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GEMINI PROGRAM MISSION REPORT

GEMINI VIII

(U)

APRIL 1966

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION MANNED SPACECRAFT CENTER

GEMINI SPACECRAFT FLIGHT HISTORY			
Mission	Description	Launch date	Major accomplishments
GT-1	Unmanned 64 orbits	Apr. 8, 1964	Demonstrated structural integrity.
GT-2	Unmanned suborbital	Jan. 19, 1965	Demonstrated heat protection and systems performance.
GT-3	Manned 3 orbits	Mar. 23, 1965	Demonstrated manned qualifications of the Gemini spacecraft.
Gemini IV	Manned 4 days	June 3, 1965	Demonstrated EVA and systems performance for 4 days in space.
Gemini V	Manned 8 days	Aug. 21, 1965	Demonstrated long-duration flight, rendezvous radar capability, and rendezvous maneuvers.
Gemini VI	Manned 2 days rendezvous (canceled after failure of GATV)	Oct. 25, 1965	Demonstrated dual countdown procedures (GAATV and GLV—spacecraft), flight performance of TLV and flight readiness of the GATV secondary propulsion system. Mission canceled after GATV failed to achieve orbit.
Gemini VII	Manned 14 days rendezvous	Dec. 4, 1965	Demonstrated 2-week duration flight and station keeping with GLV stage II, evaluated "shirt sleeve" environment, acted as the rendezvous target for spacecraft 6, and demonstrated a controlled reentry to within 7 miles of planned landing point.
Gemini VI-A	Manned 1 day	Dec. 15, 1965	Demonstrated on-time launch procedures, closed-loop rendezvous capability, and station keeping techniques with spacecraft 7.
Gemini VIII	Manned 3-day rendezvous and dock (terminated in rev. 7)	March 16, 1966	Rendezvous and docking with GATV, controlled landing, emergency recovery, multiple restart of GATV in orbit. Spacecraft mission terminated early because of an electrical short in the control system.

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GEMINI PROGRAM MISSION REPORT

GEMINI VIII

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MANNED SPACECRAFT CENTER

HOUSTON, TEXAS

APRIL 29, 1966

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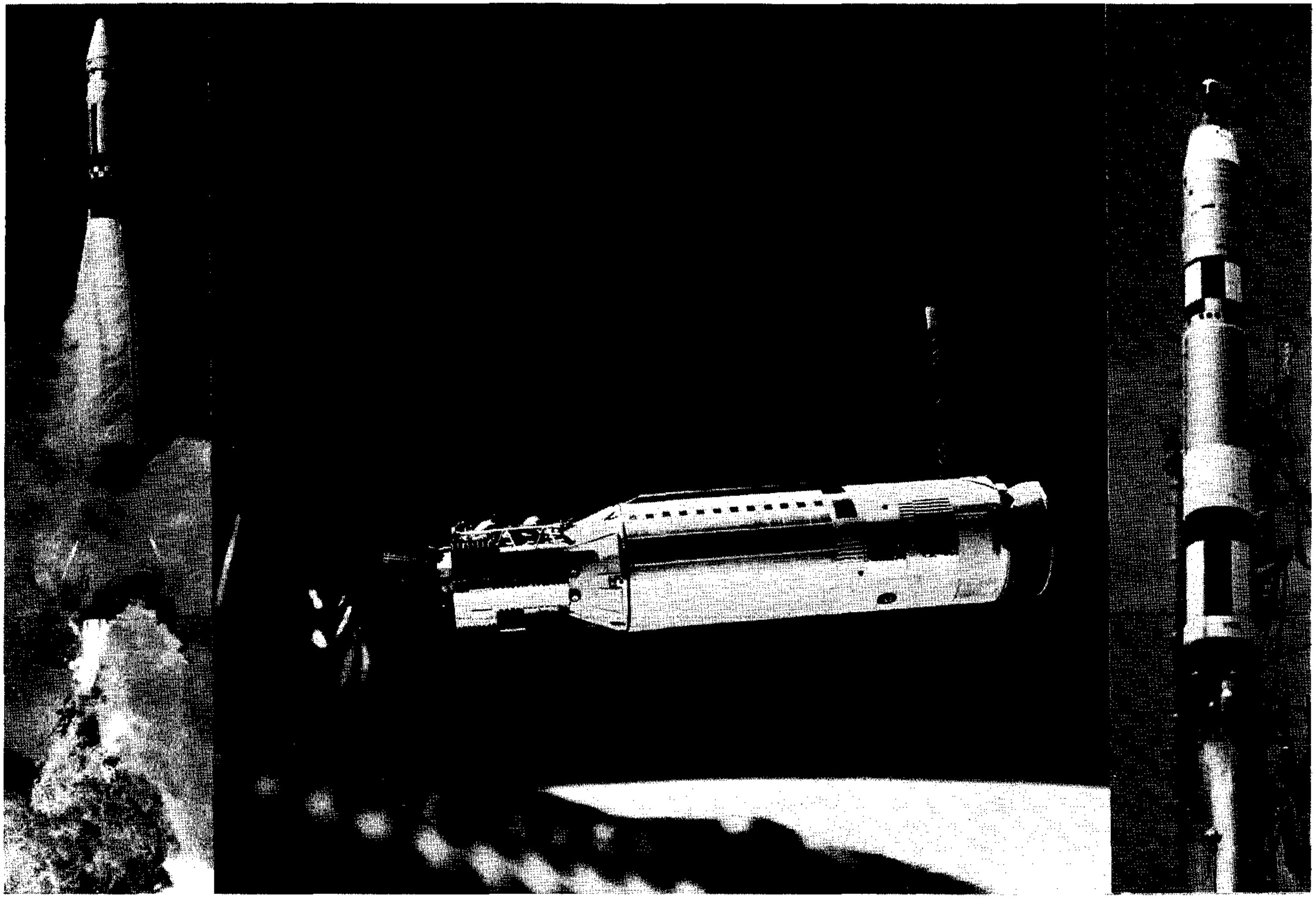

Gemini VIII Space Vehicles at lift-off and in orbit.

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1.0 MISSION SUMMARY

The sixth manned mission, designated Gemini VIII, was the second rendezvous mission and the first docking mission of the Gemini Program. The Gemini Atlas-Agena Target Vehicle was launched from Complex 14, Cape Kennedy, Florida, at 9:00:03 a.m. e.s.t. on March 16, 1966. The Gemini Space Vehicle, with Astronaut Neil A. Armstrong, command pilot, and Astronaut David R. Scott, pilot, was launched from Complex 19, Cape Kennedy, Florida, at 10:41:02 a.m. e.s.t. on March 16, 1966. The flight was scheduled as a three-day mission; however, because of a spacecraft control-system anomaly which necessitated activation of the Reentry Control System, the manned phase of the flight was concluded at approximately 13 hours 52 minutes ground elapsed time. During the anomaly period, the crew exhibited a calm attitude and deliberate manner in analyzing the problem and bringing the spacecraft back under control; they then performed a normal closed-loop reentry, controlling the spacecraft to a nominal landing. Recovery of the flight crew and the spacecraft was accomplished in the western Pacific Ocean at 25° 21' north latitude, 135° 56' east longitude as reported by the recovery ship, U.S.S. Leonard Mason. The crew demonstrated satisfactory control of the rendezvous and docking and completed the flight in good physical condition.

A primary objective of rendezvousing and docking with the Gemini Agena Target Vehicle was accomplished. The secondary objectives that were accomplished were rendezvousing and docking during the fourth revolution, evaluating the Auxiliary Tape Memory Unit, demonstrating a controlled reentry, and parking the Gemini Agena Target Vehicle. Two of the secondary objectives were partially accomplished in that some systems evaluation was conducted and two of the ten experiments were performed. Early termination of the mission precluded accomplishment of the remaining objectives of the mission.

The performance of the Gemini Atlas-Agena Target Vehicle was satisfactory for this mission. The countdown was completed with no holds and, after a nominal lift-off and launch phase, the Gemini Agena Target Vehicle was inserted into the planned coast-ellipse trajectory. The Primary Propulsion System of the Gemini Agena Target Vehicle ignited as planned and inserted the vehicle into a 161.3-nautical-mile circular orbit (referenced to a spherical earth having a radius equal to that of the launch complex). These orbital elements were within one mile of the planned orbital elements.

One hour 40 minutes 59 seconds after the successful launch of the Gemini Atlas-Agena Target Vehicle, the Gemini Spacecraft also was launched successfully. The performance of the Gemini Launch Vehicle was satisfactory in all respects. The countdown was entirely nominal with no unscheduled holds, and the lift-off occurred within one-half second of the scheduled time. First-stage flight was normal, with all planned events occurring within required limits. The first-stage offset yaw-steering technique was used to place the spacecraft into an orbital plane very close to the plane of the target-vehicle orbit.

Staging was nominal, and the second-stage flight was normal. The spacecraft was inserted into an orbit having a 86.3-nautical-mile perigee and a 146.7-nautical-mile apogee referenced to a Fischer ellipsoid earth. The perigee was 0.3 nautical mile below that planned and the apogee was 1.2 nautical miles above that planned. At spacecraft insertion, the slant range to the Gemini Agena Target Vehicle was a nominal 1060 nautical miles.

During the following period of 5 hours 52 minutes, nine maneuvers were performed by the crew to effect the rendezvous with the Gemini Agena Target Vehicle. These maneuvers were all performed using the spacecraft guidance system for attitude reference, and the entire terminal phase of rendezvous was completed using onboard-computer solutions and displays. Continuous radar lock-on was achieved at a range of 180 nautical miles and no subsequent losses of lock occurred until the radar was placed in standby at a distance of approximately 20 feet from the Gemini Agena Target Vehicle. The rendezvous phase of the mission was completed at 5 hours 58 minutes ground elapsed time when Spacecraft 8 was 150 feet from the Gemini Agena Target Vehicle and all relative motion between the two vehicles had been stopped.

After station keeping for about 36 minutes, docking with the Gemini Agena Target Vehicle was accomplished. The final docking maneuver was begun when a distance of about 2 feet separated the two vehicles. A relative velocity of about three-fourths of a foot per second was achieved at the moment of contact. The nose of the spacecraft moved into the docking adapter very smoothly and the docking and rigidizing sequence took place very quickly and with no difficulty. The docking sequence was completed at 6:33:22 ground elapsed time, with the two vehicles rigidized together.

For a period of 27 minutes after docking, the stability and control of the docked vehicles was excellent. At approximately 7:00:30 ground elapsed time, the crew noted that the spacecraft-Gemini Agena Target Vehicle combination was developing unexpected roll and yaw rates. The command pilot was able to reduce these rates to essentially zero; however, after he released the hand controller, the rates began to increase again and the crew found it difficult to effectively control the rates without excessive use of spacecraft Orbital Attitude and Maneuver System propellants. In an effort to isolate the problem and stop the excessive fuel consumption, the crew initiated the sequence to undock the spacecraft from the Gemini Agena Target Vehicle. After undocking, the spacecraft rates in roll and yaw began to increase, indicating a spacecraft problem which the crew attempted to isolate by initiating malfunction-analysis procedures. When the rates reached approximately 300 degrees per second, the crew completely deactivated the Orbital Attitude and Maneuver System and activated both rings of the Reentry Control System in the direct-direct mode. After ascertaining that spacecraft rates could be reduced using the Reentry Control System, one ring of the system was turned off to save fuel for reentry and the spacecraft rates were reduced to zero using the other ring. The crew continued the malfunction analysis and isolated the problem area to the no. 8 thruster (yaw left-roll left) in the Orbital Attitude and Maneuver System. The circuitry to this thruster had failed to an "on" condition.

The performance of the spacecraft was very satisfactory, except for the yaw-left thruster malfunction. Because this malfunction resulted in a necessity to activate the Reentry Control System, a decision was reached to terminate the flight during the seventh revolution and land in secondary recovery area no. 3 in the western Pacific Ocean.

The retrofire sequence was initiated exactly on time at 10:04:47 ground elapsed time. Spacecraft reentry and landing were nominal and the landing point achieved was less than 7 nautical miles from that planned. The crew of one of the search airplanes sighted the spacecraft descending on the main parachute. Recovery was accomplished very efficiently and the crew and spacecraft were onboard the recovery ship, U.S.S. Leonard Mason, approximately 3 hours 11 minutes after landing.

After the end of the manned phase of the mission, a flight plan was developed to exercise the Gemini Agena Target Vehicle. Eleven maneuvers using the two propulsion systems were conducted during the remainder of the mission (includes nine Secondary Propulsion System firings associated with the nine Primary Propulsion System firings). The Gemini Agena Target Vehicle and its systems operated satisfactorily during the entire mission except for the flight control system, which exhibited a yaw error accompanied by a slight pitch error during all Primary Propulsion System maneuvers. The yaw error was caused by an offset center-of-gravity of the Gemini Agena Target Vehicle in combination with the long time constant of the control system in response to attitude errors. This slow response was due to modifications to the standard Agena D control system which were necessary to provide dynamic stability of the docked combination during maneuvers with the Primary Propulsion System.

Flight control personnel were able to compensate in the final maneuvers for the yaw error and placed the Gemini Agena Target Vehicle in an orbit having a 222-nautical-mile apogee and a 220-nautical-mile perigee, or within 2 miles of the desired circular orbit.

The target-vehicle orbital altitude will gradually decrease and this vehicle can be used for an alternate rendezvous as a passive target during later missions.

2.0 INTRODUCTION

A description of the Gemini VIII mission, as well as a discussion of the evaluation of the mission results, is contained in this report. The evaluation covers the time from the start of the simultaneous countdown of the Gemini Atlas-Agena Target Vehicle and the Gemini Space Vehicle to the date of publication of the report.

Detailed discussions are found in the major sections related to each principal area of effort. Some redundancy may be found in various sections where it is required for a logical presentation of the subject matter.

Data were reduced only in areas of importance from telemetry, onboard records, and ground-based radar tracking. In evaluating the performance of the Atlas Standard Launch Vehicle and Gemini Launch Vehicle, all available data were processed. The evaluation of all vehicles involved in the mission consisted of analyzing the flight results and comparing them with the results from ground tests and from previous missions.

Section 6.1, FLIGHT CONTROL, is based on observations and evaluations made in real time, and, therefore, may not coincide with the results obtained from the detailed postflight analysis.

Brief descriptions of the ten experiments flown on this mission are presented in section 8.0, and preliminary results and conclusions on the two experiments performed are included.

The mission objectives, as set forth in the Mission Directive, formed the basis for evaluation of the flight and were of paramount consideration during preparation of this report. The primary objectives of the Gemini VIII mission were as follows:

(a) Perform rendezvous and docking with the Gemini Agena Target Vehicle.

(b) Conduct extravehicular activities.

The secondary objectives of the Gemini VIII mission were as follows:

(a) Perform rendezvous and docking with the Gemini Agena Target Vehicle during the fourth revolution.

(b) Perform docked-vehicle maneuvers using the Gemini Agena Target Vehicle Secondary Propulsion System.

(c) Conduct systems evaluation.

(d) Conduct ten experiments.

(e) Conduct docking practice.

(f) Perform a re-rendezvous.

(g) Evaluate the Auxiliary Tape Memory Unit.

(h) Demonstrate a controlled reentry.

(i) Park the Gemini Agena Target Vehicle in a 220-nautical-mile circular orbit.

At the time of publication of this report, more detailed analyses of data on the performance of the launch vehicles, Gemini Agena Target Vehicle, and the Radio Guidance System were continuing. Analyses of the spacecraft and the Inertial Guidance System were also continuing. Supplemental reports, listed in section 12.4, will be issued to provide documented results of these analyses.

The results of previous Gemini missions are reported in references 1 through 8.

3.0 VEHICLE DESCRIPTION

The space vehicle for the Gemini VIII mission consisted of Spacecraft 8 and Gemini Launch Vehicle (GLV) 8. The Gemini Atlas-Agena Target Vehicle (GAATV) consisted of Gemini Agena Target Vehicle (GATV) 5003 and Target Launch Vehicle (TLV) 5302.

The general arrangement and major reference coordinates of the Gemini Space Vehicle are shown in figure 3.0-1. Section 3.1 of this report describes the spacecraft configuration, section 3.2 describes the GLV configuration, and section 3.3 provides the space-vehicle weight and balance data.

The general arrangement and major reference coordinates of the GAATV are shown in figure 3.0-2. Section 3.4 describes the GATV configuration, including the Target Docking Adapter (TDA), section 3.5 describes the TLV configuration, and section 3.6 provides the weight and balance data of the GAATV.

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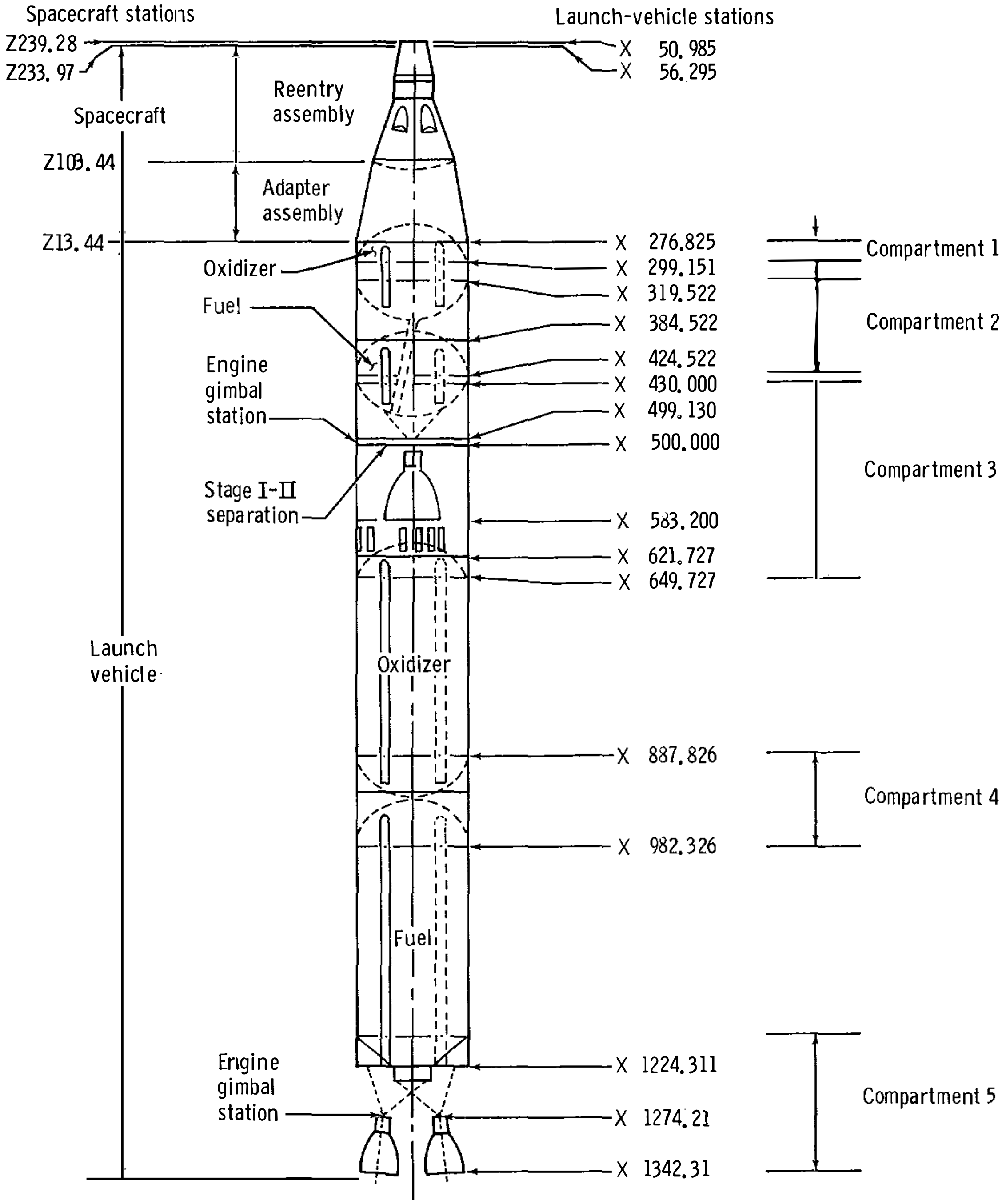

(a) Launch configuration.

Figure 3.0-1. - GLV - spacecraft relationships.

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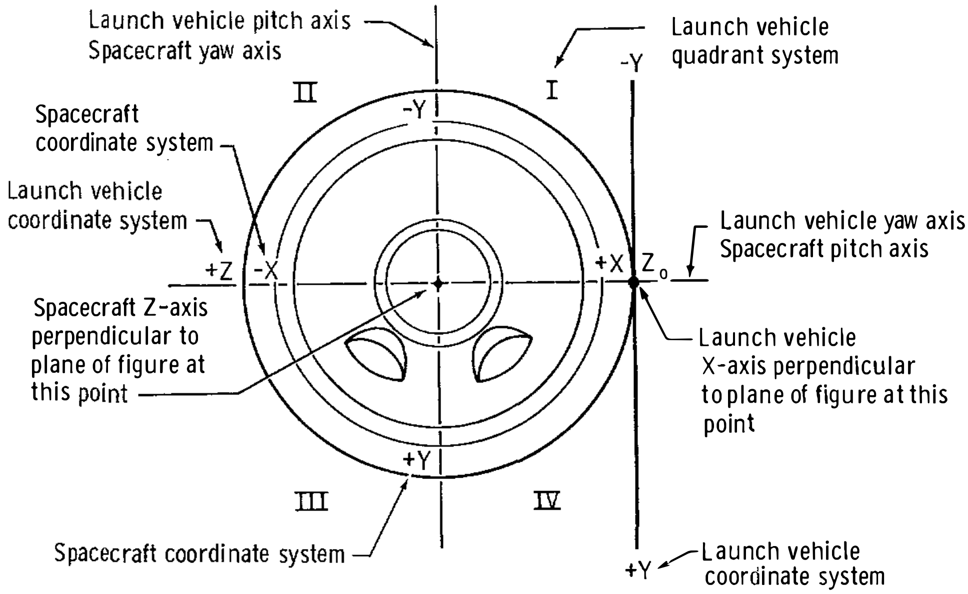
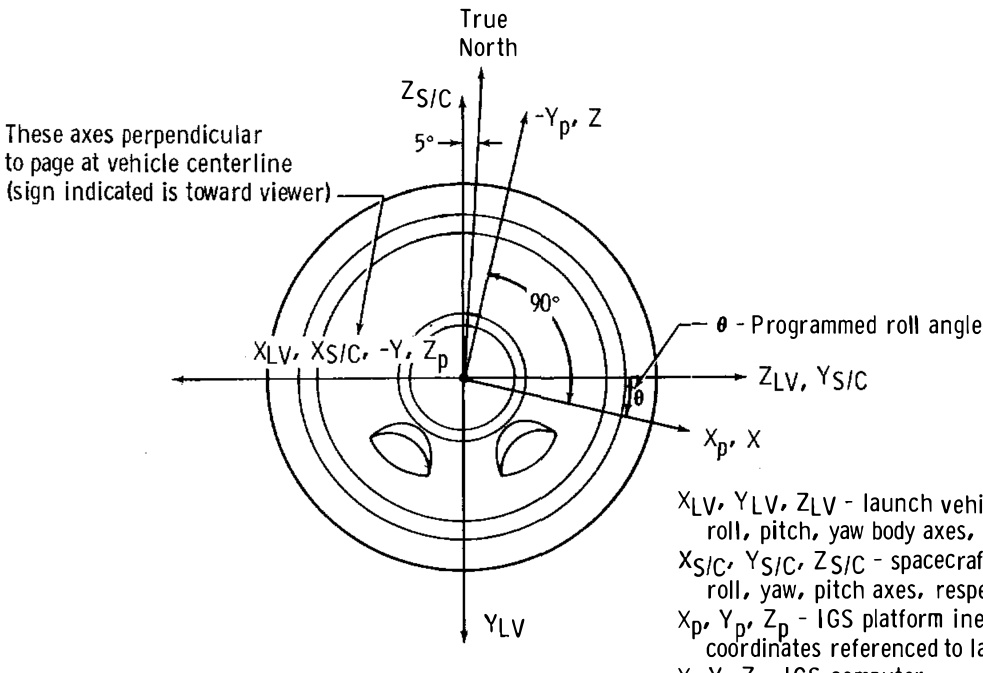


Dimensional axes



Guidance coordinates

X_{LV}, Y_{LV}, Z_{LV} - launch vehicle roll, pitch, yaw body axes, respectively
$X_{S/C}$, $Y_{S/C}$, $Z_{S/C}$ - spacecraft roll, yaw, pitch axes, respectively
X_p, Y_p, Z_p - IGS platform inertial coordinates referenced to launch stand
X, Y, Z - IGS computer computational coordinates

(b) Dimensional axes and guidance coordinates.
Figure 3.0-1. - Concluded.

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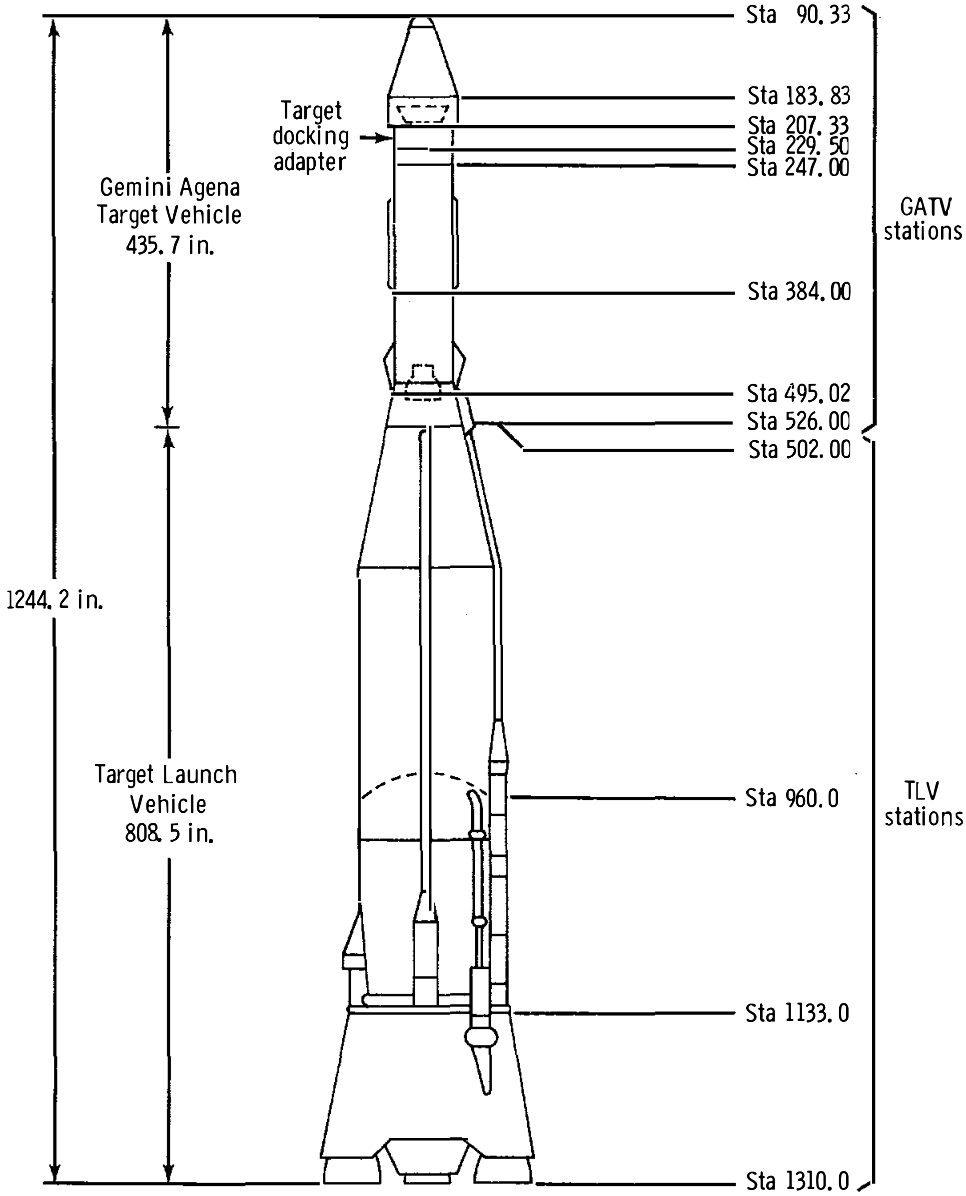

(a) Launch configuration.

Figure 3.0-2. - TLV - GATV relationships.

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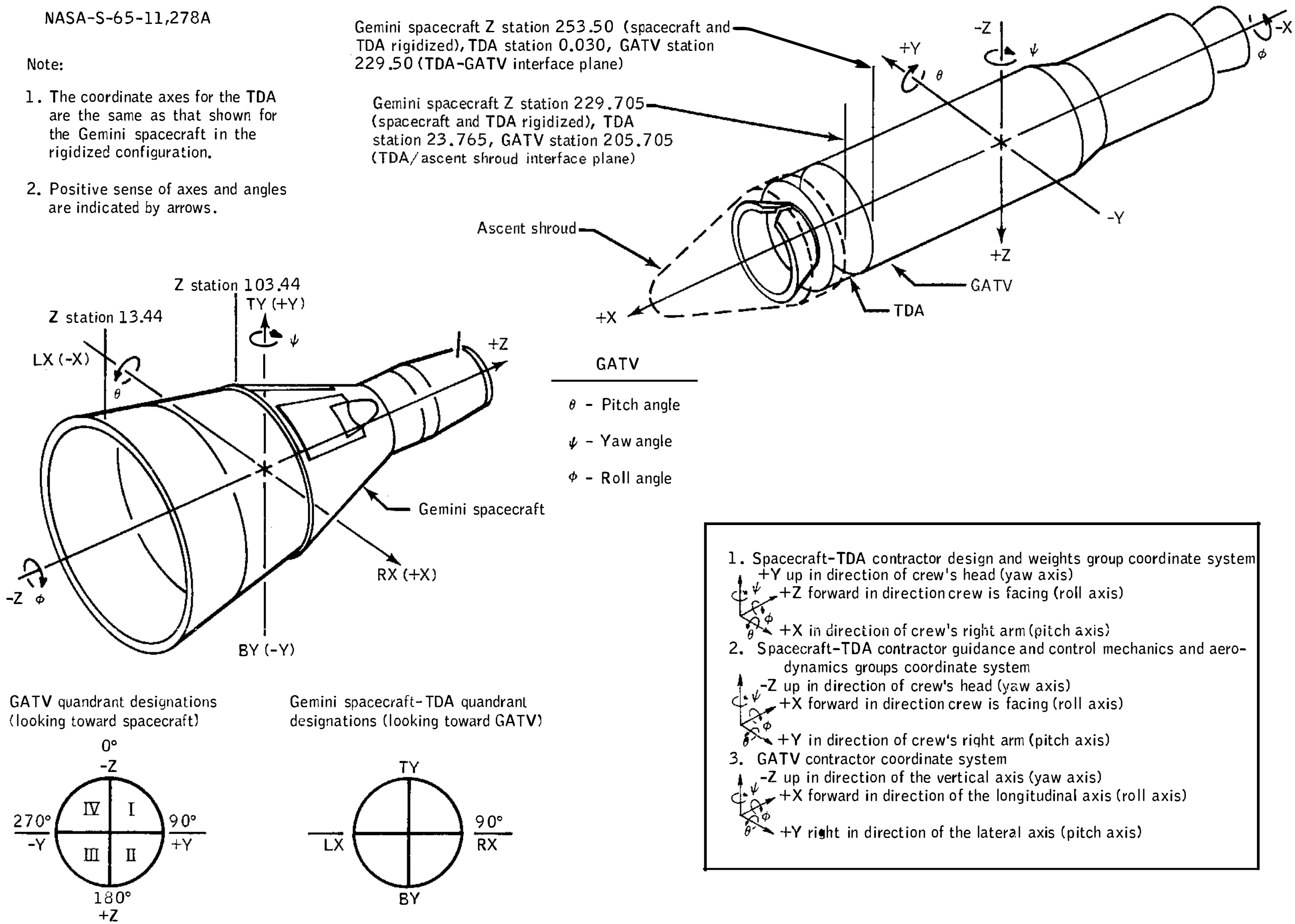

(b) Dimensional axes and guidance coordinates, GATV-TDA.

