The umbilical-release tests consisted of an electromechanical check of the complete release mechanism and boom-retraction sequence on the Atlas-Agena-spacecraft umbilicals in the umbilical tower; all quick-disconnects were in a launch configuration.

1.3.3.6 Flight Readiness Review

Upon completion of all the above test and checkout sequences, a formal review was scheduled. This review included presentation of test results from hardware and associated testing and checkout results. All component waivers, test anomalies, and deviations were reviewed by the responsible agencies. Upon their formal acceptance and approval, the final countdown was initiated.

1.3.3.7 Launch Countdown

All operations and tasks performed during the actual countdown were carefully sequenced on a lapsed-time basis to demonstrate total space-vehicle system and range (AFETR, DSIF, and SFOF) readiness. Spacecraft checks during the countdown included blockhouse and spacecraft command interrogation and verification of the general spacecraft readiness. The booster propellant level was brought up to flight capacity and verified by blockhouse instrumentation. As these tasks were completed, the launch vehicle and spacecraft remained in a flight-ready condition until the programmed launch plan was initiated and launch effected. A simplified countdown sequence for the spacecraft is contained in Figure 1.3-4.

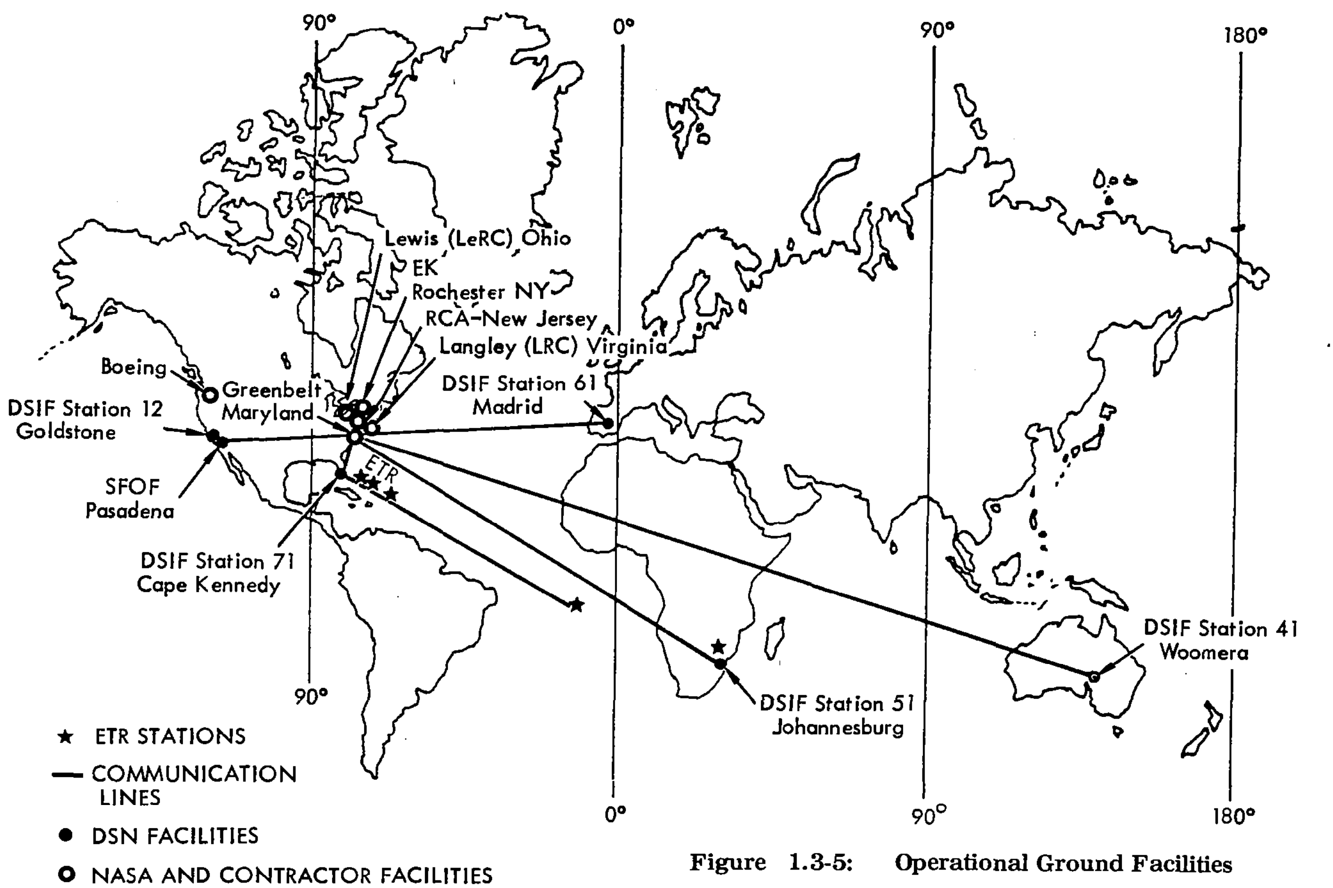


Figure 1.3-5: Operational Ground Facilities

1.3.4 MISSION OPERATIONS

The Lunar Orbiter operational ground facilities, shown in Figure 1.3-5, are located at the Air Force Eastern Test Range and at certain stations and facilities of the NASA Deep Space Net.

1.3.4.1 MISSION SUMMARY

The first officially published launch date for the Lunar Orbiter I was during the windows occurring in July, 1966. However, technical problems with the photo subsystem late in the test program delayed delivery. After minor changes and a thorough alignment check within the camera, the photo subsystem was delivered and installed in Lunar Orbiter I to support the August launch window.

Prior to the July launch date, extensive training exercises were conducted; these involved the facilities of AFETR, the Space Flight Operations Facility in Pasadena, the Seattle Operations Center, and the stations of the Deep Space Network. The month delay in launch schedule was used to enhance the proficiency of the operational personnel by additional training and simulation exercises

On August 9, the first countdown was initiated and proceeded smoothly for 2.5 hours. At this time, the 100-word-per-minute teletype printers in the SFOF failed to function properly and, shortly thereafter, the X and Y computer strings were down. In addition, the equipment mounting deck temperature of the spacecraft appeared to have exceeded the desired maximum temperature, due to problems with the ground cooling system. After a 50-minute hold, at T-60 minutes, the countdown was resumed and continued smoothly until T-7 when a scheduled 10-minute hold began. During the T-7 hold, an anomaly was noted with the propellant utilization system of the Atlas booster. This problem resulted in a series of 5-minute holds. As these holds progressed, the launch azimuth was changed at 15-minute intervals. The changes were necessitated by the changes in relative position of the Earth and Moon and also the duration of the coast period in Earth orbit. Each new launch plan also required that the spacecraft programmer core map be updated. The launch attempt was subsequently canceled by the mission director, based upon the recommendation of the launch director, to correct the erratic output signal from the propellant utilization system. The mission was subsequently canceled by the mission director to correct the erratic output signal from the propellant utilization system.

The mission was again scheduled for August 10, 19:11 GMT on the 98.6-degree azimuth required by Launch Plan 10G. Following the 7.5-hour countdown, launch occurred using Launch Plan 10H. At the time of launch (19:26:00.716 GMT) 160 minutes of the launch window remained.

Figure 1.3-6 provides a pictorial summary of the 35-day Lunar Orbiter Mission I. Significant events are shown, in Greenwich time, for each phase from the initiation of the launch countdown on August 10, through the completion of the final readout on September 14. Also shown are the phasing of the major launch vehicle sequences required to place the spacecraft on the cislunar trajectory. Since the primary photo sites were located between 90° east longitude and 45° west longitude, a period of 12 days was required for the spacecraft's orbit to precess across this area and bring each site within vertical photography range of the cameras with the proper illumination. The irregular spacing of photo sites required variations in the number at orbits between site photo sequences. The figure does not show the 55 film-set and test exposures that were taken at intervals between the primary photo sites.

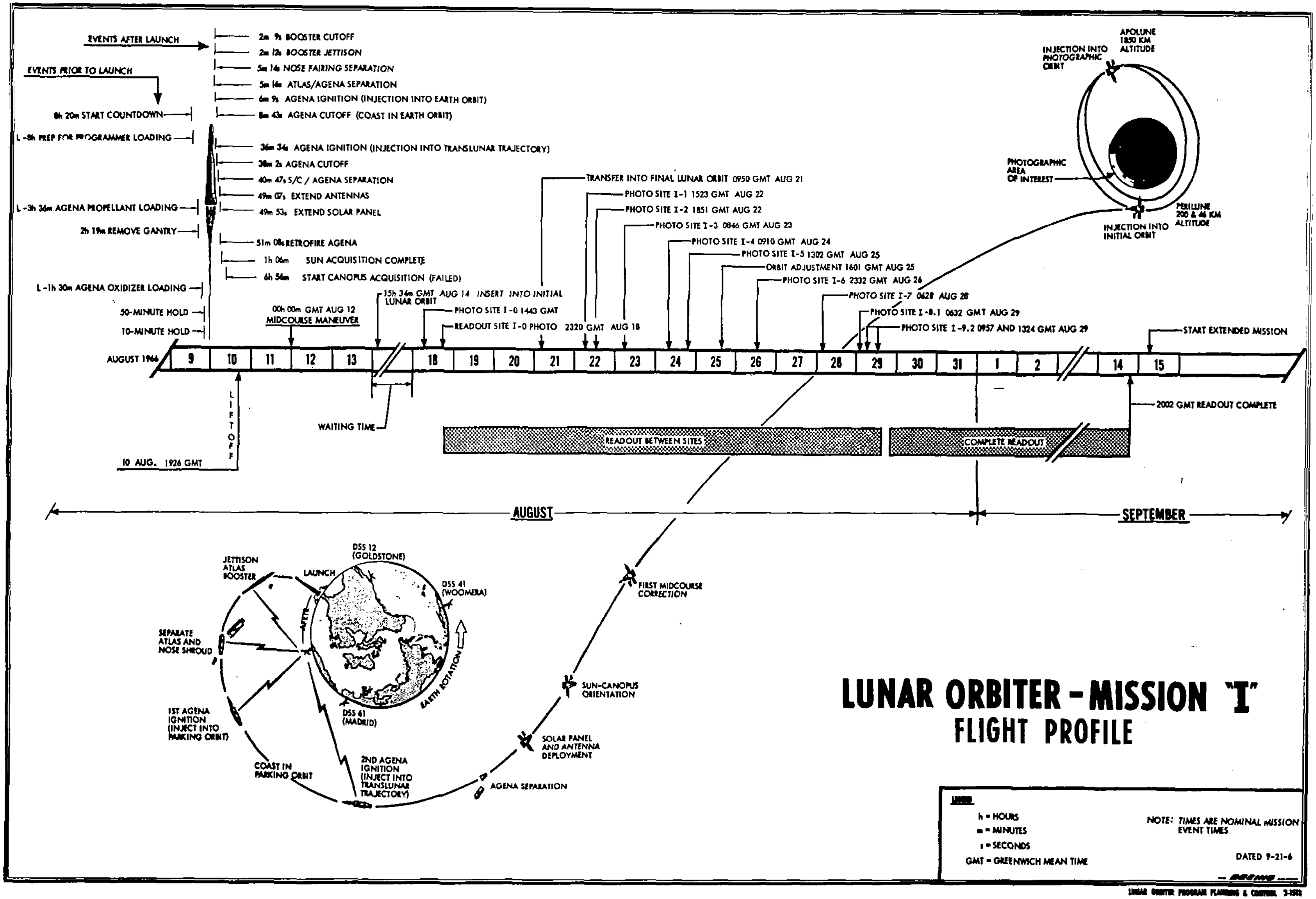


Figure 1.3-6: Flight Profile

The "readout between sites" period was limited to a maximum of one spacecraft frame or less by photo subsystem restrictions. Operation of the photo subsystem processor-dryer was controlled and integrated with the exposure sequences to enable the readout of preselected exposures. During the "complete readout" period a minimum of two spacecraft frames were read out on each orbit. Approximately one orbit each day was used to obtain extended tracking data, thermal control, and flight programmer updating.

In accordance with the requirements of Launch Plan 10-H, a launch azimuth of 99.9 degrees was required to ensure that the spacecraft could be placed on the proper cislunar trajectory to rendezvous with the Moon approximately 90 hours later. The Agena and spacecraft were first placed in a circular orbit approximately 100 miles above the Earth. After a planned coast period, based on the relative positions of the Earth, spacecraft, and Moon, the Agena engine was fired a second time to inject the spacecraft into its cislunar trajectory. Subsequent to separation of the Agena and Lunar Orbiter, programmed sequences within the spacecraft deployed the high-gain antenna, the solar panels, and the low-gain antenna. The roll attitude orientation, scheduled to be established and maintained by the star tracker acquiring the navigational star, Canopus, was accomplished by an alternate method using the Moon as a reference. This was due to stray light reflecting into the star tracker during the first attempts to acquire Canopus.

The midcourse command was executed approximately 28 hours after launch and with such precision that the planned second midcourse maneuver at launch plus 70 hours was not required.

As the spacecraft approached the Moon, it was maneuvered to the proper attitude, the velocity-control engine fired, and the spacecraft injected into the first lunar orbit. This maneuver and velocity change placed the spacecraft in a lunar ellipse having an 1867-km apolune, a 189-km perilune, and an orbit inclination of 12.15 degrees at the lunar equator.

The spacecraft was maintained in this initial orbit for 4 days while tracking data were obtained by each DSIF site in turn. The tracking data were used to:

1) Solve for harmonic coefficients representing the lunar gravity field;
2) Establish tracking data experience in an orbital flight phase;
3) Design the first maneuvers for photographic coverage of Site I-0;
4) Design a transfer maneuver for a second ellipse to yield proper photo coverage of all Mission I sites, other than Site I-0.

A series of 20 exposures was taken during Orbit 26 over the designated Site I-0 (approximately 90° E longitude and at the lunar equator). Processing of these photos within the spacecraft was controlled by a series of real-time commands so that the required photos were properly positioned at the photo subsystem readout gate for transmission to the ground. The first readout was received and reconstructed at the Goldstone DSS and rapidly transported to the SFOF for evaluation prior to initiating the command sequences for transfer to the second ellipse.

During the evaluation of these photos, the high-resolution photographs were found to be smeared while the moderate-resolution photographs were of good quality. Analysis of all available data during this period indicated that the mission should proceed essentially as planned and that special tests and photographs be taken to aid in the evaluation of the high-resolution camera system. Appropriate commands were transmitted to the spacecraft to perform the attitude maneuver and velocity change required to transfer to the second ellipse.

At the end of Orbit 45 in the initial ellipse, the velocity control engine was fired a third time to reduce the spacecraft velocity by 40 meters per second and attain a perilune altitude of approximately 56 km. The photo graphic mission was continued as the spacecraft was maneuvered by command over each of the primary sites (I-1 through I-5) located between 42.2°E longitude and 1.4°W longitude.

To provide early evaluation of the photographic information, a series of selected photographs were scanned in a priority readout mode during orbits when photographs were not taken.

Evaluation of photo subsystem performance continued during all of the priority readout periods at the SFOF and each DSS. All high-resolution photos showed that the reduction in altitude did not improve the quality of the high-resolution photos. Evaluation of test data at Eastman Kodak indicated that further reduction in perilune might improve the probability of obtaining good high-resolution photographs.

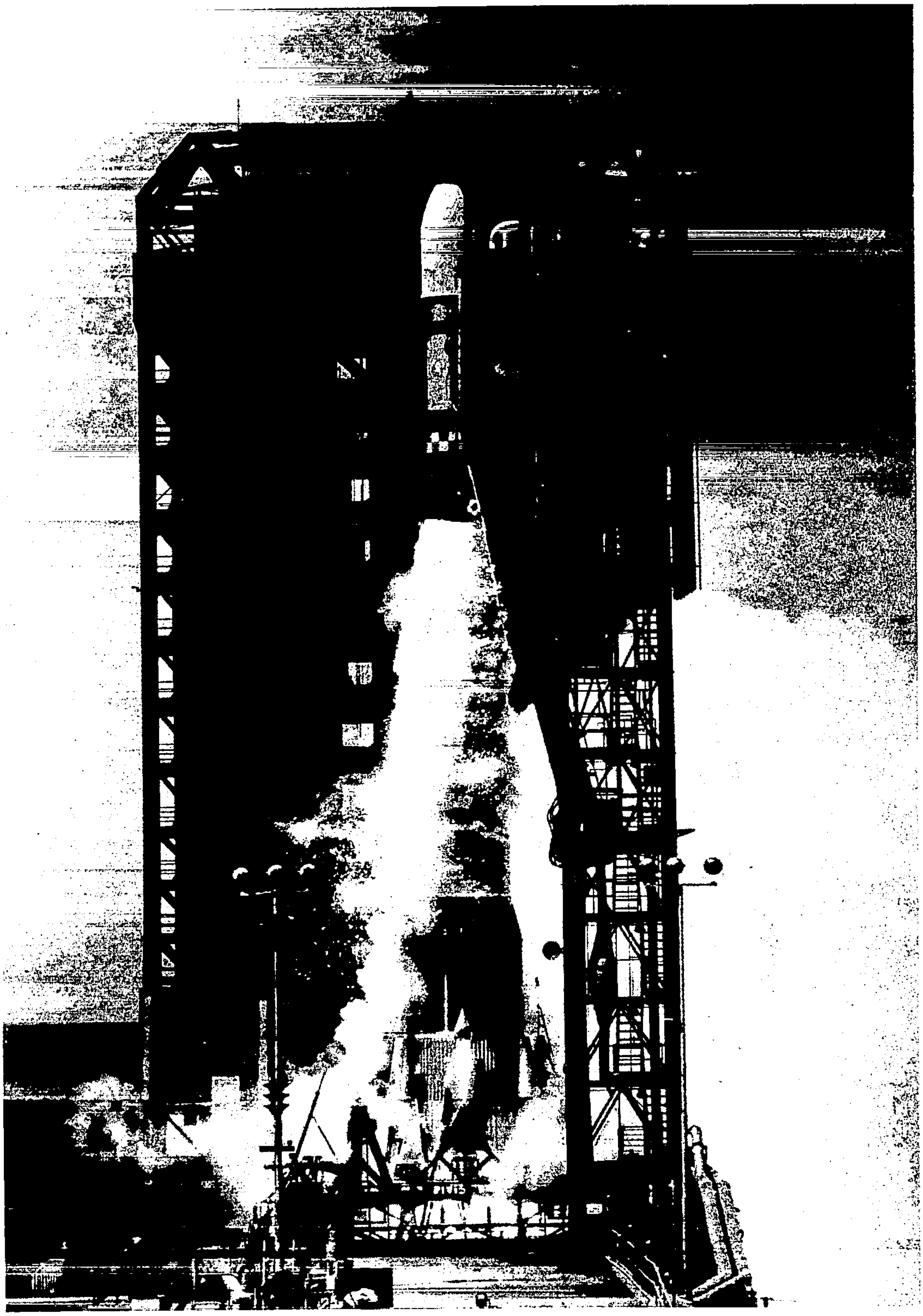
Accordingly a fourth velocity-control engine ignition was made during Orbit 30 to decrease spacecraft velocity by 5.42 meters per second for the desired orbit change. This final orbit change did not improve the quality of the high-resolution photographs and the remainder of the photo mission was completed without any additional photographic testing. Photos were taken of primary sites (I-6 through I-9.2) located between 0.2° W and 42.3° W longitude.

Photo subsystem operational constraints required the usage of some film not allocated to prime target pictures to minimize the effects of film set, Bimat stick, and Bimat dryout. This requirement was used to advantage by commanding the spacecraft to take such photographs as the Earth as seen from the vicinity of the Moon, the morning afternoon terminator (the line of demarcation between lunar day and night), and large areas on the farside of the Moon. Additional frontside photography included potential sites for subsequent Lunar Orbiter missions, large areas of scientific interest, and Apollo navigational aids. Photography continued through the last of the preselected sites (covering the area of the successful Surveyor I landing). After the last picture was taken on August 30, a "Bimat cut" command was transmitted to the spacecraft. During the next 15 days, all 211 dual-exposed frames taken by the spacecraft were read out, converted to a video signal, and transmitted to Deep Space Stations. This final readout sequence was in the reverse order of picture taking. The first photo taken by the spacecraft (August 18) was the last to be read out, transmitted, and recorded; this occurred on September 14.

1.3.4.2 LAUNCH VEHICLE PERFORMANCE

Analysis of vehicle performance, trajectory, and guidance data indicated that all launch-vehicle objectives were satisfactorily accomplished.

This was the eighth Atlas SLV-3 vehicle (Serial Number 5801) to be launched from AFETR. Performance of all Atlas subsystems was satisfactory. A ground-based guidance and command system working with the airborne



Lunar Orbiter I Launch

EVENT	PROGRAMMED TIME (+SEC)	MEASURED TIME (TIM) (+SEC)
LIFTOFF 2-IN. MOTION	0	1926:00.716 GMT
BOOSTER ENGINE CUTOFF	128.5	128.8
SUSTAINER ENGINE CUTOFF	288.8	290.9
START ASCENT SEQUENCE TIMER	291.3	294.9
VECO - UNCAGE GYROS, JETTISON H/S FAIRINGS	309.1	311.8
NOSE SHROUD EJECTION	311.1	314.3
SLV-3/AGENA SEPARATION	313.5	316.8
SEPARATION BACKUP (SEQUENCE TIMER)	337.3	341.0
INITIATE - 120 DEG/MIN PITCH RATE	334.3	348.0
TRANSFER TO -3.21 DEG/MIN PITCH RATE; PITCH H/S TO IRP	349.3	353.0
ENABLE VELOCITY METER	364.3	367.9
ARM ENGINE CONTROL	364.3	368.0
FIRST BURN IGNITION (90% P_c)	365.6	369.2
FIRST BURN CUTOFF (V/M CUTOFF SWITCH)	517.0	523.0
(ENGINE SWITCH GROUP)	517.0	523.0
START GYROCOMPASSING	532.3	535.9
TRANSFER TO -4.20 DEG/MIN PITCH RATE	532.3	535.9
TRANSFER TELEMETRY GYRO OUTPUTS TO LOW	532.3	535.9
HORIZON SENSORS TO 0.23 DEGREE BIAS POSITION	544.3	547.9
DISABLE V/M	2180.3 (a)	2180.5
TLM GYRO SIGNAL CONDITIONERS TO HIGH	2180.3	2180.4
STOP GYROCOMPASSING	2180.3	2180.4
SWITCH TIM TO V/M COUNTER	2186.3	2186.3
ENABLE V/M AND SWITCH TIM TO ACCELEROMETER	2191.3	2191.3
BOTH ISOLATION VALVES OPEN	2193.2	2193.0
SECOND BURN IGNITION (90% P_c)	2193.3	2194.4
SECOND BURN CUTOFF	2281.5	2282.2

(a) ALL PROGRAMMED EVENTS AFTER "DISABLE V/M" ARE BASED ON THE SECONDARY TIMER START DISCRETE AT PLUS 271.3 SECONDS.

Table 1.3-1: Sequence of Events

guidance and flight control systems maintained the ascent trajectory within prescribed limits. Satisfactory telemetry data was obtained during all phases for Atlas performance analysis.

Similarly, the Agena (Serial Number 6630) upper-stage booster performed satisfactorily through all phases of flight. This launch was the first flight of an Agena D vehicle that maintained propellant tank ullage control by the use of special tank sumps in place of the previously used positive continuous ullage control system.

The velocity meter accelerometer in the Agena system was enabled after the first Agena-burn period. Accelerometer pulses continued with decreasing frequency until loss of signal at the Antigua tracking station. Telemetry data from Pretoria also indicated that the accelerometer pulses were decreasing with a greater change in frequency. This phenomenon has been observed on previous Agena missions and apparently has no undesirable effect on the mission.

All timer-controlled events occurred as scheduled, and the engine burn periods were within expected tolerances. The Agena and spacecraft separation sequence, including the yaw maneuver of the Agena vehicle and the Agena retromaneuver, was recorded by the range instrumentation ship in the Indian Ocean. This data was not available when this report was prepared.

All available tracking data indicated that the launch-vehicle flight trajectory was satisfactory and that the position and velocity of the spacecraft at the time of separation was within acceptable limits. Table 1.3-1 presents significant events for the launch vehicle during the ascent trajectory. All times are referenced to the liftoff time of 19:26:00. 716 GMT.

