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THE RANGER VII MISSION*

by

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I. INTRODUCTION

The Ranger spacecraft system was conceived in 1959 as a fully attitude-stabilized platform capable of performing a variety of lunar and planetary missions through using alternate payloads mounted on top of the basic spacecraft. In addition, the launch vehicle was required to accommodate a new concept called a parking orbit. The parking orbit technique permits maximum payloads to be injected on the most efficient lunar or planetary trajectory. The technique involves two burns of the second stage of the launch vehicle system. This permits compensation for the nonideal geographical location of the launching pad and provides a more practical daily launch window.

The concept of the fully attitude-stabilized spacecraft provides the following advantages:

1. Maximum solar power generation by pointing the solar panels at the Sun.
2. Maximum communications capability by pointing a high-gain directional antenna at the Earth.
3. Establishment of a spacecraft centered coordinate system for use in midcourse maneuvers to trim trajectory accuracy and for terminal orientation.
4. Permit scientific instruments requiring direction determination and/or control to perform their observations.

Ranger Block I was intended to provide an engineering evaluation of the fully attitude-stabilized system and the parking orbit technique. The mission objectives of Ranger Block I were not achieved because of launch vehicle system problems. Ranger Block II had the primary mission of hard landing an instrumented capsule on the surface of the Moon. This objective was not achieved, but the parking orbit, the attitude stabilization and the midcourse maneuver techniques were successfully demonstrated. Considerable valuable knowledge and experience were obtained from the early Rangers which was applied to the later Mariners and Rangers. The success of Mariner II in its flyby of Venus on December 14, 1962, is a matter of record. Ranger VI and VII were similarly successful in performing all launch vehicle system and spacecraft system functions in a near perfect manner with pinpoint lunar impact accuracy.

II. RANGER BLOCK III

The Ranger Block III mission was initiated in mid-1961 with the objective of obtaining high-resolution photographs of the surface of the Moon. Such photographs were to be of value to the manned lunar landing program and to the scientific space program. The program was a direct extension of Ranger Block II requiring only a change in the spacecraft payload. Several systems using long focal length optics, spacecraft retro slowing techniques, and survivable capsules with photographic systems were studied. The approach which was selected uses fast-sequence shuttered television cameras operating into two high-power transmitters. The two multiple camera channels are used to provide a combination of redundancy, area coverage, and high resolution.

The Ranger Block III was subjected to a thorough review in December 1962. The resulting system modifications required a slip in the launch of Ranger VI to January 1964. Ranger VI appeared to be a complete success up to the last 15 min of its flight. Investigation showed that the television system had inadvertently turned on during the launch and the high voltage power supplies undoubtedly destroyed themselves upon passing through the critical pressure region. The result was that no video was received during the terminal phase of the Ranger VI flight even though turn-on indications were observed.

The launch of Ranger VII was delayed five months in order to provide protection against a repeat of the Ranger VI failure. The modifications involved simplifying and desensitizing control circuitry, interlocking control circuitry during the boost phase, and reducing the sensitivity of the camera subsystem elements to mechanical stresses and vibration. The success of these efforts is clearly demonstrated by the resulting Ranger VII photographs of the Moon.

III. THE LAUNCH PROBLEM

A lunar trajectory can be approximated by a single geocentric conic ellipse whose perigee is nearly equal to the parking-orbit radius and whose apogee is twice the lunar distance from the Earth. This Earth-Moon transfer ellipse can be thought of as rotating with the Earth and contains the effective launch site until liftoff. Thereafter, the ellipse is fixed in geocentric inertial coordinates. Geometrically, it is required that the Moon intersect the transfer ellipse at the required encounter time and this, in turn, determines the time of launch. The plane of the transfer ellipse contains the launch site at launch time and the Moon at encounter time. The geometry in this plane is completed by the direct-ascent trajectory from the actual launch site to the parking orbit and the parking orbit coast arc which allows the injection of the spacecraft to occur near perigee of the transfer ellipse to obtain maximum payload capability. To compensate for the rotation of the Earth, the launch azimuth and parking-orbit coast arc, are adjusted simultaneously to preserve the required geometrical relations. This is illustrated in Fig. 1.

To carry out a launch that meets all these conditions, a multiple-stage launch vehicle (the Atlas D/Agena B) with a radio guidance system is used. The launch site is Cape Kennedy in Florida.

The Atlas D is a one and one-half stage booster that uses liquid oxygen and kerosene as propellants. The Agena B second stage uses hydrazine and nitric acid as propellants. Two burns are required of the Agena B to accelerate to parking-orbit velocity and then, at the end of the parking-orbit coast, accelerate to the required lunar trajectory velocity. The launch-to-injection sequence is shown in Fig. 2. The liftoff weight of the total space vehicle is about 277, 000 lb and the initial thrust is about 370, 000 lb.

Several other constraints on the lunar flight plan must be satisfied. Three of the major ones are:

1. The Earth-Moon geometry must be such that the spacecraft, which uses the Sun and Earth as reference bodies, will have adequate orientation accuracy during its flight to the Moon. Fig. 3 illustrates this and shows that the periods around new and full Moon are unsatisfactory.

2. The lunar lighting conditions in the areas of interest must provide sufficient contrast for good pictures. This restricts the launches to times during first or third quarter, with third quarter being the most satisfactory.
3. The flight time must be such that lunar encounter will occur during the Goldstone Tracking Station view period. This requirement is placed upon the system for operational reasons and because of the cost of placing sensitive maser receivers and video recording equipment at more than one station.

The net result of these constraints and others, such as safety considerations for populated areas and shipping lanes, is that a launch period of about six days exists once a month. The launch window during each of these days is about 2 hr. The resulting trajectories result in flight times of about 65 to 70 hr.

IV. RANGER SPACECRAFT

The Ranger spacecraft consists of the power, telecommunications, guidance and control, propulsion, temperature control and pyrotechnics subsystems plus structure, and the television camera payload. The camera subsystem will be described in the next section. The basic spacecraft configuration is shown in Fig. 4 The design provides a fully attitude-stabilized system using solar panel power and providing high-gain directional communication with the Earth. In the stowed position, the solar panels fold up along the side of the camera tower and the high-gain antenna folds under the spacecraft structure. The basic structure is hexagonal in shape with the solar panels hinging from opposite sides of the hexagon. The antenna is hinged from a corner of the hexagon between the solar panels. The camera aperture is on the opposite side away from the high-gain antenna. The physical dimensions and approximate weight for the spacecraft and its subsystems are shown in Table 1.

Power is supplied by both batteries and solar panels. The batteries are used whenever the solar panels are not oriented at the Sun. The selection is performed automatically by power switching and logic circuitry in the power subsystem. Each solar panel is divided into three electrically isolated segments for protection against internal shorts. The 24.4 sq ft (2.27 sq m) of solar cell area provide over 200 w of raw power. This raw power is regulated to 31.5 v by the booster regulator and distributed to the power conversion electronics for the various subsystems.

The telecommunication subsystem consists of the antenna, radio, command and telemetry subsystems. Two antennas are used. On top the camera subsystem tower is an omnidirectional antenna which receives the signals transmitted from the Earth and which transmits spacecraft data to the Earth whenever the high-gain antenna is not oriented at the Earth. The high-gain antenna is used to transmit both the spacecraft signals and the camera subsystem output.

The radio subsystem contains the receiver for two-way doppler and ground commands and the transmitter for sending spacecraft signals back to the Earth. Phase modulation techniques are used in both the ground commands and the spacecraft telemetry modulation of the transmitter signal.

The command subsystem decodes the subcarrier recovered by the receiver. It provides decoded real-time commands directly to the user and stored command data to the central computer and sequencer in the guidance and control subsystem. The command unit provides eight real-time commands and six stored commands. The stored commands involve 5-bit address block and a 12-bit data block.

The telemetry subsystem provides ten channels of 110 separate measurements. Complete spacecraft engineering telemetry is made available to assess the status and performance of the various spacecraft subsystems. The telemetry subsystem is also used to verify the receipt and action upon both real-time and stored commands. A 15-point telemetry commutator in the camera subsystem modulates an IRIG channel 8 voltage-controlled oscillator, and this signal is included in the telemetry subsystem output.

The guidance and control subsystem consists of the central computer and sequencer and the guidance and attitude control units. The sequencer stores commands inserted prior to launch and by radio command and has a timing system for controlling spacecraft operation in accordance with these commands. The computer provides a velocity increment sensing system to provide midcourse motor shut-off at the prescribed time. Several command sequences are initiated by radio command and then controlled by the central computer and sequencer.

The guidance and attitude control subsystem provides the equipment to permit Sun and Earth acquisition and to attain and maintain specific command attitudes for midcourse and terminal maneuvers. Optical sensors are used to lock onto the Sun and Earth. Small cold gas jets are used to turn the spacecraft in space. Pitch and yaw turns are used to obtain and maintain Sun orientation. The spacecraft attitude control movements are illustrated in Fig. 5 .

The Sun-spacecraft line is defined as the roll axis of the spacecraft. The Earth sensor is located on the high-gain antenna hinge. Earth orientation is achieved after Sun lock by setting the hinge-angle to a nominal value and rolling the spacecraft till the Earth sensor picks up the Earth. Earth lock is maintained by automatically controlling the roll jets and the antenna hinge-angle servo.

The guidance and attitude control system contains an inertial attitude reference system. This system is used to attain and maintain a command attitude relative to the Sun-spacecraft-Earth coordinate system for the midcourse or terminal maneuver.

The propulsion system used in the midcourse maneuver is a mono-propellant system using hydrazine. A small quantity of oxidizer is used to initiate the combustion which is maintained by aluminum oxide pellets acting as a catalyst. The engine can impart a velocity change of from 10 cm/sec to approximately 60 meter/sec to the spacecraft.

Spacecraft temperature control is achieved by passive techniques involving the proper selection of surface finishes and controlling the internal heat transfer. Local temperatures are, therefore, dependent upon solar energy absorption, energy absorption from the Earth, radiation of energy into space, heat from internal power dissipation, and heat transfer to or from other spacecraft components. Extensive telemetry of temperatures throughout the spacecraft permit a check of temperature control success for adjustment on subsequent flights.

The pyrotechnic subsystem consists of the circuitry and control logic for firing the explosive actuators used in deploying the solar panels and controlling the midcourse motor. The solar panel pyrotechnic devices (squibs) are explosive-powered pin pullers. These devices release the solar panels upon command from the central computer and sequencer. The midcourse rocket motor is controlled by a set of normally closed and normally open explosive operated valves. Motor turn-on is accomplished by activating the set of normally closed valves. Shut-off is accomplished by activating the normally open valves. Squib devices are capable of only one activation.

The camera subsystem is mounted on top of the basic hexagonal spacecraft structure. A minimum of interconnection between the two systems was desired. The central computer and sequencer and the radio command unit provide commands for control of the camera subsystem. The camera subsystem provides the combined transmitter outputs from its two channels and the camera telemetry data to the spacecraft telecommunication system. Certain control and monitoring lines are brought out of the camera subsystem for arming purposes and for ground checkout and control. The camera subsystem is described in the next section.

