NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report No. 32-699

Ranger VI Mission Description and Performance

GPO PRICE $ ______

CFSTI PRICE(S) $ ______

Hard copy (HC) 3.00

Microfiche (MF) 1.30

ff 653 July 65

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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Ranger VI Mission Description and Performance

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December 15, 1966

Prepared Under Contract No. NAS 7-100
National Aeronautics & Space Administration

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ABSTRACT

The *Ranger VI* mission was launched on January 30, 1964 on a lunar impact trajectory with the mission objective of obtaining close-up television pictures of the lunar surface. A failure in the television subsystem (the nature and causes of which were thoroughly analyzed) prevented achievement of this objective. However, with this exception, the flight was successful, demonstrating a full operational capability and providing an impact on the lunar surface on February 2, 1964 at 9.3 deg north selenographic latitude, 21.5 deg east longitude, within 20 mi of the aiming point.

I. INTRODUCTION

Ranger VI was launched on January 30, 1964 at 1549:09 GMT[1] from Pad 12 at the Atlantic Missile Range (AMR), Cape Kennedy, Florida. The spacecraft was boosted aloft by a launch vehicle consisting of *Atlas D* (199) and *Agena B* (6008) that injected the spacecraft on a lunar-flyby trajectory with a miss distance of approximately 600 mi. The AMR downrange tracking and telemetry stations supported early portions of the flight. The subsequent spaceflight portion of the mission was supported by tracking and communications stations of the Deep Space Network (DSN), located in South Africa, Australia, and Southern California, and was conducted from the Space Flight Operations Complex (SFOC)[2] at JPL.

[1] GMT times are used in this report. For reference, liftoff occurred on January 30, 1964 at 7:49 AM, PST, or 10:49 AM, EST; impact occurred on February 2, 1964 at 0924 GMT or 1:24 AM, PST.

[2] The Space Flight Operations complex was an interim establishment; *Ranger VI* was its last mission. Following this mission, necessary equipments were moved to the Space Flight Operations Facility, JPL, which was used for *Ranger VII* and subsequent missions.

The *Ranger VI* mission objective was to obtain high-resolution television pictures of the lunar surface, during a terminal phase preceding lunar impact, for the benefit of scientific knowledge and the U.S. manned-lunar-landing program. Although an accurate lunar impact was achieved, an apparently early inflight failure of the television subsystem, discovered during the terminal phase when the subsystem failed to operate, prevented achievement of the mission objective. The failure is believed to have been caused by the destruction, by arcing, of high-voltage television elements during an inadvertent turnon while the spacecraft was in a critical pressure environment, shortly after *Atlas* booster-engine cutoff.

A. Mission Events

After two scheduled and two unscheduled holds (neither caused by any spacecraft problem), a nominal launch occurred approximately 37 min after the opening of the launch window on the first day of the launch period. The *Atlas* and *Agena* vehicles performed within their tolerances, injecting the spacecraft on a flyby trajectory well within midcourse correction capability. Separation, solar panel extension, Sun acquisition, Earth acquisition, and switchover to the high-gain antenna occurred close to nominal times.

The midcourse maneuver commands, specifying the roll, pitch, and velocity change calculated to correct the *Ranger VI* trajectory, were sent to the spacecraft beginning at 0724 GMT on January 31, 1964. The maneuver sequence was commanded at 0830, and the maneuver was executed without anomaly, changing the flight path from a flyby to an impact trajectory to the target area in *Mare Tranquillitatis* at 8.5 deg north latitude, 21.0 deg east longitude. Sun and Earth reacquisition following the maneuver were nominal.

Cruise-mode performance was nearly normal up to a few minutes before impact. Because of the approach geometry, no terminal maneuver was required. Therefore, the TV backup clock became the prime turnon source for the F-channel cameras, and an RTC-7 command was sent 15 min before impact to turn on the P cameras. There was no evidence of TV RF radiation from the spacecraft; no video and no detailed (90-point) diagnostic telemetry was received. The spacecraft impacted the lunar surface on February 2, 1964 at 0924:33.1 GMT at 9.3 deg north, 21.5 deg east, approximately 20 mi from the target point.

The various events of the mission are listed in Appendix A, with their nominal mission times and actual times.

Anomalies observed from the telemetry included an unscheduled TV telemetry (15-point) turnon at 140 sec after launch, lasting 69 sec, loss of a group of temperature measurements just after midcourse-motor ignition, and some minor disparities between predicted and actual temperature and other readings.

B. Project Background

Ranger VI initiated Block III of the *Ranger* Project, which began the NASA/JPL unmanned lunar program in 1959. Block III was to consist of four flight missions.

The first two *Ranger* flights, constituting Block I, were directed toward engineering test of spacecraft equipment, verification of the parking-orbit ascent trajectory concept, and the system design, using a highly elliptical Earth orbit. A payload of environment-sampling scientific instruments was flown. The two missions, launched in the Fall of 1961, experienced launch-vehicle failures that left the spacecraft in low-altitude, short-lived satellite orbits. Some scientific data were obtained and some spacecraft design elements were tested, but the mission objectives were not met.

The three Block II flights occurred in 1962. For these missions, a retromotor-slowed landing capsule containing a seismometer experiment and radio system, and two lunar-approach experiments, including approach television, were employed in conjunction with the *Ranger* basic bus. The three Block II spacecraft were sterilized. The missions collectively demonstrated launch vehicle performance and spacecraft design adequacy, but unfortunately, not on the same attempt. *Ranger III* afforded the opportunity for a full exercise of cruise operations, but the launch dispersion precluded lunar impact, and an ill-timed spacecraft failure interrupted the TV-flyby alternative attempt. *Rangers IV* and *V* suffered early disabling spacecraft failures; however, the capsule transmitters were tracked through the missions, to impact in the case of *Ranger IV*, and for eleven days in the case of *Ranger V*.

Following the conclusion of *Ranger* Block II, an exhaustive review, analysis, and evaluation of *Ranger* ensued. Block III, then under development, was deferred, and its first launch delayed. A number of improvements to the Block III spacecraft were undertaken to assure reliability (see Appendix B). Documentation, design control, quality control, and material control were enhanced and formalized; test and review efforts were intensified.

The spacecraft payload for *Ranger* Block III had been designed around a multicamera approach television system in place of the capsule and its retromotor, with some other experiments retained. This TV subsystem was redesigned, and became the sole experiment. The mission objective for the Block III missions was simplified and clarified. The flight objective was to obtain television pictures of the lunar surface with a photographic resolution of at least one order of magnitude better than that available in Earth-based photography. The first launch was scheduled for December 1963.

C. Project Description

The various efforts which supported the *Ranger VI* mission consist of four Systems: The Launch Vehicle System (which included launch operations), Spacecraft System, Deep Space Network (DSN) System, and Space Flight Operations (SFO) System. Mission relationships of these Systems are shown in Fig. 1. Each System's role under the project was defined by the Project Develop-

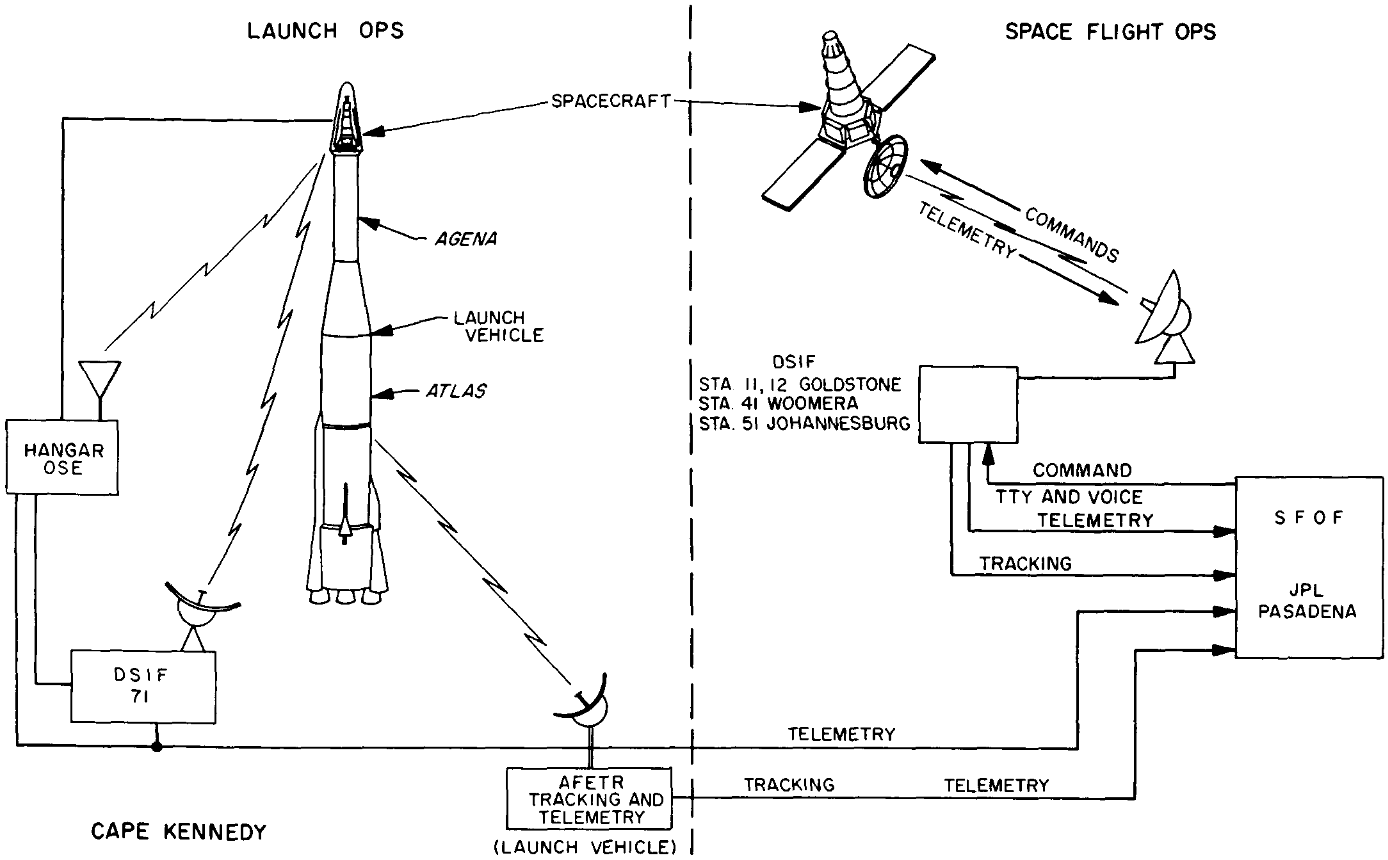


Fig. 1. *Ranger* system elements

ment Plan, and interfaces were maintained by mutual agreement under Project-Office direction.

1. Launch Vehicle System

The role of the Launch Vehicle System was to deliver the spacecraft into a specified lunar trajectory. The *Atlas D* vehicle (No. 199) was procured from General Dynamics/Astronautics by the USAF Space Systems Division for NASA Lewis Research Center. The *Agena B* (No. 6008) and adapter were made by Lockheed Missile and Space Company under contract to LeRC, the Launch Vehicle System Manager. Launch operations were conducted at the AMR, Cape Kennedy, Florida, by Goddard Space Flight Center Launch Operations (GLO).

2. Spacecraft System

The role of the Spacecraft System was to carry out the lunar flight, placing the TV cameras on the desired approach path for lunar observation, and to obtain and return the desired TV data. Activities associated with launch, cruise, midcourse maneuver, and terminal maneuver were included, encompassing telecommunications, power, mechanical and environmental support, and attitude control. The spacecraft bus was designed and assembled within JPL using broad industrial support; the TV subsystem was designed and manufactured under JPL contract by RCA Astro-Electronics Division.

3. Deep Space Network

Four major tracking stations (two at Goldstone, California, one at Woomera, Australia, and one at Johannesburg, South Africa) and the ground-communications network joining them were committed to *Ranger* Block III. This System had the responsibility of tracking and maintaining two-way communication with the spacecraft throughout the mission, and recording original data.

4. Space Flight Operations

Housed in a computation, analysis, and operations center at JPL, and connected to the Deep Space Network, this System conducted and controlled the mission. Tracking and telemetry data were analyzed, and commands were prepared for transmission to the spacecraft.

II. LAUNCH PREPARATION AND OPERATIONS

A. Spacecraft Prelaunch

The *Ranger VI* launch had been scheduled for December 1963 early in the year; spacecraft assembly and testing proceeded toward that end until October 21, 1963, when the spacecraft was ready for shipment to AMR. At that time the discovery that certain diodes used in the spacecraft and launch vehicle constituted a potential source of failure forced a postponement. Upon replacement of critical diodes and reverification of the spacecraft, it was shipped to AMR, arriving on December 23. Final assembly, checks, and system test were completed late in January for the January 30, 1964 launch. Total spacecraft prelaunch operating time was 535 hr.

1. Spacecraft Assembly and Test

Initial assembly began July 2, 1963; subsystem testing and calibration, occupying 76 hr of operation, were conducted August 7–22. The first system test was conducted without the TV subsystem, which was installed August 27. System tests No. 2, 3, and 4 were completed and all flight subsystems were installed by September 20. Modal and three-axis vibration testing followed.

Three mission tests in the 25-ft space simulator were conducted beginning September 27, October 3, and October 4, 1963. Mission test No. 2 was terminated after 13 hours because of a TV battery failure; No. 3, a repeat, was carried out with dummy TV batteries. Following SFOC/spacecraft testing and system test No. 5, the spacecraft was ready for shipment.

2. Diode Problem and Recovery

A number of 1N459 diodes, used in several spacecraft and launch vehicle applications and in critical circuits in the *Ranger* CC&S and Earth sensor, were found to be subject to short circuits caused by loose gold flakes from an internal bonding agent. These units were removed from the spacecraft and the diodes were placed.

Flight spares were tested on the spacecraft in mid-November. No TV was installed, no solder joints were broken, the flight antenna actuator and attitude-control gas systems were used; the spare CC&S was not tested. The spares performed satisfactorily. An antenna-deflection test, the Earth-sensor reverification test, TV calibration, and a redundant modal survey were also performed. On December 5, 1963 with the spacecraft once more assembled with flight equipment, system test No. 6 was performed; the SFOC compatibility test followed the next day. Mission test No. 4, a reverification performed at ambient temperature in the space simulator, was conducted December 10–13.

On completion of JPL assembly and test operations and satisfactory conclusion of the preshipment status review on December 17, the spacecraft and support equipment were shipped to AMR in five vans, leaving December 19 and arriving December 23, 1963.

3. AMR Test Operations

The *Ranger VI* spacecraft and the OSE arrived at AMR on December 23, 1963. By December 31, the following operations had been performed: (1) the OSE was installed and checked out, (2) the spacecraft hand-carried items were reassembled, (3) the TV subsystem was checked out, (4) an attitude-control subsystem test was performed, and (5) the spacecraft was given an initial system test.

On January 3, 1964, a backup-functions system test was performed, the solar panels were checked with the new solar-panel-illumination OSE, and the secondary Sun sensors were checked. The spacecraft was then transported to the Explosive Safe Facility for an attitude-control system leak test and preparations for the initial on-pad operations.

Operations from January 6–9, 1964 included the following:

1. Installation of midcourse propulsion subsystem and electrical checkout.
2. Installation of TV subsystem.
3. Weight determination and center-of-gravity adjustments.
4. Match-mate with *Agena* adapter.
5. Attitude-control gas system pressurization to flight pressure.
6. Installation of shroud.
7. Operational checkout of spacecraft in the Explosive Safe Facility.

On January 9, 1964, spacecraft operations started at Launch Complex 12. An umbilical safe check was performed. A dummy precountdown was run. Spacecraft operations were normal, except when the TV RF power came on. This noise and lock-up problem was corrected by installing a 20-db attenuator in the omni-transmitter link.

A combined radio-frequency interference (RFI) test, a dummy-run countdown, and the spacecraft joint flight acceptance composite test (J-FACT) were performed. An anomaly with the dc power supply in the launch shelter, an open fuse in the vehicle command destruct package, and an improper roll gyro sync indication occurred during these tests. As a result, the RFI test was rerun. All spacecraft operations were normal during the rerun.

Mechanical preparation for the final system test was made in hangar AM. The TV subsystem was demated and its shroud removed. The batteries were then removed and inspected. The spacecraft was installed on the system test stand.

The flight TV subsystem cameras were calibrated. During a test performed prior to installing the TV shroud, it was found that diodes in the temperature-sensor assembly were defective. A spare temperature-sensor assembly was installed; it performed correctly. The TV shroud was installed and final buttonup for flight was completed.

The solar panel and battery continuity wiring test was performed satisfactorily. To obtain more information about the gyro sync problem discovered during the J-FACT testing, the spacecraft was turned on three additional times, and the voltages and gyro outputs were recorded. Proper gyro sync was obtained on each spacecraft turnon with no abnormal indications.

The final system test was performed on January 18, 1964, utilizing full-time computer support. Some difficulty was experienced with the communications system. A damaged cable at the communications OSE was removed, redressed, and reinstalled. No other problems were noted during this test, and the spacecraft was approved for flight.

The TV subsystem was exercised in cruise mode with the battery jumper plugs installed. Battery temperature measurements were correctly obtained. A special test was performed on the TV subsystem to verify proper backup operation and to correct timing of the TV-clock-pulse watch. A solar panel exciter test was successfully accomplished. The high-gain antenna was installed, and the antenna deflection test was completed.

After final Explosive Safe Facility operations, the spacecraft was moved to the launch pad on January 25, 1964, and the electrical checkout was completed. A special test was conducted to verify proper gyro operation and to determine more accurately the midcourse-motor leak rate. The leak rate on the midcourse motor was not considered excessive. Spacecraft operation was normal.

A simulated launch was performed. Spacecraft operations were normal, and the test was considered successful. Another special test was performed to verify again proper gyro operation. Spacecraft power was turned on. No spacecraft problems were noted.

B. Launch Vehicle

The *Atlas D* (199) arrived at AMR November 30, 1963, and was erected on Complex 12 on December 10.

Agena B (6008) arrived at AMR on November 22, 1963.

Following checkout of individual systems and subsystems, the *Atlas* and *Agena* were mated on December 30, 1963.

Joint spacecraft and launch vehicle tests were conducted satisfactorily except for Agena command destruct receiver 1 which blew a fuse during a radio interference (RFI) test. The fuse was replaced. During the J-FACT testing the *Agena* ullage rocket motor circuit was found to be open. The *Atlas* programmer "safe" light was lost. This was traced to an aerospace ground equipment deficiency and was rectified by a procedural change. The channel A accelerometer mounted on the GE guidance mounting rack was noisy, and a hydraulic leak was discovered in the *Atlas* sustainer engine and rectified.

During the simulated launch on January 28, 1964, a number of problems were encountered.

The electrical cable to the *Agena* unsymmetrical dimethyl hydrazine (UDMH) drain line guillotine was not installed. The *Agena* ground 28-v backup power supply failed. A relay in the *Agena* ground-equipment battery charger failed and was replaced.

At $T - 40$ sec of the simulated launch, the pneumatic internal / external valve could not be activated. The problem was found to be in the ground relay circuitry, but could not be further isolated because the condition could not be duplicated. All four relays were replaced.

C. Countdown and Launch

Countdown operations began on January 30, 1964, at 0727 GMT (T −395 min). The spacecraft picked up the count three hours later at T − 215.

In addition to the two built-in holds at T − 60 and T − 7 min, two holds were called to deal with launch vehicle problems.

At T − 202 min, the Atlas fuel tank indicated overfill. At T − 155, a hold was called. Reference to other sensors indicated that the fuel loading was correct, and the count was resumed after 23 min.

At T − 15 min, a hold was called for repair of the GE-guidance encoder power supply. A short necessitated the replacement of several circuit cords and revalidation of the system. The count was resumed after 40 min.

Liftoff occurred at 1549:09.092 GMT, approximately 37 min into the window. Launch plan 30E was employed, entailing a 95.0-deg azimuth. Launch weather was satisfactory: 4,500-ft ceiling, 10/10 cloud cover, 7-mi visibility, no precipitation or thunderstorms. The sea-level temperature was 64°F. Winds were 10 knots from 340 deg at surface, maximum 77 knots from 254 deg at 42,000 ft, and maximum shear 18 knots per thousand ft.

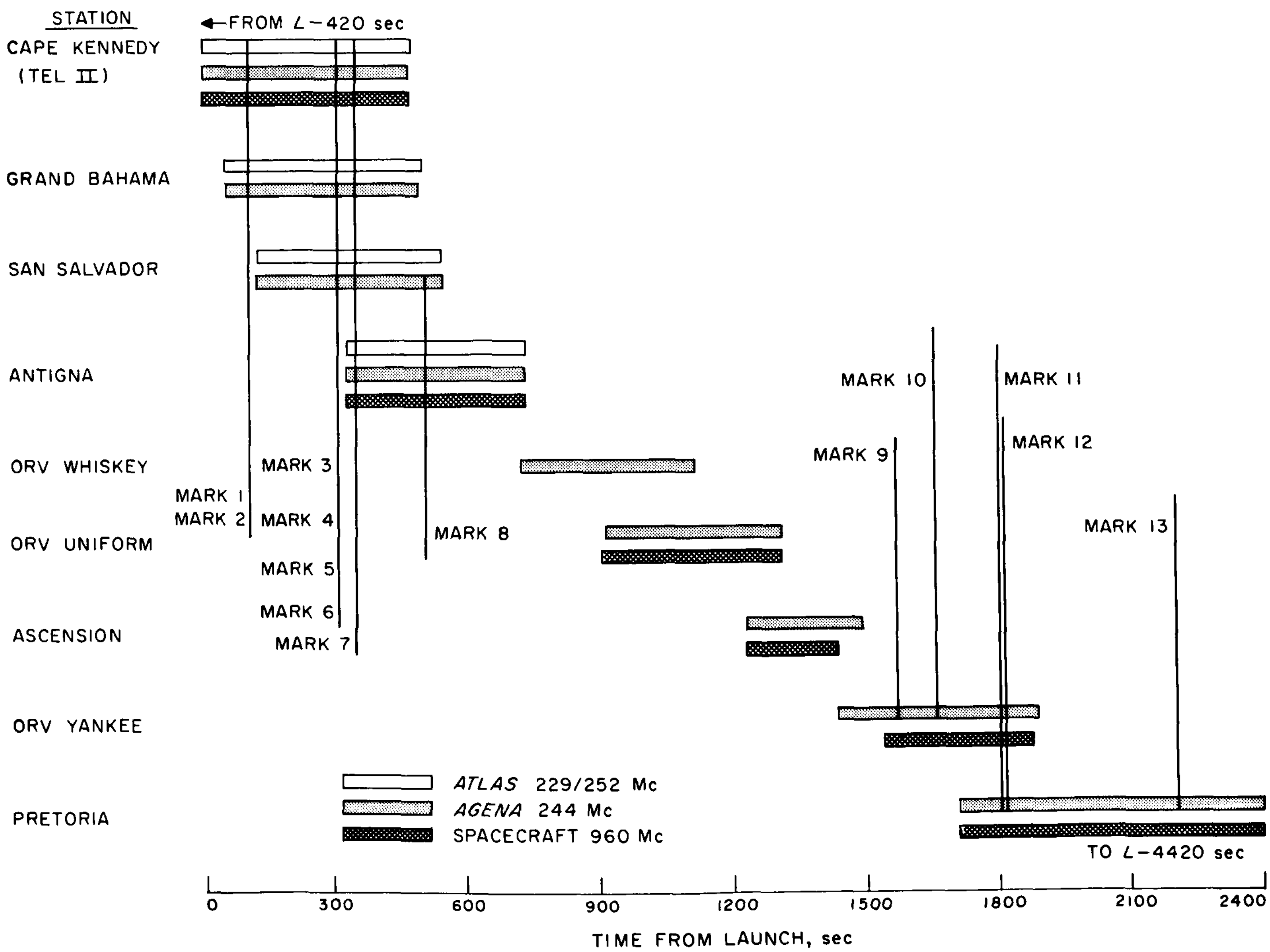


Fig. 2. *Ranger VI* AMR telemetry coverage

D. AMR Postlaunch Support

Photographic, tracking, and telemetry reception stations from the launch complex down range as far as Pretoria, South Africa, supported the *Ranger VI* mission. Twenty-eight engineering sequential cameras, in and near the immediate AMR launch area, obtained metric, engineering performance, and documentary information. Ten tracking radars covered the launch trajectory and provided data for real-time trajectory calculations. Six ground telemetry stations and three ships provided telemetry coverage variously on the *Atlas*, *Agena*, and spacecraft.

The Mark II Azusa at the Cape tracked the *Atlas* transponder to $L + 376$ sec; coverage by the C-band radar system is given in Table 1. Antigua data (37 points) were used in calculating the parking orbit; 71 points from the Pretoria radar were used in calculating the transfer trajectory.

Atlas discretes were confirmed in real time at Tel 2; *Agena* discretes were confirmed by Antigua Island, ORV *Yankee*, and Pretoria. General telemetry coverage is given in Fig. 2. Telemetry from ORV's *Uniform* and *Yankee* and from Ascension Island and Pretoria were transmitted in nonreal time by SSB radio to the Cape; the transmission was generally noisy. Air-snatch data return was also employed from the ships, *Whiskey* and *Uniform*; *Yankee* data were retransmitted to the aircraft, recorded there, and returned to the Cape.

Table 1. AMR C-band tracking

Station	Location	Coverage, sec from launch
1.16	Cape Kennedy	50-310
0.16	Patrick Air Force Base	50-310
3.16	Grand Bahama Island	60-493
5.16	San Salvador Island	138-562
	Bermuda Island	335-658
91.18	Antigua Island	386-730
886C	(ORV Uniform)	954-1280
12.16	Ascension Island	1309-1350 1380-1450
13.16	Pretoria, South Africa	2160-3630

III. LAUNCH VEHICLE SYSTEM

An *Atlas D/Agena B* vehicle system was used to launch the *Ranger VI* spacecraft from Complex 12 at Cape Kennedy, Florida.

Powered flight telemetry, tracking, and other range support services were supplied by the Atlantic Missile Range (AMR). All *Atlas* and *Agena* discretes were confirmed in real time by the range support facilities.

The *Atlas D/Agena B* is a 2½-stage vehicle in which all engines of the *Atlas* are ignited and stabilized prior to commitment to launch, while the single *Agena* engine is ignited in flight twice: first to accelerate the *Agena*/spacecraft combination to the velocity required to maintain a circular orbit about the Earth, and then, after a suitable coasting period in this parking orbit, to accelerate the *Agena*/spacecraft combination to the required injection velocity necessary to escape the Earth's gravitational field and coast to the Moon.

A. Vehicle Operation

1. *Atlas*

a. Planned Sequence for Atlas. The *Atlas D* is a 1½-stage boost vehicle containing five rocket engines that utilize a kerosene-like hydrocarbon and liquid oxygen as propellants and at launch give a thrust-to-weight ratio of approximately 1.25. This modified version of the *Atlas* missile is shown in Fig. 3.

All five engines (the two boosters, the one sustainer, and the two vernier engines) are ignited on the ground prior to liftoff to ensure maximum reliability of this stage. After the majority of the *Atlas* propellants are consumed in flight, and prior to the time the vehicle acceleration attains 7 *g*, the two outboard booster engines are shut down and jettisoned, and the vehicle continues on, powered primarily by the sustainer engine. When the required velocity for the *Atlas* portion of the flight has been achieved, the sustainer engine is shut down and, for a few seconds only, the vernier engines provide thrust to stabilize the vehicle and to achieve the precise velocity desired. When this has been accomplished, the verniers are shut down, the *Agena*/spacecraft combination is separated from the *Atlas*, and the *Atlas* is backed away from the *Agena* by two small solid-propellant retro rockets.

The *Atlas* is guided during its flight first by an onboard programmer and autopilot and later by a radio guidance system that sends correction signals to the autopilot based on information obtained from a ground-based radar tracking station. The on-board programmer and autopilot guide the vehicle from liftoff through the jettisoning of the booster engines, except for a 10-sec period when the ground radio command guidance system is turned on. After the booster engines are jettisoned, the radio guidance loop is again enabled, and the vehicle is guided by the ground-based guidance and computer system for the rest of the *Atlas* powered flight.

b. Atlas 199D Operation. *Atlas* boost phase conditions were normal and did not present any structural or thermal problems to the *Agena* or spacecraft. An analysis of the flight indicated that there were no gross abnormalities. The expected signal dropout at booster staging occurred and lasted for 451 millisec. This signal dropout is believed to be caused by the flames and gases from the sustainer engine impinging upon the jettisoned booster section, causing attenuation of the transmitted telemetry signal.

Atlas performance and guidance resulted in nearly perfect booster coast apogee conditions in which the altitude was only 16 ft low and the velocity was 0.7 ft/sec low. Booster steering capability was available but it was not necessary to use it. *Atlas* initiation of the *Agena* primary timer occurred 5.98 sec earlier than predicted; but only 0.07 sec early with reference to the coast ellipse. *Atlas* pitchdown, nose shroud ejection, and *Atlas/Agena* separation were normal.

2. *Agena*

a. Planned Sequence for Agena. The *Agena B* (Fig. 4) is a single-engine dual-start upper-stage vehicle utilizing unsymmetrical dimethyl hydrazine as fuel and inhibited red fuming nitric acid as oxidizer. When first ignited, the *Agena B* has a thrust-to-weight ratio of approximately unity. Its flight control system consists of a programmer, a reference gyro system, two horizon sensors, and a velocity meter. Elements of the flight control system are preset on the ground prior to launch. By means of the *Atlas* radio guidance system, a ground-calculated discrete command initiates the timing function for *Agena* second burn; the *Agena* receives no further guidance or control signals from the ground subsequent to separation from

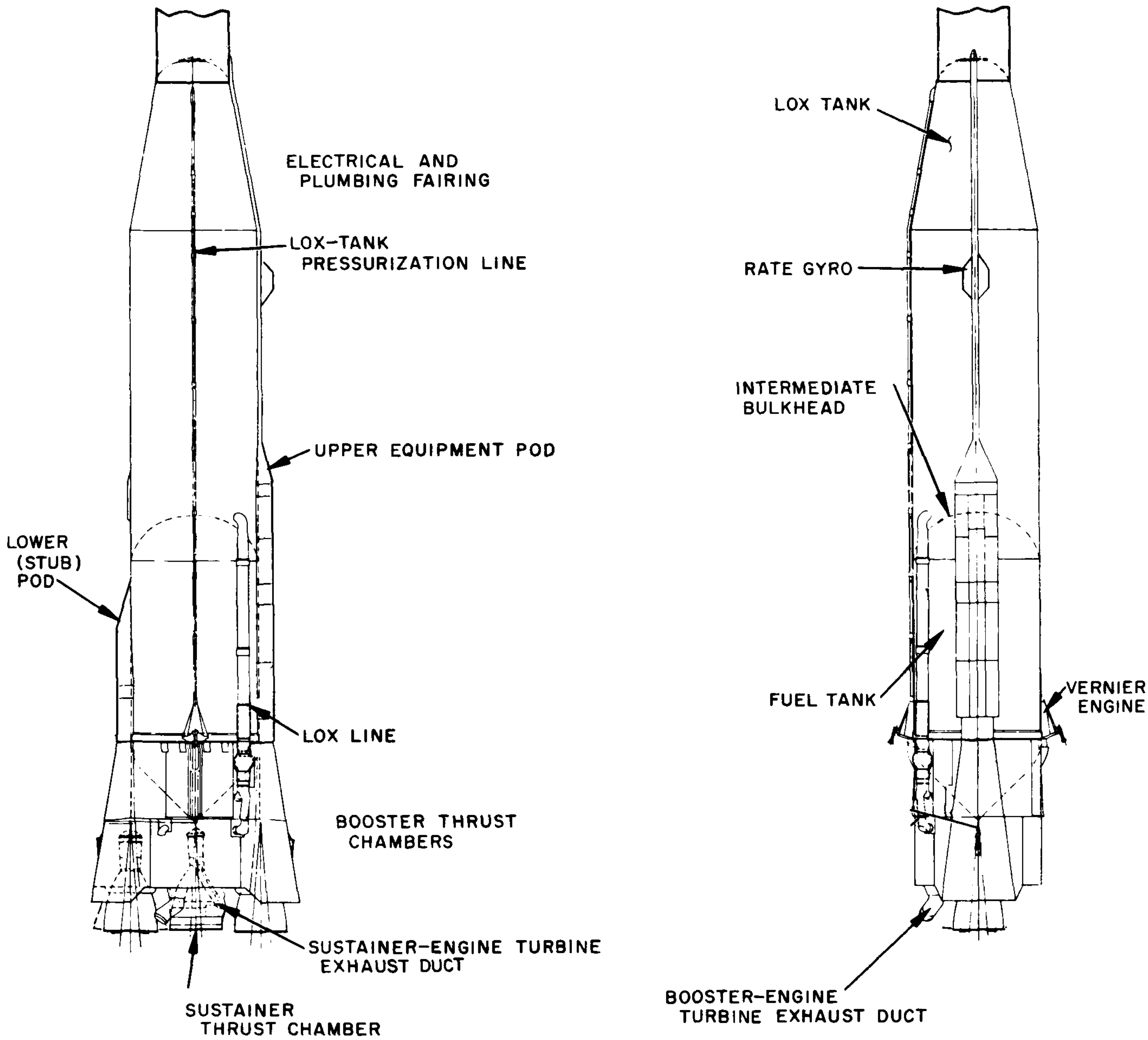


Fig. 3. *Atlas* launch vehicle

the *Atlas*. The programmer and the gyro references provide the discrete events and the basic vehicle attitude information during coasting and powered flight phases. The horizon sensors view the Earth and update the gyro information during the flight to compensate for gyro drift. The velocity meter is preset for the required velocity-to-be-gained by the *Agena* stage, and determines when the engine will be shut down. The *Agena* engine is required to ignite twice in order to accelerate the spacecraft to the required injection conditions. After attaining the required velocity, the engine is shut down, the spacecraft separates from the *Agena*, and the *Agena* executes an almost-180-deg yaw maneuver. The *Agena* is decelerated by a small solid-propellant retro rocket to prevent it from impacting on the Moon.

b. Agena B 6008 Operation. All *Agena* subsystems performed almost perfectly throughout the upper boost phase of flight. Both *Agena* engine operations were normal. Only 9 lb of control gas (34 lb was loaded, with 19 lb usage predicted) were used by the *Agena*, indicating excellent guidance and an extremely stable flight. Both sequence timers functioned within specifications. C-band radar and telemetry coverage exceeded predictions, and

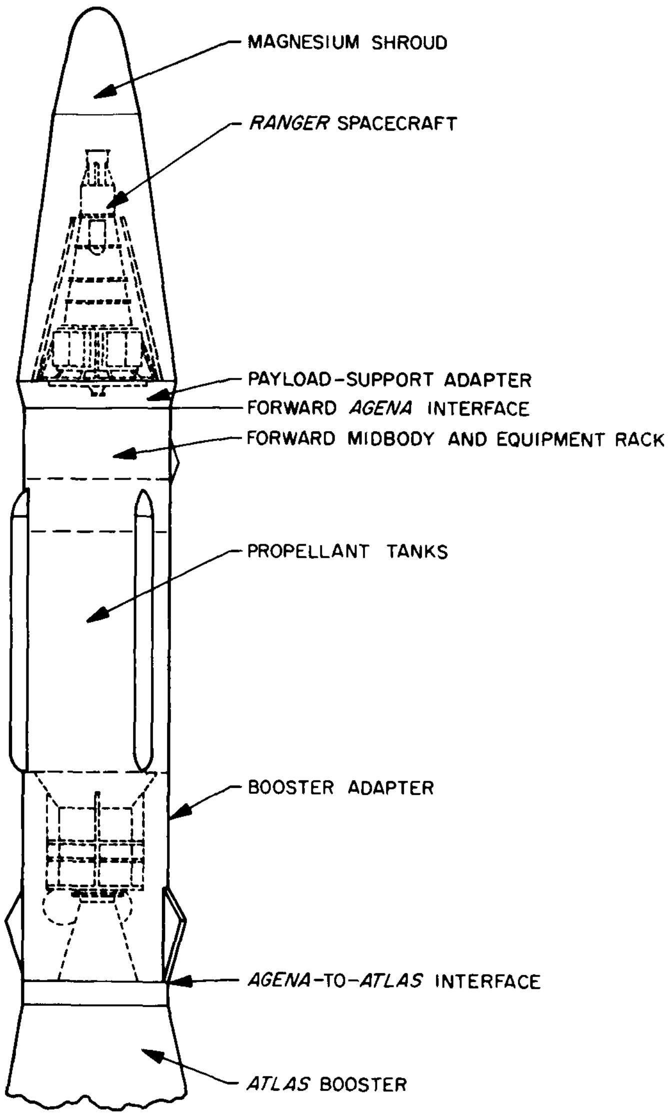


Fig. 4. *Agena* injection vehicle

Agena instrumentation properly provided all data programmed. All electrical power equipment functioned properly and power levels remained well within specifications until the last sampling at $T + 4311$ sec (6700 nm altitude). New items on board were the Hercules ullage rocket, a new power converter and new horizon sensors. All operated normally.