Figure 3.0-2. - Continued.

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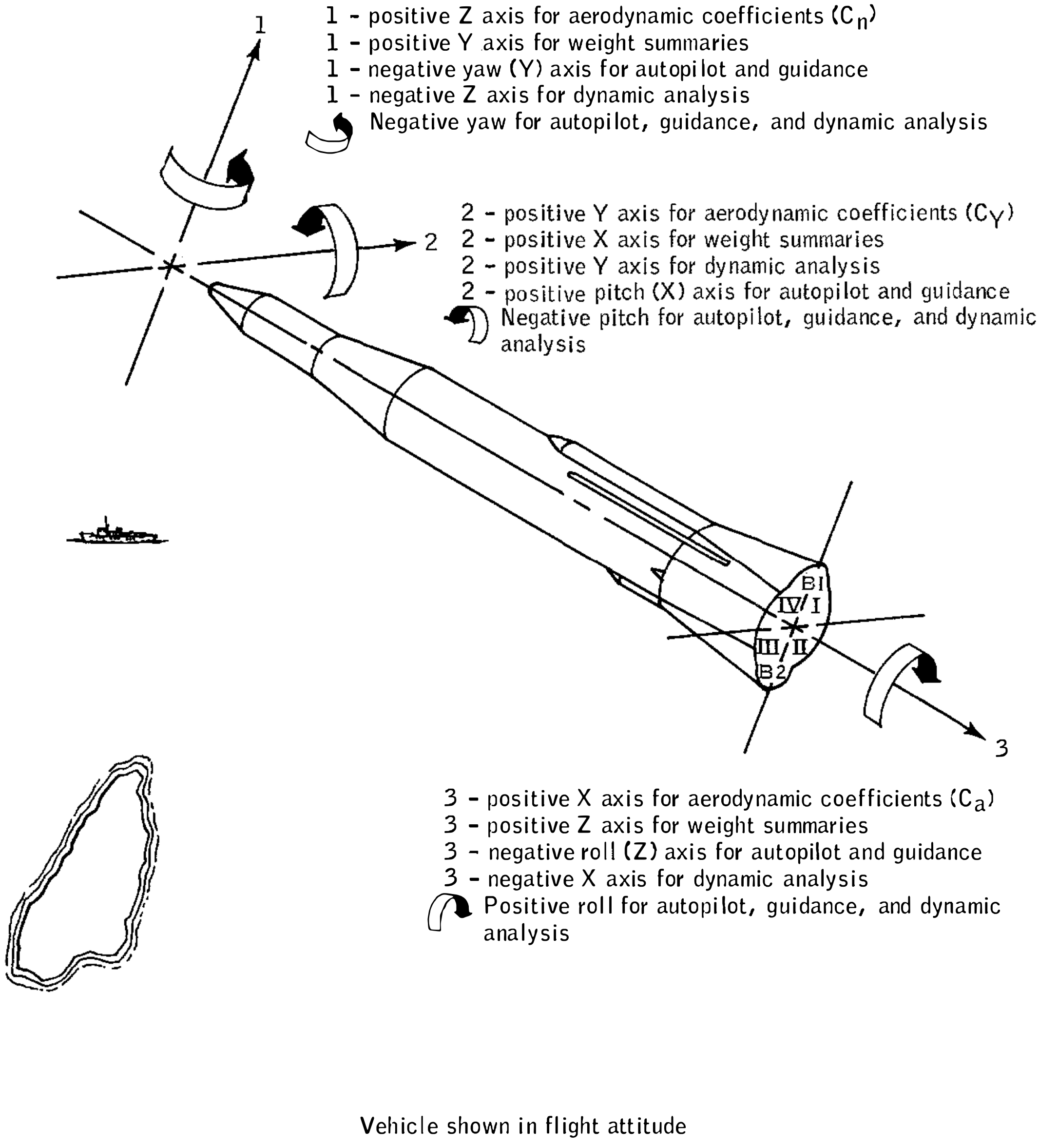

(c) Dimensional axes and guidances coordinates, TLV.

Figure 3.0-2. - Concluded.

3.1 GEMINI SPACECRAFT

The structure and major systems of Spacecraft 8 (fig. 3.1-1) were of the same general configuration as the previous Gemini spacecraft. Reference 2 provides a detailed description of the basic spacecraft (Spacecraft 2) and references 3 through 8 describe the modifications incorporated into the subsequent spacecraft. Except for the Fuel-Cell Power System and the extravehicular equipment, Spacecraft 8 most closely resembled Spacecraft 6 (ref. 7), and only the significant differences (table 3.1-I) between those two spacecraft are included in this report. Equipment associated with the Fuel Cell Power System will be compared to the Spacecraft 7 system (ref. 8), and the extravehicular equipment will be compared to Spacecraft 4 equipment (ref. 4). A detailed description of Spacecraft 8 is contained in reference 9.

3.1.1 Spacecraft Structure

The primary load-bearing structure of Spacecraft 8 was essentially the same as that of Spacecraft 6. However, some changes were incorporated to facilitate the planned extravehicular activity (EVA) (see section 3.1.2.12).

3.1.2 Major Systems

3.1.2.1 Communications System.- The following changes were required to the Communications System because of the planned EVA. At lift-off, the voice tape recorder was mounted as normal, adjacent to the pilot's right elbow; however, it was planned that during preparation for the EVA, the recorder would be relocated by the flight crew so that it was accessible to the command pilot for changing the voice tape cartridges. The recorder would have been secured with Velcro tape to the Velcro on the cabin wall. The recorder circuits were modified to permit received, as well as transmitted, voice communications to be recorded. A UHF voice transceiver was included in the Extravehicular Support Package (section 3.1.2.12) for communication between the extravehicular pilot and the command pilot. This transceiver was of the same configuration as the one to be used in the Astronaut Maneuvering Unit (AMU) during later EVA missions.

3.1.2.2 Instrumentation and Recording System.- The Instrumentation and Recording System was basically the same as the one used on Spacecraft 6. However, four additional accelerometers were installed to provide data for determining the stability of the docked Spacecraft-GATV combination during the GATV Secondary Propulsion System (SPS) firing.

3.1.2.3 Environmental Control System.- The following changes were incorporated into the Environmental Control System (ECS).

3.1.2.3.1 Cabin heat exchanger: The cabin heat exchanger and its associated fan and components were not installed.

3.1.2.3.3 Egress oxygen system: The complete egress oxygen system was deleted.

3.1.2.3.3 Cabin repressurization control: A locking device was added to the cabin-repressurization control-valve handle to prevent inadvertent opening.

3.1.2.3.4 Cabin vent-valve redundant seal stopper: A manually operated redundant seal stopper installed over the inlet of the cabin vent valve was similar to that used on Spacecraft 7, except that it could be reseated. This seal stopper provided a backup seal for the cabin pressurization in case of an inadvertent opening of the cabin vent valve.

3.1.2.3.5 Water storage tanks: Because of the use of the Fuel Cell Power System, the water storage tanks were similar in function to those installed in Spacecraft 7. However, the 3-day mission required only two tanks (fig. 3.1-2), each having a capacity of 42 pounds of water. Each tank was constructed of two aluminum half-spherical shells separated by a titanium ring. Two diaphragms were installed in each tank, one at each mating surface of the titanium ring with the aluminum shell. In tank A, 19 psia of gaseous nitrogen, and in tank B, 36 pounds of drinking water, were stored in the aluminum shells prior to launch. The purpose of the titanium ring was to preclude the destructive reaction between the aluminum shells and the acidic water from the fuel-cell sections.

In flight, the fuel-cell product water was transferred into both storage tanks, between the diaphragms, causing the diaphragms to expand and pressurize the drinking-water system. As the quantity of fuel-cell product water increased in proportion to the amount of water consumed by the flight crew, a dual pressure regulator permitted the gas in tank A to vent overboard. Thus, the water system remained pressurized at approximately 20 psia.

3.1.2.3.6 Crossfeed valve: A crossfeed valve was installed to interconnect the ECS breathing-oxygen system and the fuel-cell Reactant-Supply-System (RSS) oxygen. This arrangement was similar to that used for Spacecraft 7.

3.1.2.3.7 Coolant pumps: Two coolant pumps, an A-pump and a B-pump, were installed in each coolant loop. This arrangement was similar to the Spacecraft 7 system; however, Spacecraft 6 had only a single A-pump in each coolant loop.

3.1.2.4 Guidance and Control System.- The following changes were incorporated into the Guidance and Control System.

3.1.2.4.1 Auxiliary Tape Memory Unit: The Auxiliary Tape Memory Unit (ATMU) (fig. 3.1-3) was installed in the spacecraft adapter assembly. The ATMU increases the program-storage capability of the onboard digital computer by providing a means of reloading portions of the computer memory with various operational modes such as ascent, catchup, rendezvous, touchdown predict, and reentry. A mode selector switch (see section 3.1.2.9) enables the flight crew to select the desired ATMU operational mode. The modes available are as follows:

(a) Standby - Power is applied to the ATMU which remains in a non-operating status.

(b) Automatic - The flight crew can insert instructions in the Manual Data Insertion Unit (MDIU) and the computer will automatically command the ATMU to wind, rewind, program, or verify portions of the computer memory. The Incremental Velocity Indicator (IVI) displays the tape position and program on the X-channel and Y-channel, respectively.

(c) Wind - The ATMU will wind the tape and stop automatically at the end of the tape.

(d) Rewind - The ATMU will rewind the tape and stop automatically at the beginning of the tape.

(e) Program - Programs are read from magnetic tape and stored in the computer memory. The tape position and the program number being transferred are displayed by the IVI.

3.1.2.4.2 Operational program: The computer operational program deleted the ascent-abort reentry mode and added the touchdown-predict mode. The touchdown-predict mode could calculate the trajectory data and predict the touchdown point for a landing at any time between lift-off and planned end-of-mission. At launch, the spacecraft computer memory contained only the portions of the operational program that were applicable between lift-off and the end of the rendezvous phase. After the rendezvous phase, the ascent, catchup, and rendezvous modes were erased from the computer memory by the ATMU and replaced by the reentry and touchdown modes. The ATMU could load, verify, or reload any of these five modes (see section 3.1.2.4.1).

3.1.2.5 Time Reference System.- Except for the interface with the ATMU (see section 3.1.2.4), the Time Reference System was the same as the one used on Spacecraft 6. The time of equipment reset $\left(T_x\right)$ address command was used to provide the ATMU with a verify or a reprogram command, and when the computer-write mode was used, computer-clock and computer-write data signals were used to transfer data to the ATMU.

3.1.2.6 Electrical System.- The Electrical System (fig. 3.1-4) included a Fuel Cell Power System that was the same as the Spacecraft 7 system, except that the hydrogen regenerative cooling line and the insulation on the hydrogen supply tank were not incorporated. In addition to the pressure differential data provided by the switches and warning lights on the crew-station instrument panel, an analog readout of these pressures was also provided to the flight crew and, by telemetry, to the ground stations.

3.1.2.7 Propulsion System.- The Orbital Attitude and Maneuver System (OAMS) is shown in figure 3.1-5. The Reentry Control System (RCS) is shown in figure 3.1-6. The following changes were incorporated into the Propulsion System.

3.1.2.7.1 Oxidizer valve heaters: In the OAMS, each of the 16 oxidizer solenoid valves was provided with a thermostatically controlled redundant 1.25-watt heater.

3.1.2.7.2 OAMS reserve fuel tank: A reserve fuel tank was added to the OAMS to provide a contingency quantity of fuel because of potential gaging system inaccuracies in the primary fuel system. The reserve tank was of the same configuration as the RCS fuel tank and was mounted on the adapter-assembly internal structure. An F-package was also provided to isolate pressure from the reserve tank until after depletion of the fuel supply in the primary tank. The operation of the reserve tank and F-package was the same as for Spacecraft 7 (ref. 8).

3.1.2.8 Pyrotechnic System.- Except for the pyrotechnic devices associated with the EVA equipment and with experiments, the Pyrotechnic System was similar to the one used on Spacecraft 6. The pyrotechnic devices required for the planned EVA included three guillotines for severing the cable which retained the handholds and foot supports in the adapter section and for severing the attachment bolt that secured the Extravehicular Support Package (ESP) (see section 3.1.2.12). Also, four cable-cutter guillotines were installed for releasing equipment planned for use with experiments D-14, D-15, D-16, and S-9 (see section 8.0).

3.1.2.9 Crew-station furnishings and equipment.- The following changes were incorporated into the crew-station furnishings and equipment.

3.1.2.9.1 Controls and displays: In addition to the following changes, the crew-station controls and displays (fig. 3.1-7) also included minor changes in the nomenclature of indicators and switch positions.

(a) A panel was installed to monitor and control the ATMU and contained an ON-RESET-OFF switch, a mode selector switch, a running light, and an error light.

(b) In addition to switches for controlling the GATV, the Agena control panel also contained switches and circuit breakers for supplying power for the EVA lights and pyrotechnics and for the planned experiments S-9, D-14, and D-15. (See section 8.0.)

(c) The fuel-cell power monitor was similar to the one used for Spacecraft 7. The two fuel-cell differential-pressure warning lights incorporated into the annunciator panel monitored and warned of excessive differential pressures between the two fuel-cell reactants and between the reactants and the product water. The main-bus ammeter installed on Spacecraft 6 was removed from Spacecraft 7 and 8 to provide space for the fuel-cell monitor and control panel. Two of the six ammeters previously used to monitor the fuel-cell stack currents were changed to monitor the two main-bus currents. The ac voltmeter monitored the 26 V-ac, 400-cps system.

(d) A switch was provided for the OAMS reserve fuel tank.

(e) Two control switches were installed for starting and stopping the TDA rigidizing sequence and for initiating the docking and unrigidizing sequences (see section 3.4.12). These switches were for use by the flight crew if the automatic sequencing circuits had failed.

(f) A light was added to the digital clock to provide increased lighting for the elapsed-time display. An ON-OFF switch and dimming control was installed adjacent to the clock.

(g) Displays and controls were installed for experiments S-9, D-14, and D-15 (see section 8.0).

3.1.2.9.2 Miscellaneous equipment changes: The ejection-seat system was modified to reduce the height of the egress kit, and this change, combined with the removal of the egress oxygen system (section 3.1.2.3), required minor changes in the method of egress-kit ejection.

3.1.2.9.3 Stowage facilities: The stowage containers are shown in figure 3.1-8. Table 3.1-II lists the major items of equipment, including cameras, stowed in the containers at launch.

3.1.2.10 Landing System.- There were no significant changes to the Landing System.

3.1.2.11 Postlanding and Recovery Systems.- There were no significant changes in the Postlanding and Recovery Systems.

3.1.2.12 Extravehicular activity equipment.- The following modifications were incorporated in the spacecraft and the G4C space suits to permit EVA. In addition, the Extravehicular Life Support System (ELSS) and the ESP were provided to equip the pilot for the planned extravehicular operation.

3.1.2.12.1 Spacecraft modification for extravehicular activity: An external handrail assembly (fig. 3.1-9) was added to the exterior surface of the spacecraft adapter assembly behind the right hatch. The handrail, composed of two units, was stowed flush on the surface of the adapter during launch. The aft handrail was automatically extended to the EVA position after the spacecraft was separated from the launch vehicle. The forward handrail was to be extended by pilot actuation of a latching device. To augment the handrail, Velcro hook patches (fig. 3.1-9) were also added to serve as handholds on the external surface of the spacecraft. The patches were spaced at 1-foot intervals in the following locations:

(a) From the right hatch to the vicinity of the docking bar

(b) Circumferentially around the spacecraft at the forward and aft ends of the adapter assembly

(c) From the left hatch to the aft end of the adapter assembly and in a line parallel to the EVA handrail

Handholds and foot supports (fig. 3.1-10) were added inside the spacecraft adapter assembly to enable the pilot to don the ESP during the planned EVA. Because of load considerations and GLV dome clearance at launch-vehicle separation, a cable retention system was incorporated to retain the handholds and foot supports. The adapter-equipment-section thermal curtain was redesigned to accommodate the ESP, the handholds, and the foot supports. Floodlighting was provided in the adapter equipment section and a light was added to the forward end of the adapter assembly and was pointed aft to illuminate the adapter surface and handrail for night-side EVA. A mount was provided on the adapter assembly just behind the right hatch to support a 16-mm movie camera which was to provide external photographic coverage of the EVA. A ring (fig. 3.1-9) was installed on the forward surface of the Rendezvous and Recovery Section to provide an attaching point for the EVA tether when the spacecraft was not docked.

A hatch holding device was added to both hatches. This was a tooth and ratchet system with the tooth mounted on the center torque box of the cabin and the ratchet attached to the hatch. To provide EVA capability through either hatch, a hatch closing device and attaching eyebolts were added to the left hatch and were the same as the existing installation on the right hatch. Hatch rigging procedures were changed to insure compatibility with the hatch holding device.

3.1.2.12.2 Space suits: The G4C Gemini space suits were basically the same as the extravehicular space suit used in the Gemini IV flight. Two configurations of the basic suit were used. The intravehicular suit worn by the command pilot utilized the basic G4C pressure-garment assembly with a single-layer, lightweight cover layer. The extravehicular suit worn by the pilot utilized the basic G4C pressure-garment and helmet assemblies with the following modifications:

(a) A revised material lay-up in the cover layer provided micrometeoroid protection with increased mobility by reduction in bulk.

(b) Pressure gloves with integral micrometeoroid and thermal protection were provided in lieu of the wear-over, two-glove concept used for EVA during the Gemini IV mission.

(c) An extravehicular visor assembly, consisting of an outer visor for protection from the sun and an inner visor for thermal protection and structural strength, was added to the pilot's helmet.

3.1.2.12.3 Extravehicular Life Support System: The ELSS shown in figure 3.1-11 was designed as a semi-open-loop system utilizing externally supplied oxygen for ventilation and for removal of carbon dioxide. For operation with spacecraft oxygen, the gas was to be delivered to the ELSS through an umbilical which would also supply electrical power, communications, and telemetry, and act as a structural restraint. Approximately two-thirds of the effluent suit-ventilating stream was to be recirculated and the remainder was to be vented overboard by means of a valve which controlled the suit-loop pressure to approximately 3.7 psia. The recirculated gas would have passed through a heat exchanger for removal of excess moisture from the gas and use of the condensed moisture as a heat sink. Electrical heaters were incorporated on the primary-oxygen inlet line and on the ejector to maintain the oxygen temperature within desired limits.

A suit pressure regulator would have withdrawn oxygen from the umbilical, the ESP, or the self-contained chest-pack emergency supply when the suit pressure fell below 3.3 psi. If the primary oxygen from the spacecraft had been interrupted for any reason, a 33-minute emergency oxygen supply, contained within the ELSS chest pack, would have automatically maintained ventilation and pressurization of the extravehicular pilot. If the heat exchanger had failed, actuation of a manual bypass valve would have allowed additional dry oxygen to be supplied downstream of the heat exchanger through the ejector secondary duct into the suit. The ELSS display panel contained the malfunction-detection warning lights and tone devices, and a pressure gage for the emergency oxygen supply. Power for the oxygen heaters, pressure transducers, displays, and warning system was provided through the 25-foot umbilical when it was connected; or by a 24-volt silver-zinc battery installed in the ELSS, when on the 75-foot tether.

3.1.2.12.4 Extravehicular Support Package: The ESP (fig. 3.1-11) was designed to provide the life-support oxygen and the compressed gas for the Hand-Held Maneuvering Unit (HHMU) to enable the extravehicular pilot to maneuver independent of the spacecraft supplies. While operating from the ESP, the only tie to the spacecraft was to have been the 75-foot umbilical which included hardline communications, biomedical instrumentation wiring, and a mechanical tether having a tensile strength of 1000 pounds. The ESP also included a UHF voice transceiver for backup communications. The oxygen for life support and the Freon-14 for propulsion were stored at 5000 psi in a gaseous state in two pressure vessels similar to the ECS secondary-oxygen pressure vessels except that a heater was provided on the ESP outlet line to raise the temperature of the oxygen from the supply tank. With a nominal usage rate of 5.1 lb/hr, the ESP was capable of providing 80 minutes of support. The ESP had a self-contained battery to power the oxygen heater, to energize the oxygen and Freon-14 pressure transducers, and to power the UHF voice transceiver.

3.1.2.12.5 Hand-Held Maneuvering Unit: The HHMU was of the same general design as that used during the Gemini IV mission and would have provided a thrust of approximately 2 pounds over a 200-second time span. The major change was the use of Freon-14 instead of oxygen as the propellant. The Freon-14 was to be supplied by the ESP; consequently, the oxygen supply bottles mounted on the HHMU for the Gemini IV mission were not installed for this mission. Also, the bracket for mounting the EVA camera was not installed on the HHMU.

TABLE 3.1-I.- SPACECRAFT 8 MODIFICATIONS

System	Significant differences between the Spacecraft 8 and Spacecraft 6 configurations
Structure	EVA provisions incorporated.
Communications	No significant difference.
Instrumentation and Recording System	Onboard tape recorder was removable and could record received as well as transmitted voice communications.
Environmental Control System	(a) Cabin heat exchanger and fan removed. (b) Egress oxygen system deleted. (c) Stopper installed over inlet of cabin vent valve. (d) Two 42-pound-capacity tanks installed for storing drinking water and fuel-cell product water. (e) Valve installed for crossfeed between fuel-cell oxygen supply and ECS breathing-oxygen supply. (f) Two coolant pumps installed in each coolant loop.
Guidance and Control	(a) Auxiliary Tape Memory Unit installed. (b) Operational program loaded into computer prior to launch changed because of ATMU storage capability.
Time Reference	Interface provided between ATMU and T_x address command, computer-clock, and computer-write data signals.
Electrical	(a) Fuel Cell Power System used instead of adapter battery module and was same as Spacecraft 7 Fuel Cell Power System except hydrogen regenerative cooling line and insulation on hydrogen supply tank were not incorporated. (b) Analog readout provided for differential pressures of fuel-cell reactants and water.
Propulsion	(a) Redundant heaters added to oxidizer solenoid valves. (b) Reserve-fuel-tank system installed for OAMS.
Pyrotechnics	Seven guillotines installed for releasing EVA and experiment equipment.
Crew-station furnishings and equipment	(a) ATMU monitor and control panel installed. (b) Agena control panel modified so that it could supply power for EVA lights and pyrotechnic devices and for experiments S-9, D-14, and D-15. (c) Fuel Cell Power System monitors and controls installed.

TABLE 3.1-I.- SPACECRAFT 8 MODIFICATIONS - Concluded

System	Significant differences between the Spacecraft 8 and Spacecraft 6 configurations
Crew-station furnishings and equipment (Continued)	(d) Main-bus ammeters deleted to provide space for fuel-cell monitor and control panel. Circuits changed to permit monitoring of main-bus currents on fuel-cell stack ammeters. (e) Switch added for OAMS reserve fuel tank. (f) Two switches installed for pilot control of TDA docking, rigidizing, and unrigidizing sequences. (g) Displays and controls installed for experiments S-9, D-14, and D-15. (h) Ejection-seat system modified to reduce height of egress kit. (i) Light and dimming controls added to illuminate the elapsed-time digital-clock display.
Landing	No significant change.
Postlanding and Recovery	No significant change.
EVA equipment (compared with Gemini IV EVA equipment)	(a) Handrails and Velcro patches added to exterior surface of spacecraft. (b) Handholds and foot supports added to spacecraft adapter equipment section. (c) Adapter-equipment-section thermal curtain redesigned to accommodate EVA equipment. (d) Lights added to adapter assembly for night-side EVA. (e) Mount for 16-mm movie camera installed on adapter assembly. (f) Ring installed on forward surface of R and R section for attaching EVA tether. (g) Hatches modified to incorporate holding devices. (h) EISS provided and stowed in crew-station area. (i) ESP provided and stowed in adapter assembly. (j) Self-contained oxygen propellant tanks and camera bracket were not installed on HHMU as they had been on the Gemini IV HHMU. (k) G4C space suits worn by both crew members and the pilot wore a modified cover layer, modified pressure gloves for thermal protection, and modified EVA visor assembly.

TABLE 3.1-II.- CREW-STATION STOWAGE LIST

Stowage area (See fig. 3.1-8)	Item	Quantity
Centerline stowage container	70-mm camera	1
	16-mm camera	2
	18-mm lens, 16-mm camera	1
	75-mm lens, 16-mm camera	1
	5-mm lens, 16-mm camera	1
	16-mm film magazine	11
	Ring view finder	1
	70-mm camera	1
	70-mm film magazine	4
	Cloud-top spectrometer, Experiment S-7	1
	Mirror mounting bracket	1
Left sidewall containers	Spotmeter and exposure dial	1
	Postlanding kit assembly	1
	Personal hygiene towel	2
	Tissue dispenser	1
	Food, two-man meal	2
	Pilot's preference kit	1
	Urine receiver	1
	Urine hose and filter	1
	Clamp for urine collection device	2
	Plastic zipper bag	4

TABLE 3.1-II.- CREW-STATION STOWAGE LIST - Continued

Stowage area (See fig. 3.1-8)	Item	Quantity
Left aft stowage container	Components for EVA consisting of	1 set
	Standup electrical cable	1
	Umbilical assembly	1
	Jumper cable	2
	Electrical cable extension	1
	Dual connector	2
	Standup tether	1
	ELSS restraint assembly	2
	ELSS hose, short	1
	ELSS hose, long	1
	Penlight	2
	6-inch adjustable wrench	1
	EVA rear-view mirror	1
	EVA hand pad	2
	Knee tether	1
Left pedestal pouch	Waste container	1
	Defecation device	1
	Velcro tape, 1 by 12 in.	4
	Velcro pile, 12 in.	1
Left footwell	Helmet stowage bag	1
	Window shade, reflective	1
Right sidewall containers	Personal hygiene towel	2
	Voice tape cartridge	8
	Food, two-man meal	1

TABLE 3.1-II.- CREW-STATION STOWAGE LIST - Continued

Stowage area (See fig. 3.1-8)	Item	Quantity
Right sidewall container - concluded	Debris cutter	1
	Pilot's preference kit	1
	Penlight	2
	EVA mirror and wrist band	1
	Sunshade	1
	Urine sample bag, Experiment M-5	16
	Latex roll-on cuff (urine system)	6
	Covering for Flight Director Attitude Indicator	1
	Plastic zipper bag	4
	Medical accessory kit	1
Right aft stowage container	16-mm camera (with adapter, 3 film magazines, and EVA remote control cable)	1
	70-mm film magazine	1
	70-mm camera, super-wide angle	1
	Manual inflator, blood pressure	1
	Waste container	2
	Tissue dispenser	1
	Defecation device	4
	Voice tape cartridge	5
	Food, two-man meal	6

TABLE 3.1-II.- CREW-STATION STOWAGE LIST - Continued

Stowage area (See fig. 3.1-8)	Item	Quantity
Right aft stowage container - concluded	Velcro tape, 1 by 12 in.	1
	Circuit breaker and light assembly, 16-mm camera	2
	Urine sample bag, Experiment M-5	8
	Thermal cover, 16-mm camera	1
	35-mm camera and mounting bracket, Experiment S-1	1
Right pedastal pouch	Waste container	1
	Defecation device	1
	Velcro tape, 1 by 12 in.	1
	Velcro hook, 12 in.	1
Right footwell	Sunshade assembly	1
	Helmet stowage bag	1
	Window shade, reflective	1
Plotboard pouch	Orbital path display assembly	1
	Celestial display - Mercator	1
	Celestial display - polar	1
	Flight data book	3
	Circuit-breaker guard	1
Orbital utility pouch	Lightweight headset (with oral temperature probe installed)	2
	Food, two-man meal	1

TABLE 3.1-II.- CREW-STATION STOWAGE LIST - Concluded

Stowage area (See fig. 3.1-8)	Item	Quantity
Orbital utility pouch - concluded	Remote-control cable for EVA 16-mm camera	1
	ELSS mirror	1
	Sextant bracket	2

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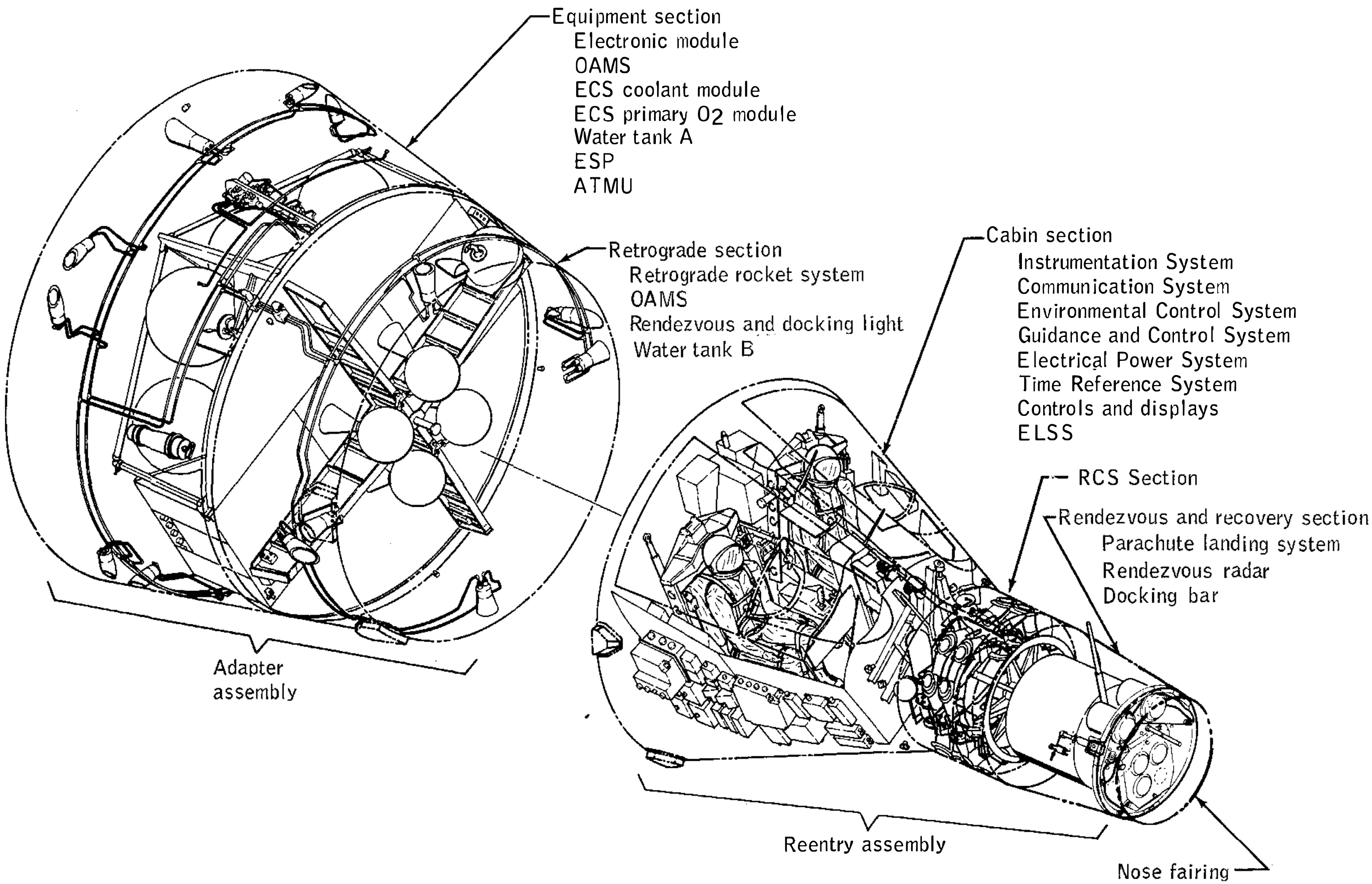

Figure 3.1-1. - Spacecraft arrangement and nomenclature.