V. TELEVISION CAMERA SUBSYSTEM

The television subsystem basically consists of six cameras, associated control and video circuitry, power system, thermal control system, and transmitters to send the video back to the earth. The assembled system without the thermal shroud is shown in Fig. 6. The TV subsystem mounted on the spacecraft was shown in Fig. 4.

Mission reliability is enhanced by providing as much isolation as possible between the wide-angle full-scan cameras and the narrower-angle partial-scan cameras. The degree of isolation and the functional relationship of the various elements of the subsystem are illustrated in Fig. 7.

Similar optics are used in both camera systems. The two basic lenses are a 25 mm f/1.0 and a 76 mm f/2.0. When used in the full-scan (F) camera system with the 11-mm square format on the vidicon target, the resulting fields of view are 25 and 8.4 deg for the 25 and 76 mm lenses respectively. The f/1.0 and f/2.0 lens speeds coupled with the 5-msec exposure and vidicon sensitive provides a dynamic range of approximately 15 to 2500 ft-lamberts scene brightness.

The video bandwidth constraints permit the 11 mm square format to be scanned in 2.5 sec with 1150 scan lines. The 25 deg field of view full-scan camera is designated the "A" camera and similarly, the 8.4 deg field of view full-scan camera is designated the "B" camera. These two cameras are read out sequentially with a total of 5.12 sec/cycle because of the frame timing and blanking requirements.

The partial-scan (P) cameras have a 2.8 mm square format on the vidicon target. Operating at the same video bandwidth as the full-scan cameras results in a 0.2 sec frame time for the 300 scan lines used. Because of the erase and target preparation time required between exposures, more than two cameras are necessary for continuous coverage. Four partial-scan cameras were decided upon and were designated P1, P2, P3 and P4. P1 and P2 were fitted with 76 mm f/2.0 lenses and P3 and P4 were fitted with 25 mm f/1.0 lenses. The resulting fields of view were 2.1 deg for P1 and P2 and 6.3 deg for P3 and P4. The sequence of operation is P1, P3, P2, and P4 so that photographs are alternately taken by a 76 mm lens and a 25 mm lens. The sequence requires a total of 0.84 sec.

The center of view of the cameras is 38 deg from the roll axis of the spacecraft looking down and away from the spacecraft structure. The individual pointing directions and fields of view are shown in Fig. 8a, b, and c. The fields of view have been adjusted for the loss due to the edge mask on the vidicon target to provide a black level reference in the video signal.

The shutters used are of the solenoid-operated sliding-aperture type and moves from one side of the target to the other each time a picture is taken. The moving blade is located as close to the focal plane as possible. While the full-scan exposure is 5 msec, the P camera exposure was shortened to 2 msec to minimize image motion. The faster sequencing of the P cameras provides photographs from a much lower altitude than the F cameras so that image motion becomes a critical factor in terminal resolution.

The vidicon used in this mission is a specially designed rugged tube produced by the Radio Corporation of America. It employs a special photoconductive surface with excellent sensitivity, high retentivity and good erasure characteristics. The gamma of the vidicon is approximately 0.75. The spectral sensitivity of the tube compared to the human eye is shown in Fig. 9. In operation, the photoconductive surface is prepared for exposure by uniformly charging the surface with the electron scanning beam. Optical exposure modifies this stored charge in accordance with the image induced photoconductive resistance changes. The electron beam then scans the surface using a conventional raster technique. The video signal is dependent upon the discharge which occurred during exposure. When the picture has been read out, the surface must be erased in order to eliminate any residual image which would interfere with the next photograph.

The camera electronics provides for the operation of the cameras, the timing of the sequences, and the amplification and mixing of the video signals. The timing is controlled by a separate camera sequencer for each channel. The video amplification includes some frequency shaping to optimize the sine-wave response of the system. There is a video combiner for each camera group whose function is to enable sequential operation of the cameras into the two transmitters. The video combiner circuitry provides a gate to block the erase video output of the cameras being prepared for exposure. The video combiner output is converted to a frequency-modulated signal for modulation of the appropriate transmitter.

The two frequency-modulated transmitters are frequency centered ±0.53 mc on either side of the spacecraft transmitter frequency. The transmitter chains provide a basic output of 60 w per channel. This output must be mixed to provide a single signal line to the spacecraft high-gain directional antenna. The mixing of the two 60 w outputs is accomplished in a four-port hybrid mixer where half the power is dissipated in a dummy load and the other half is sent to the spacecraft. In the spacecraft, the spacecraft transmitter output and the camera subsystem transmitters output are combined in a directional coupler.

The transmitter subsystem also includes 15- and 90-point telemetry processors. The 15-point system is turned on prior to launch and operates to impact. Prior to the camera subsystem turn-on, the 15-point telemetry data is transmitted by the spacecraft system. The points monitored provide information on battery voltages and currents, various camera subsystem temperatures, and the operation of a backup turn-on clock. When the camera subsystem turns on, the more detailed 90-point telemetry processor turns on and provides a variety of engineering measurements within the camera subsystem.

The camera subsystem control circuitry incorporates several methods of placing each channel into warm-up and then into full-power operation. The cameras may also be turned off rapidly in the event of an inadvertent turn-on. After the difficulties with Ranger VI, several safeguards were built in to prevent the turn-on of the system until after spacecraft separation from the Agena B. The command and control system consists of a command control unit, a distribution control unit, an electronic backup clock and control circuits within the high-current regulators, the camera sequencers, and the transmitters.

The camera subsystem may be placed into warm-up by relay closures in the spacecraft Central Computer and Sequencer or Radio Command Unit. The F camera channel can also be put into warm-up by a transistor switch activation in the backup clock. The warm-up mode is initiated when any one of the indicated commands activates the relay controlling the trigger of a silicon-control rectifier which applies battery voltage to the high-current regulator.

The camera subsystem is placed into full-power operation 80 sec after warm-up when the sequencer operates a relay controlling the high voltages to the transmitter. Should the sequencer fail to operate the relay, the spacecraft Central Computer and Sequencer performs a similar operation after a 5-min period.

The turn-off of the camera subsystem is accomplished by a radio commanded relay closure which activates circuitry to turnoff the silicon control rectifiers supplying battery voltage to the high-current regulators.

The power system for the camera subsystem consists principally of two batteries, two high-current regulators, and one low-current regulator. Batteries were selected because of the very short high-power drain required to perform the mission. The batteries are silver-zinc composition with 22 wet electrolytic cells. The batteries are sealed and ruggedized and can deliver 40 amp-hr at 34 v. The flight batteries are installed just prior to mating the spacecraft to the launch vehicle at the Eastern Test Range.

Thermal control of the camera subsystem is accomplished through a passive thermal balance technique. A polished aluminum thermal shroud is placed over the entire camera subsystem. The amount of energy received by the system from the Sun is controlled by the painted surfaces of fins attached to the shroud. Heat sinks were added in high dissipation areas to permit operation for in excess of mission requirements.

TELEVISION RECEIVING AND RECORDING EQUIPMENT

The television signals from the spacecraft are received, amplified, and separated in the ground receiver system. The signal frequency variations are then converted back to amplitude variations in two demodulators (one for each camera channel), whose outputs are the same as the video signals originally generated in the cameras. The video signals are used to control the intensity of an electron beam in a cathode-ray tube, which is scanned in unison with the electron beam in the cameras. The cathode-ray tube reconstructs the original image, which is then photographed on 35-mm film. These recording devices are similar to the commercial kinescopes used for recording television programs on film. Again, there is one recording device for each camera channel, so that two pictures are being recorded at any instant in time, one F camera and one P camera. All the functions discussed above are duplicated at both receiving sites, with one exception. One site utilizes a single film recorder to record the four P cameras, while the other site maintains two film recorders and records both camera channels.

In addition to the film recorders, another means of recording the data is used. The signals are recorded on magnetic tape at both sites. Two such recorders are used at each receiving station. In order to obtain film records from the magnetic tapes, they are played through a demodulator, and the video signal is applied to the film recorder as discussed above.

VI. THE SPACE FLIGHT OPERATIONS GROUND COMPLEX

A sizeable world-wide ground complex of men and equipment is required to carry out the space flight operations for a Ranger mission. The three main equipments, all portions of NASA's Deep Space Network, are:

1. The Deep Space Instrumentation Facility (DSIF) consisting of transmitting and receiving stations located at Goldstone, California; Woomera, Australia; and Johannesburg, South Africa. These stations, each with an 85-ft-diam antenna, provide continuous tracking and telemetry coverage of the spacecraft and have the ability to send commands to the spacecraft. In addition, a spacecraft monitoring station is located at Cape Kennedy, and a mobile tracking station is located in the Johannesburg area. The characteristics of these stations are shown in Table 2. Figure 10 shows the Goldstone station.
2. The Space Flight Operations Facility (SFOF) is located at the Jet Propulsion Laboratory in Pasadena, California. This facility houses the data processing equipment required for the telemetry and tracking data and the communications terminal equipment for communications to the receiver sites. The processed data is transmitted to the DSIF stations and to the people who analyze the data and determine the actions to be taken during the course of the flight.
3. A world-wide communications network is used to communicate between the DSIF stations and the SFOF. These are teletype and voice circuits that are used for both data and instruction transmissions. A combination of landline, microwave, submarine cable, and high-frequency radio communication is utilized as connecting links between the various locations.

In addition to the many highly trained persons required to operate the vast complex of equipments, three highly trained and specialized technical groups are necessary to conduct a space flight mission.

1. <u>The Spacecraft Data Analysis Team</u>

This team is composed of engineers who know in detail the various parts of the spacecraft and who have trained for many months during the preflight testing to read and interpret the telemetry. They monitor the performance of the spacecraft continuously during the flight and recommend appropriate actions to ensure top performance of the spacecraft.

2. <u>The Flight Path Analysis and Command Group</u>

This group is composed of trajectory engineers, tracking data analysts, and data processing specialists who use the doppler and angle data from the tracking stations to determine the flight path of the spacecraft. They also determine the required midcourse maneuver to correct the trajectory to the selected impact area. In addition, from the trajectory and other pertinent data, they determine the spacecraft attitude for use during the midcourse and terminal phases of the flight.

3. <u>The Space Science Analysis Group</u>

This group is composed of trained scientists and engineers who work with the experiment team to determine scientific trade-offs for various impact locations and terminal maneuvers and recommend specific courses of action to maximize the scientific output.

All of these many elements of the Space Flight Operations Complex are under the direction of the Space Flight Operations Director who conducts and directs the mission according to a thoroughly tested space flight operations plan.

VII. THE RANGER VII FLIGHT

Ranger VII was launched at 16:50:08 GMT on July 28, 1964. The countdown proceeded without any unscheduled holds, and the launch occurred 8 sec after the launch window opened. All launch vehicle system events occurred in a normal sequence, and the spacecraft was injected on its Earth-Moon transfer orbit at 17:20 GMT. After separation from the Agena B, the spacecraft performed a normal Sun- and then Earth-acquisition sequence. The injection accuracies were sufficient to ensure an impact on the far side of the Moon. A typical Ranger flight profile is shown in Fig. 11 .

In order to impact the desired target area of 11 deg South and 21 deg West selenocentric coordinates, it was necessary to perform a midcourse maneuver. The maneuver required a +5.56-deg roll turn and a -86.8-deg pitch turn. The midcourse motor was then ignited for a 50-sec burn to achieve a 29.89-m/sec velocity increment. The midcourse maneuver was successfully completed with Sun and Earth reacquisition at 10:58 GMT on July 29, 1964. A nominal midcourse maneuver sequence is shown in Fig.12.