Atlas Performance

Atlas SLV-3 operation was satisfactory throughout the flight. All engine thrustwise periods were normal. Peak accelerations of 6.1 and 3.1 g were recorded at booster engine cutoff (BECO) and sustainer engine cutoff (SECO), respectively. A 5-Hz longitudinal oscillation was noted after liftoff; this reached a maximum peak-to-peak level of 1.0 g and was damped out by approximately T + 19 seconds. This oscillation produced no adverse effects in the performance of either the Atlas-Agena combination or the spacecraft. The propellant-utilization system responded properly to the error-counter output signal. Calculations based on performance data indicated that there were approximately 1194 pounds of liquid oxygen and 753 pounds of fuel remaining at SECO. These propellants would have produced approximately 6.6 seconds of additional engine burn time.

Vehicle stability was maintained throughout all phases of powered flight by the Atlas flight-control system. The programmed roll and pitch maneuvers and other commanded maneuvers were satisfactorily executed. The small attitude-change transients resulting from liftoff were satisfactorily damped out on activation of the autopilot after 42 inches of vertical rise. Performance data indicated that the vehicle angular displacements and rates at vernier engine cutoff (VECO) were negligible.

Analysis of ground recorded and telemetry data indicated that both the Mod III-A ground station and Mod III-G airborne guidance equipment performed satisfactorily. All discrete and steering commands were properly transmitted by the ground station, received by the beacon and decoded and executed by the flight-control system. Beacon track was maintained until launch plus 393.6 seconds, when the received signal strength decreased to the noise level. Evaluation of all recorded data (vehicle performance telemetry, ground system monitoring, and tracking data) indicated that all components of the Atlas launch vehicle and ground supporting system operated properly throughout all phases of Atlas powered flight.

The following coast ellipse parameters and insertion parameters at VECO+2 seconds were obtained from the guidance system data:

Parameter	Value
Semimajor axis	14,512,105 feet
Semiminor axis	12,707,193 feet
Velocity magnitude	18,520 feet per second
Velocity to be gained	+0.40 foot per second

Filtered yaw velocity	-3.12 feet per second
Filtered altitude rate minus desired altitude rate	+15.25 feet per second

Agena-D Performance

Agena performance data recorded by the range instrumentation ship in the Indian Ocean was not available at the time of preparing the Lunar Orbiter I Launch Report section on Agena performance. However, since the spacecraft was acquired by the Deep Space Network at near the expected time and the cislunar trajectory was within tolerance, it must be assumed that performance of the Agena was satisfactory.

Performance of the propulsion system was satisfactory through the loss of signal from the vehicle. The first-burn ignition was initiated at the proper time by the on-board sequencer. The main combustion chamber pressure was 507.5 psia, which produced a calculated thrust of 16,072 pounds. The total propellant flow rate was computed at 55.22 pounds per second at a specific impulse of 291.0 pound-seconds per pound. The first burn period lasted for 153.8 seconds. Preliminary data was available from the Pretoria range station during the second burn period. Early analysis of this data indicated a combustion chamber pressure of 506 psia. The burn duration for the second firing of the engine was 87.8 seconds (0.9 second longer than predicted).

The guidance and flight control subsystem performance was normal during both engine-burn periods. Continuous data on vehicle longitudinal acceleration was obtained by leaving the accelerometer pendulum output connected to the velocity meter telemetry signal.

Many of the pitch and roll maneuvers during the Atlas boost phase were sensed by the caged Agena gyros. Small disturbances were also noted at BECO; those at Atlas-Agena separation were barely visible. Inertial reference package gyro disturbances at first and at second Agena burn ignition were small and quickly damped out by hydraulic actuator motion.

1.3.4.3 SPACECRAFT PERFORMANCE

The performance of the Lunar Orbiter system is best evaluated in light of program and specific mission objectives. Accordingly, the performance of each of the subsystems as it relates to these objectives is discussed in the following paragraphs.

To place the photo subsystem in the proper location and attitude at the right time to obtain the desired photographs:

1) The Lunar Orbiter must be injected into an accurate orbit about the Moon whose size and shape, as well as center of gravity and mass, are not precisely known.

2) It must perform a critical attitude maneuver by precise velocity reduction to transfer into a specified lower photographic orbit.

3) It must also continue to operate in an unknown radiation environment and in an unknown density of micrometeoroids over an extended time.

4) It must accomplish a precise attitude maneuver prior to photographing each specified site and actuate the cameras at precisely the commanded time.

5) The operations system must provide the tracking data from which to verify that assumptions made in computing the orbit parameters are valid and that, in fact, the photographic mission can be accomplished.

Failure to satisfy any of these conditions could jeopardize accomplishment of the Lunar Orbiter mission.

How well Lunar Orbiter I did in fact accomplish these critical tasks is shown in Table 1.3-2.

Lunar Orbiter I, having demonstrated that it could inject itself into a permanent orbit about the Moon, continued its primary mission of photographing selected lunar sites while, at the same time, providing additional tracking data and monitoring the environmental conditions about the Moon. Without exception, the spacecraft responded faultlessly to all commands directed to it by the operations team. Over 35 million programmer logic cycles were completed without an error during the 35-day mission. The operational event summary shown in Table 1.3.3 indicates the increase in mission activities as compared to the planned mission and a qualitative evaluation of overall spacecraft effectiveness.

Premission planning included preparation of a family of nonstandard events, including: the methods of detection, and verification; alternate procedures to be employed to reinstate the function or to compensate for an abnormal or nonstandard event; and the possible alternate mission in the event compensation was not possible. These premission plans were successfully used to carry out several alternate procedures.

Telemetry performance data, evaluated in real time to support the operation and confirmed by postmission analysis, indicated that there were several anomalous operations of interest to the subsystem analyst in evaluating spacecraft subsystem performance. Only one of these conditions, the abnormal operation of the high-resolution shutter, resulted in failure to fully satisfy the

	DESIRED TRAJECTORY		VELOCITY CHANGE (METERS PER SECOND) DESIRED	VELOCITY CHANGE (METERS PER SECOND) ACTUAL	ACTUAL TRAJECTORY	
TRANSLUNAR MIDCOURSE	AIM POINT	6509 KM	37.8	37.8	AIM POINT	6555 KM
LUNAR ORBIT INJECTION	Hp Ha INCL	199 KM 1850 KM 12.04 DEG	790.0	789.65	Hp Ha INCL	189 KM 1867 KM 12.15 DEG ±0.05
ORBIT TRANSFER	Hp Ha INCL	57.92 KM 1855 KM 12.04 DEG	40.2	40.15	Hp Ha INCL	56 KM 1853 KM 12.05 DEG ±0.10
ORBIT TRIM	Hp Ha INCL	40.0 KM 1824.0 KM 12.03 DEG	5.43	5.42	Hp Ha INCL	40.5 KM 1816.7 KM 12.0 DEG ±0.20

Table 1.3-2 Trajectory Change Summary

EVENTS	EXECUTED	PLANNED
MANEUVERS		
VELOCITY CHANGE	16	16
PHOTO	92	66
STAR MAP, CANOPUS & OTHERS	45	2
THERMAL PITCH OFF	77	10
ATTITUDE UPDATE	144	0
STORED PROGRAM COMMANDS TRANSMITTED	2522 (WORDS)	1850 (WORDS)
REAL-TIME COMMANDS TRANSMITTED	1988 (WORDS)	750 (WORDS)
MISSION I SITE PHOTOGRAPHY	156 (FRAMES)	180 (FRAMES)
FILM SET FRAMES		42
MISSION B SITES	15 (FRAMES)	
EARTH	2 (FRAMES)	
BACKSIDE	11 (FRAMES)	
NEAR PERILUNE	17 (FRAMES)	
MISCELLANEOUS & TEST *	10 (FRAMES)	
FINAL READOUT	211 (FRAMES)	212 (FRAMES)
PRIORITY READOUT	19 PERCENT	17 PERCENT

* INCLUDES 4 FRAMES WITH CAMERA THERMAL DOOR CLOSED

Table 1.3-3: Mission Event Summary

mission objectives. A brief analysis of the performance of each subsystem of the Lunar Orbiter spacecraft is presented in the following digest.

Photo Subsystem Performance

The photo subsystem is the most critical link in accomplishment of the primary objectives. An analysis of the telemetry data and the reassembled photographs indicates that the moderate-resolution camera operated satisfactory. Exposure of moderate-resolution photos of Sites I-5 and -9.2 were considered good. The remaining moderate-resolution photos ranged from slight to severe overexposure. The overexposure was predominately attributed to surface topography and albedo conditions, giving areas exceeding the luminance capability of the system. The majority of the high-resolution photographs were degraded by improper tripping of the high-resolution-camera focal-plane shutter. Analysis showed that all of the 156 high-resolution photographs of the primary photo sites had image smear. This smear took three specific forms:

1) In direction of spacecraft motion (IMC);

2) In the film advance direction;

3) Combinations of (1) and (2).

Postmission analysis of the smear characteristics, supported by a hardware test program at Eastman Kodak, confirmed that the high-resolution-camera's focal-plane shutter electronic logic-control circuitry was susceptible to radio-frequency and electromagnetic interference (RFI/EMI). The timing of the shutter operation (as determined by feature matching the high-and moderate-resolution photos and converting the measured displacement of the principle points to a timing interval) showed correlation with electronic transients attributed to the starting of the film-clamp and film-advance motors. Although the logic circuitry was susceptible to these electronic transients, nine of the filmset photos taken with the V/H sensor turned off produced high-quality nonsmeared high-resolution photos. These high-resolution photos were taken during the initial orbit on the front side and also from higher altitudes on the farside of the Moon.

The processor-dryer performed within established system tolerances. Some minor local-area film degradation was anticipated as a result of the repeated stopping and starting of the processing cycle.

Premission planning indicated that the expected temperatures and postulated power limitations would limit readout periods during the final readout phase to approximately two spacecraft frames per orbit. However, by closely monitoring the temperatures and power availability, this period was extended to the maximum time allowable between sunrise and Sun occulation. Thus, it was possible to complete the entire final readout sequence in a shorter time than had been planned.

During the priority readout phase, the experience gained in evaluating the processing and film-advance characteristics of the photo unit enabled the photo data analyst to precisely control the position of the spacecraft film in the readout gate. As the mission progressed, he was able to control the exact position of the spacecraft film at the readout gate to within a few tenths of an inch.

During the 28 days of operation only the one major photographic subsystem abnormality was observed. The high-resolution-camera focal-plane shutter control circuitry now employs an integrating technique to ensure that the signal received is a valid command pulse (amplitude and duration) and not an electronic transient. This circuit design change has been verified by test and is incorporated in the remaining photo subsystems.

Electrical Power Subsystem Performance

All electrical power for Lunar Orbiter I was provided by a solar-panel and storage-battery system. The spacecraft was placed on internal power at T-7 minutes, the batteries supplied all power demands until the solar panels were deployed and the Sun was acquired after injection into cislunar trajectory. Thereafter, the solar panels supplied the electrical power when illuminated by the Sun, and the batteries furnished power during Sun occulation. The depth of battery discharge just prior to Sunrise was nominally 30%, and the corresponding battery output voltage was 24.96 volts.

During the early part of the mission, the solar array output current exceeded all the system demands when the array was oriented within 60 degrees normal to the Sun. The excess electrical energy was dissipated through load resistors mounted on the low-gain-antenna boom. The maximum power dissipated through the load resistors was 260 watts during a 0.28 - ampere trickle charge.

On August 17th the first irregularity in the electrical power subsystem occurred during Orbit 20. An increase of 1.26 amperes in the load current after Sunset was observed in the telemetry data. This additional load was confirmed by an increase in the depth of the discharge occurring during the lunar night period and the proportional increase in amount of energy required to charge the batteries to full level. Analysis of these data indicated that the most probable cause of the additional load was a power transistor short. Although this short continued for the remainder of the flight, the additional battery-discharge level was still within the safe operating range of the system and no further effects were noted.

Communications Subsystem Performance

The Lunar Orbiter communications subsystem received and properly executed 4510 commands and transmitted the video data equivalent to 251 frame pairs during the 46 priority and 93 final readout periods. During the entire 35-day space mission, there were only two instances when early handover from one station to the next was necessary. The first occurred at Woomera Deep Space Station when trouble was experienced in acquiring the spacecraft. A slight reduction in the transmitter frequency enabled the station to receive and maintain lock on the spacecraft transponder. When the handover to the Madrid Deep Space Station occurred the problem was not evident. The second incident occurred as a result of a power supply failure at the Woomera Deep Space Station.

Transmission of the video data via the traveling-wave-tube amplifier (TWTA) and the high-gain antenna produced very satisfactory results on the films recorded at the Deep Space Stations. At all times the received signal levels remained well above the required thresholds for continual data integrity. Timing correlation between Deep Space Stations was accomplished using Lunar Orbiter I ranging data as the basic reference for the first time on the DSN. All data indicate that this function was satisfactorily completed.

Although the spacecraft thermal environment was more severe than expected, the performance of the TWTA throughout the mission exceeded or equaled the expected nominal values. The total operating time on the amplifier from launch to the end of the final readout period was 211.1 hours. During this time it was cycled 148 times. The average length of time for a final readout period was 106.5 minutes per readout. Power output during these readouts was normally between 10.5 and 12.2 watts with very little change apparent throughout the entire readout period. During the mission 907 one-degree antenna rotations were commanded and performed.

As stated in the configuration section, the high-gain-antenna boom was adjusted, with respect to the spacecraft axes, for the nominal alighment of the spacecraft with the Sun. During the mission it became necessary to operate the spacecraft off the Sunline to maintain spacecraft thermal control. The signal margin prediction program (SGNL) was used at the SFOF to determine the high-gain-antenna pointing angles necessary to ensure reception of video signals at the Deep Space Station. This program was based upon the spacecraft maintaining the Sun-Canopus reference attitude. Thus, when the spacecraft was commanded to deviate from this orientation the outputs from this program were not usable. During the mission, a method was developed to relate the actual spacecraft attitude to the Sun-Canopus reference to ensure that the high-gain antenna would be pointed toward a Deep Space Station. As demonstrated by continuous reception of the video data during transmission, the high-gain antenna was properly pointed during the readout periods.

During the early portion of the mission, the high-gain antenna was used as an alternate mode for Canopus verification. By using the procedure outlined in the nonstandard events premission plan, it was possible to verify the spacecraft roll attitude and the location of the navigation star, Canopus, relative to the spacecraft axes.

With the exception of a few periods when the signal strength decreased slightly (which has also been noted during ground testing during temperature changes), the communications system performed flawlessly during the entire mission.

Attitude-Control Subsystem Performance

The attitude-control subsystem provided the capability to store, select, command, and execute all of the 4510 commands received from the Space Flight Operations Facility. The flawless processing of these commands, the execution of 35,000,000 programmer logic cycles, and the precise execution of all attitude commands demonstrates conclusively the capability and performance of this subsystem.

During the mission there was an indication of abnormal operation within the attitude control system. During the initial Canopus-acquisition sequence when the Canopus tracker was turned on and the vehicle rolled the 360 degrees to establish the star map, the data indicated that the output level was extremely high and that the map produced was not as expected. Appropriate alternate procedures were instituted, based on the premission planning for recovery from nonstandard events. These procedures permitted the completion of the Lunar Orbiter I mission with no degradation in accomplishment of mission objectives.

Canopus Acquisition

The planned sequence of events for establishing the spacecraft attitude during the cislunar trajectory was a combination of two maneuvers. The first was an alignment upon the Sun based on the error signals obtained from the Sun sensors. The cislunar trajectory and the Sunline then established the attitude in space of the spacecraft pitch and yaw axes. Determination of the roll attitude was to be obtained and maintained by the star tracker searching for, acquiring, and locking on the Canopus. The attitude control subsystem design included an inertial reference system to maintain attitude reference and control at any time that the spacecraft was commanded off the Sunline. Thus, it was possible at all times, by using either the Canopus acquisition mode or the inertial reference mode, to determine the spacecraft roll attitude. Immediately prior to executing any commanded maneuvers the exact roll attitude of the spacecraft must be known. The spacecraft maneuver commands were given as increments from initial known attitude conditions.

The unexpectedly high star map voltage observed during the initial turn-on sequence indicated that an extremely strong light was entering the tracker optical field of view. As the spacecraft continued on its flight to the Moon, several tests were made with the Canopus tracker to gather additional data from which to analyze the performance and to determine the cause of the high signal level.

The spacecraft was commanded to roll to a computed angle in an attempt to lock on the Moon. This experiment was satisfactorily completed, indicating that the Canopus tracker itself was operating satisfactorily. Other attempts were made to acquire Canopus during the cislunar trajectory. Some of these tests were successful, others were not, but all indicated the presence of high background light. Additional tests were conducted during the first period when the Sun was occulted while the spacecraft was visible to Earth. During this test, the spacecraft was commanded to acquire Canopus. Aquisition was accomplished with normal output indications. As the spacecraft emerged into the view of the Sun, the Canopus lock was lost, apparently due to presence of the reflected light. The conclusions from this series of tests were:

1) The Canopus tracker was operating completely satisfactorily at all times;

2) The tracker could not acquire and stay locked on Canopus while the spacecraft was being illuminated by the Sun due to the reflected light;

An alternate procedure was developed and employed to maintain roll attitude control during the mission as follows: Approximately 10 minutes before Sunrise the spacecraft was commanded to a roll position where Canopus was visible to the star tracker. With the roll axis in the inertial hold mode, the roll position and roll drift rate was determined by the subsystem analysts from the telemetered tracker outputs. These factors were primary inputs for the computation of each photo or velocity change maneuver which was subsequently transmitted to the spacecraft for execution.

Velocity Control Subsystem Performance

All spacecraft velocity changes after separation from the Agena booster were accomplished by the velocity control subsystem. Spacecraft velocity changes were required at four different times during the mission: midcourse correction, initial ellipse orbit injection, transfer to the second ellipse, and the adjustment to the third ellipse.

The engine gimbal actuators performed satisfactorily in controlling the engine pointing. The response to the commands from the attitude control subsystem was as expected and the performance telemetry data indicated that attitude stability was maintained throughout all maneuvers.

Velocity control subsystem performance is summarized in the following table.

	VELOCITY CHANGE (mps)	BURN TIME (sec)	THRUST (lbs)	SPECIFIC IMPULSE LB (sec/lb)
MIDCOURSE				
PREDICT	37.8	32.7 ±1	99.8	275.2
ACTUAL	37.8	32.1	101.6	276
INJECTION				
PREDICT	790.0	588.3 ±10	99.75	274.7
ACTUAL	789.7	578.7	101.3	276
TRANSFER				
PREDICT	40.15	22.7 ±1.6	112.6	274.5
ACTUAL	40.2	22.4	113.6	276
2ND TRANSFER				
PREDICT	5.43	3.0 ±1	114	
ACTUAL	5.4	3.0	113.6	

The increase in the actual and the predicted thrust levels for the third and fourth burn periods was attributed to the nitrogen pressure regulator leakage.

Following the injection into the first ellipse, a leak in the gas regulator for the velocity control propellant tanks was observed. This leakage caused the pressure in the propellant tanks to rise at a rate of 6 psi per day.

A review of the qualification test records for this regulator showed that it had passed all of the leakage tests within the allowable specifications and that no irregularities had been observed. It was concluded that the excess leakage exhibited by the regulator was an isolated case most probably caused by a foreign particle that prevented the regulator from seating properly. The leakage rate detected was not serious enough to affect the spacecraft operation or completing the mission objectives.

On August 27th, the shutoff squib valve was activated as planned to isolate the velocity control system from the nitrogen pressurization system and no further evidence of the gas leakage was detected. This action sealed the propellant tanks at approximately 231 psi pressure behind the remaining propellants. The additional propulsion impulse remaining in the system was to be used during the extended mission as needed. The propellant tank heaters were activated on seven occasions for a total of 698 minutes.

Structures, Mechanisms, and Integration Elements Performance

All of the components of this subsystem performed as anticipated except for the thermal control elements.

Telemetry data confirmed that the antenna and solar-panel deployment was accomplished at the prescribed times. Telemetry and photographic data confirmed that the camera thermal door opened and closed at the proper times and fully supported the photographic mission.

Analysis of the vibrational data from the accelerometers located on the equipment mounting deck indicated that the launch vibrations were generally less severe than the vibrations applied during the spacecraft flight acceptance tests. The upper module accelerometer data channel produced no data. At this time there is no explanation for the failure of this accelerometer data channel.

Temperatures of the equipment mounting deck were higher than expected as a result of thermal coating characteristics. With the exception of the solar panel temperature thermistor on Panel 3, all telemetry instrumentation monitors performed satisfactorily throughout the flight. The thermistor on Panel 3 was intermittent after Orbit 34 in the first ellipse.

Two anomalies produced thermal effects during the flight: (1) a shorted electrical subsystem transistor, and (2) the thermal coating performance. Neither of these affected the overall performance of the Lunar Orbiter I mission. The shorted transistor caused a steady loss of about 15 watts of electrical energy that had to be dissipated by the spacecraft. Performance telemetry data indicated that the equipment mounting deck temperatures rose at a faster rate than had been anticipated while the solar panels were aligned with the Sun. It was believed that the thermal control coating was degrading in the presence of ultraviolet radiation at a higher rate than predicted. To maintain the spacecraft temperatures within the design range, the spacecraft was oriented off the Sunline as necessary to reduce spacecraft temperatures.

1.3.4.4 OPERATIONAL CONTROL PERFORMANCE

The most exacting standard of measuring the operational control of a complicated mission is the ability to meet mission objectives with a minimum of difficulty. The speed and accuracy with which situations were identified, evaluated, and a course of action initiated is adequate demonstration of the absolute control and discipline maintained. Premission operations and data collection planning provided the tools and procedures to accomplish this fast reaction. The inherently high flexibility of the spacecraft command programming system was repeatedly demonstrated. This is evidenced by the activities summarized in the following table:

EVENTS	EXECUTED	PLANNED
MANEUVERS		
VELOCITY CHANGE	16	16
PHOTO	92	66
STAR MAP, CANOPUS & OTHERS	45	2
THERMAL PITCH OFF	77	10
ATTITUDE UPDATE	144	0
STORED PROGRAM COMMANDS TRANSMITTED	2522 (WORDS)	1850 (WORDS)
REAL-TIME COMMANDS TRANSMITTED	1988 (WORDS)	750 (WORDS)

Preflight planning included the preparation of a document Flight Operations Plan for Lunar Orbiter. This document contained a detailed sequence of events for all activities associated with the planned mission. Included in the plan were all flight programmer core maps required to conduct the mission. The plan was supported by an operational software program, Mission Sequence of Events (SEAL). Up until the time of the attempted initial Canopus acquisition, the mission proceeded in accordance with the planned sequence of events. From this point on, it was apparent that the preflight plan would be modified extensively to satisfactorily complete the mission objectives.

The mission director, supported by his staff of mission advisers, and the flight path and spacecraft analysis and command teams took full advantage of all the design flexibility of the Lunar Orbiter hardware, software and planned procedures, and spacecraft command and programming system during the remainder of the mission.

Operational Personnel

The Flight Operations group was divided into three teams (designated red, white, and blue) to provide 24-hour coverage of the flight operations at the Space Flight Operations Facility. Sufficient overlap between the shifts was scheduled to allow detailed coordination between the on-coming and off-going system analysts. Wherever possible, the team changes were made during Earth occultation, when spacecraft activity was at a minimum. Twenty-four-hour coverage was also required at the Deep Space Stations. This coverage was provided by scheduling the three stations to sequentially support the mission during the period while the spacecraft was in view. The operations team at the SFOF was also supported, on a 24-hour standby basis, by a team of design specialists operating through the Operations Center in Seattle, Washington. The flexibility of Lunar Orbiter operational personnel was thoroughly demonstrated during the conduct of this mission. Corrective action implemented to compensate for the isolated nonstandard events encountered during the flight, coupled with the change in the operational employment of the film-set photographs, added an extensive workload to the planned functions to be accomplished during the mission. The magnitude of these deviations is best illustrated by the previously tabulated comparison of planned and actual spacecraft maneuvers during the mission (total planned : 94; total executed : 374).

Incorporation of the above activities into operational control of the mission required extensive revisions to the preplanned flight programmer core maps. The large number of revisions made it impossible to project the programmer loading plan as far in advance as desirable.

The primary task of the mission event coordinator was to coordinate and schedule predicted events, avoid conflicts, and facilitate the conduct of the mission in an orderly and optimum manner. To accomplish this function, the mission event coordinator maintained close operational contact with the Space Flight Operations Director and other members of the operations team, particularly the FPAC and SPAC directors, supervisor of net operations, and the DSIF operations project engineer. The mission sequence of events was prepared using the IBM 7094 computer and the mission sequence of events program (SEAL). To permit timely revisions to the mission events sequence, a computer program has been designed to store the maximum amount of information that will not change from mission to mission. This information consists of standard inputs, format data, and test time intervals, which are placed in a master file (common environment) that is accessible by the programmer.