One anomaly was indicated. Data from the radial vibration monitor on the *Agena*/spacecraft adapter indicated levels at that point twice those of previous *Ranger* flights during liftoff and in the transonic region. Special dynamic analyses of this anomaly were conducted.

Agena/spacecraft separation and *Agena* yaw maneuver events were normal. *Agena* retrograde rocket impulse provided a velocity decrement of approximately 45 ft/sec which was approximately 1.7 ft/sec low. Trajectory studies show the *Agena* passed the Moon on the trailing side, but within its effective gravitational field. The *Agena* orbit energy was increased in such a manner that the spent stage entered a heliocentric orbit with a computed period of 383.7 days.

B. Flight Performance

The performance of the Launch Vehicle system for the *Ranger VI* flight appeared to be completely nominal. With respect to the target specification for launch plan 30E, the combined *Atlas* and *Agena* performance resulted in a 2-sigma error injection condition, most of which was contributed by the *Agena* (midcourse velocity correction of approximately 36 m/sec), which was well within the midcourse correction capability of the spacecraft (60 m/sec). Booster performances appeared to be satisfactory through all flight phases leading to separation of the spacecraft on the lunar trajectory.

IV. SPACECRAFT PERFORMANCE

The twelve bus subsystems of the *Ranger* VI spacecraft (Fig. 5) performed well within their design tolerances throughout the flight with minor exceptions, none of which had any adverse effect on the outcome of the mission. The spacecraft properly executed all internally programmed commands and all ground commands that were transmitted during the launch and midcourse maneuver phases. The attitude control subsystem maintained the precise Sun and Earth lock required for normal solar power operation during cruise modes, and for data recovery via the high-gain communications link. The communications subsystems provided continuous tracking and telemetry signals for accurate orbit determination and real-time performance analysis.

Observed deviations from nominal performance by the Spacecraft System consist of the following:

1. The television subsystem failed to operate, and no pictures were recovered. No video signals from either TV channel were received. The first and only indication of an anomaly came at $L + 140$ sec when the TV cruise telemetry came on for about 1 min. When the TV cruise telemetry was monitored at the normal time, nominal indications were observed and standard operation of the payload was assumed.
2. A transducer insulation breakdown caused a temperature bridge power supply to be shorted out right after midcourse motor ignition, which resulted in the loss of two propulsion subsystem temperature measurements for the duration of the maneuver mode and seven cruise mode temperature measurements.
3. The TV package, the Earth sensor, and one Sun sensor operated at temperatures above predicted levels.
4. The telemetered values of attitude-control gas tank pressures and the Earth sensor light intensity did not agree with predicted values.

The spacecraft flight sequence (given in Appendix A) lists chronologically all the discrete standard and nonstandard events involving the spacecraft from countdown to lunar impact. The actual flight sequence followed the

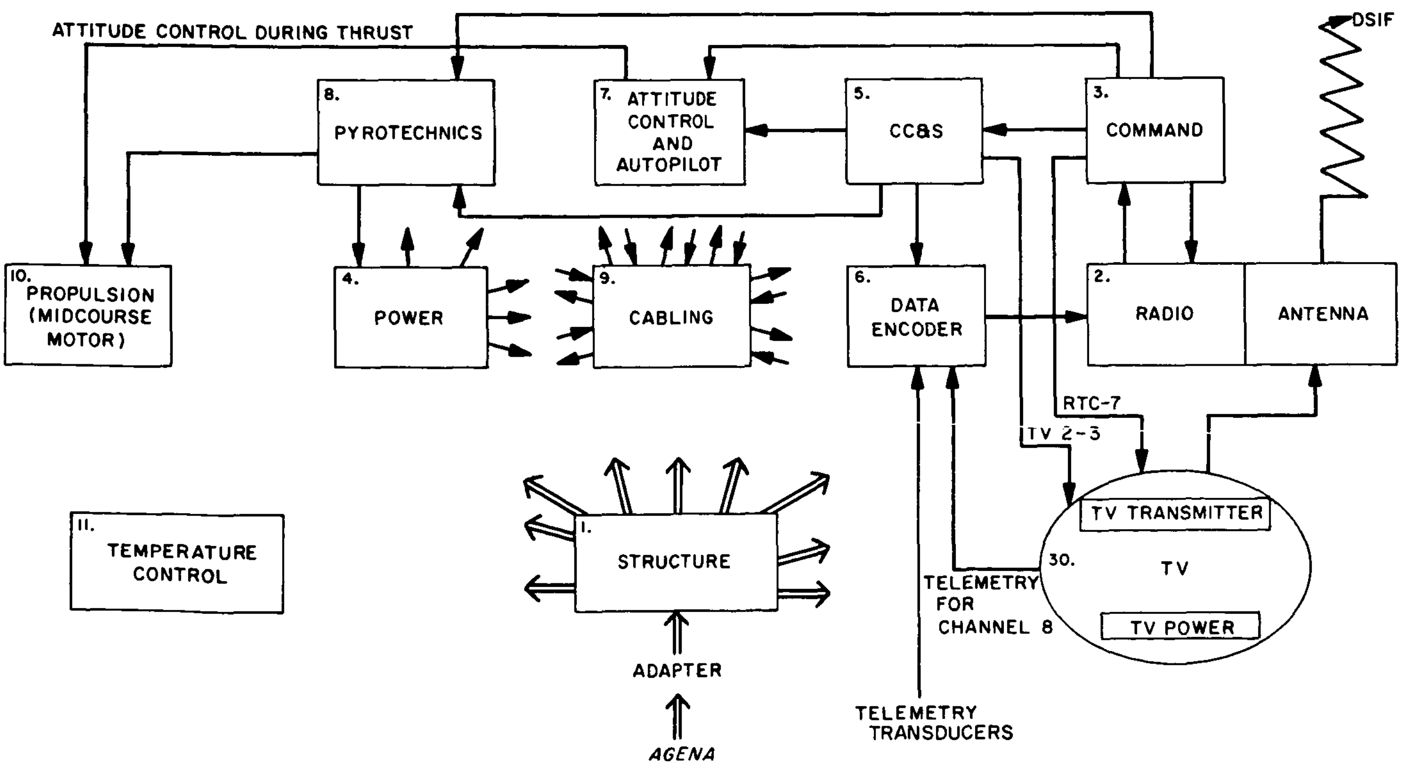


Fig. 5. *Ranger* Block III spacecraft subsystems

standard sequence of events very closely except for the absence of terminal maneuver commands and events, and the presence of the nonstandard events previously mentioned.

Spacecraft configuration, coordinates, and general arrangement of subsystems are given in Appendix C.

A. Radio Subsystem

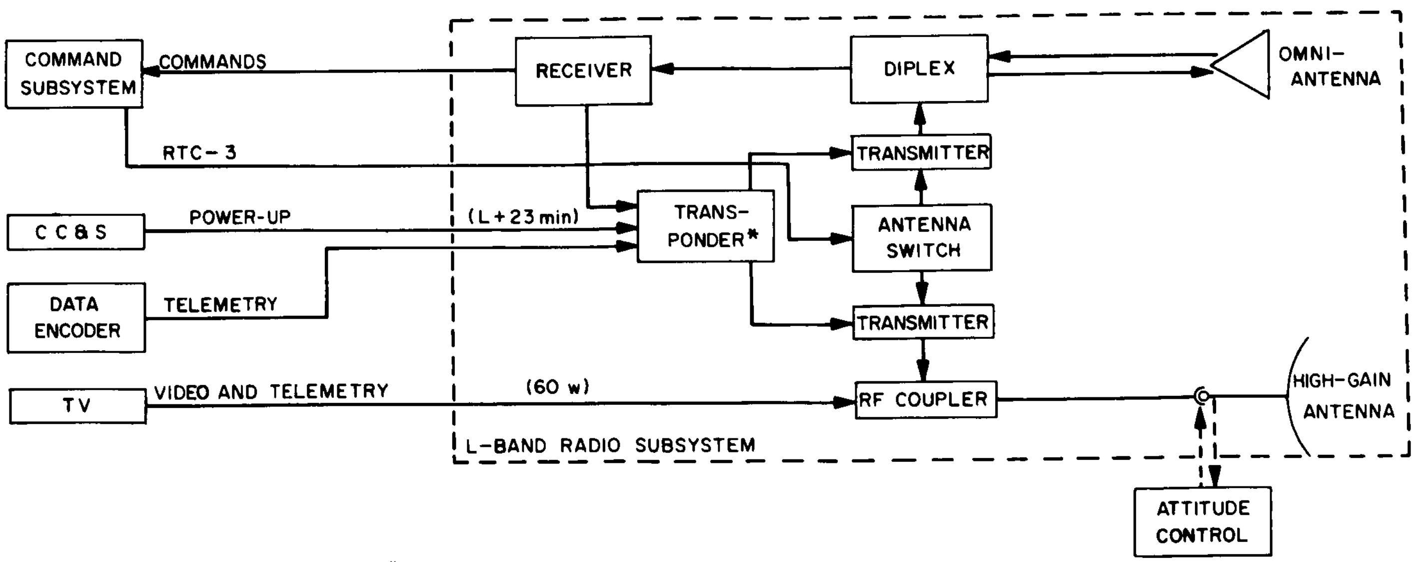
The spacecraft L-band radio subsystem is composed of five major elements: omnidirectional antenna, high-gain antenna, receiver, transmitter, and auxiliary oscillator (Fig. 6). The subsystem is designed to receive signals and commands via the omni antenna. The omni antenna is also used to transmit signals when the spacecraft is solar-oriented or nonoriented. The high-gain antenna is used to transmit signals when the spacecraft is Earth-oriented. The transmitter-receiver combination is designed to transmit a phase-modulated signal which is phase-coherent with the received signal. When no signal is received by the spacecraft, a noncoherent signal is provided to the transmitter by an auxiliary oscillator.

Operation of the transmitter was monitored by three telemetered measurements. Two antenna-drive measurements indicate the amount of power supplied to each antenna and a 250-v monitor indicates the amount of plate voltage on the RF power amplifiers. All three measurements indicated expected values through the *Ranger VI* mission, as shown in Table 2.

The receiver operation is monitored by means of five telemetered measurements. Two measurements of receiver automatic gain control (AGC) voltage with coarse and fine resolution indicate the spacecraft received signal strength, and a local-oscillator-drive measurement indicates the amount of power supplied by the receiver local oscillator. Two other measurements, coarse and fine static phase error, indicate the dc correction voltage resulting from the frequency difference between the uplink signal and the receiver voltage controlled oscillator (VCO). These measurements are generally zeroed for the transmission of a command and thus become operational aids for ground stations.

Prior to and during the launch phase the Cape station, DSIF 71, stayed in one-way lock only with the spacecraft. Two-way lock was not attempted since the need for command capability was not anticipated. Although DSIF 71 momentarily lost lock at liftoff, the station continued to receive the spacecraft signals until radio horizon which occurred approximately 8 min after liftoff.

Powerup occurred on schedule at $L + 23$ min, and the CC&S event was telemetered and received by AMR. Normal powerup operation was not confirmed by spacecraft data, however, until DSIF 51 and DSIF 59 acquired at $L + 32$ min. At this time, the 250-v monitor measurement (Table 2) indicated 255 v and the low-gain antenna drive indicated 34.8 dbm, corresponding to the normal powerup operation.



*INCLUDES MULTIPLIERS, MODULATORS, AUXILIARY OSCILLATOR

Fig. 6. L-band radio subsystem

Table 2. *Ranger VI* telemetered transmitter parameters

Time, GMT	Event	Temperature, °F	RF amplifier, v	Low-gain antenna drive, dbm	High-gain antenna drive, dbm
—	Nominal mission values	70-110	152-252 ±4	31.2-33.4 $^{+1.5}_{-1.0}$	21.4-23.6 $^{+1.4}_{-1.0}$
1338, 1/30	Countdown	76	153	—	21.6
1548	Prelaunch	76	154	31.6	—
1555	Shroud ejection	77	154	31.6	—
1612	Transmitter power up	78	255	34.8	—
1655	Sun acquisition	80	253	34.9	—
1945	Earth acquisition	91	252	34.7	—
2113	Antenna transfer	94	253	—	24.7
0821, 1/31	Antenna transfer	100	253	34.7	—
0852	End midcourse turns	98	252	34.6	—
0905	Sun reacquisition	94	253	34.7	—
0929	Earth reacquisition	98	252	34.7	—
0945	Antenna transfer	100	252	—	24.8
0750, 2/2	Pre-impact	102	255	—	25.0

During the initial rise over DSIF 51 and DSIF 59, the spacecraft appeared to be tumbling, and DSIF 51 experienced difficulty in maintaining two-way lock. Stable two-way lock did not occur until $L + 55$ min when DSIF 41 acquired the spacecraft. At this time, since the spacecraft was tumbling, the spacecraft AGC coarse and fine measurements exhibited large excursions. The spacecraft-received signal strength, however, appeared to fall within predicted tolerances.

Solar acquisition was completed at $L + 67$ min and was reflected in smaller variations in the spacecraft receiver AGC telemetry. Since the spacecraft was rolling, these variations were due simply to differences in the omni antenna gain for different clock angles. Again, the values fell within predicted tolerances.

Earth acquisition was completed at $L + 236$ min; the receiver AGC measurements showed an improvement in stability. For the remainder of the mission the spacecraft-received signal strength measurements generally fell within 3.0 db of predicted values (Table 3).

The RTC-3 command sequence was first attempted at $L + 276$ min from DSIF 41. Received signal strength at the spacecraft was observed at the Space Flight Operations Complex (SFOC) to be marginal in terms of command capability after transmission of the first RTC-0 "clear" command, and the command sequence was accordingly terminated. Spacecraft telemetry had been observed by DSIF 51, and noise in the ground communications link had made data interpretation difficult and slow. The approximately 30-db-low signal was found to be due to DSIF 41 inadvertently transmitting via its acquisition-aid antenna.

Two-way lock was transferred to DSIF 51, and the RTC-3 command sequence was initiated from that station.

The switchover was confirmed by the antenna drive power measurements. After switchover the high-gain antenna drive indicated 24.7 dbm, normal power output. The switchover was also confirmed by the DSIF-received signal strength which increased by approximately 10 db.

Throughout the cruise phase preceding the midcourse maneuver, all of the measurements fell within predicted tolerances.

Prior to the midcourse maneuver the planned maneuver was analyzed in terms of its effect on signal strength. Since the omni antenna pattern contains numerous nulls in the direction of the roll axis, the spacecraft or the DSIF received signal strengths can fall below threshold for some maneuvers. In the case of the *Ranger VI* maneuver, a momentary approach to carrier threshold data in the pitch turn was expected. It occurred as predicted, slightly exceeding expectations in magnitude.

Table 3. *Ranger VI* telemetered receiver parameters

Time, GMT	Event	Temperature, °F	Predicted receiver AGC, dbm	AGC, dbm (actual)	Phase error	Local oscillator drive, dbm
1400, 1/30	Prelaunch	76	—	−94	0	3.4
1626	DSIF acquisition; DSIF 51; Spacecraft tumbling	79	−80	−85	−50	3.5
1655	Sun acquisition; DSIF 41	80	−113	−112	−50	3.4
1945	Earth acquisition; DSIF 51	91	−95	−96	+30	3.2
0712, 1/30	Premidcourse; DSIF 12	100	−99	−102	0	2.9
1000	Postmidcourse; DSIF 12	101	−100	−104	−20	2.9
1900	Pass 2; DSIF 41	103	−106	−110	−25	2.8
0100, 2/1	Pass 2; DSIF 51	102	−107	−110	−10	2.8
1200	Pass 2; DSIF 12	102	−105	−107	0	2.8
1900	Pass 3; DSIF 41	101	−110	−113	+35	3.1
0100, 2/2	Pass 3; DSIF 51	100	−110	−113	0	3.2
0750	Pre-impact; DSIF 12	102	−108	−111	−30	3.1

Antenna-power switchover (to the omni antenna 10 min before the maneuver started, and back to the high-gain antenna 74 min after the maneuver started) was commanded, executed, and observed in the antenna power measurements.

Solar reacquisition and Earth reacquisition occurred within nominal expected times and were again evidenced by the spacecraft receiver AGC measurements which slowly settled down as attitude stabilization was achieved.

Throughout the postmidcourse phase, and until impact, all the measurements indicated normal values; no discrepancies were noted in telemetered data. There were, furthermore, no times of flight at which ground stations indicated any difficulty in transmitting to or receiving from the spacecraft. On the basis of these facts, the spacecraft radio subsystem appeared to function normally in all respects during all phases of the mission.

B. Command Subsystem

The spacecraft command subsystem consists of a command detector and a command decoder (Fig. 7). Commands are sent to the spacecraft over the radio link by means of a frequency-shift-keyed (FSK) subcarrier signal that is recovered in the spacecraft radio receiver. The command detector performs the following functions: (1) detects the FSK signals provided by the spacecraft radio receiver; (2) determines whether a command is being sent; (3) converts the detected FSK signals into a serial sequence of binary *ones* and *zeros* (which comprise the code); (4) generates various clocking signals for code processing both in the detector and the decoder; and (5) routes these binary and clocking signals to the command decoder. The command decoder receives the serial sequence of binary *ones* and *zeros* from the command detector, forms the data into command words, and determines whether the command is a real-time command (RTC) or a stored command (SC). If the command is an RTC, it is decoded to determine which RTC has been

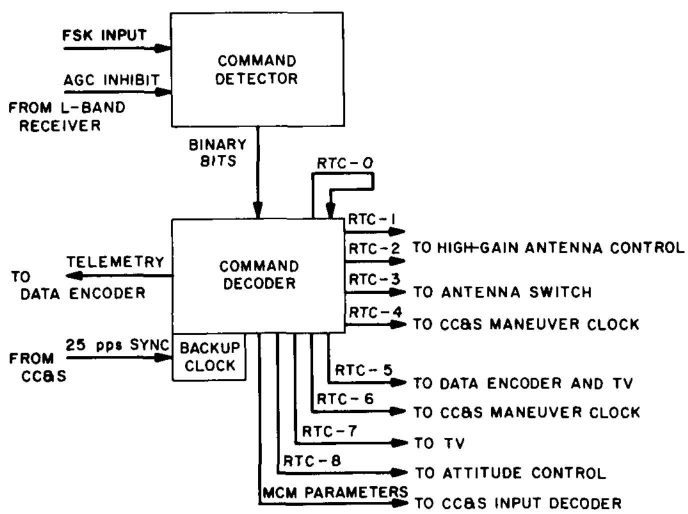


Fig. 7. Command subsystem

transmitted. The appropriate relay is closed to initiate the requisite action in the appropriate subsystem (CC&S, attitude control, etc.). If the received command is an SC, the data are clocked serially into the CC&S for further processing.

Verification that the decoder has received and acted on a command is telemetered in the form of FM pulses on a subcarrier (B-20). The subcarrier, in turn, phase-modulates the RF carrier transmitted from the spacecraft. RTC verification is in a form indicating that some relay closed while an SC is verified by telemetering each data bit as it is transferred to the CC&S. Additional verification of the commands is accomplished by the action noted on the other subsystems via telemetry. Since noise inherent in transmitting and receiving a command can modify the command itself (the probability of a bit error is $\leq 10^{-5}$), the reaction of each command is watched closely.

In every case in which a command was transmitted to *Ranger VI*, a verification of action was indicated on telemetry in the form of the proper pulses. With the exception of the last three RTC-7 commands sent, all reactions of the spacecraft were correct for all commands. In the instance of the last three RTC-7's, it is believed that there was a failure in the subsystem for which the command was intended and that the command subsystem performed normally. On the basis of this supposition, the command subsystem appeared to function normally in all respects throughout the mission.

A summary of the commands received by the command subsystem is given in Table 4.

The commands are defined as follows:

RTC-0: Clear spacecraft command subsystem.

RTC-1: Roll override.

RTC-2: Antenna hinge-angle change.

RTC-3: Antenna switchover.

RTC-4: Initiate midcourse maneuver sequence.

RTC-5: Telemetry mode change; inhibit backup clock.

RTC-6: Initiate terminal maneuver sequence.

RTC-7: Television on/emergency/on/off

SC-1: Midcourse maneuver roll duration.

SC-2: Midcourse maneuver pitch duration.

SC-3: Midcourse maneuver velocity increment.

SC-4: Terminal maneuver first pitch duration.

SC-5: Terminal maneuver yaw duration.

SC-6: Terminal maneuver second pitch duration.

Table 4. Commands received during *Ranger VI* flight

Command	Time sent, GMT	DSIF station sending	Telemetry confirmation
RTC-0	30/2025:00	51	—
RTC-0	30/2108:00	51	—
RTC-0	30/2110:00	51	—
RTC-3	30/2112:00	51	Ch. B-20 at 2112:40
RTC-0	31/0720:00	12	—
RTC-0	31/0722:00	12	—
SC-1	31/0724:00	12	Ch. B-20 at 0724:40
SC-2	31/0726:00	12	Ch. B-20 at 0726:40
SC-3	31/0728:00	12	Ch. B-20 at 0728:40
RTC-3	31/0820:00	12	Ch. B-20 at 0820:40
RTC-4	31/0830:00	12	Ch. B-20 at 0830:40
RTC-3	31/0944:00	12	Ch. B-20 at 0944:40
RTC-0	2/0811:00	12	—
RTC-0	2/0813:00	12	—
RTC-7	2/0908:00	12	Ch. B-20 at 0908:41.7
RTC-7	2/0915:29	12	Ch. B-20 at 0916:10.7
RTC-7	2/0919:21	12	Ch. B-20 at 0920:02.7

RTC-0 is a clear command whose sole function is to cycle the logic of the command subsystem to make sure it is in a state wherein it can receive a command; there are no telemetered data to assure that it was in fact received.

C. *Data Encoder*

The data encoder is required to accept, encode, and prepare for radio transmission signals corresponding to voltages, temperatures, pressures and other normally telemetered variables. The block diagram of the data encoder (Fig. 8) lists all the telemetered measurements. The data encoder uses semiconductor circuitry and relay commutators. The unit consumes approximately 10 w of power, weighs 27 lb, and occupies about 0.9 ft^3.

Because of the limited bandwidth and signal power available, both time division and frequency division

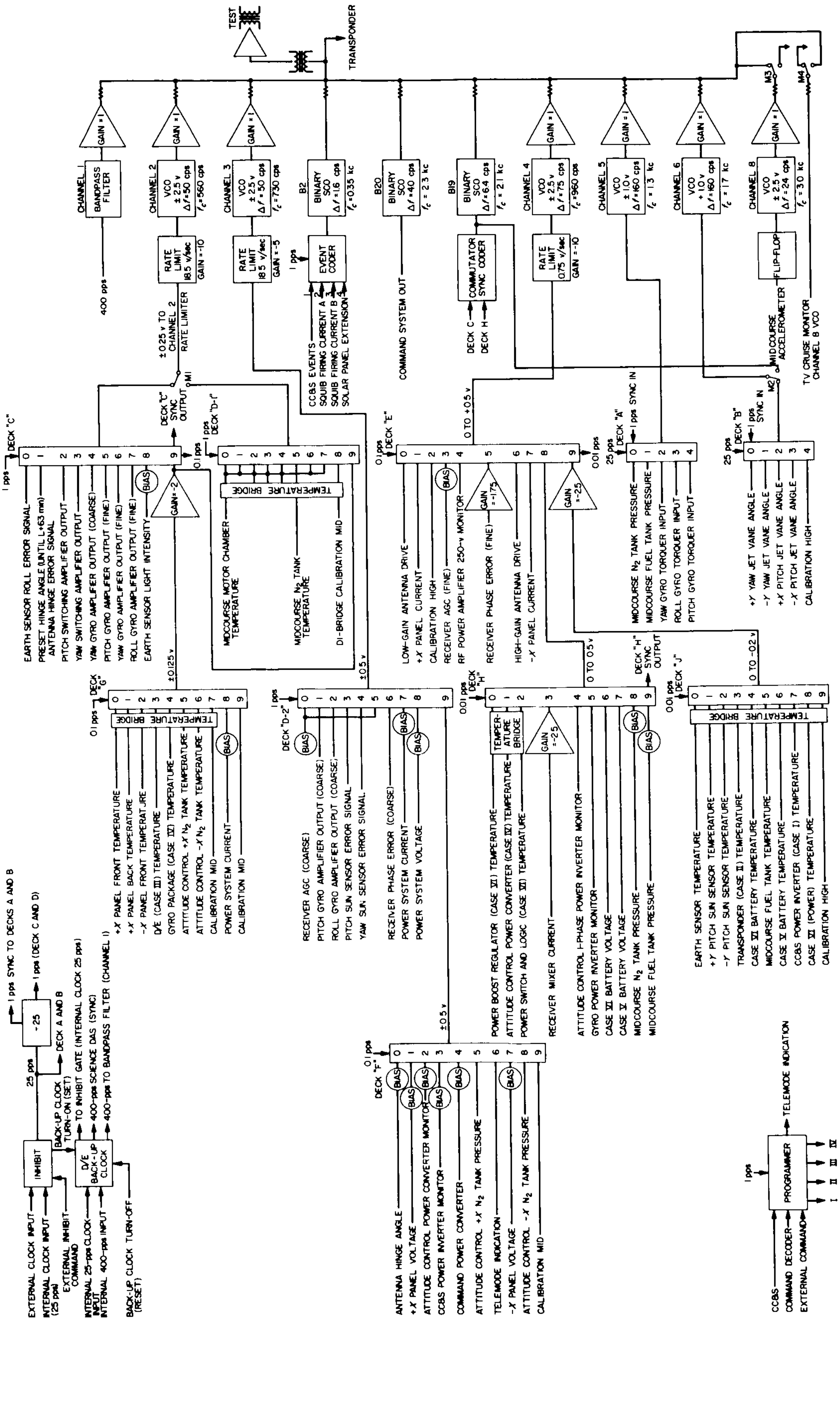


Fig. 8. Data encoder functions

multiplexing methods are employed. Time division multiplexing is obtained by commutation and subcommutation. Frequency multiplexing is accomplished by the use of ten subcarrier frequencies which are linearly summed for direct modulation of the flight transmitter. The ten channels consist of one channel for the reference timing frequency, three binary subcarrier channels, and six conventional, voltage-controlled-oscillator channels.

The major elements employed to perform this function are: (1) temperature and pressure transducers to convert physical parameters to electrical signals; (2) signal conditioning networks to tailor measurement ranges to the proper levels; (3) commutators to perform measurement selection; (4) rate-limited amplifiers to limit the rate of change of sampled signals to a value consistent with the communication system design; (5) voltage-controlled oscillators (VCOs); (6) binary oscillators; and (7) modulation-mixing networks. Also included as major elements are the data selector or programmer, the event coder, and the auxiliary clock.

1. Element Description

Resistance bridges excited by dc-to-dc converters are employed to convert resistance-type temperature transducer variations to dc voltages. Included in the temperature measurements are bridge calibration points to detect supply-voltage variations and bridge-resistor changes of value.

Voltage-dividing networks, bucking-voltage circuits, and dc amplifiers are used as signal conditioning devices to tailor measurements to the desired voltage levels. Reference calibration points are included as part of the signal-conditioning network to indicate zero shifts and sensitivity shifts of dc amplifiers, rate limiters, and voltage-controlled oscillators.

Both solid-state switches and magnetic-latching relays are used as time-division multiplexers to increase the information capability of the telemetry system. These commutators must operate properly so that a fixed sequence of measurement selection will be maintained.

Phase-locked-loop discriminators are used in the ground telemetry subsystem to utilize fully the limited power, and therefore, the limited information bandwidth. The tracking error of these discriminators is directly proportional to the rate of change of the input frequency and inversely proportional to some power of the discriminator loop bandwidth. Since the available power is limited it is impossible to increase the loop bandwidth and maintain the desired threshold performance. In order to maintain a phase error of 30 deg or less for channels 2, 3, and 4 ground discriminators, it is necessary to control the rate of change of their input frequencies. The rate-limited amplifiers are used to limit the maximum rate of input-voltage change to the voltage-controlled oscillators. The maximum allowable rate of change of frequency for channels 2 and 3 is 185 cps/sec and for channel 4 it is 11.2 cps/sec.

Ten subcarriers are used to perform frequency-division multiplexing. Channels 2, 3, 4, 5, 6, and 8 are conventional voltage controlled oscillators, channel 1 is a band-pass filter, and channels B2, B19, and B20 are binary oscillators. Engineering measurement voltages are impressed upon the voltage controlled oscillators. The channel 1 band-pass filter produces a 400-cps sine wave from a 400-pps rectangular wave provided by the CC&S. Since the 400-pps rectangular wave is the basic clock for the data encoder, this information along with the synchronization information discussed below is required by the ground telemetry system to perform automatic data reduction.

Channels B2, B19, and B20 are narrow-band oscillators to transmit binary or two-state information. Channel B2 along with the event coder transmits binary-coded information to indicate the occurrence of spacecraft events. The events are listed in Appendix A. Channel B19 is used to transmit a time reference (frame synchronization) to indicate the completion of a cycle of the commutator. This information is used by the ground telemetry station to identify the 77 commutated measurements. Channel B20 is used for ground-command verification.

The event coder encodes the occurrence of spacecraft events into four codes. Four 1-sec bit codes are used to identify specific events. The events are read into the event coder in parallel and read out serially. A priority system is incorporated in such a way that the event with the highest priority will be read out first if different events occur simultaneously.

The programmer changes the telemetry mode to meet the information requirements of the different phases of spacecraft operations. In the normal cruise mode, channel 6 transmits redundant commutator-synchronization signals and channel 8 transmits TV telemetry. During the midcourse maneuver, channel 6 transmits attitude control jet-vane information and channel 8 transmits midcourse-acceleration information.

2. Performance of the Data Encoder

The data encoder subsystem experienced a failure of the midcourse motor temperature transducer. This failure resulted in the loss of nine temperature measurements from a time very near the start of motor burn throughout the remainder of the flight. With the exception of this failure the data encoder performed to meet all the design requirements.

The commutators, the synchronization signals on channel B19, and the basic encoder timing information on channel 1 all functioned perfectly, permitting continuous automatic data reduction.

The performances of the event coder and channel B2 provided information to indicate the proper sequence and duration of events, and also provided the CC&S subsystem with information to evaluate its performance. In all, the occurrences of about 22 events were telemetered. The programmer performed its function of switching telemetry modes in response to the preprogrammed command from the CC&S.

The stability of the data encoder subcarrier frequencies is determined by monitoring reference measurements. All reference measurements with the exception of address 77 remained well within their allowed tolerance throughout the mission. (Address 77 went to a band-edge reading after the firing of the midcourse motor due to the failed transducer.)

3. Midcourse Motor Chamber Temperature Transducer

During the midcourse motor burn the motor chamber temperature transducer developed a short to the spacecraft frame. The result of this short was to cause all G deck temperature measurements to read at the upper band-edge limit of channel 2 and the D-1 deck temperature measurements to read at the low band-edge limit.

The G and D-1 deck temperature bridges utilize a common power supply. Consequently, a short in either bridge will affect the operation of the other. Figure 9 illustrates this clearly. It should be noted that with a transducer shorted, the bridge reference is destroyed and the output level is shifted to a more positive level than during normal operation. The G deck output signal is inverted by a signal-conditioning amplifier. The D-1 deck output requires no conditioning and is applied directly to the rate limiter. For this reason the short produced opposite band-edge frequencies to be read from the two decks.

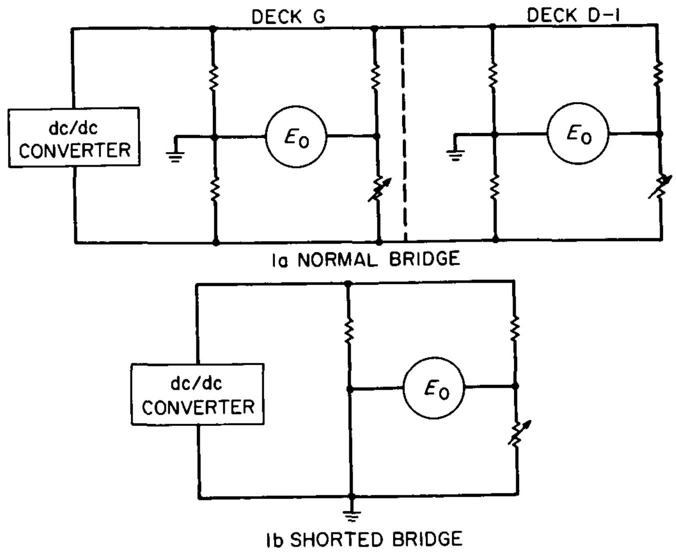


Fig. 9. G and D-1 deck temperature bridges

The exact cause of the transducer malfunction cannot be determined. It is known from laboratory tests and verification on the proof test model (PTM) spacecraft that the final value of the short was less than 2.8K ohms.

Figure 10 shows plots of the indicated midcourse motor chamber temperature and the D-1 deck bridge calibration versus time. These plots were made from data taken from channel 2 analog records in order to observe the change over the entire segment. The calibration plot was made using the calibrate measurement in addition to other D-1 or G deck measurements which indicated the condition of the D-1 deck bridge during the period of failure. The curves 1 and 2 are estimated bounding curves of all curves which could be drawn between points 35 and 38 in the region where the bridge balance is rapidly changing. The dashed curve is a typical motor temperature plot taken from motor firing tests.

These curves show that the bridge unbalance began 4 sec after motor ignition and that the indicated increase in temperature was much slower than expected.

If the typical motor temperature plot is used as representing the approximate actual temperature of the motor chamber, the point of bridge unbalance will occur at a temperature of 1100 to 1200°F. This figure is in agreement with curves of the insulating properties of magnesia and alumina as well as breakdown temperatures noted on motor firing tests with this particular transducer type.

