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LUNAR ORBITER: ITS MISSION AND CAPABILITY

by

Israel Taback

Spacecraft Manager: Lunar Orbiter

NASA LANGLEY RESEARCH CENTER, Langley Station, Va.

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Historical Background

In 1960, JPL directed a feasibility study, called the Visual Observation Instrumentation System, for lunar photography, by RCA, Eastman-Kodak Company, and Fairchild. Shortly thereafter the Langley Research Center investigated a method of lunar photography and made a study of the required trajectory. The results of these and other lunar orbit trajectory studies were subsequently published as technical notes, such as NASA TN D-1226 and NASA TN D-1780. In 1962, JPL sponsored industry studies of a Surveyor Orbiter spacecraft and a competition for a Visual Instrumentation Subsystem - this was not finalized. In early 1963, OSS funded a feasibility study with the Space Technology Laboratory concerning a spin-stabilized orbiter. A request for proposal was issued by Langley Research Center in August 1963 (L-3270, August 30, 1963). The industry proposals, in response to the RFP, were received October 4, 1963. The Boeing Company, with RCA and Eastman-Kodak Company as subcontractors, was selected for contract negotiations December 20, 1963.

Project Organization

Fig. 2 is an organizational diagram indicating the responsibilities of the major contributors to the Lunar Orbiter Program. Program direction is provided by the Office of Space Sciences, Lunar and Planetary Programs, at Headquarters level. This direction insures that the program will fulfill the requirements of manned space flight and the scientific community. Project direction under these general policy guidelines is delegated to the Langley Research Center and will be executed at this Center within a Lunar Orbiter Project Office. The responsibility of this group extends from the generation of the detailed specifications for the project through the execution and delivery of the data to the data users. This office has delegated an overall project analysis and integration responsibility to a prime contractor, who, with his subcontractors and supplies will furnish the spacecraft and necessary auxiliary ground equipment. The launch vehicle will be furnished and launched by direction of the Lewis Research Laboratories. Launch facilities will be provided by AMR, and ground-based tracking and telemetry will be accomplished through the JPL operated DSIF. Mission control is planned at the SFOF at Pasadena, which is being built and will be operated by JPL. The project organization shown is similar to that used on all of the current NASA unmanned space programs.

Project Constraints

A basic policy of relying on proven components, systems, and procedures resulted in a number of design restraints, which fortunately are not unduly restrictive. Fig. 3 lists these restraints. All are chosen so that the projected time schedule could be met, with a minimum of foreseeable developmental problems to be solved. Also, the advantages of experience and proven techniques in operating the launch, tracking, and communications complex are immediately available.

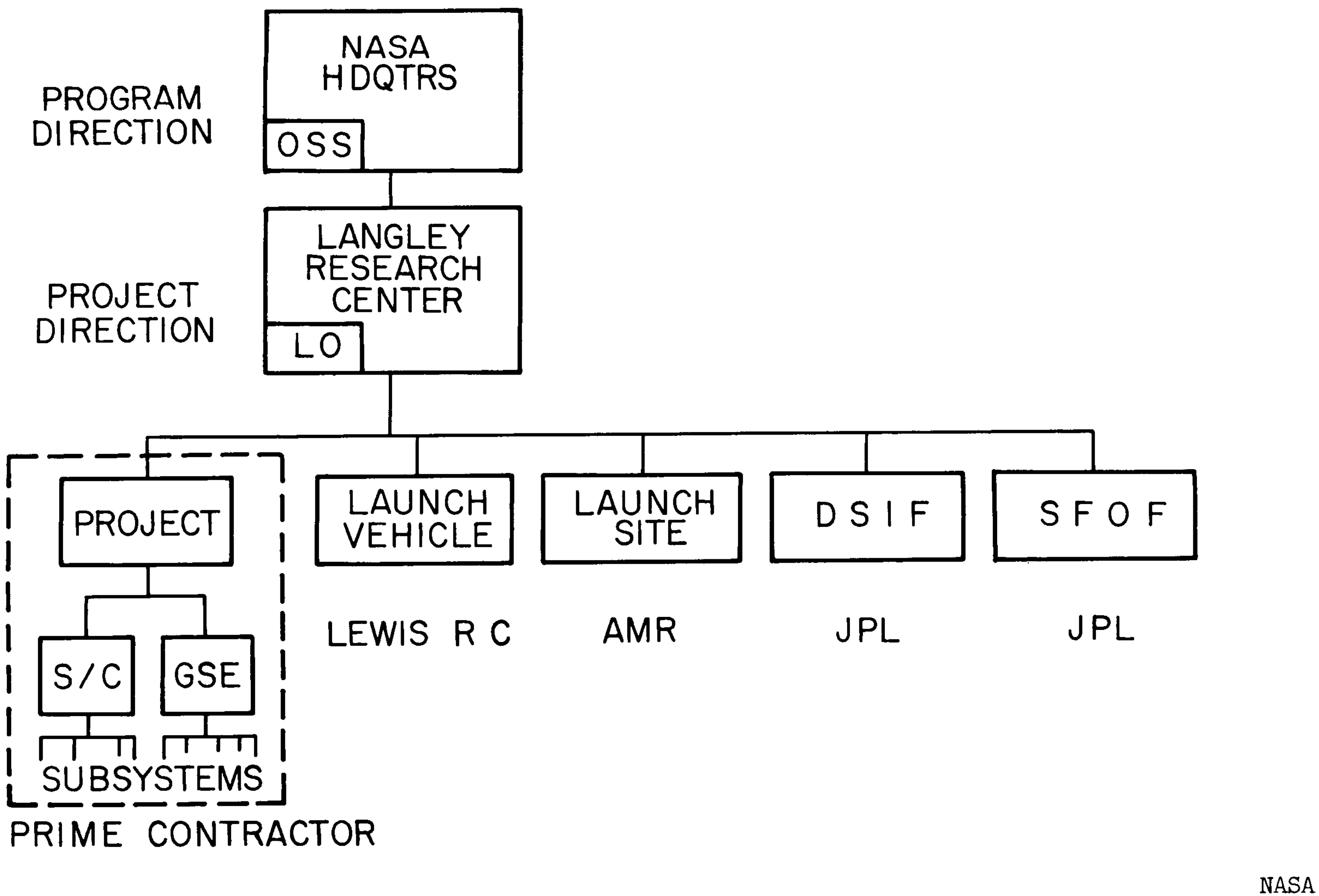

Figure 2.- Lunar orbiter project organization.

1. Launch with proven vehicle (Atlas-Agena)
2. S/C design with conservative weight limitation
3. Launch from AMR
4. Flight operations and control from DSN
5. Maximum use of space flight qualified components

Figure 3.- Mission constraints.

Project Requirements

The basic requirements for the Lunar Orbiter are aimed primarily at securing capability for photographic coverage of the lunar surface over a range of resolutions. Although this vehicle is not a "bus" concept, requirements for measuring other phenomena in the lunar environment have been imposed to extend the range of usefulness for general scientific purposes.

The photographic coverage, selected on the basis of requirements for Apollo and tempered with the knowledge of what capabilities are within the state of the art, is illustrated in Fig. 4. There is a requirement for being able to cover the entire lunar surface; however, the area of primary interest for Apollo is the area bounded by $\pm 10^\circ$ latitude and $\pm 60^\circ$ longitude. Within this area a coverage of 40,000 square kilometers is specified, at a resolution of at least 8 meters. This may be considered to be the site selection mode for Surveyor landings; within this area are approximately 100 possible 20-kilometer square potential landing sites. The coverage at a resolution of 1 meter is specified at a total of 8,000 square kilometers. In this high resolution mode it is necessary that spot coverage of areas having a radius of at least two Surveyor CEP's be obtained. This requires about 200 square kilometers. The remainder of the capacity is available for spot coverage of up to 40 similar areas, or can be used in modes too numerous to list herein. A detailed description of the proposed camera system which meets these requirements will be given in a later section of the paper.

In order to secure coverage and transmittal to earth of the primary area shown in the figure, operational times of less than 1 month are required. However, to secure the scientific objectives of selenodesy and lunar environmental measurements a design lifetime goal of 1 year is specified. The initial vehicles will be tracked by the DSN to secure a measure of the harmonics of the lunar gravitational field, and measurements of micrometeoroids and high-energy particles will be made.

Lunar Orbiter Configuration

A photograph of the mock-up is shown in Fig. 5, and another view in Fig. 6. The overall size of the spacecraft is about 5 feet in diameter and 6-1/2 feet in the folded configuration. The estimated weight is approximately 830 pounds. The in-flight configuration in translunar trajectory is shown in Fig. 7. Primary attitude reference is secured from a Sun-line-Canopus-line determination. This attitude is achieved exactly as in the Surveyor, by acquiring a sun fix and rolling until Canopus is acquired. The sun panels, directional and omni antennas are then extended. The directional antenna has one degree of freedom about

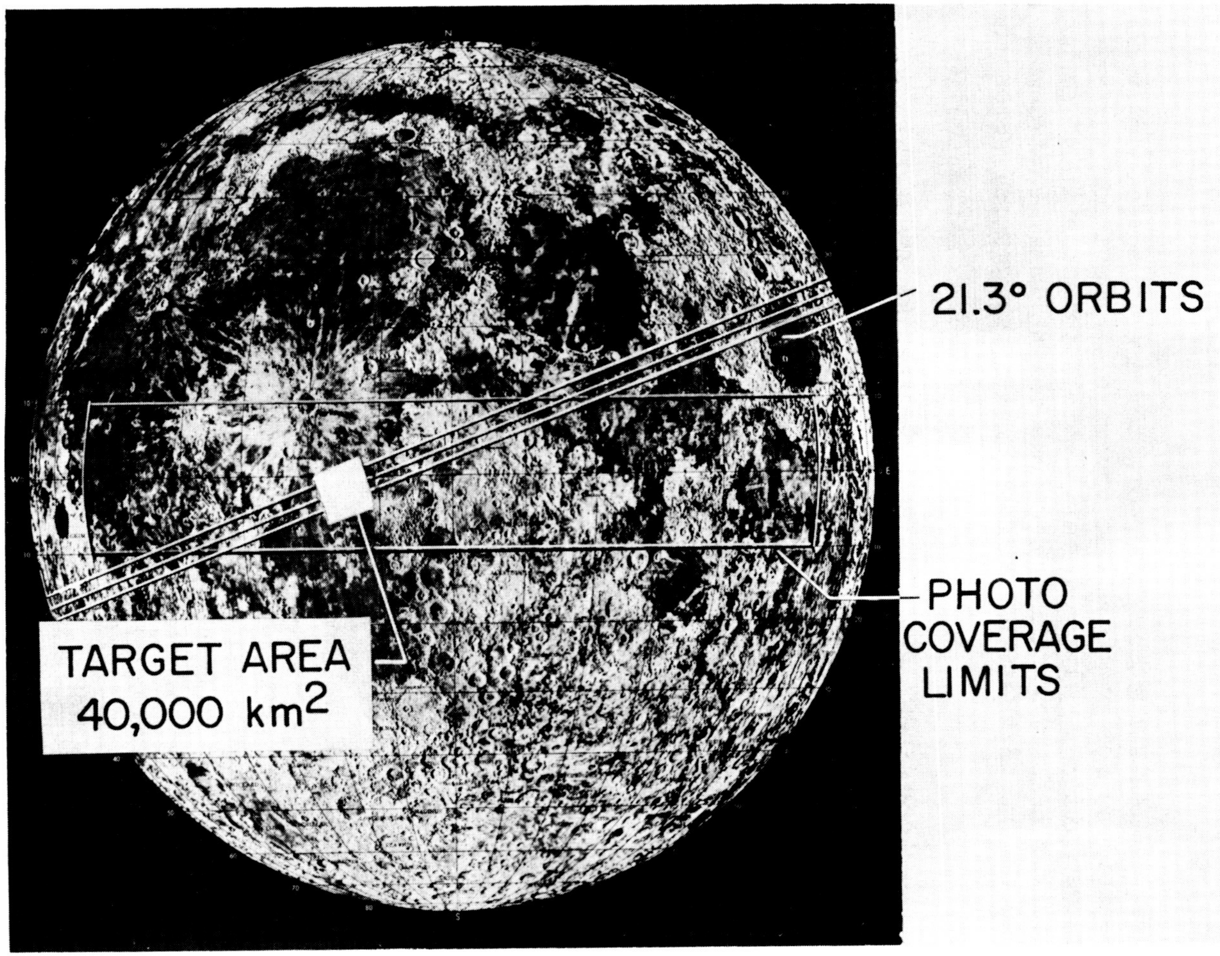

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Figure 4.- Photographic coverage and representative orbital tracks.