Copies of the sequence of events were distributed to all mission control personnel as revisions to the flight plan were determined. Early in the mission it became apparent that information and activities planned for each DSN station did not provide a complete enough picture of the overall flight plan. As a result, a greater part of the activities affecting all Deep Space Stations were transmitted to each station.

Premission planning included establishment of programmer core maps and command generation software simulation of the entire mission. Commands for countdown tests and initial loading of the flight programmer were prepared and sent to the Cape Kennedy Deep Space Station. In addition, the Mode II commands (real-time commands) were prepared prior to the mission and sent to the Goldstone, Woomera, and Madrid stations for use during the mission. It soon became apparent as the mission progressed that the preplanned core maps and command generation program simulation could not be followed precisely. Thus the plan has been changed for future missions such that the command generation program simulation of the entire mission will not be attempted and stress will be placed on making the core map as the mission progresses.

Command preparation activity during the mission became on occasion an undesirably hurried operation, due to the late definition of exact inputs or requirements, late changes, or requirements for programmer core storage in excess of an efficient level. This necessitated programmer work-around methods. While all of these requirements were met, there were some cases of spacecraft core maps being stored at the very last minute. As a result, the command programmers were unable to plan flight programmer memory utilization very far ahead. Since the analyst could not take the time to optimize core maps, the number of commands required in some sequences exceeded the minimum required.

Command preparation directives were issued in accordance with standard procedures for most of the photography and propulsion maneuvers. However, due to the pressure of time during the planning for some "film set" photographs, and in some of the spacecraft diagnostic testing activities, these exact procedures were not followed. Despite the increased number of commands required by the additional maneuvers and photographs than originally planned, the procedural method of checking the command instructions (core maps) for accuracy prior to transmission as well as the careful bookkeeping of actual programmer status prevented any serious command errors. Only a few minor instructions were not properly executed by the spacecraft on the intended schedule.

The command transmission procedures used on Mission I were very adequate although in some instances redundant. These procedures required the transmission of commands from command coordinator to the DSIF trackchief, then to the Deep Space Station manager, and finally to the senior Lunar Orbiter engineer at the site. During some of the peak spacecraft real-time command activities, the command coordinator contacted the senior Lunar Orbiter engineer at the site directly because there was insufficient time to follow procedures and get commands transferred to the spacecraft. On future missions, procedures will be revised so that the command coordinator will coordinate most of the planned activities directly with the senior Lunar Orbiter engineer.

Experience obtained during this first mission identified the requirement for an additional function of the space flight operations directors staff. During the photo taking portions, the decisions in the use of the "film set" times were in some cases approved late in relation to the time of the intended activity. Coordination between the teams became difficult when the operations team was required to plan for the change as well as monitor the execution of the normal functions of spacecraft activity. On future missions an off-line group will be formed to plan and coordinate these changes in the flight plan as a parallel effort to conducting the actual operation. Thus the operating team's work load will be considerably reduced, enabling them to concentrate on the command implementation and evaluation of the performance of the spacecraft.

Routine Operational Control

Satisfactory mission control can be divided into two basic areas. The first is continual monitoring of the spacecraft operational and performance status to ensure that it can continue to execute the mission objective. The second is transmission of the necessary instructions to the spacecraft for execution at the proper times to obtain the mission objectives. Both of these functions were satisfactorily accomplished on the entire mission.

Premission operational plans contained a detailed sequence of events for all activities associated with the 35-day mission. Included in these plans were all the flight program core maps, as a function of time, required to conduct the mission. The mission proceeded precisely according to this plan until the first attempt to acquire Canopus was initiated. The resultant nonstandard operation caused modifications to the plan for conducting the mission. Although changes in the timed sequence of events and the mode of operation were implemented, the method of conveying these commands to the spacecraft was accomplished in the normal manner.

Diagnostic test sequences were introduced to identify the Canopus star tracker and the high-resolution-camera system problems. Upon completion of these test sequences the alternate operational control procedures were integrated into the sequence of events and executed by the normal control procedures. The detail approach to this diagnostic testing is described under Diagnosis and Evaluation paragraphs.

The location of the primary photographic sites on the lunar surface, together with the Lunar Orbiter's flight path information, defined a sequence of exposure times. These precise photographic times were used as the initial conditions for determining the time to establish the roll attitude update maneuver and spacecraft photo maneuver commands to take the desired photographs.

Spacecraft telemetry data were received at the SFOF at all times when the spacecraft was in view of the Earth. To assist the subsystem analyst in continually monitoring the operational status of the spacecraft and its subsystems, automatic and computerized aids were used. These aids included the following.

Visual Alarm Monitor Panel

An immediate visual display of certain preselected telemetry channels which exceeds specified limits, this panel automatically displays any change of state of selected digital discrete channels.

Data Plotters

Milgo and Dymec plotters were installed in the operations area to provide time history plots and user program plots of any of the selected channels desired by the analyst.

High-Speed Bulk Printers

The high-speed printers provided tabulated records of telemetry channels in engineering units or computer output for use by the analyst as requested.

Raw Telemetry Printers

Sixty wpm teletype printers at each analyst position provide a printed real-time record of raw telemetry counts in hexadecimal format, in a selected edit table. During two-station view periods, two edit tables can be made available.

Processed Telemetry Printers

One-hundred-wpm teleprinters (nine total) were available that printed out the data as processed and converted by the IBM 7044. A variety of different formats can be called up and changed as the analysts desires to monitor any selected T/M channels.

Computer Tabulations and Plots

This was a working copy or permanent record of the output of the IBM 7094 or 7044 computers. The functions contained on these tabulations and plots were as requested by the individual analyst.

The requirement to continuously know the exact operational status of the spacecraft requires the computation of many parameters based on the telemetry data. In some instances it involved relationships of many data channels to define the subsystem operational status. Computer programs were developed to assist the analyst in these computations. These programs were primarily of an engineering nature rather than a data processing nature. The programs are stored on the 1301 disk and are loaded into the core by a monitor when they are to be executed. These user programs are classified into three major groups: spacecraft performance and command, mission integration control, and flight path analysis control. Two files were maintained on the disk and used extensively by the user programs. These files are called the master data table and the common environment. The master data table is a part of the disk reserved for the storage of time-tagged decommutated telemetry data. The disk is so arranged that 18 hours of telemetry data can be maintained for each measurement. The common environment is a special data file on the 1301 disk that contains current values of all parametric and similar quantities required by Lunar Orbiter user programs.

Spacecraft operational status control was established and maintained by a combination of the telemetry performance data and the SPAC user programs, which were interpreted and analyzed by the subsystem specialists. The SPAC programs used to accomplish this function are summarized as follows.

Star Identification Programs

These programs are used by the attitude control subsystem analyst to provide a means for establishing the spacecraft roll attitude whenever required. One of the subprograms produces an a priori star map simulating the anticipated output of the Canopus tracker. This data is compared by a second program that correlates the simulated map with the map obtained from the telemetry signal from the Canopus star tracker. The third program calculates the necessary maneuvers to point the spacecraft high-gain antenna to the available Deep Space Station. This latter program can also be used in reverse (i.e., to determine the spacecraft roll attitude resulting from a peak in the signal strength recorded at a Deep Space Station).

Gas Budget and Vehicle Dynamics

This program assisted the attitude control program analyst in evaluating and predicting Lunar Orbiter I attitude and velocity control subsystem performance. Spacecraft performance telemetry, design parameters for the vehicle and spacecraft were combined to determine the nitrogen gas, fuel, and oxidizer consumption and to provide a past history of the spacecraft dynamic performance. Using these same data, the program can also be used in a prediction mode to determine the nitrogen gas and change of velocity capability remaining at real or future time.

Signal Margin Predict Program

The communications subsystem analyst used this program to predict the performance of the two-way telecommunications system and the video transmitting systems. The computer program analyzes the actual strength of the received signal at the Deep Space Stations and at the spacecraft for subsequent comparison of predicted values for corresponding time periods. In the predict mode this program: (1) predicts the signal-to-noise ratio for selected telecommunication modes as a function of time; (2) computes the pointing error of the high-gain antenna for each Deep Space Station and the required rotation angle; (3) computes, based on the pointing error, the single spacecraft maneuver and associated antenna rotation(s) for each Deep Space Station view period which would minimize the pointing error of the high-gain antenna.

Electrical Power and Energy Management

The power subsystem analyst used this program to assist him in his prediction of the electrical power subsystem performance. Performance telemetry data and system characteristics were combined to evaluate the electrical power system. The major parameter calculated during each iteration was the battery state of charge. Future performance of the electrical power subsystem was predicted based on the mission profile, nominal design data, thermal program inputs, and a given sequence of events.

Thermal Management Program

The thermal analyst used this program to monitor and predict the thermal behavior of the spacecraft. Performance telemetry data was tabulated, with appropriate flagging of any questionable data. Spacecraft temperature as a function of time was predicted by the solution of a set of thermal finite difference equations subjected to a set of initial conditions, boundary conditions, and spacecraft event sequence. A status mode was also used to provide performance summary data for analysis.

Mission Integration and Control

These computer programs differed from the user programs above in that they do not serve as an analysis tool for any particular spacecraft subsystem. The outputs are used as aids in mission control and as such are used by several support areas. They satisfied the need for common information in all mission support areas and ensure that requirements and actions of each area are compatible with spacecraft design and mission objectives.

Mission Sequence of Events (SEAL)

This program provided a means by which the many events and activities essential to the conduct of the Lunar Orbiter mission were displayed in proper time in event ordered sequences. The SEAL program produced a display in two forms, a time-ordered script and a bar chart form. This data indicated when, where, and by whom specific tasks are performed. It also provided abbreviated scripts that contained selected events of interest to the Space Flight Operations Facility or the Deep Space Stations.

Spacecraft Time/Greenwich Mean Time Correlation Program

This program provided the means of correlating the 20-bit binary spacecraft time code (which is cycled every 29.127 hours) with Greenwich Mean Time. The correlation was made using the spacecraft time code at the beginning of each telemetry frame. Predicted times for use in maneuver-commands and photo sequencing, and final determination of camera on time, were made based on this time correlation function.

Command Generation and Programmer Simulation (COGL)

The COGL program was used within the SPAC operations area for the encoding and verification of command sequences, prior to their transmission to the Lunar Orbiter spacecraft. The program had four major routines: a command assembler, a programmer simulator, a teletype compatibility routine, and a control monitor. The command assembler translated the symbolic programmer command to appropriate binary commands. The programmer simulator maintained the current program format sequences. Teletype compatibility was required to ensure the command sequences were compatible with the teletype transmission mode. Finally, the control monitor served as an executive routine and directed the execution of the previous three functions.

Data and Alarm Summary (DATL)

The data and alarm summary provided the capability to display or make available the telemetry data from the master data table. The program performed the following functions:

1) Displayed in either raw counts or engineering units (depending on the analyst requirements);

2) Presented appropriate flags on the tabulated data if the data exceeded the preset alarm limits;

3) Identified any data that had been quality flagged by the command telemetry data handling system programs;

4) Provided summaries that indicated the general condition of the telemetry data.

Common Environment Utility Program

This program provided communication between the user areas and the common environment area of the 1301 disk. The program allowed the user to extract any of the data in the 1301 disk common environment area for use with the program that he had selected.

Command Transmission

To ensure that the commands received within the spacecraft were indeed the commands generated at the SFOF, a series of checks and verifications was designed into the system. The command generation sequence at the SFOF involved extensive coordination with all analysts involved – this included determining spacecraft capability to execute the desired commands, whether the command would interfere with other commands already in the spacecraft, and the exact format in which the command must be transmitted. Stored program sequences of commands were transmitted from the SFOF to the Deep Space Station by teletype messages in the form of a command triplet. The command triplet is a self-contained check on the accuracy of the command being transmitted. (Any noise or disturbance from the transmission line could affect one of the command words, but not all three in the same place. Verification circuitry of the Deep Space Station was set to compare each of the three commands.) Upon satisfactorily comparing these three commands, a verification message was trasmitted to the SFOF. The Deep Space Station, in turn, punched on tape the command received for transmission to the spacecraft.

The command as received by the spacecraft was decoded and temporarily stored in the programmer register. At the same time, the decoded command was retransmitted to the Deep Space Station via the performance telemetry system. After decommutation of the telemetry data, the command from the spacecraft was compared with that transmitted from the Deep Space Station. A satisfactory comparison of these two commands is required before the transmission of an execute tone. Upon receipt of this tone, the spacecraft shifted the stored command out of the register into the programmer for storage or for execution at the time commanded.

Real-time commands were directed by voice from the SFOF by one of two methods:

1) Designation of prepunched tape on hand at the DSS (Mode II);

2) Specification of the true bits of the command words to be set manually at the command equipment (MDE) at the DSS (Mode III).

Diagnosis and Evaluation

The approach used to identify and diagnose the in-flight anomalies experienced is described as follows:

Canopus Star Tracker

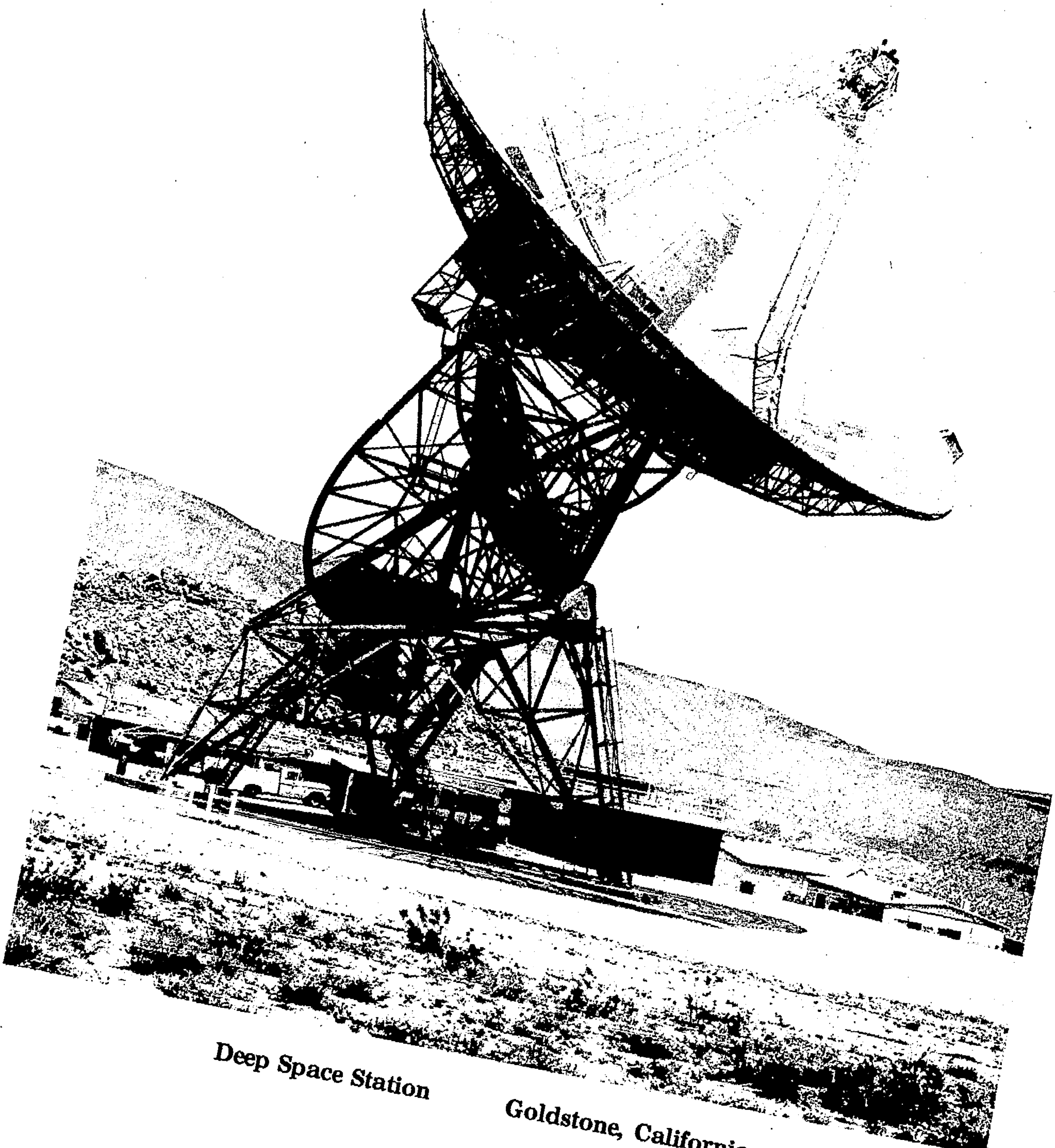
One of the failure modes considered in preflight planning was an alternate procedure to be used in the event that the navigational star, Canopus, could not be acquired. It did not, however, cover the exact type of incident encountered. An alternate method, consisting of using the high-gain antenna and monitoring the received signal strength at the Deep Space Station, was used to assist in establishing the spacecraft roll attitude during the diagnostic testing.

When the Canopus tracker was initially turned on, the mapping voltage was much higher than expected as the spacecraft was rolled through the commanded 360 degrees. During this first star mapping maneuver (which included commanding the spacecraft to execute certain maneuvers, while turning the Canopus tracker on or off to satisfy test conditions), the mapping voltage saturated when the tracker was pointed toward the Moon. However, there was no evidence that the signal change for the star Canopus had been detected. During the third attempt to make a star map, the high-gain antenna was set at 38 degrees prior to the spacecraft roll maneuver. During this roll, the antenna map and star map were made simultaneously. As in the previous test, the star tracker output saturated at the Moon, but Canopus was not seen. The data from the antenna test were used to determine the relative roll position of the star Canopus.

Additional evaluation of the data obtained indicated the possibility that light was being reflected into the star tracker from the low-gain antenna. To verify this theory, the spacecraft was rolled to an attitude that positioned the low-gain antenna in the shade. The results of this test were promising but were not conclusive and further tests were deferred.

Evaluation of the available data indicated that the star tracker was operating properly and that stray light being reflected into the tracker prevented it from identifying the stars. Therefore, it was decided to use a different celestial body to determine the roll attitude of the spacecraft for the midcourse maneuver. A maneuver was set up to verify that the star tracker could detect, lock on, and track the Moon. This experiment was successful and plans were initiated to perform the midcourse maneuver using the Moon as the reference for establishing roll attitude. The accuracy of this alternate mode of operation was demonstrated by the fact that the spacecraft required no further midcourse corrections to achieve the desired aiming point in the vicinity of the Moon.

As the spacecraft continued its journey to the Moon, additional attempts were made to acquire Canopus. Tests were also accomplished to determine the proper operation of the bright-object sensor in the star tracker circuitry. During these tests, it was possible for short periods of time to acquire Canopus, thus confirming that the tracker itself was operating satisfactorily, In addition, as the spacecraft was rolled toward the Moon and also yawed toward the Sun, the proper performance of the bright-object sensor was verified. During the first Sun occultation during the initial lunar ellipse, it was possible to again attempt to acquire Canopus with the spacecraft completely in the shade. This test was completely satisfactory and the star was identified, locked on, and tracked; however, when the spacecraft again came into the Sun the "glint" appeared, causing the loss of track signals. This last incident was a conclusive test that the theory was completely valid. For the remainder of the mission the spacecraft was operated in the "inertial hold" mode whenever it was in Sunlight. An attitude update maneuver was accomplished while the Sun was occulted to update the roll reference prior to any maneuver.



Deep Space Station Goldstone, California

High-Resolution Camera Focal-Plane Shutter

Initial photo video data readout indicated the presence of image smear on the photographs taken with the high-resolution camera. Initial interpretation of these photos indicated that the velocity-to-height (V/H) sensor apparently was triggering the shutter prematurely. The first high-resolution photograph was also a double exposure —one exposure was taken earlier than intended and the second exposure at the commanded time. Neither of these pictures showed evidence of image smear. The remaining pictures of the initial series showed evidence that the shutter was operating at an improper time. A series of tests was proposed in an attempt to obtain more information on camera operation.

To evaluate Exposure 26 (the first picture of the four-frame sequence after photographing Site I-0), ten additional exposures were made for diagnostic purposes. One test involved the use of different exposure rates with and without the V/H sensor turned on. A second test was used to determine that the V/H sensor was in fact causing the abnormal shutter operation. This was accomplished by the following:

1) Camera thermal door was open and the V/H sensor was turned on;

2) The sensor was left on for approximately 2 minutes and then turned off;

3) The camera thermal door was then closed and the camera shutter was commanded to take a picture with the door closed and to move fresh film into the camera for the next photograph.

The results of this test could reveal one of two things (1) If the V/H sensor was not causing the abnormal operation, both exposures (the moderate-and high- resolution photos) would be blank; or (2) If the V/H sensor were contributing to the abnormal operation, a high-resolution picture would be taken during the time that the thermal door was opened, but the moderate-resolution frame would be blank. Readout of these photos confirmed that a picture was taken on the high-resolution frame and not the moderate, thus confirming that abnormal operation was occurring when the V/H sensor was on.

In accordance with plans previously developed, a study was to be made after the readout of the initial-ellipse photos to determine whether to go to a lower altitude. Based upon available data, it appeared that a higher V/H sensor output voltage would be generated that would be above any noise level and thereby improve the performance of the shutter. It was decided that the mission should continue as originally planned.

Prior to the transfer to the lower orbit it was learned that one of the early frames had evidence of Bimat stick. Although none of the film handling constraints that relate to Bimat stick had been exceeded during the mission, there was some concern that the constraint was incorrectly stated. This constraint required that fresh Bimat be placed on the processor drum at least every 15 hours and was being observed by processing two frames every four orbits. On the basis of this report, a decision was made to use additional film to permit processing every orbit. This required that an additional eight photographs be taken.

As a result of the extra photos taken to evalute and analyze the high-resolution-camera shutter operation, it was necessary to revise the photographic coverage for the remainder of the sites. Instead of taking 16-frame exposures on all of the primary sites, it was determined that eight exposures would be taken of Sites 4, 6, and 8. The remaining sites would have the normal 16-frame sequence coverage.

After photography of Sites 1 and 2, priority readout data indicated that the high-resolution shutter was still operating erratically and further diagnosis was desired. In support of this investigation, tests were accomplished on a photo subsystem at Rochester, New York, while the mission was being flown. These tests did not duplicate the exact flight failure but did indicate a higher V/H rate might eliminate the problem encountered. It was recommended that the V/H sensor output be increased to at least 37.5 milliradians per second. The planned altitudes for covering Sites I-4 and I-5 were expected to provide this effective ratio. However, after evaluating the photography of Site I-3, it appeared that the spacecraft was at a higher altitude than had been anticipated. This fact was also confirmed from Site I-4 photography. Therefore, a decision was made to transfer to a third ellipse after photographing Site I-5. The third ellipse was planned to obtain the desired V/H ratio of 37.5 milliradians per second but not exceed a maximum of 50 milliradians per second.

Although it was hoped that the above series of tests would provide improved operation of the high-resolution-camera shutter, subsequent investigation indicated that the focal-plane-shutter logic control circuitry was susceptible to electromagnetic interference and therefore could not be corrected by changing the operating procedures during the flight.

During these tests many historic pictures were taken by Lunar Orbiter I. The command and maneuver requirements were developed to take, in nearly real-time, such pictures as the morning and evening terminator, the Earth as seen from the vicinity of the Moon, numerous pictures of the farside of the Moon, and additional photographs of sites of interest on the frontside. Such areas as potential targets for Mission B, major craters, and mare and upland areas useful as Apollo navigation landmarks were photographed. In most cases it was possible to satisfy all of the requirements to take these photos. In some isolated instances, however, the photo occurred when the programmer was in an improper mode of operation to supply the time of exposure and there was no timing code exposed on the film. In those cases the commanded time was used as the reference.

Spacecraft Temperature

During premission planning it was determined that high equipment-mounting-deck temperatures would require the spacecraft to fly off the Sunline. Based on this planning, it was decided that this maneuver would be implemented in real time as necessary and would not be programmed into the mission sequence of events.

The equipment-mounting-deck thermal coating underwent a greater pigmentation change due to solar radiation during the flight than was predicted from simulated space environmental test data. As a result, the spacecraft temperatures were higher than expected during the entire mission. It was concluded that the actual solar absorptivity coefficient was higher than predicted in the cislunar phase and the apparent degradation rates of the paint were more severe than anticipated.

The increased spacecraft heat absorption was counteracted by pitching the spacecraft off the solar vector by a predetermined angle whenever cooling was required. These maneuvers effectively controlled spacecraft temperatures to the desired levels.