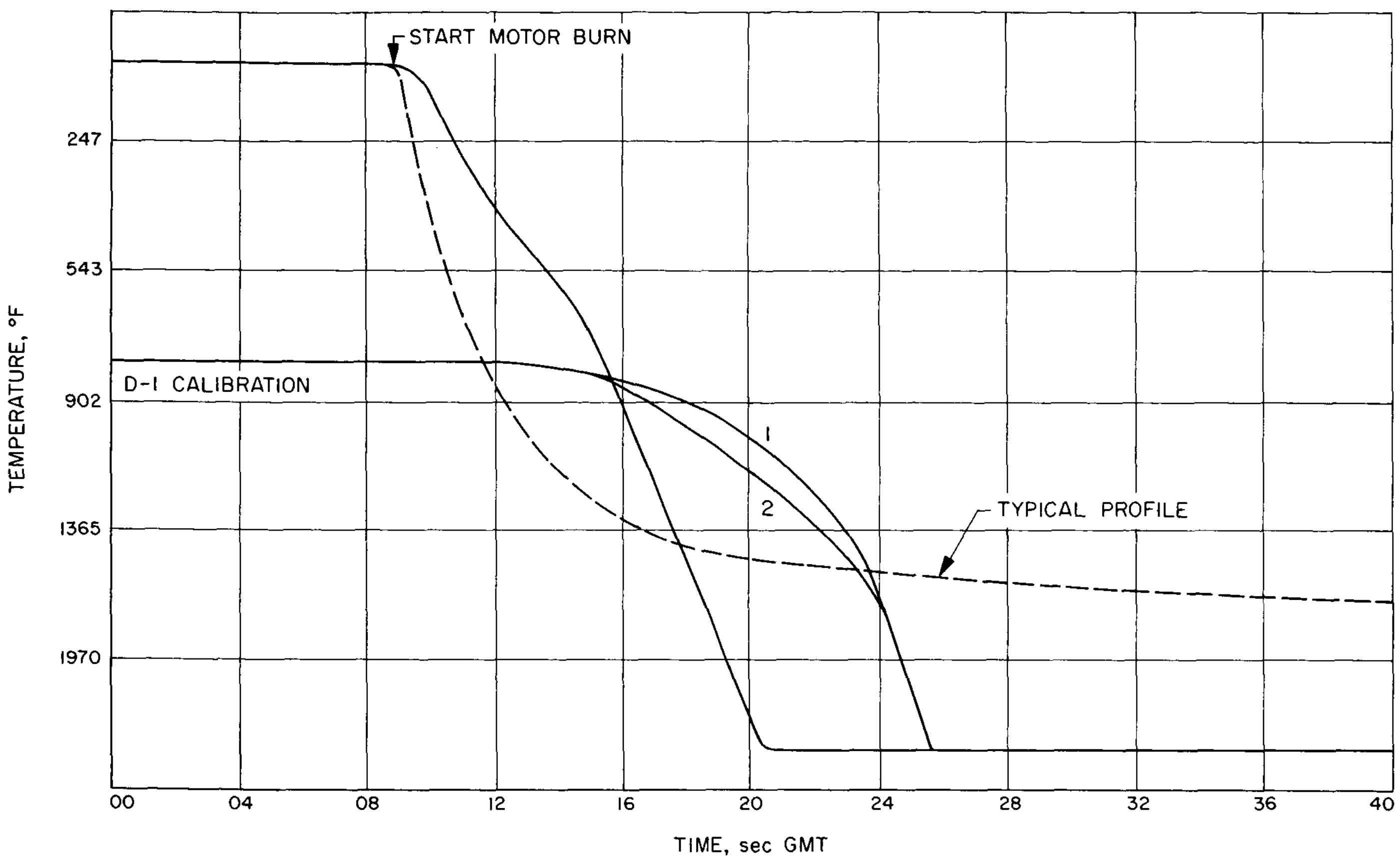


Fig. 10. Midcourse motor temperatures

A slight effect from leakage resistance was anticipated and would have resulted in no loss of data. On *Ranger VI*, however, the insulation resistance continued to decrease far below the anticipated value and remained at this low value throughout the remainder of the flight.

It is concluded from this investigation that at a time very near that of motor ignition, an insulation breakdown occurred and had two effects. The first was a shunting of the resistance element of the transducer, producing a lower-than-actual temperature indication. The second created a low resistance path from one or both sides of the temperature transducer to the spacecraft frame resulting in severe unbalance of both the D-1 and G deck temperature bridges.

4. J-Deck Frequency Shifts

During the early phases of *Ranger VI* flight small frequency shifts were present on all measurements made by the J-deck commutator. All but one of these shifts were caused by radio frequency interference (RFI) affecting the J-deck dc amplifier within the encoder. The source of the interference was the *Agena* telemetry transmitter. This effect has been experienced during testing of the *Ranger* spacecraft at times when transmission from the *Agena* telemetry transmitter is simulated.

These tests have also shown that the shielding afforded by the aerodynamic shroud is sufficient to prevent the encoder dc amplifiers from being affected by the *Agena* signal. This was also the case in the *Ranger VI* flight. Prior to shroud ejection there were no distinctive J-deck frequency shifts.

After separation of the spacecraft from the *Agena* vehicle, the J-deck measurements were free of RFI effects.

The one small frequency shift noted on address 91 occurred 30 sec after booster staging. The appearance of this frequency transient was much different from the shifts caused by *Agena* RFI and occurred prior to shroud ejection. At the time of this frequency shift a noticeable change occurred in the analog trace of the 15-point TV telemetry on channel 8. No definite cause for this shift has been determined.

All J-deck frequency shifts noted in the *Ranger VI* flight were of little consequence to the accuracy of the encoder. The maximum amplitude of the frequency shifts on the J-deck readings was approximately 2% of full scale. This amount of error is very small compared to the required accuracy of the temperature measurements during the injection phase of the flight.

D. Central Computer and Sequencer

The *Ranger VI* central computer and sequencer (CC&S) is designed to perform the following functions:

1. Provide other spacecraft subsystems with frequency references.
2. Issue launch, midcourse, and terminal maneuver sequence commands.
3. Receive and store midcourse and terminal maneuver command magnitudes.
4. Sequence and control the sense (direction) and duration of the midcourse and terminal maneuver roll, pitch, and yaw turns in accordance with the maneuver commands received and stored.
5. Control the midcourse motor burn by comparison of velocity increments received during the midcourse motor burn with the stored velocity increment command.

The CC&S achieved an exceptional record of performance during the *Ranger VI* mission. The telemetry data indicated that no malfunction nor abnormal condition occurred during the mission. Telemetry data from other subsystems using CC&S functions confirmed this. All preflight tests were uneventful in that no problems occurred. All launch phase events were verified to have taken place at their nominal times. The midcourse-maneuver functions of the CC&S were exercised and confirmed to be in agreement with the stored commands transmitted to the spacecraft. Fixed time commands during the midcourse phase were all nominal. The flight trajectory and attitude of the spacecraft were sufficiently corrected by the midcourse maneuver so that a terminal maneuver was unnecessary; therefore, terminal-maneuver functions of the CC&S were not activated and no events occurred. The CC&S power-inverter temperature monitor supplied case I temperature information throughout the mission. The temperature range was from 84°F during prelaunch to 115°F during the midcourse-maneuver phase.

The CC&S central clock's timing accuracy was much better than specified by the functional requirements. The clock error for the 65.5 hr mission was less than 0.001% as compared to the required ±0.01%.

1. Prelaunch and Launch-Phase Operations

The CC&S, upon arrival at Cape Kennedy, had a subsystem checkout before being integrated with the other subsystems. The test results indicated CC&S functions were normal, except the telemode change pulse (TM 3) was about 2 millisec wider than the maximum value of 3 millisec specified. The data encoder subsystem is compatible with a wider pulse and an examination of the mode change pulse of other CC&S units revealed widths to 6 millisec. An engineering change requirement was generated to formalize the 6-millisec pulse width.

A complete system test was conducted with no problem areas involving the CC&S. The spacecraft was moved to the launching pad where it was mated to the *Agena* and launch vehicle. The on-pad system test (J-FACT) was run with no difficulties experienced by the CC&S. A second and final hangar system test resulted in no CC&S problems. It is interesting to note that, during system tests at Pasadena and Cape Kennedy, the CC&S central clock one-pulse-per-sec frequency never deviated from WWV signals.

The prelaunch operation of the CC&S was normal. After power turnon, the H9 1000-sec sync signal on channel B19 was monitored to verify timing accuracy.

The events and corresponding times of the launch sequence are given in Appendix A. The CC&S performed within its tolerances in commanding transmitter power-up, solar panel deployment, and the acquisition operations.

The first cruise after the launch sequence was uneventful with attention primarily given to case I temperature and clock accuracy.

2. Midcourse Phase

The transmission of stored commands to the spacecraft to load the CC&S shift registers with the proper roll, pitch, and velocity information began 1 hr and 6 min before initiating the start of the midcourse phase. The final computer midcourse command generation program specified the magnitude and polarity of the roll and pitch turns and also the magnitude of the velocity increment. Turn magnitude (in deg) is transformed to time (in sec), and the velocity magnitude (in m/sec) is changed to the total number of accelerometer pulses needed to achieve the desired velocity.

The stored commands transmitted for roll (SC-1), pitch (SC-2), and velocity (SC-3) are shown in Table 5 with the values for the maneuvers included in the table. Each command contains a five-bit address followed by eleven bits for magnitude and a polarity bit, a *one* for a positive turn or a *zero* for a negative turn. The eleven bits of magnitude is a specially coded binary word unique to the CC&S shift register and is referenced to a code table which translates the 2047 combinations to decimal form.

The receipt of the stored commands by the command decoder was verified by telemetry data on analog channel B20. The CC&S issues a capacitor-cycling pulse to the attitude-control subsystem on the third bit of the SC-1 address. The presence of this pulse was verified by a B-2-1 event coincident with the third address bit.

The midcourse maneuver was initiated by RTC-4, at 08.30:00 GMT, January 31, 1964. Midcourse events were verified by B-2-1 event blips on the analog record. Because the B-2-1 events are only an indication of CC&S relay closures or openings, further verification had to be obtained from other subsystems indicating CC&S commands were actually received. To verify that the correct number of accelerometer pulses were counted down by the CC&S velocity register, a counter was connected to the channel 8 telemetry analog output. The counter indicated a total of 676 pulses during motor burn. The 676 pulses were also verified on the analog recorder. The stored command (SC) intervals were timed by stopwatch and were in agreement with the analog record as well as the predicted times.

The case I CC&S inverter temperature started to rise, from an apparently stabilized 107°F at the start of motor burn, to a high of 115°F 2 hr later. The temperature has been attributed to the heat generated by the midcourse motor burning and the 2-hr delay in reaching the apex was due to the thermal resistance of the structure and other hardware. After reaching the 115°F temperature at L + 19 hr, the temperature started decreasing slowly until it reached 111°F at $L + 32$ hr and remained at this temperature for 31 hr ($L + 63$). From $L + 63$ until impact ($L + 65$ hr, 30 min), a slow rise of temperature was indicated and reached a maximum of 113°F on the last reading 6 min before impact. The increasing temperature during the last 2½ hr of the mission is believed to have been caused by the Sun being reflected from the lunar surface.

The final event in the midcourse sequence was the last sequencing function the CC&S performed on the *Ranger VI* mission, except for the frequency references from the central clock supplied to other subsystems. From the end of midcourse until impact, the CC&S frequency reference was constantly monitored by comparing the H9 sync signal on channel B19 with the predicted computer printout times. The 1000 sec intervals for the H9 sync were printed before launch by the computer and were compared with the actual H9 sync times as indicated on the Brush recorder. The time difference between predicted and actual times was never greater than 1 sec for the entire mission.

The case I temperature monitor indicated temperatures were running slightly higher than were expected but were well within a safe operating range. The space simulation temperature tests on the proof test model spacecraft revealed CC&S temperatures were on an average 5°F lower than the flight monitor point with the exception of the transformer rectifier subassembly which was 2½–4°F warmer than the monitor point. It would be desirable to have the temperatures lowered by about 5°F on future *Ranger* missions.

Table 5. Stored commands for roll, pitch, and velocity

Maneuver	Magnitude	Seconds	Polarity	Address	Word
SC-1 roll	12.01 deg	54	negative	10101	00100101000-0
SC-2 pitch	71.06 deg	328	negative	11101	00011111010-0
SC-3 velocity	41.1605 m/sec	1351 pulses (67.5 sec.nom.)		00011	10110001000-1[a]

[a]The polarity bit on SC-3 has no meaning, but is always a one to facilitate reading the analog record.

E. Power Subsystem

The *Ranger VI* power subsystem was designed to perform the following functions:

1. Generate, store, and convert electrical power necessary for operation of all *Ranger* spacecraft subsystems with the exception of the television subsystem.
2. Perform the proper switching, conditioning, and control of the electrical power as required by the *Ranger* spacecraft subsystems.

The *Ranger* power subsystem (Fig. 11) consists of two basic groups: a power-generating system and a series of local conversion units. The raw power is supplied by two photovoltaic-cell solar panels and two primary batteries in conjunction with suitable switching, logic, and booster-regulator elements to derive a basic direct-current supply. Specialized power-conversion equipment furnishes power to the user subsystems in the following forms:

1. 2.4-kc single-phase square wave.
2. 400-cps single-phase sine wave.
3. 400-cps three-phase sine wave.
4. Regulated direct current power.
5. Unregulated direct current power.

Tests conducted at the Atlantic Missile Range employed the solar panels under artificial illumination for furnishing system power. With the exception of a performance demonstration with the *Ranger* proof test model, this marked the first time such an operation had been conducted. The test involved the electrical cabling of the solar panels to the spacecraft and the illumination of the panels with the newly designed solar-panel "exciter." The exciter consists of a bank of twenty 500-w sealed-beam tungsten lamps and their associated power control units. After final spacecraft system testing, this test was repeated with the solar panels assembled to the spacecraft and electrically connected for flight. The 100% continuity of the electrical interconnections between the solar panels and the spacecraft ring harness, including redundant paths, was verified in the final checkout.

All elements of the *Ranger VI* power subsystem performed normally within design specification and performance limits throughout the mission. Although there were a few instances where flight telemetry data did not show coincidence with preflight conditions, it has been shown that either the preflight predictions were in error or that the flight telemetry data was incorrect.

The power subsystem raw voltage, current, and power data for the *Ranger VI* mission are shown in Fig. 12. The

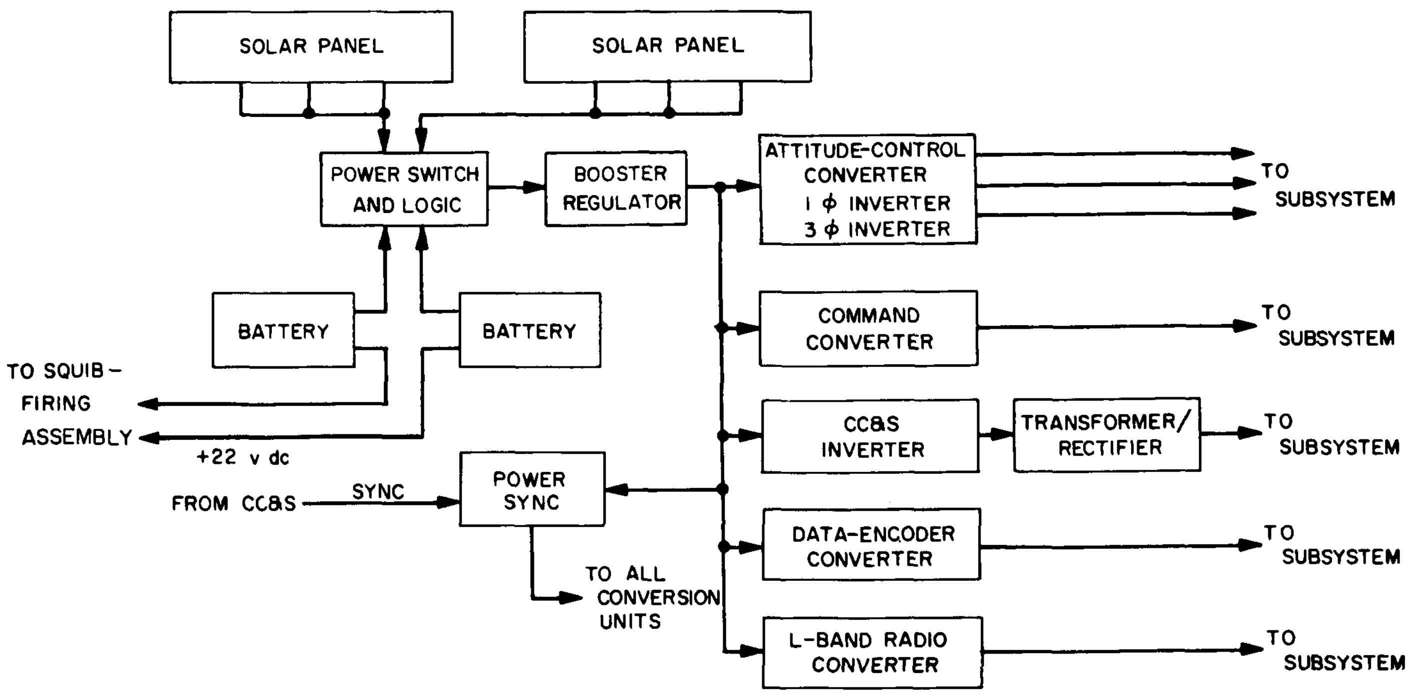


Fig. 11. Power subsystem

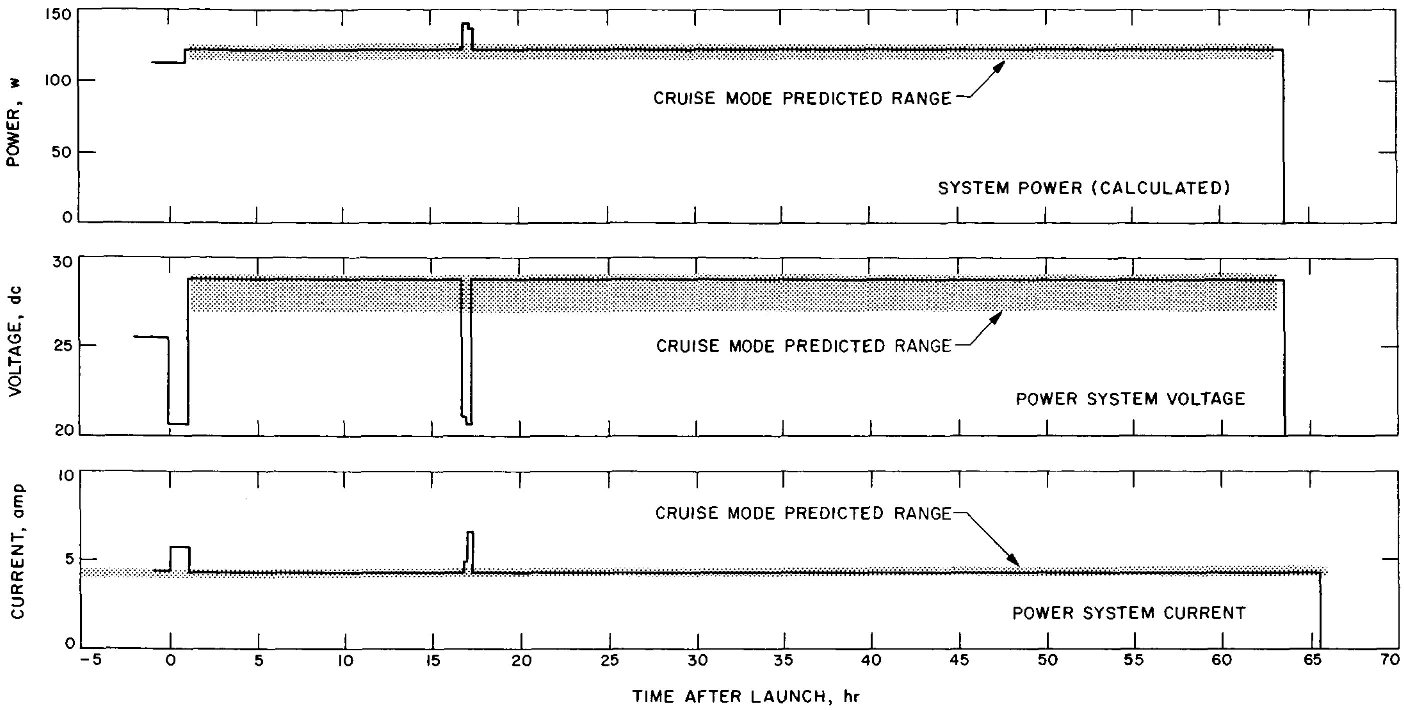


Fig. 12. System voltage, current and power — *Ranger VI* flight

solar panel currents and voltages are shown in Fig. 13. The battery voltages are shown in Figs. 14 and 15.

The spacecraft was operated on external power for preflight checkout and was switched to internal power approximately 5 min prior to launch. During the preflight period the power subsystem operated well within tolerance limits and close to the predicted levels:

1. System power: Predicted 110 w, limits 100 to 115 w, actual measured levels 110 to 111 w.
2. System voltage: Predicted 26 v, limits 25 to 27 v, actual measured levels 25.5 v.
3. System current: Predicted 4.2 amp, limits 3.8 to 4.6 amp, actual measured level 4.1 to 4.3 amp.

The subsystem was switched to internal power at 1543:50 GMT ($L - 5$ min) with the batteries as the only power source. The battery voltages of Figs. 14 and 15 show a corresponding drop from the open circuit voltage levels to the loaded voltage levels.

At 1554:15 GMT ($L+5$ min) shroud ejection occurred and an increase in the solar panel voltages from 16.1 and 16.6 v to 21.5 and 21.6 v for the $+X$ and $-X$ panels was observed.

Shroud ejection exposes the folded solar panels to the relatively low ambient light conditions. The illumination is sufficient to provide a voltage indication slightly less than the system voltage.

The CC&S initiated solar panel extension at 1649:02 GMT and panel extension was verified 64 sec later. The solar panels attained sufficient Sun orientation to supply spacecraft power at 1653:54 GMT (112 sec after start of Sun acquisition).

During the premidcourse maneuver cruise phase, at approximately $L+7$ hr, the battery voltages exceeded their preflight prediction for the upper voltage limit of 25.5 v. An analysis disclosed the method of determining this limit to be incorrect. During the tests from which the preflight predictions were derived, the operational support equipment (OSE) is connected to the spacecraft, and the OSE voltage monitoring circuit provides a sufficiently low impedance load to prevent the batteries from attaining a true open-circuit voltage, as produced during the *Ranger VI* flight.

The midcourse maneuver was initiated at 0830:00 GMT and resulted in an increase in system power consumption. During the maneuver, the spacecraft remained sufficiently Sun oriented to allow load sharing between the solar panels and batteries.

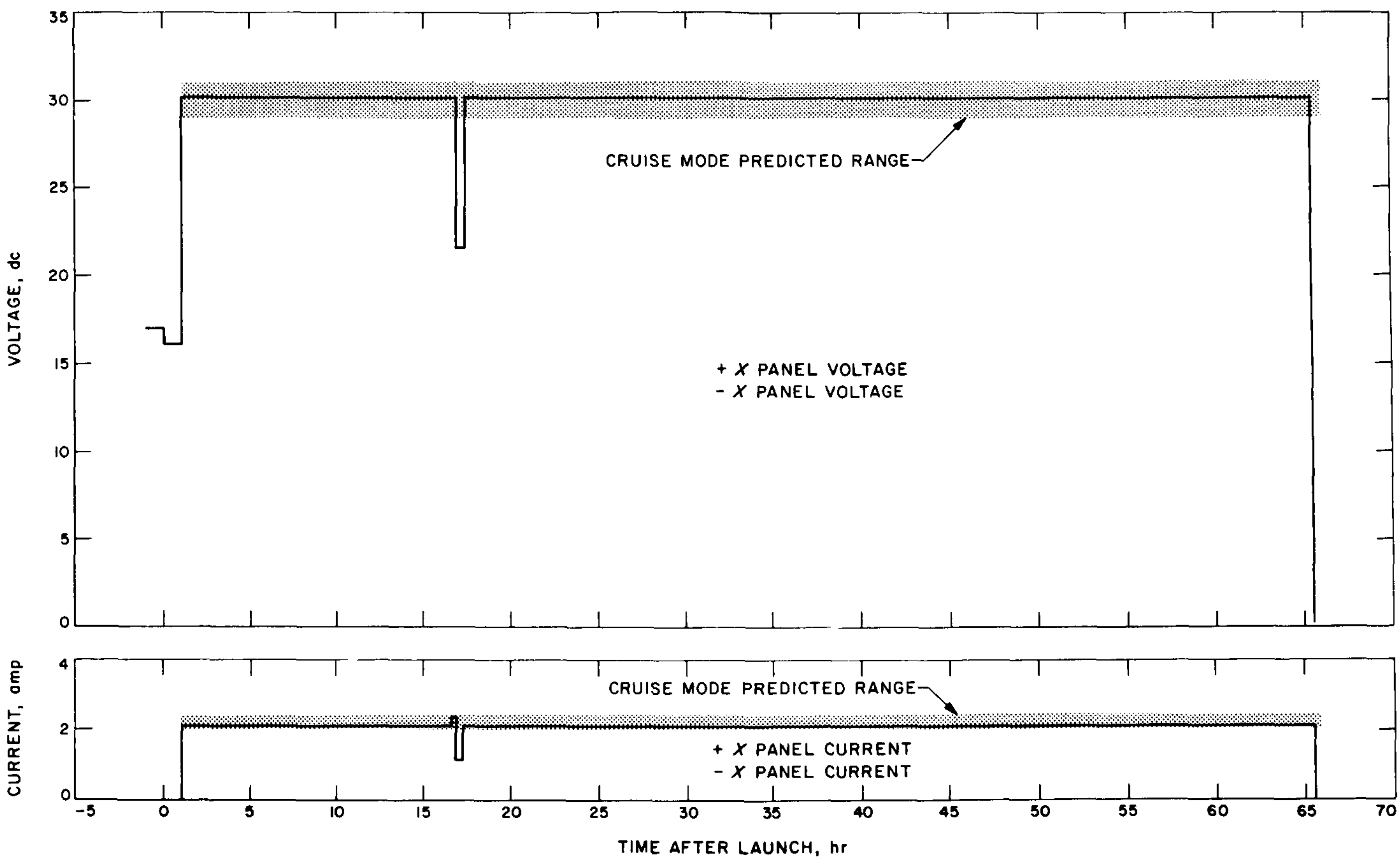


Fig. 13. Solar panel voltages and currents — *Ranger VI* flight

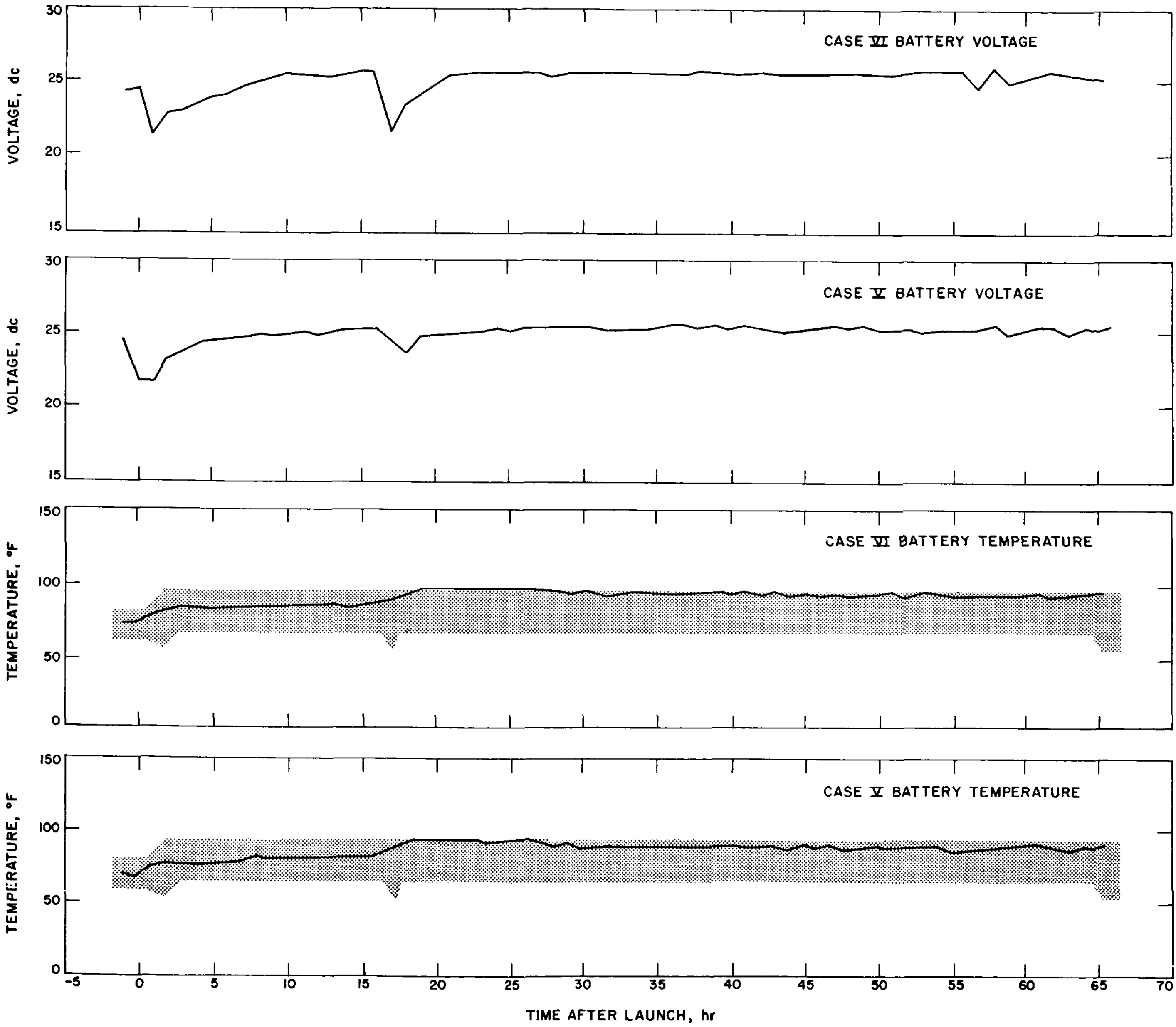


Fig. 14. Battery voltages and temperatures — *Ranger VI* flight

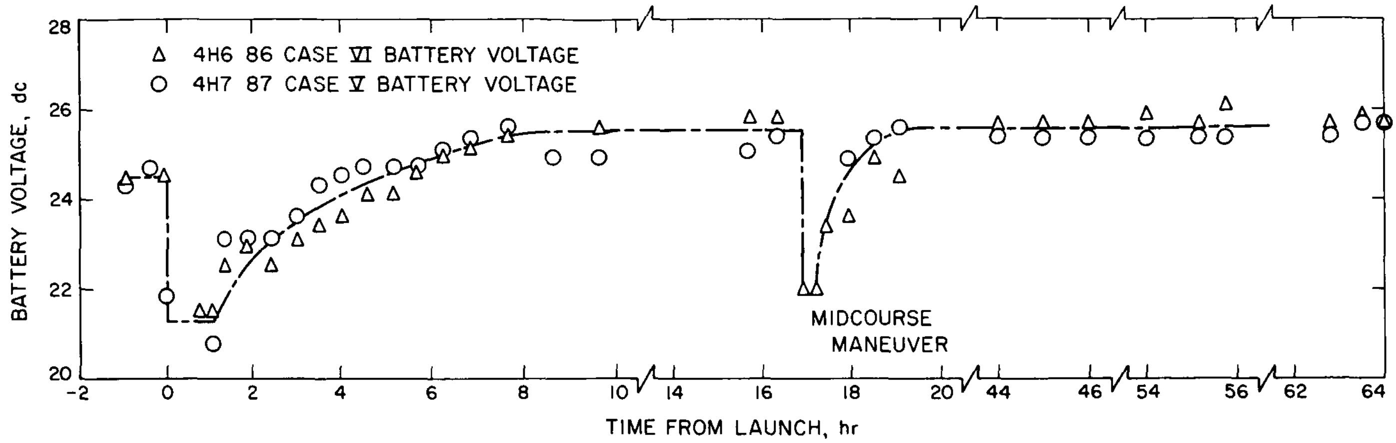


Fig. 15. Power subsystem battery voltages — *Ranger VI* flight

After postmidcourse earth reacquisition and throughout the cruise phase until lunar impact, the *Ranger VI* system power remained near preflight prediction levels. The system power averaged 121 w compared to a predicted power level of 120 w. The system voltage averaged 28.9 v compared to a predicted value of 28 v. The system current remained at 4.2 amp compared to a predicted level of 4.5 amp.

From liftoff to lunar impact the power subsystem converter elements operated at predicted levels as shown in Fig. 16. It should be noted that at liftoff, all of the power-converter monitor readings increased, as predicted, approximately 0.2 v due to the removal of the OSE monitoring circuits and the resultant reduction of the loads on the power-converter monitoring circuits.

Except for one parameter, the telemetered power subsystem temperatures remained within the predicted boundaries throughout the *Ranger VI* flight as shown in Figs. 14, 17, and 18. The high telemetry readings for the $+X$ and $-X$ solar panel front temperatures were caused by use of a nonreflecting coating on the solar-panel temperature transducers. This coating, with a higher ratio of absorptivity to emissivity than the solar cells, resulted in erroneously high temperature readings during the mission. Calibration tests performed on the spacecraft prior to flight in a 25-ft vacuum chamber and simulated solar radiation indicated that the error in the solar-panel-front temperature transducers would be a reading 16°F too high. Use of this correction factor on the telemetered temperature measurements of the solar panels, as shown in Fig. 18, show that the actual solar-panel-front temperatures were within their predicted limits.

There were no known power subsystem performance problem areas or malfunctions during the *Ranger VI* mission.

F. *Ranger VI* Attitude Control Subsystem

1. Subsystem Description

The attitude-control subsystem (Fig. 19) is designed to control the orientation of the spacecraft with respect to the Sun and Earth and to point the high-gain antenna toward the Earth. Angular orientation errors with respect to the reference bodies are sensed by optical Sun sensors and an Earth sensor. Separate Sun sensors control the pitch and yaw axes; the Earth sensor controls the roll axis and the hinge orientation of the high-gain antenna. Subsystem damping is provided by rate signals from three single-axis floated gyroscopes with torque rebalance loops.