The post-midcourse maneuver tracking indicated that the spacecraft would impact the Moon at about 10.7 deg South and 20.7 deg West selenocentric coordinates. The midcourse maneuver was required to adjust the time of arrival, as well as impact location. Calculations indicated that the time of arrival would permit nominal use of the camera subsystem backup clock to turn on the F cameras. The backup clock was set prior to launch to turn the F cameras into warm-up at 67 hr, 45 min after spacecraft Agena B separation. The actual turn-on occurred at 67 hr, 46 min, 8 sec after separation.

The terminal approach geometry was analyzed, and it was determined that no terminal maneuver orientation change was required. The factors considered were the angle of illumination of the lunar surface, the direction of the spacecraft velocity vector, and the cameras pointing directions and area coverage. A zero maneuver terminal sequence was used in order to initiate the central computer and sequencer terminal sequence counter. The central computer and sequencer provides both warm-up and full-power commands to the camera subsystem timed from initiation of the sequence. Typical approach geometry is shown in Fig. 13.

The terminal sequence was initiated by radio command at 12:25:08 GMT on July 31, 1964. The camera subsystem backup clock turned the F channel into warm-up at 13:07:15 GMT. Full power was achieved automatically by the F channel sequencer at 13:08:36 GMT after the nominal 80-sec warm-up period. The P channel was turned into warm-up by the central computer and sequencer at 13:10:49 GMT, and full power was achieved after the 80-sec warm-up at 13:13:09 GMT. The camera subsystems operated normally to impact at 13:25:49 GMT and provided approximately 4300 excellent photographs of the Moon.

VIII. RANGER VII MISSION RESULTS

The Ranger VII mission was performed so flawlessly that it has been described as a "textbook mission". The technological feat demonstrated by Ranger VII marks a significant milestone in mankind's exploration of the Moon. The tracking accuracy for Ranger VI and VII permitted refinement of the determination of the mass of both the Earth and the Moon. Figures 14 and 15 show the correlation between the independently derived values from each spacecraft mission. Combining the results of Ranger VI and VII will permit further reduction of the uncertainties.

The 4300 pictures taken by Ranger VII are obviously the most significant feat of the mission. System performance resulted in excellent quality photographs from all six cameras. The last photograph was taken by the P_3 camera at an altitude of approximately 500 m. The photograph was only partially read out before impact and showed an area of 30 by 50 m. Since the resolution of the P cameras is about 100 optical pairs, a surface resolution of approximately 0.5 m is achieved. Cratering phenomena can be observed down to the limit of detectability in the photograph.

Future Ranger missions will sample other mare areas. The objectives of such samplings will be to provide more information for the unmanned and manned lunar landing missions and to increase our scientific knowledge of the Moon. While the first step by Ranger VII has been significant, it will require many more, and far more, complicated steps before the basic questions concerning the Moon can be answered.

Table 1. Ranger spacecraft

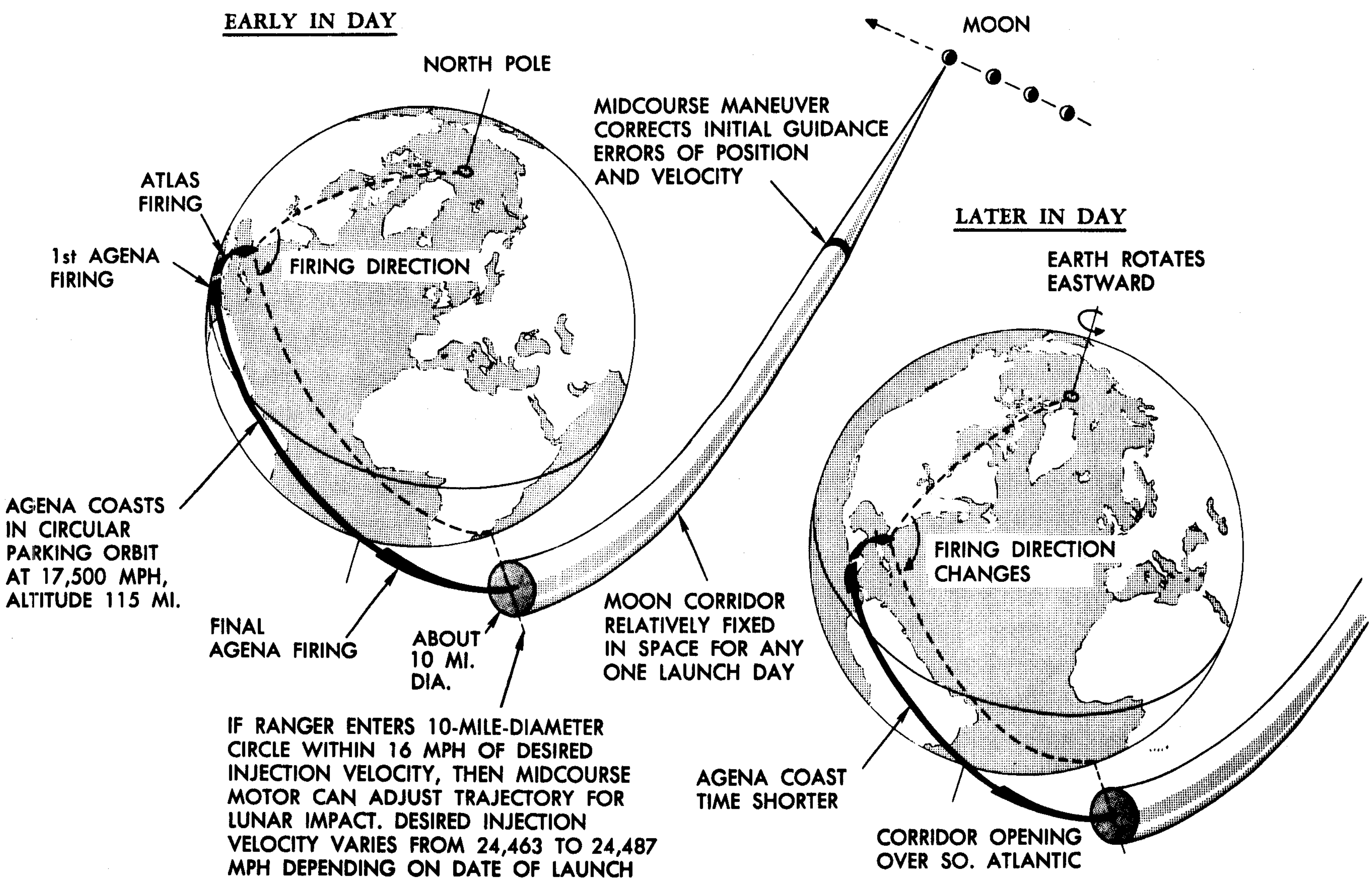
Dimensions	feet
In launch position, folded	
Diameter	5
Height	8.25
In cruise position, panels unfolded	
Span	15
Height	10.25

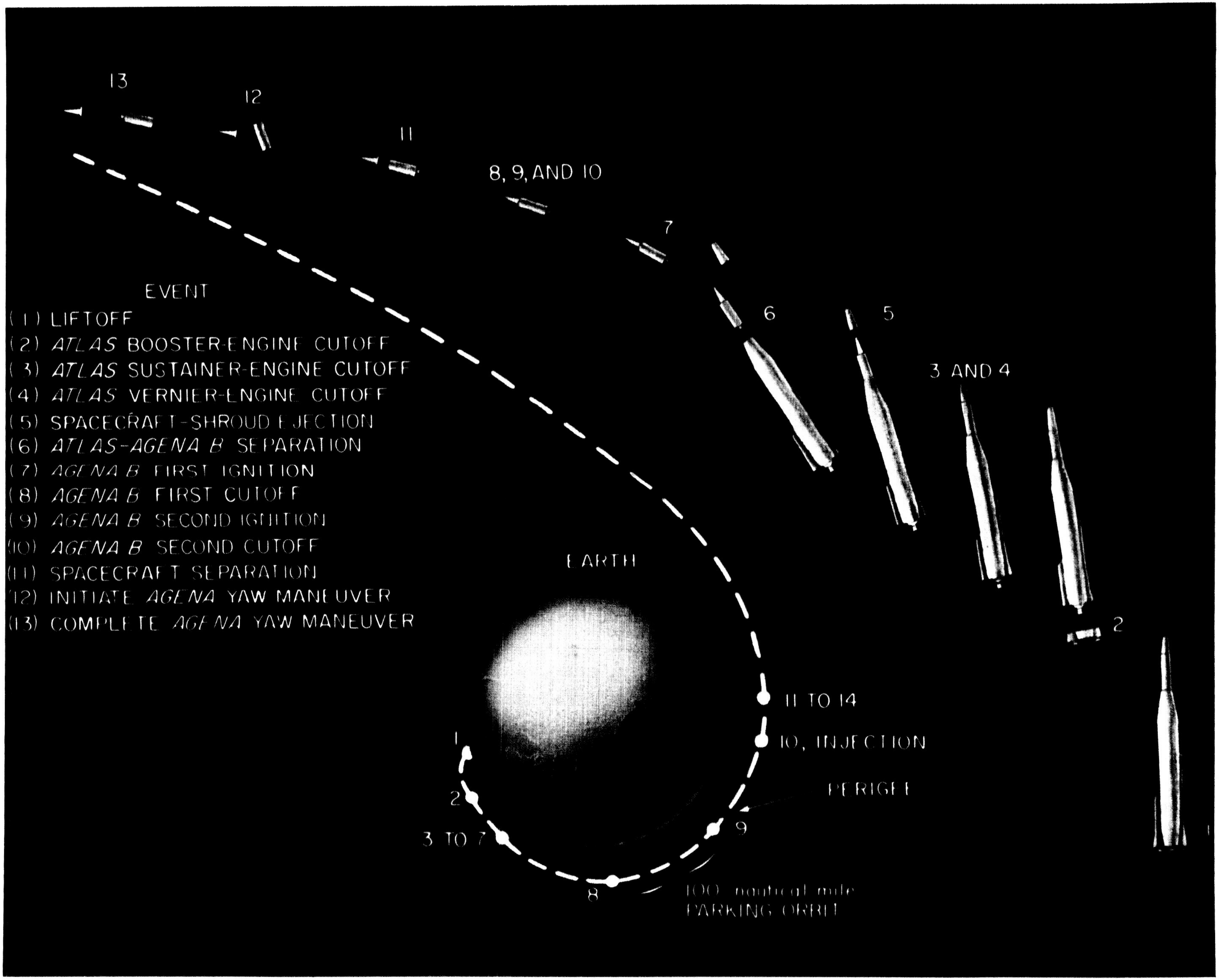
Approximate Weight	pounds
Structure	90
Solar panels	47
Electronics	154
Propulsion	45
Launch back-up battery	51
Miscellaneous equipment	37
Ranger bus total	424
TV subsystem total	382
Gross weight	806