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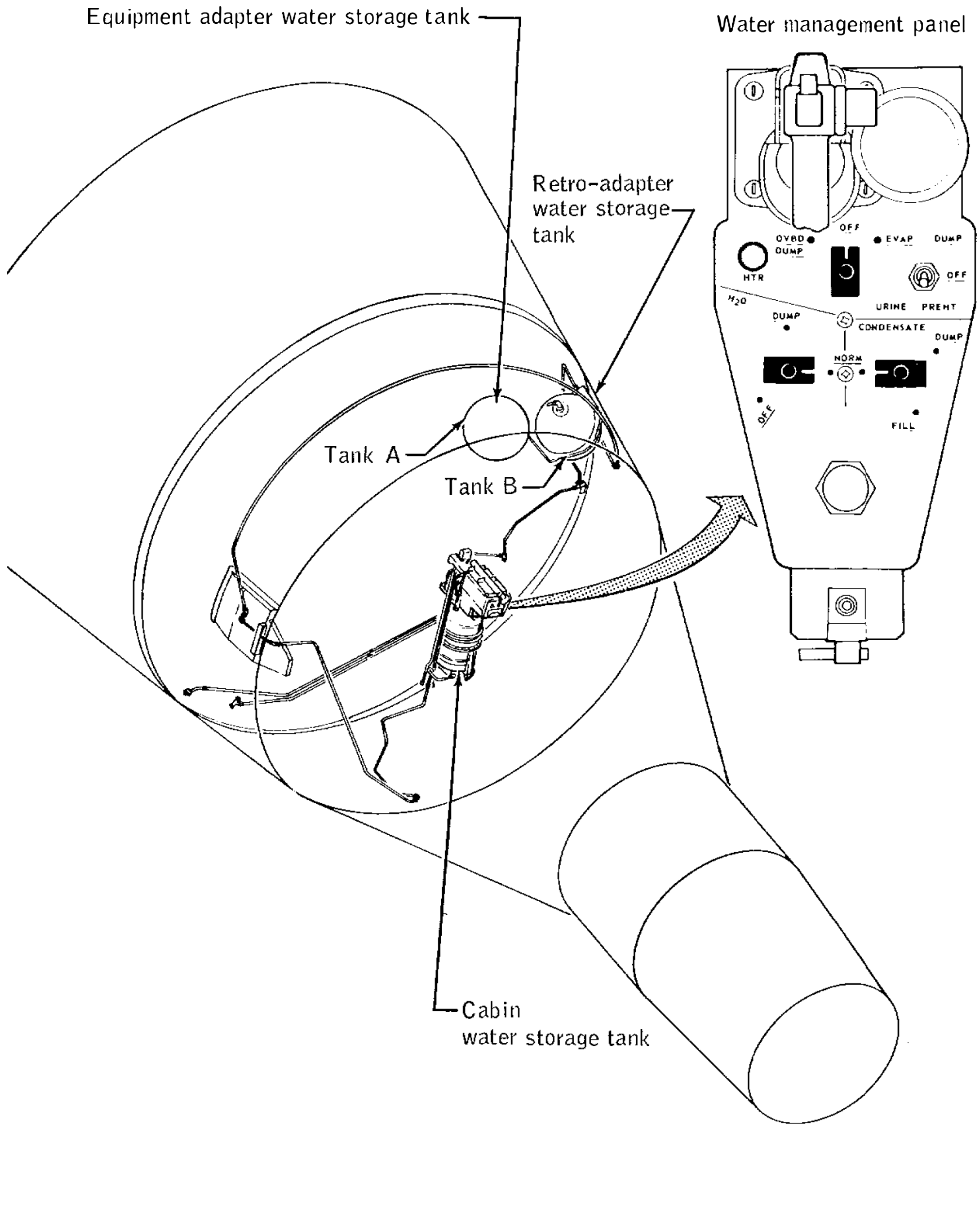

Figure 3.1-2. - Water management system.

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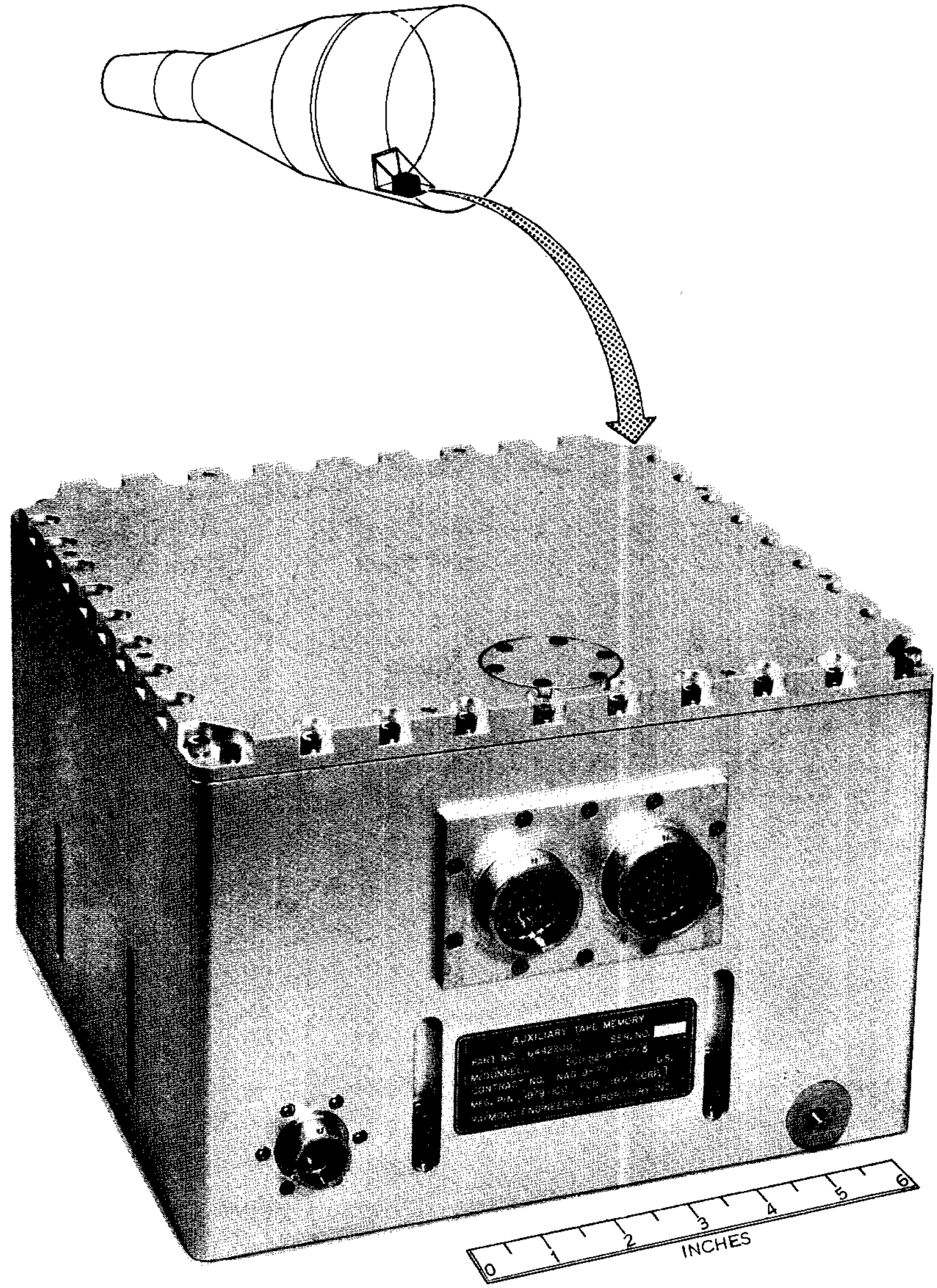

Figure 3.1-3. - Auxiliary tape memory unit.

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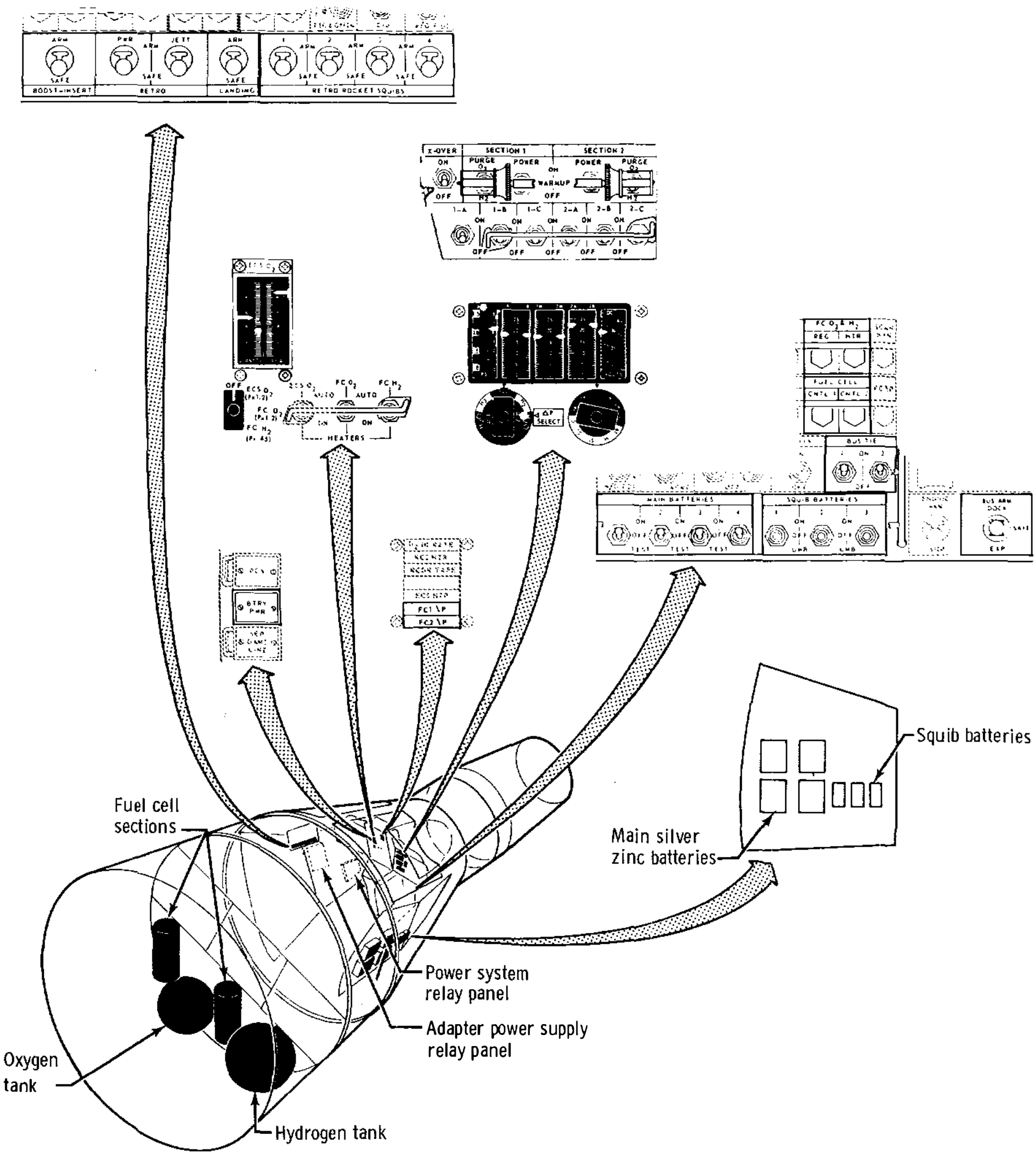

Figure 3.1-4. - Electrical system.

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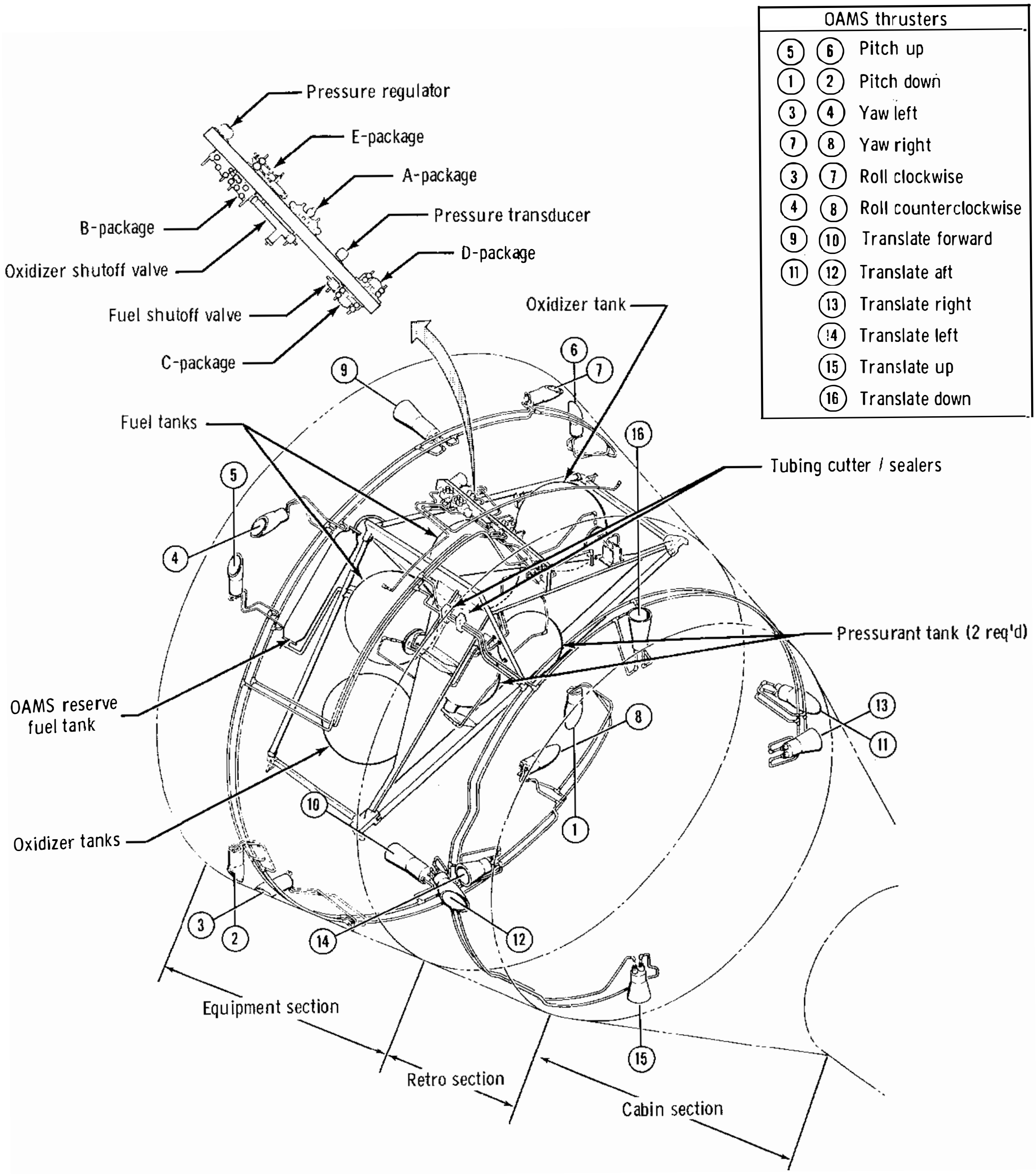

Figure 3. 1-5. - Orbital Attitude and Maneuver System.

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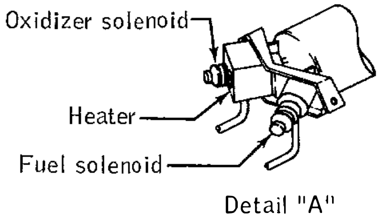

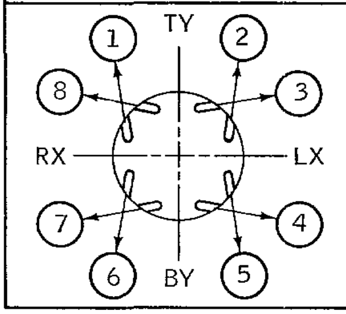

Control modes		
5	6	Pitch up
1	2	Pitch down
3	4	Yaw right
7	8	Yaw left
3	7	Roll right
4	8	Roll left

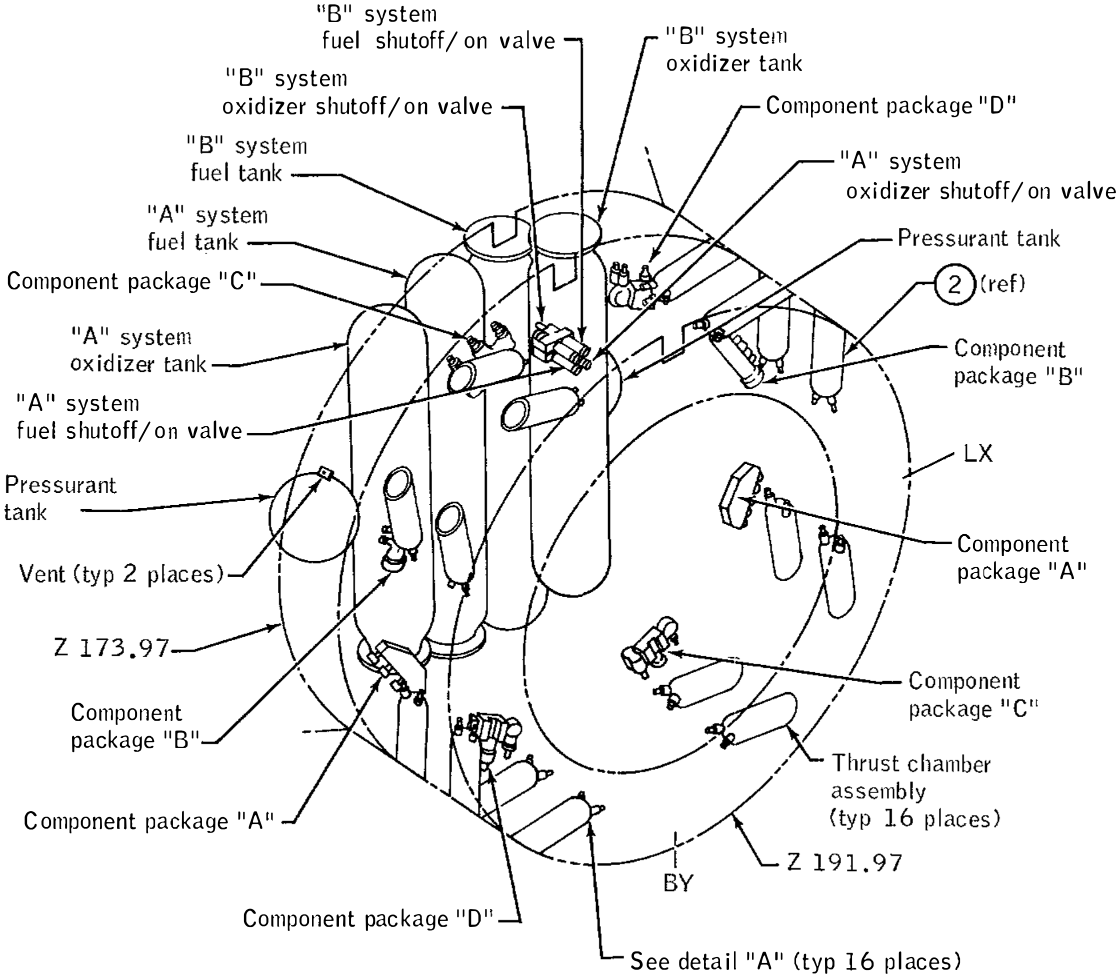

Figure 3.1-6. - Reentry Control System.

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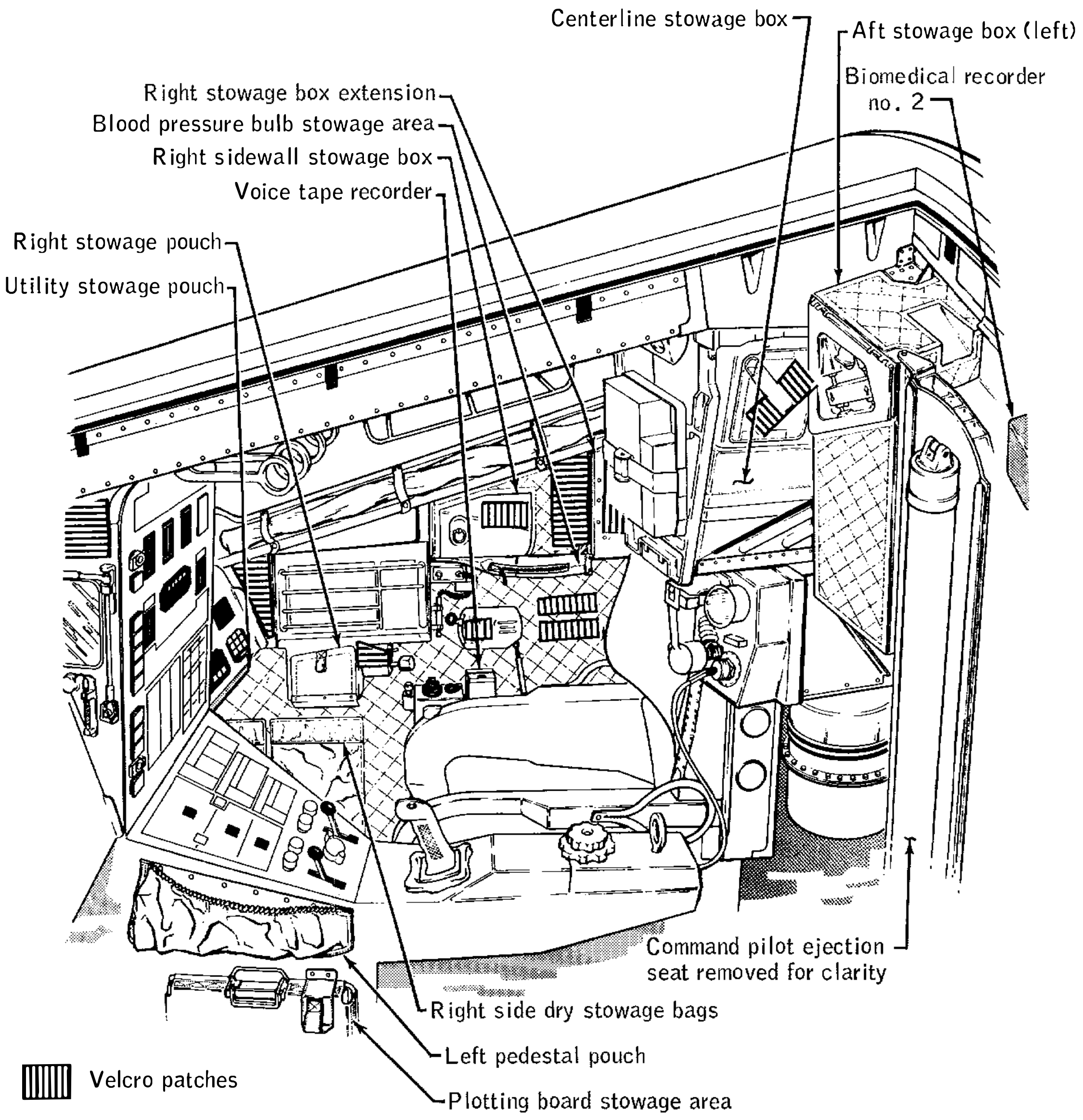

(b) View looking into pilot's side.

Figure 3.1-8. - Concluded.

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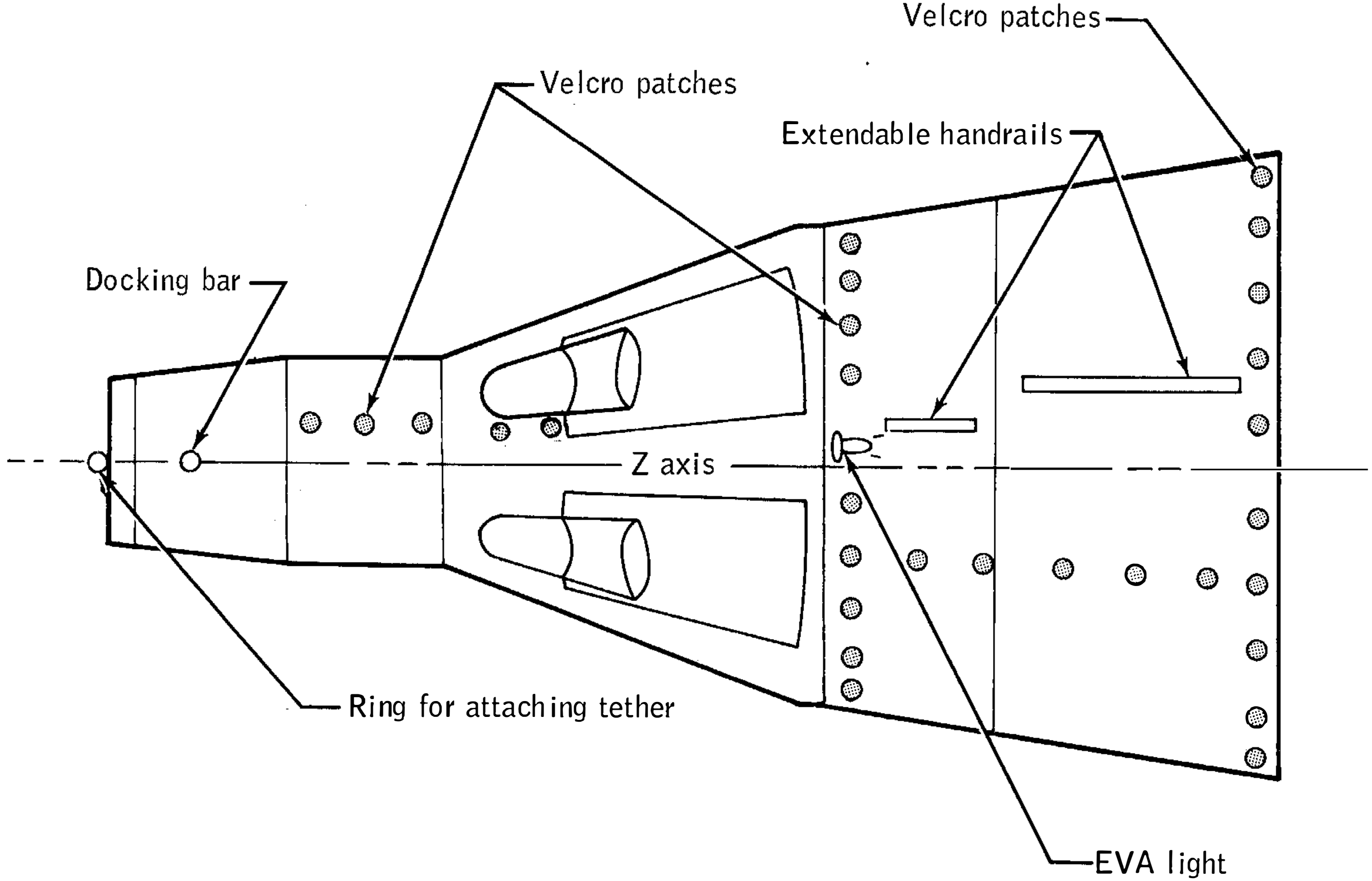

Figure 3.1-9. - Arrangement of EVA provisions on spacecraft.