These cooling maneuvers produced interacting effects with other spacecraft subsystems and required close coordination as the mission progressed. The electrical power subsystem received a reduced amount of converted solar energy during these maneuvers. (In all cases the batteries were fully charged prior to Sun occultation.) The maneuvers increased usage rate of nitrogen. Reorientation of high-gain-antenna pointing angles was required as a result of the thermal relief maneuvers to ensure video-data reception at the DSIF.

Although the temperature environment increased the workload on the anlysis team, it did not affect the ability of the spacecraft to accomplish mission objectives (except insofar as the residual supply of nitrogen gas was reduced).

Trajectory Control

The Lunar Orbiter trajectory was controlled during the booster phase and injection into cislunar orbit by the launch-vehicle guidance and control system at AFETR. After acquisition by the Deep Space Station at Woomera, Australia, trajectory control was maintained by the Space Flight Operations Facility in Pasadena, California. For the first 6 hours of this period, the Deep Space Network performed orbit determination calculations to ensure acquisition by the DSS. Guidance calculations and trajectory control were performed by the Lunar Orbiter Operations group. The trajectory was maintained by establishing a velocity vector during periods of acceleration or deceleration.

Unlike the SPAC and MIC software, the FPAC programs do not operate directly under the SFOF monitor. They operate under a submonitor or supervisory program called, "Jet Propulsion Laboratory Trajectory Monitor" (JPTRAJ), which in turn operates under the SFOF monitor. There are two advantages to using JPTRAJ: (1) The user can sequence a group of user programs where a run can be handled externally at execution time, and it is not generic for individual programs; (2) Not only is the execution order set up externally, but also any specific input computer data may be sent from one program to another during execution. The individual FPAC programs are termed computational blocks and the combinations of these programs under the JPTRAJ monitor are termed user programs. FPAC support functions are described in the following paragraphs.

The FPAC team at the SFOF monitored the Lunar Orbiter flight path, proposed flight path corrections or changes, and predicted future flight paths and orbits. FPAC provided a choice of spacecraft maneuvers for each event involving thrust orientation or camera pointing. FPAC also specified the time of occurrence for each propulsive maneuver or photo event. These functions were performed using DSN tracking data, SFOF computer services, and Lunar Orbiter software (computer programs).

FPAC activities during a normal Lunar Orbiter mission were divided for convenience into the seven phases listed below:

1) Acquisition and launch evaluation;
2) Midcourse decision;
3) Pre-midcourse;
4) Preinjection;
5) Pretransfer;
6) Prephoto;
7) Photo and readout.

Acquisition-phase FPAC activity was managed by the JPL FPAC director. A JPL FPAC team was supported by the Lunar Orbiter Boeing FPAC team during this 4- to 6-hour interval beginning shortly before launch. A transition to Boeing FPAC control followed certified acquisition of continuous tracking contact by the DSIF. The Boeing FPAC team conducted all subsequent phases with JPL support on tracking data selection, DSIF predict, handling, and related orbit determination functions.

Acquisition and Launch Evaluation

A highly redundant series of DSIF tracking predictions was provided to ensure earliest possible acquisition of tracking contact by the DSIF. Preliminary predicts, based on preflight targeted trajectory data and expected lift-off time, were sent just before liftoff. Progressively more refined predictions that reflect the recorded occurrence of certain launch events and the accumulation of AFETR tracking data were sent as the launch and cislunar injection operations proceeded.

Pre-midcourse Phase

The purposes of the pre-midcourse phase were:

1) To calculate the best or optimal orbit-injection point;
2) To find a cislunar trajectory that satisfied the computed injection constraints;
3) To recommend execution of the required midcourse maneuver.

The encounter (or end) parameters were computed based on the latest orbit determination and were used as input for the pre-midcourse guidance computations. The pre-midcourse guidance computations were made to search out the one approach hyberbola from which an injection into the desired initial ellipse could be made with a minimum velocity increment.

Various combinations of two-axis attitude maneuvers were computed to align the thrust engine axis along the desired midcourse velocity vector axis. The combinations of attitude maneuvers and the plots of the maneuver were given to SPAC to evaluate and select one for execution.

When sufficient tracking data were determined to redefine the cislunar trajectory after the first midcourse maneuver, the two phases – midcourse decision and pre-midcourse – were again used in considering the consequences of a second midcourse. These computations were repeated as necessary until the possibility of performing useful maneuvers was overshadowed by the impending injection into lunar orbit.

Following first DSIF acquisition and receipt of corresponding tracking data, orbit determination work on the cislunar trajectory was begun by both JPL and Boeing FPAC. Two computer strings were active, one for each team. Resulting predictions for subsequent acquisition by other stations were added to the inventory and transmitted at the discretion of the JPL FPAC director. The joint JPL and Boeing FPAC effort was continued until continuity of DSIF tracking was ensured.

Meanwhile, as time allowed, Boeing FPAC made preliminary evaluations of the cislunar trajectory and computed the corresponding lunar encounter conditions. Comparison of these computed encounter conditions with the planned encounter gave an early check on the success of launch and lunar injection.

Midcourse Decision

Whether or not a midcourse correction was either necessary or advisable was established by the following procedure, which was used for both first and second midcourse decisions. The state vector and its covariance matrix from the latest orbit determination were mapped forward to a specified time in the neighborhood of lunar encounter. Then various encounter parameter computations were made to find the best impulsive orbit-injection maneuver for the present uncorrected cislunar trajectory.

The Δ V required for orbit, injection, and transfer, as obtained from the preinjection guidance computations, was compared to the budgeted Δ V. If predicted encounter errors and required Δ V were both small, a midcourse correction could definitely be determined as not necessary. If the predicted Δ V required exceeded available Δ V and the uncertainties were small, a midcourse correction was definitely necessary and the pre-midcourse phase was begun at once.

Preinjection Phase

Preinjection computations were made to prepare for the orbit injection maneuver. The cislunar orbit determination computations were continually updated, the trajectory redetermined after the final midcourse maneuver, and the state vector computed at the lunar encounter time. The preinjection guidance computations were made to find the best impulsive orbit injection maneuver for the current cislunar trajectory. Attitude command computations were performed to find the smallest two-axis attitude maneuver for injection using the required Δ V of preinjection guidance computations. When the maneuver was performed, the doppler was observed and compared with predicted values.

Pretransfer Phase

Orbit determination computations were begun shortly after injection. These computations were continually updated until the lunar orbit and the important lunar harmonics had been estimated to the required accuracy. Attitude maneuvers for photography in the initial orbit were plotted for SPAC.

For the transfer maneuver a trajectory search was computed by the guidance command computations using a variation-of-parameters procedure to gain a fast but accurate search. The two-axis attitude maneuvers as well as plots were processed for SPAC. After a successful transfer, the mission status was updated and the information displayed.

Prephoto Phase

After the transfer maneuver was made and the orbit reasonably established, a lifetime study was again initiated. If the predicted lifetime was too short, an additional transfer maneuver was to be considered. If the predicted lifetime was satisfactory, the photo command computations were made. The plots for successive orbits were prepared and compared until satisfactory photo coverage could be made. The photo information attitude commands and attitude plots were prepared so that SPAC could select the proper three-axis maneuver for the first pass. The succeeding photo passes were accommodated in much the same way until the photo mission was completed.

Photo and Readout Phase

During this phase, orbit determination computations were made to update orbital parameters for more precise location of photographs taken and to support command preparation for subsequent photography.

Periodically, during this phase, readout operations were suspended to permit the acquisition of tracking data. These data were used for orbit determination computations, to update earlier information, and support the periodic preparation of DSIF prediction and spacecraft commands containing Sun occultation and antenna orientation information.

Software

The employment and use of the operational software (computer programs) in controlling the Lunar Orbiter mission has been explained in detail in the preceding SPAC and FPAC sections. Additional discussion is needed, however, in the area of the overall effect of the programs themselves.

The computer programs performed as they were designed with only a few minor exceptions. The majority of the software errors were discovered and eliminated during an intensive period of testing prior to the mission. A few errors that were discovered during the mission were either corrected as patches in the software, or workaround methods were devised.

The intensive use of the software system uncovered a number of operational shortcomings that, if left uncorrected, would cause undue pressure on operational personnel, and, therefore, increase the probability of error. Most significant of these was the lack of any software-provided pointing data for the high-gain antenna for off-Sun Canopus orientation of the spacecraft. The CORL program was modified during the mission to provide this information. Workaround methods were devised for this shortcoming and required considerable extra effort on the part of the analyst.

The DSIF computer software performed without error. Minor real-time changes were made in the JPL source deck to support the FPAC programs. These did not change the computer programs but ensured that the outputs required to support successive routines were made available and in the proper format. These communication controls were required between the orbit data generator (ODG) and the tracking data processor (TDP) programs, and within the ODPL program.

Photographic Control

Lunar Orbiter photography required a complex series of operations to initiate photography and control the processing and readout operations. The spacecraft photo subsystem was designed to provide the capability of automatically accomplishing definite sequences of events, including film exposure, film processing and drying, film transportation, and photo data readout. Each of these automatic sequences was initiated and controlled by a series of commands originating at the SFOF. The time sequencing of photographs and the supporting spacecraft maneuvers were determined by the location of the photographic site with respect to the trajectory data.

The FPAC trajectory, guidance, and maneuver control programs performed the following functions.

1) Generation of orbit traces over an area of interest in selenographic coordinates;

2) Generation of photo "footprints" on the orbit traces;

3) Computation of the spacecraft-camera geometry;

4) Generation of the attitude commands for picture taking;

5) Automatic plotting of the photo footprints and photo geometry;

6) Automatic plotting of the three-axis attitude commands in cone-clock axes to avoid antenna nulls.

The orbit determination program (ODPL) processed DSIF tracking data to determine accurate state vectors as close as possible to the photo sites. These state vectors were then used with the lunar gravity constants to compute the desired trajectories for flight path analysis. The procedure used to determine the state vectors was as follows:

1) Process short data arcs, including as much as possible of two-station tracking data;

2) Where possible, solve for the state vector by itself; if necessary, use gravity harmonics when the fit was inadequate;

3) Use a matrix tight enough to prevent divergence but loose enough not to constrain the solution.

A photo quality prediction computer program (QUAL) was employed to assist in determining the desired shutter speed for given photo sites and conditions. The predicted ground resolution was obtained for the following criteria:

1) Diameter of the cone with a four-to-one base-to-height ratio obtained at a S/N of 3:1;

2) A slope angle of a 7-by 7-meter plane surface obtained at a S/N of 1:1;

3) The 3:1 contrast tri-bar resolution obtained at a S/N of 1:1.

Mission-dependent parameters required for input to this program for each run included the following:

1) Albedo;

2) Phase angle;

3) Altitude;

4) Smear rate;

5) Smear angle;

6) Radiation dosage.

The output of the QUAL program supplied the shutter speed and other parameters for both the 80- and 610-mm cameras. Also included was a quality factor indicating the expected results for data recorded on the spacecraft film, the GRE film, or video magnetic tapes.

The output of these computer programs was used to generate the maneuver command sequences and camera settings for the spacecraft and the time to execute the photo sequence. The set of procedures outlined above was employed for all vertical photography of the primary site photographs and the film-set photographs of the front side of the Moon. Special computation were made during the mission to implement the requirement for taking the nonstandard photos which combined the Earth and the Moon's limb, the oblique film-set photos and the farside photos.

Film management was carefully planned to position particular priority readout frames at the proper location at the time the readout could be accomplished. This was satisfactorily done within the following photo subsystem constraints:

1) To avoid Bimat stick, process at least two frames every 15 hours;

2) To avoid Bimat dryout and accompanying degradation, process at least two frames every 4 hours;

3) The camera storage looper has a maximum capacity of 20 frames in addition to its normal closed thread-up length;

4) The camera looper should not be emptied to less than two extra frames;

5) Provide optimum stereo pair readout of moderate-resolution frames of each site;

6) Provide as many high-resolution frames as possible of the stereo areas;

7) To avoid film set, make single exposures on alternating orbits (every 8 hours).

The following paragraphs discuss response of the spacecraft to the commands required to satisfy the above and the successive operations necessary to produce the final lunar photographs.

Operational Photography

The photo subsystem consisted basically of three functional units: a camera, a processor-dryer, and the optical-mechanical scanner or readout mechanism. These areas were separated by film storage loopers of variable contents. Film that has been advanced through the camera goes to the camera storage looper. Film that is to be processed comes from this camera storage looper. The looper thus allows the camera and the processor-dryer to function independently. During the film processing operations, the film passes through the readout looper and the optical-mechanical scanner and onto the takeup reel. When film is read out by the optical-mechanical scanner, it passes "backwards" from the takeup reel and expands into the readout looper. Then, at the beginning of subsequent processing, the film in the readout looper advances to the takeup reel. Thus, the readout looper returns to and remains in a closed position during all processing.

Operation of the spacecraft and photo subsystem are controlled by selectable stored program commands (SPC) such as:

1) Perform Attitude Change for Photography;

2) Perform Reverse Attitude Change after Photography;

3) Prepare for Readout;

4) Stop Readout;

5) Prepare for Photography;

6) Return to Standby after Photography;

7) Process X Frames;

8) Read Out Photo Data;

9) Combinations of Film-Set Frames, Read Out Photo Data, and Process X Frames (to satisfy photo subsystem operation constraints).

At the end of the attitude control maneuver for photography, the photo subsystem should be in a "solar eclipse off" and "processor inhibit" state. To ensure that this condition exists, backup commands are transmitted to the spacecraft. The processor inhibit function also prevents any Wind Forward command and safeguards against inadvertent Bimat cut. Photo Subsystem Heater Power-Off commands ensured that the total photo subsystem power demand does not overload the spacecraft power supply. The camera program is set for the required photo sequence (16, 8, 4, or 1, slow mode or fast mode). The camera thermal door is opened and the V/H sensor is turned on. After a 2-minute warmup period for the V/H sensor, the picture-taking sequence is initiated by a camera on command. Frames are exposed at a frequency depending on the camera rate and the V/H ratio. The V/H sensor shuts off automatically after the last exposure in the multiple sequence. To ensure that the V/H sensor is off, a backup command is also performed. The photo subsystem heaters are then activated to provide thermal control for the photo subsystem. On completion of the photo sequence, the camera thermal door is closed by a command. The processor-dryer is activated by command and film begins to move through the unit. When the desired processing has been completed, the Readout Drive On command is transmitted which inhibits further processing

The photo video chain is the longest string of system elements in the series affecting the ultimate in photographic quality. During the priority readout, all readouts were of 43 minutes (one frame) or less. During the final readout, the original plan was to limit readout to 86 minutes (two frames); however, as the mission progressed, this constraint was purposely exceeded and was gradually increased from 86 to approximately 102 minutes. This was accomplished by monitoring the thermal fin plate temperature to ensure that the upper limit was not exceeded. The 102-minute readout period was equivalent to 2.37 frames. A typical sequence of real-time commands required to implement the readout function are as follows:

1) TWTA On followed by Video Modulation On and Readout Electronics On;

2) Readout Drive On (first transmission puts the line-scan tube in the focus stop position);

3) Line Scan Tube Focus (increase or decrease to provide the proper results);

4) Video Gain command (increase or decrease to provide optimum signals);

5) Readout Drive On (second transmission initiated the readout sequence).

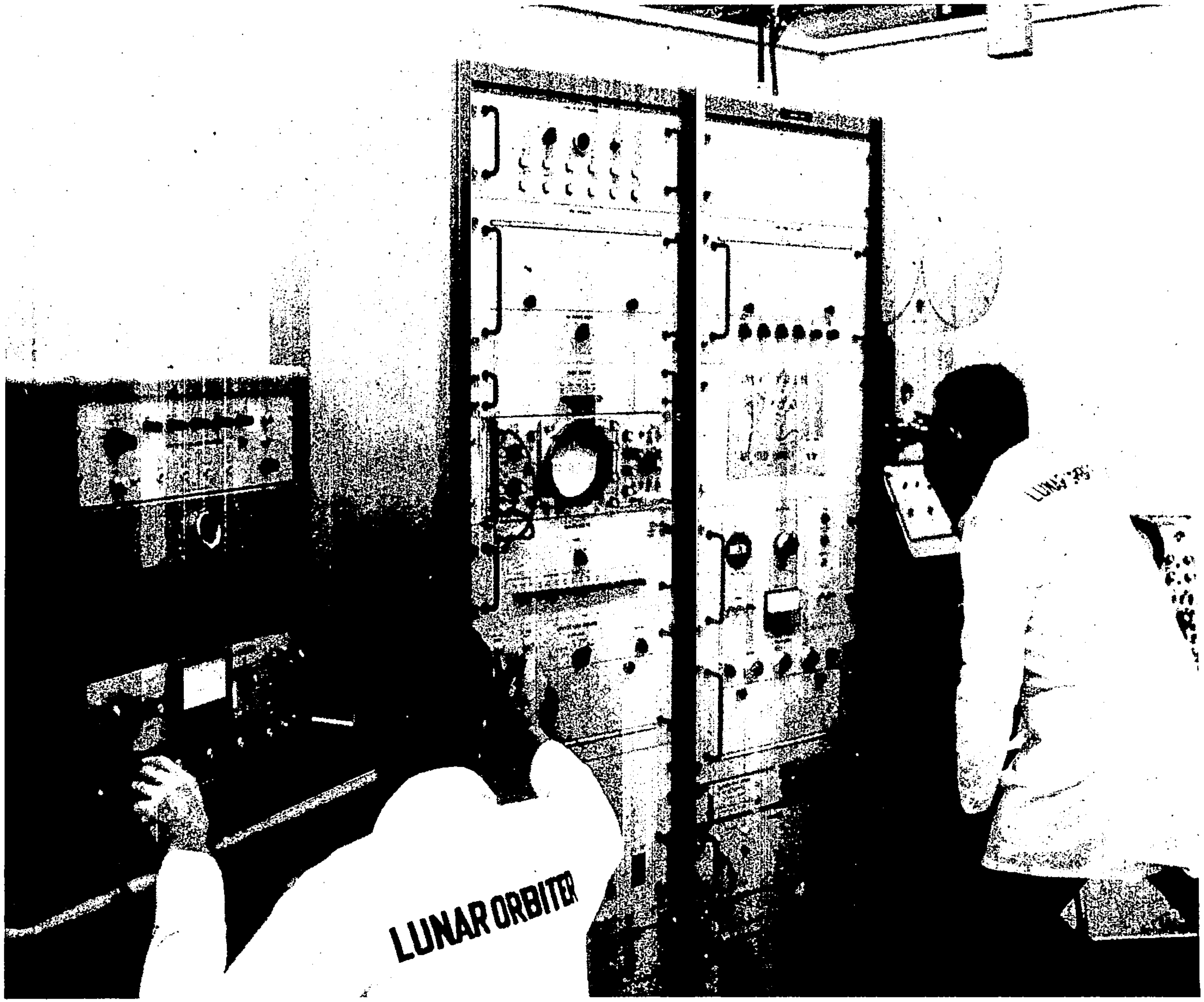
The readout sequence was terminated by the stored program command series of Readout Drive Off, Readout Drive On, and TWTA and Video Modulation Off. The Readout Drive On command in the termination sequence is used as a backup to ensure that the photo subsystem returns to the standby processing mode.

Analysis of the performance telemetry data and evaluation of the photos obtained verified that all of the above functions were satisfactorily performed during the 211 exposures and the 141 readout periods of the photographic mission.

Reconstruction

Approximately 10 minutes prior to the start of the readout sequence, a conference was held between the photo data analyst at the SFOF and the video engineer at the supporting DSIF. During this period information was exchanged relative to operational status of the GRE and operational status of the spacecraft, and other information needed for readout identification. As the readout sequence was initiated in the spacecraft, the video engineer observed and confirmed each of the command functions as they were executed by the characteristics of the readout data. The first Readout Drive On command configured the optical-mechanical scanner (OMS) in a focus stop position. In this condition, the photo video chain adjustments could be made and optimized to the prescribed levels. Additional adjustments were made in system gains to ensure that the white level was being properly clipped to set the average white level at exactly 5.0 volts and 0 volt offset. An alternate method of optimization, used at the Woomera site, consisted of setting the video subcarrier detector for a 5.0-volt input during preparation for readout. The spacecraft was commanded to the focus stop position and a white-level amplitude was measured on the GRE monitor oscilloscope. Video gain changes were transmitted to bring the gain to 5.0 volts. Both of these methods produced proper results. The first used the spacecraft as a reference while the second used the video subcarrier detector calibration as a reference. Analysis of the methods will be made to standardize the procedure for future missions.

All of the 35-mm GRE film (Type SO-349) was processed by an Oscar Fisher commercial processor using liquid X-ray developer after each readout period at each DSS. Processor performance was maintained by using a pre-exposed standard sensitometer step tablet as the primary standard. These control strips are shipped to each DSIF at regular intervals and stored under controlled environmental conditions until used. Each of the processed control strips is evaluated in terms of speed, contrast, and minimum density. These results are plotted against the average



Lunar Orbiter GRE Equipment at Deep Space Station

and 2 σ control limits for each parameter and appropriate corrective measures taken to correct for out-of-control results or indicated trends.

Following the development of the GRE film of each readout period, a number of measurements were made on the quality evaluation viewer, and reported to the SFOF. These measurements were of two types: qualitative values from the edge print data, and quantitative and qualitative image evaluation. Evaluation of the edge print data established a control on the spacecraft film processing, photo-video chain adjustment and performance, and, to a limited extent, GRE adjustment and performance and an overall test of the communications performance. The edge data gray scales permitted evaluation of the overall tone reproduction and a scale adjustment against which the image density could be measured.

Direct measurements of the image evaluation consisted of brightest and darkest flat area density, and estimated image smear and minimum resolvable object size. The video engineers were also asked for their opinions of photo quality, such as exposure, smear, etc.,. Results of their readout analysis were used to modify operating procedures, to optimize readout quality, and assist in the determination of camera exposure settings for individual photos.

Reassembly

Manual reassembly of selected GRE film was accomplished at the SFOF, Madrid, and Woomera during the mission. The film was produced by video tape playback into the GRE during nonreadout periods. These photos provided early data on the performance of the photo subsystem to the mission director. They were also used for local public information releases.

Manual reassembly was also accomplished at the Langley Research Center from film produced by video tape playback at the Photo Data Assessment Facility. The photos were used to provide transparencies and paper prints for use by NASA and the Lunar Orbiter Photo Data Screening group.

Automatic reassembly of the GRE film from all readout periods was made at Eastman Kodak facilities in Rochester, N. Y.. The processing and handling schedule was based on reassembly printing as the first function performed. It was further planned that the reassembly printer output would remain current with the daily receipt of GRE film. However, during the final readout period the reassembly printer output could not keep up with the daily receipts.

Early in the mission the film from one GRE could not be automatically reassembled because the density of the image reference in the edge data pattern was too high. This was caused by flare in the GRE kinescope and existed in different degrees on each GRE. Other than this there were no major problems encountered during the reassembly operations. Spot checks of the reassembly negatives indicated that the printer operated within the allowable tolerances.

Copying

Photo copying at Eastman Kodak included the making of negative transparencies and positive transparencies by successive generation copies from the original 35-mm film. In addition, the 9.5-inch reassembled negative was copied to produce successive generation positive transparencies, negative transparencies, and paper prints. As a result of the degradation in the high-resolution photos, the requirement to make paper prints of these photos was deleted during the final readout.

The limitations of the reassembly printer output also limited the quantity of 35-mm film available for copying, causing delays in the delivery schedule. In addition, the copying of the 9.5-inch negatives was interrupted for a total of 14 working days by programs having a higher priority. The 35-mm copying requirement was completed on October 12th and the 9.5-inch requirement on October 28th.

A total of 267 video magnetic tapes were recorded at the three Deep Space Stations and shipped to the Photo Data Assessment Facility at Langley Research Center. These tapes were used to selectively produce the following:

1) A duplicate copy of the original;
2) An analog tape copy containing only the video data;
3) One GRE film for each analog tape;
4) Two additional GRE films as priority permitted.

1.3.4.5 GROUND SYSTEM PERFORMANCE

The Deep Space Network (DSN) provided all facilities necessary to sustain the flight operations requirement of the Lunar Orbiter mission. This was accomplished through a complex consisting of several Deep Space Stations (DSS), the Space Flight Operations Facility (SFOF), and the ground communications system (GCS), which provided the interconnection between all of these facilities.

Other than minor circuit outages normally expected during routine operation of the DSN-GCS over extended periods, the only major circuit outage of significance was the loss of the three teletype lines and the high-speed data circuits for a period of 20 minutes between the SFOF and the Madrid Deep Space Station. The outage was attributed to the commercial carrier at the Madrid facilities. This problem is of a nonrepetitive nature and requires no further action.