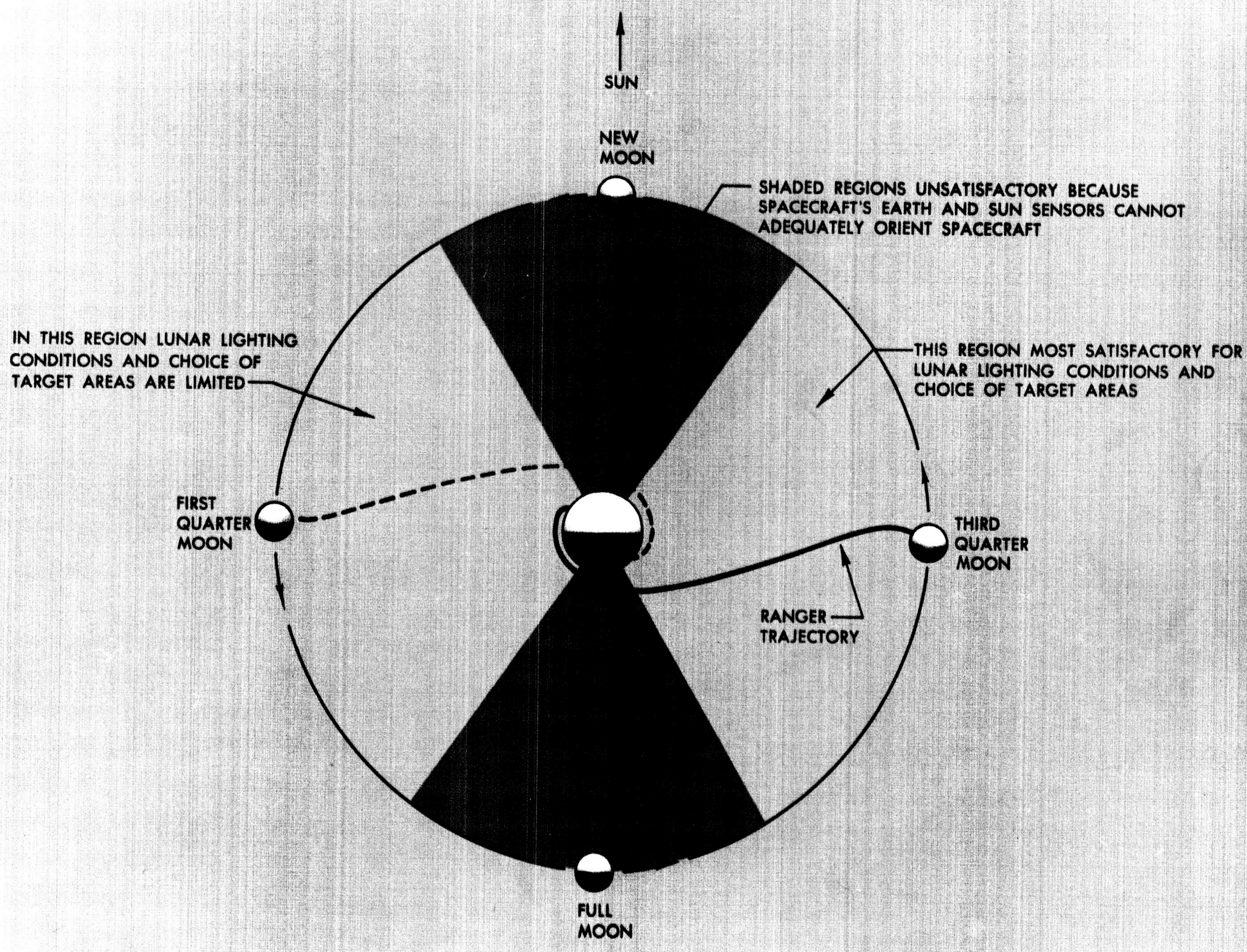
Table 2. Capabilities and characteristics

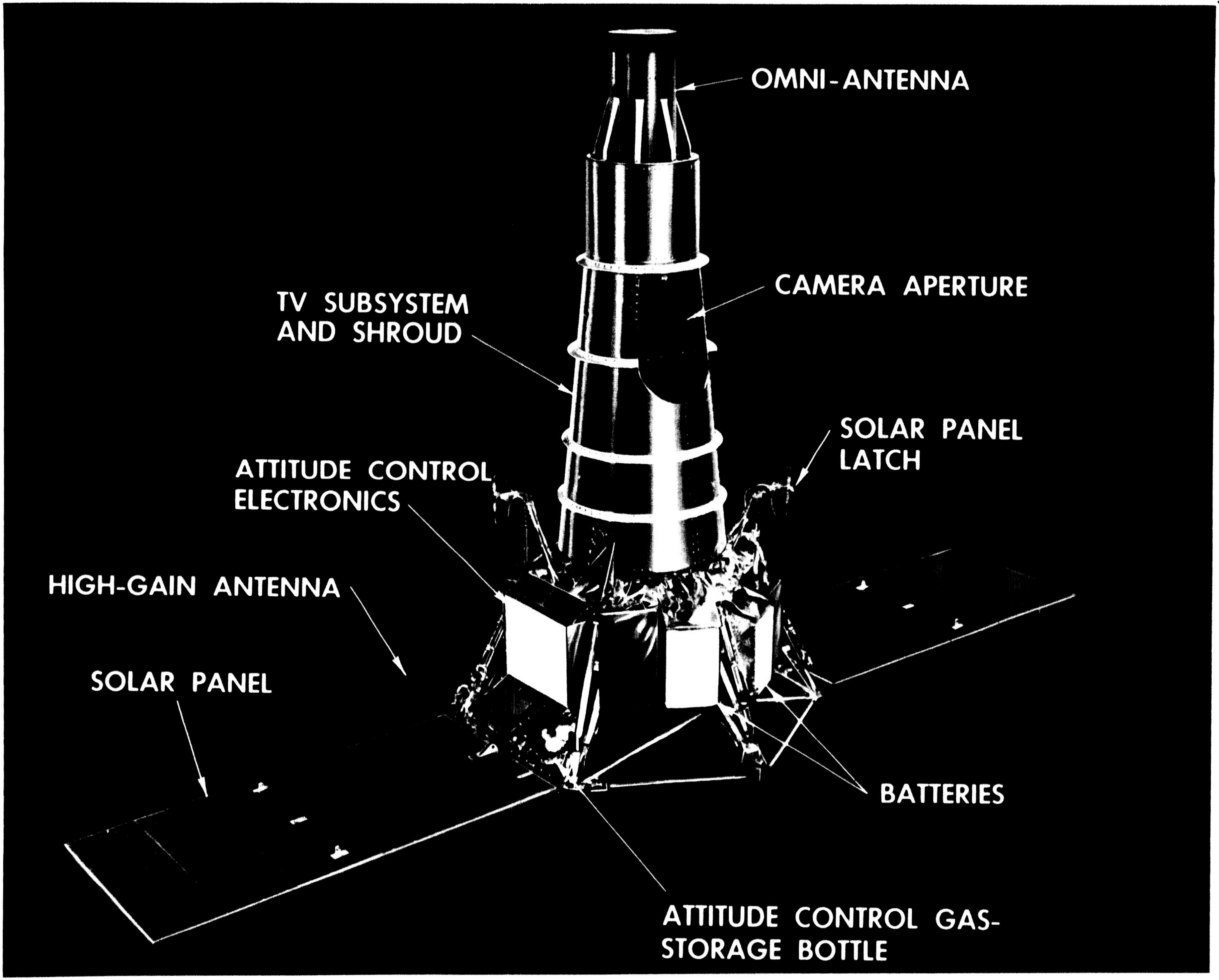
Characteristic	Spacecraft Monitoring Station	Mobile Tracking Station	Goldstone Pioneer Station	Goldstone Echo Station	Woomera Station, Australia	Johannesburg Station, South Africa
Antenna size	6-ft (Az-El) (no angle data)	10-ft (Az-El)	85-ft Polar (HA-Dec)	85-ft Polar (HA-Dec)	85-ft Polar (HA-Dec)	85-ft Polar (HA-Dec)
Maximum angular rate	Manually operated	20 deg/sec in both axes	0.7 deg/sec in both axes	0.7 deg/sec in both axes	0.7 deg/sec in both axes	0.7 deg/sec in both axes
Antenna gain (960 Mc)						
Tracking feed	---	23.5±0.2 db	---	---	43.7±0.9 db	43.7±0.9 db
Horn feed	20.5 db	---	45.7±0.8 db	45.7±0.8 db	---	---
Transmitter power	---	25 w	---	200 w (50-w backup)	200 w	200 w
Data transmission						
Angles-doppler	---	Near-real time	Near-real time†	Near-real time†	Near-real time	Near-real time
Telemetry	Real time*	None	Record only	Near-real time Real time*	Near-real time Real time*	Near-real time Real time*
Decommutated telemetry	No	No	No	Yes	Yes	Yes
Command capability	No	No	No	Yes	Yes	Yes

*Sent to the Telemetry Processing Station (TPS) via wide-band telephone line.
†Angle data not the result of autotrack operation.


EARLY IN DAY
NORTH POLE
ATLAS FIRING
1st AGENA FIRING
FIRING DIRECTION
AGENA COASTS IN CIRCULAR PARKING ORBIT AT 17,500 MPH, ALTITUDE 115 MI.
FINAL AGENA FIRING
ABOUT 10 MI. DIA.
IF RANGER ENTERS 10-MILE-DIAMETER CIRCLE WITHIN 16 MPH OF DESIRED INJECTION VELOCITY, THEN MIDCOURSE MOTOR CAN ADJUST TRAJECTORY FOR LUNAR IMPACT. DESIRED INJECTION VELOCITY VARIES FROM 24,463 TO 24,487 MPH DEPENDING ON DATE OF LAUNCH
MIDCOURSE MANEUVER CORRECTS INITIAL GUIDANCE ERRORS OF POSITION AND VELOCITY
MOON
MOON CORRIDOR RELATIVELY FIXED IN SPACE FOR ANY ONE LAUNCH DAY
LATER IN DAY
EARTH ROTATES EASTWARD
FIRING DIRECTION CHANGES
AGENA COAST TIME SHORTER
CORRIDOR OPENING OVER SO. ATLANTIC


13
12
11
8, 9, AND 10
7
6
5
3 AND 4
2
1
EVENT
(1) LIFTOFF
(2) ATLAS BOOSTER-ENGINE CUTOFF
(3) ATLAS SUSTAINER-ENGINE CUTOFF
(4) ATLAS VERNIER-ENGINE CUTOFF
(5) SPACECRAFT-SHROUD EJECTION
(6) ATLAS-AGENA B SEPARATION
(7) AGENA B FIRST IGNITION
(8) AGENA B FIRST CUTOFF
(9) AGENA B SECOND IGNITION
(10) AGENA B SECOND CUTOFF
(11) SPACECRAFT SEPARATION
(12) INITIATE AGENA YAW MANEUVER
(13) COMPLETE AGENA YAW MANEUVER
EARTH
11 TO 14
10, INJECTION
PERIGEE
9
8
3 TO 7
2
1
100 nautical mile
PARKING ORBIT