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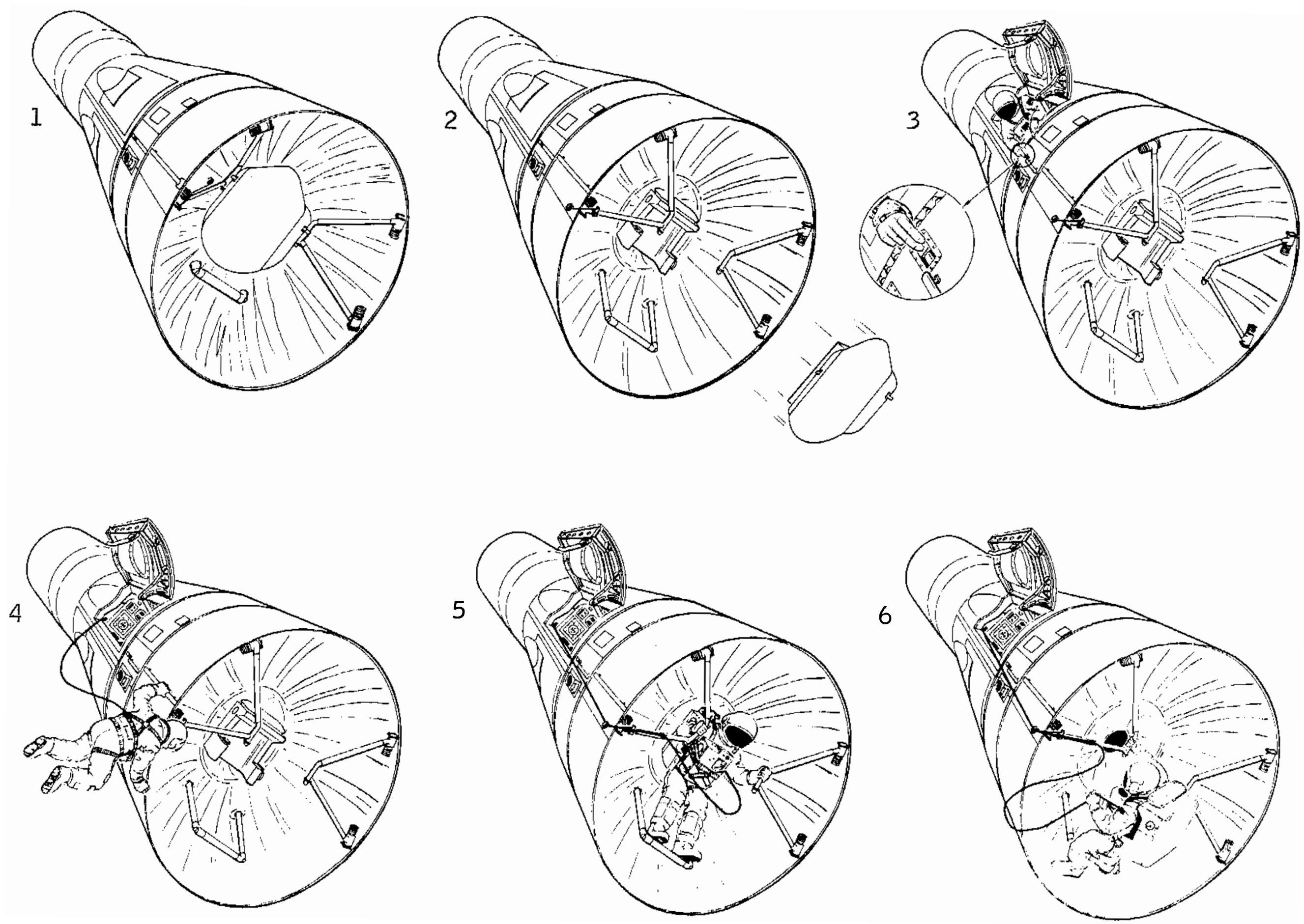

Figure 3.1-10. - Planned sequence for donning extravehicular support package.

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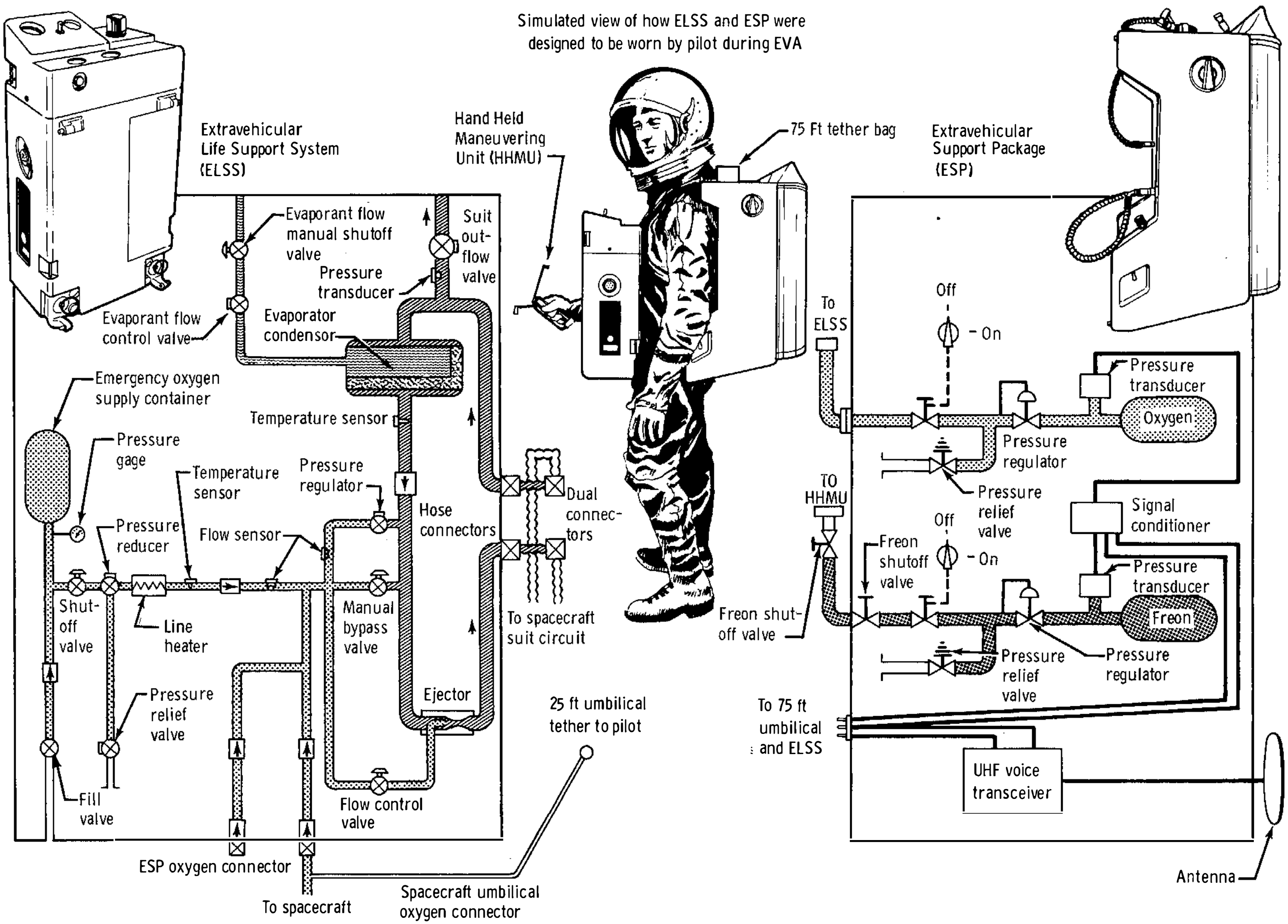

Figure 3. 1-11. - Extravehicular equipment.

3.2 GEMINI LAUNCH VEHICLE

The Gemini Launch Vehicle (GLV-8) was of the same basic configuration as those used for the previous Gemini missions. Table 3.2-I lists the significant differences between GLV-8 and GLV-6 (ref. 7). These modifications are further described in the following paragraphs.

3.2.1 Structure

The cutouts for telemetry antennas were reduced from four to two. On GLV-6, cover plates had been installed over the superfluous cutouts.

3.2.2 Major Systems

3.2.2.1 Propulsion System.- An improved propellant injector was added to the Stage II engine. This injector, developed as part of the Gemini Stability Improvement Program (GEMSIP), used cooled-tip ejector baffles to provide combustion stability in the thrust chamber.

3.2.2.2 Flight Control System.- The time for gain change no. 1 was changed from lift-off (LO) + 110 seconds to LO + 105 seconds.

3.2.2.3 Radio Guidance System.- There were no significant changes to the Radio Guidance System.

3.2.2.4 Hydraulic System.- There were no significant changes to the Hydraulic System.

3.2.2.5 Electrical System.- Stiffeners were added to strengthen the telemetry antenna.

3.2.2.6 Malfunction Detection System.- There were no significant changes to the Malfunction Detection System.

3.2.2.7 Instrumentation System.- There were no significant changes to the Instrumentation System.

3.2.2.8 Range Safety and Ordnance Systems.- There were no significant changes to the Range Safety and Ordnance Systems.

TABLE 3.2-I.- GLV-8 MODIFICATIONS

System	Significant differences between the GLV-8 and GLV-6 configurations
Structure	Telemetry cutouts reduced from four to two.
Propulsion	Improved injector installed on Stage II engine
Flight Controls	Time for gain change no. 1 changed from 110 seconds to 105 seconds after lift-off
Radio Guidance	No significant change
Hydraulics	No significant change
Electrical	Stiffeners added to telemetry antenna
Malfunction Detection	No significant change
Instrumentation	No significant change
Range Safety and Ordnance	No significant change

3.3 WEIGHT AND BALANCE DATA

Weight and balance data for the Gemini VIII Space Vehicle are as follows:

Condition	Weight (including spacecraft), lb (a)	Center-of-gravity location, in. (a), (b)		
		X	Y	Z
Ignition	345 359	774.7	-0.049	59.96
Lift-off	341 671	775	-0.050	59.95
Stage I burnout (BECO)	85 276	349	-0.202	59.836
Stage II start of steady-state combustion	73 790	343	-0.079	59.021
Stage II engine shutdown (SECO)	14 326	288	-0.300	59.700

[a] Weights and center-of-gravity data were obtained from the GLV contractor.

[b] Refer to figure 3.0-1 for the Gemini Space Vehicle coordinate system. Along the X-axis, the center-of-gravity is referenced to GLV station 0.00. Along the X-axis, the center-of-gravity location is referenced to buttock line 0.00 (vertical centerline of horizontal vehicle). Along the Z-axis, the center-of-gravity is referenced to waterline 0.00 (60 inches below the horizontal centerline of the horizontal vehicle).

Spacecraft 8 weight and balance data are as follows:

Condition	Weight, lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Launch, gross weight	8351.31	0.16	+1.66	105.12
Retrograde	5726.36	0.26	-1.27	129.27
Reentry (0.05g)	4879.89	0.1	-1.47	136.21
Main parachute deployment	4454.89	0.09	-1.61	129.14
Touchdown (no parachute)	4344.08	0.10	-1.66	127.08

[a]Refer to figure 3.0-1 for spacecraft coordinate system. The X-axis and the Y-axis are referenced to the centerline of the spacecraft. The Z-axis is referenced to a plane located 13.44 inches aft of the launch vehicle—spacecraft separation plane.

3.4 GEMINI AGENA TARGET VEHICLE

The Gemini Agena Target Vehicle (GATV 5003) for the Gemini VIII mission was similar to GATV 5002 used for the Gemini VI mission (ref. 6). Table 3.4-I lists the significant differences between these two vehicles. These modifications are further described in the following paragraphs.

3.4.1. Structure

3.4.1.1 Gemini Agena Target Vehicle.- To preclude the possibility of the jettisoned aerodynamic shroud locking onto the coils of the initial separation spring on the GATV, the spring mount and cover were modified.

3.4.1.2 Target Docking Adapter.- Modifications were added to the TDA to complete the circuits from the RIGID-OFF-STOP and the OFF-UNDOCK switches on the spacecraft instrument panel (see section 3.1.2.9). These circuits provided the flight crew with the capability of controlling the rigidizing, unrigidizing, and docking sequences if the automatic sequencing circuits or command system had failed. The modifications added two hardline umbilicals, two limit switches which sensed spacecraft separation from the TDA, and wiring changes to the relay panel. One of the parallel wires in the spacecraft-to-GATV ARM-STOP circuitry was used to facilitate this modification.

An RFI filter was added external to the mooring drive motor. Another RFI filter was added external to the latch-release actuator to replace a previously installed internal filter.

To facilitate the planned extravehicular activity (EVA), three Velcro patches were added to the external surface of the TDA in line with the top acquisition light. Brackets and a fairing were also installed for mounting the micrometeorite collector (Experiment S-10).

To provide an apparent increased intensity and greater range of the acquisition lights mounted on the TDA, the flash rate was changed from 65 to 55 flashes per minute and a reflector was added to the lower light to decrease the cone angle.

3.4.2 Major Systems

3.4.2.1 Propulsion System.- The Primary Propulsion System (PPS) was modified to insure that an adequate amount of oxidizer entered the engine thrust chamber prior to the initiation of fuel flow. This was accomplished by altering the engine electrical control circuits to the configuration shown in figure 3.4-1. As a result of this modification, the pilot-operated solenoid valve that controlled the main fuel valve was not energized until the oxidizer feed pressure (OFP) at the main oxidizer valve or the oxidizer manifold pressure (OMP) at the injector was sufficient to insure that an oxidizer preflow of 5 to 8 pounds had been provided. However, in actual practice, operation of the OFP switch alone will provide a preflow of 11 to 14 pounds. The fuel and oxidizer main valves, which had been modified for GATV 5002, were changed to the standard Agena configuration and the turbine-overspeed electronic gate was inhibited from cutting off the engine during the ascent maneuver. Also the method of turbine-overspeed engine cutoff was changed as shown in figure 3.4-1. As a result of these changes, the expected engine start sequence was as shown on table 3.4-II.

3.4.2.2 Electrical System.- The changes in the PPS required circuit modifications within the Electrical System. The modifications included rewiring of relays and connector pins in the aft safe/arm junction box and the addition of diodes for spike suppression across the oxidizer and fuel solenoids. A new junction box was installed to permit pressure-switch control of the pilot-operated solenoid valve. In addition, shock mounting was provided for various electrical junction boxes and components located in the GATV aft section.

3.4.2.3 Flight Control System.- As a result of the modification to inhibit PPS turbine-overspeed shutdown during the ascent phase of flight, a relay was added and a patch panel was rewired in the flight-control junction box.

3.4.2.4 Communications and Command System.- To improve reliability and overall performance of the command system, minor circuit changes and component mounting modifications were incorporated in the command controller and in the programmer. A filter box was added to reduce transients on the power line when the C-band, S-band, and telemetry systems were turned on and off. A 9-hour plug, instead of the 3-hour plug used on GATV 5002, was used in the emergency reset timer (ERT) which, when it times out, normally energizes or turns on the L-band transponder, C-band and S-band transponders, tape recorder, and telemetry system, and also enables the UHF to receive ground commands. At lift-off, the GATV 5003 programmer memory was loaded with all zeros while the GATV 5002 (used on Gemini VI mission) programmer memory was loaded with two commands: (1) ERT reset, and (2) L-band off. Antenna locations are shown in figure 3.4-2.

3.4.2.5 Range Safety System.- There were no significant changes to the Range Safety System.

TABLE 3.4-I.- GATV-5003 MODIFICATIONS

System	Significant differences between GATV 5003 (Gemini VIII mission) and GATV 5002 (Gemini VI mission) configurations
Structure	(a) Spring mount and cover modified to prevent possible interference during jettisoning of aerodynamic shroud.
	(b) Two hardline umbilicals and two limit switches added with wiring changes to complete circuits from TDA control switches on spacecraft instrument panel.
	(c) RFI filters added to mooring-drive motor and latch-release actuator in TDA.
	(d) Velcro patches and mounting bracket for micrometeorite collector (Experiment S-10) installed on TDA.
	(e) Acquisition lights mounted on TDA modified to decrease flash rate and to add reflector to lower light.
Propulsion	(a) PPS main oxidizer and fuel valves modified to standard Agena configuration.
	(b) Two pressure switches installed in PPS oxidizer system.
	(c) Circuit installed to inhibit turbine-overspeed electronic gate during ascent phase of the flight.
Electrical	(a) Wiring changes incorporated to complete circuits for PPS modifications.
	(b) Pilot-operated solenoid-valve junction box installed.
	(c) Shock mounting provided for electrical junction boxes and components located in GATV aft section.

TABLE 3.4-I.- GATV-5003 MODIFICATIONS - Concluded

System	Significant differences between GATV 5003 (Gemini VIII mission) and GATV 5002 (Gemini VI mission) configurations
Flight Control	Wiring changes and relay added to flight-control junction box to complete inhibit circuit for turbine-overspeed electronic gate.
Communications and Command	Minor circuit and component mounting modifications for improved reliability of command controller and programmer.
Range Safety	No significant change.

TABLE 3.4-II.- NOMINAL PPS START SEQUENCE

Function	Time, seconds
Fire signal	0.0
Oxidizer gas generator valve open	0.040
Fuel gas generator valve open	0.075
Gas generator ignition	0.210
Main oxidizer valve open	0.400
Start gas generator/pump bootstrapping	0.600 to 0.800
Oxidizer manifold pressure (OMP) switch actuates	0.875 to 0.950
Pilot-operated shut-off valve, pilot open	OMP + 0.020
Fuel valve starts to open	1.020 to 1.050
Fuel enters thrust chamber	1.100
Ignition	1.115
Main fuel valve full open	1.170
Steady-state performance	15.0 to 20.0

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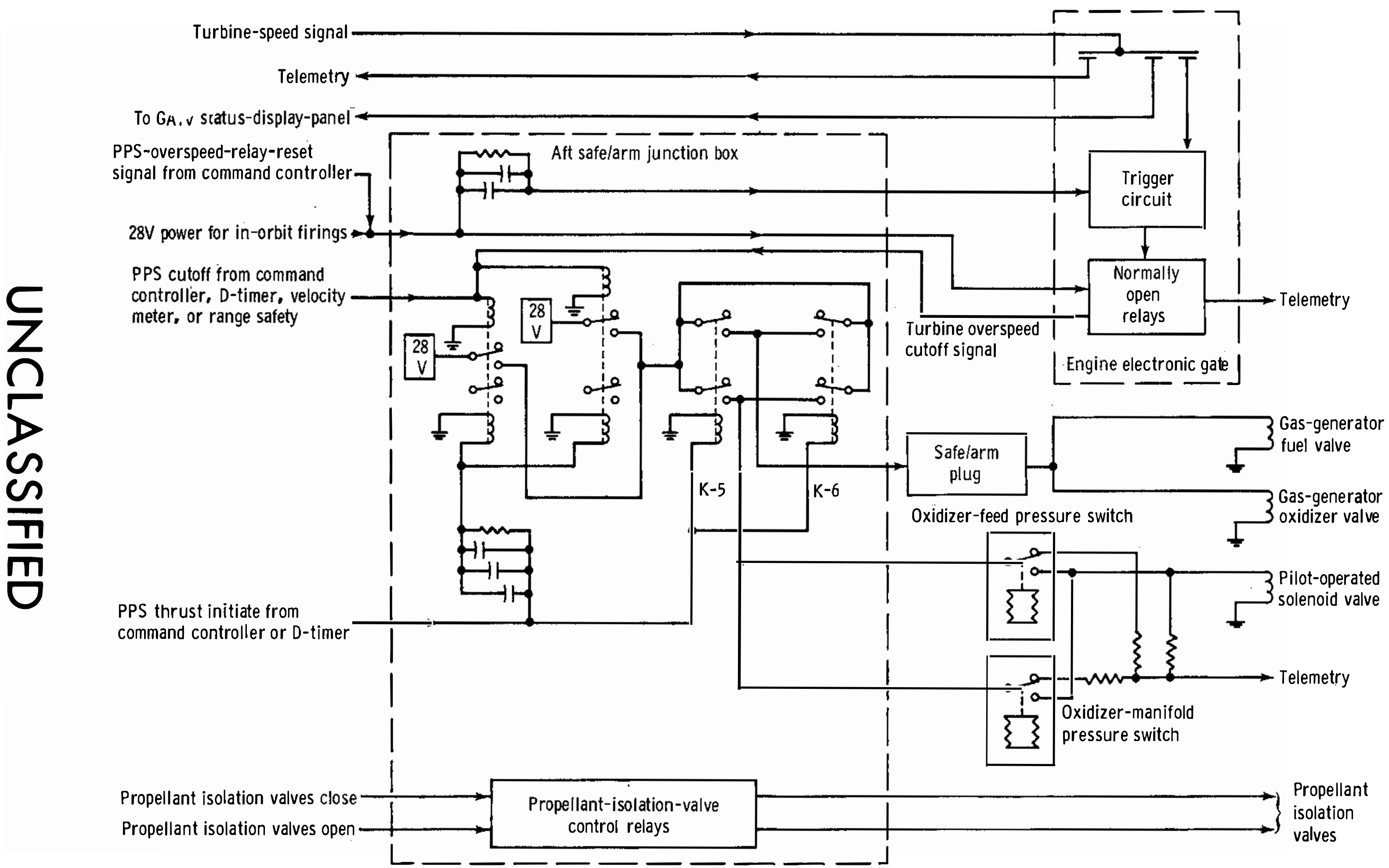

Figure 3.4-1. - GATV Primary Propulsion System control circuits.

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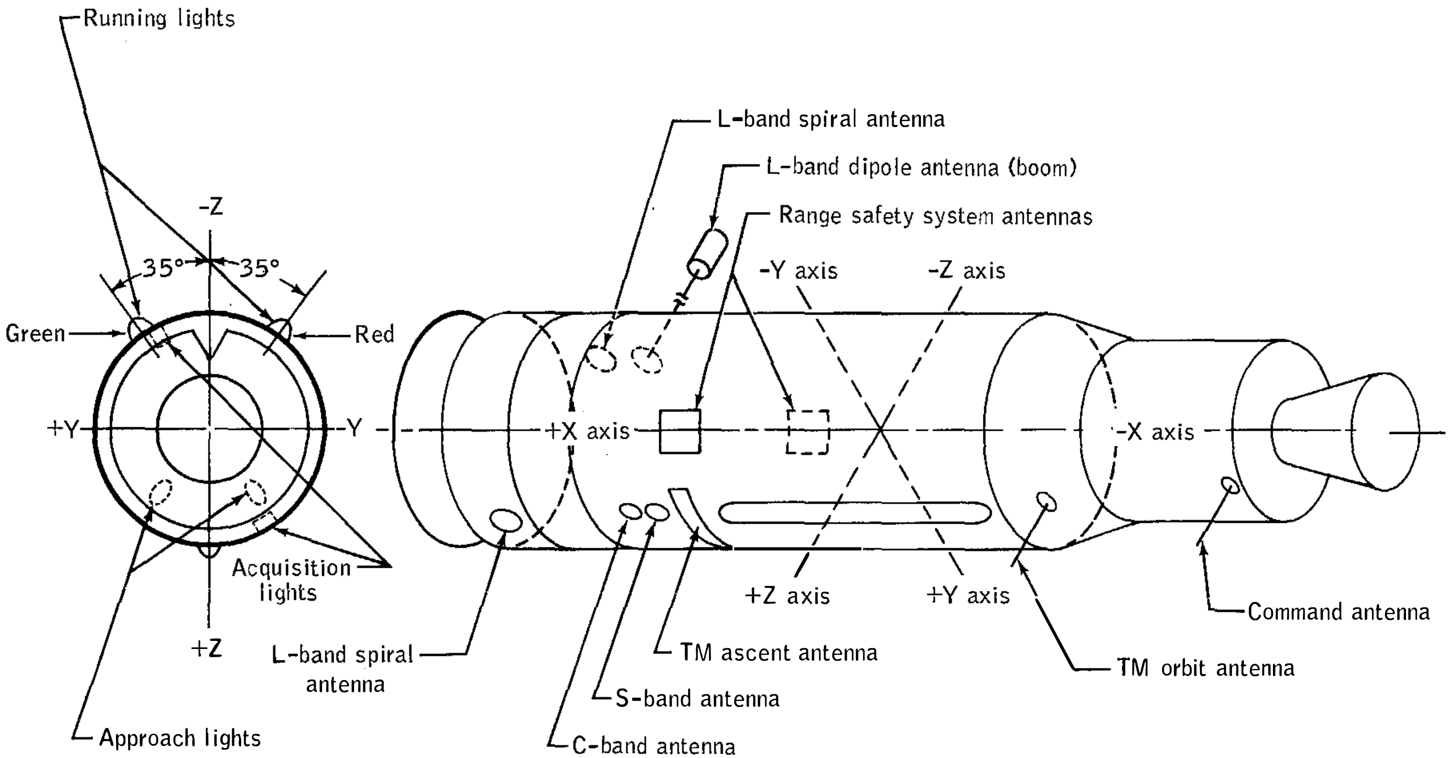

Figure 3.4-2. - GATV antenna locations.

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3.5 TARGET LAUNCH VEHICLE

The Target Launch Vehicle (TLV-5302) was an Atlas Standard Launch Vehicle (SLV-3) and was of the same basic configuration as TLV-5301 used for the Gemini VI mission (ref. 6). Table 3.5-I lists the significant differences between TLV-5302 and TLV-5301. These modifications are further described in the following paragraphs.

3.5.1 Structure

There were no significant changes in the TLV structure.

3.5.2 Major Systems

3.5.2.1 Propulsion System.- In the vernier-engine fuel-purge system, the orifices were removed and fuel check valves having internal orificing were installed.

3.5.2.2 Guidance System.- In the rate-beacon klystron, the insulating washer material was changed from mica to Kapton.

3.5.2.3 Flight Control System.- In the autopilot circuitry, special-quality diodes were installed. In the programmer circuit, a redundant electrical path was provided around the safing contacts of the 28-volt relay and through the safe/arm switch when the programmer was in the armed condition.

3.5.2.4 Electrical System.- In the electrical distribution box (D-box), two parallel isolation diodes were added in the automatic fuel cutoff (AFCO) line and two were also added in the manual fuel cutoff (MFCO) line. Also in the D-box, an unnecessary filter capacitor was deleted from the 28-volt power line to the autopilot programmer, the motion limit-switch circuitry and destructor circuitry were modified to provide greater reliability, and current-limiting resistors were added to the monitoring circuits of the battery for the Range Safety Command and Instrumentation Systems.

3.5.2.5 Pneumatic System.- In the propellant pressurization system, the thick-skinned helium-storage spheres were replaced by lightweight, pressure-welded storage spheres. In the propellant-tank relief valves, the silicon/fiberglass diaphragms were replaced with silicon/Dacron diaphragms.

3.5.2.6 Instrumentation System.- There were no significant changes in the Instrumentation System.

3.5.2.7 Range Safety System.- In the Range Safety System, the destructor unit was replaced by an improved model.

TABLE 3.5-I.- TLV-5302 MODIFICATIONS

System	Significant differences between TLV-5302 (Gemini VIII mission) and TLV-5301 (Gemini VI mission) configurations
Structure	No significant change.
Propulsion	Orifices removed from vernier-engine fuel-purge system, and check valves with internal orificing installed.
Guidance	Washer material in rate-beacon klystron changed from mica to Kapton.
Flight Control	(a) Special-quality diodes used in autopilot circuitry.
	(b) Redundant electrical path provided around 28-volt relay safing contacts.
Electrical	(a) Two parallel isolation diodes added to AFCO line and two to MFCO line in D-box.
	(b) Filter capacitor deleted from power line to autopilot programmer.
	(c) Motion limit-switch circuitry and destructor circuitry modified for greater reliability.
	(d) Current-limiting resistors added to RSC/Instrumentation System battery monitoring circuits.
Pneumatics	(a) Thick-skinned helium storage spheres replaced by lightweight spheres.
	(b) Silicon/fiberglass diaphragms in propellant-tank relief valves replaced by silicon/Dacron diaphragms.
Instrumentation	No significant change.
Range safety	Destructor unit replaced by improved model.

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3.6 WEIGHT AND BALANCE DATA

Weight and balance data for the Gemini Atlas-Agena Target Vehicle are as follows:

Condition	Weight (including GATV), lb (a)	Center-of-gravity location, in. (a)		
		X	Y	Z
Ignition	281 805	-	-	-
Lift-off	279 387	845.13	-0.48	-0.39
Booster engine cutoff (BECO)	73 565	849.21	-1.72	-1.45
Sustainer engine shutdown (SECO)	26 815	573.44	-2.01	-3.28

[a]Refer to figure 3.0-2(c) for GAATV coordinate system.

Gemini Agena Target Vehicle weight and balance data are as follows:

Condition	Weight, lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Launch, gross weight	18 097	339.6	+0.5	0
Separation	17 686	337.0	+0.5	0
Insertion weight (in-orbit)	7 116	343.0	+1.2	-0.10

[a]Refer to figure 3.0-2(b) for GATV coordinate system.

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4.0 MISSION DESCRIPTION

4.1 ACTUAL MISSION

The Gemini VIII mission was initiated at lift-off of the Gemini Atlas-Agena Target Vehicle (GAATV) on March 16, 1966, at 15:00:03.127 Greenwich mean time (G.m.t.). During vertical flight, the vehicle was rolled from a pad azimuth of 105 degrees to a flight azimuth of 84.36 degrees. Sustainer steering was used to obtain the desired longitude of the ascending mode and inclination angle. No booster steering was required.

The flight-controller and range-safety plotboards all indicated a nominal Target Launch Vehicle (TLV) flight. The inertial flight-path angle was slightly depressed at approximately 40 000 feet by a wind shear at this altitude. A slight crossrange deviation was noted at Gemini Agena Target Vehicle (GATV) separation; however, this was well within a 3-sigma tolerance. Separation was smooth with low angular rates.

The GATV performed as planned, executing the 90 deg/min pitchdown rate after separation and continuing this rate until the D-timer started the -3.99 deg/min orbital geocentric pitch rate. The GATV achieved a near-circular orbit with a perigee of 158.8 nautical miles and an apogee of 161.3 nautical miles (referenced to a spherical earth with a radius equal to the radius of Launch Complex 19) 250.9 seconds after vernier engine cutoff (VECO).

One hour 40 minutes 59.262 seconds after GAATV lift-off, the GLV was launched with lift-off at 16:41:02.389 G.m.t. on a rendezvous launch azimuth of 99.9 degrees. The preflight nominal azimuth had been calculated to be 98.7 degrees, but minor deviations in the GAATV launch trajectory required a 1.2-degree shift in launch azimuth to effect a nominal rendezvous. The flight-controller plotboards indicated a launch trajectory that was nominal in every respect, except for a slight deviation in inertial flight-path angle. This deviation was caused by the wind shear at approximately 40 000 feet. The earlier launch of the GAATV for the Gemini VIII mission had experienced a similar deviation for the same reason.