Control torques about the spacecraft axes are produced by a cold nitrogen gas-expulsion subsystem. This subsystem utilizes four gas-expulsion jets per axis, two for the clockwise direction and two for the counterclockwise direction, for a subsystem total of twelve jets. Gas flow through individual jets is controlled by solenoid valves. Each solenoid valve pair is operated by a signal from a switching amplifier having a bistable output switch. The switching amplifier inputs are the rate and position signals for the respective vehicle axes. When the sum of the rate and position signals exceeds a predetermined value, or deadband, the amplifier actuates the pair of solenoid valves that will produce a torque opposing the direction of the rate-plus-position error. A controlled hysteresis in the bistable output switches prevents the valves from turning off until the error signal reaches a prescribed level below the turnon level. This hysteresis

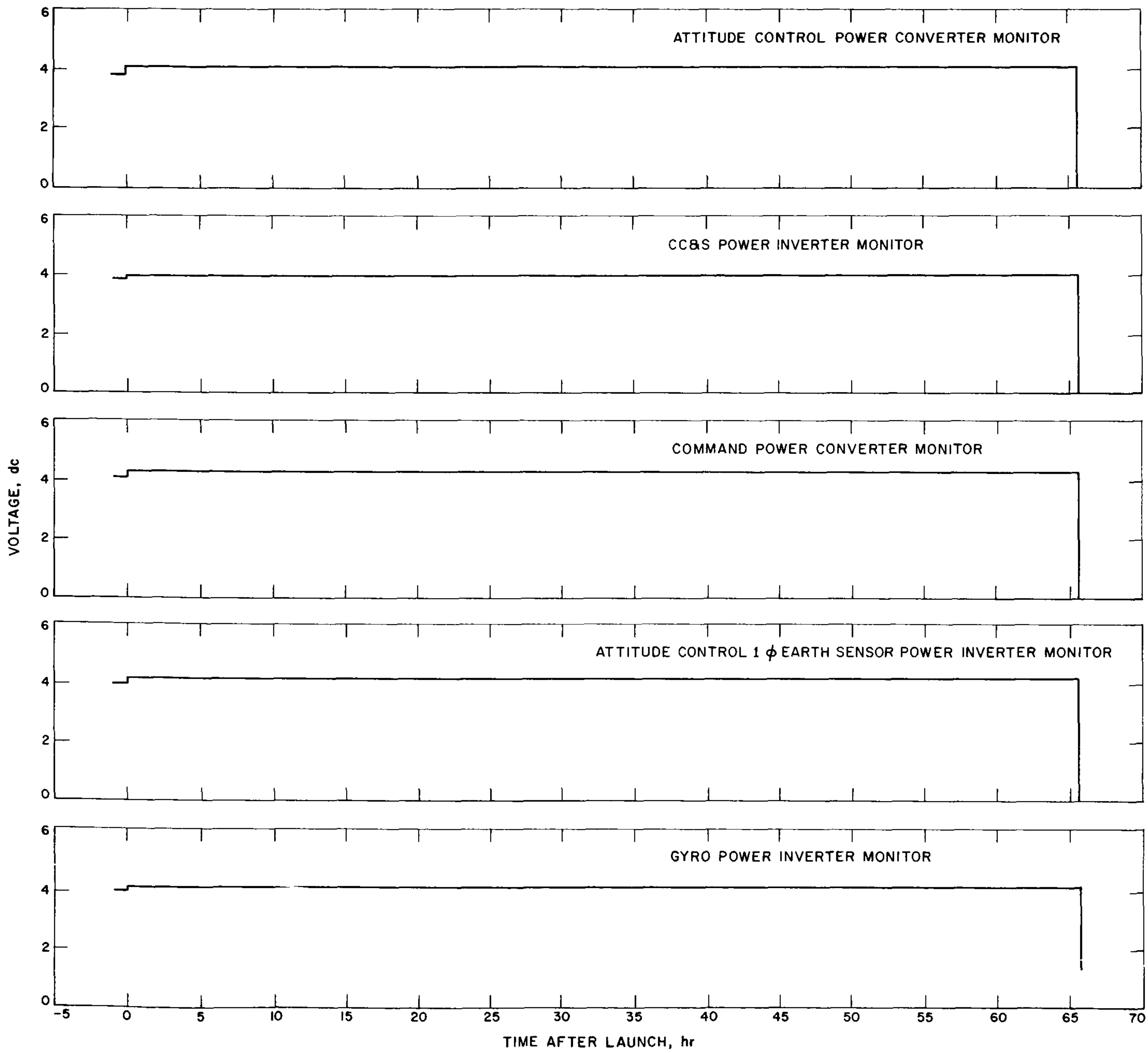


Fig. 16. Power subsystem monitor voltages — *Ranger VI* flight

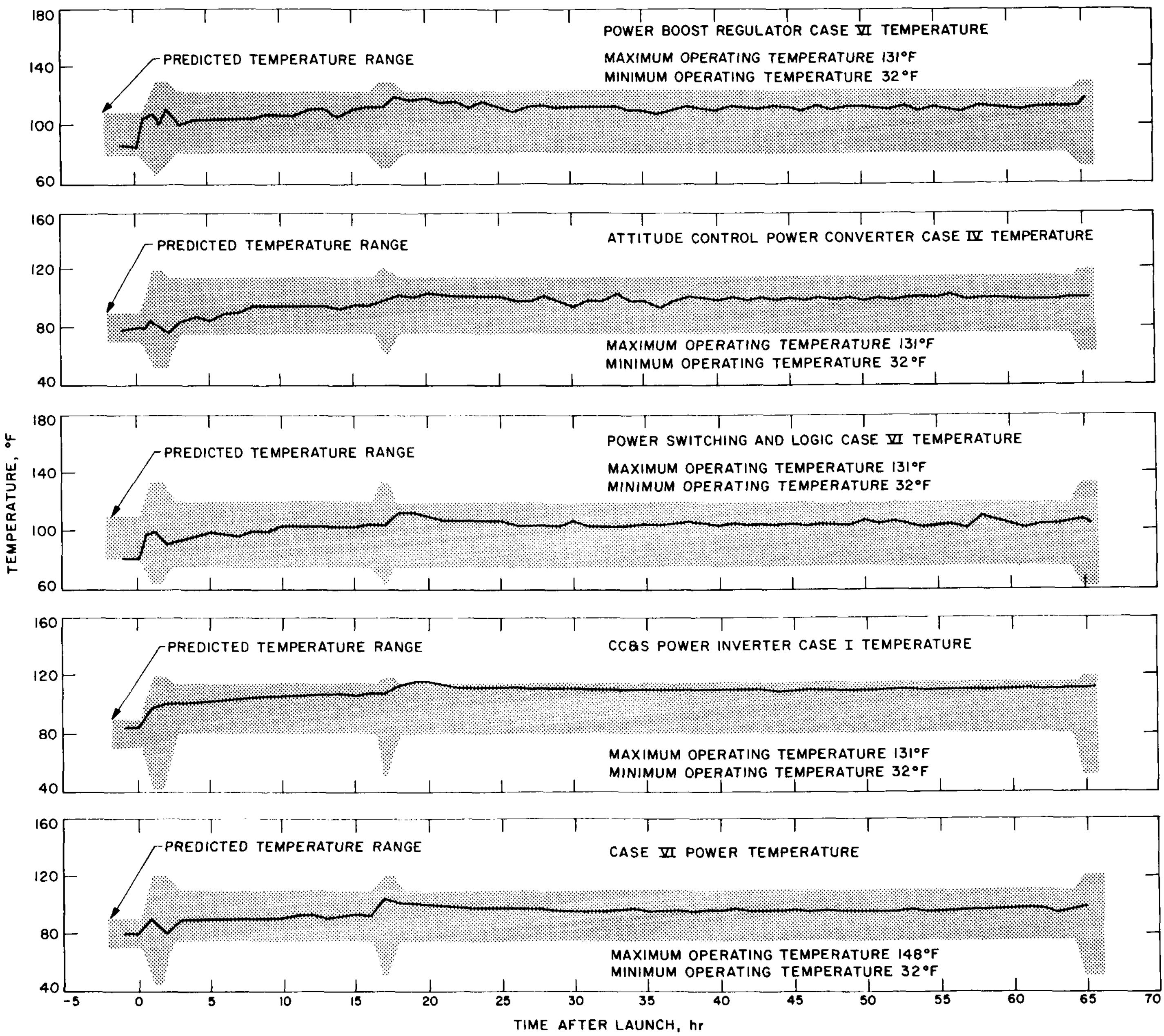


Fig. 17. Power subsystem temperatures — *Ranger VI* flight

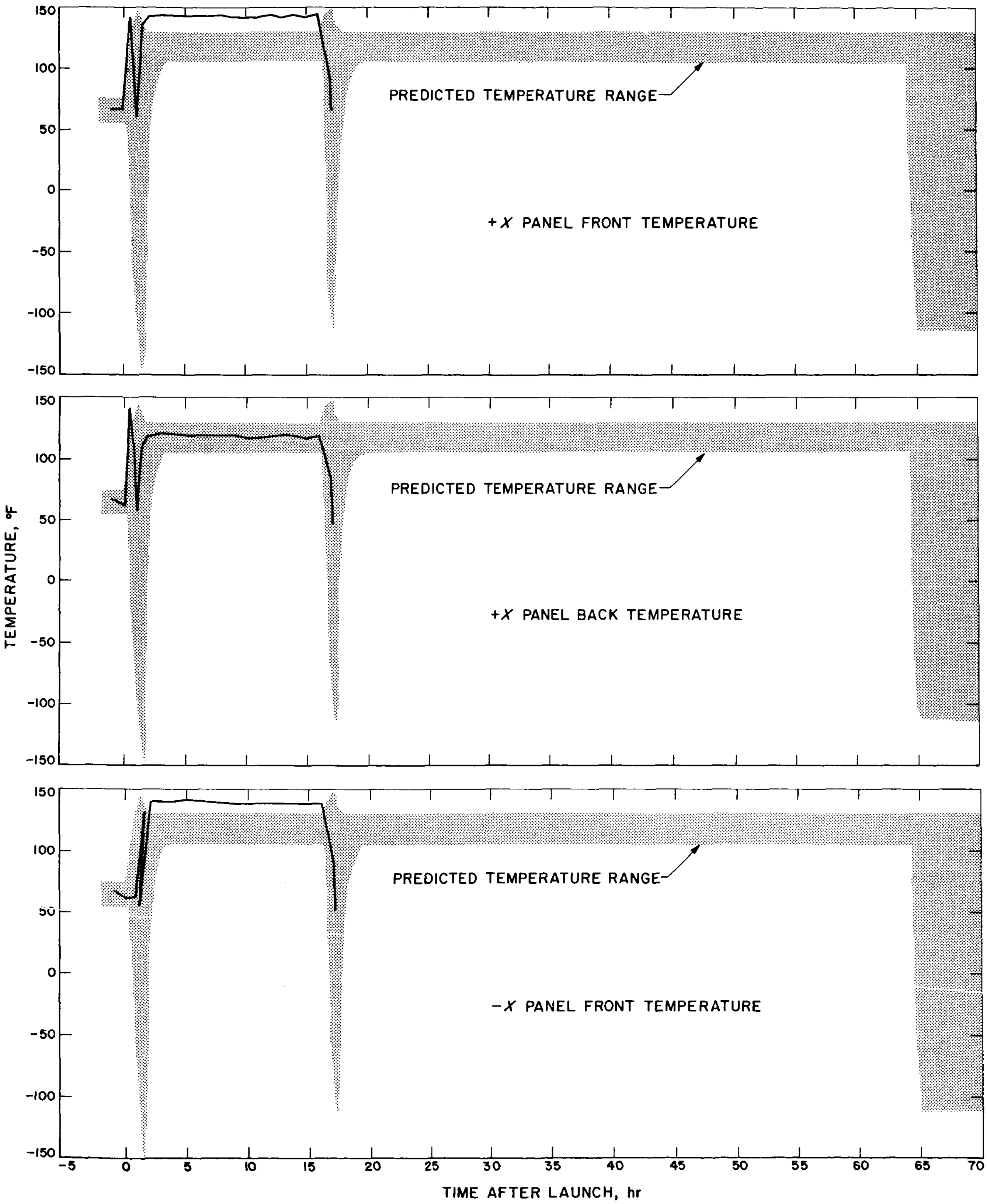


Fig. 18. Solar panel temperatures — *Ranger VI* flight

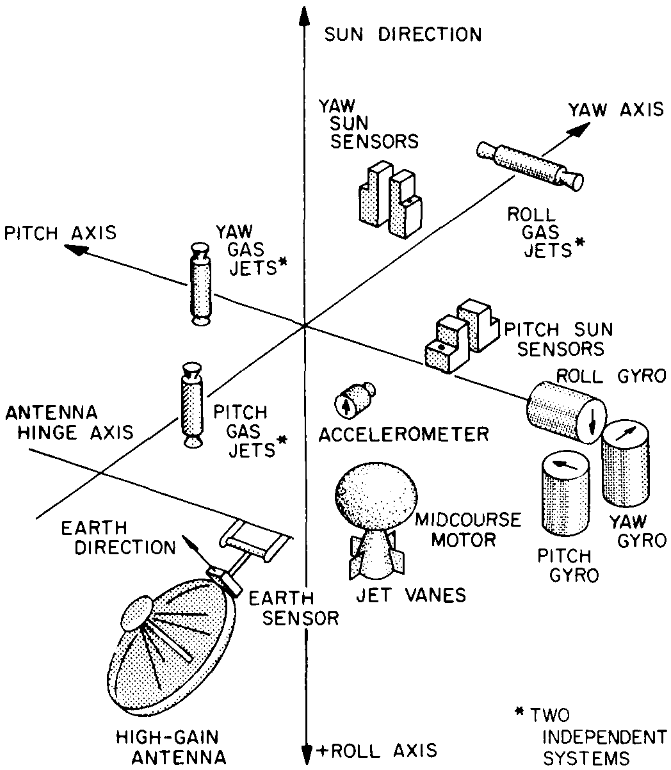


Fig. 19. Attitude control elements

establishes a minimum impulse that is imparted to the spacecraft on each valve actuation and thus prevents possible valve chatter.

Reorientation of the spacecraft for the midcourse velocity correction is accomplished by a roll/pitch turn sequence in response to commands from the CC&S. These commands disable the Sun and Earth sensor and connect a command current generator to the respective gyro torquer; the gyro senses the difference between the command rate and the actual spacecraft rate. Position error is obtained by inserting an integrating capacitor in the gyro-rebalance loop. When the turn command is removed, the gyro retains the position reference.

A midcourse autopilot maintains the spacecraft orientation during the midcourse motor firing. Spacecraft rate and position are sensed by the gyros, and control torques are produced by jet-vane control of the midcourse motor thrust.

The *Ranger* attitude control subsystem requires angular-rate and angle information about each of the three major axes: pitch, yaw, and roll. Angular-rate information is used during the acquisition and cruise periods for stabilization of the system. During the maneuver period for midcourse correction, the angle information is also required.

The gyro subassembly derives this information by a unique application of three single-degree-of-freedom, floated integrating gyros. Each gyro is filled with a low-viscosity, high-density fluid which provides full flotation at 115°F. A low-viscosity fluid has been selected which permits operation of the gyro without the use of heaters. Damping and precession axis restraint are accomplished by a torque feedback loop. The restraint provided by this electronic loop remains relatively constant, independent of changes of fluid viscosity due to temperature.

The magnitude of the velocity increment added during the midcourse maneuver for midcourse correction is measured by means of an accelerometer. This accelerometer has a pendulous, force-balance, flexure-suspended, proof-mass design. A pulse torque rebalance loop is provided by the associated electronics. The pulse is directly proportional to the applied acceleration.

The digital computer in the CC&S stores the velocity increment correction, which is transmitted from an Earth command. Upon activation of the midcourse motor, the constant acceleration of the spacecraft is measured by the accelerometer, and the digital-pulse output is matched with the stored digital information to derive the motor cutoff command. A functional diagram of the subsystem is given in Fig. 20.

2. Inflight Performance

The inflight performance of the *Ranger VI* attitude control subsystem was normal in all respects and is summarized below.

Acquisition. When the CC&S issued the Sun-acquisition command at 1652:01 GMT ($L + 63$ min), the spacecraft tumbling rates were 5.0, −9.6, and 2.6 mrad/sec in pitch, yaw, and roll, respectively. The spacecraft acquired in pitch and yaw within $3\frac{1}{3}$ min after receipt of the acquisition command. The roll rate was reduced to 0.9 mrad/sec within 10 sec.

Upon receipt of the Earth-acquisition command at 1920:02 GMT ($L + 211$ min), the spacecraft was accelerated to a roll-search rate of −3.8 mrad/sec.

After a 20-min, 40-sec roll search, the Earth sensor detected Earth light, and it completed the acquisition 25 min after receipt of the command. It is estimated that the spacecraft rolled through a 270-deg angle in the roll search.

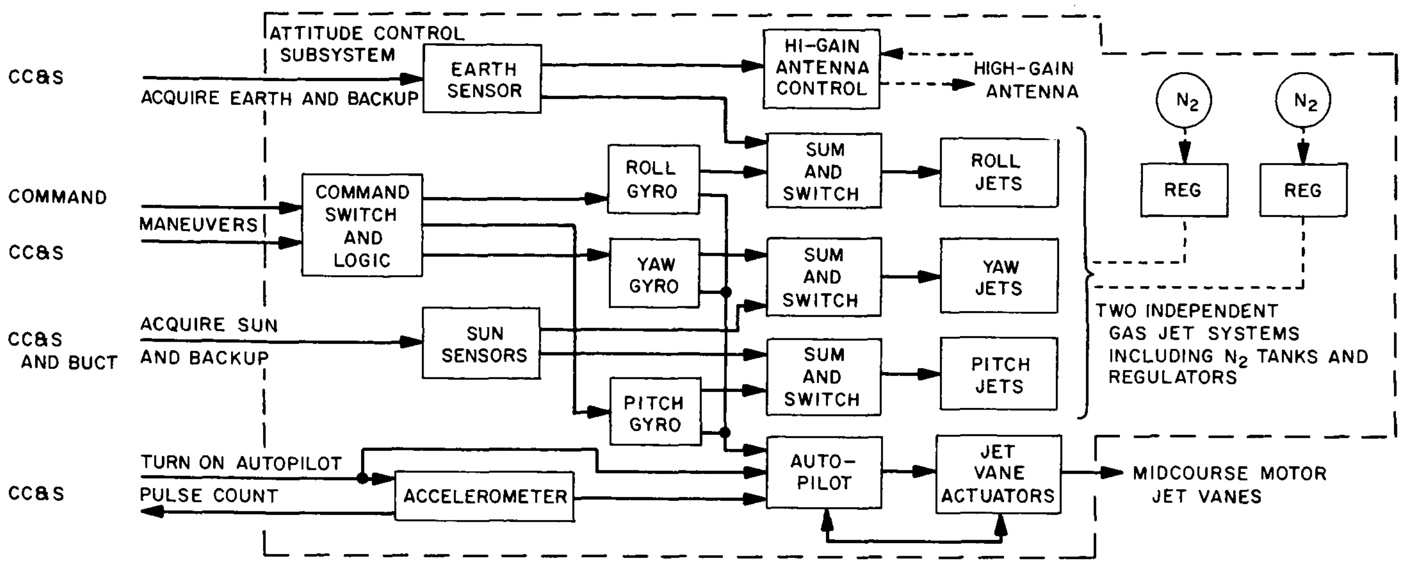


Fig. 20. Attitude control subsystem

The Earth sensor measured Earth brightness consistently 2 to 4 times the expected value (Fig. 21). This is accounted for by the difference in color temperature between the calibration source and the Earth, and by the accuracy of the calibration meters. There is also an uncertainty in the predicted Earth brightness because of variation of the Earth's albedo.

Cruise phases. The attitude-control subsystem stabilized the spacecraft position within the nominal $\pm$ 2.8-mrad deadband in pitch and yaw. Roll position stabilization was within the normal deadband limits, which vary with the distance from Earth. The limit cycle velocity increment was within the design maximum of 90 mrad/sec. An external pitch torque of -170 dyne-cm was determined from the limit cycle data. No steady external torques were evident from the roll and yaw data.

The hinge subsystem tracked the Earth within the 1.25-deg deadband of the hinge-angle servo during cruise operation. During Sun acquisition the antenna was extended to the preset angle of 135 deg inserted before launch. During the midcourse maneuver sequence, the antenna moved in a normal manner to the 180-deg exit angle and then to the 122-deg reacquire angle at Sun reacquisition.

The spacecraft acceleration rate provided by the gas subsystem was within the specification range of 0.54 to 0.66 mrad/sec^2. Based on the observed 60- to 90-mrad/sec velocity increment, the gas consumption is estimated to be 0.1 to 0.4 lb, which is insignificant compared to the 1.15-lb calculated worst-case value. No measurement of the actual consumption is possible due to the 0.08-lb resolution of the telemetered measurements and the resolution of the gas reservoir temperature measurements.

A comparison of pitch and yaw limit cycle increment distribution for the *Ranger VI* and earlier spacecraft is shown in Table 6. It is of interest to note the close agreement of the mean values for *Rangers III, V,* and *VI.*

The operating temperatures of the attitude control subsystem elements were within the operating limits at all times during the *Ranger VI* mission. The $-Y$ Sun sensor temperature exceeded the upper predicted value by approximately 2°F but is not considered a design or problem area. The Earth sensor temperature exceeded its upper prediction limit by 11°F and is to be the subject of further analysis and possible design modification although it was below its upper operating limit at all times.

Table 6. Pitch and yaw values for Block III Ranger flights

Spacecraft	Limit cycle velocity Yaw		Increment-microradian/second Pitch	
	mean	σ	mean	σ
Ranger VI flight	26.9	8.5	41.5	9.5
Ranger V flight	37.9	12.6	30.8	12.2
Ranger III flight	26.9	5.0		
Ranger III simulation	27.2	11.5		

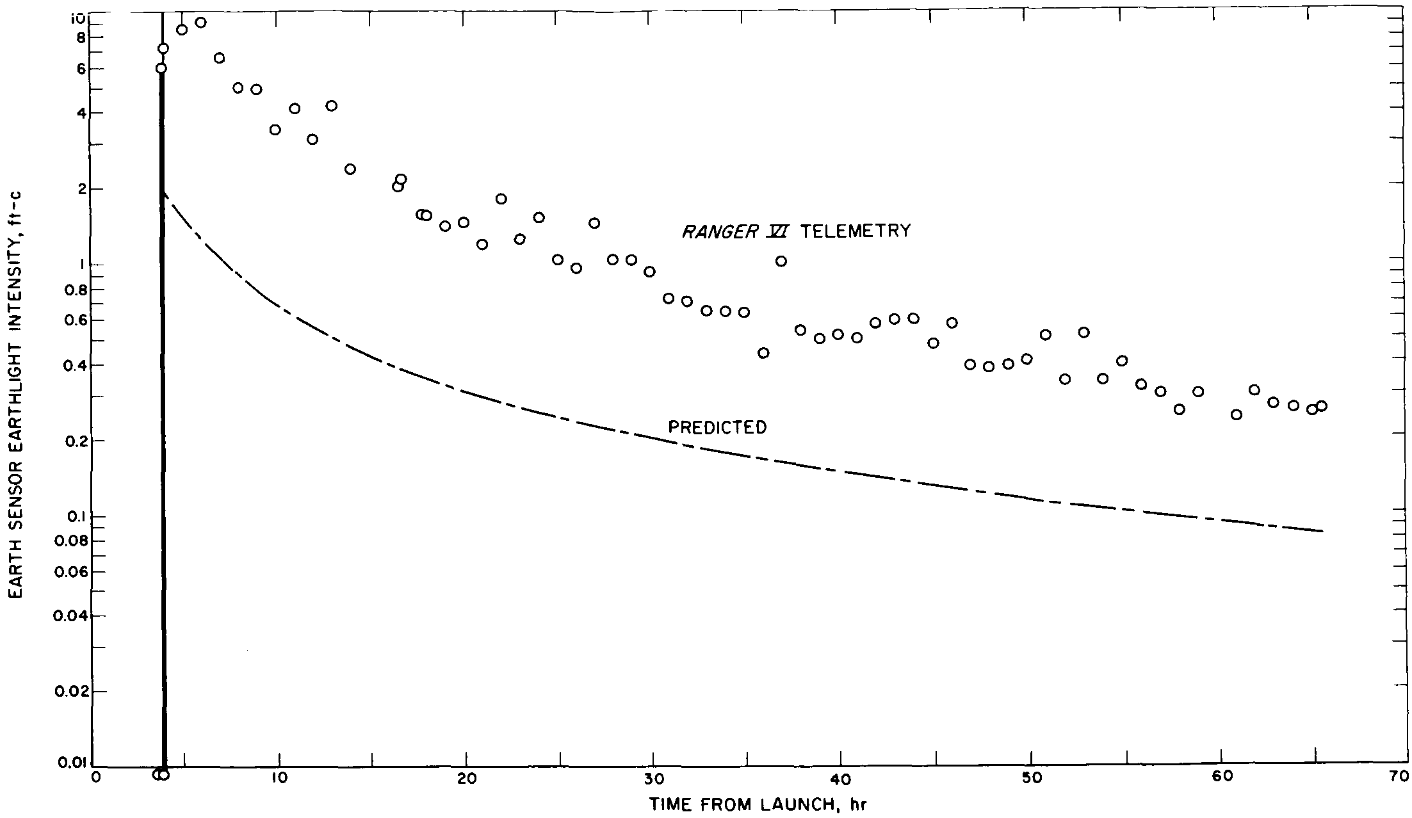


Fig. 21. Earth sensor Earthlight intensity — *Ranger VI* flight

Midcourse maneuver. The midcourse maneuver required for the trajectory correction consisted of a 54-sec negative polarity roll turn and a 328-sec negative polarity pitch turn. The limit cycle positions at the start of the roll turn were: roll, +3.8 mrad; pitch, +0.5 mrad, and yaw, +2.0 mrad. Both turns were executed in a normal manner, and the rate transients compared well with the expected values. The predominant source of error in the midcourse maneuver is the initial roll limit cycle position. Since this initial condition was small compared to its maximum predicted value of 11.3 mrad, a very small midcourse point error resulted.

The autopilot transients compared well with predicted values from an analog computer study.

Reacquisition. Upon receipt of the Sun-reacquisition command, the spacecraft accelerated to a pitch rate of 4.9 mrad/sec and reacquired the Sun in 5 min. The yaw rate and position remained at null during the reacquisition.

At the end of the midcourse-motor burn, the residual roll rate was 100 mrad/sec. This rate reduced the roll position error relative to the Earth from 114 deg (introduced by the midcourse roll turn) to approximately −1 deg at the time of Earth reacquisition, so that the Earth sensor detected the Earth as soon as it was turned on, and the acquisition was completed within 80 sec.

G. Midcourse Propulsion Subsystem

The midcourse propulsion subsystem (Figs. 22 and 23) is designed to provide to the spacecraft a controlled velocity increment of up to 60 m/sec for the purpose of correcting trajectory errors which normally result from spacecraft injection guidance errors. A small, monopropellant-hydrazine propulsion subsystem delivers a 50-lb thrust to the spacecraft. The propulsion subsystem is capable of delivering a variable total impulse in response to signals from an integrating accelerometer circuit. It is functionally a regulated-gas-pressure-fed constant-thrust rocket. Principal subsystem components consist of a high-pressure gas reservoir, a gas pressure regulator, a propellant tank and bladder, a rocket engine, and an ignition cartridge containing a small amount of nitrogen tetroxide to initiate propellant decomposition.

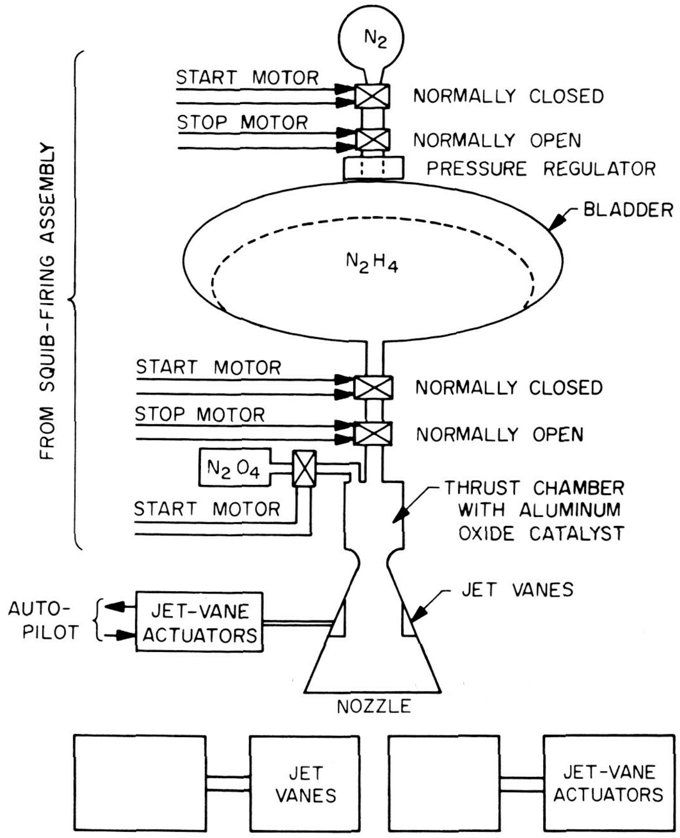


Fig. 22. Midcourse propulsion subsystem

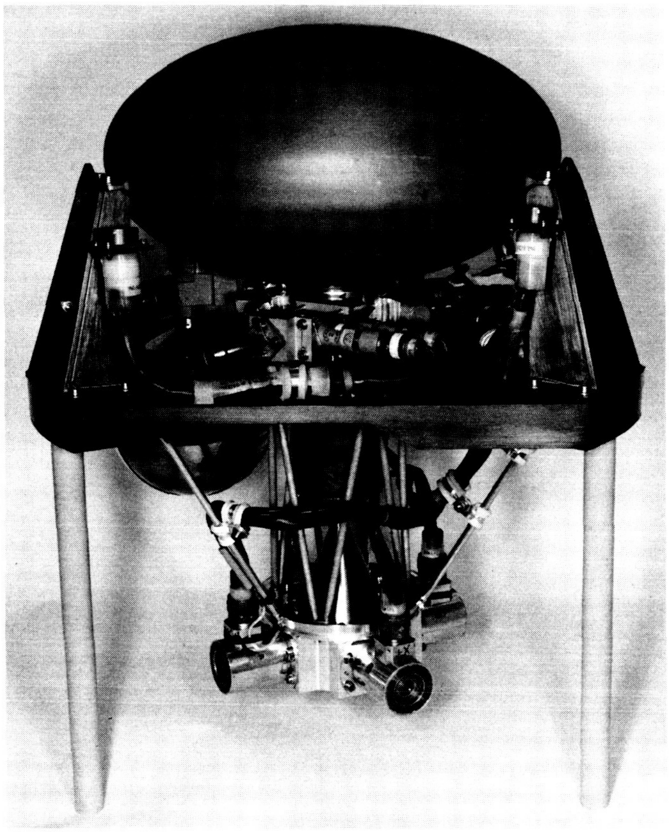


Fig. 23. Midcourse motor

The flight performance of the propulsion subsystem was normal. The premidcourse normalized pressures were 3152 psia in the high-pressure nitrogen reservoir and 255 psia in the fuel tank. These pressures remained constant until the motor-burn period, indicating no leakage from either tank. The calculated maximum velocity increment capability was 60.17 m/sec, with a tailoff velocity increment of 0.0925 m/sec. A normal motor burn was accomplished at 0857:08 GMT on January 31, 1964. A velocity increment of 41.27 m/sec (135.1 ft/sec) was commanded and involved a motor-burn time of 69 sec.

Shortly after ignition of the propulsion subsystem during the *Ranger VI* midcourse maneuver, the motor chamber wall temperature transducer underwent a shift and shorted when the chamber reached steady-state temperature (see Section IV-C for further details). The temperature-time history until the shift occurred indicated that a normal ignition had been obtained. The loss of the chamber temperature transducer invalidated the nitrogen-tank-temperature measurement, since a common reference bridge had been severely unbalanced. As a result of this problem on *Ranger VI*, a design change was accomplished for subsequent spacecraft which electrically disconnects this transducer from the data encoder subsystem, thus eliminating any data from this measurement.

Two pressure measurements were available during the majority of the motor-burn period. The plot of these two parameters is presented in Fig. 24. The fuel tank pressure undergoes an abrupt change at ignition since high-pressure nitrogen gas is released to the pressure regulator at this time. Regulated pressure is normally very stable; the scattered data points are believed to be caused by instrumentation noise which is typical of the high-sampling-rate parameters. The nitrogen tank pressure-decay curve is typical of such a process with limited heat transfer occurring between the gas and its surroundings during the blowdown process. A positive shutoff was obtained at the end of motor burn; the pressures remained stable until lunar impact.

H. Spacecraft Structure

The spacecraft bus structure is a redundant truss-type design of aluminum and magnesium alloys. In configuration, it appears as a series of concentric hexagons, the

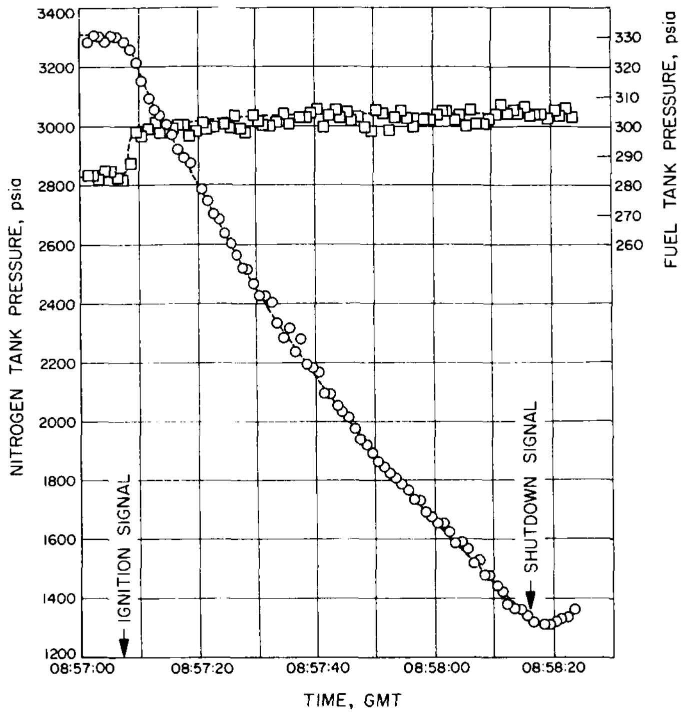


Fig. 24. Pressure measurement, motor-burn period

lower and larger of which comprises the attachment and separation plane from the *Agena* adapter. This plane, or base, is surmounted by a smaller double-walled hexagonal structure which houses and supports the electronics cases around its outer periphery, and the midcourse propulsion system within its inner periphery. Appendix C consists of drawings showing structural details. The structure provides stable reference for the Sun-sensor units, TV subsystem, directional-antenna/Earth-sensor complex, midcourse propulsion subsystem, and attitude-control subsystem.

The high-gain antenna is a parabolic dish constructed of fabricated sheet aluminum-alloy ribs emanating radially from the center and supported at midpoint and the outer diameter by sheet-metal rings. The dish surface is covered by a black anodized aluminum mesh, held in conformation by the radial ribs and the mid and outer rings. The antenna feed is mounted at the center of the concave side of the dish, supported by four fiberglass, tubular struts. The antenna is driven and maintained at the required attitude by the antenna gearbox through a yoke and arm attached to the convex side of the antenna. In its retracted position, the antenna nests in the rearward end of the spacecraft, just above the separation plane. During ascent, the antenna is protected from vibration effects by snubbing elements provided in the *Agena* adapter.

The solar-panel substrates are fabricated of spot-welded and seam-welded aluminum-alloy sheets, providing structural support and protection for the solar cells. The cells are mounted on a flat, rectangular aluminum-alloy sheet, whose flatness and rigidity are maintained by corrugated aluminum-alloy sheets in the longitudinal direction and by cross-bracing in the transverse direction. The cross braces also provide the mounting and the heat sink for the zener diodes of the power subsystem.

During the *Ranger VI* mission, the spacecraft bus structural subsystem, the high-gain antenna structure, and the solar-panel substrates all performed their functions satisfactorily from both a structural and mechanical standpoint. There were no known anomalies.

I. Temperature Control Subsystem

The purpose of the temperature control subsystem is to provide a spacecraft temperature environment that is within the operating limits for the spacecraft hardware and electronic components. This environment is to be provided for all spacecraft attitudes and conditions except when an Earth shadow or non-Sun-oriented attitude of greater than 60-min duration is experienced by the spacecraft. Violation of these constraints on the trajectory will, in general, result in an overcooling of some components and, perhaps, an overheating of others.