Considerable difficulty was experienced in the scheduling of communications resources to support Lunar Orbiter I. This was primarily due to scheduling relationships with other user projects. The realignment of schedules in real time became a daily task.

Space Flight Operations Facility (SFOF)

The Space Flight Operations Facility exercised command control of all Lunar Orbiter flight operations during the mission. In addition, the SFOF provided the data processing, communications, display, and support capabilities that were necessary to perform such analysis, evaluation, and interpretation required to support the mission to completion. Separate areas were used for mission control, spacecraft performance analysis and command (SPAC), flight path analysis and command (FPAC), and computer control. Communications between these areas consisted of an operational voice control system, closed-circuit television, interphone, telephone, and personal conferences. Computer systems, teletype, visual displays, bulk printers, administrative printers, and plotting boards were all used as tools in support of the analysis of spacecraft status and performance.

Telemetry processing system (TPS) and central computing complex (CCC) sections of the data processing system provided telemetry data processing, tracking data processing, command generation transmission and verification, and prediction generation and transmission to support the Lunar Orbiter mission. Hardware performance of associated computers and the data processing system was outstanding. Some intermittent problems with the 1301-disk files were encountered, which consisted primarily of format and parity errors. Overall operational performance was good. Problems encountered were due to dual-mission support and insufficient time to conduct extensive Lunar Orbiter mission simulation exercises after the conclusion of the Surveyor I mission. The majority of the problems were of a procedural type and were quickly resolved.

DSN Intracommunications System (DSN/ICS)

The DSN/ICS provides the capability for transferring all types of information required for spaceflight operations within the SFOF. This system includes all voice communications, closed-circuit television, and distribution of teletype and high-speed data to designated areas for use throughout the SFOF.

Some minor problems were experienced in support of the nonstandard requests for special configurations. These nonstandard requests (i.e., special patches, and real-time modification of standard operating procedures) sometimes resulted in deviation from the system design. However, there were relatively few of these occurrences in the overall operational performance, which was satisfactory.

Tracking Data Quality Determination

During the first 6 hours, the DSN was responsible for both orbit determination and data quality determination, as well as the history of data quality and analysis throughout the remainder of the mission. Jet Propulsion Laboratory personnel handled the initial orbit determination. The orbits were determined within the allowable time given in the sequence of events and showed a nominal injection that was subsequently verified by later orbit-determination computations. One anomaly occurred in this phase involving the data from the X2 counter at Woomera. Several attempts to fix it failed, resulting in the decision to delete these data from the orbit-determination computations since there was sufficient overlap on the X1 data block. All data used for the midcourse-maneuver orbit-determination calculation had been evaluated in the tracking data quality determination and were assessed as good.

During the post-midcourse phase, the data showed perturbations resulting from the spacecraft pitch and yaw maneuvers as they were being executed. These maneuvers added small accelerations to the spacecraft (all approximately in the same direction so that there was no cancellation effect). All of the maneuvers were clearly visible in doppler data residuals and were of concern to orbit-determination personnel. All other doppler data obtained from midcourse to orbit injection was considered good. The accuracy of the data was well below 0.01 Hz for the 1-minute sample rate and, therefore, met the commitment specifications of 0.2 Hz.

Excellent tracking data was obtained after orbit injection and during the initial orbit. The data-quality determination was consistent among all three stations.

Deep Space Station (DSS)

The Deep Space Stations (Goldstone, California; Woomera, Australia; and Madrid, Spain) supported the Lunar Orbiter mission by:

1) Obtaining and processing telemetry and video data from the spacecraft;

2) Transmitting commands to the spacecraft;

3) Communicating and transmitting both processed and raw data to higher user facilities.

Real-time tracking and telemetry data were transmitted through the ground communications system. The video data were recorded on video magnetic tapes and, by mission-dependent equipment, on 35-mm film. All physical material, such as processed films, video tapes, logs, and other reports were sent to the appropriate destinations via air transportation.

The overall performance of the Deep Space Stations during Lunar Orbiter Mission I was excellent. All commitments were met, and the incidence of error was low. For the first 35 days of the mission, tracking data was obtained for a total of 1,003.53 station hours (a 20% increase over the 816 hours committed to support the mission). In addition to its normal function of supplying data for the mission, Goldstone Deep Space Station provided an additional service during the early photographic period. Special reruns of the Ground Reconstruction Equipment were made, using the video tape data as an input. These were hurriedly transported to the SFOF, where they were evaluated by the subsystem analysts and mission personnel to determine the course of action to be taken.

Four minutes of photo video and performance telemetry data were lost during readout sequence 114 recorded at the Madrid DSIF on September 10th. Up lock to the spacecraft was lost by a problem with the DSS transmitter.

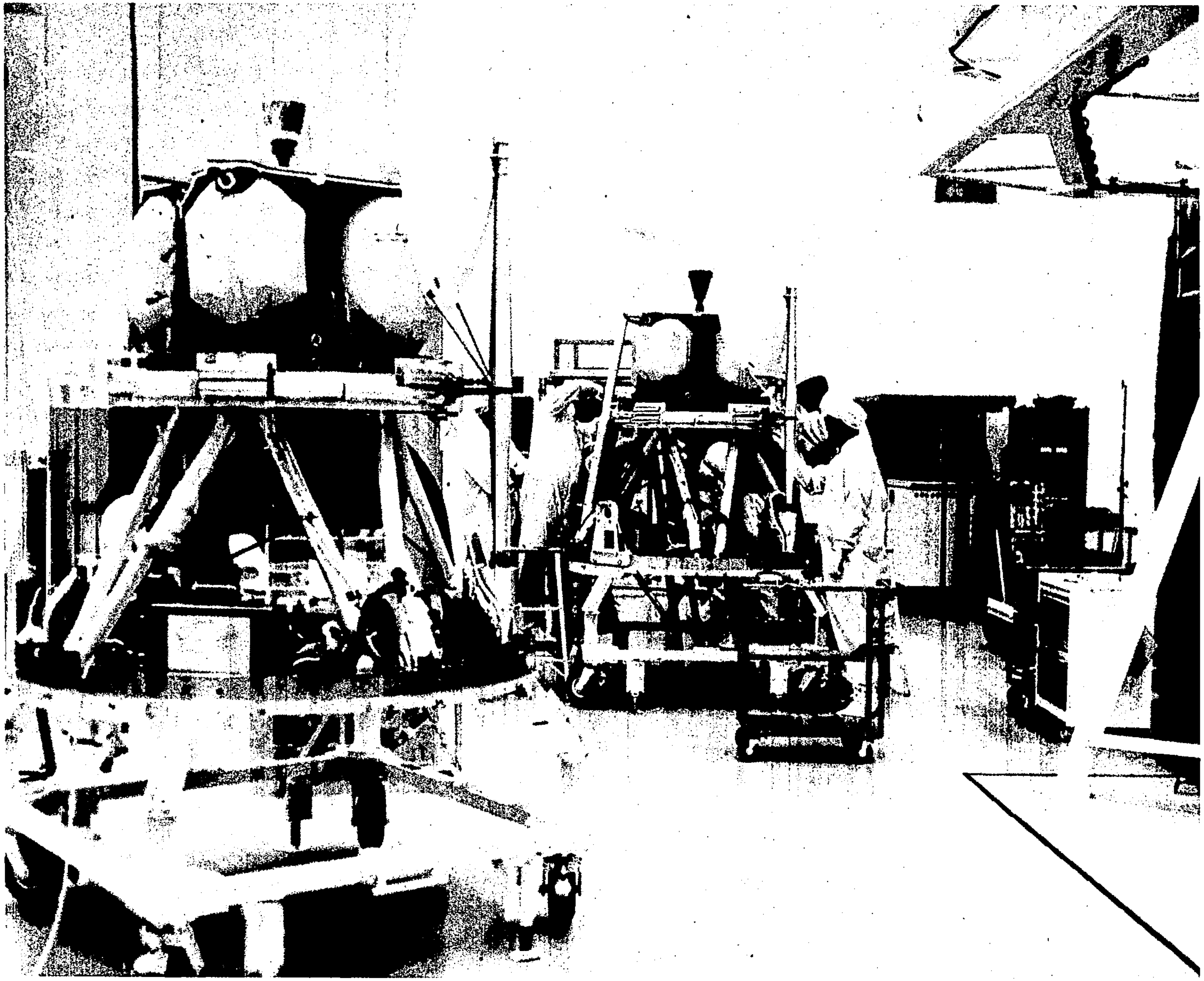
1.3.5 LOGISTICS

The Lunar Orbiter logistics plan furnished all spares and support at the Deep Space Stations as well as providing for the transportation of GRE film and video magnetic tapes to Rochester, New York and Hampton, Virginia, respectively. Environmental requirements for the different types of materials required special handling during the transit period to and from overseas Deep Space Stations. Studies were conducted to determine the optimum method of satisfying all of these requirements with a minimum of degradation in the materials being transported.

1.3.5.1 SPARE PARTS SUPPORT

Limited spare parts inventories were maintained at AFETR for spacecraft maintenance. Major components were planned to be returned to their source for repair whenever necessary. Spare parts were prepositioned at the supplier's facility for the necessary repair of spacecraft flight components.

Drawer-level communication and telemetry spares were provided at each Deep Space Station for any unit requiring removal for repair or calibration. Additional drawer-level spares were not furnished to support the redundant installed equipment. In addition to drawer or module spares, small replaceable assemblies and repair parts were located at the Deep Space Stations for repair purposes.



Lunar Orbiter Clean-Room Operation
(Hanger S at Cape Kennedy)

The ground reconstruction electronics was supported with drawer-level spares for drawers on the picture receiving line. Repair parts were available for replacement in the drawers and in the auxiliary ground reconstruction electronic test equipment (AGRETE).

1.3.5.2 EXPENDABLE SUPPLIES SUPPORT

Expendable supplies are those items consumed during the operation of the Lunar Orbiter equipment for all tests and flight operations. Included were such items as the fuels and gases (GFE) and photo subsystem film and Bimat used aboard the spacecraft. Major ground support items were data recording papers, video magnetic tapes, recording 35-mm film, and sufficient film developing and processing chemicals. These supplies were delivered to all using stations before they were needed for the appropriate operational phase. Immediately prior to the mission, the inventories were evaluated and replenished as necessary.

1.3.5.3 VIDEO TAPE SHIPMENTS

All video magnetic tape recordings of the photographic data were shipped from the Deep Space Station to the Boeing-operated Langley Photo Data Assessment Facility, Hampton, Virginia for use by NASA.

Environmental requirements for protection of the video magnetic tapes necessitated protection against electrical and magnetic fields while being transported. The Netic Data Preserver container was designed to provide the environmental protection for information stored on magnetic tapes. Preaddressed fiberboard containers were provided in advance to reduce the work load at the Deep Space Station.

Commercial air freight service was used for all shipments to all Deep Space Stations. Charter airplane service was required to transport the tapes from the Goldstone and Woomera Deep Space Stations to the nearest commercial airline terminal. The services of Emery Air Freight Transporters were secured to expedite the delivery of this data at all transfer points. Accountability was maintained on all shipments from the time that they left the Deep Space Station until they arrived at the Langley Facility.

1.3.5.4 GRE FILM SHIPMENTS

Each of the Deep Space Stations was supplied with adequate shipping containers for transportation of the GRE film. The plans also included the safeguarding or insuring against loss of a complete readout period. This was accomplished by requiring the GRE films from a single spacecraft transmission to be shipped from the station in separate shipments. Under normal circumstances, daily shipments were planned from each Deep Space Station at the end of their respective view period.

Commercial air freight facilities were used for the transportation of GRE film. The film shipments were coordinated with the shipment of video tapes to the maximum extent permissible. Chartered airplane service was also used to deliver the film from the Deep Space Station to the closest airline facility. Emery Air Freight Transporters services were also employed to expedite the handling at all transfer points.

Positive accountability was maintained at the Seattle Operations Center of all shipments from the time that they left the Deep Space Station to the time that they were delivered to the Eastman Kodak facility at New York.

1.4 MISSION DATA

PHOTO SITE	SPACECRAFT EXPOSURES PLANNED	ACTUAL
I-0	20	20
I-1	16	16
I-2	16	16
I-3	16	16
I-4	16	8*
I-5	16	16
I-6	16	8*
I-7	16	16
I-8.1	16	8*
I-9.2a	16	16
I-9.2b	16	16
SUBTOTAL	180	156
OTHER PHOTOGRAPHS		
NEARSIDE	42	17
FARSIDE	0	11
MISSION B	0	15
EARTH	0	2
MISCELLANEOUS AND TEST	0	6
CAMERA DOOR CLOSED	0	4
SUBTOTAL	42	55
TOTAL EXPOSURES	212	211

*EXPOSURES REDUCED TO COMPENSATE FOR ANALYSIS TEST OF PHOTO SMEAR

Table 1.4-1: Photo Coverage Summary

Although the Moon is our closest space neighbor, its exact surface characteristics and terrain have been defined only at the relatively gross level obtained by astronomical study over the past several centuries. Limitations of equipment and peripheral influence, such as the Earth's atmosphere, have resulted in lunar surface object resolution determination of approximately 500 meters. Recently the Ranger and Surveyor programs and the Russian Luna series have provided closeup, high-resolution pictures of small areas of the Moon's surface. These programs are the result of technology advances and the increasing rate of acquiring scientific knowledge of the last decade.

During its 35-day mission Lunar Orbiter I contributed an enormous quantity of data on the topography of the lunar surface. More than 262,000 square kilometers of the lunar surface visible to the Earth were photographed to resolutions ranging from 35 meters in the initial ellipse to 8 meters in the second and third ellipses. The resolution of these photos was considerably better than the 8-meter design specification when the exposure was correct. In addition, more than 3,000,000 square kilometers of the lunar surface not visible from the Earth (farside) were photographed to a resolution of approximately 250 meters (plus about 400,000 square kilometers from the high-resolution camera to a resolution of 30 meters). The farside photography also included lunar topographic features that were identified and named by Russian scientists. Except for the loss of the high-resolution photography of the primary sites, and the change in coverage area for Sites I-4, -6 and -8.1, all of the required photographic data was obtained. Extensive area of interest photos on both the frontside and farside were taken during the film-set photography.

Lunar environmental data were obtained during all periods of visibility and from the accumulative effect of periods of occultation. The Orbiter also recorded the increase in radiation levels caused by the proton event of September 2, 1966 and suffered no degradation in performance from the exposure, except some thermal paint degradation, which raised temperatures about 3°F.

Over 1,000 station hours of tracking data were obtained during the 35-day mission which will contribute extensive selenographic information from which to further define the lunar gravitational field.

Performance telemetry data were received at the SFOF during all periods of spacecraft visibility. Data were obtained from the AFETR via DSS-71 during the launch phase and from the DSN during the remainder of the mission.

A summary and discussion of each type of data is contained in the following sections.

1.4.1 PHOTOGRAPHIC DATA

During the Lunar Orbiter I mission a total of 413 high- and moderate-resolution photographs were taken. This included 262,000 square kilometers of the side of the Moon seen from Earth and over 3,000,000 square kilometers of the farside. (Farside photos were taken at higher altitudes, thereby covering greater areas with each photo.) All of the primary photo sites were photographed as planned, except for Sites I-4, I-6, and I-8.1.

The overall photographic coverage is summarized in Table 1.4-1: The 8-frame slow-mode sequence for Sites I-4 and -6 resulted in a 60% increase in surface area coverage. A 36% reduction in surface area coverage of Site I-8.1 resulted from the 8-frame fast-mode sequence.

Examination of second-generation GRE film copies (selected at random) have been made under magnification of 20 to 30 diameters with no significant reduction in photographic information content and the readout scan lines were visible. This is equivalent to 150 to 225 diameter enlargement of the original spacecraft film and is directly indicative of the high quality of the original negatives.

Lunar orbital photography was made particularly difficult by uncertainties in knowledge of the Moon's surface characteristics and its photometric function, both of which are critical to photography. The Moon has unique reflectance characteristics unlike any encountered in terrestial photography. The wide range of reflectance can and did produce photographic images that exceeded the density limitations of the spacecraft film in adjacent areas (thus obliterating all detail in these areas) while exhibiting excellent detail in the surrounding areas.

Overall quality of the moderate-resolution (80-mm lens) photography was good to excellent. The resolution of these photos, as determined by microdensitometer measurements of GRE film copies, was consistently better than the 8-meter design specification when the exposure was correct. Stereo coverage was obtained as planned during the normal primary site photography in both the fast and slow modes. In addition, side overlap stereo was obtained on Site I-9.2 during the successive photo orbits. The moderate-resolution photos revealed topographic and geologic data and characteristics not previously known to exist on the lunar surface.

PHOTO SITE	EXPOSURES	IMAGE SMEAR DIRECTION		USABLE FRAMES
		IMC	IMC AND FILM ADVANCE	
I-0	20	0	19	0
I-1	16	13	3	12
I-2	16	13	3	12
I-3	16	13	3	12
I-4	8	7	1	6
I-5	16	15	1	15
I-6	8	8	0	7
I-7	16	16	0	14
I-8.1	8	7	1	7
I-9.2a	16	14	2	8
I-9.2b	16	15	1	13
TOTAL	156	121	34	105

Table 1.4-2: **High-Resolution Summary**

SITE	S/C EXPOSURE	ANGLE OF INCIDENCE	PHASE ANGLE	ALTITUDE KM	ORBIT	CENTER OF COVERAGE	
						LONGITUDE	LATITUDE
I-0	5 - 20 21 - 24	68.4 - 59.9 55.0 - 53.2	62.9 62.9	216 - 208 208 - 208	26 I 26 I	88.2°E 98.6°E	1.1°N 1.4°S
I-1	52 - 67	62.4 - 60.2	60.9	53 - 54	9	42.0°E	0.9°S
I-2	68 - 83	67.0 - 64.8	65.4	52 - 51	10	35.5°E	0.1°N
I-3	85 - 100	69.3 - 67.0	69.2	54 - 53	14	26.3°E	0.4°N
I-4	105 - 112	70.4 - 66.5	68.5	49 - 47	21	13.6°E	0.1°N
I-5	118 - 133	70.3 - 68.3	68.6	49 - 48	29	1.5°W	0.1°N
I-6	141 - 148	54.5 - 50.7	54.1	44 - 51	39	2.2°W	3.8°S
I-7	157 - 172	57.9 - 55.8	58.3	46 - 48	48	22.1°W	3.3°S
I-8	176 - 183	59.6 - 58.6	59.3	49 - 50	55	36.5°W	3.0°S
I-9.2a	184 - 199	65.2 - 63.2	63.6	45 - 46	56	43.3°W	2.0°S
I-9.2b	200 - 215	63.5 - 61.5	62.2	46 - 47	57	43.3°W	2.5°S
B-2	48 & 49	81.7 & 80.9	70.3	80 & 78	5	28.8°E	3.0°N
B-4	84	83.3	70.3	78	12	14.6°E	3.0°N
B-5	103	79.1	70.3	66	18	8.1°E	2.1°N
B-7	113 & 114	83.7 & 83.3	70.3	75 & 74	23	5.3°W	2.8°N
B-8	134 & 135	79.7 & 78.8	70.3	57 & 55	31 & 32	16.0°W	1.6°N
B-9	149 & 151	72.9 & 68.7	70.3	47 & 43	40 & 42	24.0°W	0.2°N
B-10	150	77.7	70.3	54	41	30.9°W	0.8°N
B-11	153 - 156	75.1 - 73.2	70.3	50 - 48	46	36.1°W	0.3°N

Table 1.4-3: **Photo Site Coverage**

The high-resolution photos were degraded by different amounts, depending upon the magnitude and direction at smear on the spacecraft film. Although the 610-mm lens had the same f-stop opening and shutter speed as the 80-mm lens, the high-resolution photos were generally underexposed as a result of the 24% difference in the light transmission characteristics of the two lenses. Many photos contained local areas, such as slopes facing the Sun and the higher albedo uplands, with reflectivity characteristics which severely overexposed these areas of the moderate-resolution photos. The reduction in light transmission produced recogniziable detail in the corresponding areas of the high-resolution photos where the detail may be completely lost in the moderate-resolution photos. (An example of this condition is shown in Figures 1.4-6 and 1.4-7). Table 1.4-2 summarizes the general appearance of the high-resolution photos of the primary photo sites. The usable frames are defined as those photos that have smear in the IMC direction only and which may be moderately underexposed. When used in conjunction with the moderate-resolution photos some additional qualitative data can be obtained. The ability to mosaic successive high-resolution photos provides increased visibility of large areas of the primary sites at higher magnification, without employing a magnification lens or photo enlargements, thereby augmenting the qualitative evaluation of the primary site characteristics. Nine of the film-set high-resolution photos, taken with the V/H sensor off, produced high-quality photos. These were taken in the initial orbit on the nearside and from an altitude at 1300 km on the farside. These photos showed surface characteristics that have not been previously observed on the lunar surface.

Table 1.4-3 summarizes photographic coverage of the primary sites for Mission I and also the corresponding information on selected Mission B sites. The latter coverage was obtained in satisfying the photo subsystem constraints. The angle of incidence is defined as the angle between the Sun's rays and the normal to the lunar surface. The phase angle is the angle between the camera axis and the Sun's rays. The angle ranges are for the first and last frame of the sequence, respectively. Center of coverage is the geometric center of the single-or multiframe exposure sequence. Figures 1.4-2 through 1.4-8 are representative photos of lunar features observed at the primary photo sites.

Combined coverage of the eleven moderate-resolution photographs of the farside of the Moon is illustrated in Figure 1.4-1 and significant supporting data is shown in Table 1.4-4. Figures 1.4-11 through -13 are typical of the farside photos obtained.

The resolution capability of these moderate-resolution photographs is approximately 240 meters and approximately 30 meters for the five good high-resolution photographs. Correlation between some of these photos and the previously published Russian observations of the farside was possible.

Table 1.4-5 provides a summary of the location of the nearside film set photography as well as identification and supporting data for these photos. Figures 1.4-9 and -10 are examples at this photography.

The following photographs, Figures 1.4-2 through 1.4-13, are representative of portions of the primary photo sites identified for this mission. Also included are representative farside photography, frontside areas of interest, and examples of moderate resolution and accompanying high resolution. Each photo contains a descriptive caption which contains identification, location, and scale factor information.