Vehicle closed-loop steering was good in that it corrected an out-of-plane velocity of approximately 350 ft/sec. An erratic pitchdown rate was observed near second-stage engine cutoff (SECO); however, its effect was minor (see section 5.2.5). At 27.4 seconds after SECO, the crew performed an 8.0-second separation thrust using the Orbital

Attitude and Maneuver System (OAMS) and the spacecraft was then in an elliptical orbit which had a perigee of 86.3 nautical miles and an apogee of 146.7 nautical miles with the spacecraft trailing the GATV by approximately 1060 nautical miles.

During the period after insertion and before rendezvous, the crew completed the insertion checklist, reconstituted one meal, and successfully carried out a fuel-cell purge. Experiment S-9 (Nuclear Emulsion) packages were activated and sequences 1 and 2 of Experiment S-3 (Frog Egg Growth) were also performed during this prerendezvous period.

The maneuvers for rendezvous with the GATV consisted of five mid-course maneuvers and four terminal-phase maneuvers. The first mid-course maneuver was a height adjustment $\left(N_{H1}\right)$ performed using forward-firing thrusters in a retrograde direction with attitude control in the platform mode. This maneuver was accomplished using the platform for attitude reference and for determining the applied thrust. The maneuver was preceeded by a 15-minute platform alignment, as were all mid-course maneuvers except the vernier height adjustment. The maneuver was initiated over the Texas network station during the first revolution at 1:34:37 g.e.t. and lowered the spacecraft apogee from 147 to 145.5 nautical miles.

The second mid-course maneuver was a phase-adjust maneuver $\left(N_{C1}\right)$ performed in a posigrade direction using the aft-firing thrusters, again with the aid of the platform and computer but with attitude control in the rate-command mode. The maneuver was initiated during revolution 2 at 2:18:26 g.e.t., out of range of network stations. The maneuver increased the perigee from 86.3 to 113.5 nautical miles.

The third mid-course maneuver was a plane change $\left(N_{PC}\right)$ performed with the aft-firing thrusters directed in a southerly direction using the platform and computer, with attitude control in the rate-command mode. The maneuver was initiated over the Hawaii network station during revolution 2 at 2:45:53 g.e.t.

The fourth mid-course maneuver was a vernier height adjustment $\left(N_{H2}\right)$ performed in a posigrade direction using the aft-firing thrusters in the rate-command mode. This maneuver was accomplished using the platform for attitude reference but on a delta-time basis. The maneuver was initiated over the Guaymas network station during revolution 2 at 3:03:42 g.e.t.

The fifth and final mid-course maneuver was a coelliptical maneuver $\left(N_{SR}\right)$ performed 21 degrees pitched down from the posigrade direction using aft-firing thrusters in the rate-command mode and using the platform and computer displays for reference. The maneuver was initiated over the Tananarive station during revolution 3 at 3:48:10 g.e.t., approximately 35 seconds later than planned. Following completion of mid-course maneuvers, the spacecraft orbit had a perigee of 144.6 nautical miles and an apogee of 146.5 nautical miles. The difference in height between the GATV and spacecraft after these maneuvers varied between 13.5 nautical miles and 14.7 nautical miles, and the two vehicles were orbiting in very nearly the same plane. Initial contact, prior to the terminal-phase maneuvers, between the GATV and spacecraft was made by the rendezvous radar at a range of 180 nautical miles, followed by optical contact at 76 nautical miles.

The terminal-phase-initiate (TPI) maneuver was performed with the aft-firing thrusters at an effective pitch-up angle of 31.3 degrees and a yaw-left angle of 16.8 degrees with attitude control in the rate-command mode. This maneuver was accomplished closed loop and was preceeded by a 13-minute platform alignment. The maneuver was initiated just prior to telemetry acquisition by the Tananarive station during revolution 4 at 5:14:56 g.e.t. Two intermediate corrections were performed at 12 and 24 minutes after TPI when the central angle (ωt), through which the GATV was to travel from the initiation of the terminal phase to rendezvous, equaled 81.8 and 33.6 degrees, respectively.

Terminal-phase maneuvers were completed with the performance of braking using forward-firing thrusters with attitude control in the rate-command mode. Braking maneuvers were initiated over the Coastal Sentry Quebec tracking ship during revolution 4 at 5:43:09 g.e.t. The braking maneuvers were performed with one major maneuver and eight short firings over the next 10 minutes. Braking maneuvers were accomplished visually, but using rendezvous-radar data for measuring the range and range rate. At the conclusion of the braking maneuvers, the range between the spacecraft and GATV was 150 feet and there was no relative velocity between the two vehicles.

Following the conclusion of braking maneuvers, station keeping was accomplished at ranges varying between 150 and 50 feet for approximately 36 minutes prior to docking. During station keeping, the flight crew used the telescopic feature of the sextant to observe the GATV status-display panel and monitor the GATV status. A 13-minute blunt-end-forward (BEF) platform alignment was accomplished during station keeping in both platform and pulse control modes. Docking was successfully completed over the Rose Knot Victor tracking ship during the fifth revolution at 6:33:22 g.e.t.

Following completion of docking, a command was sent from the spacecraft directing the GATV Attitude Control System (ACS) to yaw the docked vehicles to the right. A 90-degree maneuver was completed in 55 seconds with a yaw rate slightly greater than 1.5 deg/sec.

At 7:00:00 g.e.t., out of range of network stations and with the docked spacecraft and GATV configured for the platform parallelism test, the GATV recorder was commanded ON. Shortly after this time, at 7:00:26.7 g.e.t., roll and yaw rates began to develop; however, there was no visual or audible evidence of spacecraft thruster firing noted by the crew. To isolate the source of the anomaly, the GATV ACS was deactivated by a command from the crew and the spacecraft OAMS was activated. The roll rates initially were reduced, but then began to increase upon release of the hand controller. The GATV ACS was again commanded ON to determine if GATV thrusters would reduce the angular rate. No improvement was noted and the ACS was commanded OFF at 7:12:38.6 g.e.t. An effort was then made to isolate the problem by switching to secondary attitude control electronics with no success. At 7:15:12.3 g.e.t., when rates were reduced sufficiently to avoid recontact, the vehicles were undocked with a separation thrust using the forward-firing thrusters.

After undocking, the angular rates immediately started to increase, thus verifying that the problem was in the spacecraft attitude control system. As rates increased to 30 deg/sec, the crew selected the OAMS rate-command mode. Rates were reduced a slight amount; however, the Attitude Control and Maneuver Electronics (ACME) bias power was inadvertently interrupted, which deactivated the hand controller and prevented the crew from controlling the spacecraft. As rates began increasing toward a level of 300 deg/sec, the crew activated the Reentry Control System (RCS) in the previously selected OAMS rate-command mode; however, the hand controller was inoperative because ACME bias power was off, and no control could be obtained. Subsequently the OAMS circuit breakers were opened, the RCS was placed in DIRECT-DIRECT, and the rates were controlled using both rings of the RCS. After the crew determined that control was available with the RCS in DIRECT-DIRECT, the RCS A-ring was turned off. Angular rates were slowly decreased using the RCS B-ring and the spacecraft was finally brought to a stable attitude at 7:25:30 g.e.t. Response from the hand controller was regained by resetting ACME circuit breakers and switches. Control of the spacecraft with the OAMS was later re-established after deactivating thruster no. 8 of the OAMS.

A decision was made to terminate the mission in the seventh revolution with recovery in the secondary landing area in the western Pacific Ocean 500 miles east of Okinawa. Prior to retrofire, the pre-retrofire checklist was completed, a fuel-cell purge was successfully accomplished, and a 22-minute platform alignment was performed. Countdown of the event timer was started over the Rose Knot Victor tracking ship on revolution 7, followed by retrofire near the Kano network station at 10:04:47 g.e.t. Reentry was nominal and landing occurred within 7 miles of the planned landing point.

The crew of one of the rescue aircraft sighted the spacecraft while it was on the main parachute. Pararescuemen, although hampered by a rougher-than-anticipated sea state, attached and inflated the flotation collar within 45 minutes after spacecraft landing. Recovery of the spacecraft and crew was accomplished by the destroyer U.S.S. Leonard F. Mason approximately 3 hours after touchdown.

After reentry of the spacecraft, the GATV was commanded from the ground to carry out a series of maneuvers (table 4.3-VIII). Section 4.3.2.2 contains a description of these maneuvers. The GATV was left in a near-circular parking orbit with a perigee of 217.6 nautical miles and an apogee of 220.4 nautical miles for possible rendezvous activities in future missions. The acquisition lights were programmed to turn on 123 days after GAATV lift-off.

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4.2 SEQUENCE OF EVENTS

The times at which major events were planned and executed are presented in table 4.2-I for the Gemini Space Vehicle and in table 4.2-II for the Gemini Atlas-Agena Target Vehicle.

TABLE 4.2-I.- SEQUENCE OF EVENTS - GEMINI SPACE VEHICLE

Event	Planned time, g.e.t.	Actual time, g.e.t.	Difference, sec
Launch phase, sec			
Stage I engine ignition signal (87FS1)	-3.40	-3.39	+0.01
Stage I MDTCPS makes subassembly 1	-2.30	-2.30	0.00
Stage I MDTCPS makes subassembly 2	-2.30	-2.37	-0.07
TCPS subassembly 1 and subassembly 2 make	-2.20	-2.29	-0.09
Shutdown lockout (backup)	-0.10	-0.10	0.00
Lift-off (pad disconnect separation)	0.00 (16:41:02 G.m.t.)	0.00 (16:41:02.39 G.m.t.)	0.00 0.39
Roll program start	8.48	8.48	0.00
Roll program end	20.48	20.47	-0.01
Pitch program rate no. 1 start	23.04	22.98	-0.06
Pitch program rate no. 1 end, no. 2 start	88.32	88.24	-0.08
Control system gain change no. 1	104.96	104.76	-0.20
First IGS update sent	105.00	105.00	0.00
Pitch program rate no. 2 end, no. 3 start	119.04	118.87	-0.17
Stage I engine shutdown circuitry armed	144.64	144.41	-0.23
Second IGS update sent	145.00	145.00	0.00
Stage I MDTCPS unmake	153.63	154.58	+0.95
BECO (stage I engine shutdown (87FS2))	153.71	154.61	+0.90
Staging switches actuate	153.71	154.61	+0.90
Signals from Stage I rate gyro package to flight control system discontinued	153.71	154.61	+0.90
Hydraulic switchover lockout	153.71	154.61	+0.90
Stage II engine ignition signal (91FS1)	153.71	154.61	+0.90
Control system gain change	153.71	154.61	+0.90
Stage separation begins	154.61	155.29	+0.68
Stage II engine MDFJPS make	154.61	155.27	+0.66
Pitch program rate no. 3 ends	162.56	161.72	-0.84
RGS guidance enable	162.56	161.65	-0.91
First guidance command signal received by TARS	169.00	168.40	-0.60
Stage II engine shutdown circuitry armed	317.44	316.29	-1.15
SECO (stage II engine shutdown (91FS2))	335.59	337.54	+1.95
Redundant stage II shutdown	335.59	337.56	+1.97
Stage II MDFJPS break	335.89	337.68	+1.79
OAMS on	355.59	362.94	+7.35
Spacecraft separation (shape-charge fired)	355.59	365.66	+10.07
OAMS off	368.97	370.94	+1.97

TABLE 4.2-I.- SEQUENCE OF EVENTS - GEMINI SPACE VEHICLE - Concluded

Event	Planned time, g.e.t.	Actual time, g.e.t.	Difference, sec
Orbital phase, hr:min:sec			
Height-adjust maneuver	01:34:37	01:34:36	-1
Phase-adjust maneuver	02:18:25	02:18:26	+1
Plane-change maneuver	02:45:47	02:45:53	+6
Vernier-height-adjust maneuver	03:03:41	03:03:42	+1
Coelliptic maneuver	03:47:35	03:48:10	+35
Terminal-phase-initiation maneuver	05:13:13	05:14:56	+103
82° corrective maneuver	(a)	05:27:26	(a)
33° corrective maneuver	(a)	05:39:20	(a)
Braking maneuver	05:45:36	05:43:09	-147
Docking	-	06:33:16	-
Rigidizing	-	06:33:22	-
90° yaw maneuver	-	[b]06:49:00	-
Thruster anomaly start	-	07:00:26.7	-
Undocking	-	07:15:12.3	-
Activate Reentry Control System	-	07:16:25.1	-
TCA no. 8 circuit breaker off	-	07:18:15.7	-
Rates under control	-	07:25:30	-
Reentry phase, hr:min:sec			
Adapter equipment section separation	10:03:47	10:03:48	+1
Retrofire initiation	10:04:47	10:04:47	0
Begin blackout	10:29:30	(c)	-
Guidance initiate	10:30:00	(c)	-
End blackout	10:34:43	(c)	-
Drogue parachute deployment	10:36:32	10:36:47	+15
Pilot parachute deploy/main parachute deploy	10:38:03	10:38:08	+5
Landing	10:42:02	10:41:26	-36
Parachute jettison	(a)	10:41:34	(a)

[a]Not applicable.

[b]Time is approximated because of missing data.

[c]Not available

TABLE 4.2-II.- GEMINI ATLAS-AGENA TARGET VEHICLE SEQUENCE OF EVENTS

Event	Planned time from lift-off	Actual time from lift-off	Difference, sec
Launch phase, sec			
Lift-off	0.00	0.00 (15:00:03.13) G.m.t.	0.00
Roll program start	2.00	2.05	+0.05
Roll program end	15.00	15.05	+0.05
Pitch program start	15.00	15.20	+0.20
Booster engine cutoff (BECO)	131.00	129.79	-1.21
Booster separation	134.00	132.75	-1.25
Primary sequencer (D-timer) start	277.38	282.08	+4.70
Sustainer engine cutoff (SECO)	279.96	283.68	+3.72
Vernier engine cutoff (VECO)	300.18	303.94	+3.76
TLV-GATV separation (retrorocket fire)	303.00	308.30	+5.30
Horizon sensor roll control start	305.50	310.70	+5.20
Pitchdown 90 deg/min start	338.38	343.00	+4.62
Pitchdown 90 deg/min stop	351.38	356.00	+4.62
3.99 deg/min orbit rate start	351.38	356.00	+4.62
SPS engine ignition	353.38	358.00	+4.62
PPS gas generator valve open	370.58	375.97	+5.39
PPS engine ignition (90-percent P_c)	371.88	377.50	+5.62
SPS engine cutoff	373.38	378.00	+4.62
Nose-shroud jettison squibs fired	381.38	386.71	+5.33
PPS engine cutoff (VMCO)	556.08	560.40	+4.32

TABLE 4.2-II.- GEMINI ATLAS-AGENA TARGET VEHICLE SEQUENCE OF EVENTS - Concluded

Event	Planned time from lift-off	Actual time from lift-off	Difference, sec
Orbital phase, hr:min:sec			
Height-adjust maneuver	21:42:47	21:42:47	0
Height-adjust maneuver	27:03:36	27:03:35	-1
Plane-change maneuver	39:16:25	39:16:26	+1
Minimum-impulse maneuver	44:01:25	44:01:23	-2
Plane-change maneuver	47:39:20	47:39:19	-1
Heignt-adjust maneuver	50:46:53	50:46:52	-1
Height-adjust maneuver	54:39:09	54:39:08	-1
Height-adjust maneuver	59:28:00	59:27:59	-1
Calibration maneuver	64:30:48	64:30:47	-1
Inclination-adjust maneuver	67:38:49	67:38:48	-1

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4.3 FLIGHT TRAJECTORIES

The launch and orbital trajectories referred to as planned are either preflight-calculated nominal trajectories (refs. 10 through 12) or trajectories based on nominal outputs from the Real-Time Computer Complex (RTCC) at the Mission Control Center-Houston (MCC-H) and planned attitudes and sequences as determined in real time in the Auxiliary Computer Room (ACR). The actual trajectories are based on the Manned Space Flight Network tracking data and actual attitude and sequences, as determined by airborne instrumentation. The Patrick Air Force Base atmosphere was used for altitudes below 25 nautical miles, and the 1959 ARDC model atmosphere was used for altitudes above 25 nautical miles for all trajectories except the actual launch phase. For the launch phase, the current atmosphere was used, as measured up to 25-nautical-miles altitude at the time of launch. The earth model for all trajectories contained geodetic and gravitational constants representing the Fischer ellipsoid. A ground track of the mission from Gemini Space Vehicle lift-off to retrofire and landing is shown in figure 4.3-1. Gemini Space Vehicle launch, orbit, rendezvous, and reentry trajectory curves are presented in figures 4.3-2 to 4.3-5. Gemini Atlas-Agena Target Vehicle (GAATV) launch and orbit curves are presented in figures 4.3-6 and 4.3-7.

4.3.1 Gemini Spacecraft

4.3.1.1 Launch.- The launch trajectory data shown in figure 4.3-2 are based on the real-time output of the Range-Safety Impact Prediction Computer (IP 3600) and the Guided Missile Computer Facility (GMCF). The IP 3600 used data from the Missile Trajectory Measurement System (MISTRAM), and FPS-16 radars. The GMCF used data from the GE Mod III radar. Data from these tracking facilities were used during the time periods listed in the following table:

Facility	Time from lift-off, sec
IP 3600 (FPS-16)	0 to 40
GMCF (GE Mod III)	40 to 383

The actual launch trajectory, as compared with the planned launch trajectory in figure 4.3-2, was slightly low in altitude, velocity, and flight-path angle during Stage I powered flight. At first-stage engine cutoff (BECO), the altitude, velocity, and flight-path angle were low by 2243 feet, 40 ft/sec, and 0.12 degrees, respectively. After BECO, the Radio Guidance System (RGS) corrected the errors accumulated during Stage I flight and guided Stage II to a near-nominal insertion. At second-stage engine cutoff (SECO), altitude and flight-path angle were low by 174 feet and 0.07 degrees, respectively, and velocity was high by 8 ft/sec. Actual SECO conditions are based on GE Mod III guidance radar data. At spacecraft separation, the actual altitude and flight-path angle were low by 1200 feet and 0.04 degrees, respectively, and velocity was high by 10 ft/sec. Table 4.3-I contains a comparison of planned and actual conditions at BECO, SECO, and spacecraft separation. The preliminary conditions at spacecraft separation were obtained by integrating the Antigua tracking station vector after insertion back to the time of separation as determined during the mission, through the planned velocity changes (ΔV's) and attitudes. The planned 10 ft/sec ΔV in reference 10 was changed prior to separation to a 5 ft/sec ΔV in order to reduce part of the 10 ft/sec overspeed. The final conditions were obtained by integrating the first-orbit best-estimate trajectory (BET) back through the actual ΔV's and attitudes to spacecraft separation as determined by telemetry. (NOTE: The BET used first-revolution tracking data from the Grand Bahama Island tracking station through Eglin Air Force Base.)

The GE Mod III and MISTRAM radar tracking data after SECO were used to compute a go—no-go for spacecraft insertion by averaging 10 seconds of data starting at SECO + 5 seconds. The go—no-go condition obtained from GE Mod III contained velocity and flight-path angle that were high by 7 ft/sec and low by 0.12 degrees, respectively, when compared to the more accurate orbital ephemeris data obtained later. The conditions obtained from MISTRAM showed velocity and flight-path angle to be high by 3 ft/sec and low by 0.09 degree, respectively, when compared to the later ephemeris data.

4.3.1.2 Orbit.- Because the main objective of the Gemini VIII mission was to rendezvous and dock with the Gemini Agena Target Vehicle (GATV), the orbit phase will be described in more detail in the rendezvous section, 4.3.1.2.1. Table 4.3-II and figure 4.3-3 show the planned and actual orbital elements after each maneuver and table 4.3-IV shows the orbital elements from insertion to retrofire. The planned elements shown in these tables were obtained from orbital ephemerides generated using the sequences in reference 10, and the actual elements were obtained by integrating the Gemini tracking network vectors after each mid-course and terminal-phase rendezvous maneuver.

Over the Indian Ocean, in the fifth revolution, the spacecraft and GATV began rolling while in the docked configuration, due to a short in the circuitry to the Orbital Attitude and Maneuver System (OAMS) no. 8 thruster. Shortly thereafter, the spacecraft was separated from the GATV and the spacecraft flight was terminated.

4.3.1.2.1 Rendezvous trajectory description: The planned trajectory as presented in table 4.3-III and figures 4.3-4(a) and 4.3-4(b) was taken from the real-time solution obtained using the Texas revolution 1 vector for the GATV and the Carnarvon revolution 1 vector for the spacecraft.

The ground-commanded maneuvers were determined from various Spacecraft 8 and GATV vectors as the plan was updated after each maneuver. The actual trajectory during the rendezvous phase was reconstructed utilizing BET anchor vectors (see reference 10). The Spacecraft 8 vector was determined prior to the first maneuver (Group A in reference 10). Maneuvers as derived from Inertial Guidance System (IGS) postflight analysis were applied as instantaneous changes in velocity until rendezvous. The GATV vector was a BET as given in reference 10, attachment 1. All perigee and apogee altitudes presented here are referenced to a spherical earth with Launch Complex 19 as the reference radius.

The ground computations, after Spacecraft 8 insertion, indicated a fairly nominal situation for effecting a fourth-orbit rendezvous. Because lift-off was on time, the only anomalies indicated were a very slight overspeed of about 2 ft/sec at spacecraft insertion and an out-of-plane condition requiring a plane change of about 26 ft/sec. At spacecraft insertion the range between the vehicles was nominal at about 1060 nautical miles; however, because of small dispersions in the GLV powered ascent, the spacecraft was about 3.5 nautical miles north of the target plane. In addition, a slight out-of-plane velocity error of about 5 ft/sec shifted the common nodal crossing to about 2 minutes from the nominal time.

At 1:34:37 spacecraft ground elapsed time (g.e.t.), a height adjustment (N_{H1}) was performed to correct the spacecraft insertion overspeed. This retrograde maneuver of 3.1 ft/sec with the forward-firing thrusters lowered the spacecraft apogee from about 147 to 145.5 nautical miles (13.5 to 14.7 nautical miles below the GATV orbit). The scheduled phase-adjust maneuver (N_{C1}) was performed at 2:18:26 g.e.t. near the second apogee. The horizontal, posigrade ΔV of 50.6 ft/sec was applied with the aft-firing thrusters. The resultant altitude at perigee was about 113.5 nautical miles.

The plane-change maneuver (N_{PC}) for placing the spacecraft into the target plane was performed at 2:45:53 g.e.t. The thrust of 26.6 ft/sec to the southeast (yaw = 90.6 degrees) was made with the aft-firing thrusters.

When the Carnarvon revolution 2 vector for the spacecraft was processed, following the N_{C1} maneuver, computations of the plan based on this vector indicated an unexpected change in the time of terminal phase initiation (TPI). Prior solutions had indicated a time for TPI of near the nominal 5:04:09 g.e.t., about 2 minutes after sunset on the spacecraft. The Carnarvon revolution 2 solution gave 5:00:52 g.e.t. and the Hawaii revolution 2 solution moved the time still further back to 4:56:11 g.e.t. This indicated that the N_{C1} and/or the N_{PC} maneuvers had not been accomplished accurately because the Carnarvon and Hawaii vectors were thought to be acceptable. In an attempt to move the TPI time toward the nominal and to achieve the planned differential altitude of 15 nautical miles between the spacecraft and GATV at apogee of the spacecraft orbit, the flight dynamics controllers scheduled a vernier height-adjust maneuver of 2 ft/sec to be applied at second perigee. This maneuver had to be performed before any further tracking from the California, White Sands, and Eglin Air Force Base stations could be processed. Therefore, at 3:03:42 g.e.t., the crew performed a posigrade maneuver of 2.3 ft/sec. Subsequent tracking data from the Grand Turk, Antigua, and Ascension stations proved that this maneuver should not have been performed because the Carnarvon revolution 2 vectors and the Hawaii revolution 2 vectors apparently had been unusually poor; thus, the terminal-phase-initiate time shifted forward to about 5:13:00 g.e.t. instead of the desired time of 05:04:00 g.e.t. The impact of this anomaly was that the lighting conditions for terminal phase were not as planned. However, because the onboard radar and computer systems were functioning properly, the lighting requirements were not essential. At 3:48:10 g.e.t., the coelliptic maneuver N_{SR} was accomplished. The crew performed this maneuver about 35 seconds late because of a problem in clearing the Incremental Velocity Indicators. This delay had no significant effect on the trajectory. The actual ΔV of 61.6 ft/sec was applied at a pitch-down attitude of 21.3 degrees and with the aft-firing thrusters. The resultant spacecraft orbit was about 146.5 by 144.6 nautical miles and the altitude differential (Δh) between the spacecraft and GATV orbits was about 13.5 nautical miles. Prior to TPI, the Δh varied from 13.5 to 14.7 nautical miles with a value of about 13.4 nautical miles at TPI.

The TPI maneuver was begun at 5:14:56 g.e.t. when the elevation angle to the GATV was approximately 26.8 degrees and the range was about 29 nautical miles. A total ΔV of 27.3 ft/sec was applied with the aft-firing thrusters at an effective pitch up of 31.3 degrees and yaw left of 16.8 degrees.

The intermediate corrections, at ωt = 81.8 degrees and 33.6 degrees, were applied 12 and 24 minutes later and required about 20 and 16 ft/sec ΔV, respectively.

The terminal-phase-finalize (TPF) maneuver was initiated at 5:43:09 g.e.t. and braking thrusts were applied intermittently over the next 10 minutes. An effective resultant velocity of about 41 ft/sec was added to the spacecraft orbit; however, because of the semi-optical approach technique, at least twice this amount of ΔV capability was expended in fuel. By 5:55:00 g.e.t., the spacecraft was about 150 feet from the GATV and the crew was station keeping.

The total translation cost of propellants for the terminal phase amounted to about the equivalent of 150 ft/sec change in velocity and, because of the fairly large intermediate corrections and braking maneuvers, this represented about 50 ft/sec more than the preflight nominal. The expected one-sigma additional cost for this type of translation maneuver had been predicted to be equivalent to about 75 ft/sec.

The total translational cost of the rendezvous maneuvers, including terminal phase, was 295 ft/sec, about 90 ft/sec greater than the preflight nominal, but under 10 ft/sec less than a one-sigma deviation.

4.3.1.3 Reentry.- The mission was terminated early with reentry during revolution 7 in the secondary landing area near the coast of China. The planned and actual reentry trajectories are shown in figure 4.3-5. The planned trajectory was determined by integrating the Ascension tracking station vector in revolution 7 through planned retrofire sequences determined by the RTCC and assuming a 55-degree contour-line bank-angle reentry according to Math Flow 7 (ref. 13). The actual trajectory was obtained by integrating the Ascension tracking station vector through the actual retrofire sequence and attitudes, as determined from telemetry records, to landing and applying the appropriate lift vectors determined from the roll-attitude angles recorded from the onboard guidance.

The landing point for this trajectory agrees with the landing point in the onboard computer at 50 000 feet (see section 5.1.5.2.3) and the peak g-loads agree with the telemetry data within 0.06g at analogous times. Blackout times were not available; however, the parachute deployment altitudes at recorded sequence times agree with those reported in section 5.1.11.

4.3.2 Gemini Atlas-Agena Target Vehicle

4.3.2.1 Launch.- The launch trajectory data presented in figure 4.3-6 are based on the real-time output of the Range-Safety Impact Prediction Computer (IP 3600) and the Bermuda tracking radar. Data from these tracking facilities were used during the time periods listed in the following table:

Facility	Time from lift-off, sec
IP 3600 (TPQ-18, FPQ-6, FPS-16)	0 to 317
IP 3600, BDA (TPQ-18, FPS-16)	317 to 418
IP 3600, BDA (FPS-16)	418 to 621

The actual launch trajectory, as compared with the planned trajectory in figure 4.3-6, was essentially nominal throughout the GAATV launch phase. The differences noted in table 4.3-V are not representative of errors or dispersions (see section 5.5.5) because the Target Launch Vehicle targets for coast-ellipse orbital elements rather than for a specific position and velocity. Table 4.3-VI presents the targeting parameters and osculating elements at GAATV vernier engine cutoff (VECO) and GATV insertion.

4.3.2.2 Orbit.- The GATV was placed into the desired orbit for the planned Gemini Space Vehicle launch and rendezvous (see section 4.3.1.2.1). Table 4.3-V contains a comparison of the planned and actual insertion conditions of the GATV. The preliminary conditions were obtained by integrating the Canary Island tracking station vector back to the GATV Primary Propulsion System (PPS) cutoff obtained in real time. The final conditions were obtained by integrating the Canary Island vector back to the PPS cutoff obtained from telemetry records.

In the fifth revolution, approximately 27 minutes after docking, the two vehicles began rolling. The spacecraft was separated from the GATV shortly thereafter, terminating the docked phase of the mission. Subsequently, the GATV was stabilized and placed in a parking orbit for possible use as a target during later missions. Table 4.3-VII contains the maneuvers performed by the GATV. Figure 4.3-7 shows the apogee and perigee altitudes, and table 4.3-VIII presents the orbital parameters before and after each maneuver. Table 4.3-IX contains the orbital parameters for every twelfth revolution after insertion until the GATV placed in the final parking orbit.

4.3.3 Gemini Launch Vehicle Second Stage

The second stage of the Gemini Launch Vehicle was inserted into an orbit with apogee and perigee altitudes of 146.5 and 86.3 nautical miles, respectively. The Gemini network tracking radars and the North American Air Defense Command (NORAD) network tracking sensors were able to skin-track the second stage during the ensuing 29-hour orbital lifetime. The Goddard Space Flight Center predicted reentry in revolution 20 with a predicted impact point of 6.24 degrees, north latitude, and 110.69 degrees, west longitude, in the Pacific Ocean.