The thermal control subsystem on *Ranger* Block III is a passive system with no moving parts. Through precise regulation of the radiating characteristics of the external surfaces, the relationship between the input energy and the energy lost to space is regulated to yield the proper spacecraft temperatures.

The philosophical approach taken on *Ranger* is the result of several years of flight and test experience with similar spacecraft criteria, i.e., Sun-oriented, constant-power dissipation, and short-transit-time flight profile.

One of the major problem areas encountered has been the effects of reflected solar energy. The *Ranger* Block III design is aimed at eliminating or minimizing solar reflections as much as possible. Black paint or black cloth are used on surfaces which might reflect to the spacecraft; polished surfaces are used where practical to direct the reflected energy away from the spacecraft.

There are uncertainties and unaccountable variations both in directly absorbed solar energy and in internal power dissipation. Since these are the only sources of input energy, an attempt is made to have each source account for approximately half of the total. In this way, uncertainties and variations in either source produce minimum variations in spacecraft temperatures.

A side product of the use of significant amounts of solar energy is that a relatively larger amount of energy must be radiated to space which, in turn, causes higher average emittance on the radiating surface. Since uncertainties in emittance measurements seem to be a fixed increment regardless of the absolute magnitude of the emittance, these uncertainties are reflected in smaller temperature uncertainties with increasing average emittance.

Another effect of a higher average emittance is the possibility of using a low absorptance-high emittance surface on the large radiating area to minimize the change in solar load when the spacecraft is in a non-Sun-oriented maneuver.

Space simulators are at best rather inexact duplications of the space environment. One area of difference is the spectrum of the solar simulator, which in general is very different from the actual spectrum of the Sun. However, by use of surfaces which exhibit similar properties regardless of spectrum, the validity of solar-simulator tests is greatly enhanced. Such surfaces are called thermally grey, and black paint is one of them. In addition, black absorbs almost all of the incident energy, leading to very little reflected solar energy. For these reasons, *Ranger* uses black paint on most of the sunlit areas where energy input and/or solar reflections are important.

Thermal shields (good insulators) are used over large areas of the spacecraft to control the amount of solar energy absorbed and to close off openings in order to help raise the remaining average emittance.

The *Ranger VI* thermal control subsystem maintained the temperatures within the operating limits at all times. Most components were within the predicted temperature bands. The Earth sensor, −Y pitch Sun sensor, the midcourse fuel tank, and the case VI battery exceeded the upper predicted temperatures by a few degrees. Part of the reason for these higher-than-predicted temperatures was the fact that the TV subsystem ran some 40°F higher than predicted, thereby creating an additional heat load on the spacecraft.

Table 7 shows the flight temperatures of the various components as received from telemetry, compared with

Table 7. *Ranger VI* flight temperature summary

Address	Function	Launch temperature, °F	Pre-midcourse temperature, °F	Post-midcourse temperature, °F	Pre-impact temperature, °F	Predicted cruise temperature limits, °F	Operating limits, °F
70	+X solar panel (front)	65	128	[a]	[a]	105-131	112-131
71	+X solar panel (back)	72	120	[a]	[a]	105-131	112-131
72	−X solar panel (front)	63	124	[a]	[a]	105-131	112-131
73	Data encoder	73	95	[a]	[a]	70-110	32-131
74	Gyro package	82	104	[a]	[a]	75-115	40-131
75	+X attitude control N_2 bottle	65	90	[a]	[a]	65-105	32-148
76	−X attitude control N_2 bottle	62	80	[a]	[a]	65-105	32-148
80	Booster regulator	90	112	122	114	80-125	32-131
81	Attitude control inverter	82	98	102	100	75-115	32-131
82	Power switch and logic	83	104	104	109	75-120	32-131
90	Earth sensor	63	85	77	88	50-75	14-94
91	+Y pitch sensor	62	78	87	83	65-90	40-140
92	−Y pitch sensor	60	91	75	94	65-90	40-140
93	Transponder	76	101	98	102	70-110	32-148
94	Case V battery	70	88	94	92	65-95	50-130
95	Midcourse fuel tank	73	88	114	101	70-95	32-148
96	Case VI battery	70	85	90	90	65-95	50-130
97	CC & S inverter	84	108	113	112	80-110	32-131
98	Case VI chassis	79	94	102	100	75-110	32-148

[a] G-temperature bridge and all of address 70's readings shorted out after this time by midcourse motor burn.

the predicted temperatures and the component temperature limits. Unfortunately, after the midcourse motor firing, no address 7X measurements were obtainable, due to shorting of the data encoder circuitry.

Ranger VI did not go through the Earth's shadow, so that there was no severe cooling of any components. However, due to the tumbling motion of the spacecraft prior to Sun acquisition some components did experience some rapid heating and cooling transients. Most notable was the +X solar panel, which went from 60°F at launch to 143°F, back down to 50°F, then up to a cruise temperature of 126°F.

Prior to the midcourse maneuver, all components except the Earth sensor and the −Y pitch Sun sensor were within the predicted temperature bands. These two items are geographically in the same area and are interrelated thermally.

Following the firing of the midcourse motor, temperatures within the bus generally rose an average of about 10°F. These temperatures gradually fell until after about 20 hr they were within 5°F of their premidcourse values. The excessively long time required for the temperatures to return to their premidcourse values was due to the unexpected heat load imposed by the hot TV subsystem. Again, however, no components were near their operating temperature limits.

For the duration of the flight, temperatures settled into their premidcourse range until the last two hours prior to impact. At this time the effects of the hot lunar surface were felt. The external components (Sun sensors and Earth sensor) and case VI all rose slightly in temperature. Because of the spacecraft attitude, case VI was the only one directly exposed to the influence of the Moon.

In summary, the performance of the thermal control subsystem was satisfactory.

J. Solar Panel Extension and Support Subsystem

The solar-panel hinge, latch-support structure, pyrotechnic pin-puller, point damper, and actuator have the two functions of supporting the solar panels against the launch-phase dynamic loads and deploying the panels on internal command after spacecraft/launch vehicle separation.

The solar-panel actuators were predicted to open within 60 to 70 sec after the B-2-1 indication of the CC&S solar-panel extension command. The B-2-1 blip occurred at 1649:02 GMT. The B-2-4 blip, indicating that the +X panel was deployed, occurred at 1650:06 (64 sec after the CC&S command).

The nominal deployment time and proper solar-panel performance throughout the flight indicate normal performance of solar-panel extension and support system during the flight.

K. Miscellaneous Timing and Arming Functions

1. Backup Command Timer (BUCT)

The BUCT is a hydraulic device which, through mechanical linkage, actuates a switch assembly to complete a common circuit. It provides spacecraft on-board command-signal redundancy for four crucial commands. The BUCT is installed between the spacecraft and the *Agena* adapter and is set into operation by the removal, at spacecraft separation, of the restraint provided by the *Agena* adapter.

The command sequencing and nominal times with respect to spacecraft/*Agena* separation are as follows:

1. Arm the squib firing assembly (SFA) at S + 2.5 min (backup).
2. Turn on channel 8 telemetry at S + 17 min (primary).
3. Deploy solar panels at S + 45 min (backup).
4. Initiate Sun acquisition at S + 60 min (backup).

These functions were successfully accomplished, presumably by primary command.

Item 2 was successfully accomplished by the BUCT at the specified time, although there was an earlier channel 8 anomaly.

2. TV Backup Clock Turnon Switch

A microswitch located at the base of the spacecraft and actuated by spacecraft/*Agena* separation, causes turnon of the TV backup clock. This function was satisfactorily performed during this mission.

L. Pyrotechnic Subsystem

The pyrotechnic subsystem (Fig. 25) consists of four squib-actuated pin pullers, three squib-actuated valves,

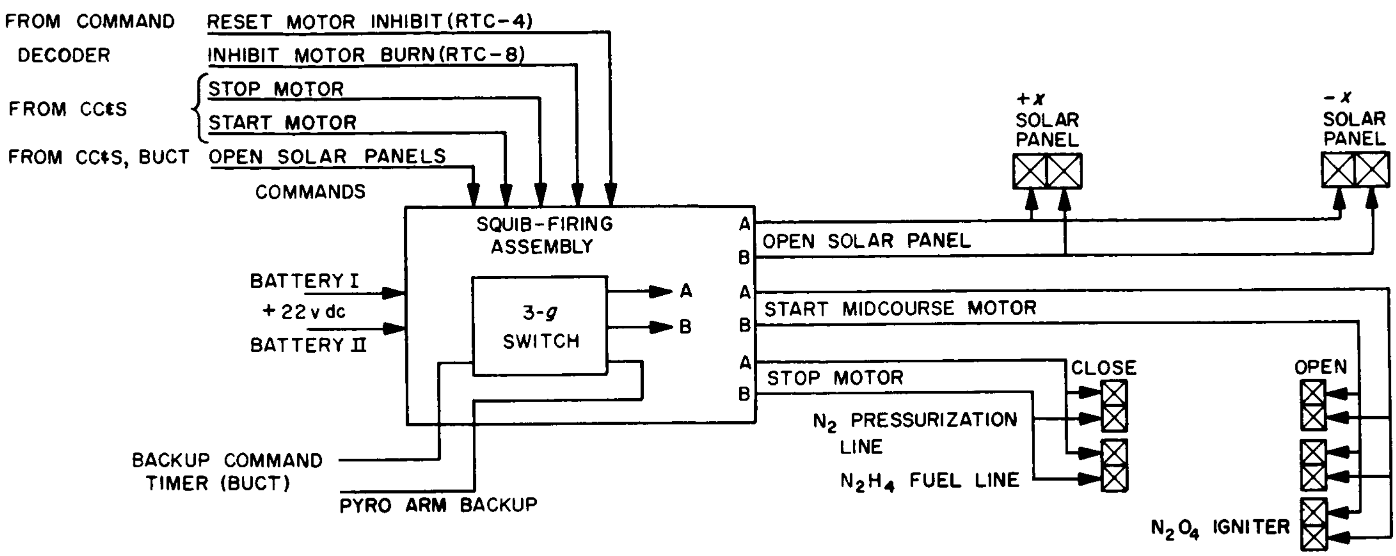


Fig. 25. Pyrotechnic subsystem

and a squib-firing assembly (SFA). The SFA is a redundant unit employing separate battery sources, armed by separate g-switches during the boost phase, and fires redundant squibs upon command from the CC&S. The arming of the unit is backed up by the BUCT, and, in the event of a CC&S failure, the BUCT will command the SFA to fire the solar-panel squibs.

The arming of the SFA was accomplished as attested to by the fact that the squib firing events did occur. Whether the g-switches or the BUCT armed the SFA is not known since no telemetry gives this information.

The time for command of solar-panel extension was 1649:02 (62 min after CC&S inhibit release). B-2-1, B-2-2, and B-2-3 event blips observed at the time indicate that the CC&S sent the command to the SFA which, in turn, sent the current to the A and B redundant squib circuits.

Nominally, the nitrogen, fuel, and oxidizer valves on the midcourse motor are opened by redundant squib firings on command from the CC&S at 26.5 min after maneuver initiation. The nitrogen and fuel shutoff valves are closed by redundant squib firings on command from the CC&S when the accelerometer pulse count reaches the stored value. The B-2-1, B-2-2, and B-2-3 blips at 0857:08 and 0858:17 GMT indicate that both A and B squib circuits fired for both motor ignition and cutoff. The start and stop of accelerometer pulses at these times verify that the pyrotechnic subsystem performed its designed task.

M. Electronic Packaging and Cabling

The electronic assemblies for *Ranger VI* provided conservative mechanical support for electronic components in order to ensure proper operation throughout the various environmental exposures. The assemblies consisted of structural units which were integrated into the spacecraft by bolting to the six bays of the structure. Each electronic assembly consisted of one or more functional subsystems consisting of subassemblies of various widths. These assemblies, having a standard cross section, were bolted into a metal chassis and connected by pigtailed cable harnesses into the system "ring" harness. The subassemblies were designed to perform as an integral part of the spacecraft structure and to provide a highly conductive thermal path from the components to the primary thermal-control surfaces.

1. Packaging

By means of electronic packaging specifications, the following system goals were achieved:

1. Materials were restricted to a few whose space-environment behavior had been evaluated and understood.

2. Hard-mounted circuit boards were employed for added structural strength and to minimize temperature gradients from components to temperature control surfaces. This technique provided an operating margin in the dynamic and temperature environments without weight penalty. Further, the

circuit-board design permitted optimum maintainability and modification without reducing subassembly integrity, and provided for simple and unlimited replacement.

3. Since a standard subassembly cross section and a defined chassis mounting were used, the requirements for the thermal and dynamic environments could be evaluated and understood prior to subassembly layout and design, resulting in improved end reliability.
4. Conformal coating was employed to obtain an assured complete insulation of all electrical conductors, which reduced the hazards from peripatetic space trash.
5. Hidden solder joints and trapped air voids were eliminated.
6. Connector pin retention tests were used to obtain interconnection integrity.
7. Test-equipment connections were provided on the assemblies to permit electrical testing without hardware degradation.

2. Cabling

The function of the cabling subsystem was to interconnect the various assemblies. The cables were built in accordance with JPL specifications and properly fitted; samples of wire were tested environmentally prior to final assembly. The spacecraft cabling was divided into three groups: the subsystem-interconnection harness usually referred to as the "ring harness," the squib harness, and the assembly harness, which is also referred to as case harness. The ring harness separated the wires into two bundles for "clean" and "dirty" signal characteristics to eliminate or minimize noise pickup. The squib harnesses were physically separated from other harnesses to prevent impulse signals from affecting squibs. The assembly harnesses allowed environmental and functional testing as well as checkout of the assemblies prior to incorporation into the total spacecraft system.

During the *Ranger VI* mission, the spacecraft electronic packaging and cabling areas performed within their design expectations and no anomalies were recorded.

N. Television Subsystem

The TV subsystem was designed to provide the equipment to fulfill the mission objective of obtaining high-resolution video pictures of the lunar surface. The design objective was to have a system capable of photographic resolution of 0.5 to 5 m in the final picture.

Two separate chains of equipment comprised the subsystem in order to give increased reliability. Each chain contained slow-scan video cameras, camera electronics, sequencing circuits, transmitters, power supplies, and control circuitry. The two chains were essentially similar with the exception of the camera configurations. The F chain contained two fully scanned 1-in.-diameter vidicons, while the P chain contained four partially scanned 1-in.-diameter vidicons. The cameras of each chain were exposed and read out in sequence, with the two chains operating simultaneously. The video output was utilized to modulate the FM transmitters for transmission of the signal to the Earth-based receiving stations. For real-time subsystem performance analysis, diagnostic telemetry was transmitted through the spacecraft-bus telemetry system by the bus transponder, and also combined with the video signal for transmission by the TV transmitters.

The TV subsystem is electrically complete and independent of the spacecraft bus with three exceptions: commands are received from the spacecraft command receiver and the on-board CC&S; the spacecraft data encoder receives and passes on TV subsystem diagnostic telemetry; and the spacecraft directional antenna is used by the TV transmitters.

1. Subsystem Description

The major assemblies of the subsystem are: cameras, camera and control sequencer, telecommunications, electrical power, command and control, backup clock, and structure with associated passive thermal control. The electrical block diagram is shown in Fig. 26.

a. Cameras. The camera system is composed of six vidicons which operate in a slow-scan mode. The F channel has two fully scanned cameras (400 optical line pairs) F_a and F_b and the P channel has four partially scanned cameras (100 optical line pairs) P_1, P_2, P_3, and P_4. The camera tube is an electrostatically focused and deflected 1-in.-diameter vidicon with an antimony-sulfide/oxygen-sulfide photoconductor target.

Each camera consists of a camera head assembly (vidicon, shutter, lens, preamplifier, and housing) and its individual camera-electronics assembly. The received image is focused on the vidicon-photoconductor target by the lens and exposed by the shutter. The shutter utilized is an electromagnetically driven, linearly actuated slit, located in front of the vidicon focal plane. The image

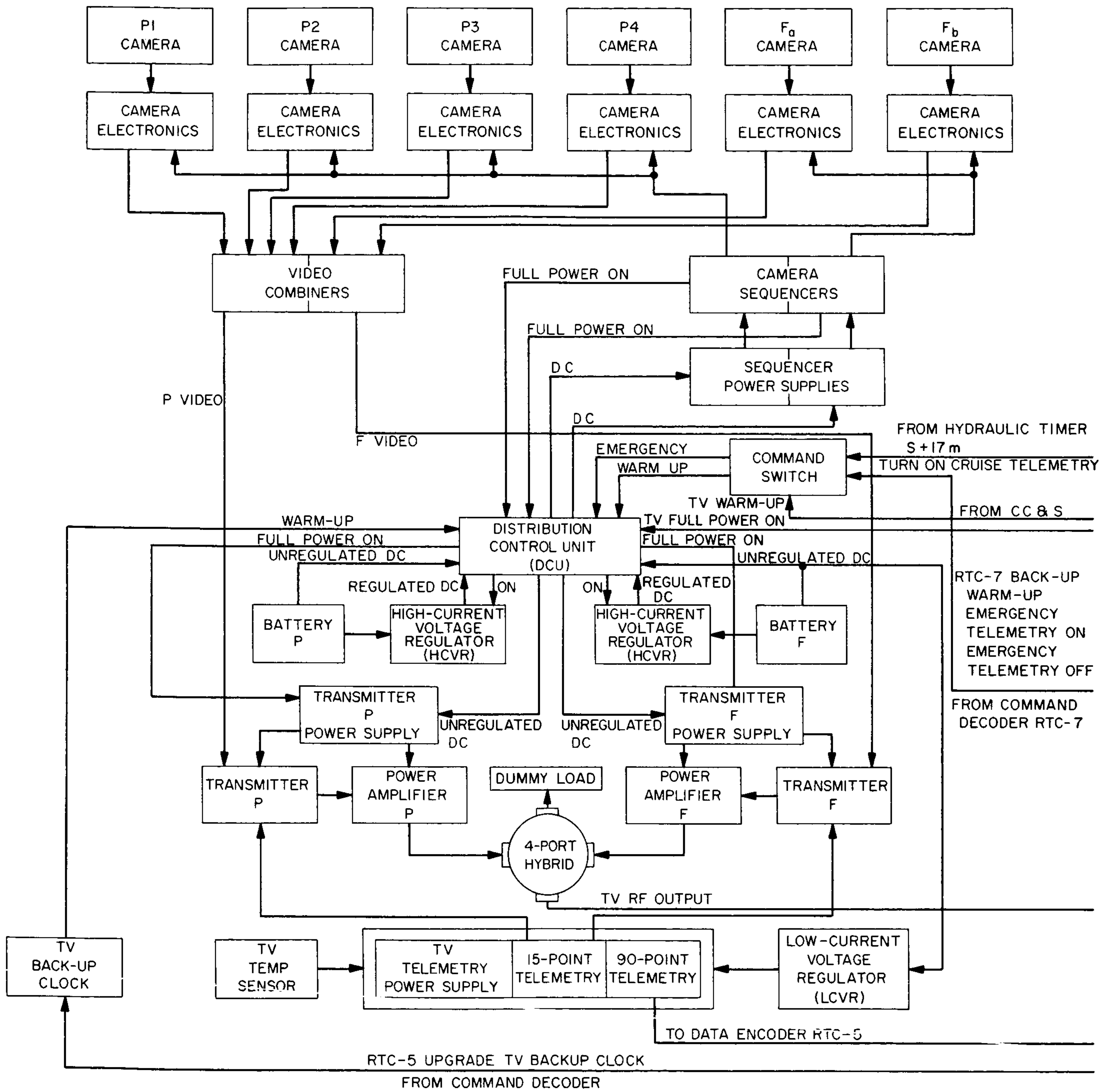


Fig. 26. Television subsystem

formed on the photoconductor causes target-resistance variations equivalent to the image brightness. After exposure, a high-frequency electron-beam signal scans the target and restores the charges on the target. During the beam-scan, the charge current is conducted off the target as video signal and coupled to a preamplifier. The signal is passed through the preamplifier and a video amplifier and gated in a video combiner, then sent to the respective transmitter. The associated camera electronics supplies the necessary operating voltage, sweep signals, focus signals, and shutter pulses for the camera head assembly. The nominal camera fields of view are indicated in Fig. 27.

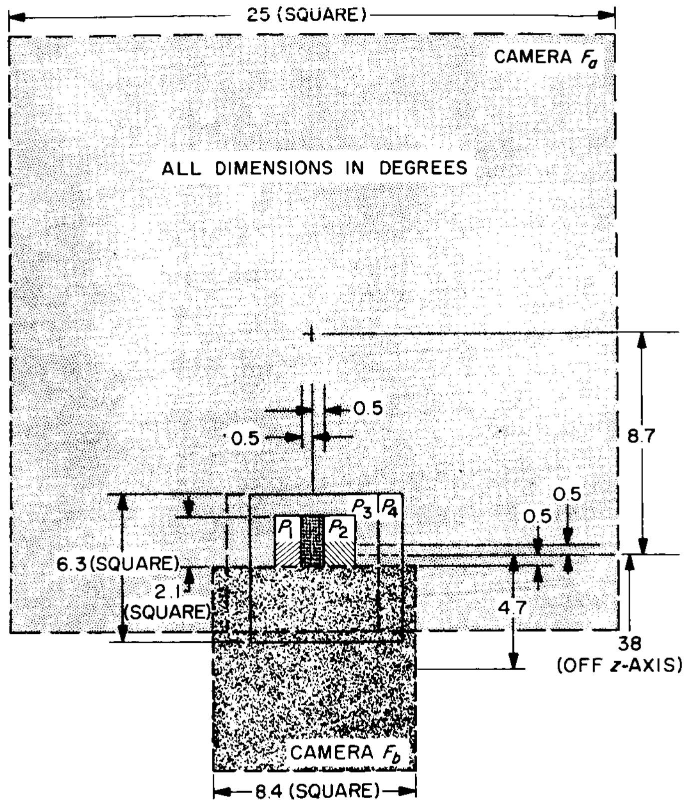


Fig. 27. TV camera fields of view

The dynamic ranges of the cameras are set to cover a possible lunar-luminance range from 20 to 2700 ft-L. Cameras F_a, P_3, and P_4 cover 20 to 650 ft-L, and cameras F_b, P_1, and P_2 cover 80 to 2700 ft-L. The F-type cameras have a 5-millisec shutter-exposure time. Camera F_a has a 25-mm-focal-length lens and camera F_b has a 75-mm lens. The P-channel cameras have 2-millisec shutter-exposure times; cameras P_1 and P_2 have 75-mm lenses, and cameras P_3 and P_4 have 25-mm lenses.

The F-channel camera utilizes 1152 active horizontal scan lines over a vidicon faceplate area of 0.44 × 0.44 in. The horizontal scan rate is 450 cps, with 0.22 millisec apportioned to horizontal blanking. The active scan lines, plus 46.6 millisec for vertical blanking, add up to a 2.56-sec frame rate. At the end of the active scan, the camera enters an erase cycle to prepare the target for the next exposure. The two F cameras are alternately scanned and erased, so that while one is being scanned and read out, the other is being erased and prepared for the next exposure.

The P-channel camera utilizes 290 horizontal scan lines over a vidicon faceplate area of 0.11 in. × 0.11 in. The horizontal line rate is 1500 cps, with 111.1 microsec allocated to horizontal blanking. The vertical scan and a 6.6-millisec blanking period occupy 0.2 sec. At the end of the active vertical scan, the camera enters an erase cycle of 0.64 sec to prepare the target for the next exposure. The four P cameras are sequentially scanned and erased, so that while one is being scanned the remaining three are in various portions of the erase cycle. A 40-millisec pulse is used to separate each sequence of the four P-channel cameras. Therefore, the total time period per sequence is 0.84 sec.

b. Camera and control sequencer. The sequencer consists of separate links for the two channels. Each link contains a video combiner, control programmer, camera sequencer, and sequencer power supply. The group generates synchronizing signals for the individual cameras, controls camera exposure and readout, combines the video from the individual cameras with sync and tone code signals (camera F_b video is identified by a 144-kc tone burst) and applies the composite video signal to the respective RF transmitter channel modulator.

c. Telecommunications. The communications equipment consists of two transmitting channels and an RF combining section. Each channel contains an L-band FM transmitter, intermediate power amplifier, 60-w power amplifier, telemetry processor, and transmitter power supply. A four-port hybrid ring in the RF section combines the outputs of the two transmitting channels. The combined output is fed through an RF directional coupler to the high-gain antenna. Each transmitter contains a modulator, two frequency multipliers, and an intermediate power amplifier. Each transmitter operates within a bandwidth of approximately 900 kc. A 160-kc band between the transmitter output frequencies is reserved for the spacecraft bus transponder transmitter.

Two commutated telemetry signals are provided. Fifteen-point telemetry, sampling at one point per sec, carries critical temperatures and voltages and the backup clock position during the cruise and terminal phases. Output of the 15-point sampling switch drives a channel 8 IRIG subcarrier oscillator connected via an amplifier and transformer with the spacecraft data encoder.

Terminal mode telemetry is activated 5 min before TV subsystem full-power activation (nominally 15 min before impact) by a warmup command from the CC&S, or the backup clock. A 90-point sampling switch operating at a rate of three points per sec samples subsystem parameters to provide the detailed diagnostic telemetry.

The output of this switch drives two 225-kc voltage controlled oscillators (VCO) connected in parallel. The output from one VCO is summed with the video at the input of transmitting channel F; the output of the other is summed with the video at the input of transmitting channel P.

An emergency mode is provided to permit the ground station to receive the terminal mode telemetry in the event a system malfunction prevents normal reception. This mode is activated by a second RTC-7 from Earth. In the emergency mode, the video and 225-kc telemetry signals are switched out and the 90-point terminal mode telemetry signals are used to pulse-amplitude modulate (PAM) the P-channel transmitter directly. The channel F transmitter continues to transmit video from the P and F cameras.

d. Electrical power. The power assembly for each channel includes an individual battery and high-current voltage regulator for supplying unregulated and regulated power to the operating assemblies. Each battery consists of 22 series-connected silver-zinc-oxide cells, having a capacity of 40 amp-hr. The high-current voltage regulators supply current at 27.5 ± 0.5 v dc, as well as the unregulated battery voltage between 30.5 and 36 v dc; each regulator has a silicon-control rectifier turnon device to switch on prime power to the system. A low-current voltage regulator on the P battery provides power to the cruise-mode telemetry.

e. Command switch and control circuits. The command switch and control circuits of the *Ranger* TV subsystem transfer commands from the spacecraft CC&S and command decoder and the TV camera sequencer to the appropriate TV-subsystem addressees. Commands are in accordance with Table 8.

RTC-7 and command switch functions. RTC-7, which is a 150-millisec contact closure generated by the spacecraft command subsystem, activates a command switch amplifier which provides a 4-amp pulse to operate the command switch. The pulse causes the 4-position command switch to advance one position, making and breaking the 4 sets of contacts associated with each command switch position, providing the functions shown in Table 9.

Preprogrammed commands and control circuit functions. In addition to the applicable commands shown in

Table 8. TV subsystem commands

Command	Initial command		Backup command		Remarks
	Source	Preprogrammed	Source	Preprogrammed	
Operational clock on	(separation)	Yes	None	—	—
Cruise mode	Hydraulic (S + 17 min)	Yes	First RTC-7 (warmup command)	No	Backup command 66 hours after launch
Warmup	CC&S	Yes	First RTC-7 (backup clock)	No (Yes*)	(*Channel F only)
Clock inhibit	RTC-5	No	None	—	To prevent clock turn-on in the event of abnormal trajectory
Power amplifiers on	Camera sequencer	Yes	CC&S	Yes	Terminal mode
Emergency	Second RTC-7	No	None	—	PAM telemetry only
Emergency off (normal operation re-established)	Third RTC-7	No	None	—	Return to normal operation
System turnoff (return to zero mode)	Fourth RTC-7	No	None	—	To reset command switch and turn system off. In the event of premature TV initiation or casualty

Table 9. Command switch functions

Switch position	Function
Zero	The starting position, battery output not applied
Warmup	Turns on high-current regulators
	Activates the cruise mode telemetry relay
	Indicates command switch position via telemetry
Emergency On	Maintains the warmup signal
	Energizes emergency mode telemetry relays for one transmitter
	Indicates command switch position via telemetry
Emergency Off	Maintains the warmup signal
	De-energizes emergency telemetry mode relays
	Indicates command switch position via telemetry
Zero	Turns off TV subsystem

Table 8, additional monitoring circuits are provided to indicate application of regulated and unregulated power.

Distribution control unit (DCU). The DCU connects regulated and unregulated voltages from the two high-current voltage regulators and distributes them within the subsystem. It also provides the necessary backup clock interface with the subsystem.

Among other features, the DCU provides primary power fuse protection and interface circuits for the clock, including:

1. A clock activation signal.
2. Two separate telemetry routings.
3. Clock-operated routing.
4. Circuits for turning on either or both high-current regulators.

The CC&S warmup command is routed to the DCU. This signal is used to activate the two high-current regulators.

Activation of cruise-mode operation energizes the telemetry power relay. This routes the output of the low-current regulator into the telemetry and temperature sensor chassis. In the event the relay fails to energize, voltage would be applied to the telemetry and temperature sensors from high-current regulator F during terminal-mode operation.

Necessary voltages to energize the power control unit (PCU) are also provided by means of the DCU.

The DCU (and, therefore, the TV subsystem) is designed to operate with or without the clock.

f. TV backup clock. The backup clock is a solid-state timer with an accuracy of $\pm$ 5 min. It is designed to initiate channel F warmup at a preset interval after *Agena*-spacecraft separation, when it is activated. The TV backup clock backs up turn on commands from the spacecraft CC&S and the ground. In the *Ranger VI* mission, the TV backup clock's setting (64¾ hr) and the corrected trajectory (see Section VI) were such that its impulse came first and initiated TV operations. The clock is so mechanized that it can be disabled by an RTC-5. Telemetry pulses at 16-hr intervals verify clock operation.

g. Structure and thermal control. Subsystem external structure is that of the frustum of a right circular cone topped by a cylindrical section. The primary strength of the structure is provided by an internal box span consisting of stiffened panel sections supported by eight longerons. The electrical assemblies are mounted on structural decks supported by the longerons. The cameras are mounted on a solid machined camera bracket housing within the top cylindrical section, with a view port cut in the external thermal shroud.

The thermal control is entirely passive, with a thermal shield (mounted outside the structure body) and fins used to control the radiative exchange of energy between the TV subsystem and other sources (or links) of thermal energy. The thermal mass of the structure is the primary heat sink during the terminal mode of operation.

2. Sequence of Critical Events

The actual sequence of critical events involving the *Ranger VI* TV subsystem is summarized in Table 10. The events in Table 10 are discussed in the following paragraphs.

1. The unscheduled appearance of the TV cruise telemetry immediately after booster engine jettison was the first and only symptom of nonstandard operation in the TV subsystem until the terminal phase of the mission. A likely mechanism for this inadvertent turnon is for the command switch to be stepped to the warmup position which would normally also activate the high-current regulators and start the 5-min timers in the sequencer, which would lead to full power operation. The cruise telemetry

Table 10. Television subsystem critical event times

Item	Mission time[a]	TV subsystem event
1	$L + 140$ sec	Cruise telemetry turnon for 68 sec (nonstandard).
2	S	Backup clock started on signal from separation switch.
3	$S + 17$ min	Cruise telemetry turnon by bus backup command timer.
4	Telemode II	TV channel 8 removed from bus carrier until mode III.
5	$S + 64$ hr 45 min	F-channel warmup sequence commanded by TV backup clock.
6	$I - 15$ min	P-channel warmup sequence by command switch on receipt of RTC-7.
7	$I - 8$ min	Second RTC-7 to step command switch to *emergency*.
8	$I - 5$ min	Third RTC-7 to step command switch to *emergency off*.

[a] L = launch; S = separation; I = impact.

on channel 8 contains no measurement which will identify the failure mechanism. No RF signal from either video transmitter was observed or recorded during this period. (A brief dropout of the bus RF signal which started just before channel 8 came on was determined to be a normal occurrence.)

2. The backup turnon clock start signal is generated by a microswitch closure when the spacecraft is separated from the *Agena*. Verification of clock operation is obtained by pulses every 16 hr which produce one-volt step changes on segment 9 of the cruise TV commutator. The occurrence of these step changes at 16, 32, 48, and 64 hr after separation indicate nominal clock performance in flight.

3. Normal cruise telemetry turnon is initiated by the bus backup command timer at 17 min after separation. This event appeared normal in all respects. The commutator started at the position where it had stopped at the end of the previous inadvertent turnon. All voltage and temperature measurements indicated normal TV status at this time. This means that if the command switch had been stepped into warmup earlier, it had subsequently been cycled to the *off* position.

4. The mixing of channel 8 data for modulation on the bus carrier is controlled entirely in the bus data encoder. (Switching is normally done to obtain midcourse accelerometer data on channel 8 during motor firing.) The TV cruise telemetry was switched back onto the bus carrier when the data encoder stepped to mode III at the end of the midcourse maneuver.

5. The only verification that the backup clock commanded the TV F-channel into warmup was a slight drop in F-battery terminal voltage (channel 8, segment 5). Normal warmup activates the high-current regulator, sequencer, and amplifier filaments and normally produces more of a voltage drop than that observed. A critical failure was indicated when no F-channel full-power video signal appeared after 5 min.

6. The first RTC-7 was a standard command to step the command switch to *warmup*. Since it was sent less than 5 min after the backup clock F-channel warm-up command, no positive TV system failure had been observed when this command was sent. The command switch activates drive circuits to turn on both TV channels. In this case, P-channel warmup was verified by a normal step change on channel 8, segment 11 in response to a pulse generated by the P-channel sequencer 30 sec after its activation. (No comparable measurement is available for the F-channel sequencer.) A slight drop in P-channel battery voltage was also observed.

7. Five minutes after the first RTC-7 was received by the spacecraft, the 5-min timer in the P-channel sequencer should have initiated the application of high voltage to the power amplifiers for full-power operation. The absence of any full-power video signal at this time indicated failures in both TV channels. The second RTC-7 was intended to step the command switch to the *emergency* position in the hope that some 90-point telemetry data would be recorded for later failure analysis. The bus command decoder verified command receipt, but without the 90-point data there is no way to verify whether the command switch was actually stepped.

8. The third RTC-7 was sent to step the command switch to *emergency off* and return the P channel to normal video after 3 min in *emergency* mode. Just as with the second RTC-7, a real time command monitor blip was telemetered by the bus, but the response of the TV subsystem cannot be determined.

Immediately after the mission, a failure investigation team was formed to determine the nature and cause of the TV subsystem failure. It is believed that the high-voltage supplies for both TV channels failed because of arcing that occurred as the result of power turnon in a critical pressure region during the period when the cruise telemetry was inadvertently turned on. The exact mechanism for the TV subsystem and channel 8 turnon and turnoff has not been determined. However, a transient, a short, or a static discharge are all possible initial causes of turnon.

V. DEEP SPACE NETWORK SYSTEM

The Deep Space Network (DSN) is a precision communication system designed to communicate with, and permit control of, spacecraft traveling 10,000 miles from Earth and beyond. It consists of the Deep Space Instrumentation Facility (DSIF), with communications and tracking stations based around the world; the Space Flight Operations Complex (SFOC), the command and control center; and the Ground Communication System (GCS) that connects all parts of the DSN by telephone and radio-teletype.