SUN
NEW MOON
SHADED REGIONS UNSATISFACTORY BECAUSE SPACECRAFT'S EARTH AND SUN SENSORS CANNOT ADEQUATELY ORIENT SPACECRAFT
IN THIS REGION LUNAR LIGHTING CONDITIONS AND CHOICE OF TARGET AREAS ARE LIMITED
THIS REGION MOST SATISFACTORY FOR LUNAR LIGHTING CONDITIONS AND CHOICE OF TARGET AREAS
FIRST QUARTER MOON
THIRD QUARTER MOON
RANGER TRAJECTORY
FULL MOON


OMNI-ANTENNA
CAMERA APERTURE
TV SUBSYSTEM AND SHROUD
SOLAR PANEL LATCH
ATTITUDE CONTROL ELECTRONICS
HIGH-GAIN ANTENNA
SOLAR PANEL
BATTERIES
ATTITUDE CONTROL GAS-STORAGE BOTTLE

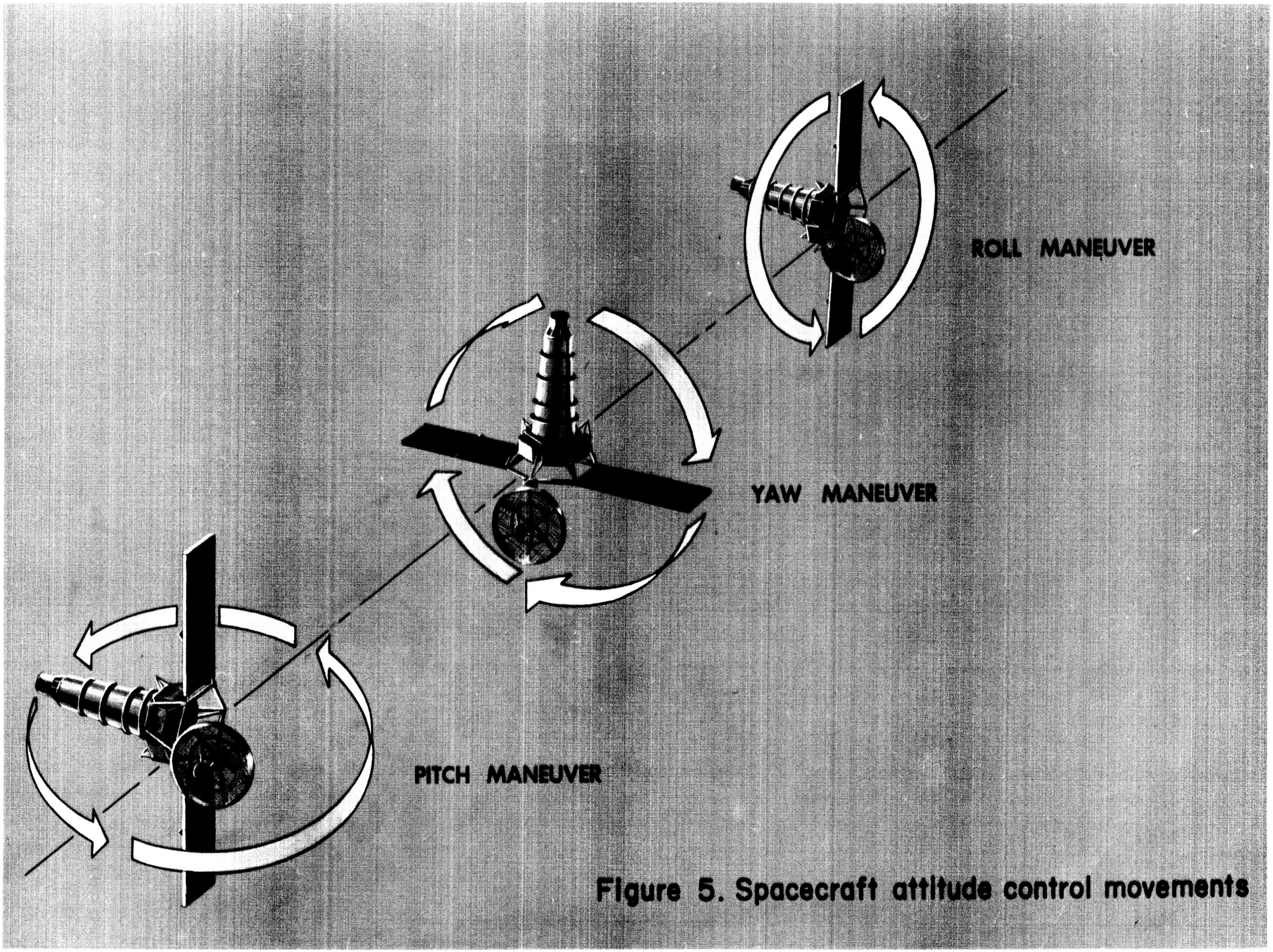
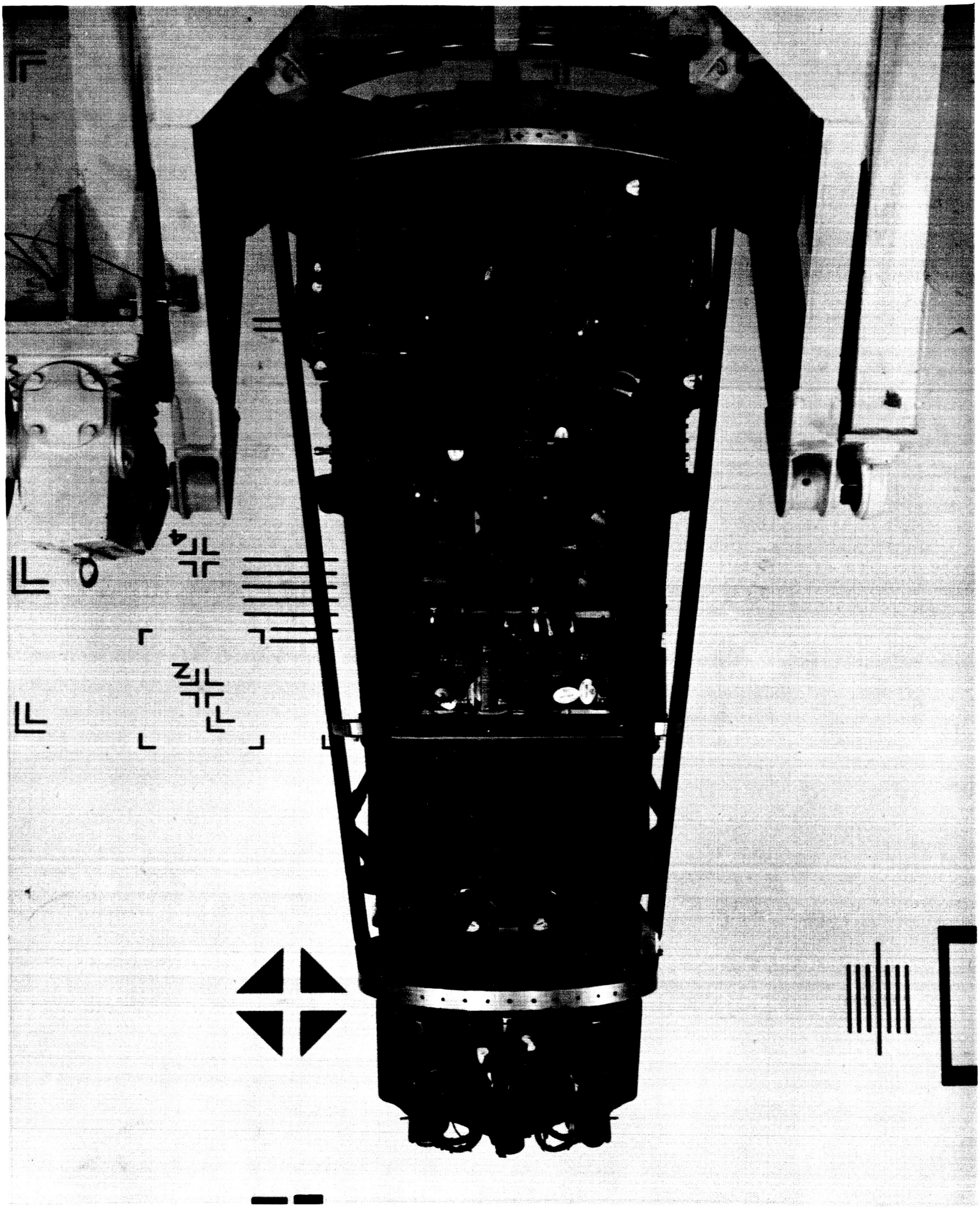
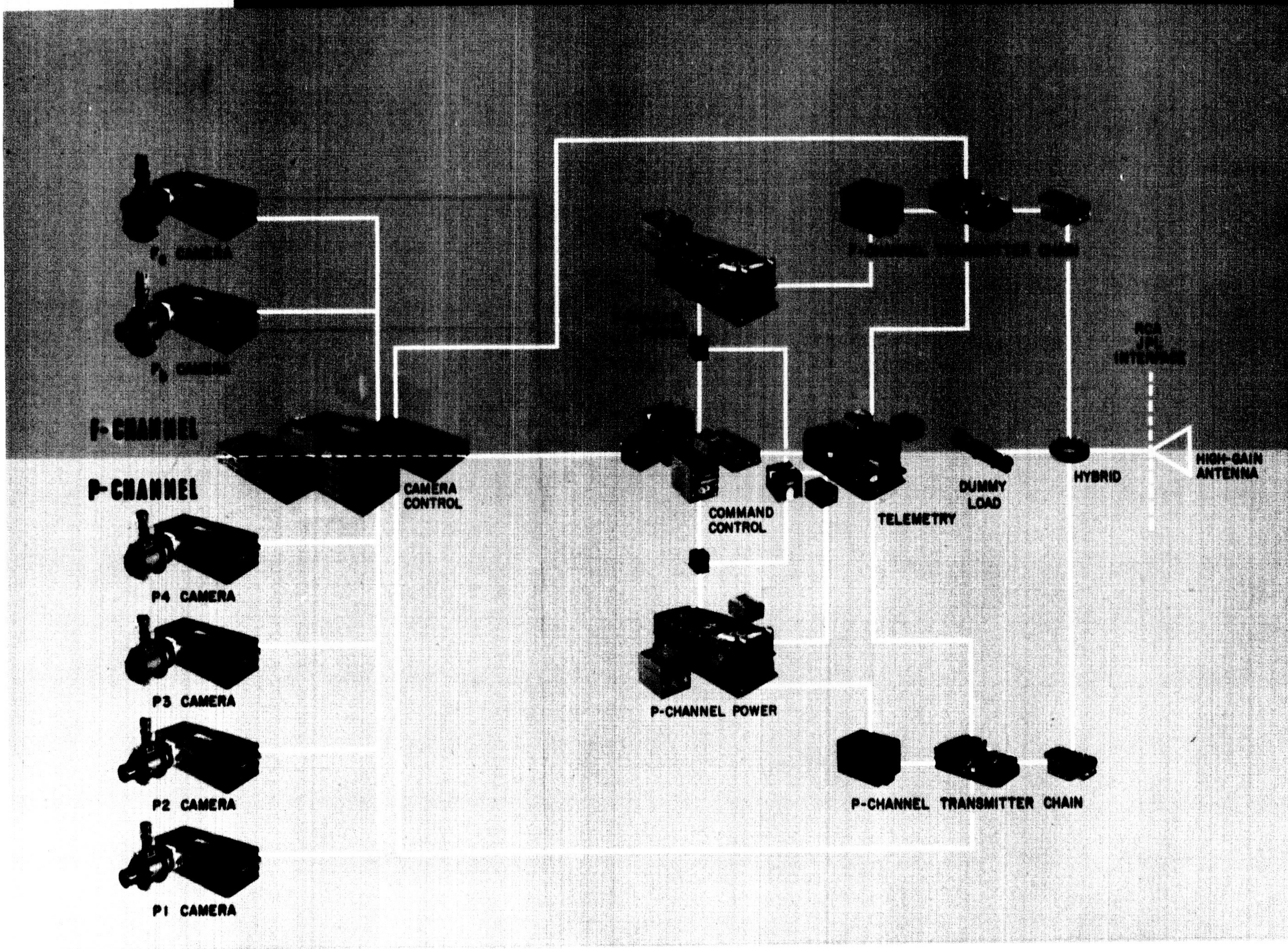
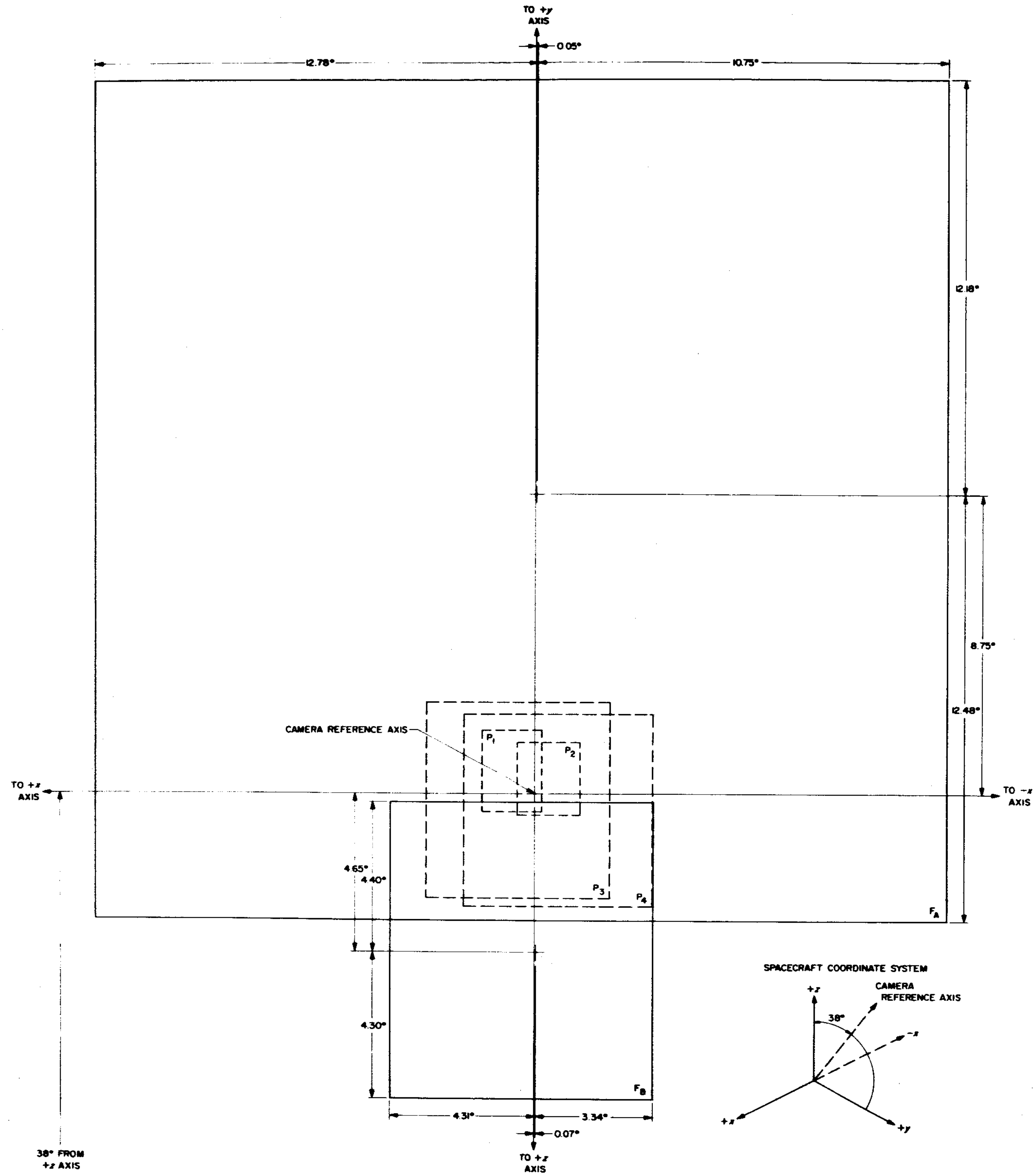