TABLE 4.3-I.- COMPARISON OF PLANNED AND ACTUAL GEMINI SPACE VEHICLE TRAJECTORY PARAMETERS

Condition	Planned	Actual	
		Preliminary	Final
BECO			
Time from lift-off, sec	153.71	Not computed	154.62
Geodetic latitude, deg north	28.36	Not computed	28.36
Longitude, deg west	79.63	Not computed	79.61
Altitude, feet	211 136	Not computed	208 893
Altitude, n. mi.	34.7	Not computed	34.4
Range, n. mi.	49.8	Not computed	51.4
Space-fixed velocity, ft/sec	9960	Not computed	9920
Space-fixed flight-path angle, deg	19.47	Not computed	19.35
Space-fixed heading angle, deg east of North . . .	99.16	Not computed	98.47
SECO			
Time from lift-off, sec	335.59	Not computed	337.54
Geodetic latitude, deg north	27.06	Not computed	27.09
Longitude, deg west	72.13	Not computed	72.04
Altitude, feet	527 299	Not computed	527 125
Altitude, n. mi.	86.7	Not computed	86.7
Range, n. mi.	456	Not computed	462
Space-fixed velocity, ft/sec	25 647	Not computed	25 655
Space-fixed flight-path angle, deg	0.0	Not computed	-0.07
Space-fixed heading angle, deg east of North . . .	100.90	Not computed	101.56
Spacecraft separation			
Time from lift-off, sec	355.59	357.56	365.66
Geodetic latitude, deg north	26.76	26.81	26.70
Longitude, deg west	70.53	70.54	70.03
Altitude, feet	526 934	526 352	525 734
Altitude, n. mi.	86.6	86.6	86.5
Range, n. mi.	544.0	543.9	572.7

TABLE 4.3-I.- COMPARISON OF PLANNED AND ACTUAL GEMINI SPACE VEHICLE TRAJECTORY PARAMETERS - Concluded

Condition	Planned	Actual	
		Preliminary	Final
Spacecraft separation - concluded			
Space-fixed velocity, ft/sec	25 728	25 737	25 738
Space-fixed flight-path angle, deg	0.00	0.05	0.04
Space-fixed heading angle, deg east of North . . .	101.66	101.64	101.88
Maximum conditions			
Altitude, statute miles	188.2	185.6	185.5
Altitude, n. mi.	163.8	161.4	161.3
Space-fixed velocity, ft/sec	25 738	25 742	25 743
Earth-fixed velocity, ft/sec	24 370	24 375	24 377
Exit acceleration, g	7.4	7.4	7.4
Exit dynamic pressure, lb/ft^2	747	677	677
Reentry deceleration, g (ephemeris data)	5.06	5.34	5.34
Reentry deceleration, g (telemetry data)	N/A	N/A	5.41
Reentry dynamic pressure, lb/ft^2	340	359	359
Landing point			
Latitude, north	25°15'	[a]25°22'	[b]25°12'
Longitude, east	136°00'	[a]135°56'	[b]136°05'

[a]Pickup point reported by the recovery ship, U.S.S. Leonard F. Mason.

[b]IGS coordinates in the onboard computer.

TABLE 4.3-II.- COMPARISON OF SPACECRAFT ORBITAL ELEMENTS BEFORE AND AFTER MANEUVERS

Maneuver	Condition	Before maneuver			After maneuver		
		Planned	Actual		Planned	Actual	
			Preliminary (a)	Final (b)		Preliminary (a)	Final (b)
Height adjust (N_{H1})	Apogee, n. mi. . . .	145.5	146.4	146.7	145.8	145.5	144.7
	Perigee, n. mi. . . .	86.6	86.7	86.3	86.5	86.6	86.3
	Inclination, deg. . .	28.87	28.94	29.02	28.87	28.94	29.07
	Period, min	88.76	--	88.83	88.77	--	88.80
Phase adjust (N_{C1})	Apogee, n. mi. . . .	145.8	145.5	144.7	145.8	145.5	144.7
	Perigee, n. mi.	86.5	86.6	86.3	116.1	114.6	113.3
	Inclination, deg. . .	28.87	28.94	29.02	28.87	28.92	29.07
	Period, min	88.77	--	88.80	89.32	--	89.35
Plane change (N_{PC})	Apogee, n. mi. . . .	145.8	145.5	144.7	145.8	145.5	144.7
	Perigee, n. mi. . . .	116.1	114.6	113.3	116.1	114.6	113.3
	Inclination, deg. . .	28.87	28.92	29.02	28.87	28.92	29.02
	Period, min	89.32	--	89.35	89.32	--	89.35
Vernier height adjust (N_{H2})	Apogee, n. mi. . . .	145.8	145.5	144.7	145.8	145.5	144.8
	Perigee, n. mi. . . .	116.1	114.6	113.3	116.1	114.6	113.3
	Inclination, deg. . .	28.87	28.92	29.02	28.87	28.92	29.02
	Period, min	89.32	--	89.35	89.32	--	89.35

[a]Preliminary elements are RTCC values obtained during the mission. The altitude is measured above the Launch Complex 19 earth radius. Period was not available.

[b]The altitude is computed above the Fischer Ellipsoid.

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TABLE 4.3-II.- COMPARISON OF SPACECRAFT ORBITAL ELEMENTS BEFORE AND AFTER MANEUVERS - Concluded

Maneuver	Condition	Before maneuver			After maneuver		
		Planned	Actual		Planned	Actual	
			Preliminary (a)	Final (b)		Preliminary (a)	Final (b)
Coelliptical maneuver (N_{SR})	Apogee, n. mi. . . .	145.7	145.5	144.8	145.7	146.6	146.7
	Perigee, n. mi. . . .	116.1	114.6	113.3	144.1	146.6	143.9
	Inclination, deg. . .	28.87	28.92	29.02	28.87	28.89	29.02
	Period, min	89.89	--	89.35	89.89	--	90.02
Terminal phase initiate (TPI)	Apogee, n. mi. . . .	145.7	146.6	146.7	163.0	161.4	159.0
	Perigee, n. mi. . . .	144.1	146.6	143.9	145.0	146.2	145.9
	Inclination, deg. . .	28.87	28.89	29.02	28.87	28.89	29.02
	Period, min	89.89	--	90.02	90.18	--	90.14
Terminal phase finalize (TPF) (braking)	Apogee, n. mi. . . .	163.0	161.4	159.0	161.1	161.4	161.1
	Perigee, n. mi. . . .	145.0	146.2	145.9	158.9	159.8	158.6
	Inclination, deg. . .	28.87	28.89	29.02	28.87	28.89	29.02
	Period, min	90.18	--	90.14	90.44	--	90.55

[a] Preliminary elements are RTCC values obtained during the mission. The altitude is measured above the Launch Complex 19 earth radius. Period was not available.

[b] The altitude is computed above the Fischer Ellipsoid.

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TABLE 4.3-III.- SPACECRAFT RENDEZVOUS MANEUVERS

Maneuver	Planned	Ground commanded	Actual
Height adjust (N_{H1})			
G.e.t., hr:min:sec	1:34:37	1:34:37	1:34:36.4
ΔV, ft/sec	3.3	2.9	3.1
Pitch, deg	0.0	0.0	-3.4
Yaw, deg	0.0	0.0	2.0
Δt_B, sec	6.0	5.0	5.5
Phase adjust (N_{C1})			
G.e.t., hr:min:sec	2:18:26	2:18:25	2:18:25.8
ΔV, ft/sec	50.3	50.6	50.6
Pitch, deg	0.0	0.0	0.4
Yaw, deg	0.0	0.0	0.2
Δt_B, sec	68.0	68.0	68.0
Plane change (N_{PC})			
G.e.t., hr:min:sec	2:46:14	2:45:47	2:45:52.8
ΔV, ft/sec	27.0	26.2	26.6
Pitch, deg	0.0	0.0	0.6
Yaw, deg	90.0	90.0	90.6
Δt_B, sec	36.0	35.0	35.5

TABLE 4.3-III.- SPACECRAFT RENDEZVOUS MANEUVERS - Continued

Maneuver	Planned	Ground commanded	Actual
Vernier height adjust (N_{H2})			
G.e.t., hr:min:sec	Not scheduled	3:03:41	3:03:42.2
ΔV, ft/sec	Not scheduled	2.0	2.3
Pitch, deg	Not scheduled	0.0	-4.7
Yaw, deg	Not scheduled	0.0	1.6
Δt_B, sec	Not scheduled	3.0	3.0
Coelliptical (N_{SR})			
G.e.t., hr:min:sec	3:47:32	3:47:35	3:48:09.7
ΔV, ft/sec	59.7	61.2	61.6
Pitch, deg	-19.8	-21.4	-21.3
Yaw, deg	0.0	0.0	-0.1
Δt_B, sec	80.0	82.0	82.5
Terminal phase initiate (TPI)		(NOT USED)	
G.e.t., hr:min:sec	5:05:07	5:13:13	5:14:55.7
ΔV, ft/sec	32.0	32.6	27.3
Pitch, deg	27.0	29.0	31.3
Yaw, deg	-1.0	-11.3	-16.8
Δt_B, sec	42.0	43.0	37.0

TABLE 4.3-III.- SPACECRAFT RENDEZVOUS MANEUVERS - Concluded

Maneuver	Planned	Ground commanded	Actual
82 degree correction (COR_1)		(NOT SENT)	
G.e.t., hr:min:sec	N/A		5:27:26.0
ΔV, ft/sec	N/A		[a]20
Pitch, deg	N/A		[b]50
Yaw, deg	N/A		3
Δt_B, sec	N/A		20
33 degree correction (COR_2)		(NOT SENT)	
G.e.t., hr:min:sec	N/A		5:39:19.9
ΔV, ft/sec	N/A		[a]-16
Pitch, deg	N/A		[b]-81
Yaw, deg	N/A		-138
Δt_B, sec	N/A		-16
Terminal phase finalize (TPF)		(NOT SENT)	
G.e.t., hr:min:sec	5:37:48	5:45:36	[c]5:43:09
ΔV, ft/sec	42.6		[d]41.2
Pitch, deg	55.7		[e]-65
Yaw, deg	177.6		-171
Δt_B, sec	67		[c]-600

[a]Approximate total ΔV expended because maneuvers were made along all three body axes with separate thrusters.

[b]Approximate line-of-sight angles to target during corrections.

[c]Braking initiated at this time, lasted intermittently for about 10 minutes as command pilot made semi-optical approach.

[d]This is the resultant ΔV applied during the braking; however, the total ΔV expended during semi-optical approach was about twice this number.

[e]Approximate look angle to target at time of braking initiation.

TABLE 4.3-IV.- COMPARISON OF SPACECRAFT ORBITAL ELEMENTS

Revolution	Condition	Planned	Actual	
			Preliminary (a)	Final (b)
1 (Insertion)	Apogee, n. mi.	145.5	146.4	146.7
	Perigee, n. mi. . . .	86.6	86.7	86.3
	Inclination, deg . . .	28.87	28.94	29.07
	Period, min	88.76	-	88.83
4 (Before TPI)	Apogee, n. mi.	145.7	146.6	146.7
	Perigee, n. mi. . . .	144.1	146.6	143.9
	Inclination, deg . . .	28.87	28.89	29.02
	Period, min	89.89	-	90.02
4 (Post-rendezvous)	Apogee, n. mi.	161.1	161.4	161.1
	Perigee, n. mi. . . .	158.9	159.8	158.6
	Inclination, deg . . .	28.87	28.89	29.02
	Period, min	90.55	-	90.55
7 (Pre-retrofire)	Apogee, n. mi.	161.1	161.4	161.3
	Perigee, n. mi. . . .	158.9	159.7	157.5
	Inclination, deg . . .	28.82	28.89	29.02
	Period, min	90.44	-	90.55

[a]Preliminary elements are RTCC values obtained during the mission. The altitude is measured above the Launch Complex 19 earth radius. Period was not available.

[b]The altitude is computed above the Fischer Ellipsoid.

TABLE 4.3-V.- COMPARISON OF PLANNED AND ACTUAL GAATV TRAJECTORY PARAMETERS

Condition	Planned	Actual	Difference
BECO			
Time from lift-off, sec	130.00	129.79	-0.21
Geodetic latitude, deg north	28.56	28.55	-.01
Longitude, deg west	79.74	79.74	0.00
Altitude, feet	196 647	190 500	-6 147
Altitude, n. mi.	32.4	31.4	-1.0
Range, n. mi.	42.8	43.3	+0.5
Space-fixed velocity, ft/sec	9 811	9 715	-96
Space-fixed flight-path angle, deg	21.33	20.87	-0.46
Space-fixed heading angle, deg east of North	85.40	85.77	+0.37
SECO			
Time from lift-off, sec	279.96	283.68	+3.71
Geodetic latitude, deg north	28.90	28.88	-0.02
Longitude, deg west	74.64	74.47	-0.17
Altitude, feet	655 562	654 190	-1 372
Altitude, n. mi.	107.9	107.6	-0.3
Range, n. mi.	312.3	321.5	+9.2
Space-fixed velocity, ft/sec	17 637	17 630	-7
Space-fixed flight-path angle, deg	10.22	10.14	-0.08
Space-fixed heading angle, deg east of North	87.10	86.97	-0.13
VECO			
Time from lift-off, sec	300.18	303.94	+3.76
Geodetic latitude, deg north	28.95	28.92	-0.03
Longitude, deg west	73.67	73.60	-0.07
Altitude, feet	715 616	709 380	-6 236
Altitude, n. mi	117.8	116.7	-1.1
Range, n. mi.	363.7	368.2	+4.5
Space-fixed velocity, ft/sec	17 560	17 588	+28
Space-fixed flight-path angle, deg	9.20	9.31	+0.11
Space-fixed heading angle, deg east of North	87.61	87.66	+0.05

TABLE 4.3-V.- COMPARISON OF PLANNED AND ACTUAL GAATV TRAJECTORY PARAMETERS - Concluded

Condition	Planned	Actual	Difference
PPS start			
Time from lift-off, sec	371.88	377.50	+5.62
Geodetic latitude, deg North	29.02	29.00	-0.02
Longitude, deg West	70.24	70.16	-0.08
Altitude, feet	875 460	875 825	+365
Altitude, n. mi.	144.1	144.2	+0.1
Range, n. mi.	544.1	549.1	+5.0
Space-fixed velocity, ft/sec	17 287	17 297	+10
Space-fixed flight-path angle, deg	5.48	5.60	+0.12
Space-fixed heading angle, deg east of North	89.42	89.39	-0.03
Condition	Planned	Actual: Preliminary (a)	Actual: Final (b)
Insertion			
Time from lift-off, sec	556.28	558.00	560.40
Geodetic latitude, deg North	28.65	28.60	28.57
Longitude, deg West	59.64	59.65	59.37
Altitude, feet	980 432	980 359	980 141
Altitude, n. mi.	161.4	161.3	161.3
Range, n. mi.	1102.6	1104.6	1117.6
Space-fixed velocity, ft/sec	25 367	25 366	25 366
Space-fixed flight-path angle, deg	0.003	0.002	0.003
Space-fixed heading angle, deg east of North	94.86	95.13	95.27
Condition	Planned	Actual	Difference
Maximum Conditions			
Altitude, statute miles	254.2	466.1	466.1
Altitude, n. mi.	221.0	405.3	405.3
Space-fixed velocity, ft/sec	25 373	25 374	25 374
Earth-fixed velocity, ft/sec	23 988	23 988	23 988
Exit acceleration, g	6.3	N/A	6.0
Exit dynamic pressure, lb/ft^2	946	N/A	884

[a]Preliminary elements are RTCC values obtained during the mission. The altitude is measured above the Launch Complex 19 earth radius. Period was not available.

[b]The altitude is computed above the Fischer Ellipsoid.

TABLE 4.3-VI.- COMPARISON OF PLANNED AND ACTUAL GAATV CUTOFF PARAMETERS

Condition	Planned	Actual	Difference
VECO Targeting Parameters			
Semi-major axis, n. mi.	2330.7	2332.1	+1.4
Eccentricity	0.5436	0.5427	-0.0009
Inclination, deg	28.87	28.85	-0.02
Inertial ascent node, deg	68.15	68.16	+0.01
VECO Osculating Elements			
Apogee altitude, n. mi.	158.1	158.2	+0.1
Perigee altitude, n. mi.	-2376.9	-2374.1	-2.8
Period, min	47.07	47.12	+0.05
Inclination, deg	28.87	28.85	-0.02
True Anomaly, deg	172.09	171.96	-0.13
Argument of perigee, deg	-86.43	-86.22	-0.21
Latitude of perigee, deg south	29.34	29.31	-0.03
Longitude of perigee, deg east	108.73	108.93	+0.20
Latitude of apogee, deg north	28.96	28.93	-0.03
Longitude of apogee, deg west	77.17	76.98	-0.19
Insertion Osculating Elements			
Semi-major axis, n. mi.	3603.3	3603.0	-0.3
Eccentricity	0.0007	0.0006	-0.0001
Inclination, deg	28.87	28.86	-0.01
Inertial ascent node, deg	68.20	67.63	-0.57
Apogee altitude, n. mi.	166.18	165.55	-0.63
Perigee altitude, n. mi.	161.40	161.36	-0.04
Period, min	90.49	90.47	-0.02
True anomaly	4.34	4.44	+0.10
Argument of perigee, deg	94.52	95.21	+0.69
Latitude of perigee, deg north	28.93	28.89	-0.04
Longitude of perigee, deg west	86.99	86.81	-0.17
Latitude of apogee, deg south	28.93	28.89	-0.04
Longitude of apogee, deg east	81.66	81.85	0.19

TABLE 4.3-VII.- GATV MANEUVERS

Condition	Ground commanded	Actual[a]	Actual[b]
Height-adjust maneuvers			
Engine	PPSC	PPSC	-
Maneuver initiate, g.e.t., hr:min:sec	21:42:47	21:42:47	-
ΔT burn, SPS mode C, sec	70.0	70.2	-
Δt burn, PPS, sec	2.2	1.2	-
ΔV, ft/sec	104.4	103.7	104
Pitch, deg	0	-	-5.1
Yaw, deg	0	-	4.5
Height-adjust maneuvers			
Engine	PPSC	PPSC	-
Maneuver initiate, g.e.t., hr:min:sec	27:03:36	27:03:35	-
ΔT burn, SPS mode C, sec	70.0	70.1	-
Δt burn, PPS, sec	2.0	1.1	-
ΔV, ft/sec	104.0	106.7	105
Pitch, deg	0.0	-	-5.1
Yaw, deg	0.0	-	4.8
Plane-change maneuvers			
Engine	PPSA	PPSA	-
Maneuver initiate, g.e.t., hr:min:sec	39:16:25	29:16:26	-
ΔT burns, SPS mode A, sec	22.0	22.0	-
Δt burn, PPS, sec	19.6	19.3	-
ΔV, ft/sec	1600	1601.1	1628
Pitch, deg	0.[illegible]	-	-1.2
Yaw, deg	-91.8	-	-84.9
Minimum-impulse maneuvers			
Engine	PPSC	PPSC	-
Maneuver initiate, g.e.t., hr:min:sec	44:01:25	44:01.23	-
ΔT burn, SPS mode C, sec	70.0	70.0	-
Δt burn, PPS, sec	1.0	0.8	-
ΔV, ft/sec	96	96	96
Pitch, deg	0.0	-	3.7
Yaw, deg	180	-	-175.1

[a]Based on telemetry data.

[b]Based on radar tracking data.

TABLE 4.3-VII.- GATV MANEUVERS - Continued

Condition	Ground commanded	Actual[a]	Actual[b]
Plane-change maneuver			
Engine	PPSA	PPSA	-
Maneuver initiate, g.e.t., hr:min:sec	47:39:20	47:39:19	-
ΔT burn, SPS mode A, sec	22.0	22.0	-
Δt burn, PPS, sec	7.4	8.1	-
ΔV, ft/sec	789	791.1	778
Pitch, deg	0	-	-2.9
Yaw, deg	-90.9	-	-73.1
Height-adjust maneuver			
Engine	PPSA	PPSA	
Maneuver initiate, g.e.t., hr:min:sec	50:46:53	50:46:52	
ΔT burn, SPS mode A, sec	22.0	22.0	
Δt burn, PPS, sec	2.5	2.5	
ΔV, ft/sec	272	272	273
Pitch, deg	0	-	-0.2
Yaw, deg	180	-	-171.8
Height-adjust maneuver			
Engine	PPSA	PPSA	-
Maneuver initiate, g.e.t., hr:min:sec	54:39:09	54:39:08	-
ΔT burn, SPS mode A, sec	22.0	22.0	-
Δt burn, PPS, sec	2.2	2.2	-
ΔV, ft/sec	247.7	247.7	248
Pitch, deg	0	-	-3.7
Yaw, deg	0	-	6.7
Height-adjust maneuver			-
Engine	PPSA	PPSA	-
Maneuver initiate, g.e.t., hr:min:sec	59:28:00	59:27:59	-
ΔT burn, SPS mode A, sec	22.0	22.1	-
ΔV, ft/sec	309.1	309.1	310
Pitch, deg	0	-	1.2
Yaw, deg	180	-	-172.5

[a]Based on telemetry data.

[b]Based on radar tracking data.

TABLE 4.3-VII.- GATV MANEUVERS - Concluded

Condition	Ground commanded	Actual[a]	Actual[b]
Calibration maneuver			
Engine	SPS2	SPS2	-
Maneuver initiate, g.e.t., hr:min:sec	64:30:48	64:30:47	-
Δt burn, sec	20	21	-
ΔV, ft/sec	63	-	57
Pitch, deg	0	-	0.5
Yaw, deg	90	-	89.4
Inclination-adjust maneuver			
Engine	SPS2	SPS2	-
Maneuver initiate, g.e.t., hr:min:sec	67:38:46	67:38:48	-
Δt burn, sec	48	51	-
ΔV, ft/sec	152.7	-	145
Pitch, deg	0	-	0.1
Yaw, deg	90	-	91.7

[a]Based on telemetry data.

[b]Based on radar tracking data.

TABLE 4.3-VIII.- COMPARISON OF GATV ORBITAL ELEMENTS FOR MANEUVER

Maneuver	Condition	Before maneuver		After maneuver	
		Actual		Actual	
		Preliminary (a)	Final (b)	Preliminary (a)	Final (b)
Height adjust	Apogee, n. mi.	161.4	161.0	219.8	218.3
	Perigee, n. mi. . . .	159.8	157.5	159.9	160.0
	Inclination, deg . . .	28.89	29.02	28.88	29.02
	Period, min	-	90.53	-	91.56
Height adjust	Apogee, n. mi.	219.8	218.3	219.9	219.8
	Perigee, n. mi. . . .	159.9	160.0	219.7	217.7
	Inclination, deg . . .	28.88	29.02	28.89	29.02
	Period, min	-	91.56	-	92.79
Plane change	Apogee, n. mi.	219.9	219.8	336.0	336.7
	Perigee, n. mi. . . .	219.7	217.5	219.8	221.1
	Inclination, deg . . .	28.89	29.02	30.68	30.78
	Period, min	-	92.79	-	94.94
Height adjust (minimum impulse)	Apogee, n. mi.	336.0	335.9	278.9	278.7
	Perigee, n. mi. . . .	219.8	221.1	219.8	219.7
	Inclination, deg . . .	30.68	30.78	30.68	30.78
	Period, min	-	99.94	-	93.83
Plane change	Apogee, n. mi.	278.9	278.7	383.8	381.2
	Perigee, n. mi. . . .	219.8	219.7	257.6	255.5
	Inclination, deg . . .	30.68	30.78	28.97	29.13
	Period, min	-	93.83	-	96.63
Height adjust	Apogee, n. mi.	383.8	381.2	258.0	256.0
	Perigee, n. mi. . . .	257.6	255.5	219.2	217.0
	Inclination, deg . . .	28.97	29.13	28.93	29.06
	Period, min	-	96.63	-	93.48
Height adjust	Apogee, n. mi.	258.0	256.0	406.6	405.3
	Perigee, n. mi. . . .	219.2	217.0	221.4	218.7
	Inclination, deg . . .	28.93	29.06	28.84	29.04
	Period, min	-	93.48	-	96.36

[a]Preliminary elements are RTCC values obtained during the mission. The altitude is measured above the Launch Complex 19 earth radius. Period was not available.

[b]The altitude is computed above the Fischer Ellipsoid.

TABLE 4.3-VIII.- COMPARISON OF GATV ORBITAL ELEMENTS FOR MANEUVERS - Concluded

Maneuver	Condition	Before maneuver		After maneuver	
		Actual		Actual	
		Preliminary (a)	Final (b)	Preliminary (a)	Final (b)
Height adjust	Apogee, n. mi.	406.6	405.3	223.1	220.2
	Perigee, n. mi. . . .	221.4	218.7	220.0	218.3
	Inclination, deg . . .	28.84	29.04	28.89	29.06
	Period, min	-	96.36	-	92.84
Calibration burn	Apogee, n. mi.	223.1	220.2	224.0	223.1
	Perigee, n. mi. . . .	220.0	218.3	219.9	218.5
	Inclination, deg . . .	28.89	29.06	28.90	29.03
	Period, min	-	92.84	-	92.86
Inclination adjust	Apogee, n. mi.	224.0	223.1	221.9	220.4
	Perigee, n. mi. . . .	219.9	218.5	219.9	217.6
	Inclination, deg . . .	28.90	29.03	28.90	29.03
	Period, min	-	92.86	-	92.82

[a]Preliminary elements are RTCC values obtained during the mission. The altitude is measured above the Launch Complex 19 earth radius. Period was not available.

[b]The altitude is computed above the Fischer Ellipsoid.

TABLE 4.3-IX.- COMPARISON OF GATV ORBITAL ELEMENTS

Revolution	Condition	Planned	Actual	
			Preliminary (a)	Final (b)
1 (Insertion)	Apogee, n. mi.	161.4	161.4	161.3
	Perigee, n. mi.	158.9	159.9	158.8
	Inclination, deg	28.87	28.89	29.02
	Period, min	90.44	-	90.57
16	Apogee, n. mi.	161.1	161.4	161.0
	Perigee, n. mi.	158.4	159.8	157.5
	Inclination, deg	28.87	28.89	29.02
	Period, min	90.43	-	90.53
24	Apogee, n. mi.	160.9	219.9	219.8
	Perigee, n. mi.	158.2	219.7	217.5
	Inclination, deg	28.87	28.89	29.02
	Period, min	90.43	-	92.79
36	Apogee, n. mi.	160.7	406.6	405.3
	Perigee, n. mi.	158.0	221.4	218.7
	Inclination, deg	28.87	28.84	29.04
	Period, min	90.42	-	96.36
48	Apogee, n. mi.	N/A	221.9	220.4
	Perigee, n. mi.		219.9	217.6
	Inclination, deg		28.90	29.03
	Period, min		-	92.82

[a]Preliminary elements are RTCC values obtained during the mission. The altitude is measured above the Launch Complex 19 earth radius. Period was not available.

[b]The altitude is computed above the Fischer Ellipsoid.

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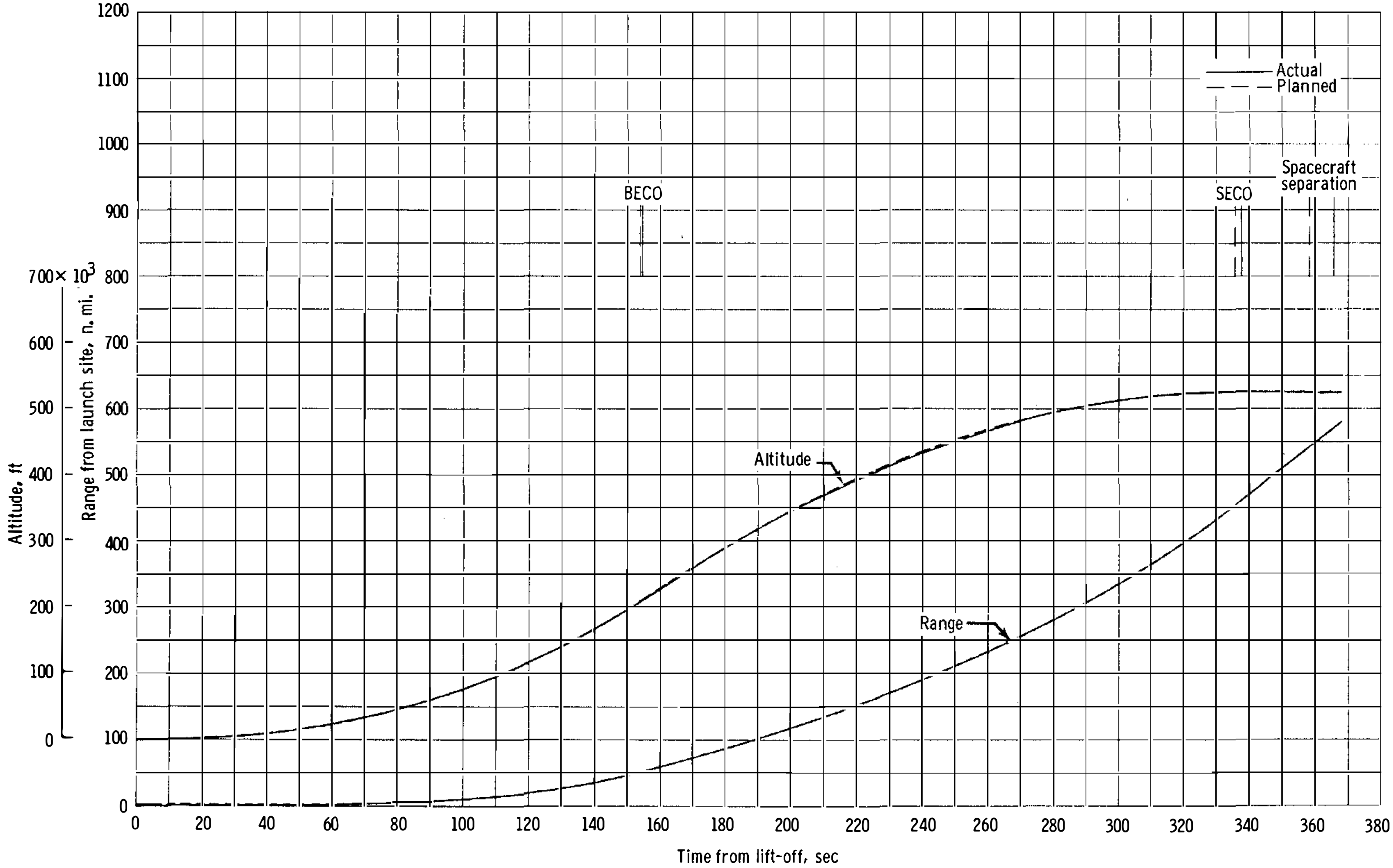
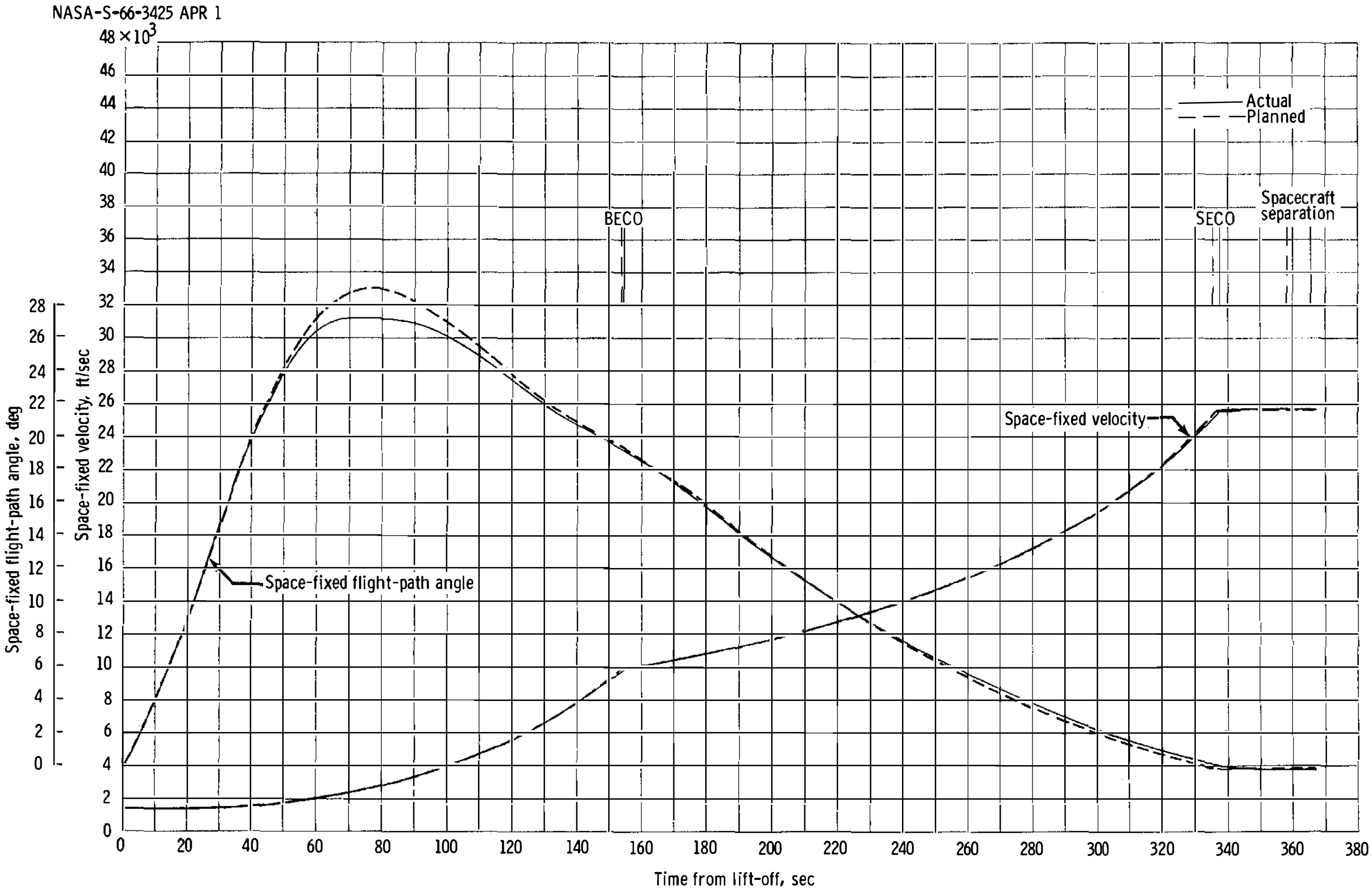


(a) Altitude and range.