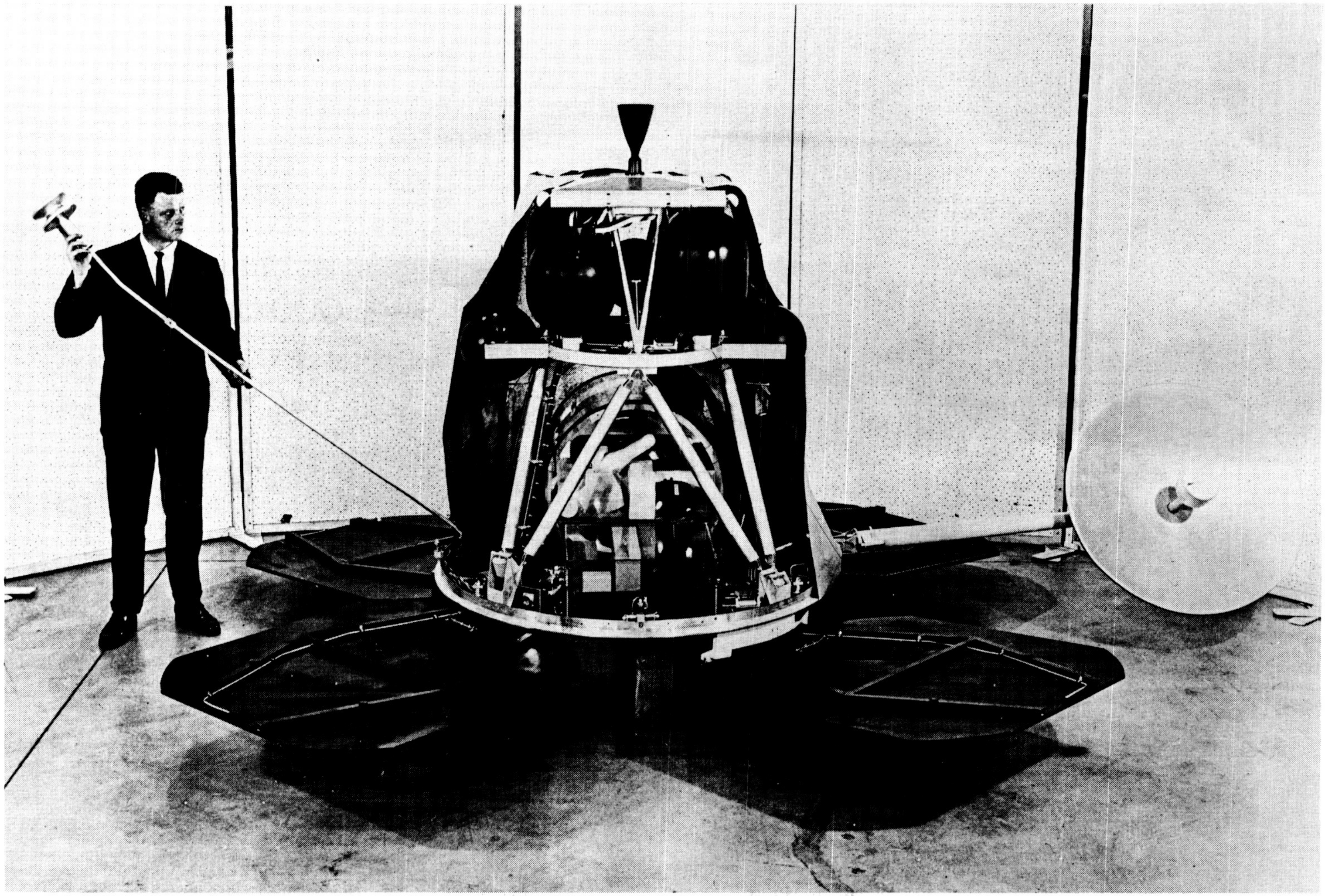

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Figure 5.- Lunar orbiter.

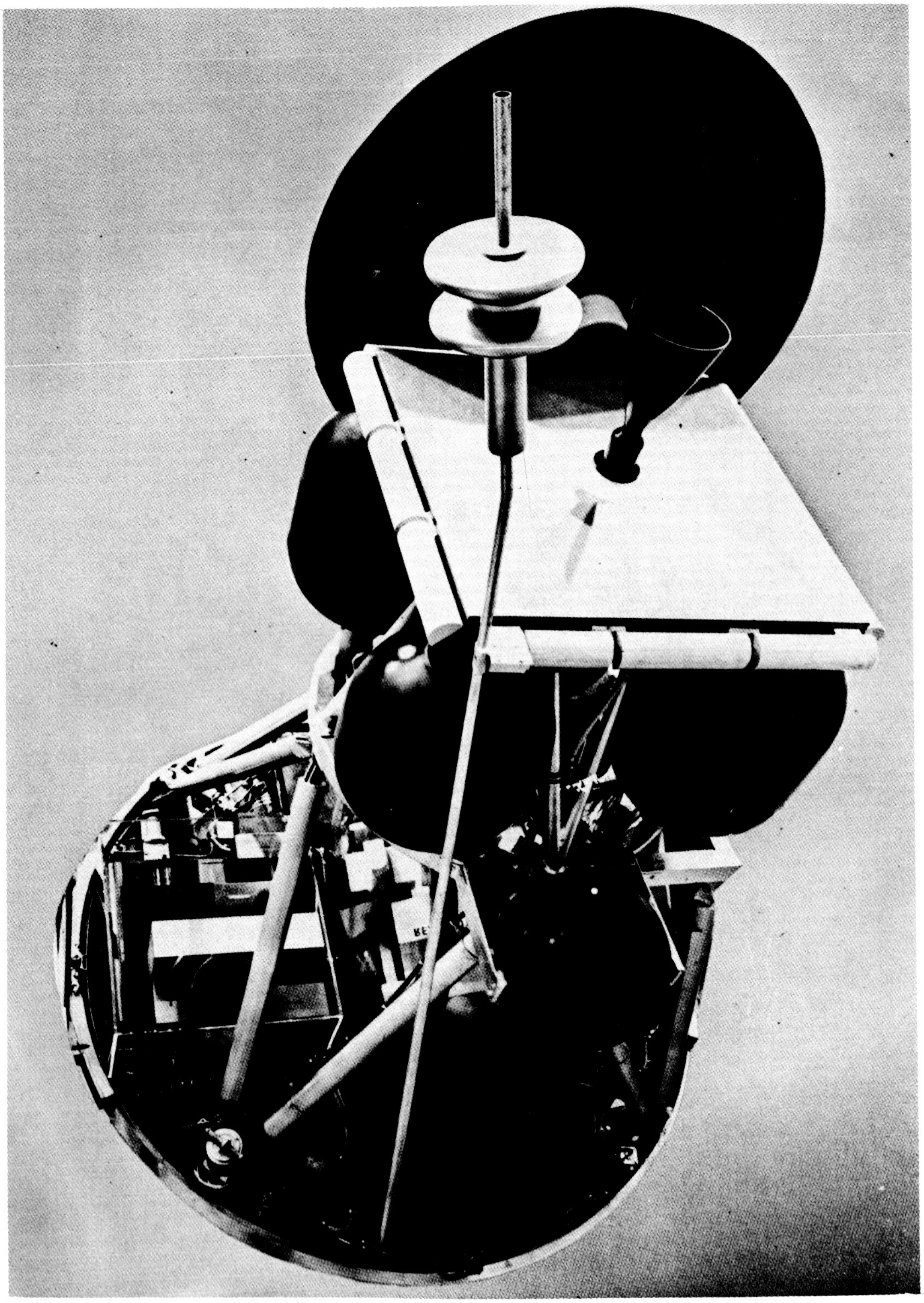

Figure 6.- Lunar orbiter, folded configuration.

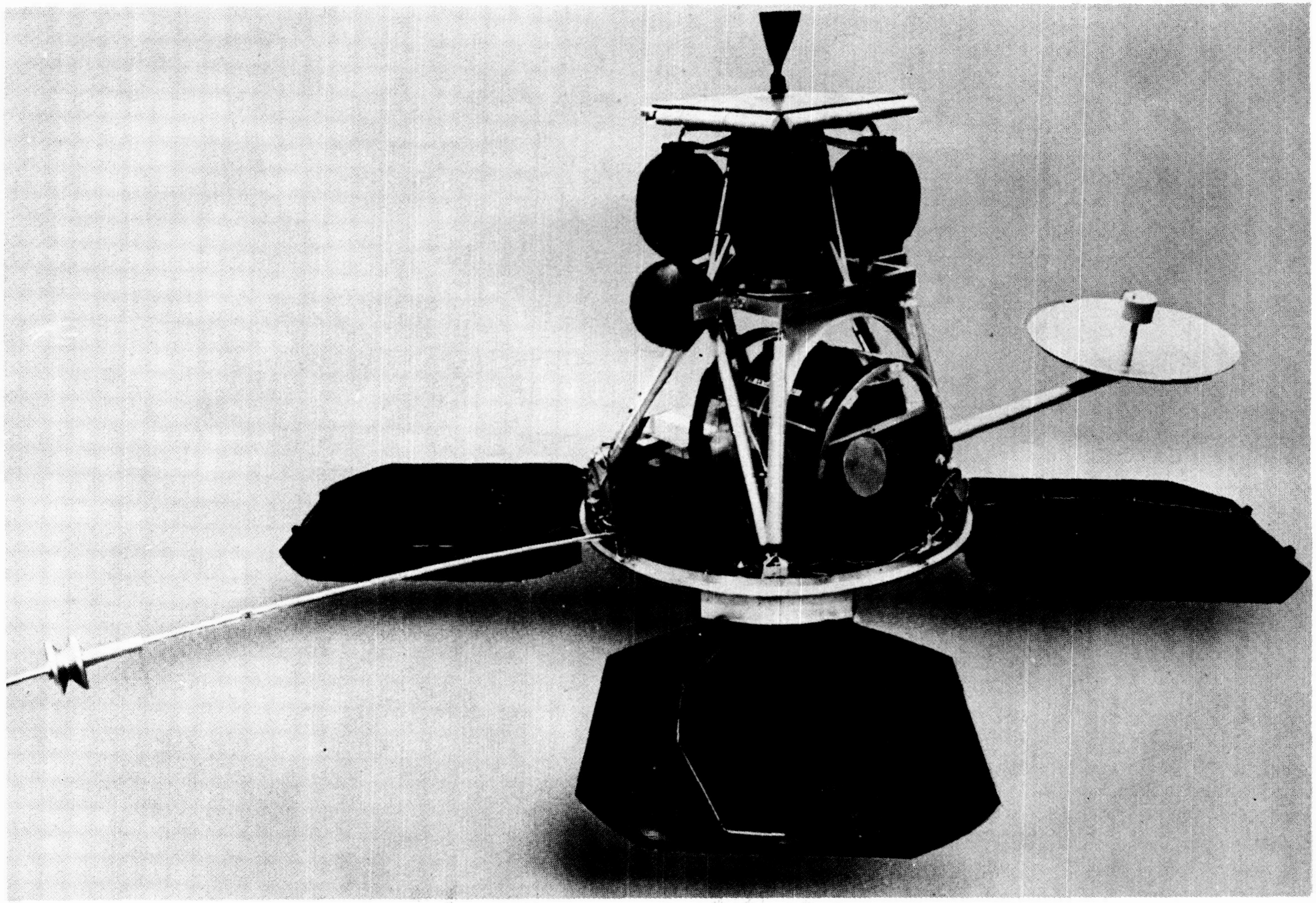

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Figure 7.- Lunar orbiter, flight configuration.

an axis perpendicular to the lunar orbital plane, and is either clocked or commanded to aim toward earth. Midcourse and retroattitude changes are made with reference to an inertial reference point, and midcourse and lunar orbit retro velocity changes are monitored by a precise accelerometer.

Because of the almost complete similarity between Ranger, Surveyor, and Orbiter launch, injection, and translunar trajectories, these will not be discussed. Significant differences exist however because of the nature of the photographic mission. These are the launch window constraints, lunar orbit maneuvers and corrections, photographic system and lighting constraints, and the video data transmission constraints.