Figure 5. Spacecraft attitude control movements

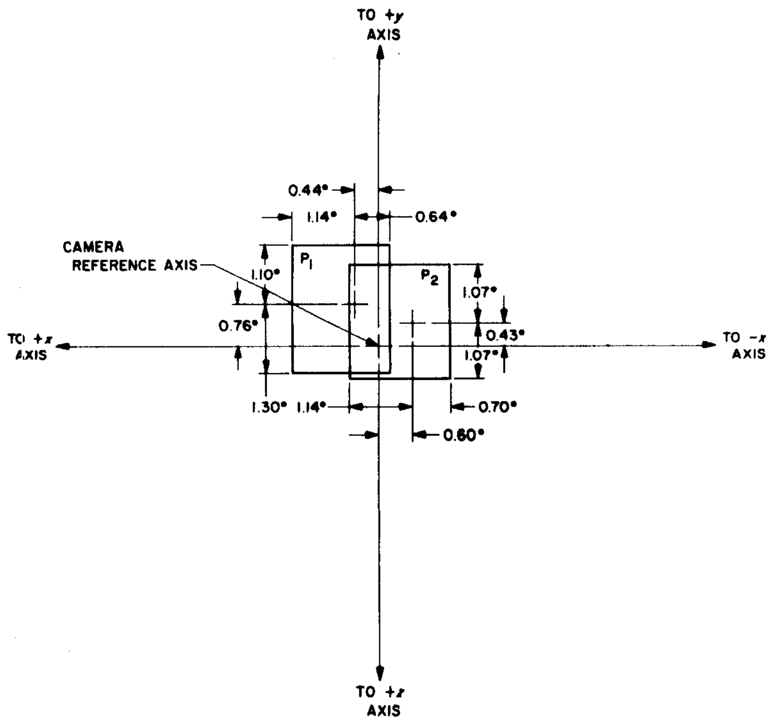


RANGER T V SUBSYSTEM

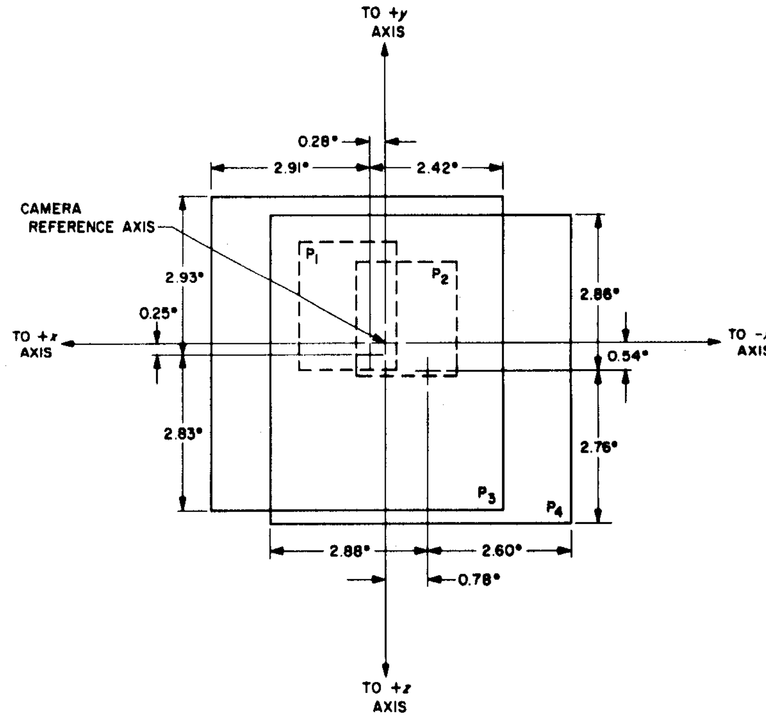




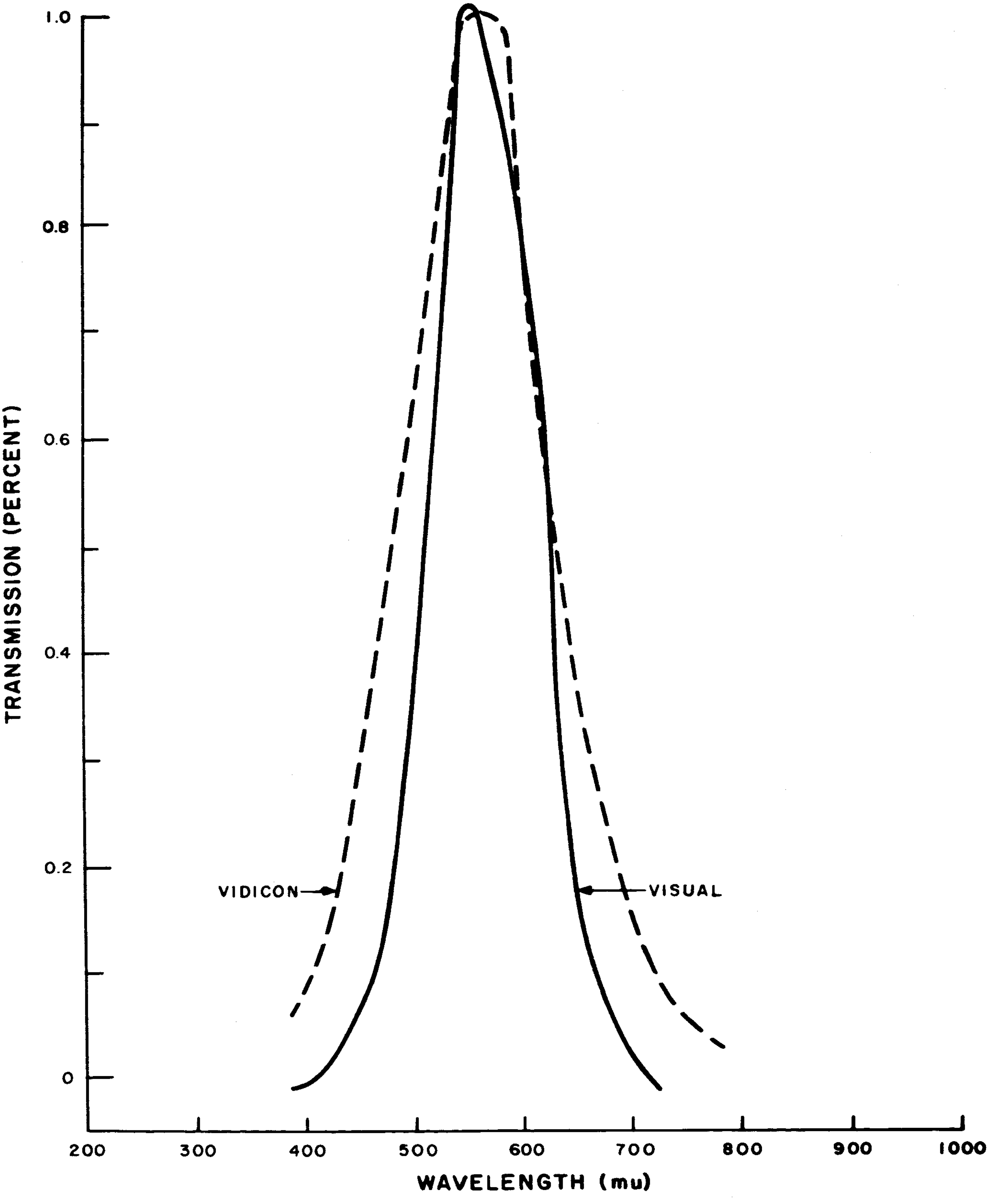
***Ranger VII* camera fields of view (a) Cameras A and B**