A. DSIF

The DSIF provides the critical communications link between spacecraft and Earth. It tracks the flight of all deep space probes launched by the United States, issues commands to the spacecraft, and receives the engineering and scientific measurements made by the spacecraft. Continuous coverage during missions is provided by antennas located approximately 120 deg apart in longitude; accordingly, stations are located in Australia and South Africa, as well as in the United States; the foreign stations are maintained and operated by cooperating agencies in these countries. A block diagram of the Goldstone Echo station (DSIF 12), in *Ranger VI* configuration, is given in Fig. 28.

The DSIF support for *Ranger VI* consisted of permanent stations located at Goldstone, California (DSIF 11 and 12), Woomera, Australia (DSIF 41), and Johannesburg, South Africa (DSIF 51); a spacecraft monitoring station (DSIF 71) at Cape Kennedy, Florida, used for spacecraft prelaunch checkout and initial telemetry reception; and the mobile tracking station (DSIF 59), located near DSIF 51, used to provide post-injection tracking and acquisition data. Equipment at the six stations is listed in Table 11.

1. Mission Preparation

Prior to the *Ranger VI* mission and its associated tests, mission-oriented equipment was installed throughout the DSIF. This equipment included a new telemetry decommutator and teletype encoder at each station; a 200-w transmitter and command system at Woomera; special subcarrier oscillators and a 10-cps modulation system throughout the DSIF; and a video recording system at each of the overseas stations.

On October 1, 1963 the DSIF began checkout and preparation of all station subsystems. A number of tests were conducted during the months preceding the mission, the most significant being compatibility, net integration and operational readiness. The final operational

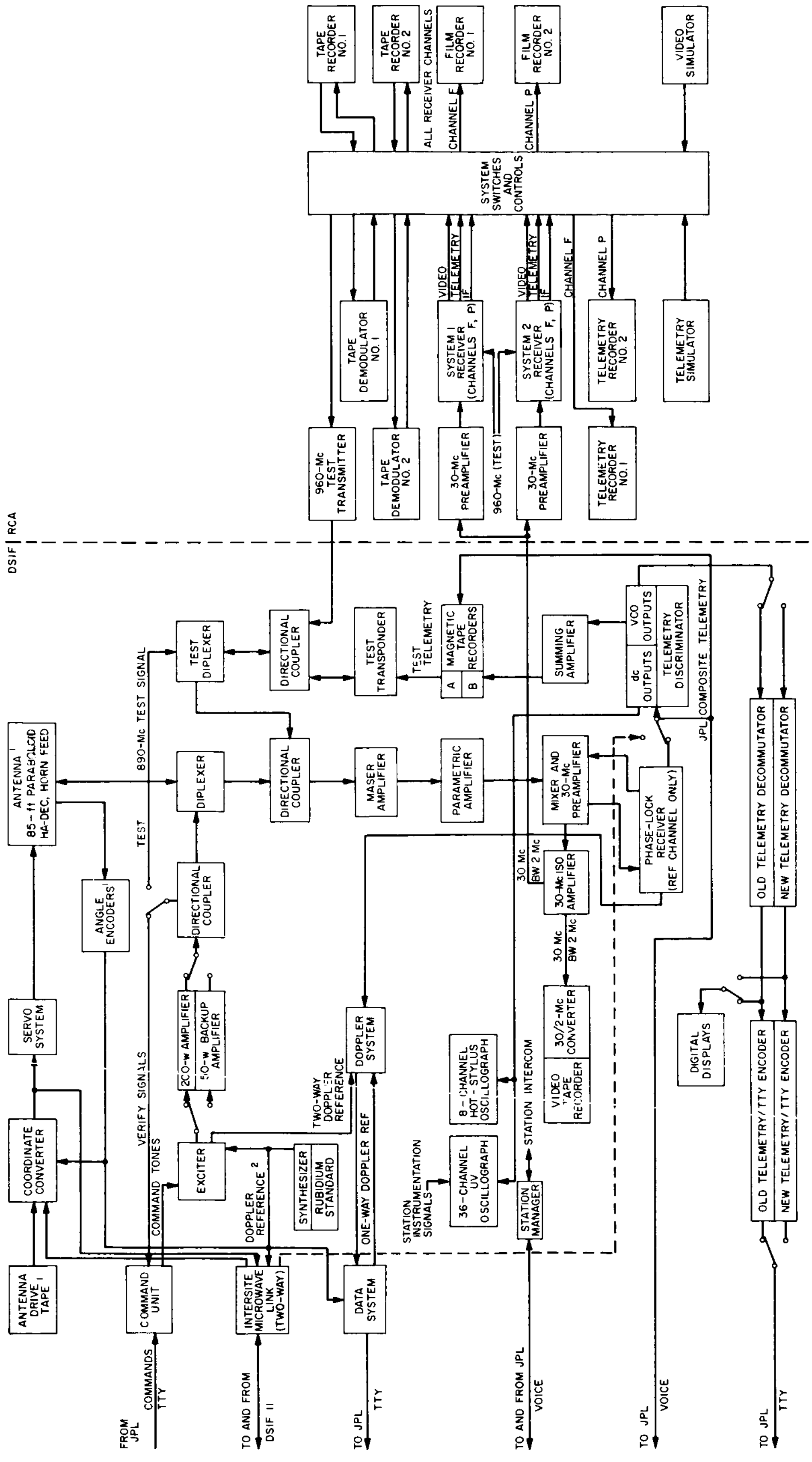


Fig. 28. Goldstone Echo station

[1] ANGLE DATA NOT FROM AUTO-TRACKING

[2] FOR COHERENT TWO-WAY OPERATION WITH DSIF II

Table 11. DSIF L-band master equipment list

Equipment	DSIF Stations: Goldstone 11	Goldstone 12	Australia 41	South Africa 51	South Africa 59	Cape 71
Antennas:						
85-ft paraboloid HA-DEC	O	O	O	O		
10-ft paraboloid AZ-EL					O	
6-ft paraboloid AZ-EL						O
Low noise amplifiers:						
Maser	O	O				
Paramp	O	O	O	O		
Feeds and diplexers:						
Tracking feed			O	O	O	
Horn feed	O	O				
Acquisition aid			O	O		
Dipole						O
Diplexer		O	O	O	O	O
Receiver:						
960-Mc GSDS modified					O	O
960-Mc GSDS	O	O	O	O		
Transmitter:						
50-w backup		O				
10-kw (operated 200 w for *Ranger* Block III)		O	O	O		
25-w					O	O[a]
Rubidium standard		O				
Synthesizer		O				
Doppler:						
One-way		O	O	O	O	O
Two-way		O	O	O	O	
Two-way two-station noncoherent						

Equipment	DSIF Stations: Goldstone 11	Goldstone 12	Australia 41	South Africa 51	South Africa 59	Cape 71
Recording:						
7 Ch magnetic tape	2	2	2	2	2	2
Strip chart:						
36 Ch ultraviolet	O	O	O	O	O	O
8 Ch hot stylus	O	O	O	O	O	
Acquisition aids:						
10-cps modulator		O	O	O		
Mission-oriented equipment:						
Command system		O	O	O		
Command interrupt		O				
Telemetry decommutator/encoder		O	O	O		
Telemetry discriminator		10	10	10	10	
Prime RCA TV GSE		O				
Secondary RCA TV GSE	O					
JPL TV GSE		O[b]	O	O		
Prime test equipment:						
Test transponder	O	O	O	O	O	O
Closed-loop RF system	O	O	O	O	O	O
Bit error checker		O				
Optical star tracker		O				
Miscellaneous:						
Intersite microwave	O	O				
Coordinate converter	O	O				

[a] Transmitter used for prelaunch only.

[b] Redundant system backup mag tape converter/FR-800.

readiness test involving the entire DSIF was conducted on January 24 and 25, 1964. There were no problems of any consequence during the station preparedness tests, and all systems were "in the green" for the *Ranger VI* launch.

2. Launch to Midcourse

Launch of the *Ranger VI* spacecraft occurred at 1549:09 GMT on January 30, 1964. DSIF 71 was in RF lock at launch and until 1557 when the spacecraft set below the horizon. Approximately 3 min after launch, the spacecraft data analysis team reported an indication that the TV subsystem had inadvertently turned on for about 1 min. DSIF 51 was alerted and instructed to transmit the TV turnoff command if necessary. Because of its flight path, the spacecraft was not acquired by DSIF 51 until 1620, at which time there was no indication of a video signal from the spacecraft. DSIF 41 upon acquisition at 1634 confirmed the fact that there was no evidence of a turnon. Consequently the TV subsystem was assumed to be in a standard mode of operation.

Event blips were observed by DSIF 41 at near nominal times, confirming solar-panel extension and Sun and

Earth acquisition. A planned sequence of initial command transmission from DSIF 41 was abandoned when it was discovered that the station was inadvertently transmitting via the acquisition-aid antenna. The first ground commands were sent to the spacecraft by DSIF 51 on January 30, 1964. Two "clear commands" (RTC-0) were sent at 2108 and 2110, followed by an antenna changeover command (RTC-3) at 2112. This last command switched the spacecraft transmitter from the omni antenna to the high-gain antenna.

3. Midcourse to Impact

Preliminary spacecraft orbit computations indicated that a trajectory correction was required to achieve lunar impact in the preselected target area. At 0720 GMT on January 31, DSIF 12 began the procedure required for transmission of the midcourse maneuver commands.

All guidance commands were correctly received by the spacecraft, antenna changeover was effected and the midcourse-maneuver-execute command was initiated at 0830. The midcourse maneuver began at 0830:40 and was completed at 0858:17. The two-way doppler shift during the retro-motor firing indicated a perfect maneuver had been executed.

After the maneuver, the spacecraft, responding to the CC&S commands, reacquired the Sun and Earth. DSIF 12 then sent the command to switch the spacecraft transmitter from the omni antenna to the high-gain antenna (RTC-3).

The *Ranger VI* spacecraft was now in a cruise mode proceeding on a lunar-impact trajectory. Based on subsequent orbital computations using postmidcourse tracking data, it was decided a terminal maneuver would not be necessary.

On February 2, at impact minus 19 min, television channel A went into warmup as initiated by the TV backup clock. At about impact minus 15 min TV channel B went into warmup as the result of a backup command (RTC-7) sent by DSIF 12. Both TV channels should have indicated full power at impact minus 10 min and video should have been received at Goldstone, but neither occurred. Another RTC-7 command was sent at impact minus 9 min in an effort to force at least one TV channel into full power, but it had no effect. A third and final RTC-7 command was sent at impact minus 5 min in what proved to be a futile attempt to turn on the warmup mode. Impact occurred at 0924:33.

4. Tracking Performance

In general, the quality of the tracking data received from the DSIF stations was excellent, although there were some problems. A summary of the tracking data used in the orbit determination program, together with the noise statistics, is presented in Table 12.

The angular data showed that the correction polynomials used in the orbit determination program to describe the angular pointing error were not adequate. Large biases remained in the hour-angle residuals after these corrections had been applied.

The doppler-tracking data was excellent except for the first pass at Johannesburg and the latter part of the third pass at Woomera. The doppler data from the mobile tracking station (MTS) was unusable in the orbit determination program because the counter was recycling at odd times. Almost all doppler data from the Johannesburg first pass was lost due to poor station performance. The orbit determination program was able to use only 40 of the 244 points of 5-sec sample data taken at Johannesburg on the first pass. A total of 3½ hr of doppler data was lost during the Woomera third pass because of transmitter voltage control oscillator (VCO) instability, and the precision bias doppler loop showing a false lock condition. The quality of the doppler data from both Goldstone stations was excellent throughout the mission. A reduction in doppler noise on this mission in comparison to previous missions was obtained by using the frequency synthesizer rather than the VCO, which may be seen by comparing the noise statistics of Echo data (Table 12) on January 31 and February 1.

A new transfer procedure, which consisted of transferring the spacecraft from one station to the other without going to the one-way doppler mode, was successfully effected several times without loss of ground station lock. Some data was lost however, due to the incorrect use of the data conditioning code. In several instances the two stations involved in the transfer reported good one-station two-way doppler for simultaneous periods.

5. Equipment Problems

Summarized below are the equipment problems encountered at each DSIF station during the *Ranger VI* mission.

a. Mobile tracking station. The only serious failures occurred during countdown, namely a voltage regulator diode failure causing a 3-min power outage, and the failure of a collimation tower polarization drive which

Table 12. Summary of tracking data used in *Ranger VI* spacecraft orbit computations

DSIF station	Data type	Beginning date/GMT	Ending date/GMT	Number of points	Standard deviation	Root mean squared, rms
Premidcourse						
Echo	Cc 3	31/0635	31/0646	31	0.0229 cps	0.0542 cps
Woomera	Cc 3	30/1659	30/2034	136	0.0259 cps	0.0283 cps
	HA	30/1647	30/2301	340	0.0139 deg	0.0373 deg
	Dec	30/1647	30/2301	342	0.0072 deg	0.0210 deg
Johannesburg	HA	30/1621	30/1630	108	0.0227 deg	0.0521 deg
	Dec	30/1621	30/1630	108	0.0228 deg	0.0231 deg
	Cc 3	30/1626	31/0620	485	0.0361 cps	0.0374 cps
	HA	30/1915	31/0633	577	0.0143 deg	0.0244 deg
	Dec	30/1915	31/0633	576	0.0105 deg	0.0109 deg
Postmidcourse						
Pioneer	Cc 3	31/0907	31/1600	396	0.0237 cps	0.0237 cps
	Cc 3	01/0742	01/1728	571	0.0146 cps	0.0154 cps
Echo	Cc 3	31/0906	31/1600	384	0.0146 cps	0.0152 cps
	Cc 3	01/0612	01/1724	665	0.0146 cps	0.0146 cps
Woomera	Cc 3	31/1608	31/2103	230	0.0310 cps	0.0317 cps
	Cc 3	01/1736	01/2038	161	0.0587 cps	0.0611 cps
Johannesburg	Cc 3	31/2106	01/0555	397	0.0310 cps	0.0310 cps
	Cc 3	01/2355	02/0154	100	0.0467 cps	0.0468 cps

caused a slight delay in countdown. During operations, 2 min of tracking data were lost due to corrosion of the card contacts on the doppler-shift-register cage causing the tape punch to run away. Lack of communication circuits delayed much of the information obtained by this station; there were at one stage twenty tapes waiting to be transmitted including valuable channel 8 data.

b. Goldstone Pioneer station. There were no equipment problems during the first tracking period of the *Ranger VI*. During the second pass at 0745 on February 1 the reperforator unit failed due to a bad relay. Repairs were made and the unit was back in operation at 0816. Occasional bad samples in the data subsystem were caused by dirt in the transmitter/distributor. The dirt accumulates in the transmitter distributor when tape from the high-speed punch is run onto the floor to store it for simultaneous transmission. Near the end of the second tracking period, the oscillograph recorder failed and was replaced with a spare unit prior to the third pass. Some difficulty was encountered before the third tracking period due to a last-minute requirement for recording spin modulation on both the oscillographs and the Ampex recorders. The equipment had to be set up at the same time that the replacement oscillograph was being aligned.

The maser/paramp subsystem operated normally, with no failures or unusual occurrences. The receiver subsystem operated normally during the entire mission. The servo subsystem encountered some difficulty with "bad commands" received from the coordinate converter. The same servo problem was encountered during compatibility test No. 3.

c. Goldstone Echo station. During the first pass on January 30, there was a decommutator malfunction. The failure was found to be in the B19 discriminator which was replaced. Later a channel B20 discriminator failed and was replaced. A TTY punch was replaced to correct a garbling of data lines, and a bad dc amplifier in the ZG-ZC line was replaced.

On the second pass, the only failure was a TTY reperforator which was repaired and reinstalled.

Prior to countdown for the third pass the parametric amplifier failed. A new klystron was installed and tuned during countdown. All systems operated in a satisfactory manner during the entire mission.

d. Woomera station. The channel 4 discriminator output was noisy and off the scale on the high-frequency end during the first pass. Since usable data was being received, the station was requested not to change the discriminator. The discriminator was replaced after the first pass and no further trouble was experienced.

When the station changed to the second pass RF configuration the parametric amplifier showed low maximum gain and had a tendency to oscillate. For this reason the cage-mounted paramp was used for the remainder of the track.

At 2020 on the third pass the transmitter VCO started to drift and later showed jumps of up to 18 cps. In view of the importance of telemetry data, two-way lock was not broken to replace the VCO. This condition persisted until 2145 when the VCO apparently became stable. The apparent reason for the instability was an unsoldered lead inside the module between the input and the 29⅔-Mc distribution amplifier. At 2142 the doppler bias loop went into a false lock condition which could not be corrected. This condition was also allowed to persist because of the importance of telemetry data. A post mission check disclosed the X90 module was outside specifications and the (30 + 1) module contained an intermittent coaxial connection. Two digits on the telemetry digital printer gave an occasional random incorrect printout in the last few hours of the mission. Mechanical adjustment and the relay driver cards in the print mechanism were the apparent cause for the random printout.

e. Johannesburg station. There were very few equipment faults during the mission, and those that occurred were rectified rapidly.

The oscillograph recorder was out of action for the latter part of the first pass on January 31 due to a lamp failure. A card in the Beckman decommutation, which had caused faulty rate 4 readouts, was replaced after the first track. Although the Ransome decommutator was not used, several cards were replaced during countdown to keep it operational.

No faults occurred with the receiver, although it did drop lock on a number of occasions for no apparent reason. These dropouts, most of which were of very short duration, only occurred when the station was in two-way lock, thus the transmitter VCO which was replaced later was, in all probability, the cause.

There were no severe problems during tracking, but due to a fault in the gear train, the hour angle followup package was replaced during a countdown when the counters failed to follow.

The acquisition panel which had been used for *Mariner* missions was installed prior to the third pass to facilitate finer adjustment of transmitter VCO frequency. While in two-way lock on the second pass, the transmitter was kicked off by a safety switch in the Klystron body current circuit. It was restarted and operated satisfactorily for the remainder of the mission. During the third pass, while in two-way lock, an 80-cycle modulation was observed on the receiver dynamic phase error signal. This spurious modulation was tracked to a faulty transmitter VCO which was replaced, and the problem cured. The receiver dropout could also have been caused by this faulty VCO.

B. Space Flight Operations Complex

The Space Flight Operations Complex (SFOC) includes a number of areas at JPL in which mission operations are conducted. All spacecraft command and monitoring functions took place in the SFOC, where spacecraft data were analyzed, evaluated, and interpreted. Additional support was provided by the Central Computing Facility, which reduced all *Ranger VI* tracking and telemetry data to usable form. Communications were controlled by the communications center, which handled all communication circuits providing data flow to or from any DSIF stations and/or operational unit at JPL.

The extent to which decisions are correct and effectively acted upon is a measure of the effectiveness of the SFOC. The SFOC includes a display system, television output of certain cameras with an audio status line for an internal/external Laboratory information system, access control and facility security, standby maintenance-personnel support, technical-area-assistants support for the spacecraft data analysis team and the flight path analysis and command group, and the necessary capability for correction of any facility housekeeping failures or problems.

There were no failures of any consequence during the *Ranger VI* mission. Several minor equipment failures occurred and were fixed in near-real time during the mission.

1. Central Computing Complex

This complex consisted of two IBM 7094 computers, three IBM 1401 computers, an SC-4020 plotter, a PDP-1 computer, the telemetry processing station, and the personnel required to operate and maintain the equipment.

During the days prior to the *Ranger VI* launch, the complex executed a launch-checkout sequence of events which included testing and shakedown of both software and hardware. The completion of this checkout indicated a state of mission readiness for the complex.

In general, all computer programs performed well during the mission. The orbit determination and trajectory computation effort was very satisfactory and all scheduled tasks were completed. The computation of the midcourse and terminal maneuver commands proved to be excellent. Real-time display of raw and converted engineering telemetry data, including television subsystem data, was supplied to the spacecraft performance analysis area by the PDP-1 computer and the telemetry processing station. Bulk processing, in the form of printed listings and plots, of engineering telemetry data on the IBM 7094's was satisfactory although more computer time was consumed than had been anticipated.

The computers and associated equipment had a good record of reliability during the course of the mission. The few equipment problems which occurred were minor and caused little or no delay to the operations due to quick repair and/or duplicate or backup-hardware capabilities which were available.

Postflight processing of tracking and telemetry data began immediately upon completion of the mission.

2. Communications Center

The performance of the communications center during the flight was quite effective. The communications failures experienced within the SFOC were due to terminating apparatus only, and were of a type and quantity well within normal expectations. Mechanical failures of teletype equipment, tube and semiconductor failures, plus minor technical adjustment problems constituted all of these failures.

C. Ground Communications System

The DSN Ground Communications System consists of voice and normal and high data rate teletype circuits provided by the NASA world-wide communications network between each overseas DSIF station and the SFOC, and teletype and voice circuits between the SFOC, Goldstone stations and Cape Kennedy, and a microwave link between the SFOC and Goldstone.

The high-frequency radio communication link between Sydney and Hawaii provided a problem, primarily due to intermittent distortion. A new submarine cable was expected to be operational before the next mission, and thus eliminate this problem.

The circuits to Johannesburg provided the most difficult communications problem. It was hoped the Sydney-Pretoria "back-door" circuit installed prior to *Ranger VI* would at least increase the quantity and rate of data flow, but unfortunately it did not help much. Another radio teletype circuit will be implemented for the next mission, in an effort to correct deficiencies in teletype and voice communications.

VI. SPACE FLIGHT OPERATIONS

The Space Flight Operation (SFO) System is made up of the equipment, computer programs, and technical and operational personnel which conduct the flight mission. Supported through the ground communications network by the DSIF and AMR tracking and telemetry facilities, and occupying the Space Flight Operation Complex (SFOC) at JPL, the SFO System determined the spacecraft trajectory, defined and calculated the midcourse trajectory correction, obtained and evaluated telemetered data on the spacecraft, generated commands for transmission to the spacecraft, and would have evaluated video picture quality had video data been obtained. During the *Ranger VI* mission the SFO System operated in a nominal fashion.

Prior to the mission, a number of compatibility, training, and readiness tests were conducted, according to a comprehensive SFO test plan. They included tests of spacecraft/DSIF/SFOC compatibility and DSIF/SFOC operational integration, SFOC personnel training, operational command procedures, and operational readiness.

Rapid and vigorous failure analysis was employed in conjunction with the testing program to ensure and verify correction of any problems encountered, and complete readiness was demonstrated prior to the mission.

Ranger VI space-flight operations began at approximately 1000 GMT on January 30, 1964 when comunications were established with AMR some 5½ hr before launch; they terminated with spacecraft lunar impact some 72 hr later, although postflight analysis and data reduction continued for several months.

The Central Computing Facility and the DSIF completed the mission with no critical failures. The communications, although not causing any critical mission problems, were not satisfactory to the overseas DSIF stations because of propagation difficulties along the voice and teletype paths. The communications problem was the greatest SFO System difficulty encountered during the flight. The equipment within the SFOC operated satisfactorily. No backup mode of equipment operation was required during the entire mission.

The Spacecraft Data Analysis Team continuously monitored the spacecraft data as received and provided the SFO director with spacecraft status and analytical information during the entire mission. All real-time recommendations for commands to the spacecraft were correct. The postflight analysis efforts of this team included, among other results, the conclusions presented in Section IV.

The Space Science Analysis and Command group provided information relative to midcourse and terminal maneuvers since these maneuvers would affect the quality of the expected video pictures. The principal investigator and the co-investigators were important operational additions to the mission team; their contributions to the mission proved to be highly effective.

The primary Flight Path Analysis and Command function, to provide orbit information of satisfactory quality as required within the flight sequence, was accomplished.

The premidcourse orbit computations indicated that *Ranger VI* would miss the lunar surface on the leading side and that closest approach would occur on the non-visible side approximately 2550-km from the Moon's center.

Midcourse maneuver parameters were computed which would change the spacecraft trajectory such that *Ranger VI* would impact a point chosen by the investigators on the visible side of the Moon. The aiming point chosen was near the light/shadow terminator so that desirable lighting conditions would prevail for the TV pictures.

Postmidcourse maneuver orbit computations indicated that *Ranger VI* would impact the Moon in the desired target area.

VII. FLIGHT PATH

A. Launch Phase

Ranger VI was launched from AMR, Cape Kennedy, Florida, at 1549:09.09 GMT on Thursday, January 30, 1964, using the *Atlas D/Agena B* launch vehicle. After liftoff, the booster rolled to an azimuth of 95.0 deg (east of north) and performed a programmed pitch maneuver until booster cutoff. During sustainer and vernier stages, adjustments in vehicle attitude and engine-cutoff times were commanded as required by the ground guidance computer to adjust the altitude and velocity at *Atlas* vernier-engine cutoff. After *Atlas/Agena* separation, there was a short coast period prior to the first ignition of the *Agena* engine. At a preset value of sensed velocity increase, the *Agena* engine was cut off. At this time the *Agena*/spacecraft combination was coasting in a near-circular parking orbit in a southeasterly direction at an altitude of 188 km and an inertial speed of 7.80 km/sec. After an orbit-coast time of 17.78 min, determined by the ground guidance computer and transmitted to the *Agena* during the *Atlas* vernier stage, a second ignition of the *Agena* engine occurred. About 88 sec later, the *Agena* engine was cut off with the *Agena*/spacecraft combination in a nominal Earth–Moon transfer orbit.

B. Cruise Phase

Injection occurred at 1616:42 GMT near the western coast of South Africa at a geocentric latitude and longitude of −7.19 and 8.05 deg, respectively. The *Agena* and spacecraft were at an altitude of 192 km and traveling at an inertial speed of 10.968 km/sec. The *Agena* separated from the spacecraft 2.6 min after injection, then performed a programmed 180-deg yaw maneuver and ignited its retro rocket. The retrorocket impulse was designed to eliminate interference with the spacecraft operation and reduce the chance of the *Agena* impacting the Moon. Tracking data indicated that the *Agena* missed the lunar surface by approximately 3630 km on the northern trailing edge.

Within an hour after injection, the spacecraft was receding from the Earth in almost a radial direction with decreasing speed. This reduced the geocentric angular rate of the spacecraft (in inertial coordinates) until, at 1.5 hr after injection, the angular rate of the Earth's rotation exceeded that of the spacecraft. This caused the Earth track of the spacecraft (Fig. 29) to reverse its direction from increasing to decreasing Earth longitude.

C. Midcourse Maneuver

Prior to the flight, *Ranger VI*'s selected impact site had been designated as the "most desirable" target for the launch day of January 30. Several of the factors leading to this choice are presented in Fig. 30, which presents the portion of the Moon available to impact in terms of the initial flight conditions[4].

The criterion of solar illumination of the impact site led to the choice of the zone between 50 deg and 80 deg from the subsolar point. In order to avoid loss of Earth-lock by the Earth sensor, the Earth-probe-near-limb angle was limited to a minimum of 15 deg, which selected a region within 75 deg of the sub-Earth point. The intersection of these two regions, for the February 2 encounter, located the general desirable impact area for that date.

In addition, the mission reliability would be increased if a terminal maneuver was not required to change the camera-pointing directions from the cruise-mode orientation. Generally, this situation is present in the early part of the launch period; the terminal maneuver is more desirable each day that the flight is delayed.

For the January 30, 1964 launch there were two nominal prelaunch aiming points selected, one at a latitude of 8.5 deg and a longitude of 21.0 deg in the Sea of Tranquillity and an alternate location in the Sea of Vapors at 13 deg latitude and 3 deg longitude. The alternate aiming point was selected because the Sea of Tranquillity was but 19 deg from the terminator and it was feared that the actual target uncertainties due to midcourse execution and orbit determination errors might dictate the selection of the alternate point, 35 deg from the terminator.

During the mission, it was clear from the first orbit that the premidcourse trajectory lay well within the 60 m/sec correction capability. Relative to the size of

[4]For convenience, these parameters are presented in a plane defined by the miss parameter **B**. This parameter is nearly a linear function of changes at injection conditions and is defined as the vector from the target's center of mass normal to the incoming asymptote of the osculating conic at closest approach to the target body. $\mathbf{S}_I$ is defined as a unit vector in the direction of the incoming asymptote. In the plane normal to $\mathbf{S}_I$, referred to as the **B**-plane, the unit vector **T** is parallel to the plane of the true lunar equator, and **R** completes a right-hand orthogonal system to describe **B**. This technique is rigorously described in JPL External Publication No. 674 (August, 1957), by W. Kizner.

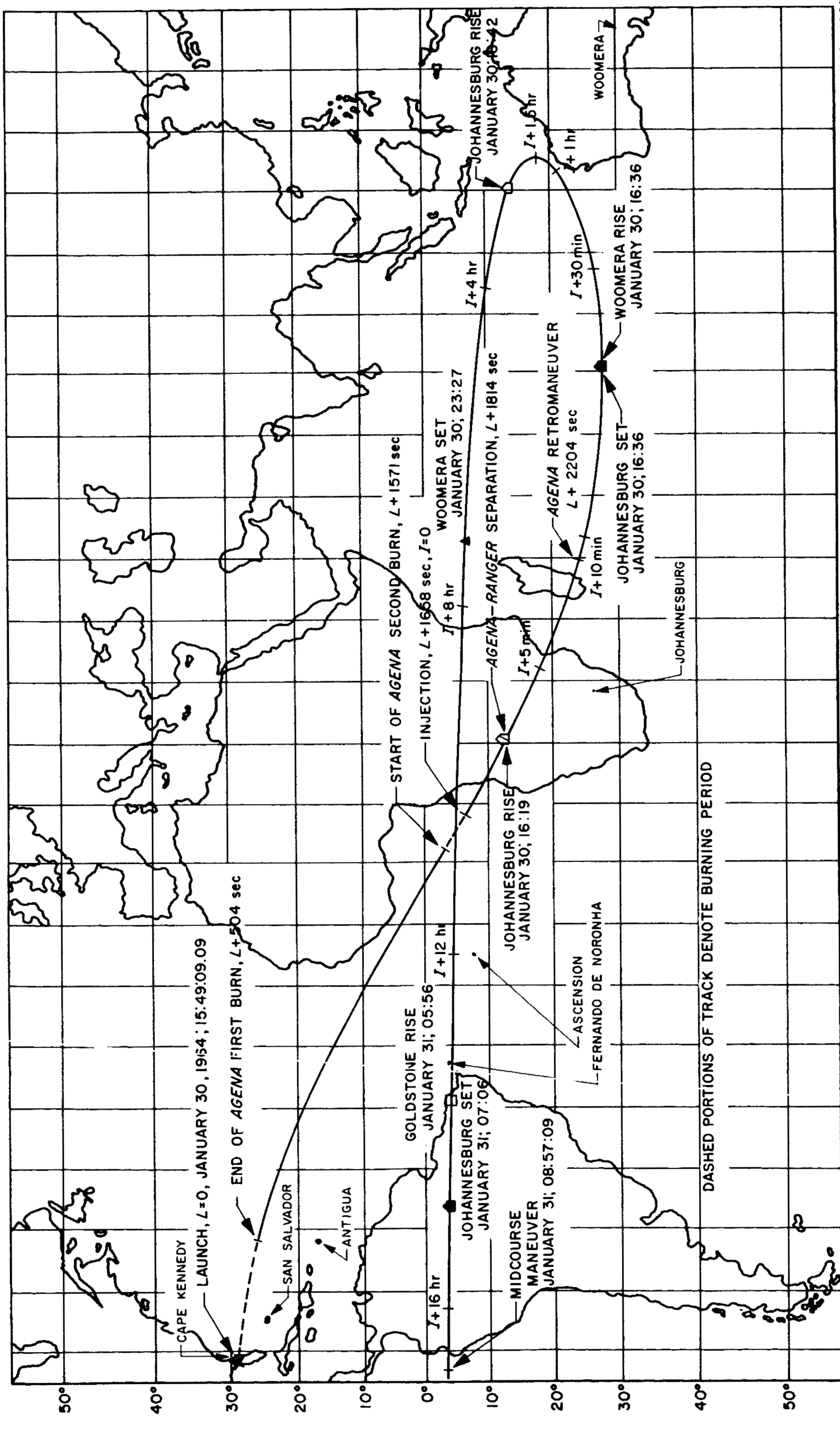


Fig. 29. Earth track of *Ranger VI* trajectory

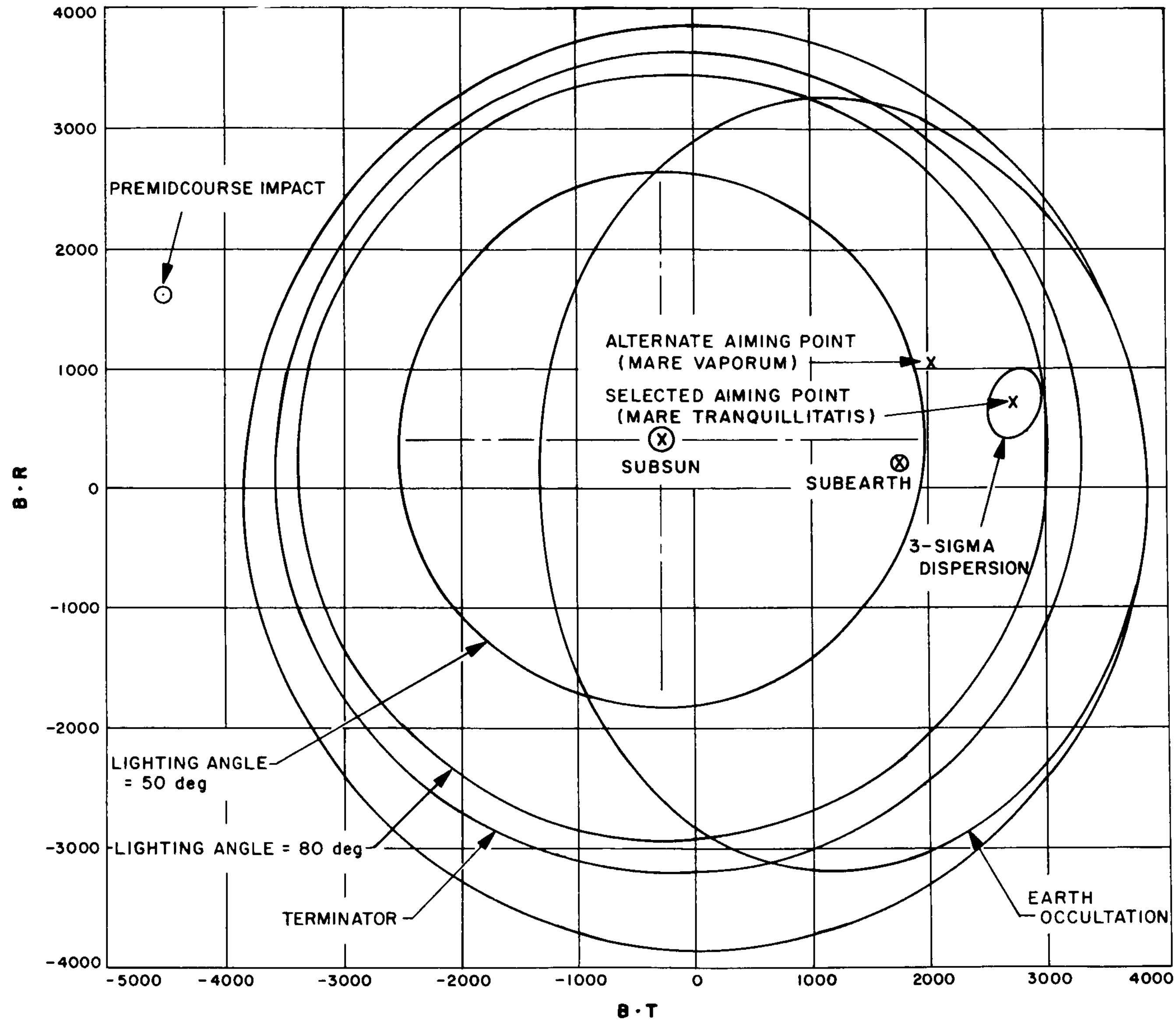


Fig. 30. Midcourse maneuver and site selection factors

the correction to be performed each orbit determination yielded the same result and the figures quoted here refer only to the nominal premidcourse orbit. This orbit provided the basis for the final computation of the midcourse maneuver. Table 13 details the estimate of the premaneuver orbit, the desired target parameters and the change in the target parameters required to attain the desired terminal conditions.