Figure 4.3-2. - Trajectory paramenters for GLV-spacecraft launch phase.



(b) Space-fixed velocity and flight-path angle.

Figure 4.3-2. - Continued.

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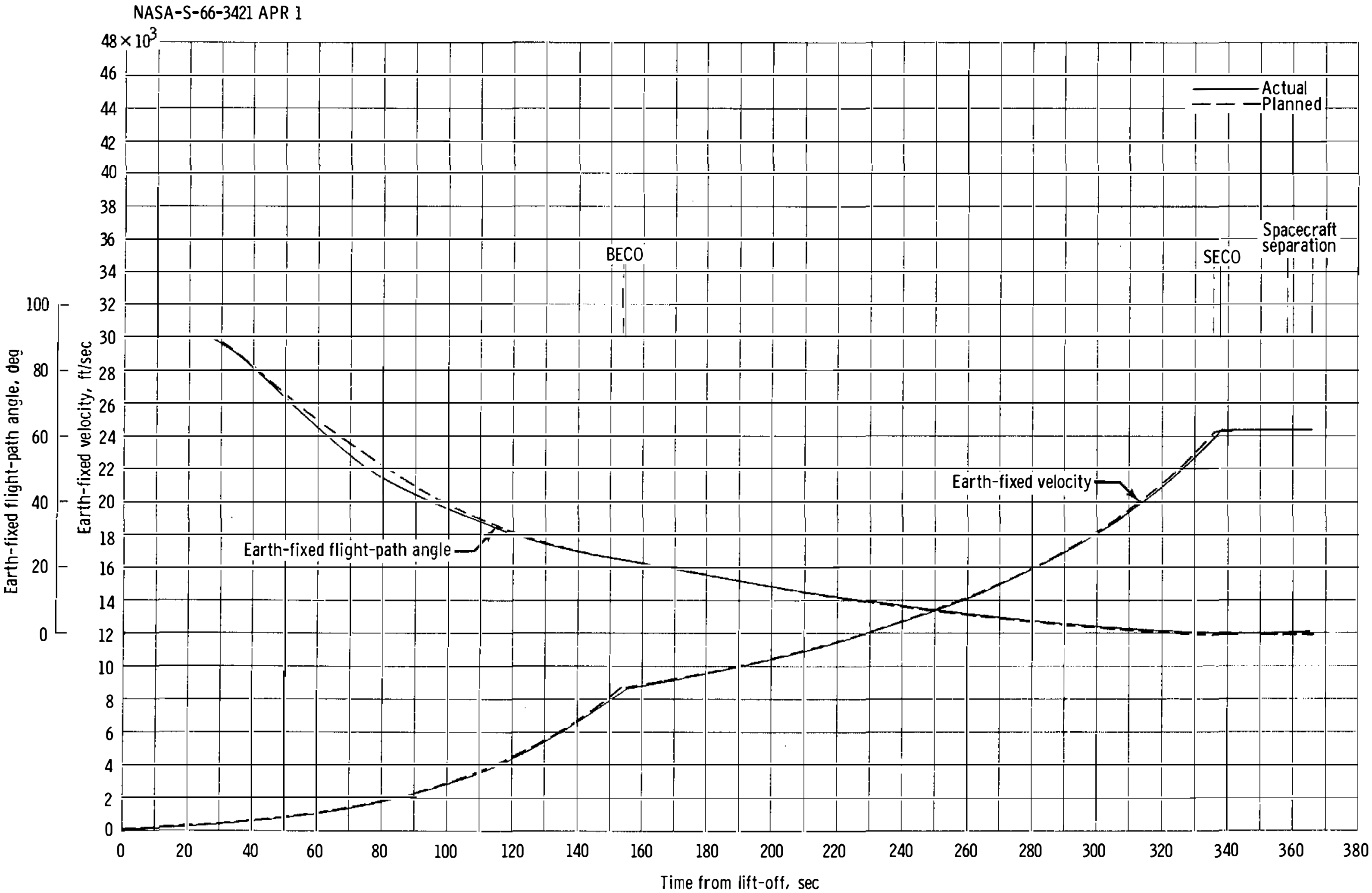

(c) Earth-fixed velocity and flight-path angle.

Figure 4.3-2.- Continued.

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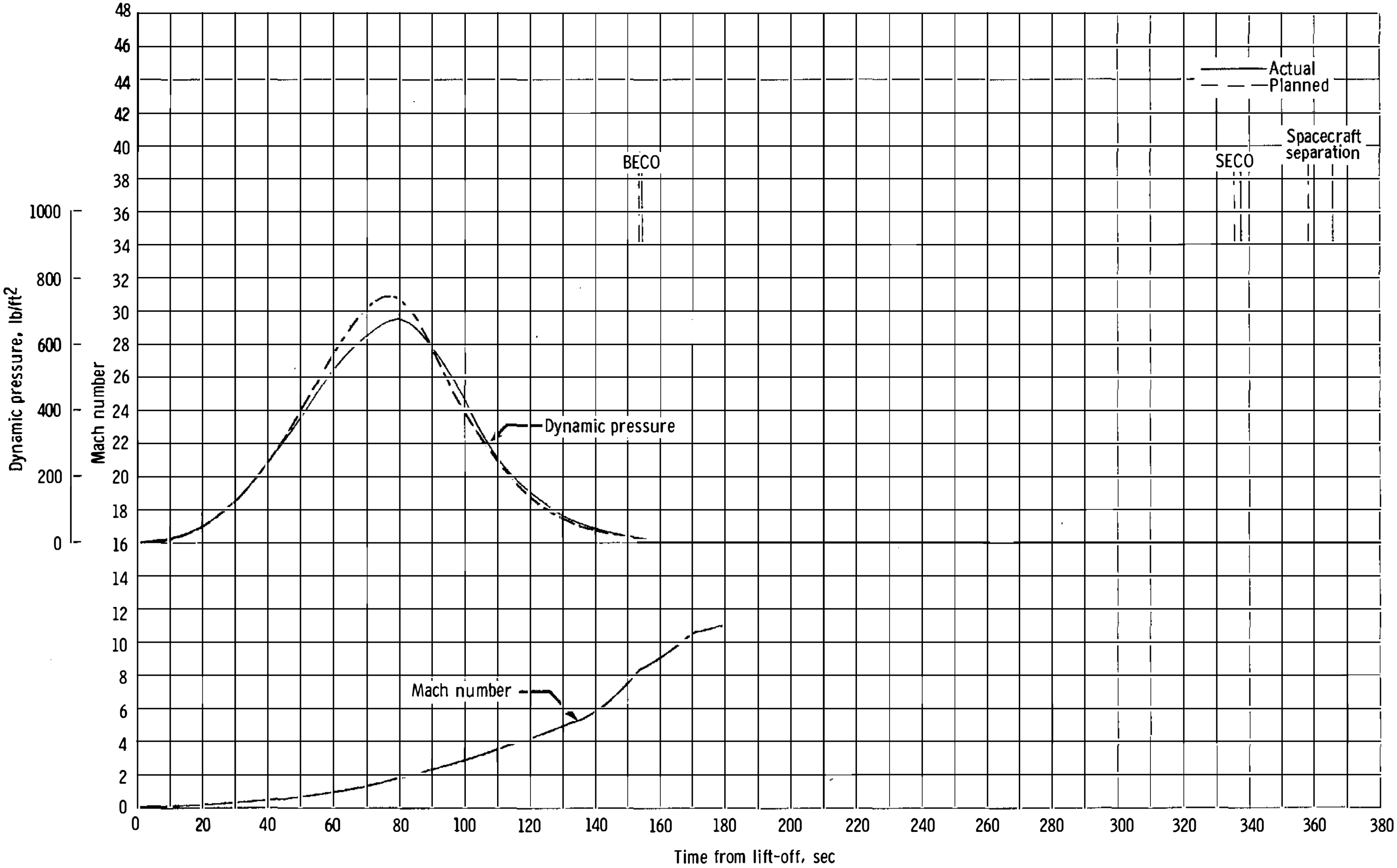

(d) Dynamic pressure and Mach number.

Figure 4.3-2. - Continued.

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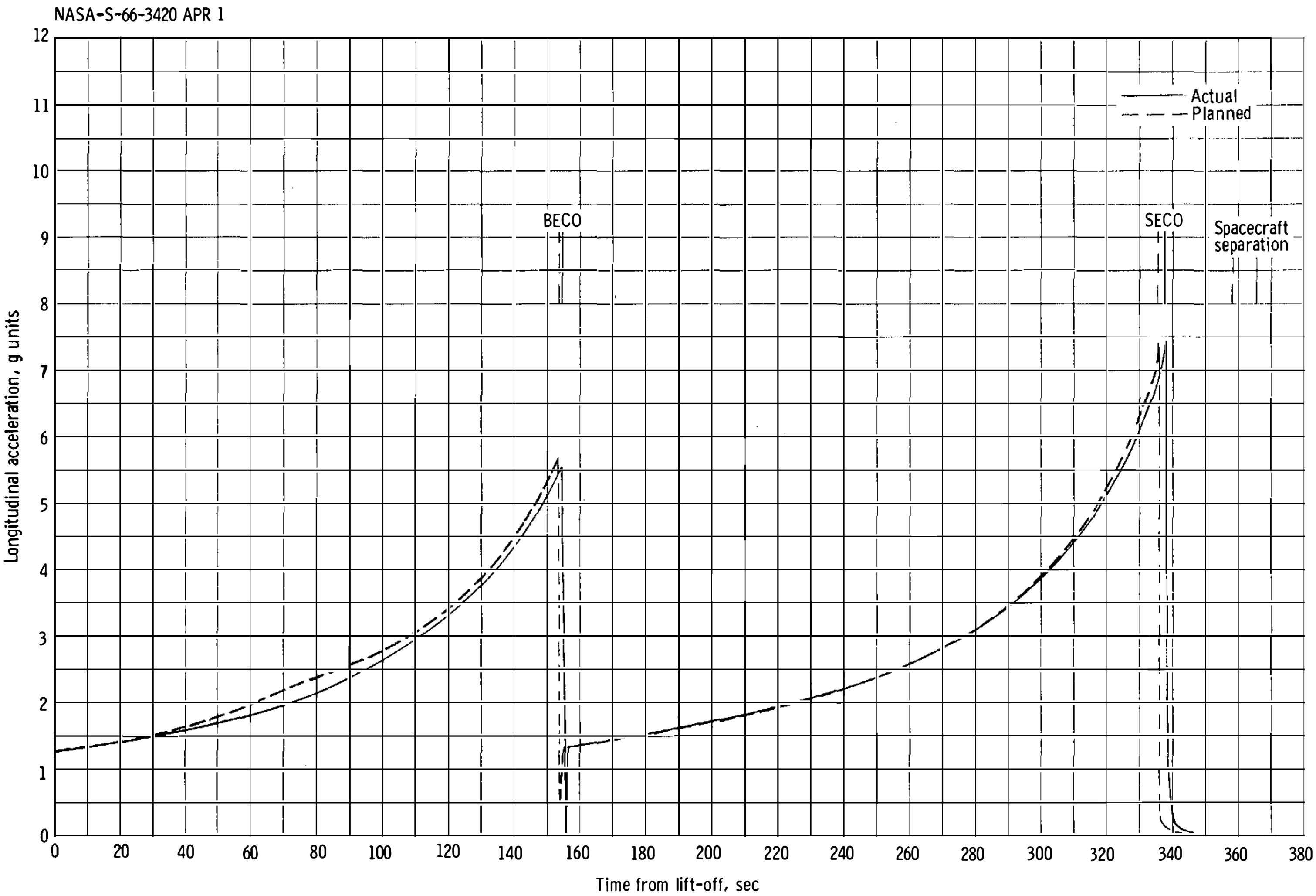

(e) Longitudinal acceleration.

Figure 4.3-2.- Concluded.

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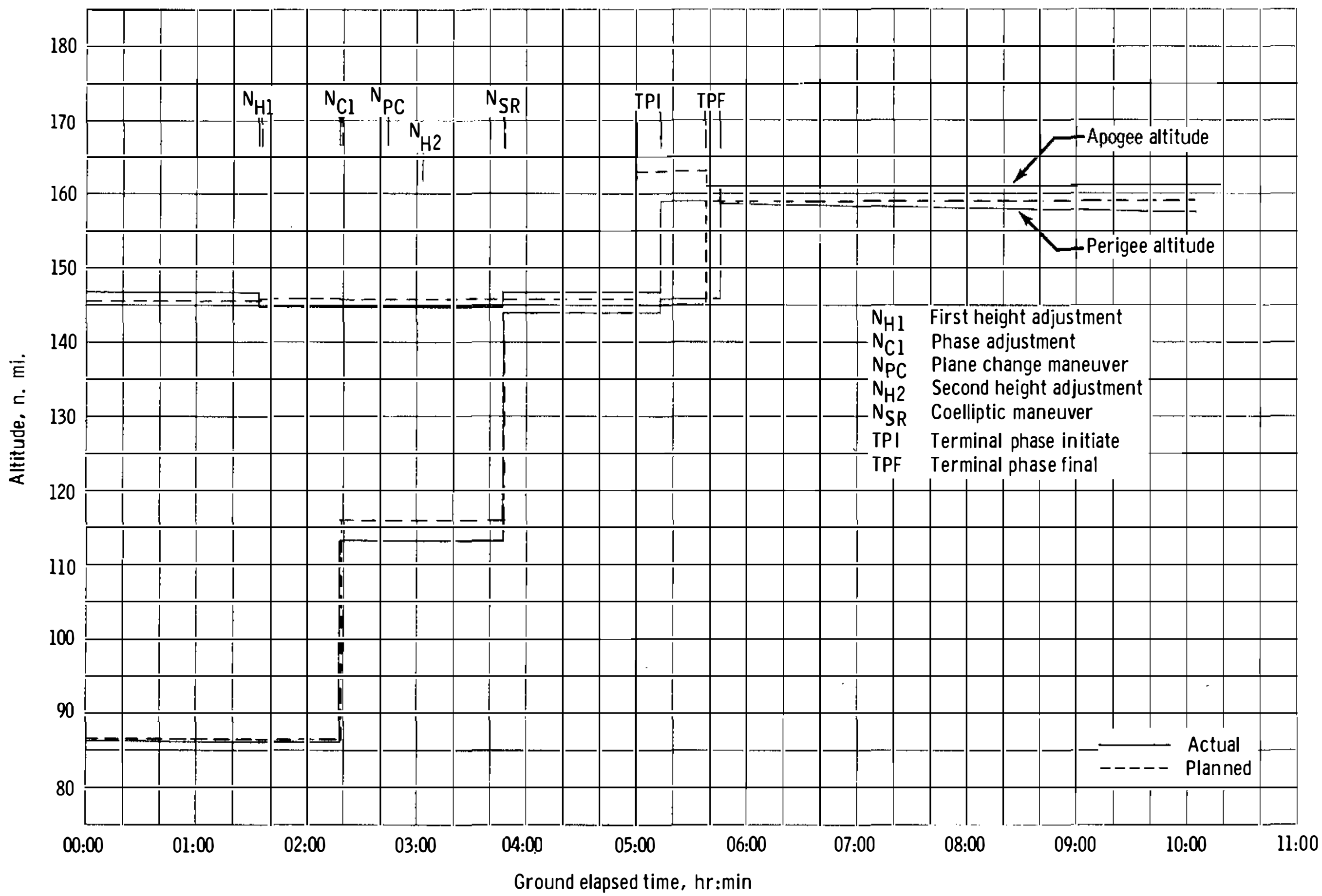

Figure 4.3-3. - Apogee and perigee altitude for the Gemini VIII mission.

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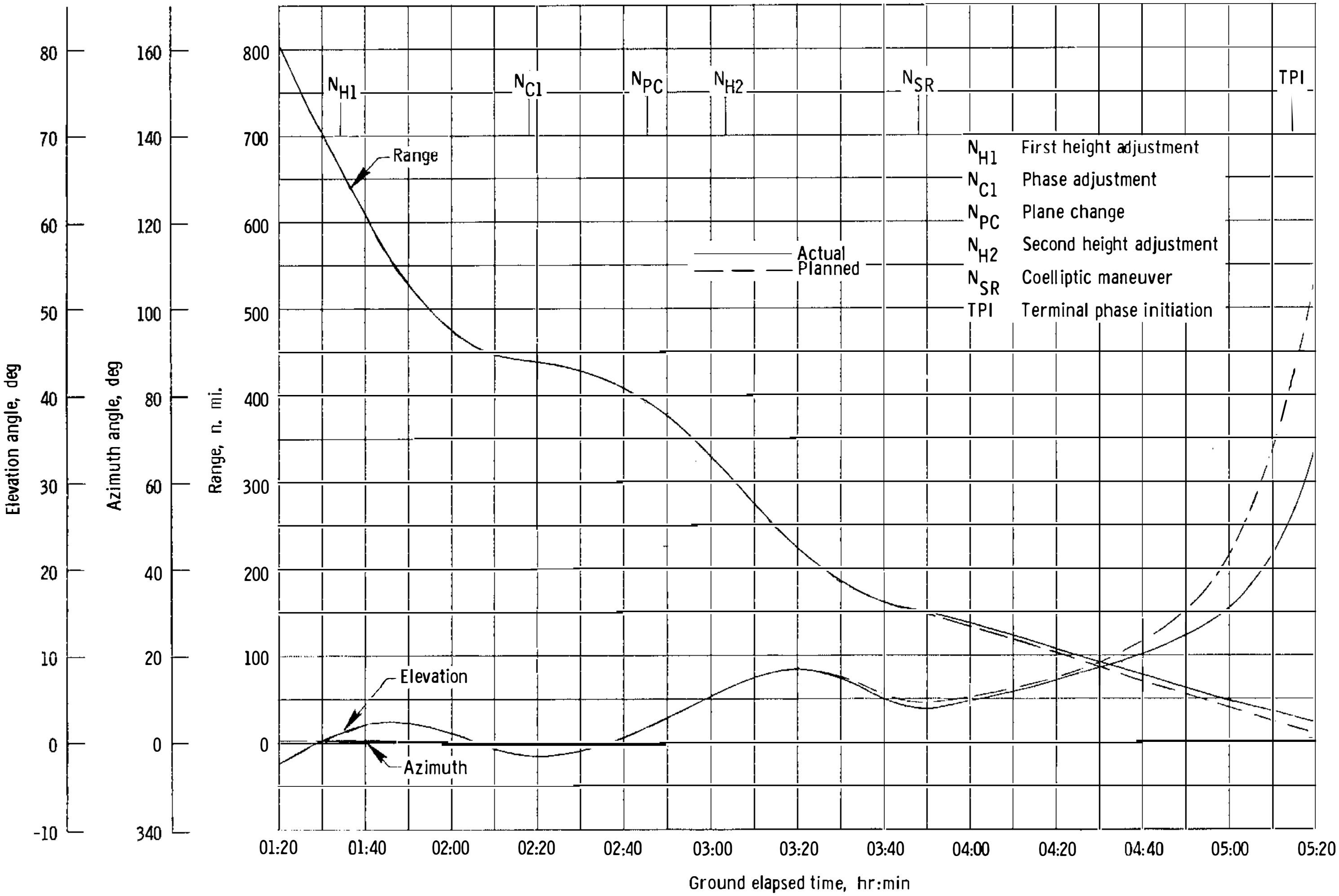

(a) Relative range, azimuth, and elevation from spacecraft 8 to GATV during midcourse maneuvers.

Figure 4.3-4. - Rendezvous during the Gemini VIII mission.

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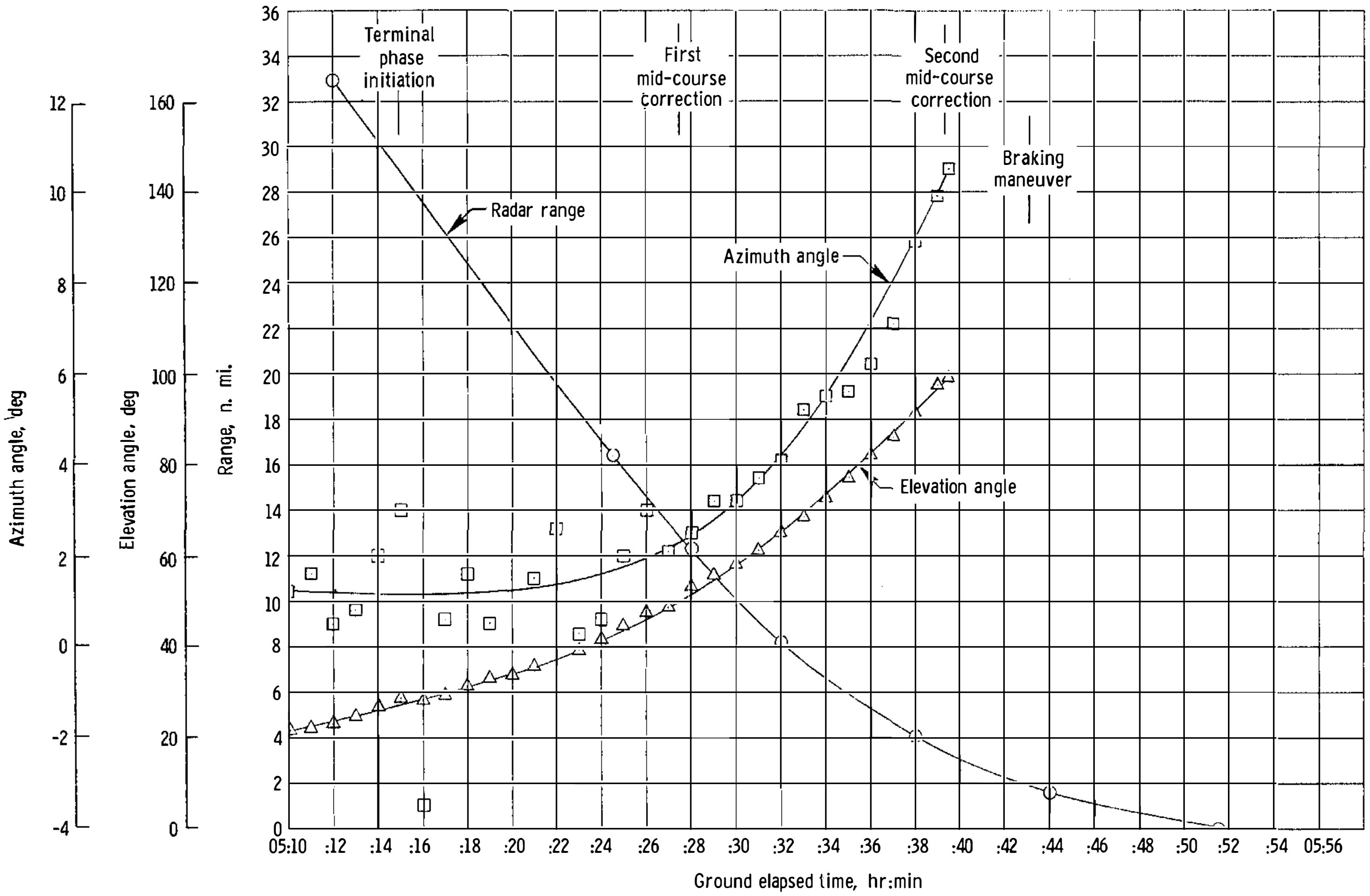
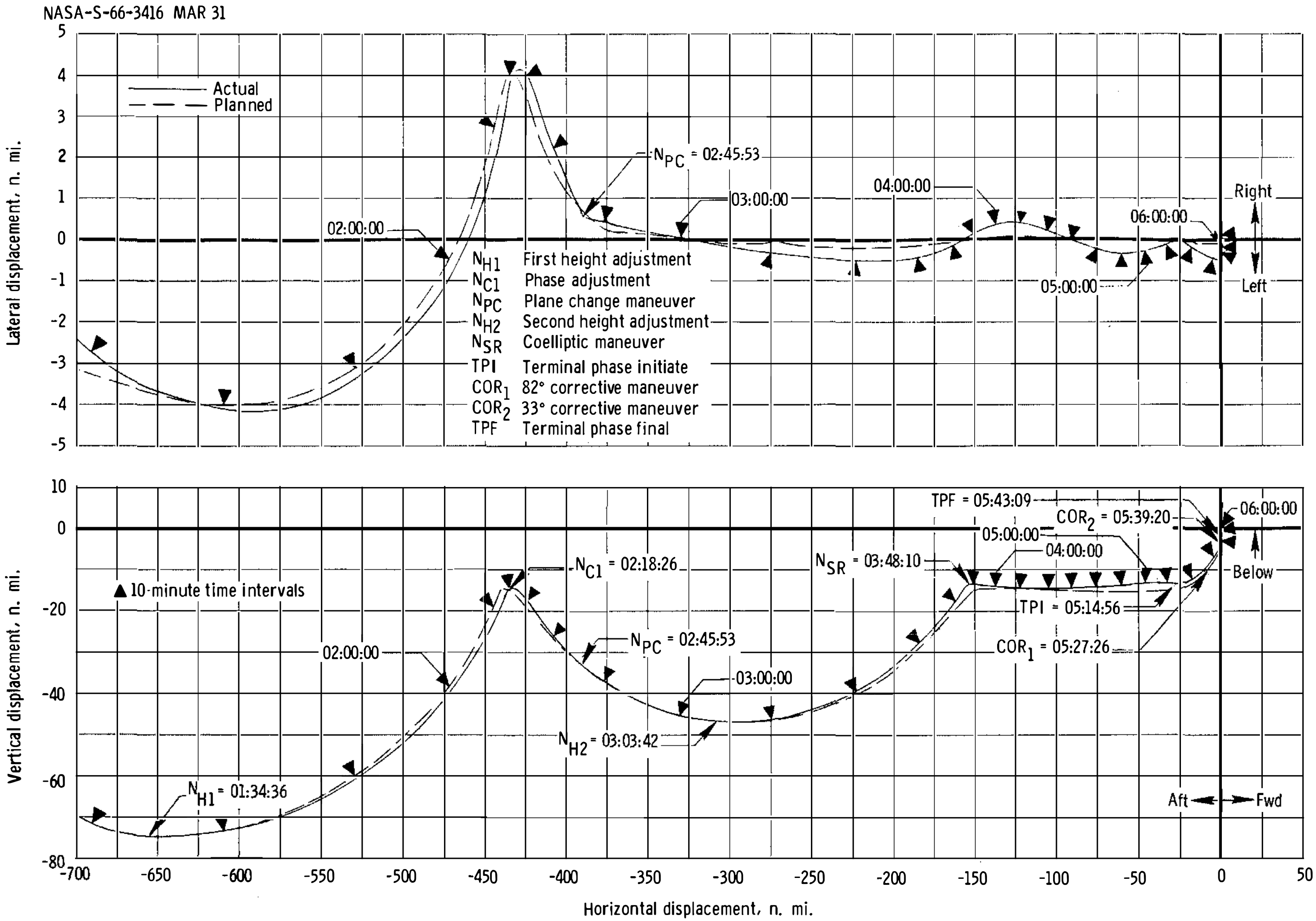


(b) Relative range, azimuth, and elevation from spacecraft 8 to GATV during terminal phase maneuvers.

Figure 4.3-4. - Continued.



(c) Relative trajectory profile, measured from GATV to spacecraft in curvilinear coordinate system.

Figure 4.3-4. - Concluded.