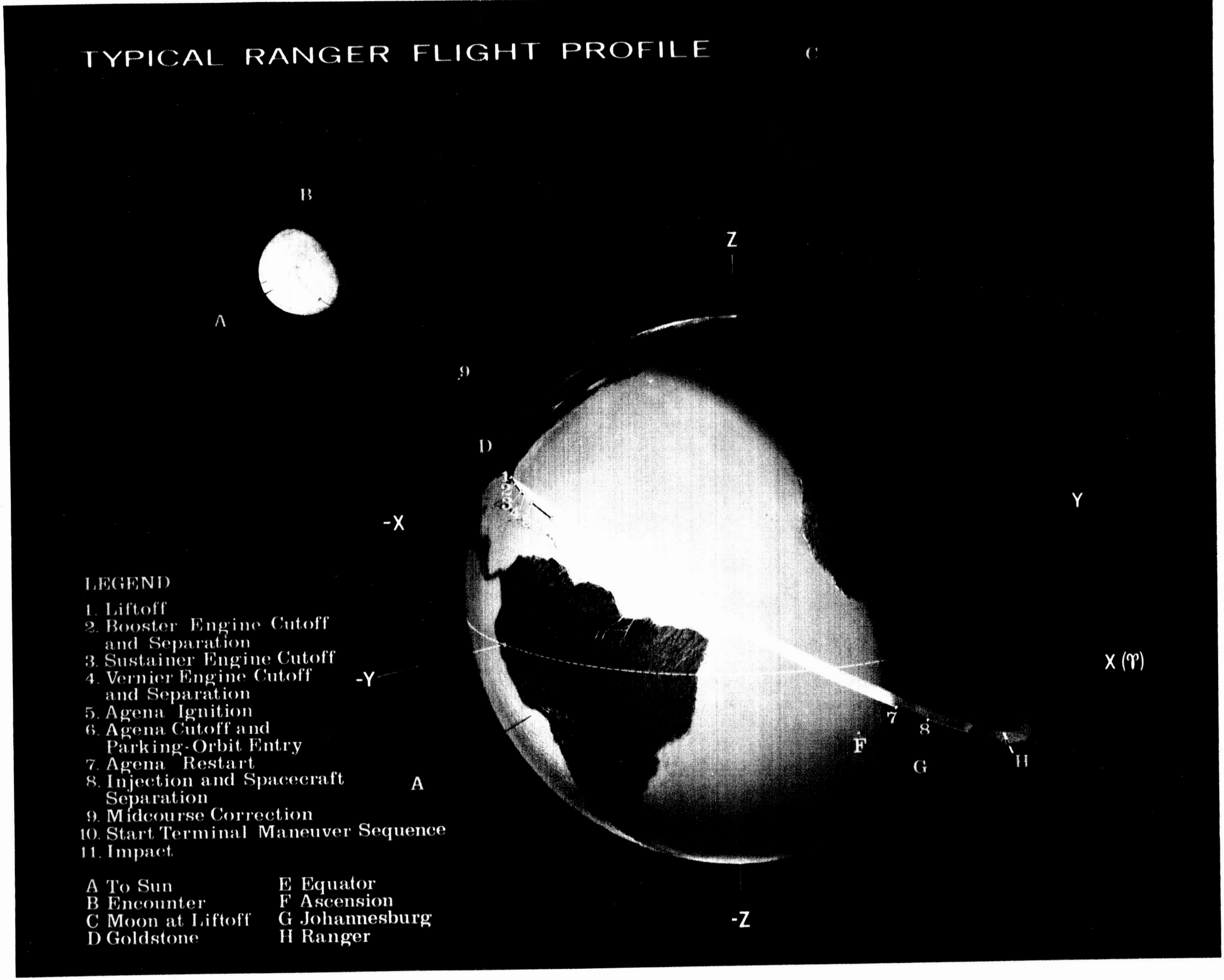


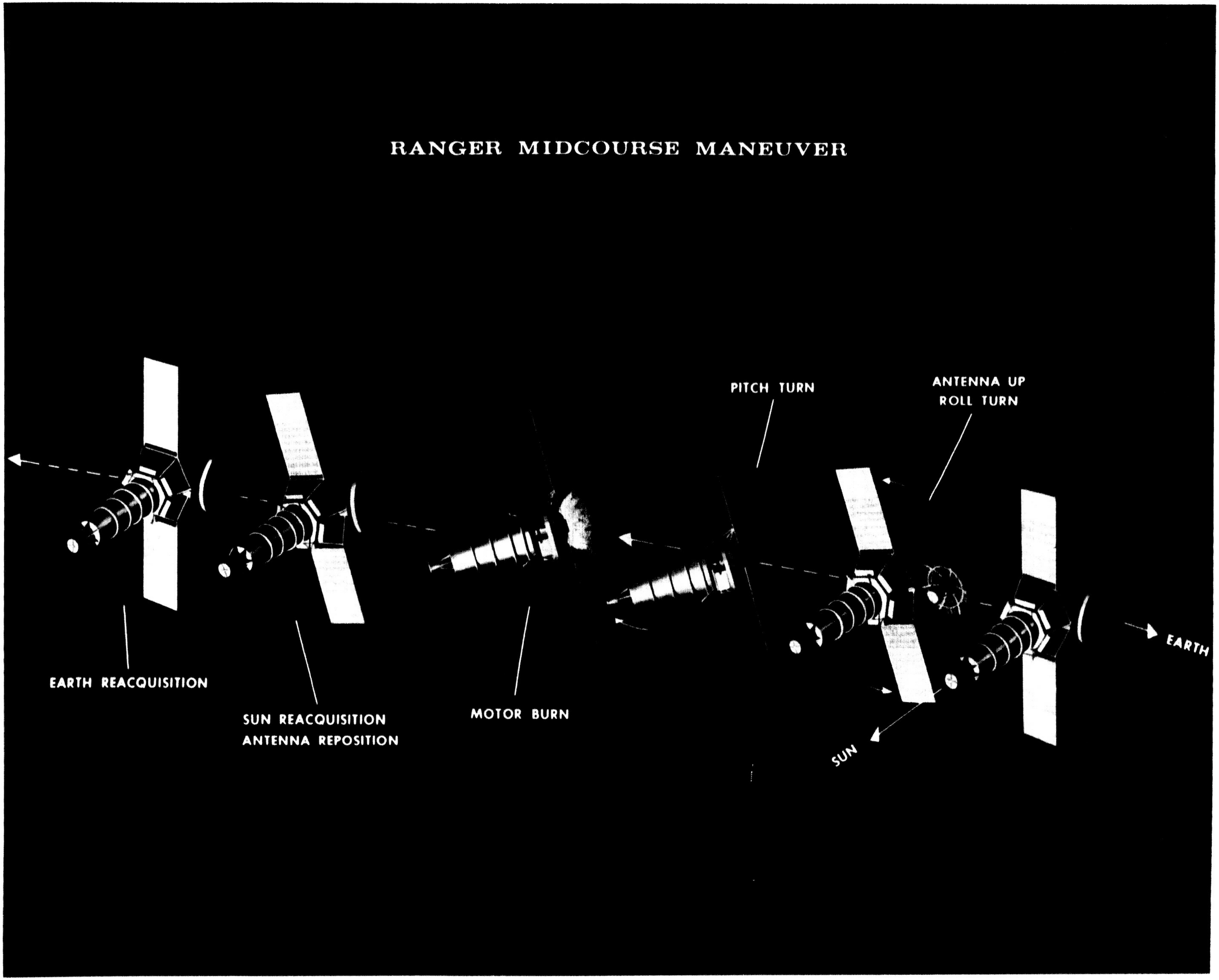
(b) Cameras P_1 and P_2

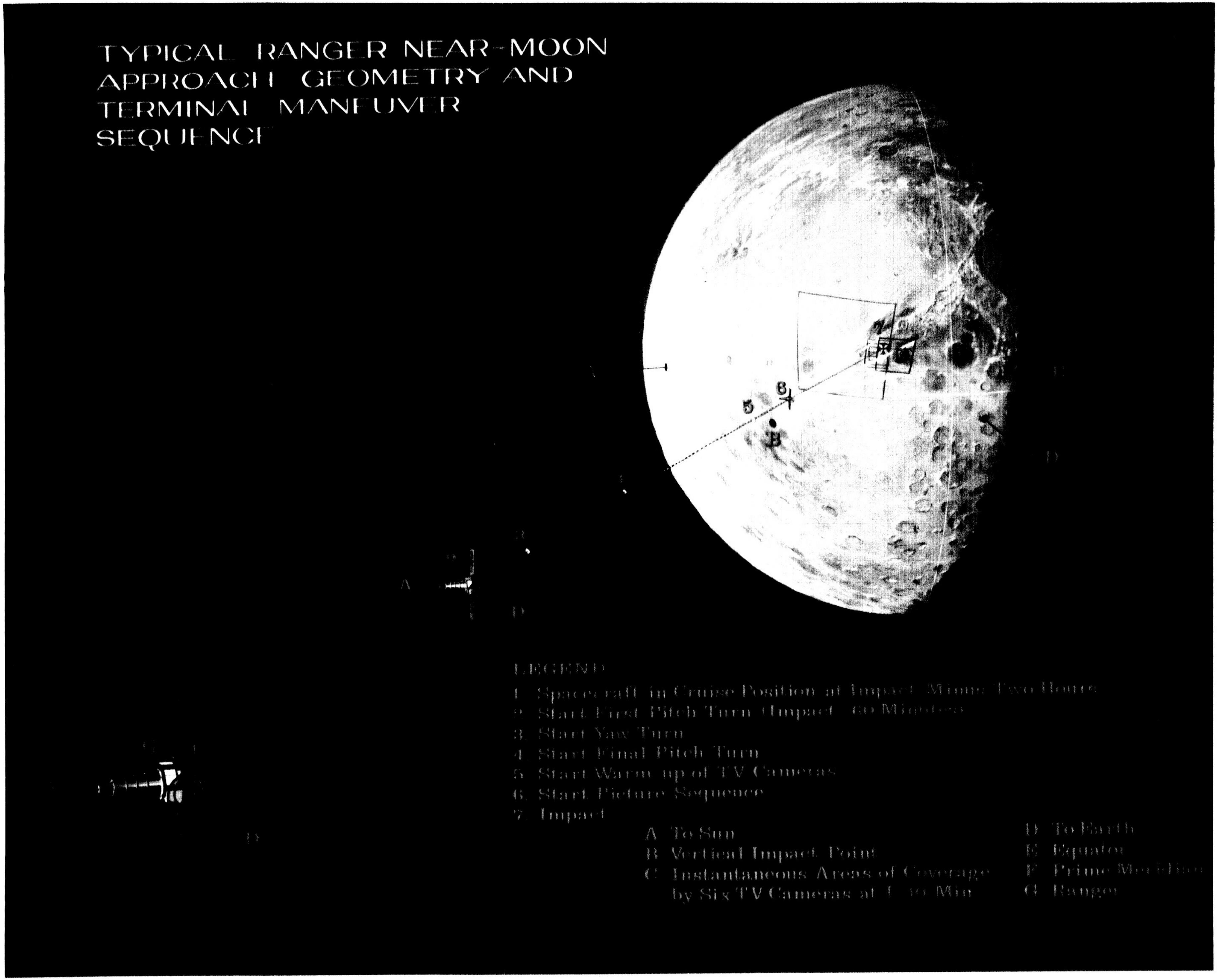


(c) Cameras P_3 and P_4

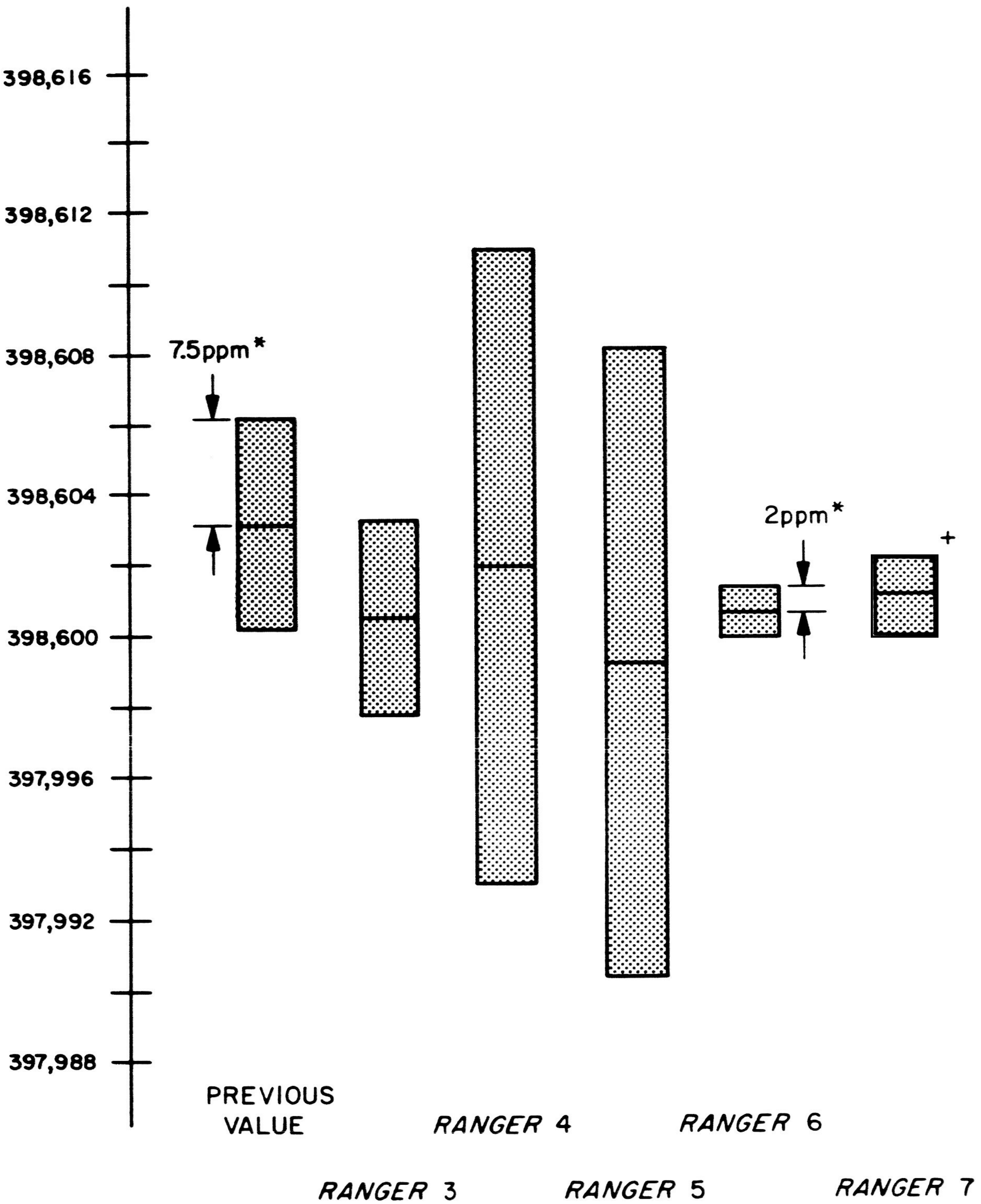

TRANSMISSION (PERCENT)
1.0
0.8
0.6
0.4
0.2
0
VIDICON
VISUAL
200
300
400
500
600
700
800
900
1000
WAVELENGTH (mu)


TYPICAL RANGER FLIGHT PROFILE
C
B
A
Z
9
D
-X
Y
-Y
X (♈)
7
8
F
G
H
A
-Z
LEGEND
1. Liftoff
2. Booster Engine Cutoff and Separation
3. Sustainer Engine Cutoff
4. Vernier Engine Cutoff and Separation
5. Agena Ignition
6. Agena Cutoff and Parking-Orbit Entry
7. Agena Restart
8. Injection and Spacecraft Separation
9. Midcourse Correction
10. Start Terminal Maneuver Sequence
11. Impact
A To Sun
B Encounter
C Moon at Liftoff
D Goldstone
E Equator
F Ascension
G Johannesburg
H Ranger


RANGER MIDCOURSE MANEUVER
PITCH TURN
ANTENNA UP