Table 13. Premaneuver and postmaneuver target conditions

Parameter	Premaneuver orbit	Prelaunch aiming point	Sea of Tranquility		Sea of Vapors	
			Desired	Perturbation required	Desired	Perturbation required
B · RT, km	1660	−536	−725	−2385	−1100	−2760
B · TT, km	−4518	498	2740	−7258	1950	−6468
T_f, hr	64.64	65.53	65.29	0.66	65.25	0.66

Also included in Table 13 is the prelaunch target point. It may be noted that this point is some 2300 km away from the primary post midcourse aiming point. The prelaunch target conditions are chosen so as to optimize the probability of impacting in the visible lighted portion of the Moon even in the event a spacecraft malfunction occurs which precludes the performance of the midcourse trajectory correction. Likewise, the flight-time of the prelaunch targeted trajectory is biased from the final desired to enhance the likelihood that the TV package will be activated automatically at an acceptable time even if no command capability is ever established with the spacecraft.

Nominally the flight-time is adjusted during the midcourse maneuver so that impact will occur 15 min after the TV backup clock applies full power to the F-channel cameras. If, however, the roll axis turns to within 40 deg of the probe-Earth line at any time during the performance of the nominal maneuver, consideration is given to modifying the flight-time from the nominal desired to a value such that an antenna constraint violation does not occur or at least occurs in a minimum fashion. For *Ranger VI* it was considered acceptable to have the cameras automatically activated anywhere in the interval 11–34 min before nominal impact. The unmodified maneuver required yielded a severe antenna constraint violation; the roll axis entered the null cone 50 sec after the start of the pitch turn and did not emerge until the motor burn had been completed and Sun reacquisition had started. Moreover, adjusting the flight-time within the acceptable window yielded no appreciable improvement. Because of the ineffectiveness of any possible modification and because it was anticipated that most of the telemetry could be received in spite of the violation, the unmodified flight time of 65.129 hr from injection to impact was selected. This completed the postmaneuver aiming point selection.

1. Use of RTC-8

Consideration was given to the possibility of sending RTC-8, interrupting the midcourse sequence, in the event a malfunction occurred during the turning sequence of the midcourse maneuver. This was a particularly difficult decision to make in real-time for the maneuver, once stopped, could not be attempted again; the spacecraft would be committed to the premidcourse trajectory.

The premidcourse trajectory took the probe on a west-side flyby of the Moon having a closest approach distance of 2550 km (altitude of 812 km) at a subprobe point of latitude −10 deg, longitude −168 deg. On this trajectory, a very satisfactory mapping of the unexplored portion of the Moon could be obtained, although the pictures would by no means have met the mission objectives of high resolution pictures. In order to obtain this back-side mapping, however, a terminal maneuver would have had to be performed. Without this reorientation of the spacecraft, the camera field of view would not have intercepted the Moon. If a real malfunction occurs during the midcourse, the likelihood of being able to successfully perform a terminal maneuver is cast in grave doubt; except in unusual circumstance then, RTC-8 is not to be sent if a terminal maneuver is obligatory on the premidcourse trajectory. Because of these considerations it was decided that no postmidcourse trajectory could be less advantageous than the existing course and the decision was made not to send RTC-8 under any circumstances.

2. Midcourse Maneuver Execution

Approximately 3 hr before the initiation of the maneuver, at 0830:00 GMT January 31, the final computation was made, using the latest estimate of the injection conditions. The important parameters of the required maneuver are given in Table 14.

The premidcourse and postmidcourse trajectories near the Moon and the impact sites are shown in Fig. 31.

The midcourse motor was ignited at 0857:08 GMT on January 31, 1964 at which time the spacecraft was at a geocentric distance of 170,000 km and traveling with an inertial speed of 1.891 km/sec relative to Earth. At the end of a 67-sec burn duration of the midcourse motor, the geocentric distance had increased to 170,125 km and the inertial speed relative to Earth was reduced to 1.852 km/sec. Telemetry data received at the Goldstone tracking station and relayed to the operations facility at JPL gave direct readings on the duration and polarity of the turns and the length of the motor burn. These measurements observed in real time during the performance of the maneuver all had the correct polarity and, within the accuracy of these measurements, were of the

Table 14. *Ranger VI* commanded maneuver

Maneuver	Magnitude	Duration	Initiated at GMT
Roll turn	−11.96 deg	54	083044
Pitch turn	−70.98 deg	328	084009
Velocity increment	41.27 m/sec	67	085709

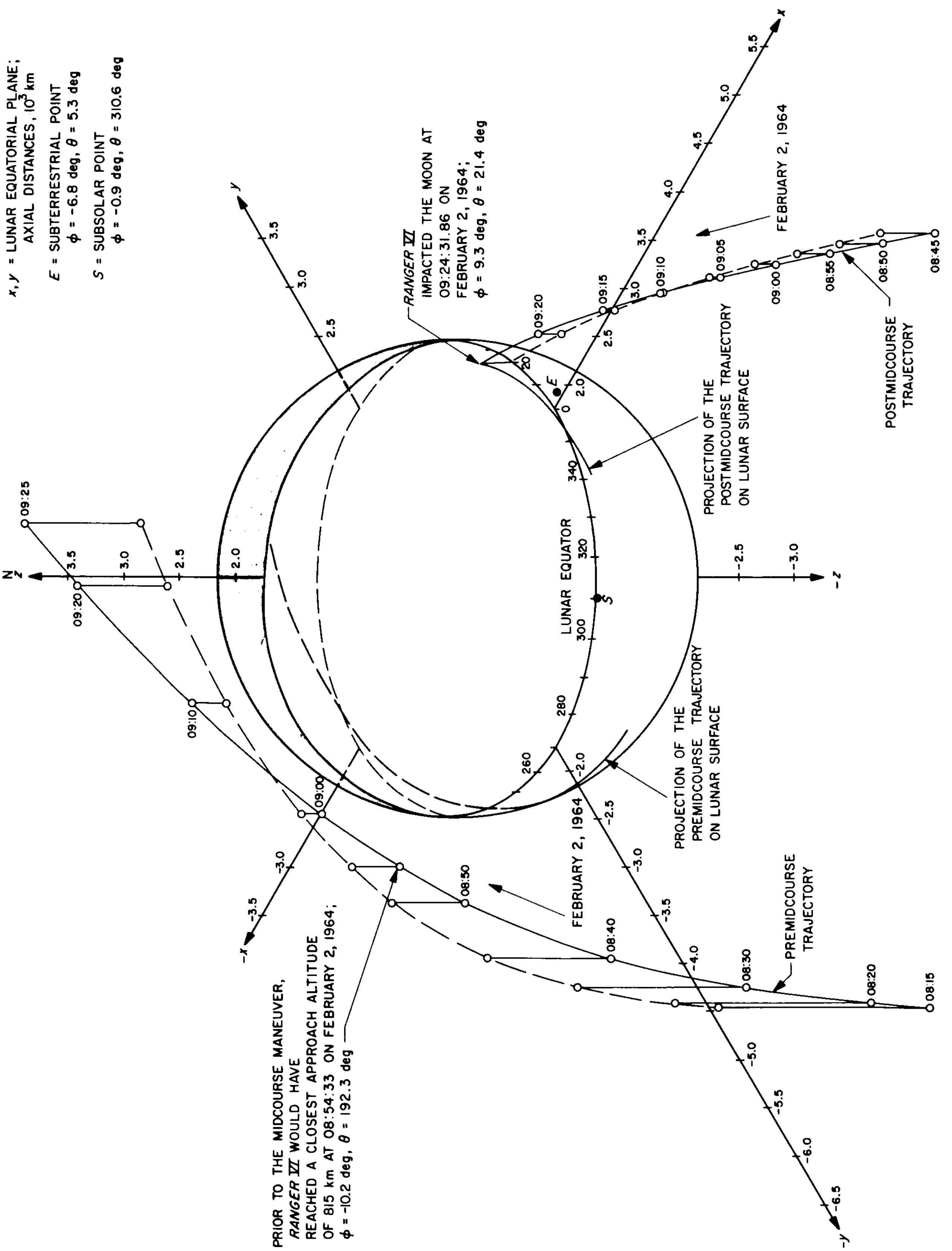


Fig. 31. Ranger VI premidcourse and postmidcourse maneuver encounter trajectories

exact commanded duration. This, coupled with the real-time doppler reduction, gave almost immediate verification that the maneuver had been performed correctly.

D. *Postmidcourse Cruise*

Following the midcourse maneuver, the spacecraft reacquired the Sun and Earth, thus returning to the cruise mode. At about 59 hr from injection and at a geocentric distance of 364,000 km, the spacecraft inertial speed relative to the Earth reached a minimum value of 1.029 km/sec. At this point, the spacecraft was about 38,090 km from the lunar surface with an inertial speed of 1.305 km/sec relative to the Moon. Because of the lunar gravitational field, the spacecraft velocity then began to increase.

Postmidcourse tracking data up to within 1 hr before impact were analyzed and resolved the lunar encounter conditions to a high degree of accuracy with the lunar impact to occur at 9.39 deg N latitude and 21.51 deg E longitude with a flight time from injection of 65.59 hr. The encounter conditions combined with the corresponding postmidcourse initial conditions are presented in Table 15. The space trace of the trajectory from injection to impact is given in Fig. 32.

E. *Terminal Maneuver Considerations*

At the time the terminal maneuver was being considered, the following data represented the best estimate of the impact parameters:

Impact latitude = 8.00 deg
Impact longitude = 20.11 deg
GMT of impact = 0924:40 on February 2, 1964
Automatic camera turnon = 0911:12 on February 2, 1964

Figure 33 depicts the impact geometry with the cameras in the cruise-mode orientation. The **C** vector represents the central pointing direction of the four P cameras, *A* and *B* represent the pointing directions for the 25 deg and 8.4 deg field of view F cameras. A terminal maneuver, if performed, would have aligned the **C** vector with the impact velocity vector. In the cruise mode the **C** vector was 12.5 deg from the velocity vector, and with the path angle shown of 42.5 deg, an impact velocity of 2.66 km/sec and a shutter speed of 2 m/sec, the resultant blurring due to camera motion was 1.85 m. That is, the center of the field of view at the time the shutter closed would be observing a point on the surface 1.85 m away from the point viewed at the time the shutter opened.

Table 15. Orbit used for maneuver determination and last postmidcourse orbit

Parameter[a]	Premidcourse	Postmidcourse
Epoch	January 30, 1964; 1616:41 GMT	January 31, 1964; 0858:21 GMT
Earth—fixed sphericals		
R	6567.0215 km	170130.42 km
ϕ	−6.8722887 deg	−3.5973897 deg
θ	7.473279 deg	276.78513 deg
V	10.551346 km/sec	12.154424 km/sec
γ	1.2443904 deg	8.5337484 deg
σ	118.93418 deg	271.02087 deg
Inertial cartesian coordinates		
X	6103.6462 km	−169769.50 km
Y	2292.1199 km	−2954.4363 km
Z	−785.78784 km	−10674.843 km
X	3.7715114 km/sec	−1.8068983 km/sec
Y	8.9531506 km/sec	−0.39521089 km/sec
Z	−5.0943372 km/sec	0.10056300 km/sec
Impact parameters		
Impact epoch	February 2, 1964; 0854:31 GMT	February 2, 1964; 0924:32.18 GMT
Selenocentric altitude	813 km	—
Selenocentric latitude	−9.6 deg	9.39 deg
Selenocentric longitude	191.1 deg	21.51 deg
Time of flight from injection	64.6318 hr[b]	65.590004 hr[e]
B	4813 km[c]	2859 km[f]
B·TT[d]	−4518 km	2754 km
B·TR[d]	1660 km	−768 km

Definition of terms

R	Probe radius distance, km
ϕ	Probe geocentric latitude, deg
θ	Probe east longitude, deg
V	Probe Earth-fixed velocity, km/sec
γ	Path angle of the probe Earth-fixed velocity vector with respect to local horizontal, deg
σ	Azimuth angle of the probe Earth-fixed velocity vector measured east of true north, deg
x, y, z	Vernal equinox Cartesian coordinates in a geocentric equatorial system. The origin is the center of the celestial body. The principal direction (X) is the vernal equinox direction of date, and the principal plane (X, Y) is the Earth equatorial plane of date. Z is along the direction of the Earth's spin axis of date, km.
x, y, z	First time derivatives of X, Y, and Z respectively: i.e., Cartesian components of the probe space-fixed velocity vector, km/sec.

[a] See definition of terms.
[b] 1σ uncertainty of 66 sec.
[c] 1σ uncertainty of 79.5 km.
[d] B·TT and B·TR are referenced to the true lunar equator.
[e] 1σ uncertainty of 1.6 sec.
[f] 1σ uncertainty of 21.3 km.

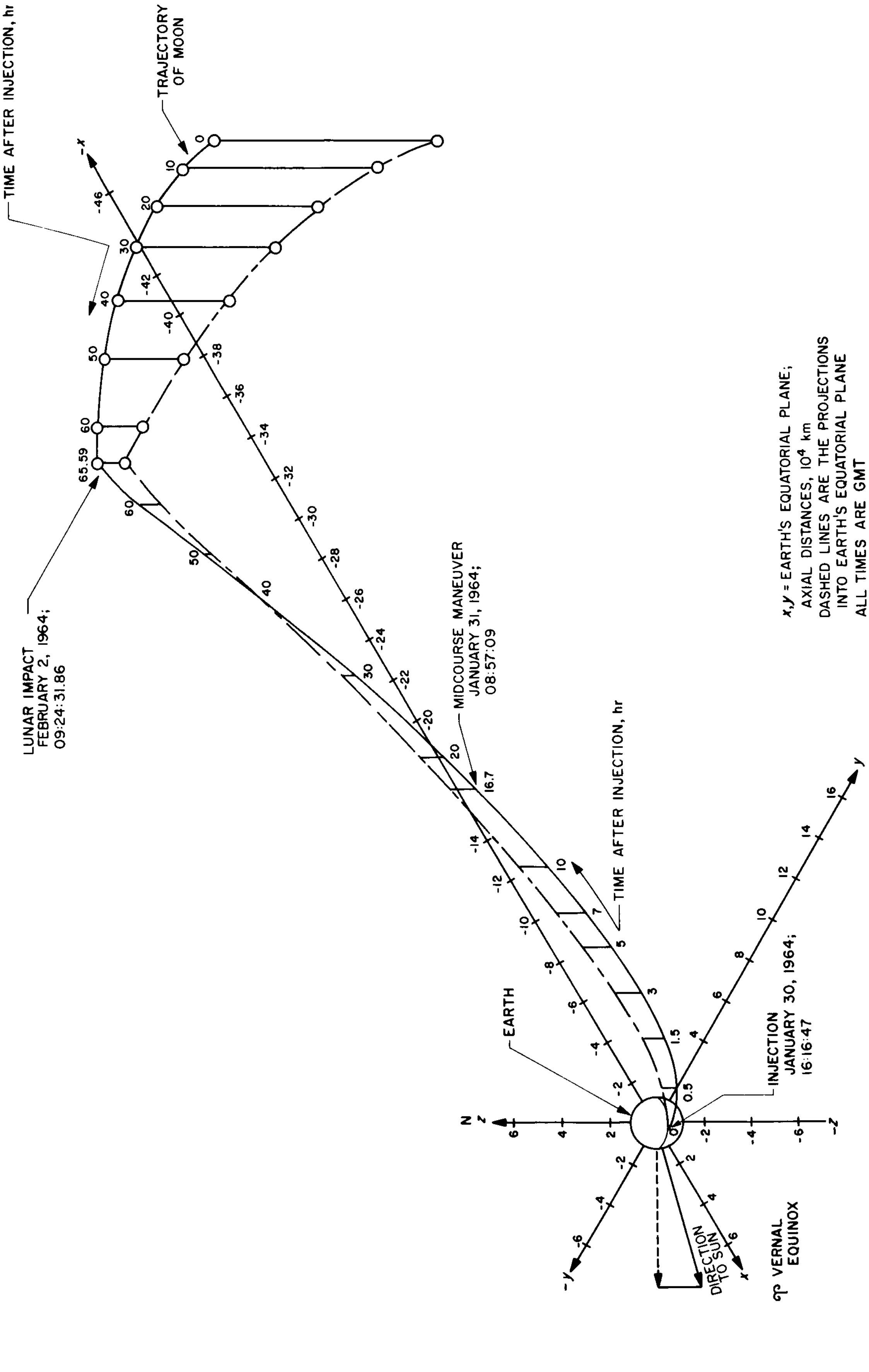


Fig. 32. *Ranger VI transfer trajectory*

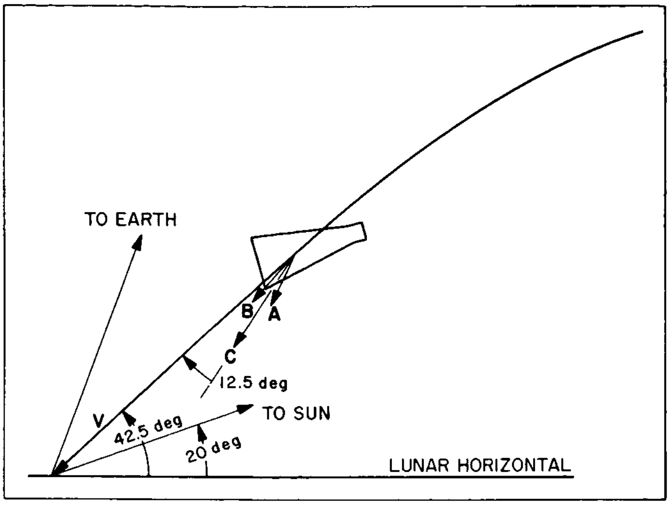


Fig. 33. No-terminal approach geometry

This is an acceptable level of blur; an amount 3 times this figure could probably be tolerated and still meet the mission objectives.

In addition, the scientific experimenters pointed out that the no terminal viewing geometry was such as to allow direct observation of shadows from objects having a 30 deg slope from the horizontal. If the cameras were aligned with the velocity vector, this situation would have been destroyed. These considerations coupled with the possibility of a spacecraft failure during the terminal maneuver led to the decision not to perform a terminal maneuver.

F. Encounter

During the encounter, the spacecraft was subjected to increasing acceleration due to the pull of the lunar gravity field.

At an hour before impact, the speed of the probe relative to the Moon had increased to 1.610 km/sec and was at a lunar altitude of 6180 km.

The spacecraft approached the Moon in direct motion along a hyperbolic trajectory with the incoming asymptote direction at an angle of −11.440 deg to the lunar equator and with the orbit plane inclined 19.80 deg to the lunar equator. At 0924:33 GMT on February 2, 1964, *Ranger VI* crashed onto the Sea of Tranquillity at selenocentric latitude and longitude of 9.33 and 21.52 deg, respectively, with an impact speed of 8.656 km/sec and at a path angle of −48.49 deg.

The trace of the trajectory on the lunar surface from injection to impact is given in Fig. 34, while the traces of the lunar approach portions of the premidcourse and postmidcourse orbits are illustrated in Fig. 31.

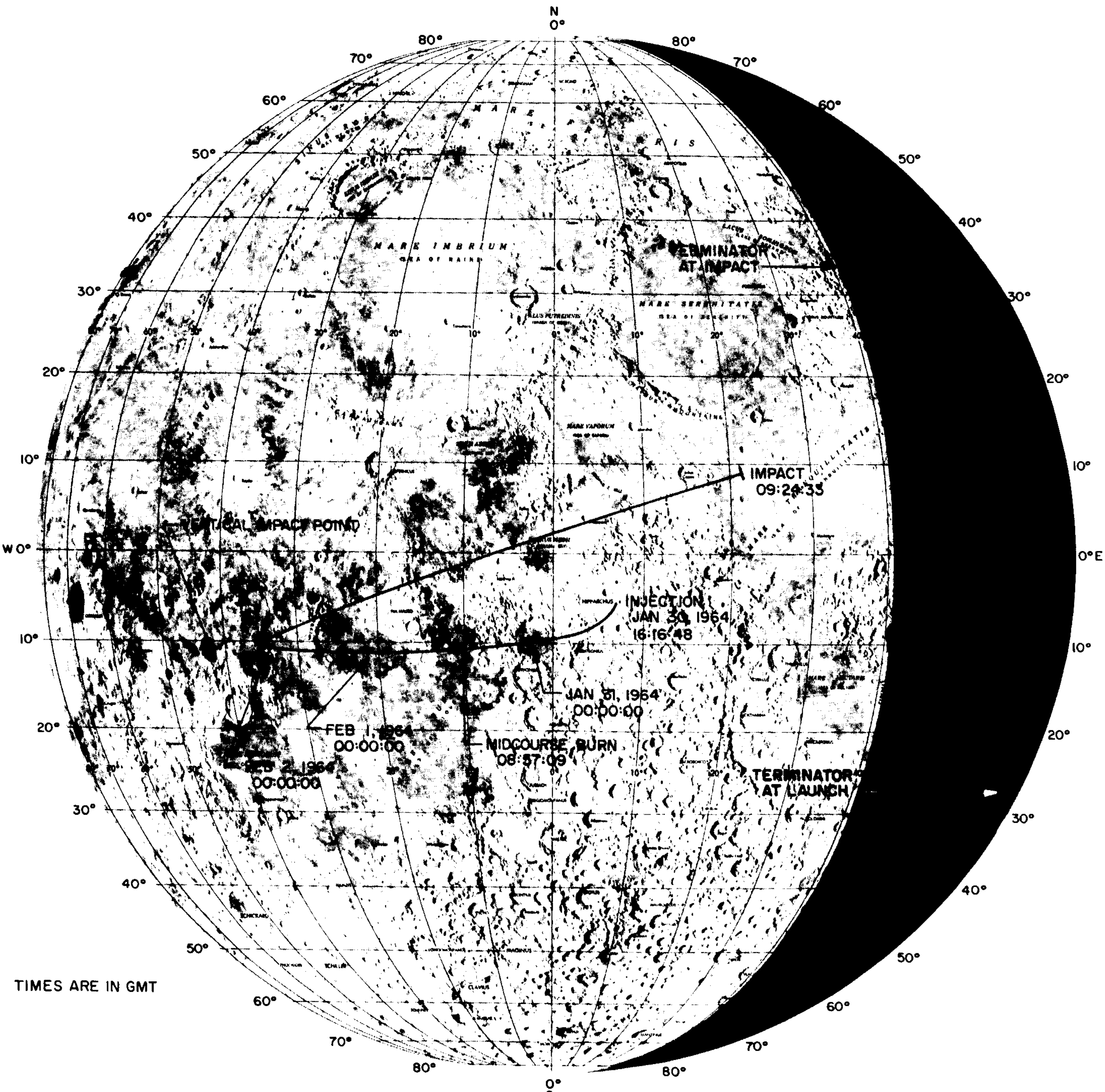


Fig. 34. Lunar track of *Ranger VI* trajectory

VIII. TELEVISION SUBSYSTEM FAILURE

The *Ranger VI* did not accomplish its mission objective of obtaining high-resolution television pictures of the lunar surface, although the spacecraft successfully accomplished all flight functions and achieved a lunar impact close to the desired site. An investigation was conducted and, as a result, certain changes were recommended and carried out.

A. Anomaly Indications

All prelaunch, launch, and flight measurements and indications were nominal except for the following:

1. Seven temperature measurements were lost.
2. The Earth sensor's running temperature was above nominal.
3. The TV subsystem stabilized at higher-than-expected temperature.
4. The TV cruise mode telemetry (channel 8) came on at an unscheduled time (booster staging), remained on for 67 sec, then turned off (Table 16). It then came on again at the scheduled time (17 min after separation from the *Agena*) and apparently functioned properly.
5. Both battery voltage readings were low, indicating that more than the cruise-mode-telemetry power load existed during part or all of the inadvertent turnon period.

Telemetry indications at the 16-, 32-, 48-, and 64-hr postseparation times indicated that the backup clock was functioning correctly. F-channel turnon was accomplished at the scheduled time as indicated by a drop in the F-battery voltage.

The response to the RTC-7 command to set the P channel to warmup was normal. This command would have set the F channel to warmup also, if the clock had not already done so. A blip on the B-20 telemetry channel indicated that the RTC-7 command had been received by the spacecraft and the relay coil in the command decoder that actuates the command switch in the TV subsystem had been energized. A drop in the P-channel battery voltage, indicated by a telemetry measurement, confirmed that a load had been applied to the P battery.

Table 16. Channel 8 telemetry during inadvertent turnon

Point	Telemetry data	Telemetered value			
		Frame 1[a]	Frame 2[a]	Frame 3	Frame 4
1	"P" low-current regulator input, v	—	30.6	33	33.2
2	"P" low-current regulator output, v	—	26.8	27.5	27.5
3	"F" battery internal temperature, °F	69	60.4	68.8	68.8
4	"P" battery internal temperature, °F	69	69	69.9	69.9
5	"F" battery unregulated output, v	33.2	31.4	33.6	33.6
6	"P" battery unregulated output, v	31.4	30.3	33.6	33.6
7	Top-hat −Y axis temperature, °F	49	44	65	65
8	Shroud −Y axis temperature, °F	64	56	64	65
9	TV clock indicator (0 = pre-16 hr)	0	0	0	0
10	Spare	0	0	0	—
11	"P" 5-min. accumulator (0 = pre-warmup)	0	0	0	—
12	Full-scale reference	4.65	4.65	4.65	—
13	Zero reference	0	0	0	—
14	Frame reference	1.15	1.15	1.15	—
15	Frame reference	1.15	1.15	1.15	—

[a] Frames 1 and 2 were noisy and difficult to read; values given are 3-reading averages.

Both F and P channels should have gone into full power 5 min after each had been turned on for warmup. No video or RF transmissions were received by either of the two ground recording stations monitoring the TV channels. Two more RTC-7 commands were sent to the spacecraft and verified, but no 90-point telemetry or video signals were received and the spacecraft impacted on the Moon with no video or RF having been received.

B. Investigation Apparatus

On February 2, 1964 an investigating committee, or technical review team, was formed of cognizant personnel of JPL and the TV subsystem contractor. The committee was under the direction of the TV subsystem Project Engineer assisted by the subsystem contractor's Project Manager. The committee was charged with investigating all phases of the program to determine all possible causes leading to the TV malfunction, and establishing preventive measures. This committee's initial tasks included telemetry data search and analysis, spacecraft construction review, and an analysis of the environment and conditions of the flight.

On February 4, 1964 a Section Chiefs' Committee was established to provide audit, extension, and suggestions to the investigation. This committee conducted a number of general reviews, and contributed to the studies and final report of the technical review team, which was completed on February 14, 1964.

On February 3, NASA had established an independent board to review the findings of the technical review team and to consider management decisions arising from the investigation. The report of this board was filed March 17, 1964. In April and May 1964, the subcommittee on NASA Oversight of the House Committee on Science and Astronautics conducted an investigation of project *Ranger*. The subject of this effort, similarly, was management; the Committee's report was released June 16, 1964.

C. Failure-Mode Conclusions

After exhaustive analysis and experimentation, it was not possible to determine any one single simple failure or malfunction that would adequately explain all the observed events and indications. However, the following conclusions were drawn from the pertinent flight data and supporting information:

1. There was an inadvertent turnon of both of the TV channels during the boost phase. This turnon caused high-voltage arcing because of the critical pressure conditions existing at that time in the flight, and burned out the transmitter power supplies and possibly some, if not all, of the camera circuitry.
2. The telemetry channel 8 turnon at booster jettison (1551:30 GMT) was probably due to the same cause that turned on the TV subsystem.
3. The exact mechanism for the TV subsystem and the channel 8 turnon and turnoff was not defined. However, a transient, a short, or a static discharge are all possible initial causes of turnon.

D. Results

In accordance with the recommendations of the technical review and the other reviews of the *Ranger VI* failure, the *Ranger B* (subsequently *Ranger VII*) attempt was rescheduled to permit modifications of design and procedure. These recommended modifications included: (1) reducing the transient sensitivity of the TV control circuits and the mechanical sensitivity of the TV subsystem; (2) simplifying TV-subsystem circuitry and command usage to increase reliability; (3) improving telemetry coverage of the TV subsystem, and modifying its thermal control coatings, and (4) increasing the emphasis on quality control. In addition, the midcourse-motor temperature transducer was eliminated to protect other measurements on the same temperature bridge.

Thus the results of the *Ranger VI* flight anomalies, major and minor, were the modifications which contributed to the subsequent *Ranger VII* success.

APPENDIX A
Spacecraft Mission Events

Event	Nominal mission time	Actual GMT
January 30, 1964		
Turn on spacecraft external power.	$T-210$ min	1022
Transmit two RTC-0 commands and one RTC-2.	$T-125$ min	1221
Preset antenna hinge to 135 deg.		
Frequency Report:	$T-90$ min	1300
Transponder frequency on auxiliary oscillator drive: 960.028625 Mc.		(1249)
Ground transmitter frequency at 0 SPE volts: 890.047290 Mc.		(1246)
Transponder frequency at 0 SPE volts: 960.051034 Mc.		(1248)
Case 2 temperature: 75° F.		(1252)
Frequency Report:	$T-35$ min	1428
Transponder frequency on auxiliary oscillator drive: 960.028599 Mc.		(1408)
Ground transmitter frequency at 0 SPE volts: 890.047290 Mc.		(1418)
Transponder frequency at 0 SPE volts: 960.051034 Mc.		(1419)
Ground transmitter frequency corresponding to average no-signal transponder SPE voltage: 890.047860 Mc.		(1415)
Transponder frequency for above: 960.051591 Mc.		(1416)
Transponder average no-signal SPE voltage: −0.0078 v.		(1412)
Case 2 temperature: 76° F.		(14:15)
Rate 4 (H-9) sync pulse ends.	$T-10$ min	1515:27
Spacecraft to internal power.	$T-5$ min	1543:50
Release inhibit, start launch counter.	$T-2$ min	1547:02
Clear relays event (B-2-1 blip).	$T-1$ min	1548:01.8
Liftoff (umbilical removed).	L	1549:09.1
Monitor voltages increase 5% (OSE load removed).		

Event	Nominal mission time	Actual GMT
Address 66 frequency drops 2.8 cps (OSE load removed).		
Squib firing assembly armed by G-/switch (no telemetry).	$L+2$ min	
Start of 0.75-sec dropout of 960-Mc RF link.	$L+140$ sec	1551:29.3
Channel 8 telemetry on.	(anomaly)	1551.29.7
Channel 8 telemetry off.	(anomaly)	1552:37.5
Shroud ejection, antenna coupler removed.	$L+306$ sec	1554:15.1
Transmitter-power-up command (B-2-1).	$L+23$ min	1612:01.8
Plate voltage from 150 to 250 v, omni drive to 35 dbm.		
Spacecraft/*Agena* electrical disconnect (remove spacecraft telemetry from *Agena* carrier).	$L+1809$ sec	1619:18.2
Spacecraft/*Agena* separation.	$L+1814$ sec	1619:23.2
Start hydraulic backup command timer.		
Start backup turnon timer.		
Relays unclamped.		
Spacecraft receiver in lock with DSIF 51 transmitter.		1620:31
Arm squib firing assembly (backup command, no telemetry).	$S+2.5$ min	1622
Command TV cruise telemetry turnon.	$S+17$ min	1636:34
Channel 8 telemetry on.		
Spacecraft receiver out of lock.		1642:13
Two-way lock with DSIF 41 transmitter.		1644:01
Command solar-panel extension (B-2-1).	$L+60$ min	1649:01.8
Fire solar-panel squibs (B-2-2, B-2-3).		
Solar panels extended (B-2-4).		1650:06
Command Sun acquisition.	$L+63$ min	1652:01.3
DC power on, extend antenna, start Sun acquisition.		
B-2-1 blip readout.		1652:01.8

APPENDIX A (Cont'd)

Spacecraft Mission Events

Event	Nominal mission time	Actual GMT
Spacecraft on solar power.		1653:54
Sun acquisition complete.		1655:30
Antenna at acquisition angle (135 deg).		1657:13
Command solar panel extension (backup, no effect).	S + 45 min	1705
Command Sun acquisition (backup, no effect).	S + 60 min	1720
Command Earth acquisition (B-2-1).	L + 211 min	1920:02
Earth-sensor power on, start roll search.		
Earth-sensor threshold signal.		1940:40
Earth acquisition complete.		1945:00
Initiate RTC-0. (Command sequence stopped because of marginal ground transmitter power.)	L + 270 min	2025:00
Initiate RTC-0.	L + 321 min	2108:00
Initiate RTC-0.		2110:00
Initiate RTC-3.		2112:00
RTC monitor blip (B-20).		2112:40
Switch spacecraft transmitter from omni to high-gain antenna, high-gain to drive to normal 25 dbm.		
January 31, 1964		
Initiate RTC-0.	M − 70 min	0720:00
Initiate RTC-0.		0722:00
Initiate SC-1 (Roll turn: 54 sec, neg.).		0724:00
Stored command readout: 25-0450-0 (B-20).		0724:40
Attitude-control capacitor cycling command (B-2-1).		0724:44
Initiate SC-2 (pitch turn: 328 sec, neg.).		0726:00
Stored command readout: 35-0372-0.		0726:40
Initiate SC-3 (velocity increment: 135.1 fps).		0728:00
Stored command readout: 03-2610-1.		0728:40
Backup clock 16-hour pulse.	S + 16 hr	0819:54
Initiate RTC-3.	M − 10 min	0820:00
RTC monitor blip (B-20).		0820:40
Switch spacecraft transmitter from high-gain to omni; low-gain drive to 35 dbm.		
Initiate RTC-4.		0830:00
RTC monitor blip (B-20).	M	0830:40
Start maneuver counter.		
Command attitude control to maneuver mode, start roll turn (B-2-1).	M + 5 sec	0830:45
Earth-sensor power off, start antenna exit, autopilot on, start negative roll turn.		
Command roll turn stop after 54 sec (B-2-1).	M + 59 sec	0831:39
Stop roll turn.		
Antenna at exit angle.		0832:38
Command pitch turn start (B-2-1).	M + 9 min, 30 sec	0840:09
Start negative pitch turn.		
To power sharing mode (panels and battery).		0843:58
Command pitch turn stop after 328 sec. Telemode II (B-2-1).	M + 14 min 58 sec	0845:38
Stop pitch turn.		
Telemode II.		
Command motor ignition (B-2-1), start shutoff computer countdown.	M + 26 min 30 sec	0857:08
Fire motor ignition squibs (B-2-2, B-2-3).		
Motor ignition.		
Jet-vane control, gate-accelerometer pulses to CC&S.		
Temperature bridges "G" and "D-1," power supply shorted out.	(Anomaly)	0857:20

APPENDIX A (Cont'd)

Spacecraft Mission Events

Event	Nominal mission time	Actual GMT
Command motor cutoff after 1351 accelerometer pulses (B-2-1).	*M* + 27 min 37 sec	0858:17
Fire motor cutoff squibs (B-2-2, B-2-3).		
Motor cutoff.		
Command Sun reacquisition, Telemode III (B-2-1).	*M* + 30 min	0900:39
Autopilot *off*, start Sun reacquisition, start antenna to reacq. angle.		
Switch to solar panel power.		0901:54
Antenna at reacquisition angle (122 deg).		0902:50
Sun reacquisition complete.		0905:20
Command Earth reacquisition (B-2-1).	*M* + 58 min	0928:39
Earth sensor power on, start roll search.		
Earth sensor threshold signal.		
Earth sensor threshold signal.		0928:45
Earth reacquisition complete.		0929:42
Initiate RTC-3.	*M* + 75 min	0944:00
RTC monitor blip (B-20).		0944:40
Switch transmitter from omni to high-gain antenna; high-gain drive to 25 dbm.		
February 1, 1964		
Backup clock 32-hr pulse.	*S* + 32 hr	0020:11
Backup clock 48-hr pulse.	*S* + 48 hr	1620:32
February 2, 1964		
Initiate RTC-0.	*I* − 73 min	0811:00
Initiate RTC-0.		0813:00
Backup clock 64-hr pulse.	*S* + 64 hr	0820:41
Backup clock commands F-channel turnon to start sequencer, initiate warmup.	*S* + 64¾ hr	0905:42
Initiate RTC-7.		0908:00
RTC monitor blip (B-20).	*I* − 15 min	0908:41.7
Step command switch; start P-channel sequencer, initiate warmup.		
P-channel accumulator 30-sec pulse.		0909:15
F-channel full power command (no verification).	*S* + 64 hr 50 min (Anomaly)	(0910:40)
P-channel full power command (no verification).	*I* − 10 min (Anomaly)	(0913:42)
Initiate RTC-7.		0915:29
RTC monitor blip (B-20).		0916:10.7
Step command switch to emergency mode (no verification).	(Anomaly)	
Initiate RTC-7.		0919:21
RTC monitor blip (B-20).		0920:02.7
Step command switch to *Emergency Off* mode (no verification).	(Anomaly)	
Impact (end of signal).	*I*	0924:33.1

LEGEND

T = predicted time of launch
L = liftoff, launch
S = spacecraft/Agena separation
M = midcourse maneuver (RTC-4 receipt)
I = lunar impact

OSE: operational support equipment
RTC: real-time command
SC: stored command
( ) indicates GMT unverified

APPENDIX B

Spacecraft Improvements for *Ranger VI*

1. Block III Design — The Product of
   a. Initial *Ranger VI* design
   b. Product improvement program after RA-4 design evaluation vehicle
   c. Design review during December and January
   d. Initial design frozen — changes by ECR only

2. Design
   a. Circuit improvements
   b. Redundancy

3. Final Design Review
   a. Prior to start of flight acceptance testing
   b. Mechanization of approved changes, parts and materials usage, Q.A.