A typical series of maneuvers starting at lunar retro is illustrated in Fig. 8. Normal injection and retro occur at 950 kilometers. After a number of orbits in which low-resolution photography may be acquired, stored on film, and then read out and transmitted, a retro for placing perilune above the region of interest is commanded. In the high-resolution photographic mode the attitude of the spacecraft is commanded from the reference attitude so that the vehicle axes are alined with the direction of flight. A precise yaw control, based on sensing of transverse image motion by a two-axis V/H sensor, is commanded. Pitch and roll attitude are open loop and depend upon integrating the gyro signals in these axes from the inertial reference position.

The launch window constraints are outlined in Fig. 9(a). For a selected target area, launch time must be selected so that there is a proper angle of sun illumination when the spacecraft arrives at target. This allows two launch windows per month (a.m. and p.m. illumination). The duration of each window is prescribed by the tolerance on sun angle, and could be as long as 1 day. A selected launch window prescribes the lunar declination, and in Fig. 9(b) is shown the loci of injection points for a range of launch azimuths from AMR.

The lighting constraints, target covering capability, and sensitivity of the photographic system can be derived in detail as follows: Fig. 10 (from JPL Rep. 32-384) is a plot of the lunar photometric function, where ϕ is a measure of the reflectivity of the surface, g is the phase angle (angle between line of sight and sun line), and α is the inclination of the surface normal with respect to the line of sight. The case wherein these angles all lie in one plane is illustrated in Fig. 10. The plot gives the sensitivity of the photographic system with lighting angle to changes in slope of the surface. For example, at a constant value of $g = 60^{\circ}$, if the slope α changes $+15^{\circ}$ (away from the sun), the light intensity drops to one-half the intensity at $\alpha = 0$. The enclosed area illustrates the operating conditions for a downward looking low-altitude camera having a square field of view of $\pm 30^{\circ}$. The low-resolution camera planned for the lunar orbiter has a field of view of about $\pm 20^{\circ}$ and will cover the operating range shown within the shaded area. There is good sensitivity to slopes within this area; however, the absolute intensity of a flat lunar surface may vary by almost 2:1 across the frame.

Fig. 11 illustrates how the lighting conditions for 20° inclination orbital paths progress with respect to the lunar surface. Posigrade

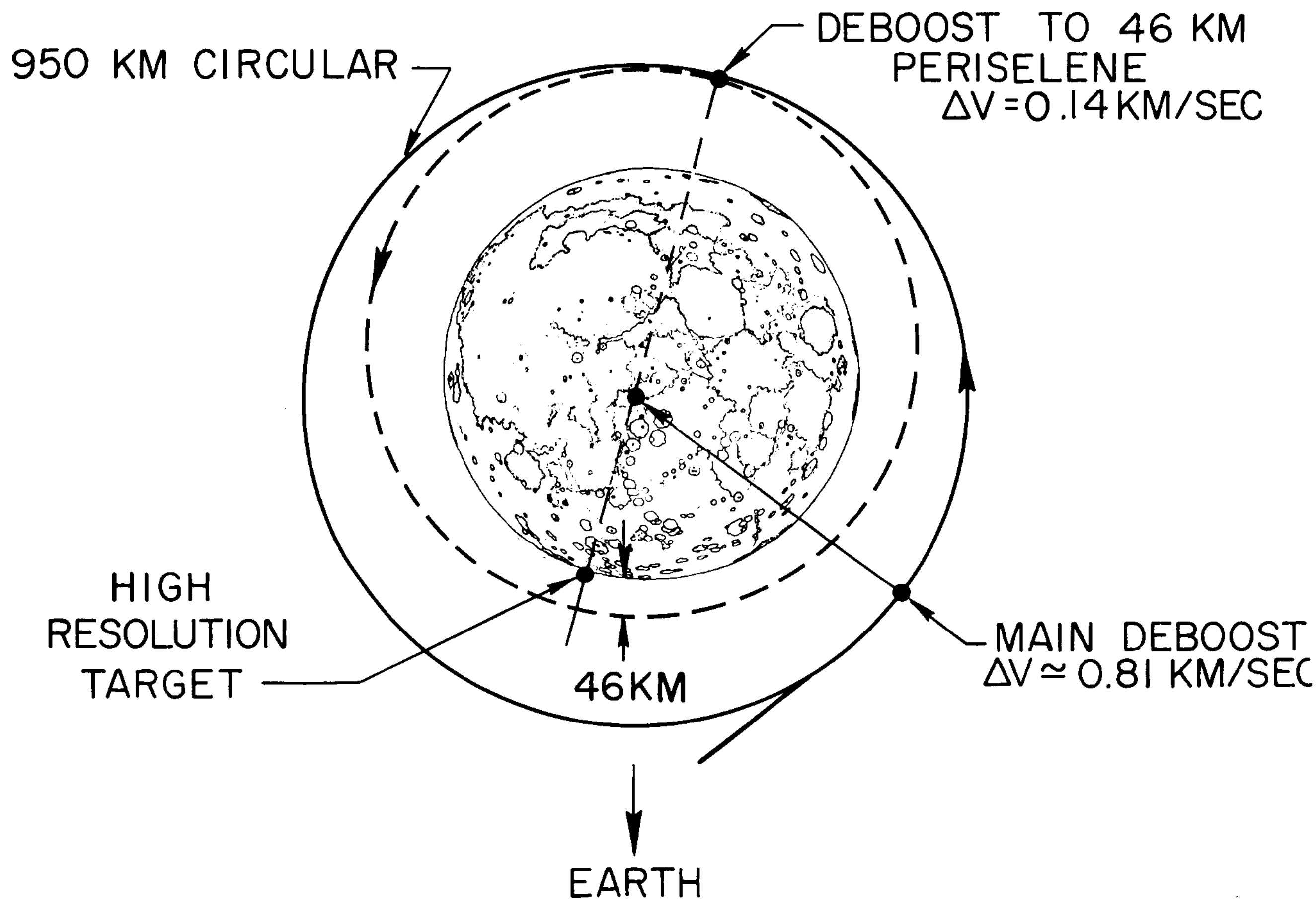

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Figure 8.- Typical lunar orbit shown in plane of approach hyperbola.

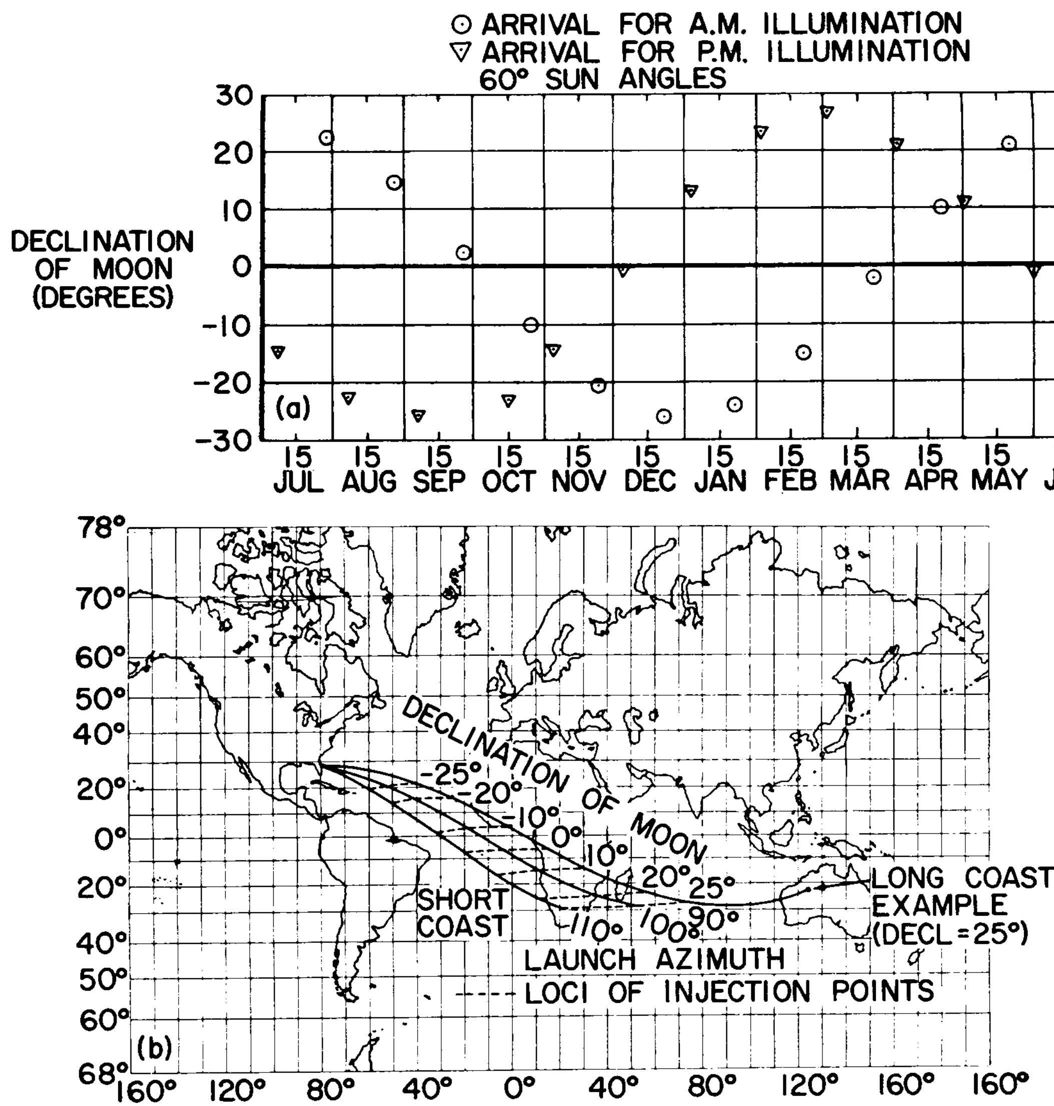

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Figure 9.- Arrival opportunities and injection positions.

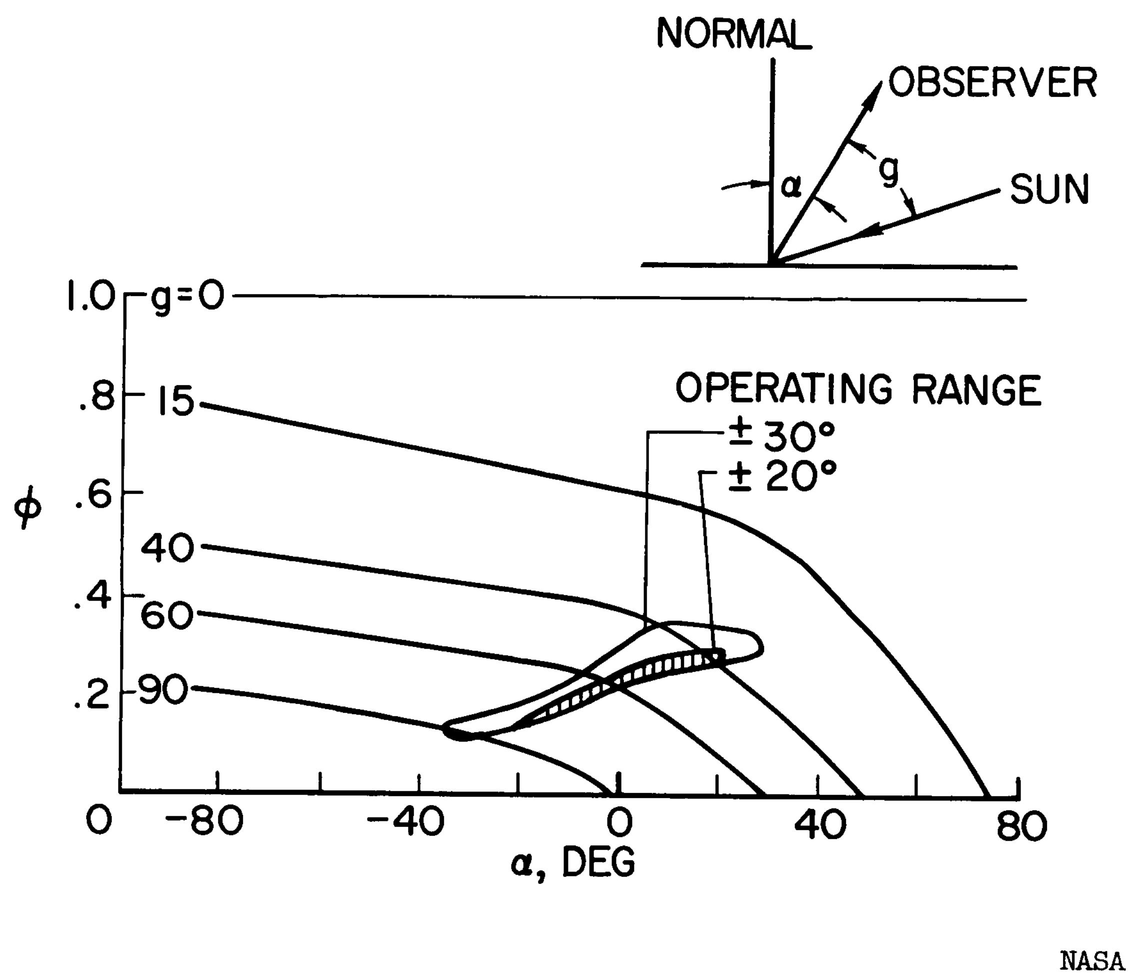

Figure 10.- Lunar photometric function.