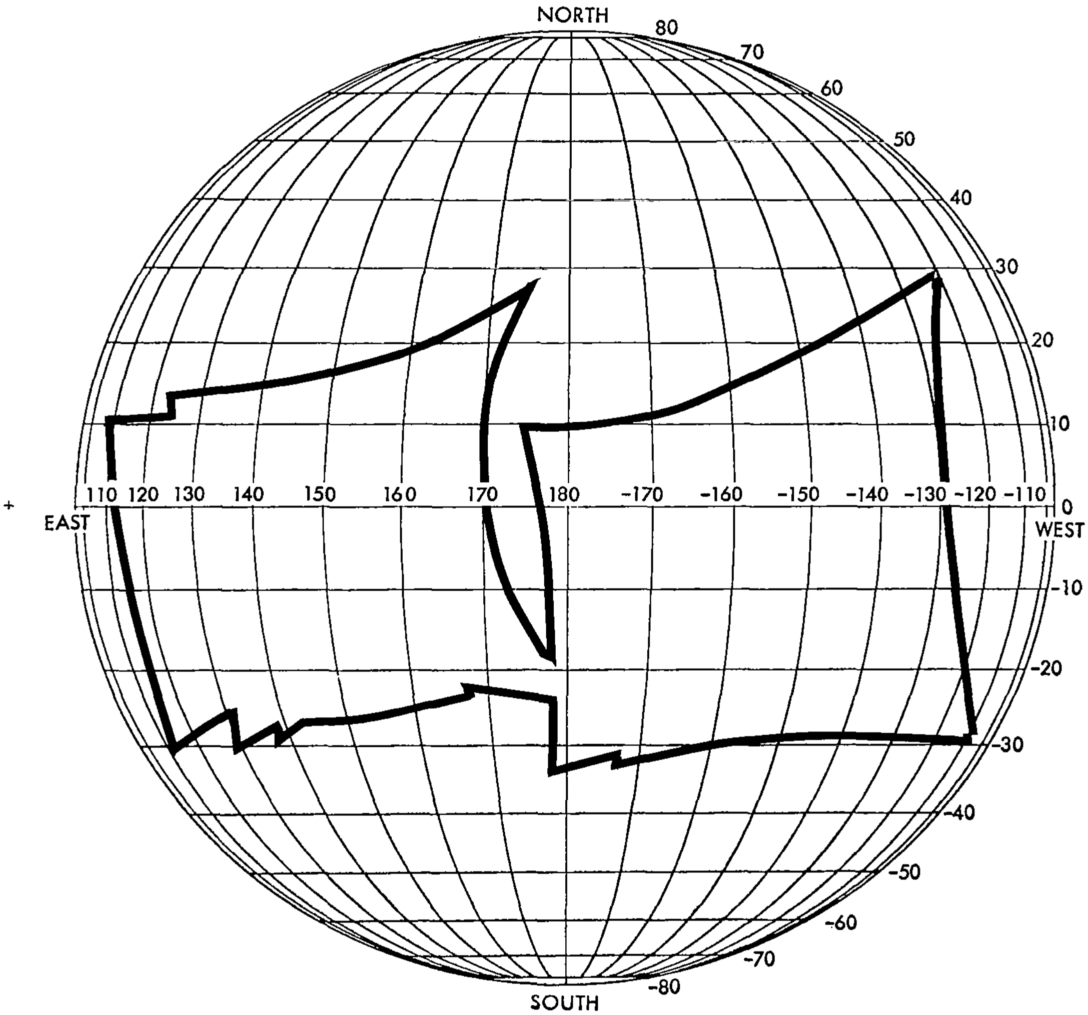


Figure 1.4-1: Lunar Orbiter I Farside Photographic Summary

SPACECRAFT EXPOSURE NO.	LOCATION OF CENTER LONGITUDE	LATITUDE	ALTITUDE km	FRAMELET WIDTH km MODERATE	HIGH	PHASE ANGLE	ANGLE OF INCIDENCE
28	154.3°W	10.4°S	1301	42		70.3	67.4
30*	163.3°W	10.3°S	1295	42	5.4	70.3	65.9
35	162.8°W	9.4°S	1335	43		70.3	69.9
36*	162.3°W	9.3°S	1340	43	5.6	70.3	70.4
37	157.9°W	8.7°S	1377	44		70.3	74.6
38*	157.6°W	8.6°S	1381	44	5.8	70.3	75.1
39	149.2°W	7.3°S	1448	46		70.3	83.2
40	148.9°W	7.2°S	1450	47		70.3	83.5
115*	144.2°E	7.6°S	1379	44	5.8	70.3	70.3
116	152.3°E	6.1°S	1454	47		70.3	78.3
136*	128.6°S	7.9°S	1321	42	5.5	70.3	68.8

* HIGH-RESOLUTION PHOTOGRAPH OF EXCELLENT QUALITY.

Table 1.4-4: Farside Supporting Data

S/C EXPOSURE	LOCATION OF CENTER		ALTITUDE KM	FRAMELET WIDTH KM	PHASE ANGLE	ANGLE OF INCIDENCE
	LONGITUDE	LATITUDE				
25	76.1°E	1.3°N	225	7.2	70.3	72.6
26*	71.8°E	1.7°N	226	7.2 0.9	70.3	73.3
27*	69.7°E	1.7°N	226	7.2 0.9	70.3	73.5
29*	61.0°E	1.8°N	227	7.3 0.9	70.3	74.7
31*	47.7°E	2.8°N	239	7.7 1.0	70.3	80.7
32	49.7°E	2.4°N	234	7.6	70.3	78.7
33	56.5°E	1.0°N	220	7.0	70.3	71.7
34	58.0°E	0.7°N	218	7.0	70.3	70.4
41	38.8°E	3.4°N	252	8.1	70.3	85.9
42	31.5°E	4.4°N	263	8.5	70.3	89.5
44	42.4°E	2.0°N	68	2.2	70.3	74.9
46	26.6°E	4.0°N	96	3.1	70.3	87.1
47	27.1°E	4.0°N	95	3.1	70.3	86.7
50	42.3°E	0.4°S	55	1.8	70.3	64.5
51	42.7°E	0.5°S	55	1.8	70.3	64.0
137	20.8°W	1.4°N	54	1.7	70.3	79.8
138	18.4°W	0.7°N	50	1.6	70.3	75.7
139	20.8°W	0.4°N	48	1.5	70.3	74.5
140	20.1°W	0.2°S	46	1.5	70.3	72.1
173	22.4°W	4.7°S	54	1.7	70.3	53.7
174	57.8°W	1.6°N	70	2.2	70.3	85.3
175	35.3°W	3.3°S	49	1.6	70.3	59.6

*Nonsmeared high-resolution photo.

Table 1.4-5: Frontside Filmset Coverage Summary

SCALE

Figure 1.4-2 Moderate-Resolution Frame 62

Site: I-1
Location of Center: 42.28° E longitude
0.93° S latitude

Altitude: 53 km

Exposure: 11 of 16
Phase Angle: 60.9 degrees
Incidence Angle: 60.9 degrees

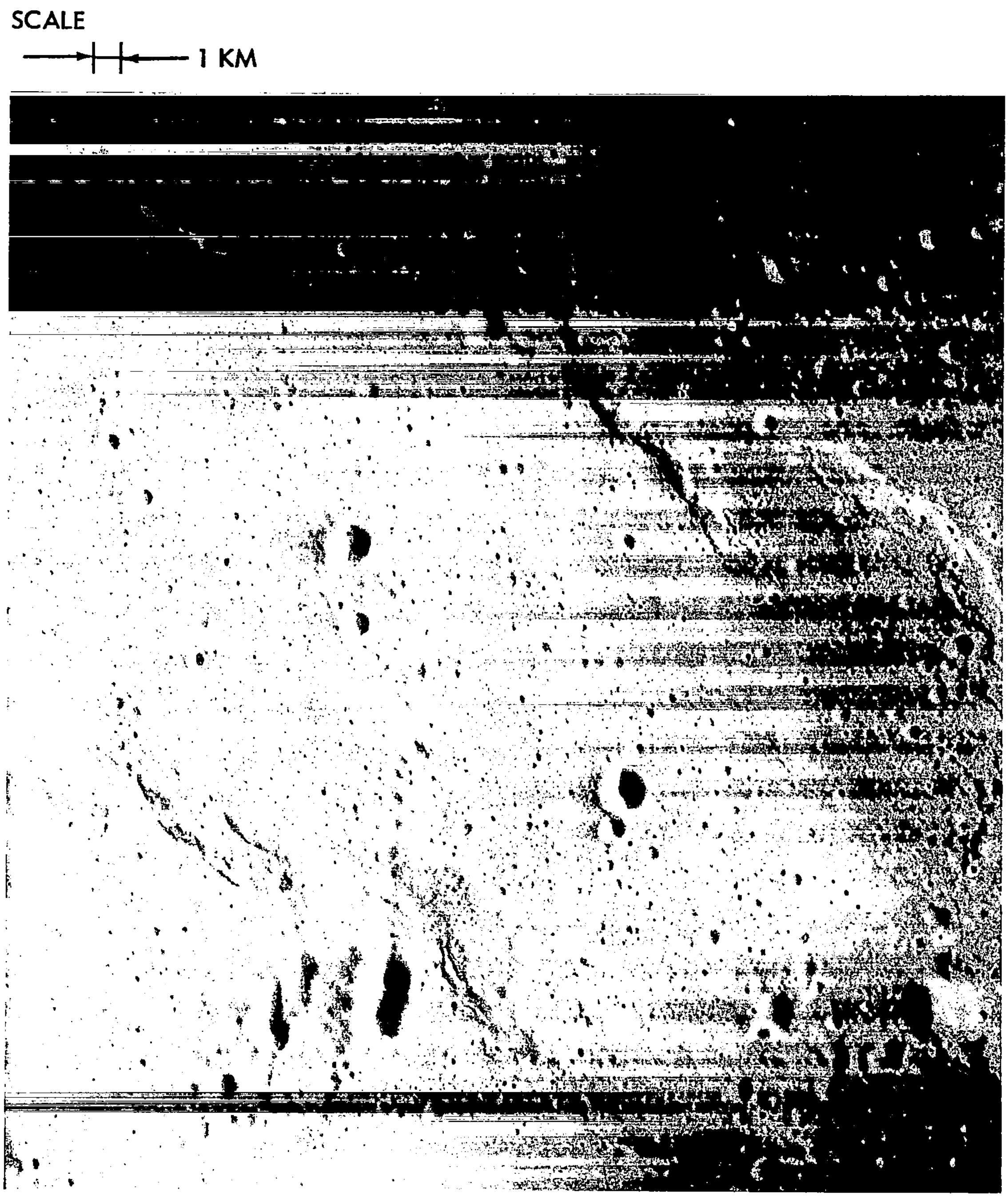


Figure 1.4-3 Moderate-Resolution Frame 90

Site: I-3
Location of Center: 25.79° E longitude
0.63° N latitude
Altitude: 53 km
Exposure: 6 of 16
Phase Angle: 69.2 degrees
Incidence Angle: 68.5 degrees

Figure 1.4-4 Moderate-Resolution Frame 123
(Planned Area of Surveyor C Landing Site)

Site: I-5	Altitude: 48 km	Exposure: 6 of 16
Location of Center:	1.87° W longitude	Phase Angle: 68.6 degrees
	0.11° N latitude	Incidence Angle: 69.6 degrees

SCALE

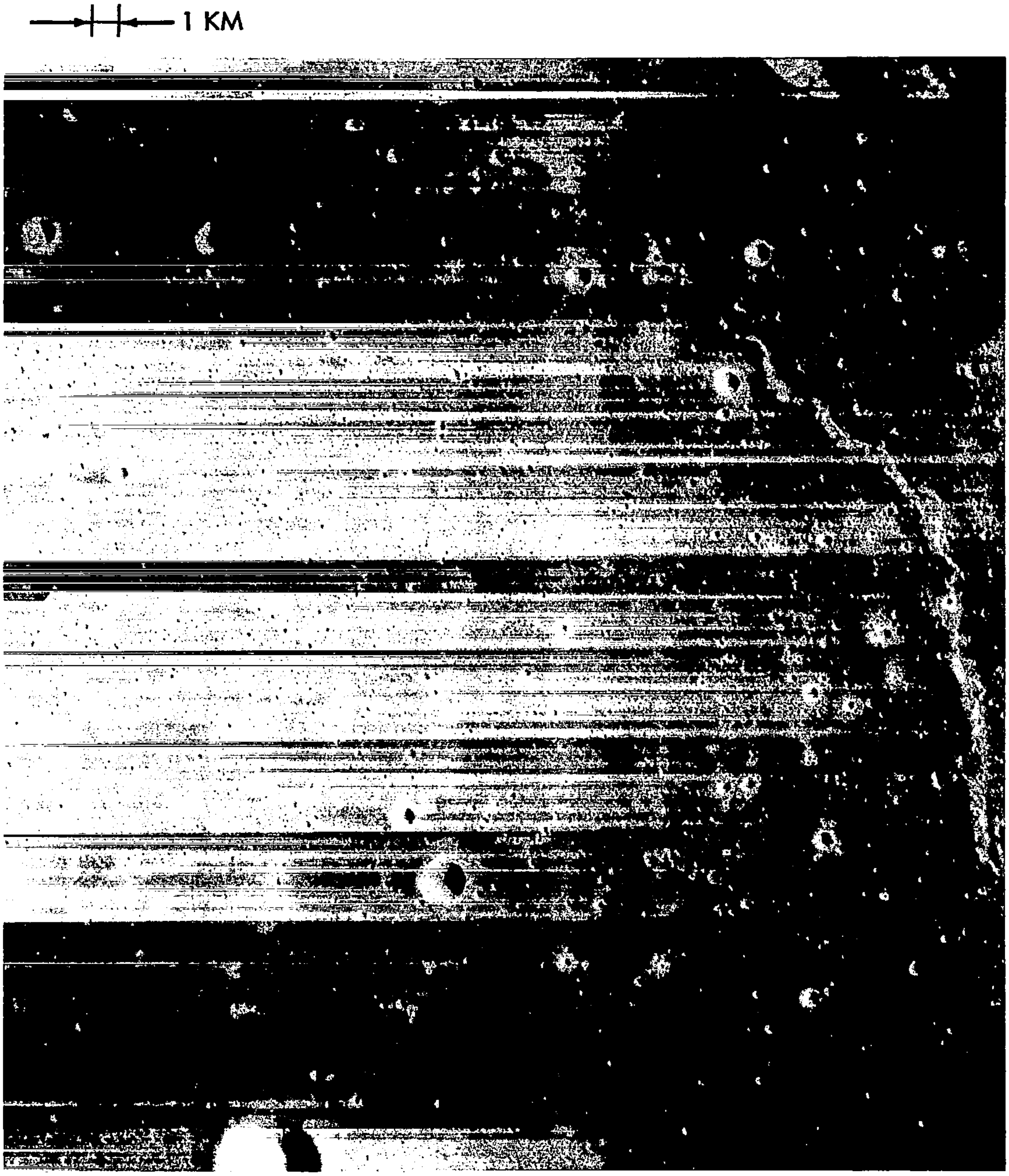


Figure 1.4-5 Moderate-Resolution Frame 208 (Surveyor I Landing Site)

Site: I-9.2 Altitude: 46 km Exposure: 9 of 16

Location of Center: 43.52° W longitude Phase Angle: 62.2 degrees

2.38° S latitude Incidence Angle: 62.4 degrees

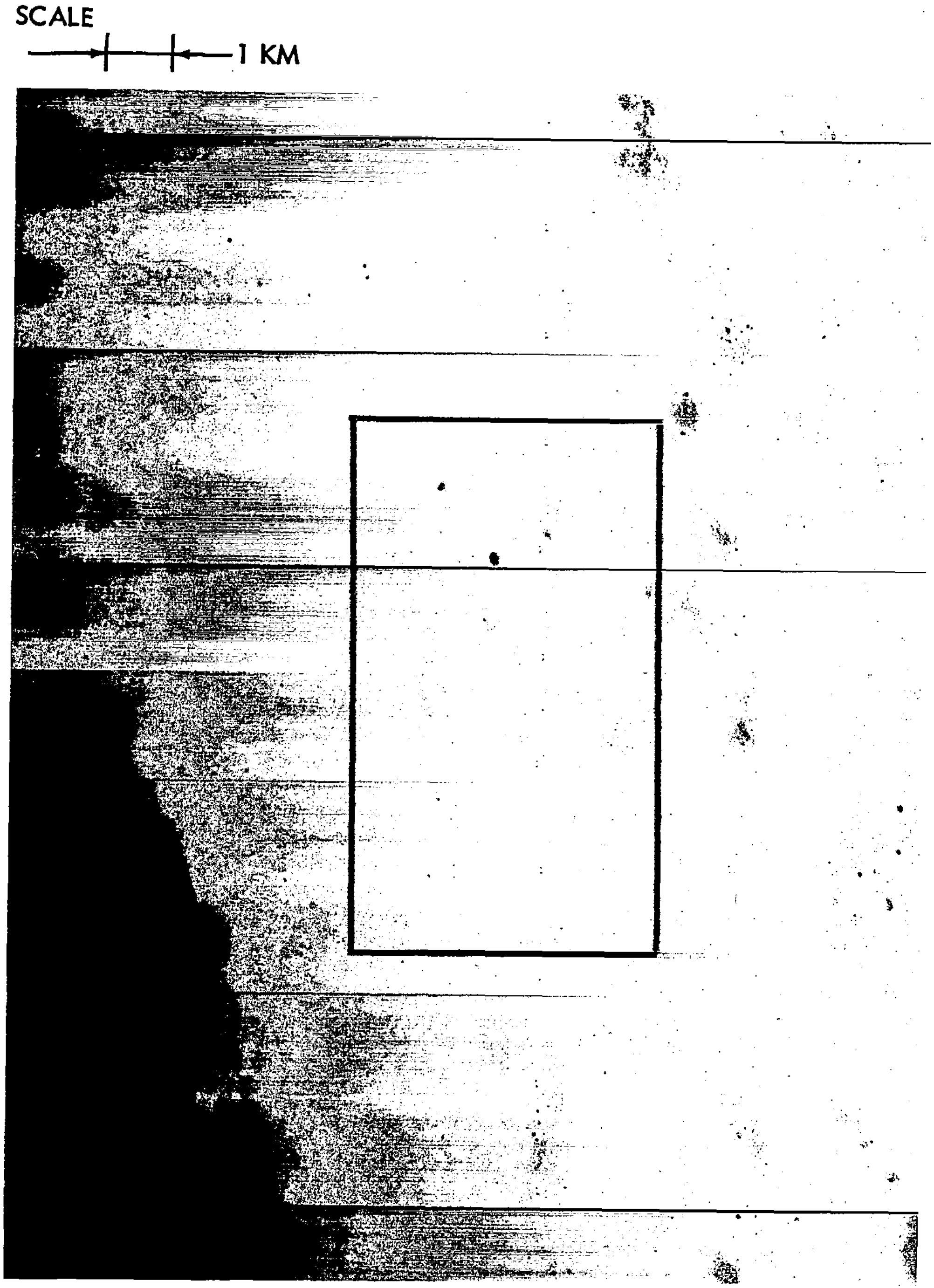


Figure 1.4-6 Moderate-Resolution Frame 70

Site: I-2 Altitude: 52 km Exposure: 3 of 16
Location of Center: 34.70° E longitude Phase Angle: 65.4 degrees
0.25° N latitude Incidence Angle: 66.7 degrees

(Detail within indicated area degraded by high reflectivity of surface.)

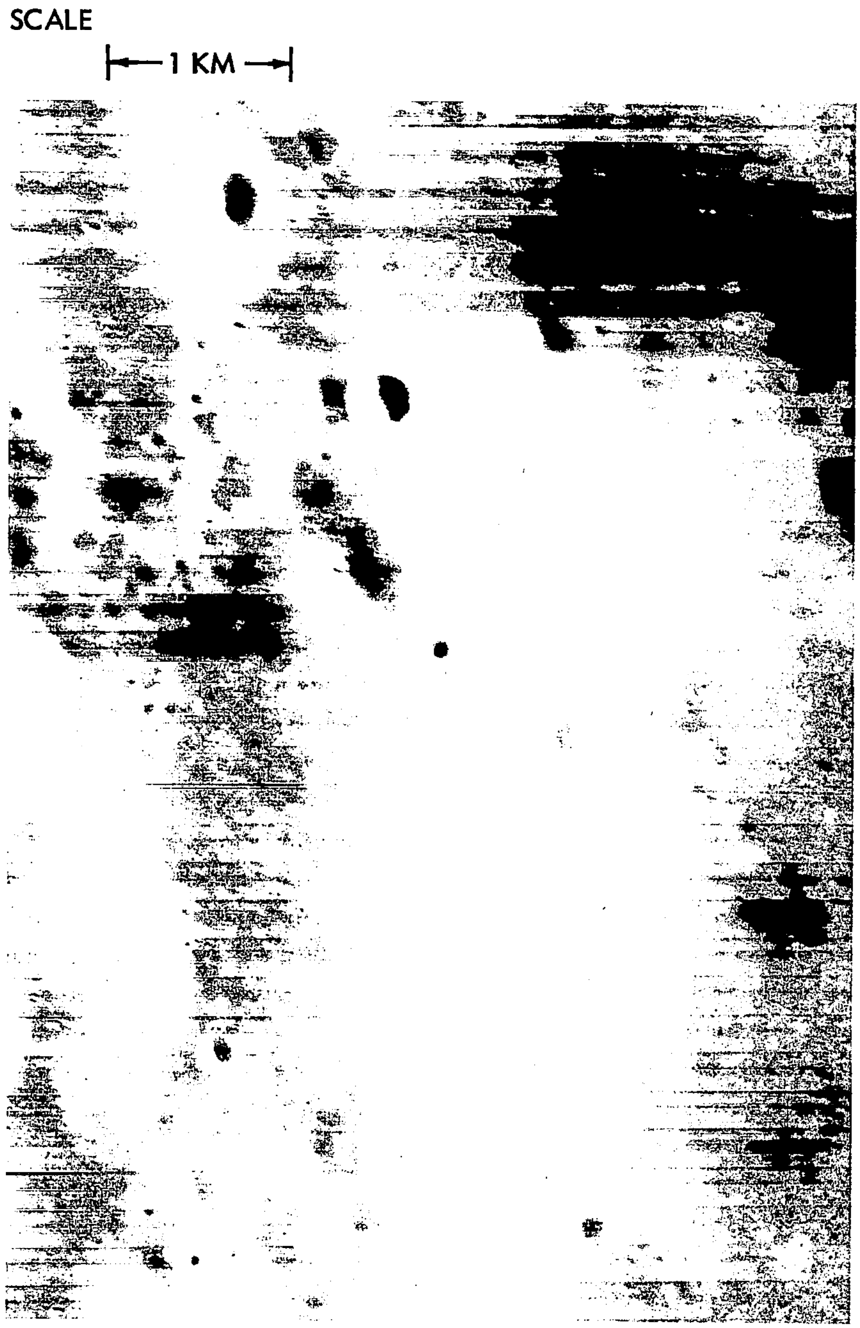


Figure 1.4-7 Section of High-Resolution Frame 70
(Smear in direction of spacecraft motion only)

Shows surface details in high-resolution photo for the area shown in Figure 1.4-7 where detail is lost in highlight.