4. Qualification
   a. More thorough margin tests
   b. Updated environmental specs
   c. Approved test specs
   d. Written acceptance criteria

5. Fabrication of flight hardware
   a. Parts quality and screening
   b. Release drawings
   c. Increased inspection (quality and quantity)
   d. Documentation
   e. Failure reports

6. Communications
   a. Redesign directional coupler
   b. Reduce noise sensitivity
   c. Add extra ¼-watt cavity
   d. Connector change for multiplication

7. Central computer and sequencer
   a. Re-layout of some circuit boards
   b. Redesign critical circuits
   c. Delete unnecessary circuits
   d. Establish rigorous margin tests

8. Attitude control
   a. Dual gas system, welded construction
   b. Narrower field on Earth sensor
   c. Re-layout of electronics
   d. Remove autopilot integrator
   e. Eight pre-set hinge angles
   f. RTC-8 Sun reacquire

9. Command
   a. Redesign some elements
   b. Reactivate additional command functions (Earth acquisition, Sun acquisition, inhibit TV clock, maneuver inhibit)

10. Structure
    a. Basic hex aluminum instead of magnesium
    b. Mono ball panel support and hydraulic damper
    c. Pin-puller modified and electrically isolated from frame
    d. Hydraulic timer backup for most launch-phase events

11. Pyrotechnic network
    a. Separate squib circuits off each battery
    b. Open returns after firing

12. Data encoder
    a. Redesigned VCO's
    b. Redesigned backup clock

13. Power
    a. Mark IV solar panels
    b. Second battery: 45 amp-hr, 26 lb
    c. Redesign power switch and logic

14. Propulsion
    a. Replace pressure regulator
    b. O-ring seals change to metallic seals
    c. Delete visual pressure gages
    d. Improve manifold to tank attachment

15. Television
    a. Two independent systems
    b. Dynamic range extended to 20–2600 ft-L by using *f*/1 lenses on one full scan and two partial scans
    c. Add backup clock turnon
    d. Increase camera-erase frequency
    e. Pressurize dummy load
    f. Shutter improvement

APPENDIX C
Spacecraft Configurations and Interfaces

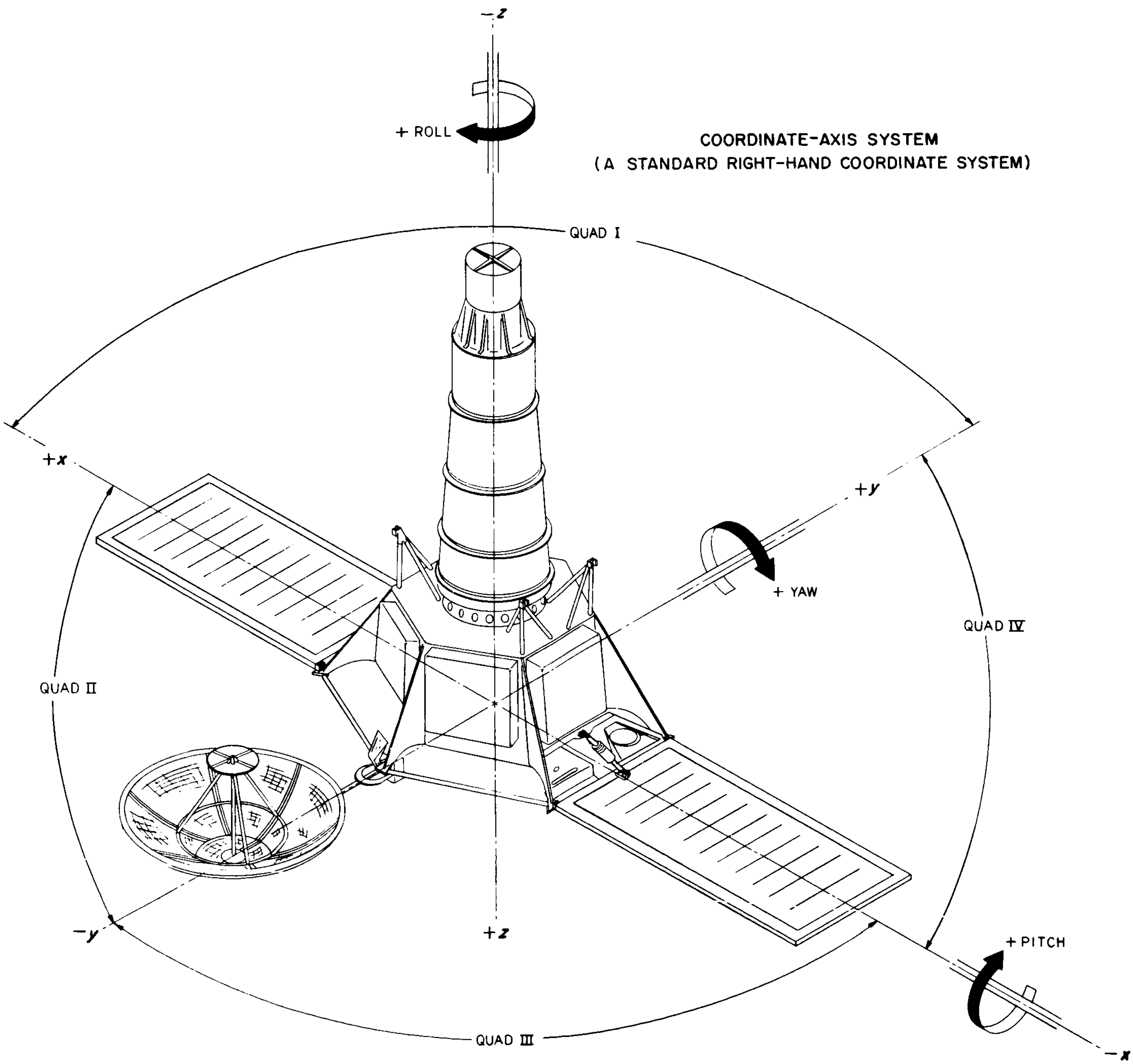


Fig. C-1. Coordinate axis system

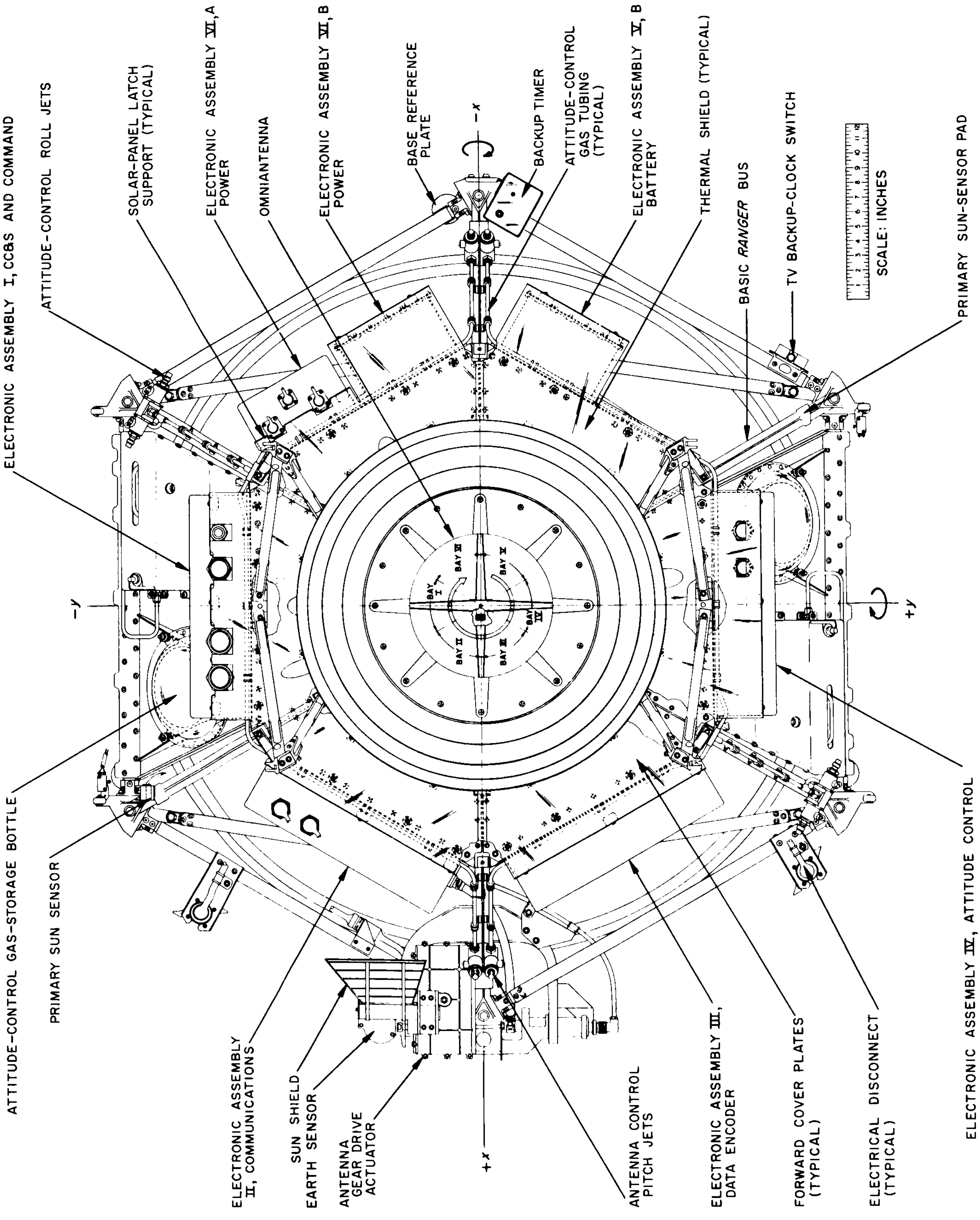


Fig. C-2a. Spacecraft configuration, *Ranger VI*, top view

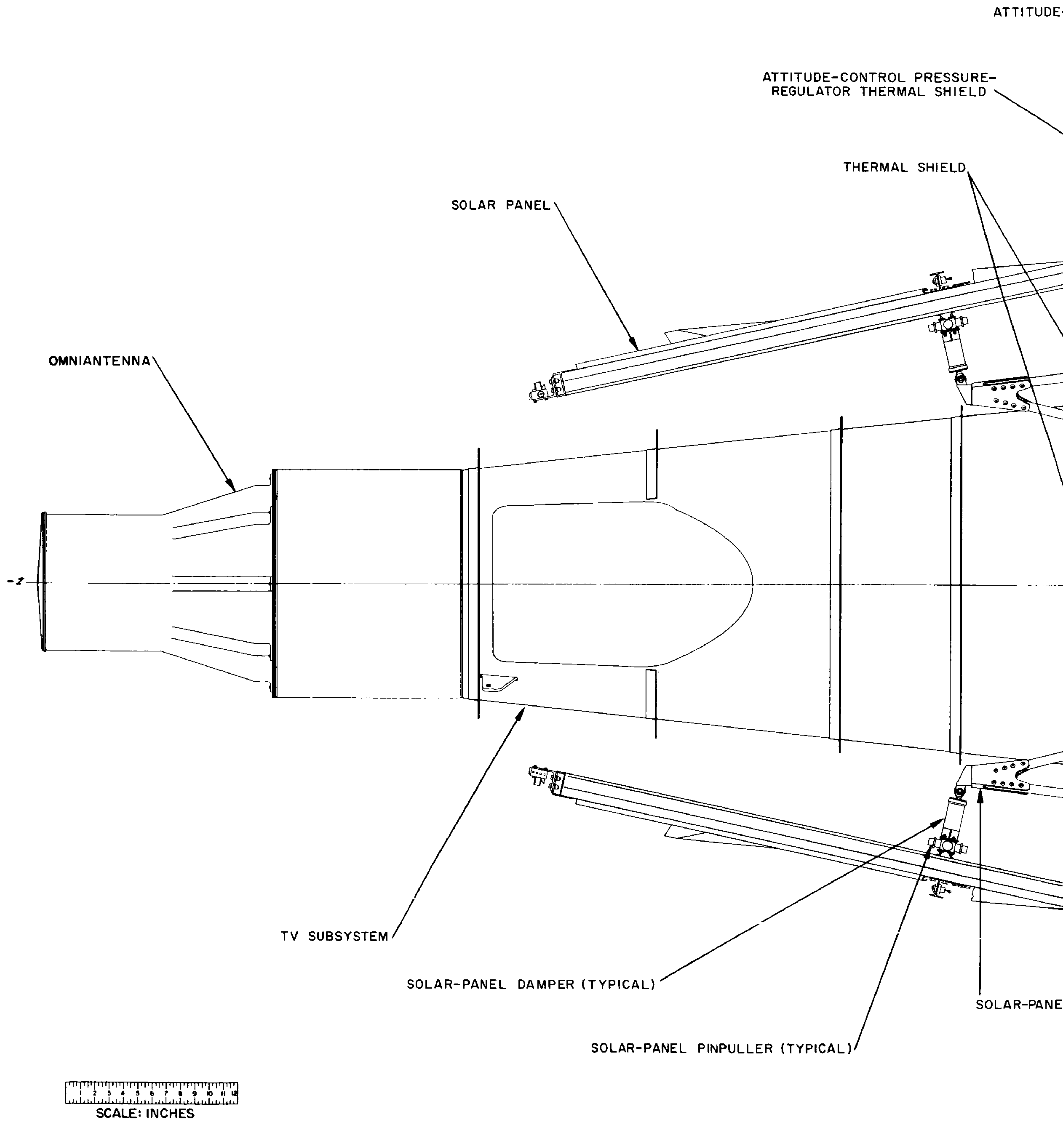
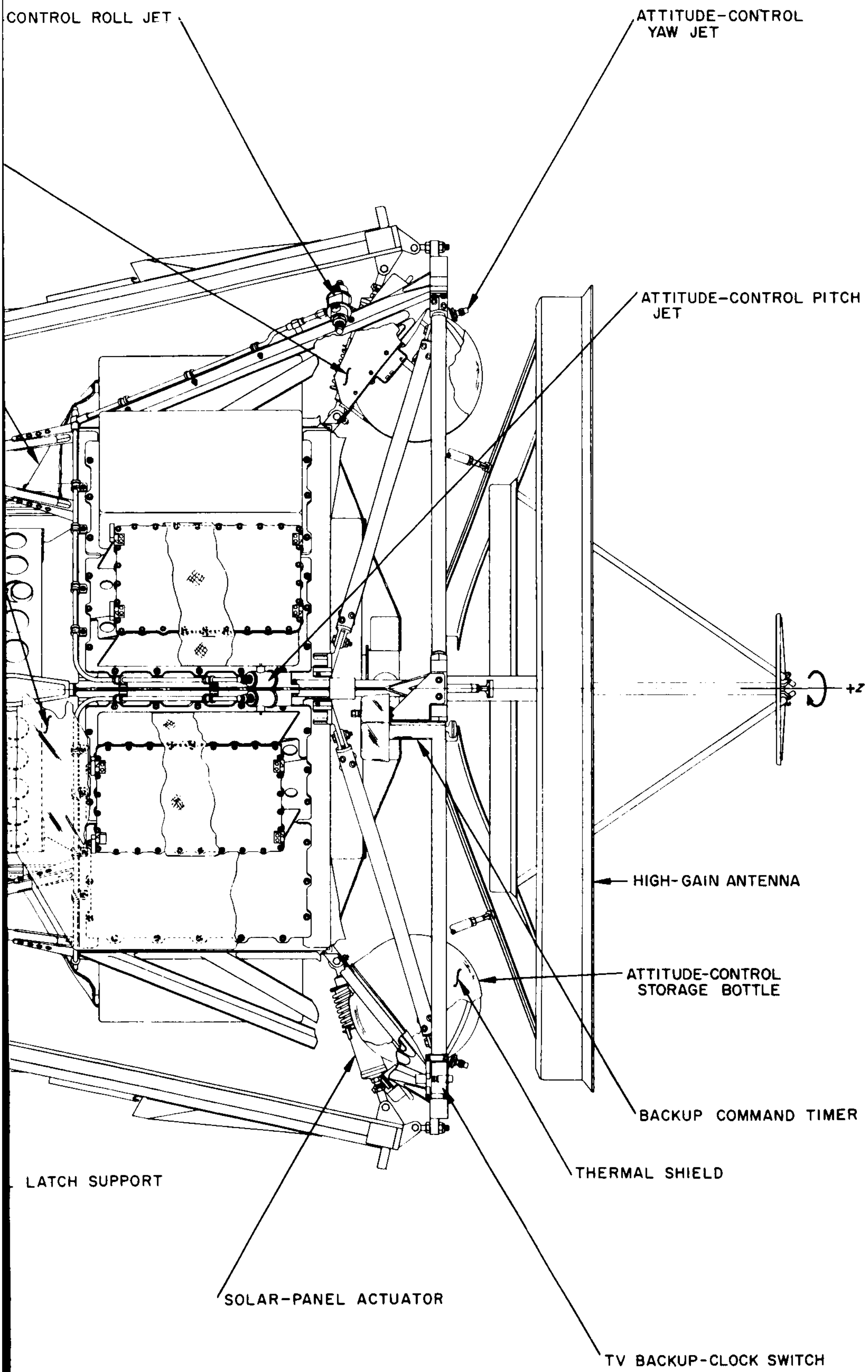


Fig. C-2c. Yaw axis


CONTROL ROLL JET
ATTITUDE-CONTROL YAW JET
ATTITUDE-CONTROL PITCH JET
+Z
HIGH-GAIN ANTENNA
ATTITUDE-CONTROL STORAGE BOTTLE
BACKUP COMMAND TIMER
THERMAL SHIELD
LATCH SUPPORT
SOLAR-PANEL ACTUATOR
TV BACKUP-CLOCK SWITCH

70-2

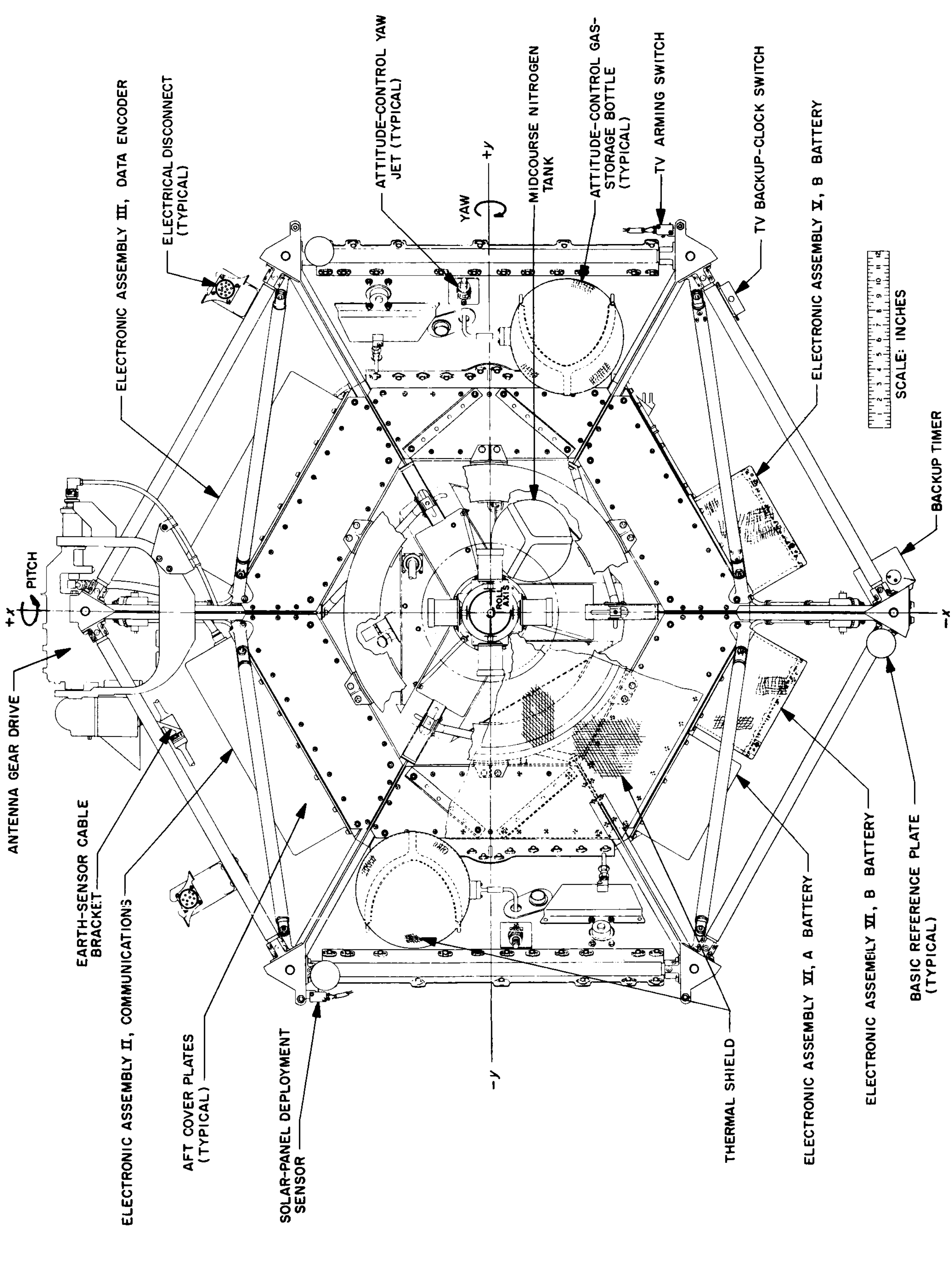


Fig. C-2d. Spacecraft configuration, *Ranger VI*, bottom view

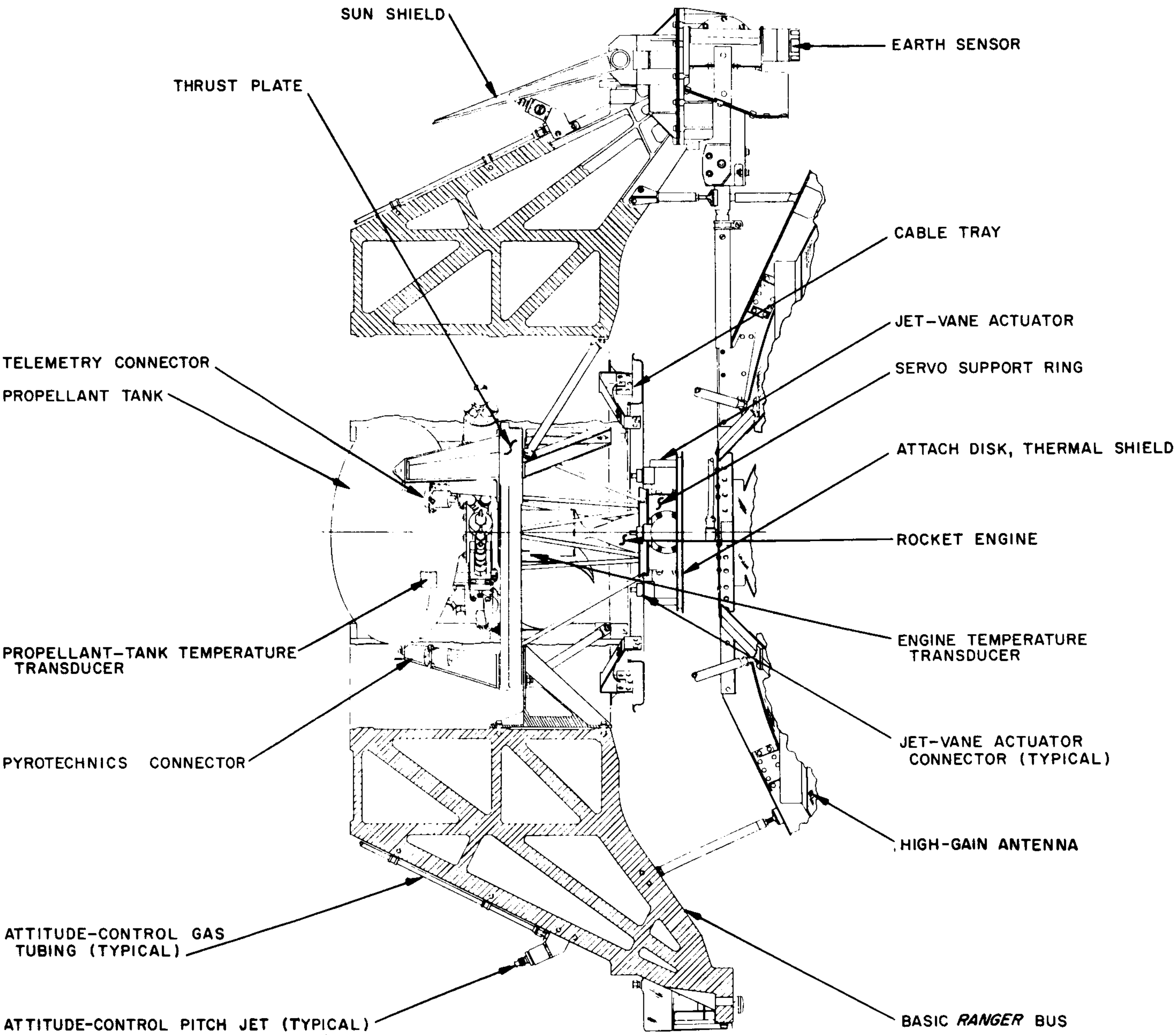


Fig. C-3. Spacecraft bus interior and spaceframe section

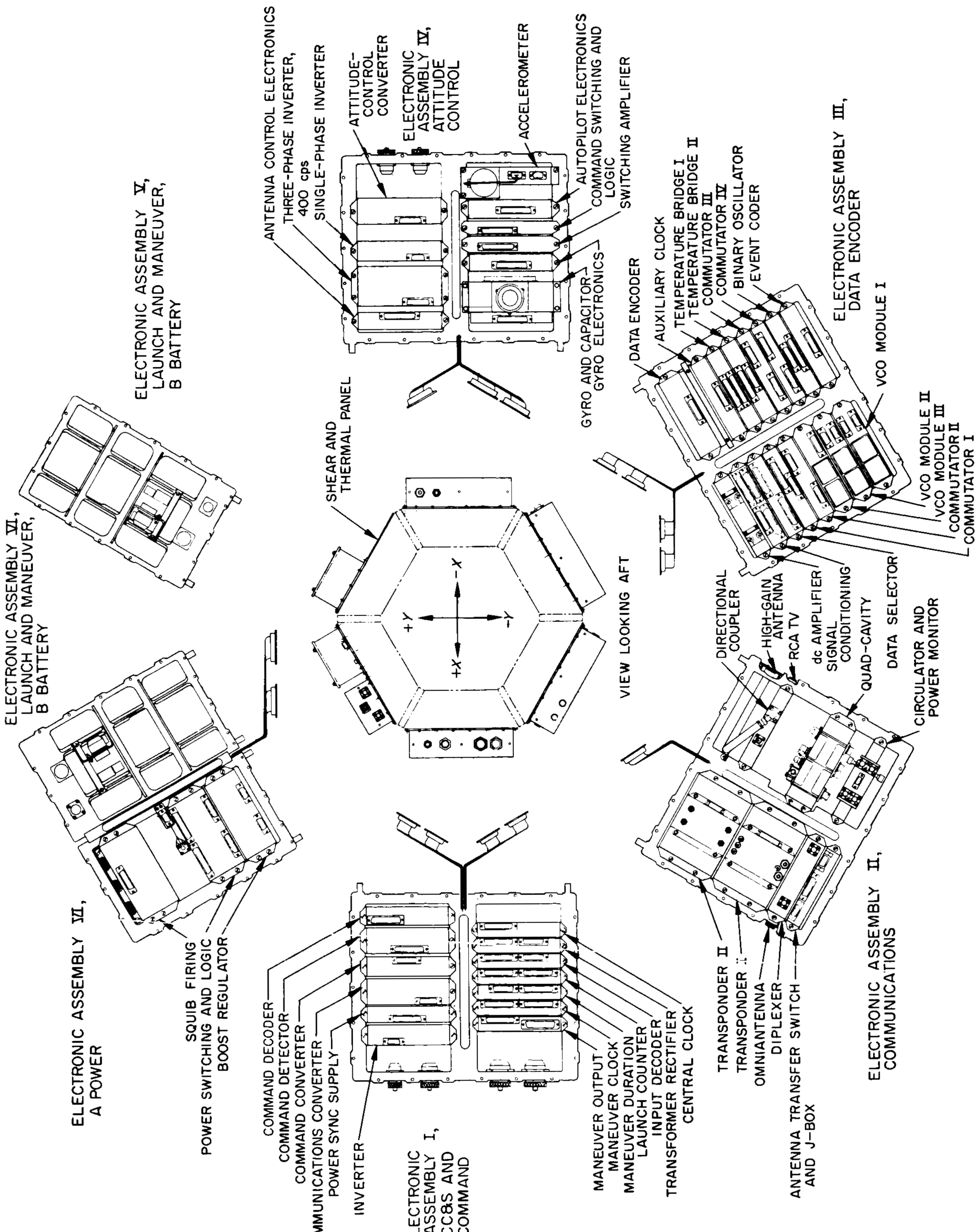


Fig. C-4. Electronics assembly locations

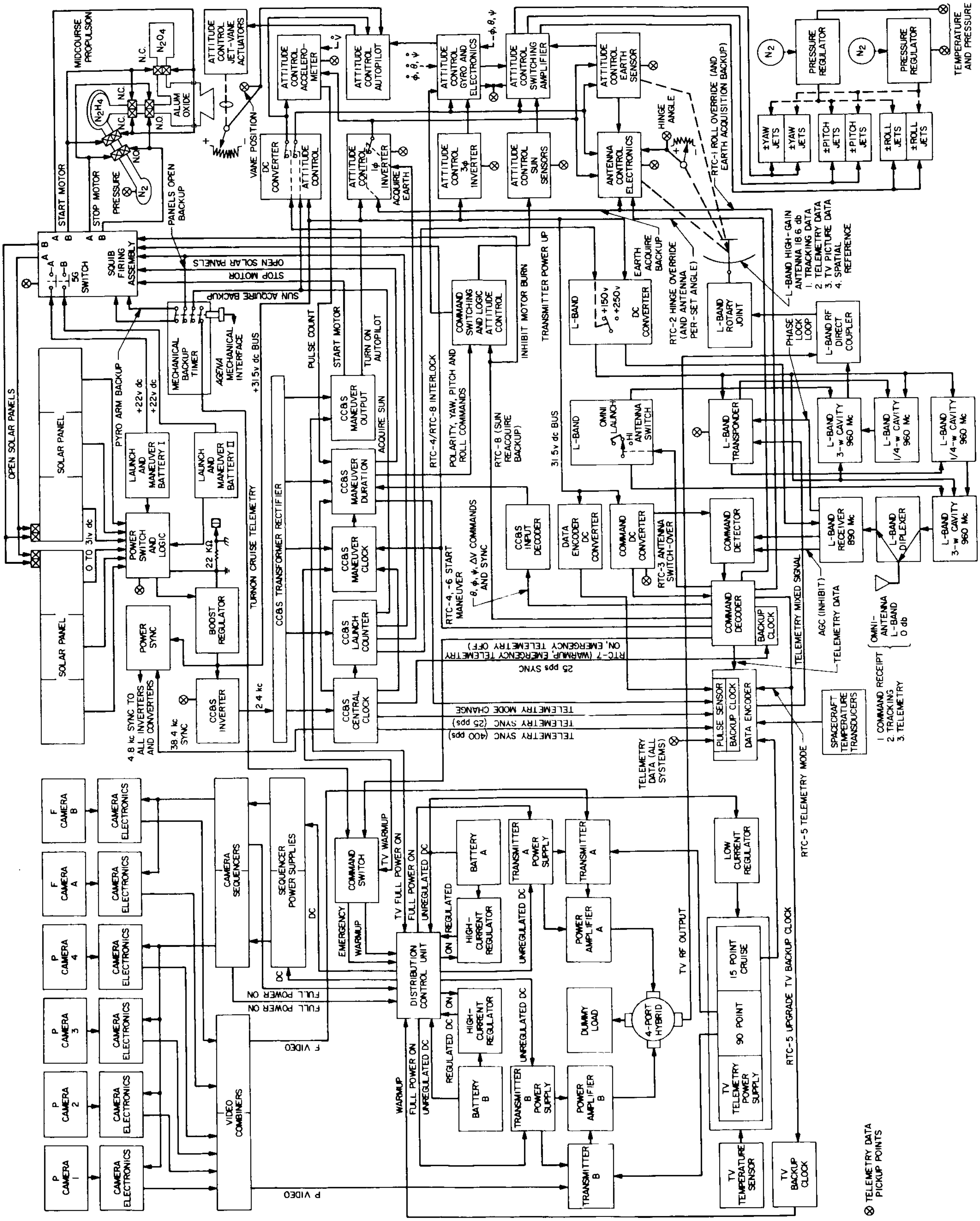


Fig. C-5. Spacecraft block diagram

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