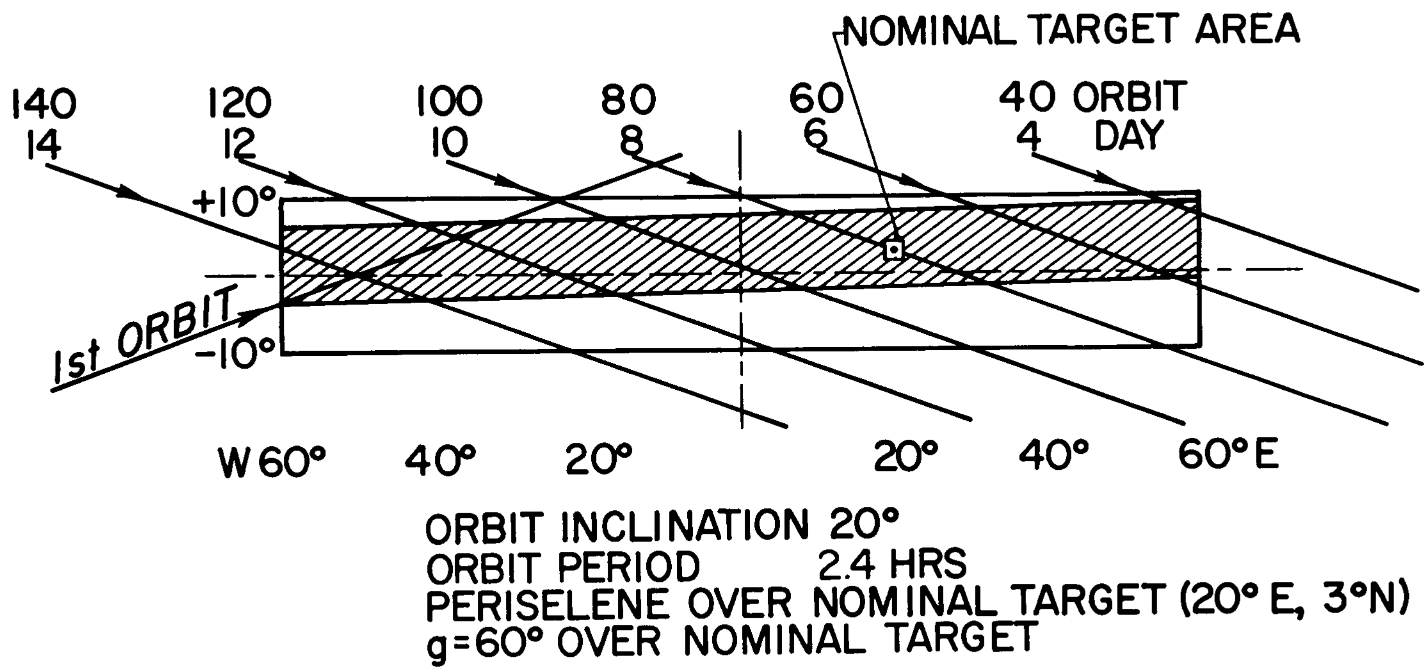

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Figure 11.- Target selection area for 20° inclination orbit.

orbits are shown with the initial ascending nodal point at about 50° W. longitude. An arbitrary target has been selected and is shown at 20° E., 3° N. An initial waiting time of 4 days is required in order for the descending node portion of the orbit to begin intersecting the region of interest. After an 8-day waiting period, the 80th orbit intercepts the target. The timing of this event is such that there is proper illumination (g angle equal to 60°). It is also possible on this orbit to take satisfactory pictures over the range included by the shaded area. In this region, because the pitch angle of the spacecraft is held constant, the value of g does not change. However, the mean surface inclination does change so that a range of α is encountered. The range of α shown is $\pm 10^\circ$. This is a variation which affects the quality of the picture but is still within the specifications. It should be noted that as the orbit rotates with respect to the lunar surface at 13° per day and the illumination rotates in the same direction at the rate of 12° per day there is a slight vernier effect which shifts the latitude over which satisfactory pictures can be taken. It may be concluded from this figure that it is possible to photograph on a single flight any targets located within the shaded area of Fig. 11 with a maximum waiting time of about 15 days. The latitude of the usable area can be shifted by properly timing the orbits with respect to the illumination. On any one flight, targets will be restricted to within a similar area.

The video data transmission windows are illustrated in Fig. 12 for low-inclination orbit. This figure shows the time per orbit wherein the spacecraft is simultaneously in view of both the sun and earth. This is a basic restraint as only in sunlight is sufficient power available to activate the video data transmission system. An average visibility time of about 46 minutes per orbit is available during the month. This is sufficient to transmit one frame of high- and low-resolution photography. The total photographic data stored on the spacecraft can be transmitted to earth in about 10 days.

The following material will cover in some detail a number of the specific subsystems which make up the spacecraft. Because of the high reliance on proven concepts, techniques, and equipment, only a few descriptive remarks will be made concerning each system; however, some attempt will be made to describe the design and capabilities of the photographic data acquisition system and allied communications link.

Subsystem descriptions.-

1. Structural subsystem is shown in Fig. 13. This is a simple truss taking loads from the engine support module to the equipment mount plate, and allows access to all internal equipment without disassembly.

2. The power subsystem is shown in Fig. 14. Prime power is supplied by solar panels and stored by nickel-cadmium batteries. Almost all of the electrical subsystems have been previously developed or qualified in other space programs such as Relay, Tiros, and Ranger.

3. The velocity control system is shown in Fig. 15. This is a completely self-contained module, employing a Marquardt 100-pound thrust

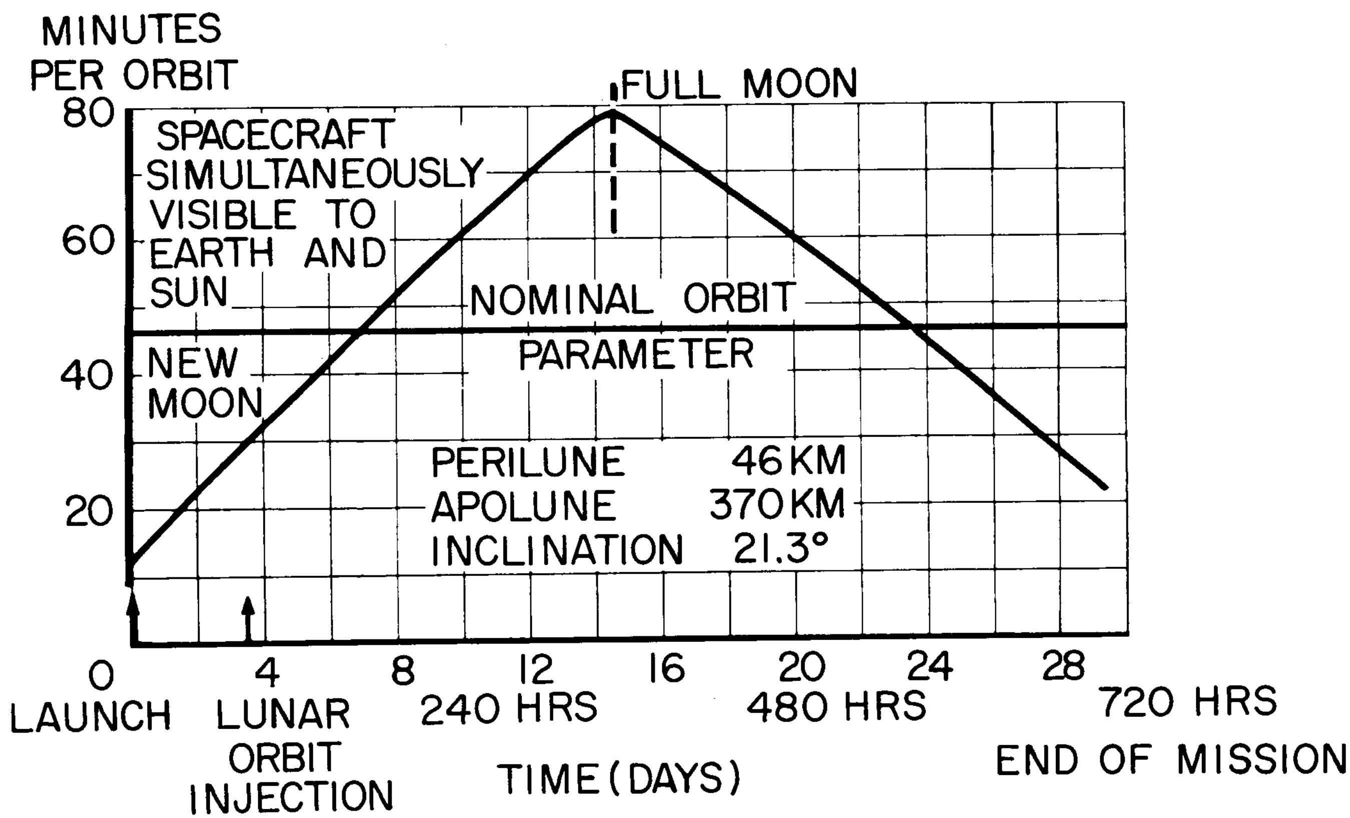

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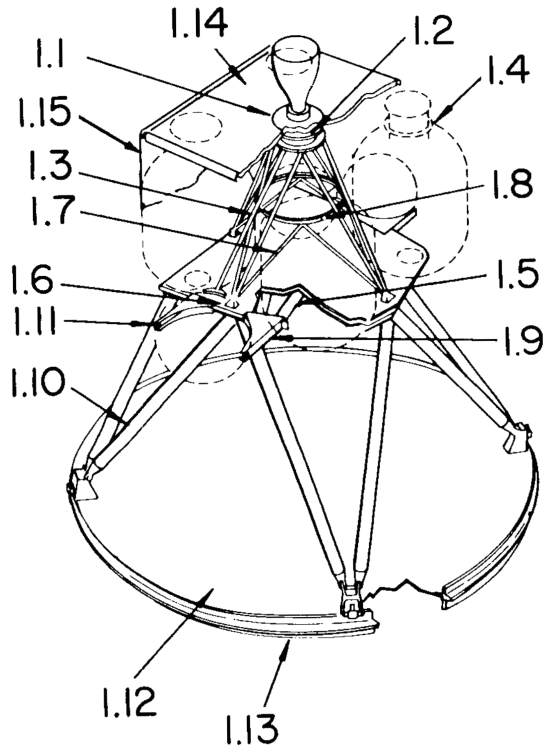
Figure 12.- Photo data communication time.