SCALE

1 KM

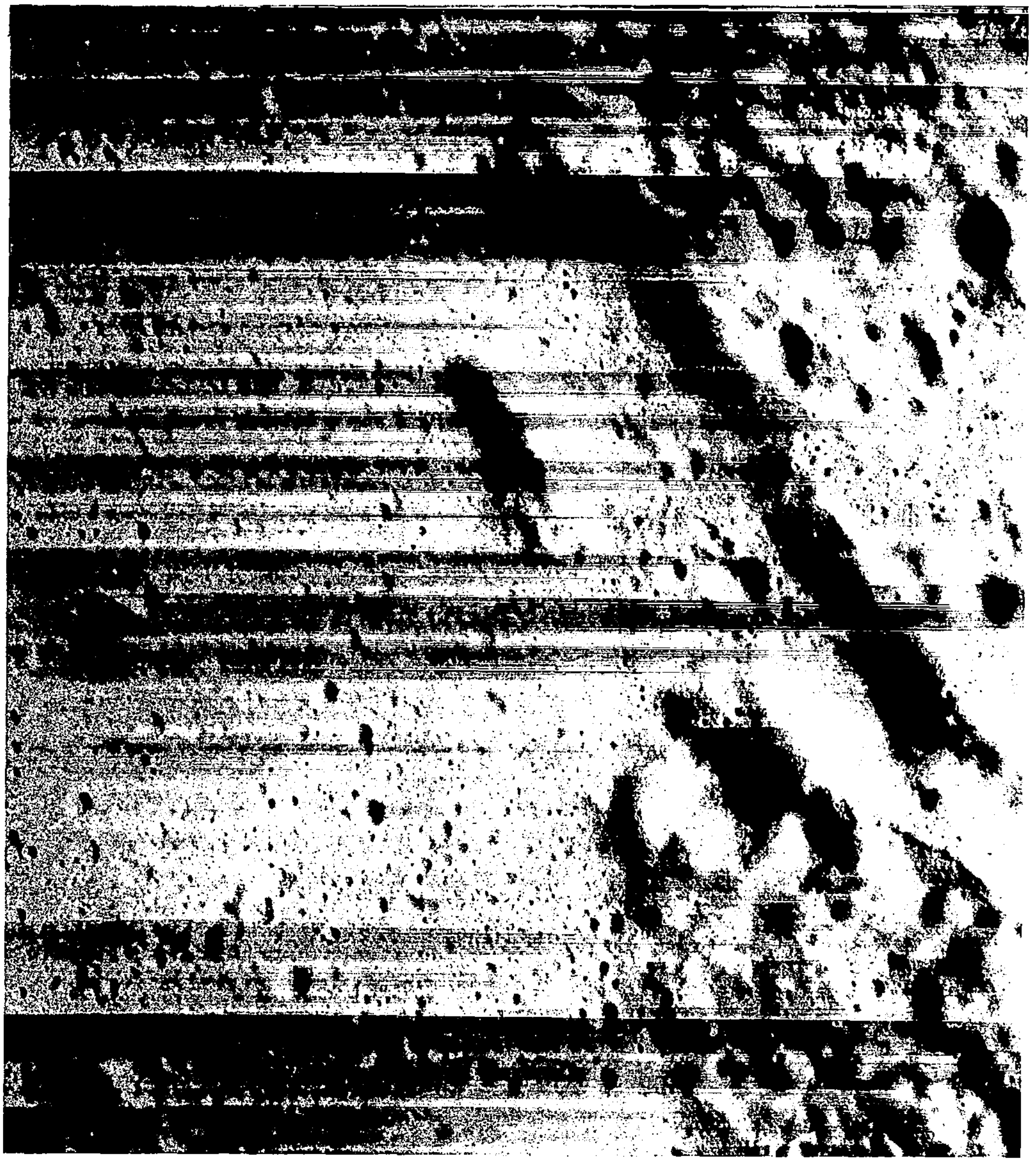


Figure 1.4-8 Moderate-Resolution Frame 108

Site: I-4
Altitude: 48 km
Exposure: 4 of 8

Location of Center: 13.24° E longitude
0.05° N latitude

Phase Angle: 68.5 degrees
Incidence Angle: 68.7 degrees

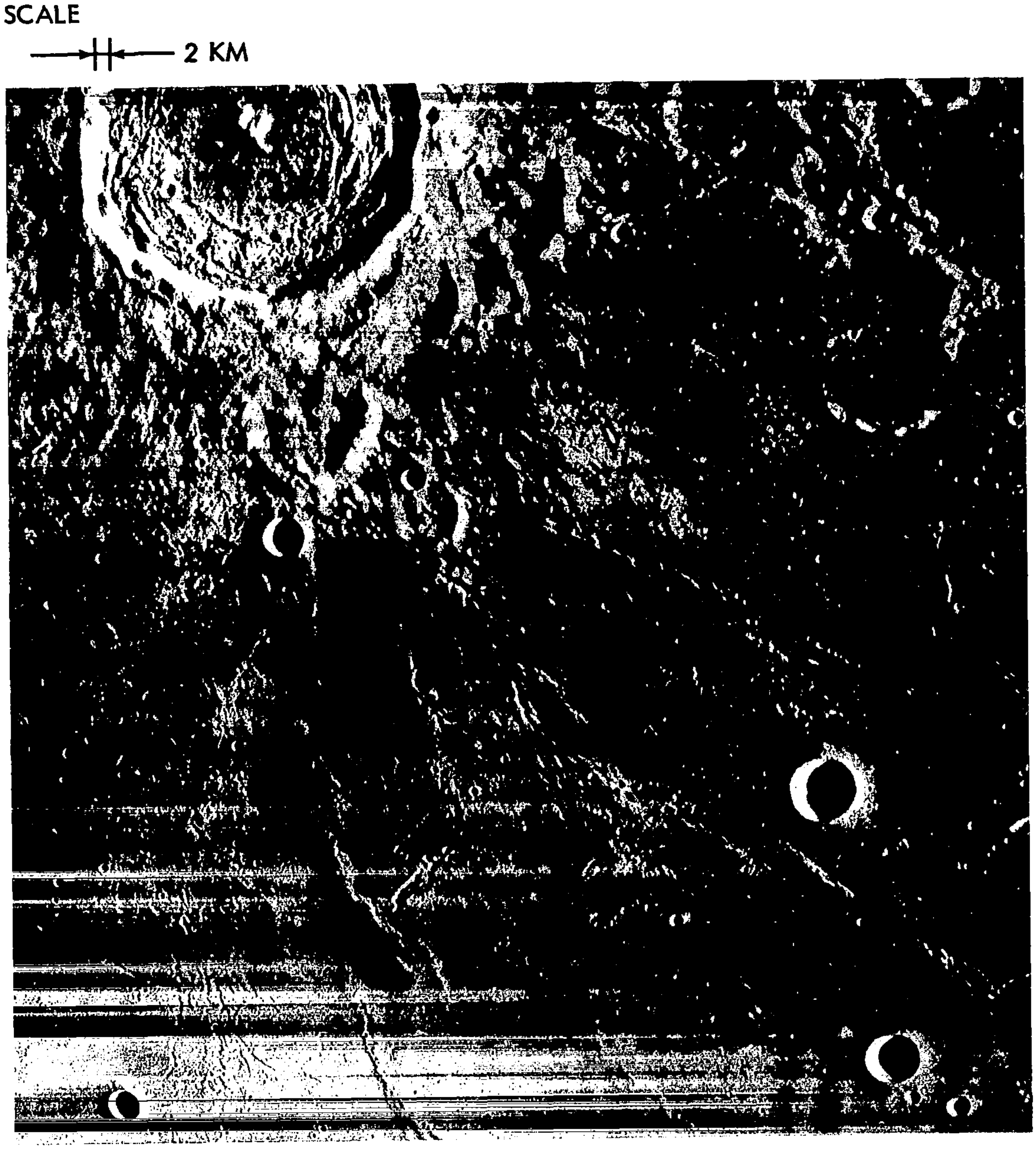


Figure 1.4-9 Moderate-Resolution Frame 31

Site: Frontside Film Set Altitude: 239 km Exposure: 1 of 1
Location of Center: 47.68° E longitude Phase Angle: 70.3 degrees
2.76° N latitude Incidence Angle: 80.7 degrees

Shows most of the crater Taruntius and
the northern section of Mare Fecunditatis.

SCALE

1KM

Figure 1.4-10 Moderate-Resolution Frame 114

Site: Frontside Film Set Altitude: 74 km Exposure: 1 of 1
Location of Center: 4.93° W longitude Phase Angle: 70.3 degrees
2.79° N latitude Incidence Angle: 83.3 degrees

Mare Area Northwest of Sinus Medii.

SCALE

10 KM

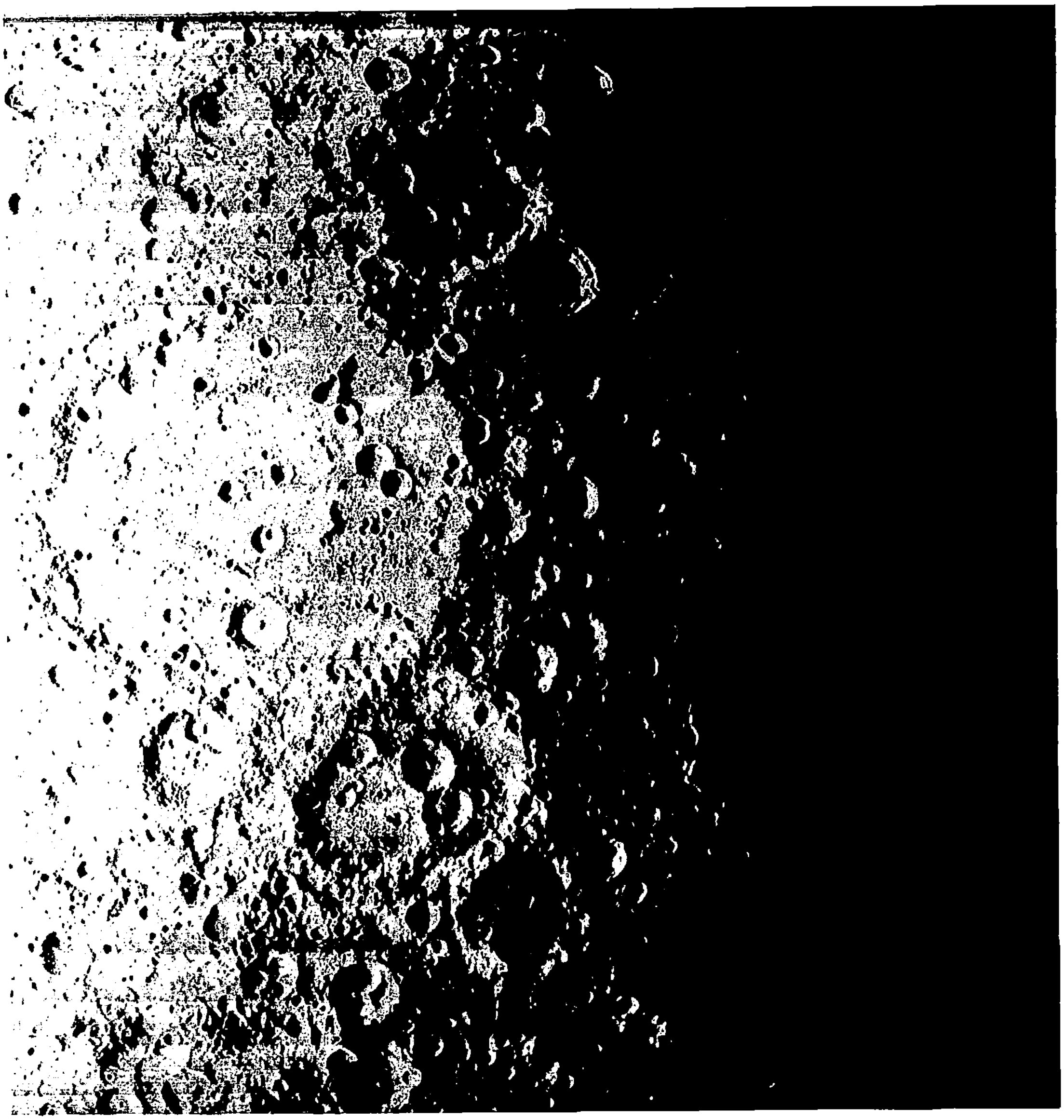


Figure 1.4-11 Moderate-Resolution Frame 40

Site: Farside Film Set Altitude: 1450 km Exposure: 1 of 1
Location of Center: 148.92° W longitude Phase Angle: 70.3 degrees
7.20° S latitude Incidence Angle: 83.5 degrees

Evening terminator: Line of demarcation between lunar night and day.

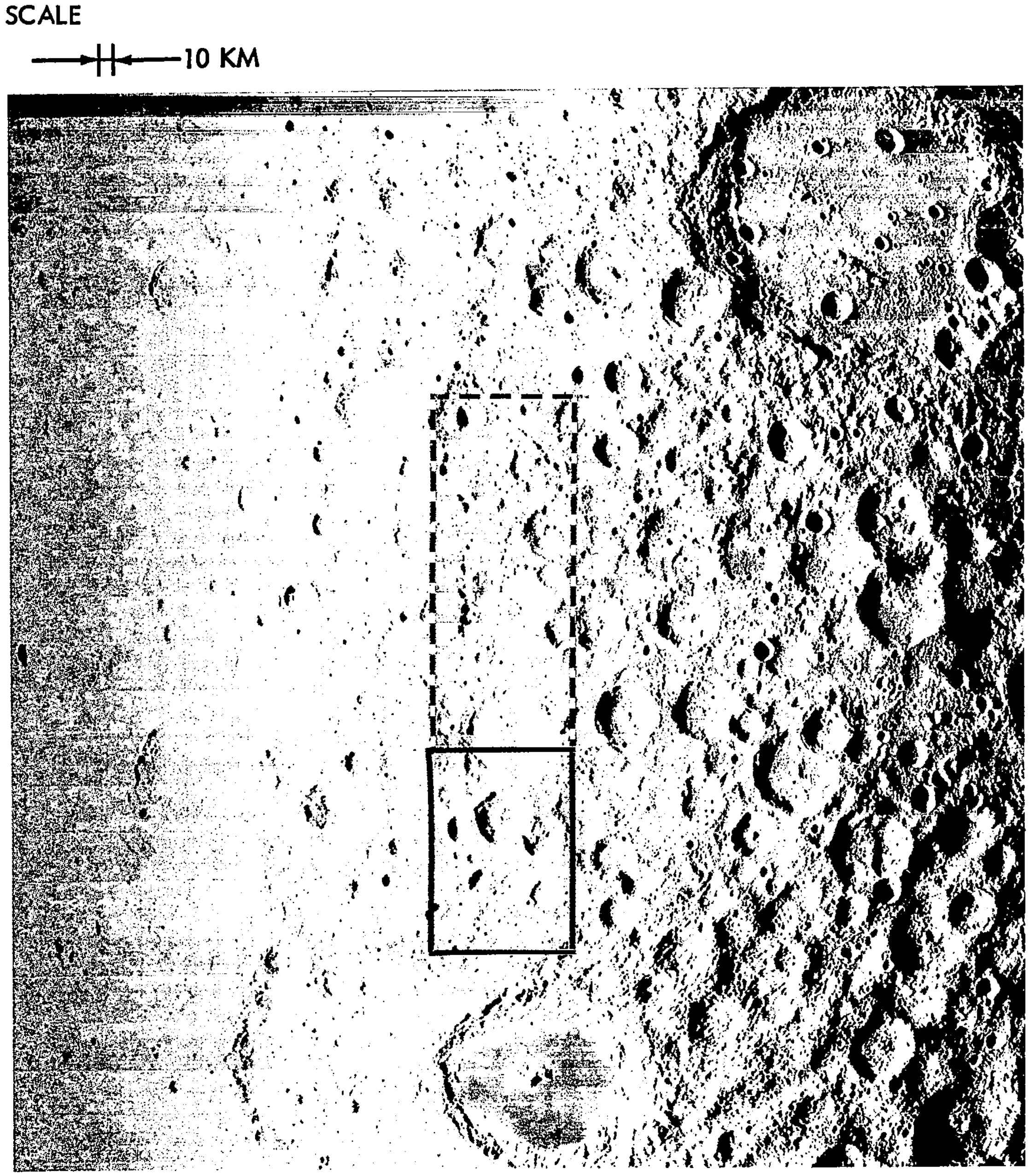


Figure 1.4-12 Moderate-Resolution Frame 136

Site: Farside Film Set: Altitude: 1320 km
Location of Center: 128.64° E longitude
7.95° S latitude

Exposure: 1 of 1
Phase Angle: 70.3 degrees
Incidence Angle: 68.8 degrees

Dark centered crater in lower center was named Tsiolkovsky by Russian scientists. Large area is indicated coverage of complete high-resolution photo.

Figure 1.4-13 Section of High-Resolution Frame 136

Shows area outlined by solid line in Figure 1.4-12 with greater lunar surface detail on crater floors.

STATE CHANGES IN (DF04 AND DF05) RADIATION MONITORS

GMT TIME OF CHANGE				DATE		NEW READING
DAYS	HOURS	MINUTES	SECONDS	(LOCAL)	COUNTER	(RADS)
222	21	16	49	8-10	DF04	0.25
222	21	47	18	8-10	DF04	0.50
222	22	9	0	8-10	DF04	0.75
222	22	52	0	8-10	DF04	1.00
228	2	25	43	8-15	DF04	1.25
228	20	25	8	8-16	DF05	0.5
235	2	18	2	8-22	DF04	1.50
238	7	3	21	8-26	DF05	1.0
240	23	20	37	8-28	DF04	1.75
241	7	18	19	8-29	DF05	1.5
242	0	14	0	8-29	DF05	2.0
242	7	35	59	8-30	DF05	2.5
244	6	47	12	8-31	DF05	3.0
245	8	59	18	9-2	DF04	2.0
PROTON EVENT OF SEPTEMBER 2, 1966 (SEE FIGURE 1.4-14)						
247	10	25	-	9-4	DF05	134.0
247	13	00	-	9-4	DF05	134.5
247	16	25	53	9-4	DF05	135.0
247	20	44	0	9-4	DF05	135.5
248	2	18	-	9-4	DF05	136.0
248	9	18	30	9-5	DF05	136.5
248	23	-	-	9-5	DF05	137.0
249	2	-	-	9-5	DF04	10.25
250	6	37	36	9-6	DF05	137.5
254	23	59	23	9-11	DF04	10.50
257	3	7	-	9-13	DF05	138.0

Table 1.4-6:
State Changes in (DF04 and DF05) Radiation Monitors

1.4.2 ENVIRONMENTAL DATA

Two types of telemetry instrumentation were installed on Lunar Orbiter I to monitor lunar environment conditions by measuring the micrometeoroid impacts on the spacecraft and the radiation dosage level at two specific points adjacent to the photo subsystem.

1.4.2.1 MICROMETEOROID DATA

Telemetry data obtained during the first 35 days of the photographic mission indicated that all micrometeoroid detectors were intact at the end of that period, and that no hits had been recorded.

1.4.2.2 RADIATION DATA

Radiation encountered during the Lunar Orbiter mission came from three sources: Van Allen belts, radiation galactic-cosmic radiation, and solar flare radiation. It was expected that the amounts of radiation received from the Van Allen belts and from galactic-cosmic sources would have little or no effect on the photographic mission. It is possible that solar flare radiation could present a hazard to mission goals.

Two radiation dosimeters were mounted adjacent to the camera system. Dosimeter 1, (DF04) located near the film cassette, had a sensitivity of 0.25 rad per count, with a capacity of 0 to 255 counts. Dosimeter 2, (DF05) located near the camera looper, had a sensitivity of 0.5 rad per count and a similar capacity of 0 to 255 counts. Due to the inherent shielding of the spacecraft, the photo subsystem structure, and the 2-grams-per-centimeter aluminum shielding provided in the film supply cassette, it was estimated that solar flares of magnitude two or less would have a negligible effect on the undeveloped film. Flares of magnitude three or greater would produce considerable fog on the film.

During Lunar Orbiter I's mission, the radiation dosimetry measurement system (RDMS) functioned normally and provided data on the Earth's trapped radiation belts and the radiation environment encountered by the spacecraft during cislunar and lunar orbiting mission phases.

Dosimeter 1 (DF04) indicated a total accumulation of 1.0 rad during penetration of the Van Allen belts. Dosimeter 2 (DF05) was not turned on until the Van Allen belts were essentially passed. For the next 18 days, the two dosimeters indicated a normal background penetrating galactic-cosmic radiation, plus the dosimeter noise level ("dark current").

On August 29th, a small solar proton event occurred that resulted in a total dosage of 1.5 rads at the camera looper.

On September 2nd, after the Bimat was cut, a large solar proton event occurred that gave total dosage of 8 rads at the film cassette location and 135 rads at the camera looper location during a period of 2 days. Any unprocessed film in the loopers would have been seriously degraded by this radiation dose, but the film in the cassette would have remained undamaged. For the remainder of the mission, only the galactic cosmic ray background in the dosimeter dark current has been observed.

Data gathered during Lunar Orbiter I's mission are presented in Table 1.4-6. The table gives the state change time for the two dosimeters prior to the solar proton events of September 2, 1966. This same table also identifies the increase in radiation after the solar proton event. Figure 1.4-14 shows the increase in radiation monitored in the 2-day proton event beginning on September 2nd.

Spacecraft instrumentation designed to monitor the lunar environment operated satisfactorily throughout the mission and provided accurate data.

1.4.3 SELENOGRAPHIC DATA

Preliminary values for lunar gravitational constants were determined from the tracking data during the mission to satisfy operational mission control requirements. A total of 1,003 station hours of tracking data was obtained by the Deep Space Network during the 35-day photographic mission. This data will be evaluated by NASA to more accurately define the lunar gravitational field.

The tracking data monitor program displayed its output on the teletype lines in the form of angle and doppler, pseudoresiduals relative to an on-site trajectory program or to SFOF predictions, detrended doppler pseudoresiduals, and doppler standard deviation. In general, this program functioned satisfactorily for mission control, considering that it was the first operational employment of the computer programs to support the determination of lunar orbits. It was of particular value in confirming the accuracy of the propulsion maneuvers.

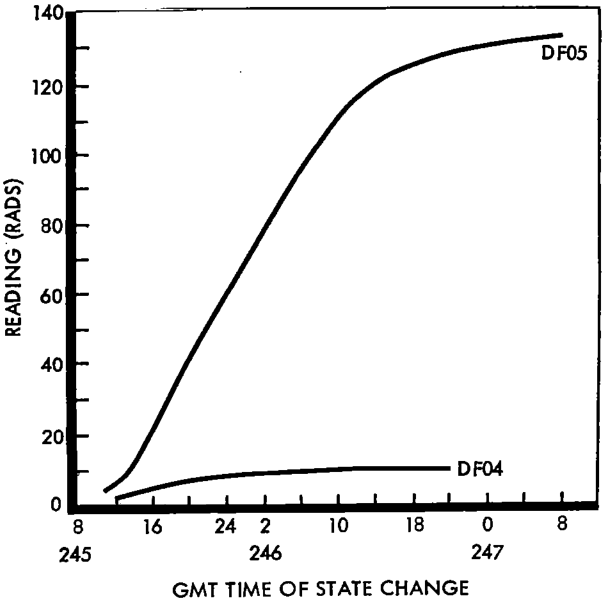


Figure 1.4-14: Radiation Buildup During Proton Event Beginning September 2, 1966.

Tracking data were generally well handled at the DSIF site and within the ground communications system with relatively little loss to the users because of garbling in the transmission. The DSIF-transmitted frequencies, data-tape, and data-monitor logs were kept up in near real time. Occasionally frequency input errors delayed the tracking data quality determination function. These were usually found to be keypunch errors.

Included in the design of any maneuver in the orbital trajectory phase was a prediction of orbital behavior after the maneuver execution. The predictions were based on the postmaneuver design state and a set of lunar harmonics considered to be the most accurate representation of the Moon. From a given state epoch, and a set of lunar harmonics, the state and Kepler elements at any later effort were easily determined using the mean element integrating computer program (LIFL). After maneuver execution, the orbit determination group solved for the actual orbit at various epochs. A measure of the accuracy of the actual maneuver was then made by comparing the orbital elements of the orbit determination solutions for those predicted at the time of the maneuver. Further, an estimate of the validity of the lunar harmonics was made by noting any divergence between the predicted behavior of the elements and the orbit determination solutions.