ROLL TURN
EARTH
EARTH REACQUISITION
SUN REACQUISITION
ANTENNA REPOSITION
MOTOR BURN
SUN

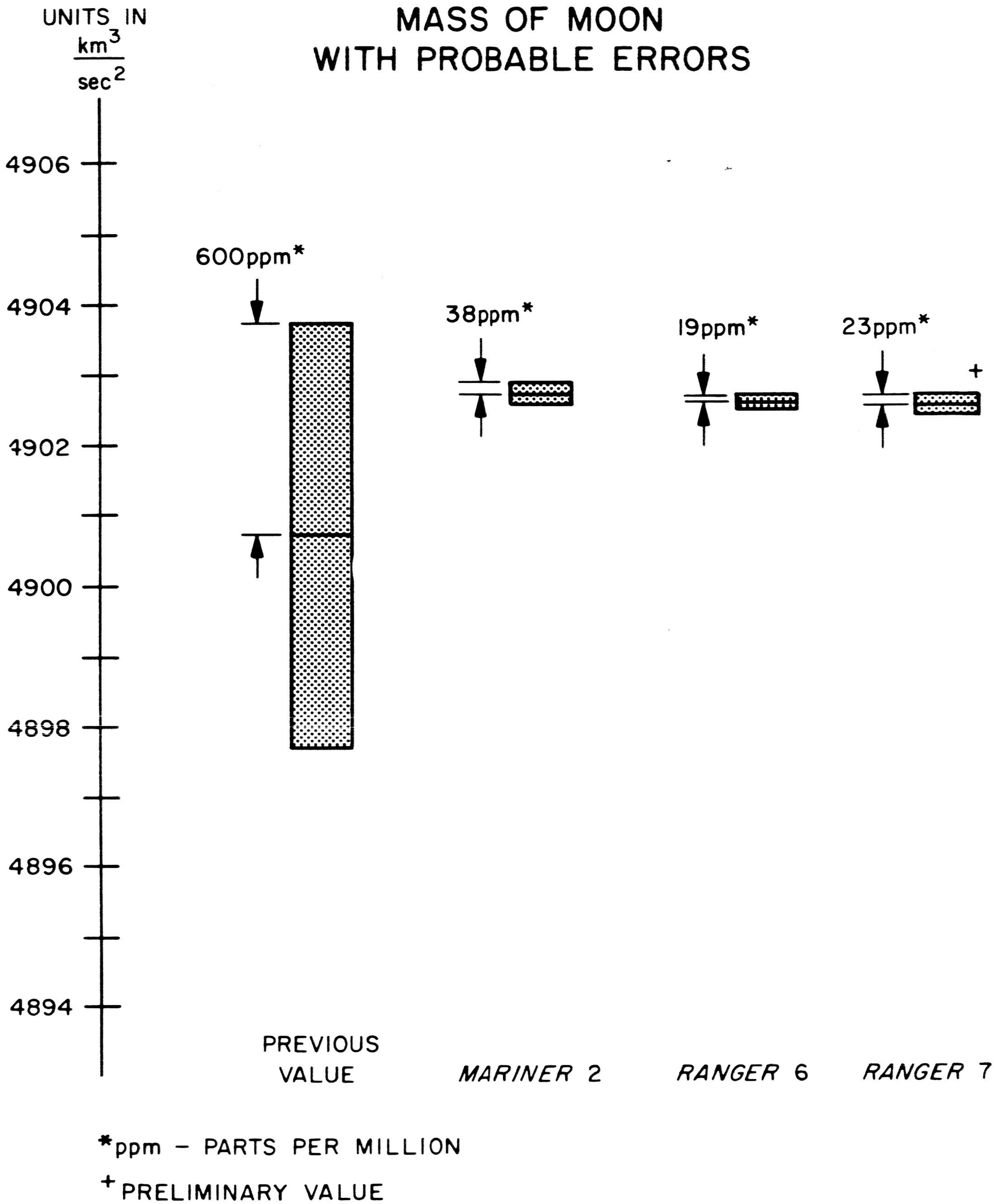

TYPICAL RANGER NEAR-MOON APPROACH GEOMETRY AND TERMINAL MANEUVER SEQUENCE
LEGEND
1. Spacecraft in Cruise Position at Impact Minus Two Hours
2. Start First Pitch Turn (Impact 60 Minutes
3. Start Yaw Turn
4. Start Final Pitch Turn
5. Start Warm up of TV Cameras
6. Start Picture Sequence
7. Impact
A To Sun
B Vertical Impact Point
C Instantaneous Areas of Coverage by Six TV Cameras at I Min
D To Earth
E Equator
F Prime Meridian
G Ranger

MASS OF EARTH WITH PROBABLE ERRORS



*ppm = PARTS PER MILLION

+PRELIMINARY VALUE


MASS OF MOON
WITH PROBABLE ERRORS
UNITS IN $\frac{km^3}{sec^2}$
4906
4904
4902
4900
4898
4896
4894
600ppm*
38ppm*
19ppm*
23ppm*
+
PREVIOUS VALUE
MARINER 2
RANGER 6
RANGER 7
*ppm - PARTS PER MILLION
+PRELIMINARY VALUE