1.1 ENGINE MOUNT ADAPTER
1.2 ENGINE MOUNT RING
1.3 ENGINE SUPPORT TRUSS
1.4 TANK LATERAL SUPPORT
1.5 TANK MOUNT FITTINGS
1.6 TANK DECK
1.7 BOTTLE SUPPORT TRUSS
1.8 BOTTLE SUPPORT TUBES
1.9 BOTTLE SUPPORT BRACKETS
1.10 PRIMARY SUPPORT TRUSS
1.11 SUPPORT GUSSET
1.12 EQUIPMENT MOUNT PLATE
1.13 PAYLOAD-ADAPTER ATTACH RING
1.14 BASE HEAT SHIELD
1.15 EQUIPMENT THERMAL SHIELD

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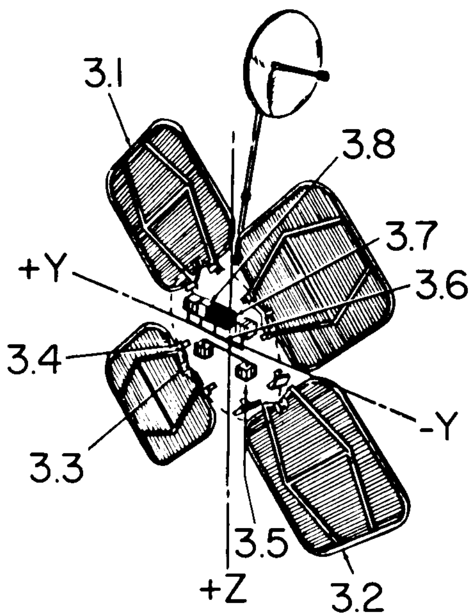
Figure 13.- Spacecraft structure.



3.1 SOLAR CELLS
3.2 SOLAR PANEL STRUCTURE
3.3 SOLAR PANEL DEPLOYMENT ACTUATORS
3.4 SOLAR PANEL DEPLOYMENT STRUCTURE
3.5 BATTERY CELLS
3.6 BATTERY STRUCT, WIRING
3.7 BUS VOLTAGE REGULATOR
3.8 BATTERY CHARGER CONTROL AND
BOOST REGULATOR

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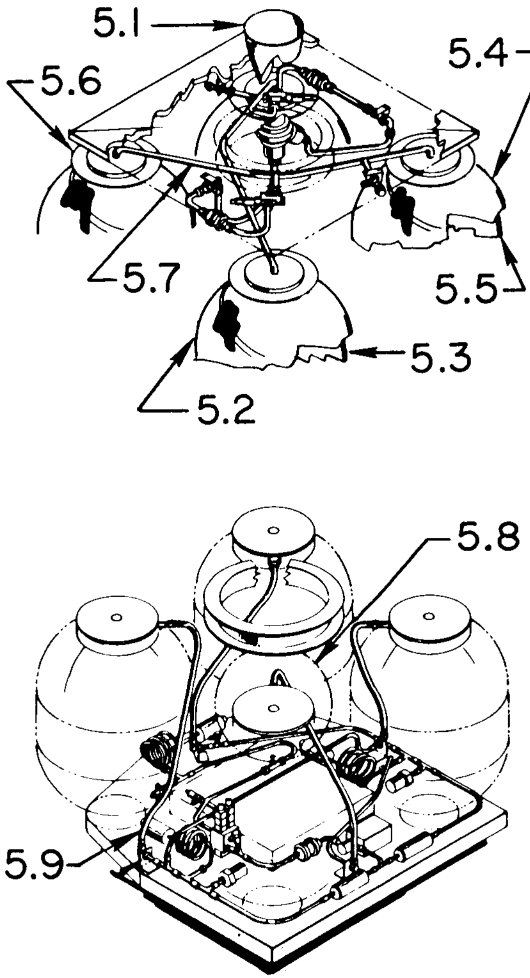
Figure 14.- Power subsystem.



5.1 LIQUID ROCKET ENGINE
5.2 FUEL TANK PRESSURE SHELLS
5.3 FUEL ORIENTATION & EXPULSION
5.4 OXIDIZER TANK PRESSURE SHELLS
5.5 OXIDIZER ORIENTATION & EXPULSION
5.6 PROPELLANT TANK ATTACH. & FITTINGS
5.7 PROPELLANT FEED SYSTEM
5.8 PRESSURE GAS BOTTLE
5.9 PRESSURIZATION SYSTEM

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Figure 15.- Velocity control.

(MA-109-XAA) bipropellant engine. The fuel is nitrogen tetroxide + aerozine 50, pressure-fed by nitrogen gas. Most of the components, such as valves, regulators, and filters, have undergone qualification tests for Apollo systems. The main engine is still under development; however, it appears to be the best possible choice at present.

4. The Attitude Control System is shown in Fig. 16. It may be broken into basic sensors, switching logic, and thrusters. Again, almost all components have been qualified at the component level. The prime reference sensors are sun sensors and a Canopus sensor. A three-gyro package containing a ΔV accelerometer is used to provide references when departures from prime attitude are required; the accelerometer is used to control midcourse ΔV corrections and for engine control during lunar orbit changes. A V/H sensor, which is packaged with the camera, provides a yaw signal to the attitude control logic during the photographic mode. There are two sets of thrusters - 1-pound jets located on the solar panels (used during engine burn) and 0.05-pound jets on the main structure used for normal control (engine off). Thrust is secured from pressurized nitrogen gas.

5. Fig. 17 shows the communication system block diagram. The main elements are the antennas, transponder, data conditioner, decoder, and programer. The system will operate at S-band, transmitting 10 watts through the directional antenna in the video data transmission mode, and 1/2 watt in the nonvideo mode through the low-gain antenna. The block diagram is self-explanatory. The video data bandwidth is about 1/4 megacycle, and is impressed on a vestigial single sideband subcarrier which then phase-modulates the carrier. Commands consist of 21 bit words. These are received and retransmitted for verification to the DSN, then gated into the programer or directly used as necessary. The programer allows for up to 16 hours of stored program for control of the spacecraft and internal subsystems.

The frequency spectrum transmitted is shown in Fig. 18. The video data extend from 80 kilocycles up to where it is attenuated by 6 dB at 310 kilocycles. The telemetry is carried at 30 kilocycles and a video pilot tone for synchronous detection is generated at 38.75 kilocycles (1/8 of the 310-kc video subcarrier). The command gates and data are sent on subcarriers ranging in frequency from 560 to 1300 cps, not shown in this figure.

Fig. 19 is a view of the camera package installed on the spacecraft, and Fig. 20 is a photograph of the available ground reconstruction equipment. Because these systems are unique to the Orbiter and the photographic requirements have designed the spacecraft to a large degree, a detailed description will be given of the proposed camera, readout system, ground reconstruction equipment, and the format and coverage which can be secured with a normal film loading. Again it should be mentioned that although a complete system has not been qualified, almost all of the major components have been qualified either for space or aircraft use.

Fig. 21 is a schematic of the camera system. Two lenses are employed, a 24-inch focal length, f/5.6 for high-resolution pictures and a Biogon 3-inch, f/4.5 lens for simultaneous, nested, low-resolution pictures.

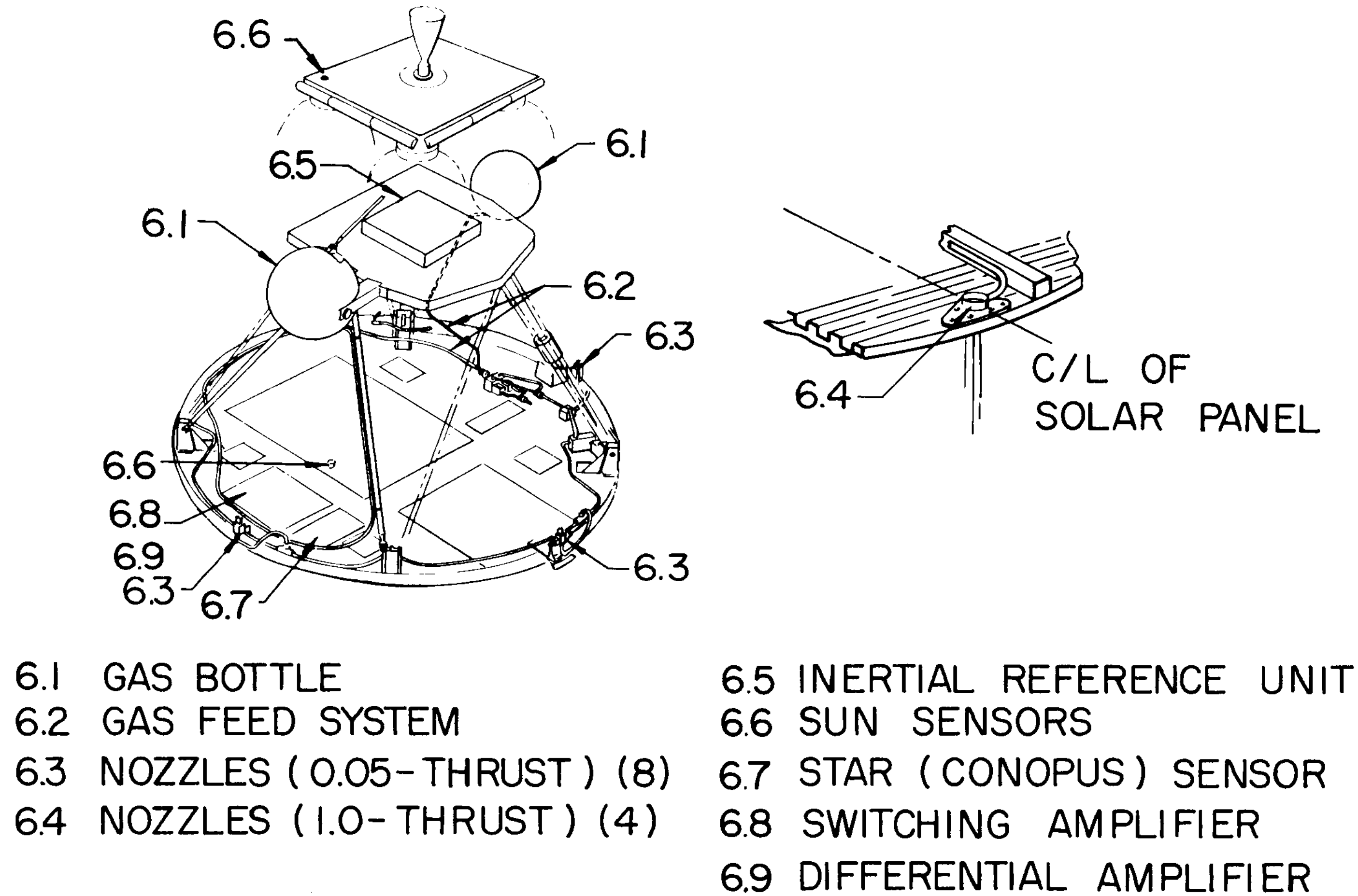

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Figure 16.- Attitude control.