The characteristics of the lunar orbits of Lunar Orbiter I are presented in Figures 1.4-15 through 1.4-19. These illustrations are histories of perilune radius, apolune

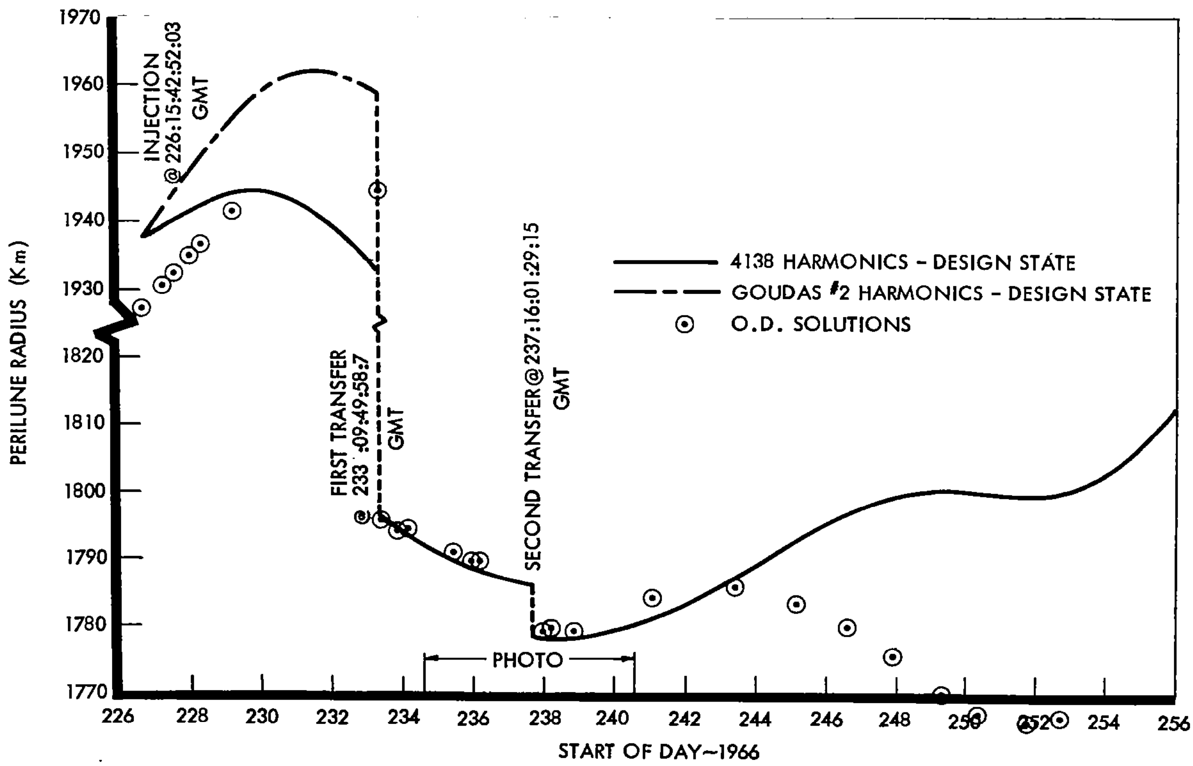


Figure 1.4-15: Lunar Orbiter Mission I Perilune Radius History

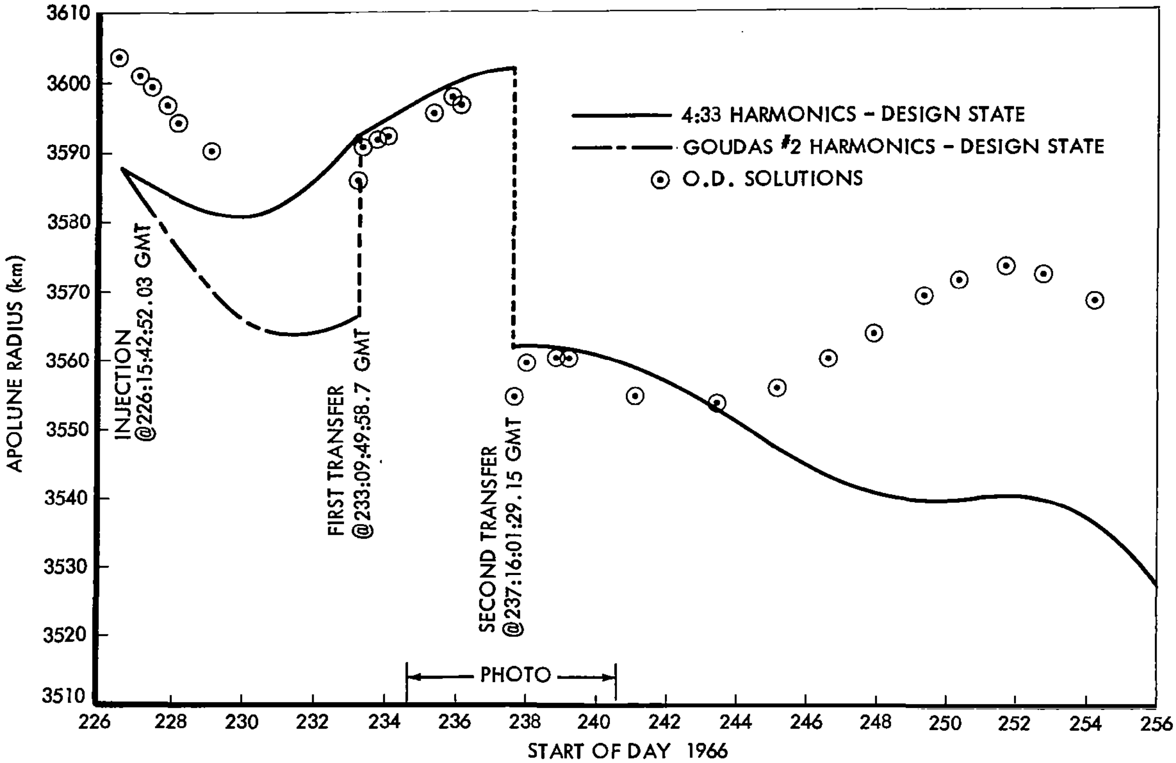


Figure 1.4-16: Lunar Orbiter Mission I Apolune Radius History

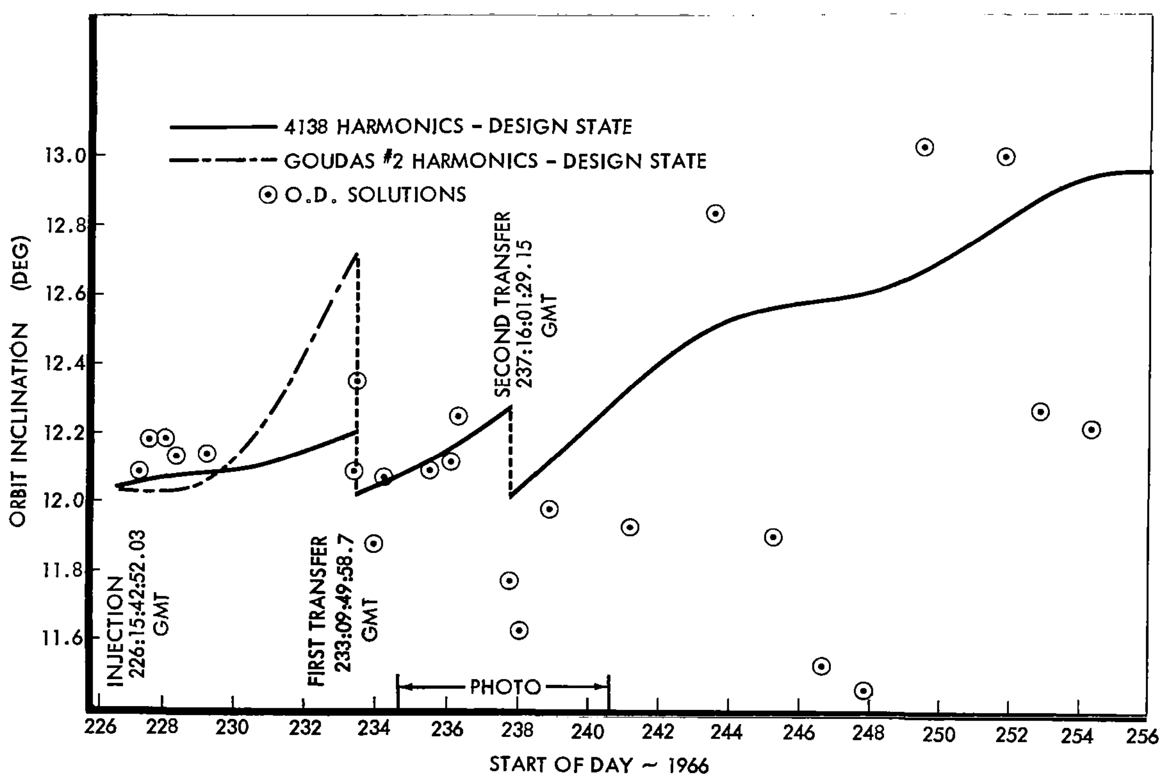


Figure 1.4-17: Lunar Orbiter Mission I Orbit Inclination History

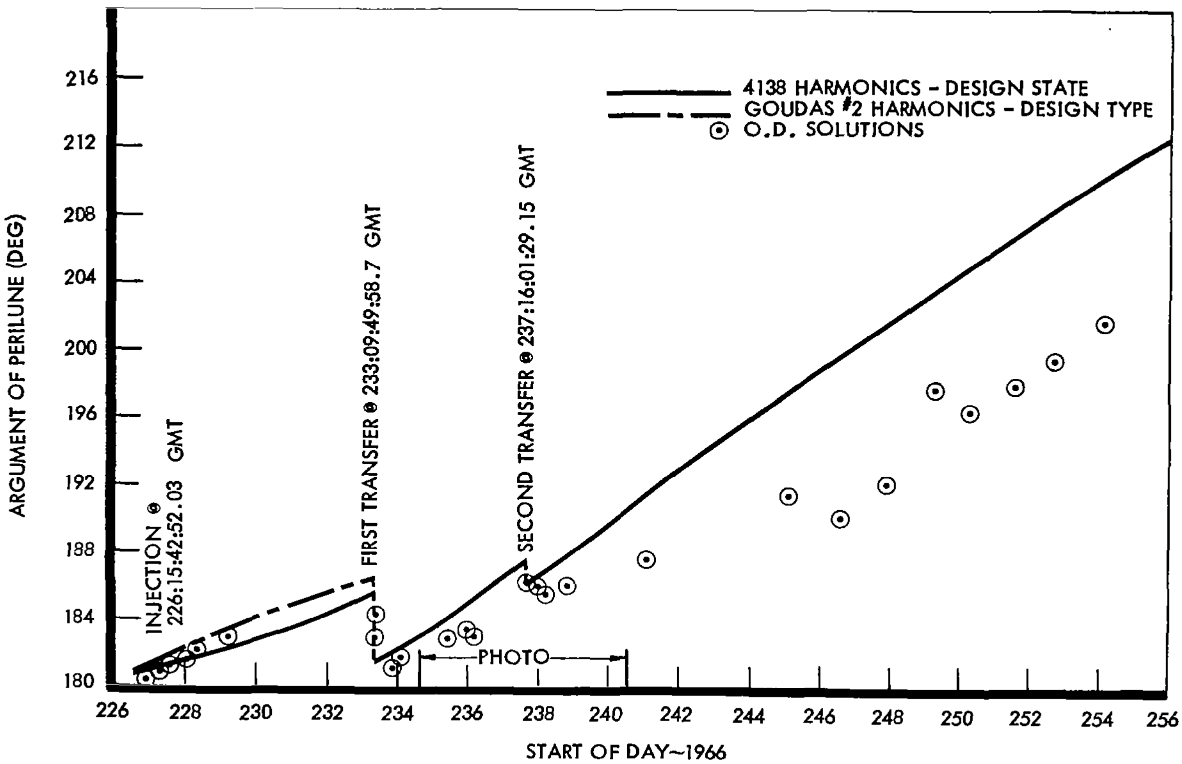


Figure 1.4-18: Lunar Orbiter Mission I Argument of Perilune History

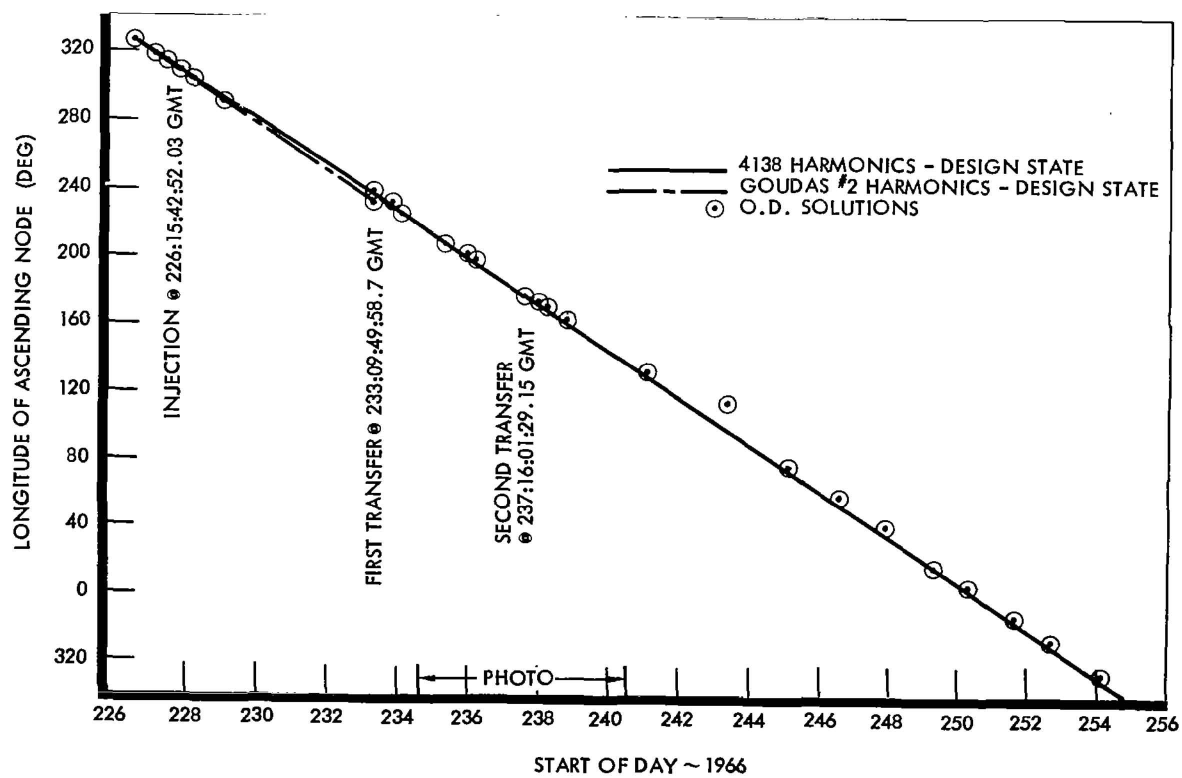


Figure 1.4-19: Lunar Orbiter Mission I Longitude of Ascending Node History

radius, orbit inclination, argument of perilune, and longitude of the ascending node. To clearly show the complete mission, these figures cover the 30-day period from lunar injection (Day 226 to Day 256) and, thus, include all three ellipses. The predictions were all based on a set of harmonics computed by the orbit determination group on Day 231 (known as the OD 4138 harmonics). These harmonics were not solved for until 5 days after injection; a prediction for the first ellipse was based on Goudas 2 harmonics.

Tracking data were recorded at the Deep Space Stations and the Space Flight Operations Facility to satisfy requirements for the selenographic data. The Deep Space Station recording was a five-level teletype paper tape. During the mission, the tracking data were transmitted to the SFOF via normal teletype messages. At the Space Flight Operations Facility teletype data were received by communications terminal equipment and passed to the raw data table on the 1301 disk by the IBM 7044 I/O processor. These data were processed by the TTYX program to seperate the telemetry data and tracking data in the messages received, and stored on the tracking raw data file on disk. The tracking data processor (TDP) program generated the master tracking data table on the 1301 disk by smoothing and sorting the data from the tracking raw data file by Deep Space Station identification. The output of this program was also recorded on magnetic tape and identified as the tracking data deliverable to NASA. An orbit data generator routine extracted selected master data file tracking data, smoothed it, sorted it according to time, and inserted it in the orbit determination program input file. Upon command from the FPAC area, orbit parameters were computed or predicted, based upon the selected data from the orbit determination program input file and the orbit determination program, and inserted into the data display files for subsequent display by the user.

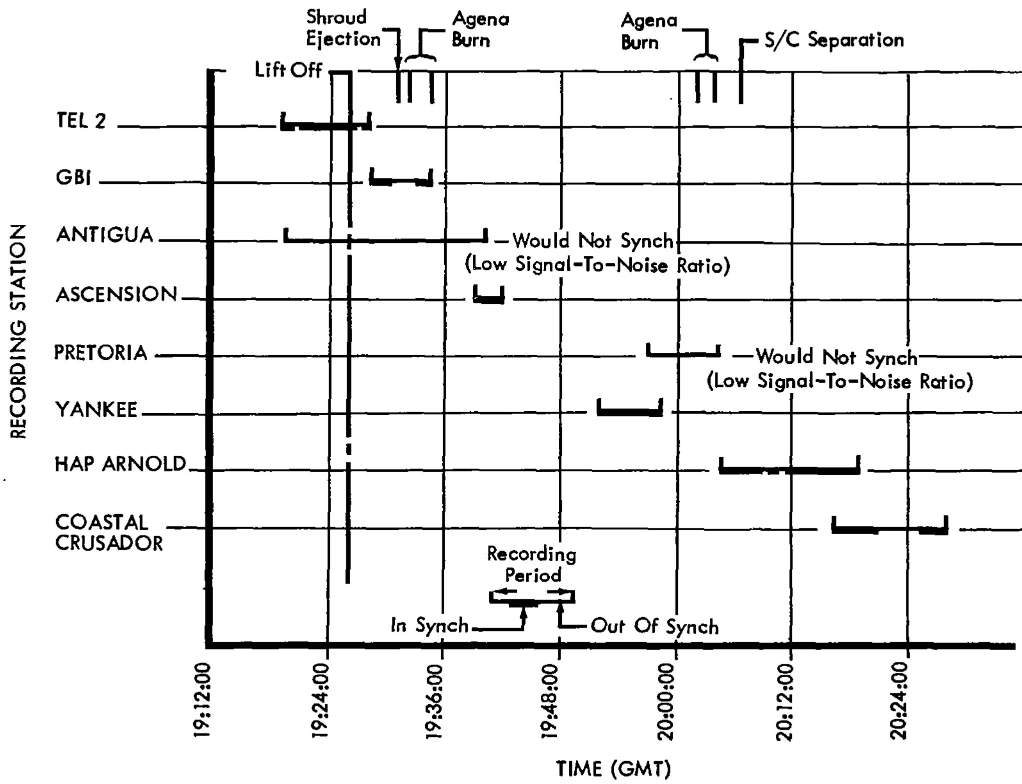
The raw tracking data paper tapes recorded at each Deep Space Station and the output of the tracking data processor at the Space Flight Operations Facility, recorded on magnetic tape, were collected and delivered to NASA for follow-on selenodetic analysis purposes.

1.4.4 TELEMETRY DATA

The performance telemetry data was recorded through all phases of the flight. Figure 1.4-20 summarizes the data obtained via "S" band transmission as recorded by the AFETR instrumentation stations and ships. Evaluation of the reduced performance data indicated that the values were within the anticipated operational limits. Electrical power subsystem measurements confirmed the expected buildup beginning after solar panel deployment and Sun acquisition.

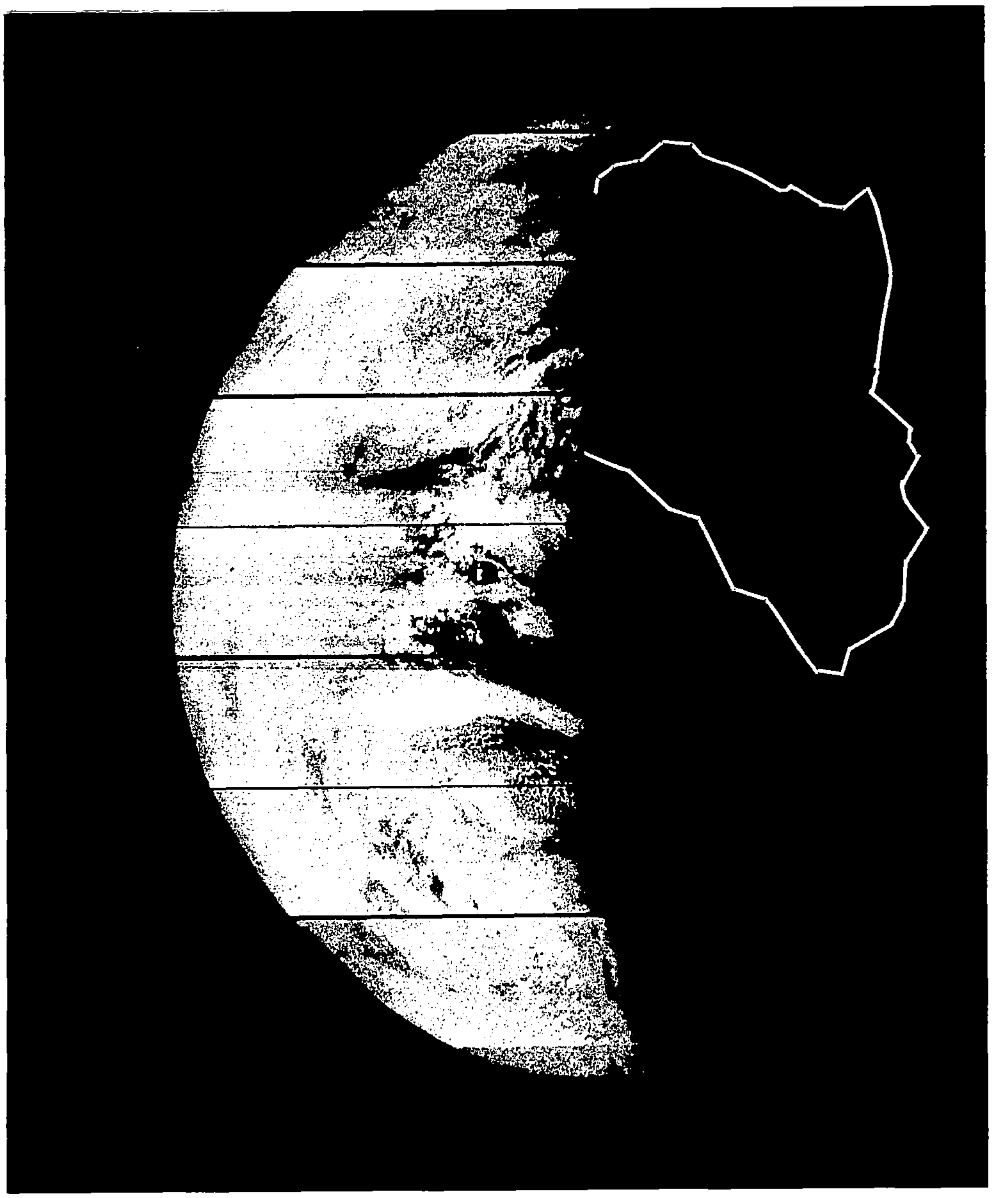
The DSN received and recorded performance telemetry data during all periods of spacecraft visibility, except for a 4-minute period on September 2, 1966 when the uplink lock was lost by the Madrid station due to a problem with the ground transmitter.

In all cases, the data was available for the subsystem analyst to continuously monitor the operational status of all spacecraft subsystems and environmental conditions. The vibrational accelerometer mounted on the upper module and connected to channel 12 of the Agena telemetry did not provide any data. One thermistor located on panel 3 provided intermittent data subsequent to Orbit 34 in the first ellipse. All other data channels provided reliable data during the entire mission.



Station	Data Record Min:Sec	Data Reduced Min:Sec
TEL 2	7:30	6:33
GBI	6:07	3:26
ANTIGUA		0:00
ASCENSION	3:00	3:00
PRETORIA		0:00
YANKEE	6:56	6:56
HAP ARNOLD	13:53	11:46
COASTAL CRUSADOR	11:31	6:54

Figure 1.4-20: AFETR Telemetry Summary (S-Band)



The Earth (Africa added)

1.5 MISSION EVALUATION

Lunar Orbiter I made significant additions to the operational techniques and data required to land a man on the Moon and return him safely to Earth. These accomplishments included, but were not limited to:

1) Accurately established the predicted initial elliptical lunar orbit with transfers to additional lower perilune altitudes.

2) Provided large-area photographic coverage (including stereo photography) of nine potential Apollo landing sites.

3) Provided extensive photographic coverage of the farside of the Moon and additional areas on the frontside (including potential future Lunar Orbiter sites and potential navigation landmarks for Apollo).

4) Eliminated some Mission B sites from further consideration as potential Apollo sites based on the evaluation of photo coverage from this mission.

5) Demonstrated multiple-restart capability of a liquid-propellant rocket system over extended periods. (Engine developed for Apollo program)

6) Demonstrated successful application of a gimbaled engine thrust vector control system in space.

7) Demonstrated the ability to reliably command and control the spacecraft attitude during 374 maneuvers.

8) Demonstrated the application of a ranging system for establishment of space vectors and ranging techniques for synchronizing DSS clocks.

9) Provided data from which to determine the lunar model coefficients for a 12-degree orbit inclination.

10) Provided initial oblique detailed photos of the lunar surface.

11) Provided initial photos of the Earth as seen from the vicinity of the Moon.

The extensive maneuvering capability of the spacecraft, the flexibility of the operational command and control, and the adaptability of the computer programs were thoroughly demonstrated during the mission. These included the rapid generation and evaluation of commands, their transmission to and execution by the spacecraft. The commands were based on the changing requirements of nonstandard events and real time definition of sites for film-set photography. At all times the subsystem analysts were provided with performance telemetry data and subsystem performance computations to support their analysis of nonstandard events.

Lunar environmental data obtained, including the measurement of a solar flare, contributed to the further definition of the environmental conditions expected during lunar missions.

Extensive single-and two-station tracking data, obtained over extended periods in each of three lunar orbits, is being used in conjunction with data from the extended mission to refine the mathematical model of the Moon's gravitational field. The accuracy of achieving the desired lunar orbit by executing commanded attitude and velocity changes confirmed the validity of assumptions used in orbit computations. This additional confidence can be used to more precisely plan future Lunar Orbiter missions and also extrapolated to support other types of lunar exploration missions.

During the Lunar Orbiter I mission, the spacecraft encountered five incidents that had varying operational effects on subsystem performance. Only one of these incidents, the operation of the high-resolution-camera shutter, effected the accomplishment of the mission objectives and degraded the photo data obtained. This problem could not be circumvented by inflight procedural changes because the shutter control circuitry was susceptible to electromagnetic interference. The remaining four incidents were satisfactorily evaluated and controlled through the inherent flexibility of the mission command and control concept. The space flight operations director implemented minor changes in operating or spacecraft control procedures so that the irregularities produced no degradation in mission objectives or the data obtained.

Evaluation of the reconstructed moderate-resolution photos of the primary sites showed examples where the 80-mm-system detection requirements were met at each site. In addition, the photos also showed the general photographic problem created by the variation of lunar topography and albedo evident at all sites. These surface characteristics resulted in wide variations of exposure in a single photo and in many cases the resultant exposure in adjacent areas exceeded the system limitations at both extremes. Photographs taken on both the near and farside of the moon revealed surface characteristics that have not been previously observed. The high-resolution photographs provide limited data that can provide additional qualitative data when used in conjunction with the corresponding moderate-resolution photos. In general the moderate resolution photography from Lunar Orbiter I provided an enormous amount of data that can be used to support the selection of potential Apollo landing sites, assist in geologic evaluation of the lunar surface, and provide additional visibility for preparation of lunar maps.

NASA-Langley, 1967 —— 30