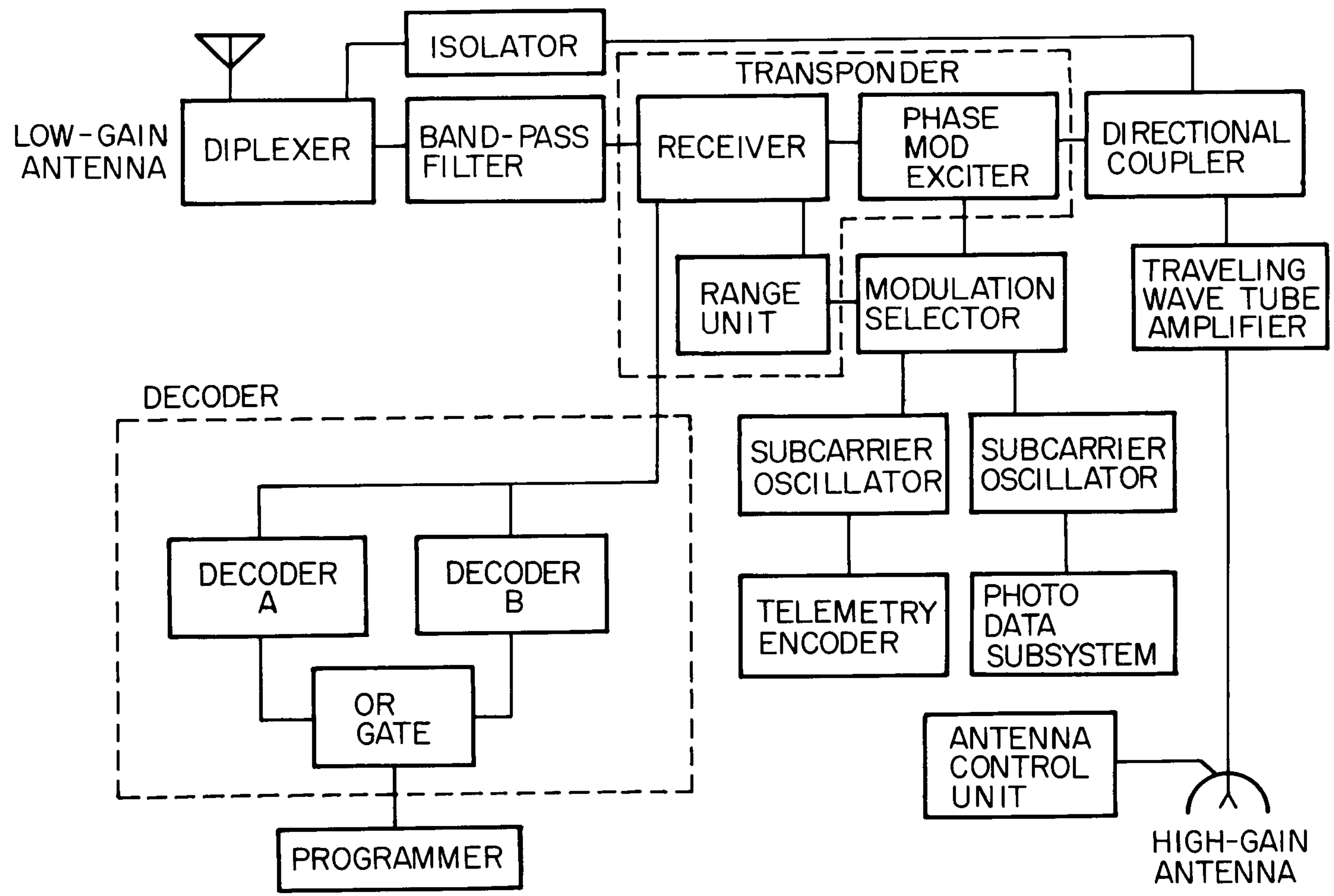

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Figure 17.- Communications subsystem.

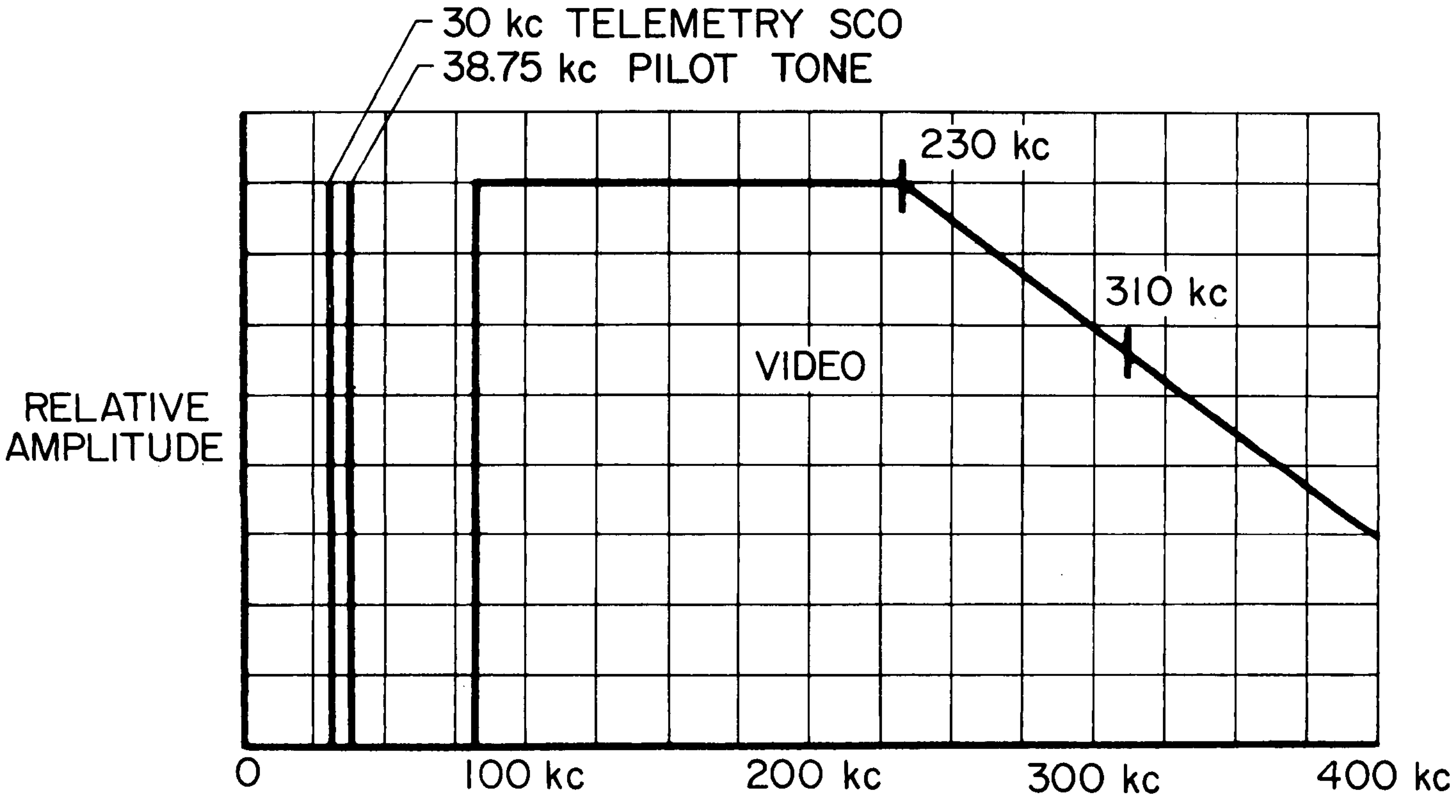

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Figure 18.- Frequency spectrum.

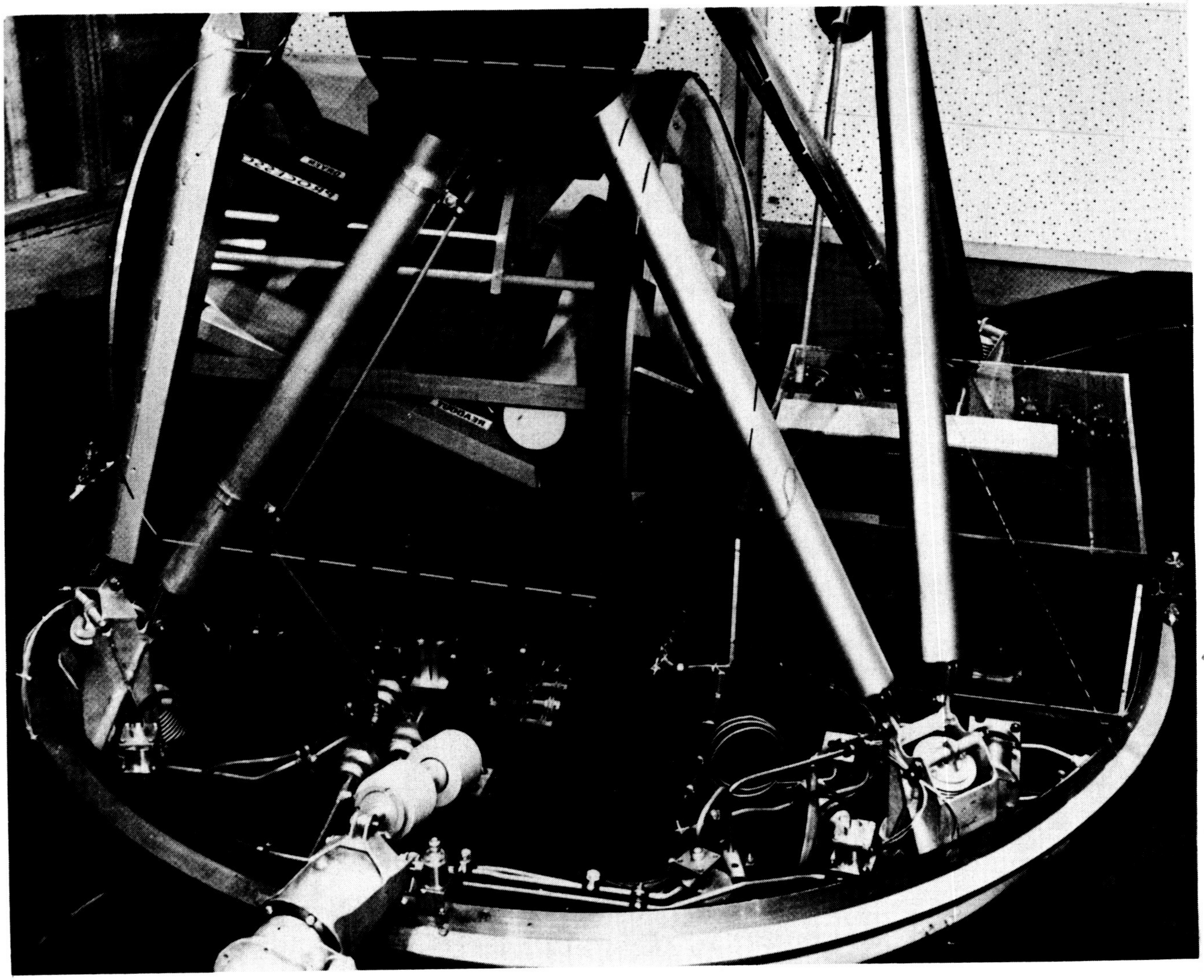

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Figure 19.- Lunar orbiter photo system.

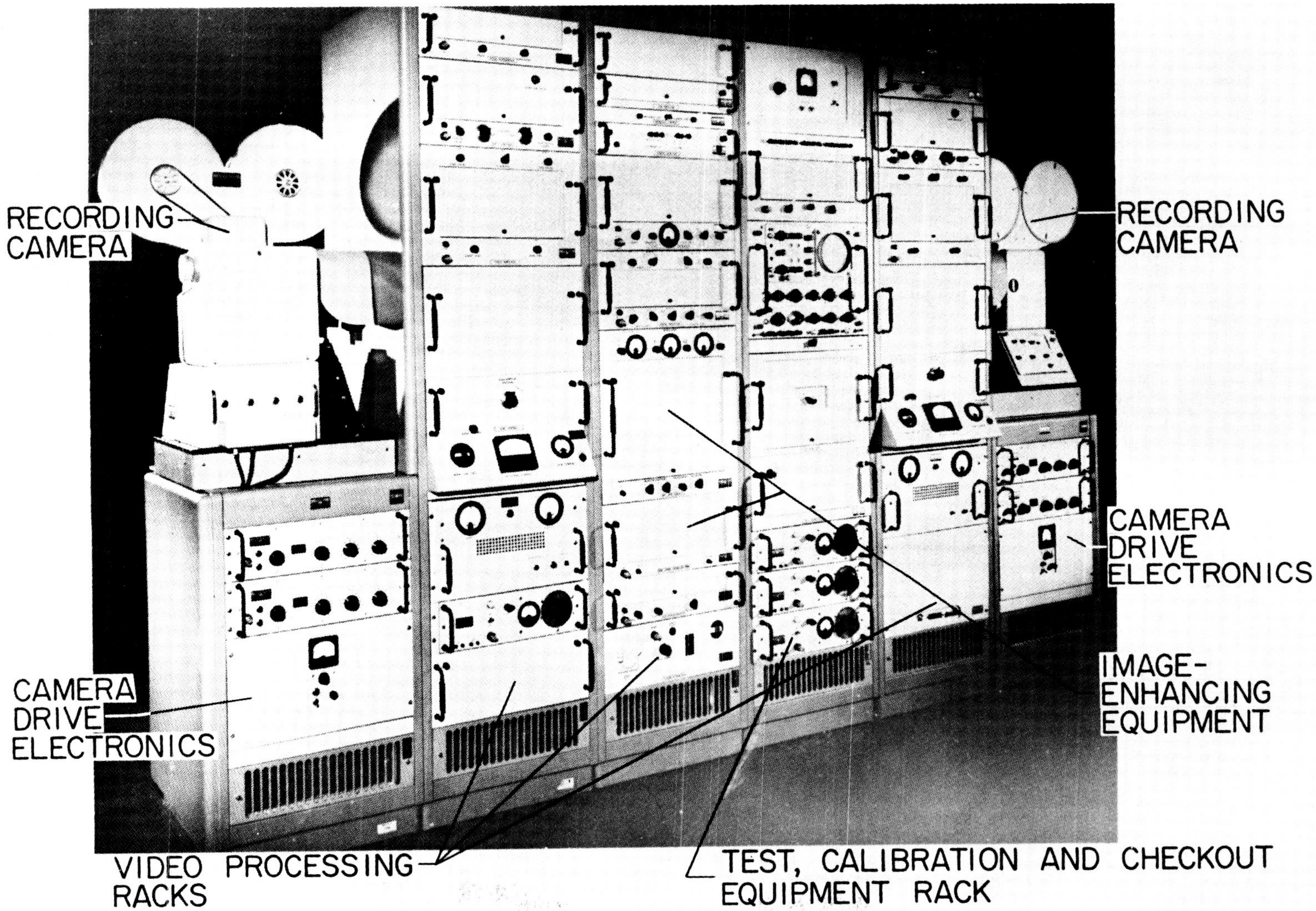

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Figure 20.- Photographic system ground recording equipment.

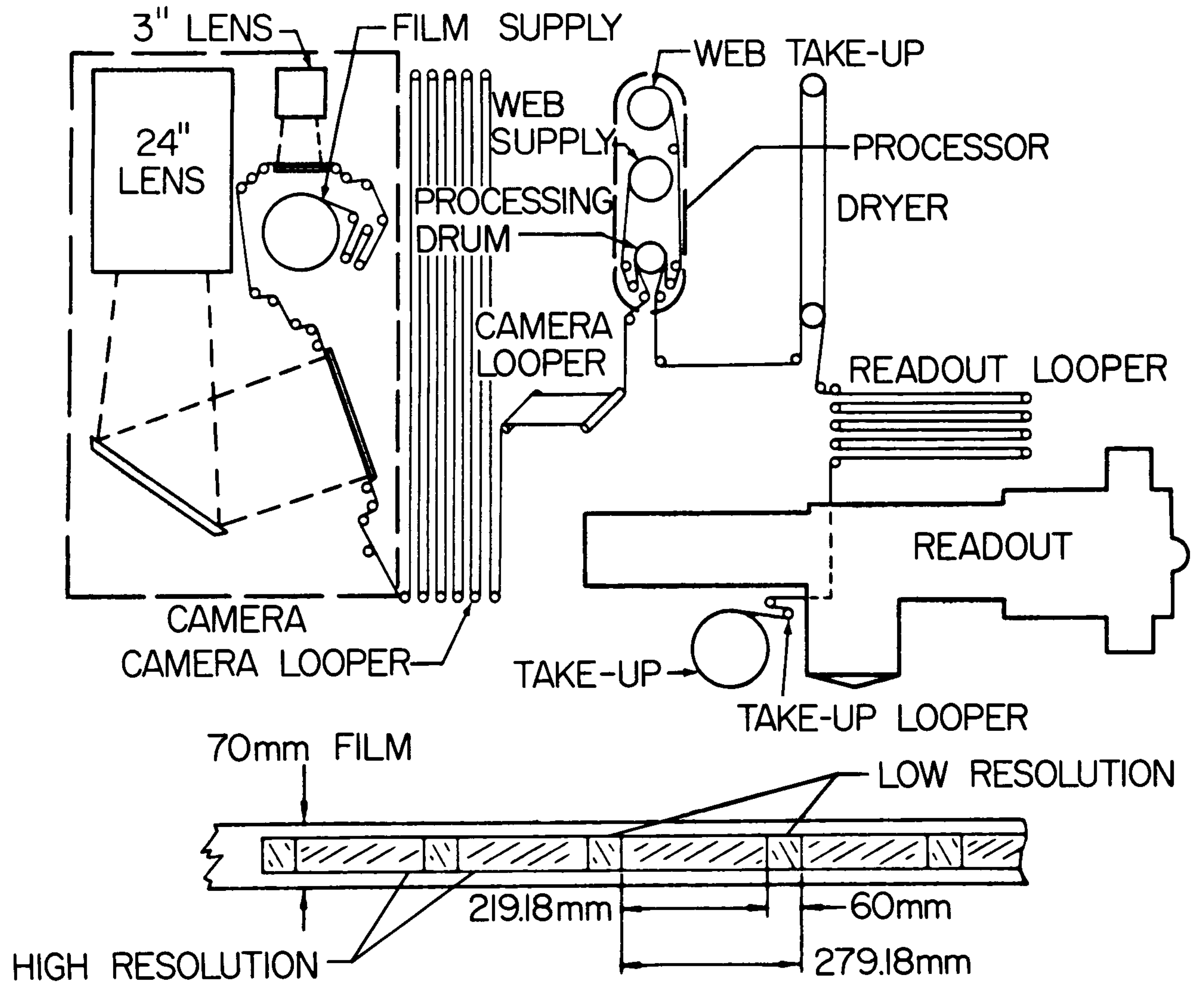

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Figure 21.- Photographic system and film format.

The film used is 70-mm Kodak SO 243 aerial film which is preprinted along one edge with grey scales, resolution bars, and other pertinent information. The 24-inch lens produces a frame approximately 60 by 219 mm (corresponding to 4.5 by 16.4 km on the lunar surface from an altitude of 46 km). The 3-inch lens produces a 60- by 60-mm frame which corresponds to a 36- by 36-km square from the same altitude. After exposure, the film is stored on loopers. The film is then passed through a Bi-mat processor at a speed of about 76 mm/minute. Enough looper storage is provided to store all of the film taken during one orbit; and the processor can process all of the film before the next orbit. The processor is a simple laminator which presses the SO 243 into contact with Kodak SO 111. The film is delaminated after processing, dried and passed through the inactive readout onto a storage spool. At any time film can be read out passing back through the readout onto the readout looper. The capacity of this looper is about four frames. To provide image motion compensation during exposure, the film platens are moved in the flight direction at a speed commanded by a V/H sensor. Transverse image motion compensation is not necessary as the spacecraft yaw angle is commanded to zero in the photographic mode.

SO 243 film is desirable for this mission because it is highly resistant to radiation. It is not significantly affected by the Van Allen belt even without shielding, and with a very small amount of shielding can be subjected to the worst solar flare recorded with fairly minor degradation of quality. Fig. 22 indicates the dose in Rads which a film roll will experience during various major solar flares. It is estimated that the spacecraft structure provides 2 gm/cm^2 of shielding. For the worst case encountered and measured to date the film would have been exposed to 600 Rads over a 15-day period. With minimum shielding this could be reduced to 200 Rads or less. Fig. 23 shows that 200 Rads can be tolerated by the SO 243 film and still provide enough contrast for a reasonably good photograph. Higher speed film such as ASA 80 could withstand a dose of only 3 Rads or so and therefore could not be shielded against a major solar flare without subjecting the spacecraft to prohibitive weight penalties.

Fig. 24 is a schematic of the spacecraft readout system. The light source for film scanning is a CBS line scan tube having its phosphor on a revolving drum. The generated spot of light is demagnified and produces scanning lines on the film of approximately 2.5-mm length. The demagnifying optical system is moved so that successive line scans are displaced until 66 mm is scanned, then the film is advanced 2.5 mm. The next series of line scans occurs in the opposite direction as shown. Collecting optics lead the transmitted light into a photomultiplier. The signal then is conditioned by a video amplifier which makes the signal compatible with the vehicle communications modulator. A separate synch package provides spot sweep voltages to drive the line scan tube and synchronization pulses for the ground equipment.

The ground reconstruction equipment accepts the video signal and displays the video data line by line on a kinescope face. The displayed image is recorded on a continuously moving 35-mm film strip. Each ground equipment has capabilities for two redundant recordings, complete self-checkout, calibration, and provision for a limited amount of image

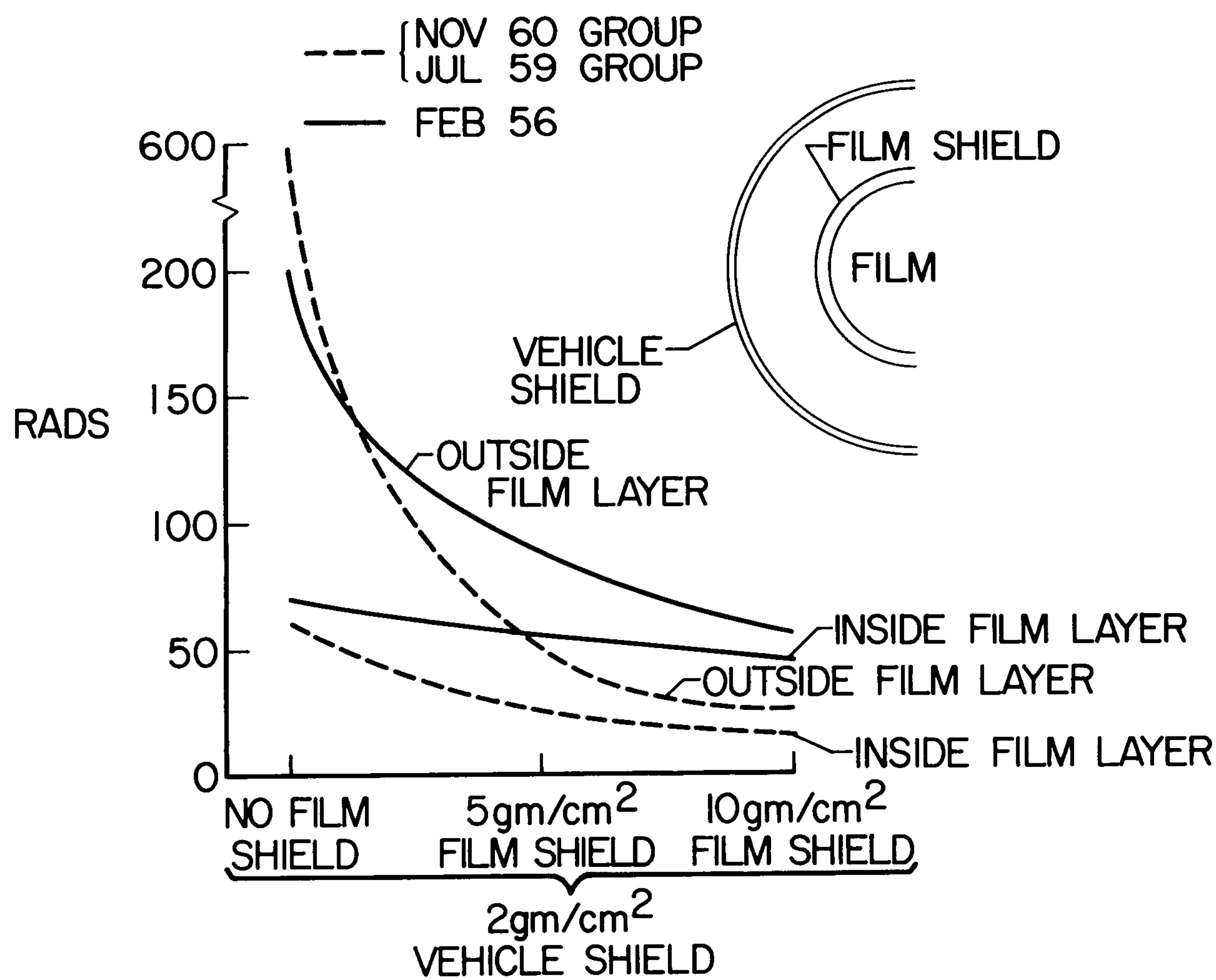

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Figure 22.- Flare dose versus shielding (H_2O).

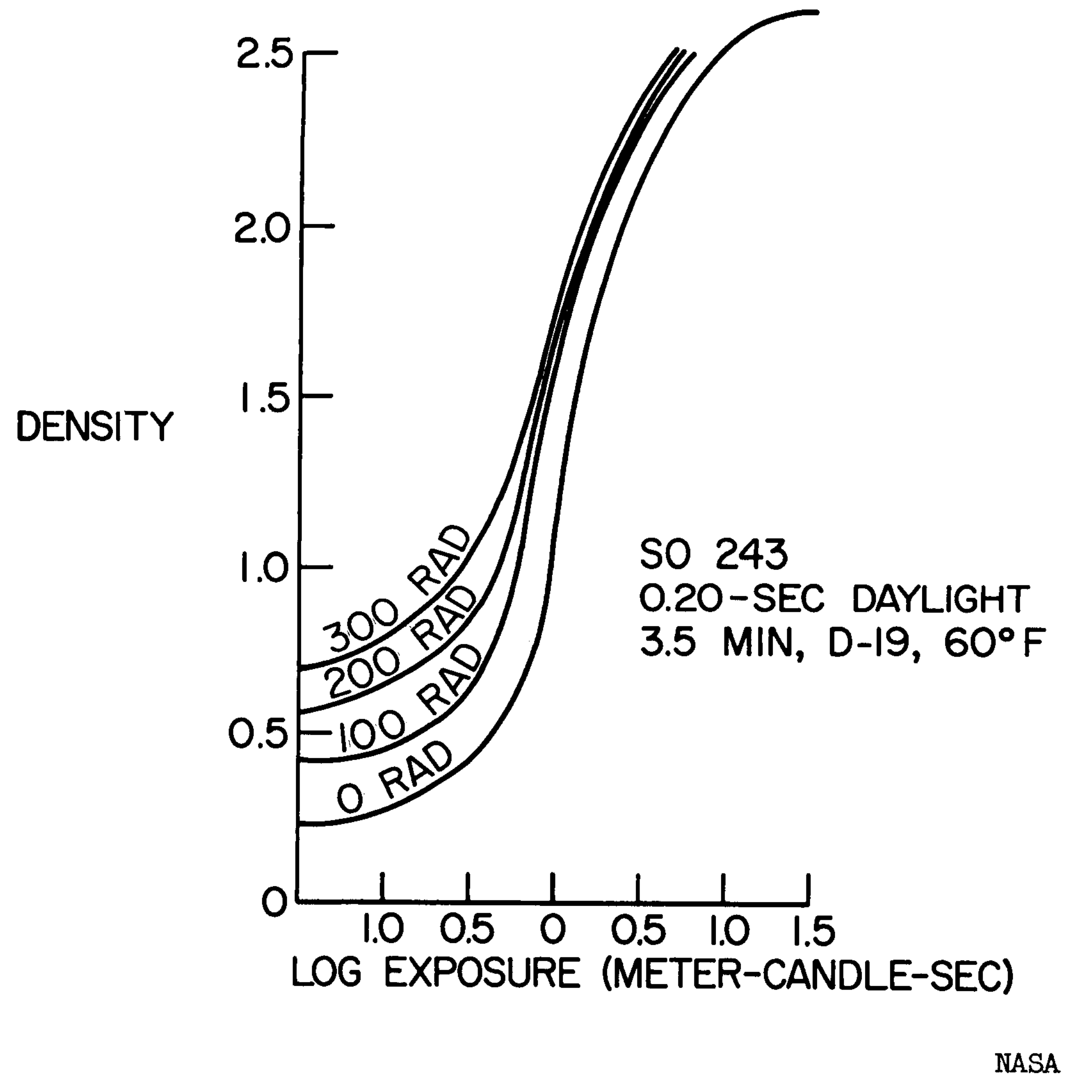

Figure 23.- Radiation effect on SO 243 film.

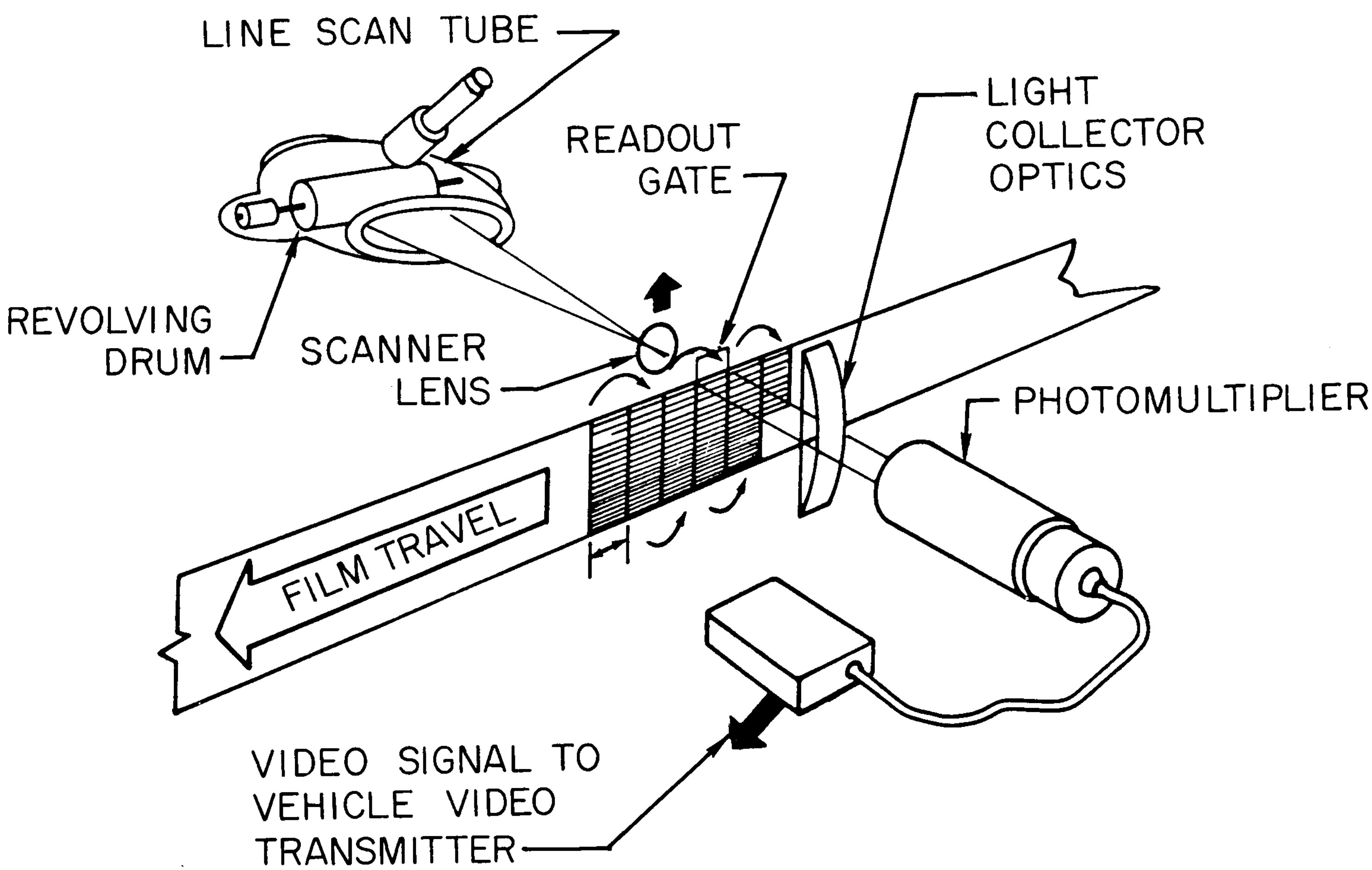

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Figure 24.- Photographic system readout schematic.

enhancement. A number of these equipments are available, as they have been used in an earlier program. Modifications are required in sweep speeds and film-drive speeds to make the equipment suitable for the orbiter program. The GRE equipment will be located at a number of DSN sites. The 35-mm film strips are delivered to a central facility where a reassembly printer is employed to regenerate the basic photographs.

Fig. 25 shows a sample of the lunar surface coverage capability which can be secured with the overall system on any one flight. The diagram is drawn for a lunar orbit inclination of about 20°. Successive orbital passes are identified by orbit number; successive frames are numbered for the low-resolution case. Each low-resolution frame, covering 36 by 36 km is overlapped 50 percent by a successive frame. This permits complete stereo coverage at about 8 meters resolution. Centered in each low-resolution frame is a one-meter resolution frame of 4.5 × 16 km. A total of 200 × 200 km can in this manner be covered in 13 orbits, taking pictures during odd orbits. The 45 × 45 km high-resolution area is shown within the low-resolution coverage. This coverage of 2,000 square kilometers can be secured in 3 orbits, taking 11 frames per orbit.

The coverage shown is only illustrative. It is probable that a number of smaller areas would be covered during only one flight, for example, both the area about a landed Surveyor and an interesting lunar eruption could be covered.

CONCLUSIONS

The Lunar Orbiter is designed to provide timely information regarding the lunar surface and environment. This information should be valuable for Apollo and for general scientific purposes. Because of a high reliance on proven techniques and equipment, there is every reason to believe that this effort will provide the necessary data with a minimum of developmental problems.

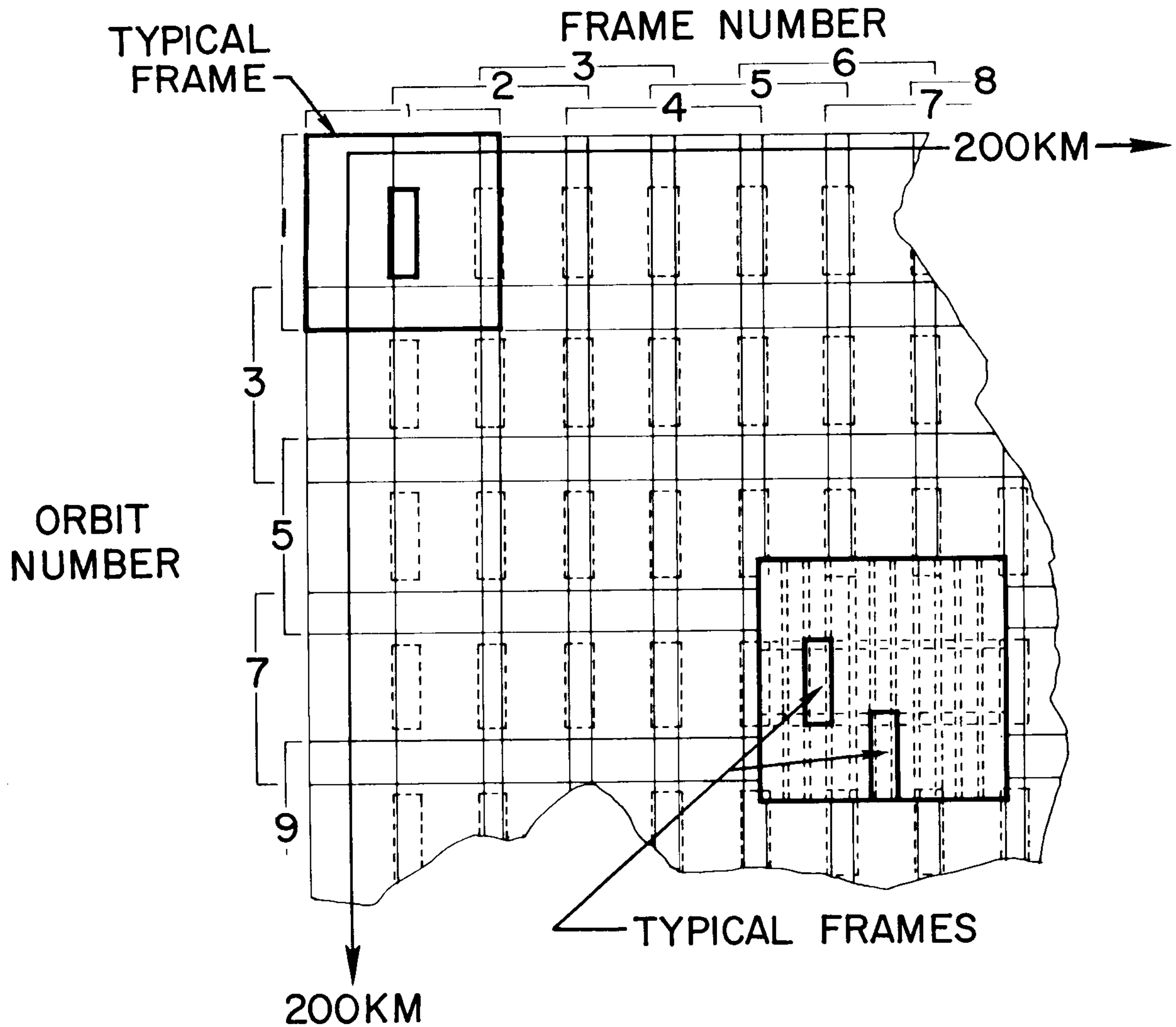

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Figure 25.- Lunar surface photo coverage.

NASA-Langley, 1964