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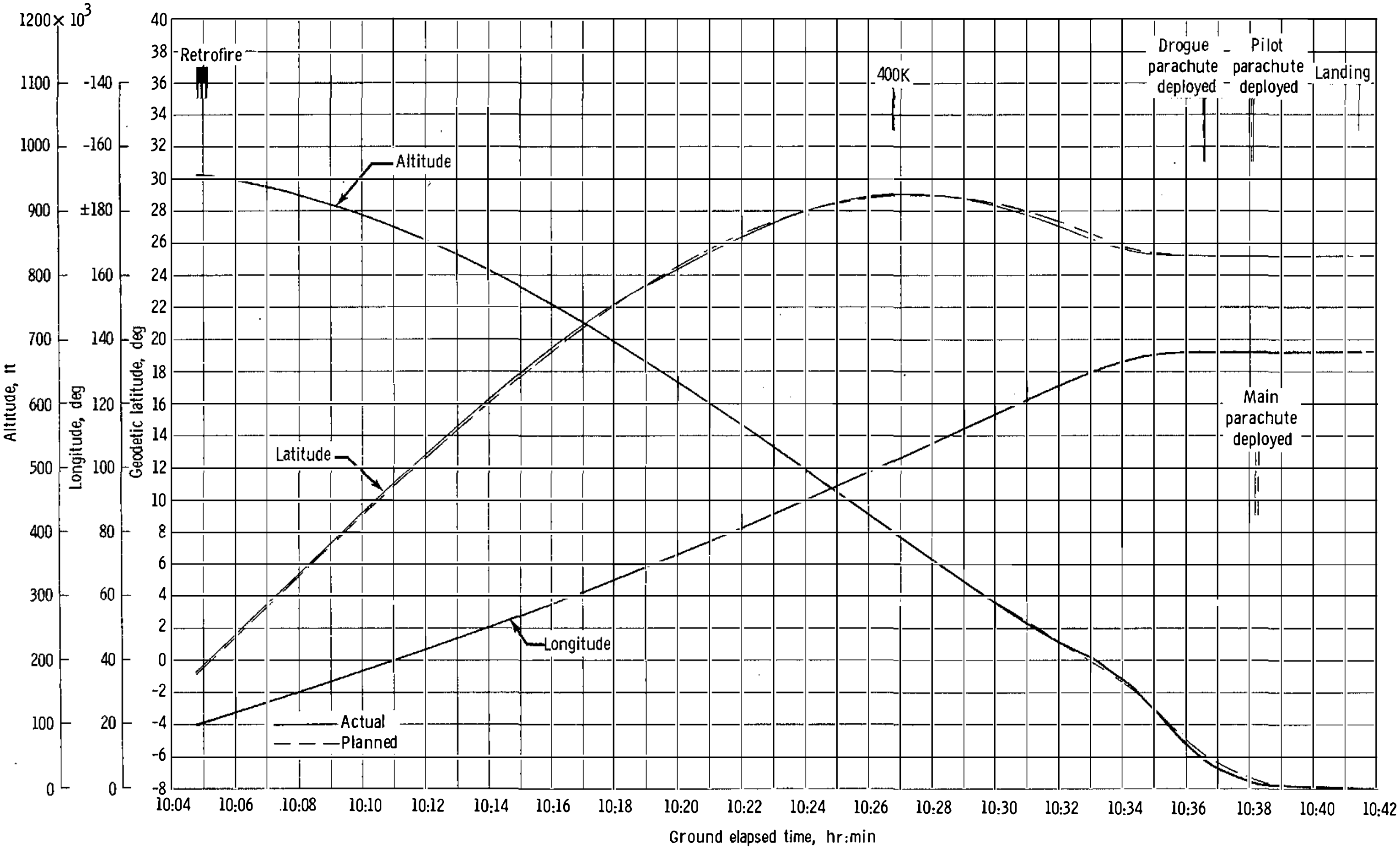

(a) Latitude, longitude, and altitude.

Figure 4.3-5. - Trajectory parameters for the Gemini VIII mission reentry phase.

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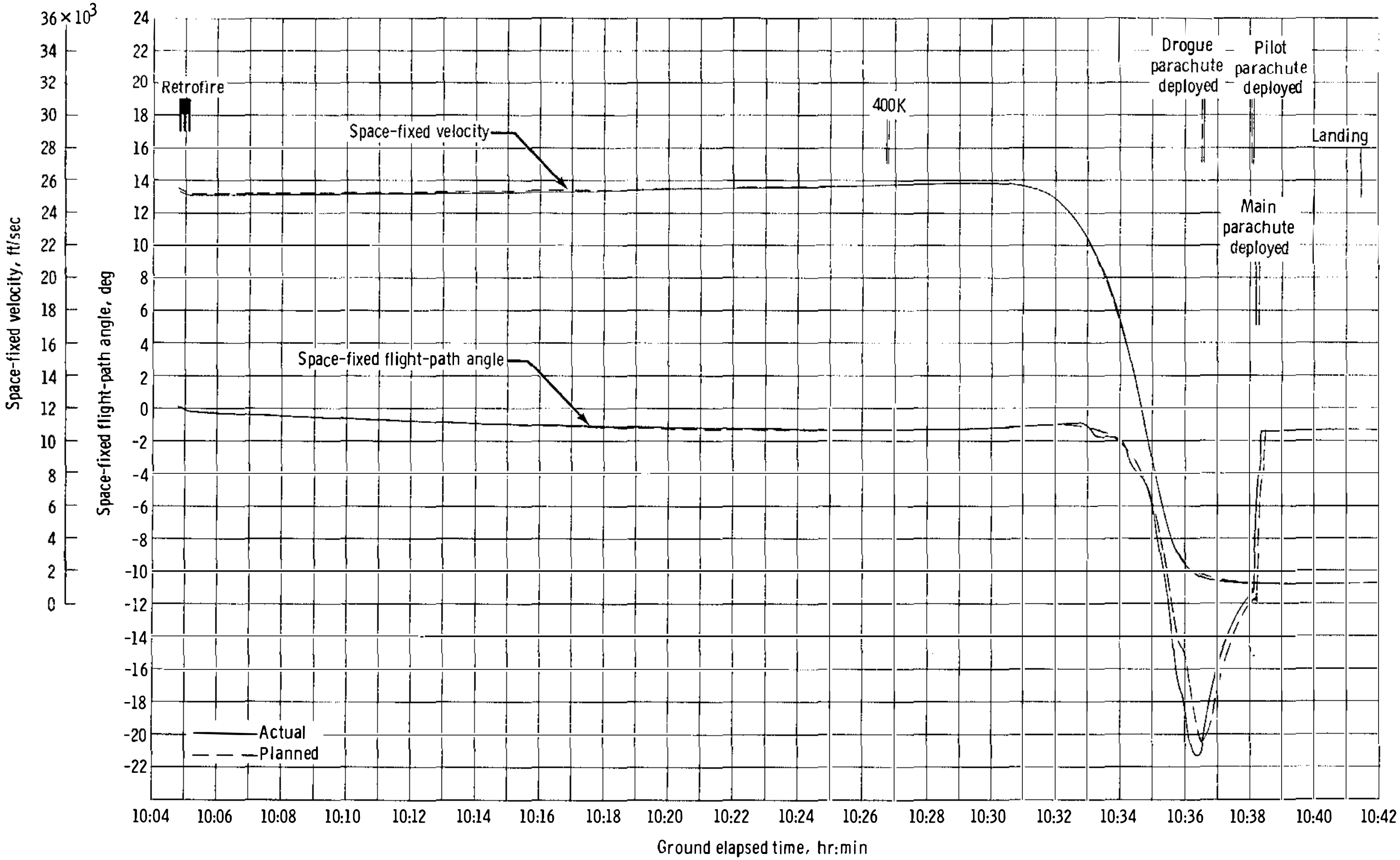

(b) Space-fixed velocity and flight-path angle.

Figure 4.3-5. - Continued.

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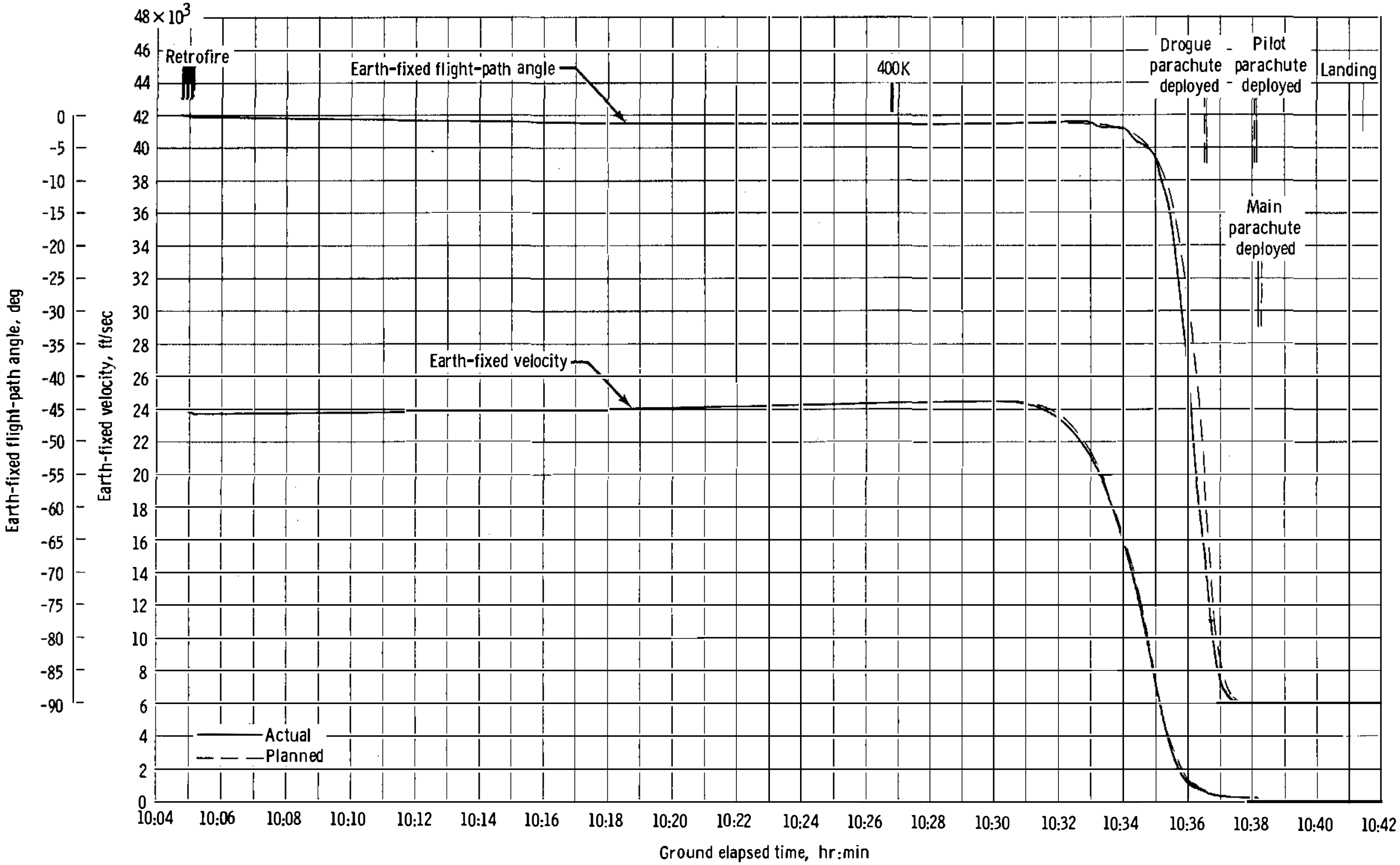

(c) Earth-fixed velocity and flight-path angle.

Figure 4.3-5. - Continued.

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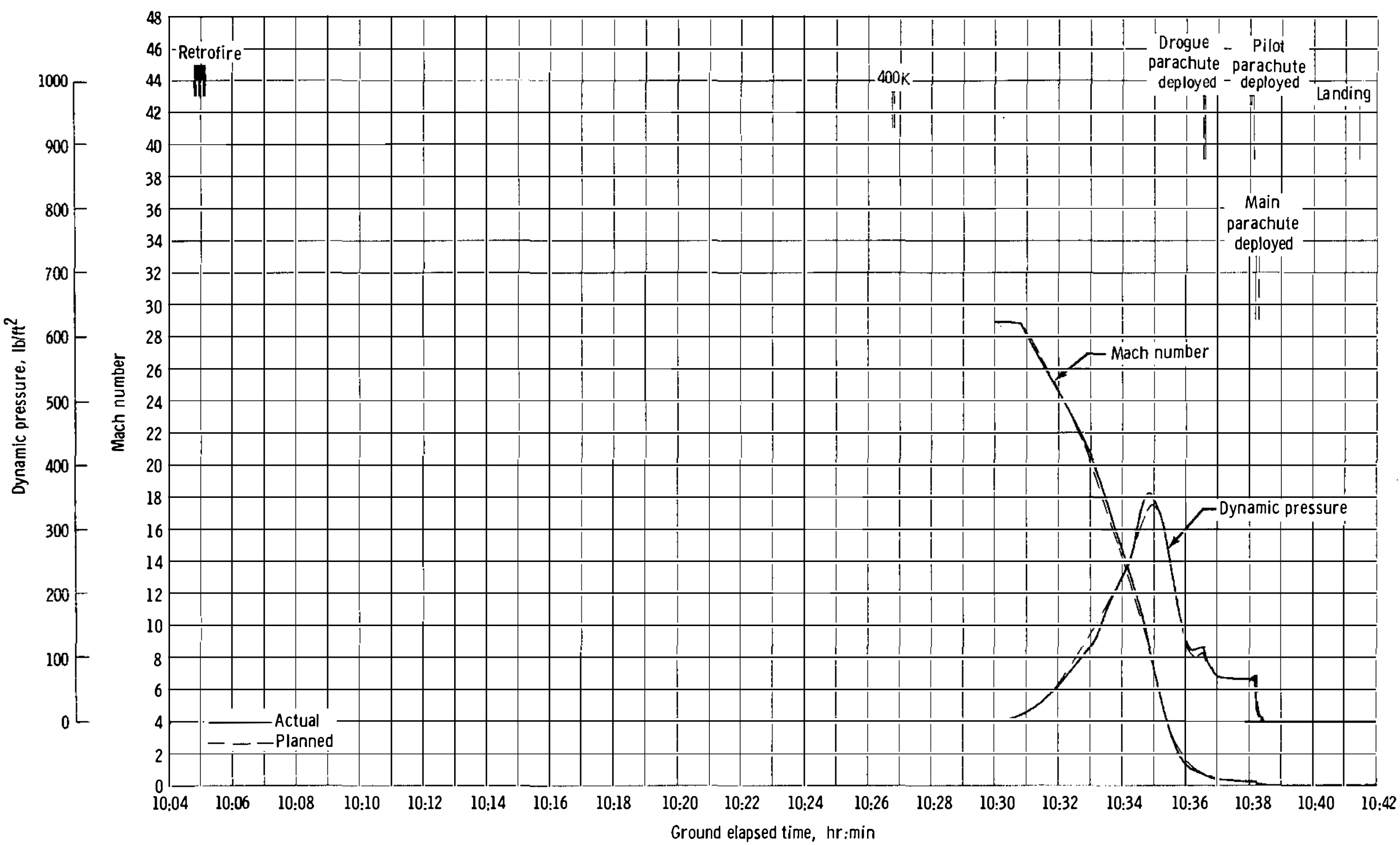

(d) Dynamic pressure and Mach number.

Figure 4.3-5. - Continued.

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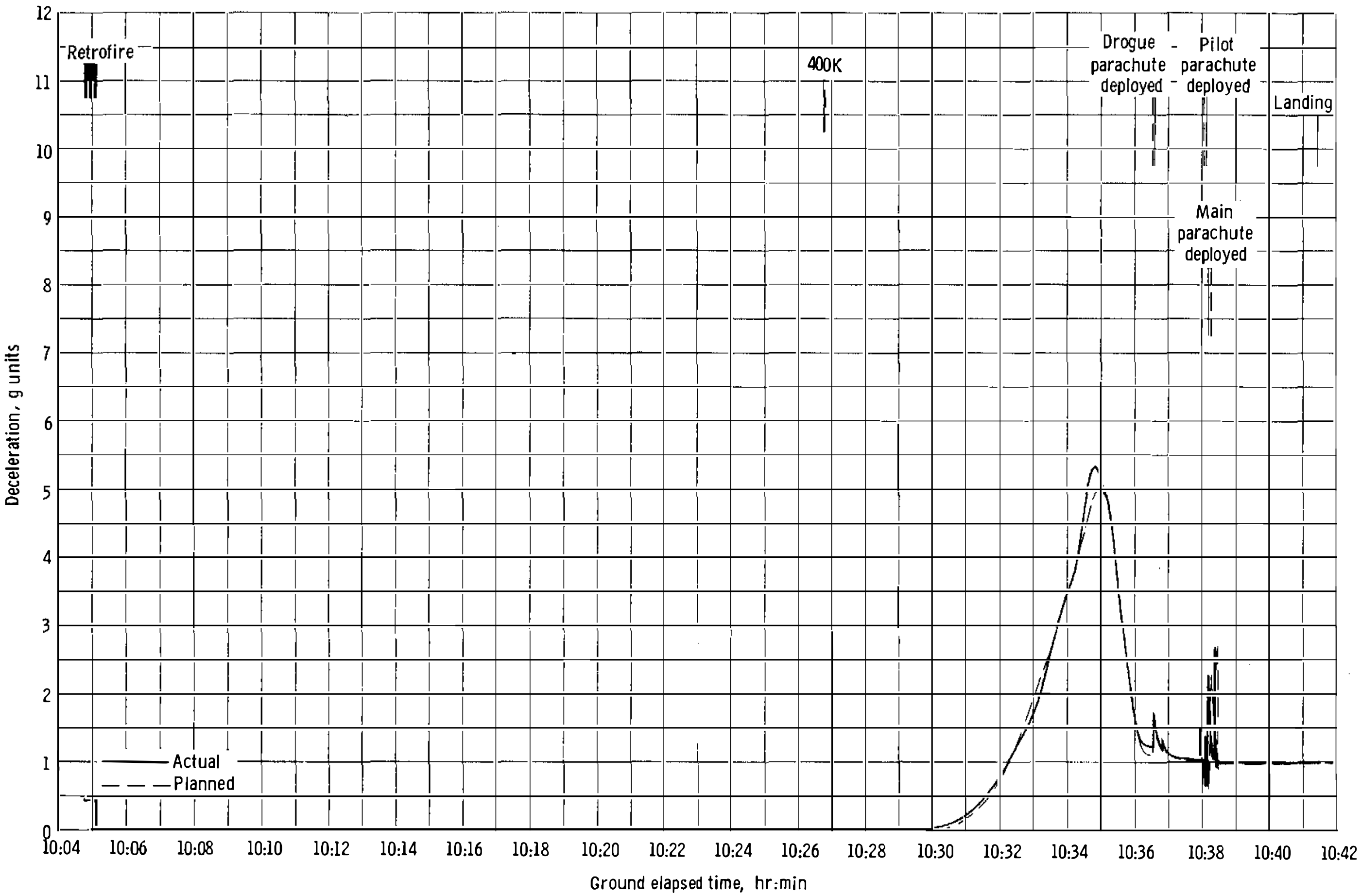

(e) Longitudinal deceleration.

Figure 4.3-5. - Concluded.

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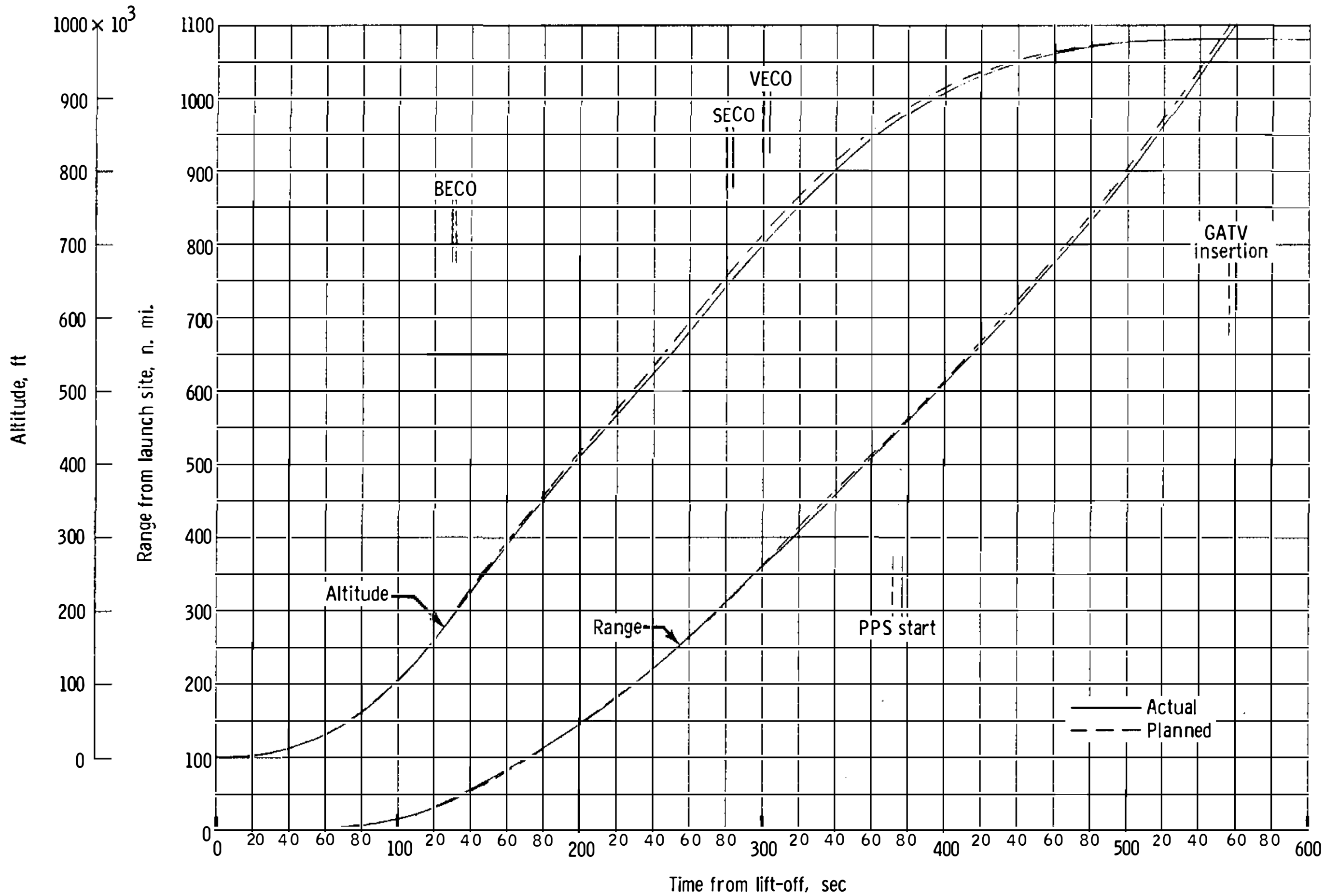
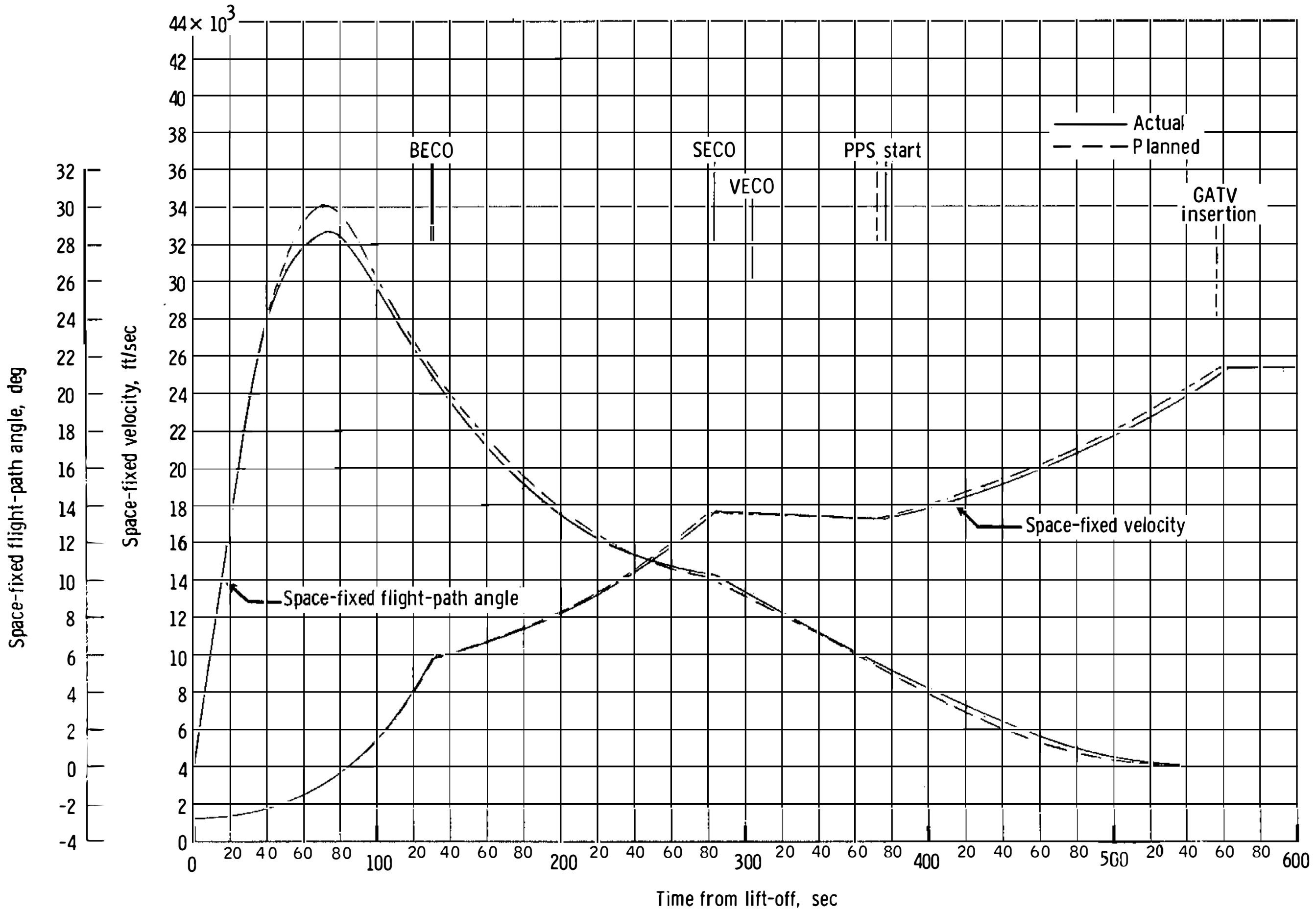


(a) Altitude and range.

Figure 4.3-6.- Trajectory parameters for the GAATV launch phase.

NASA-S-66-3429 APR 2



(b) Space-fixed velocity and flight-path angle.

Figure 4.3-6. - Continued.

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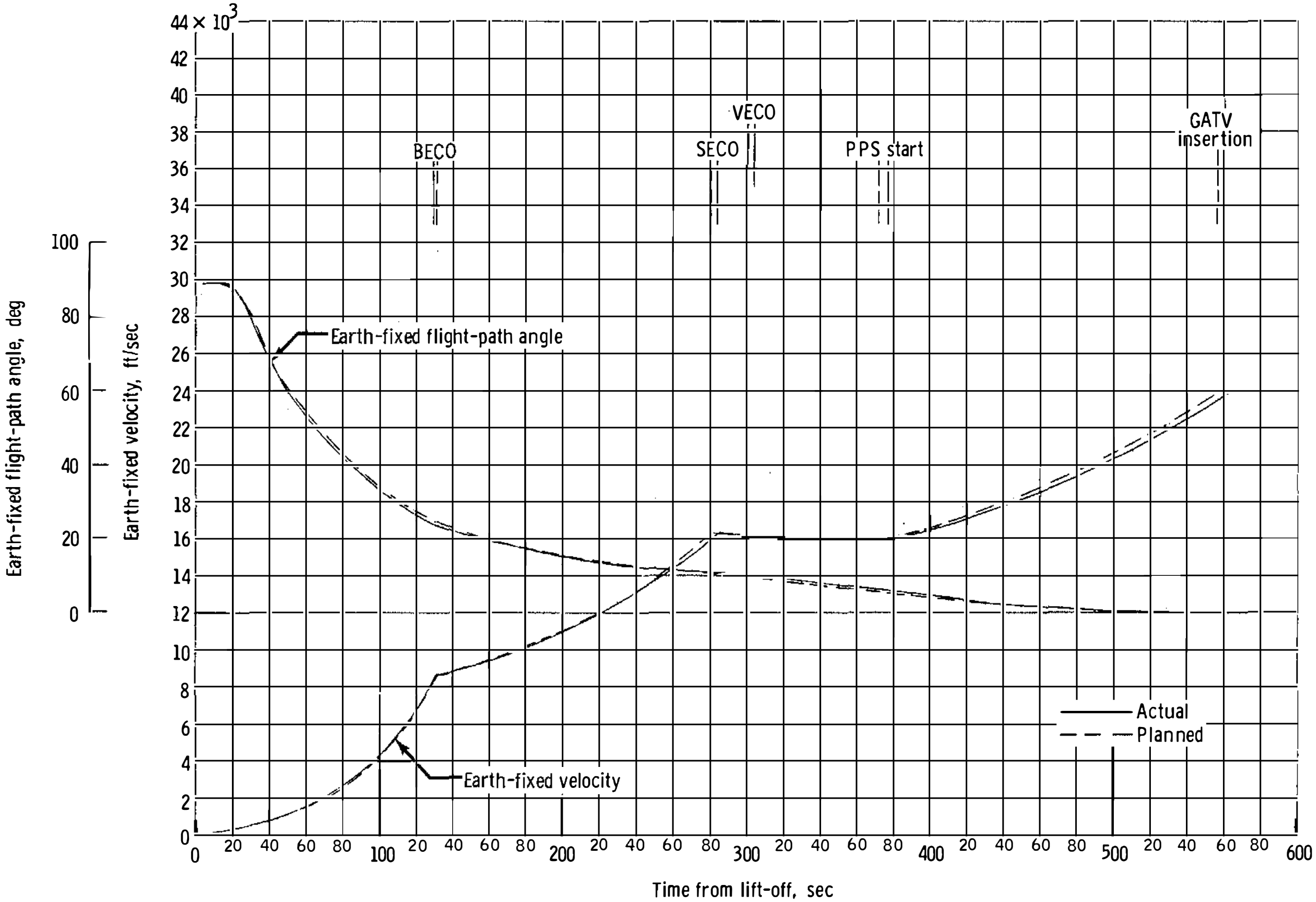

(c) Earth-fixed velocity and flight-path angle.

Figure 4.3-6.- Continued.

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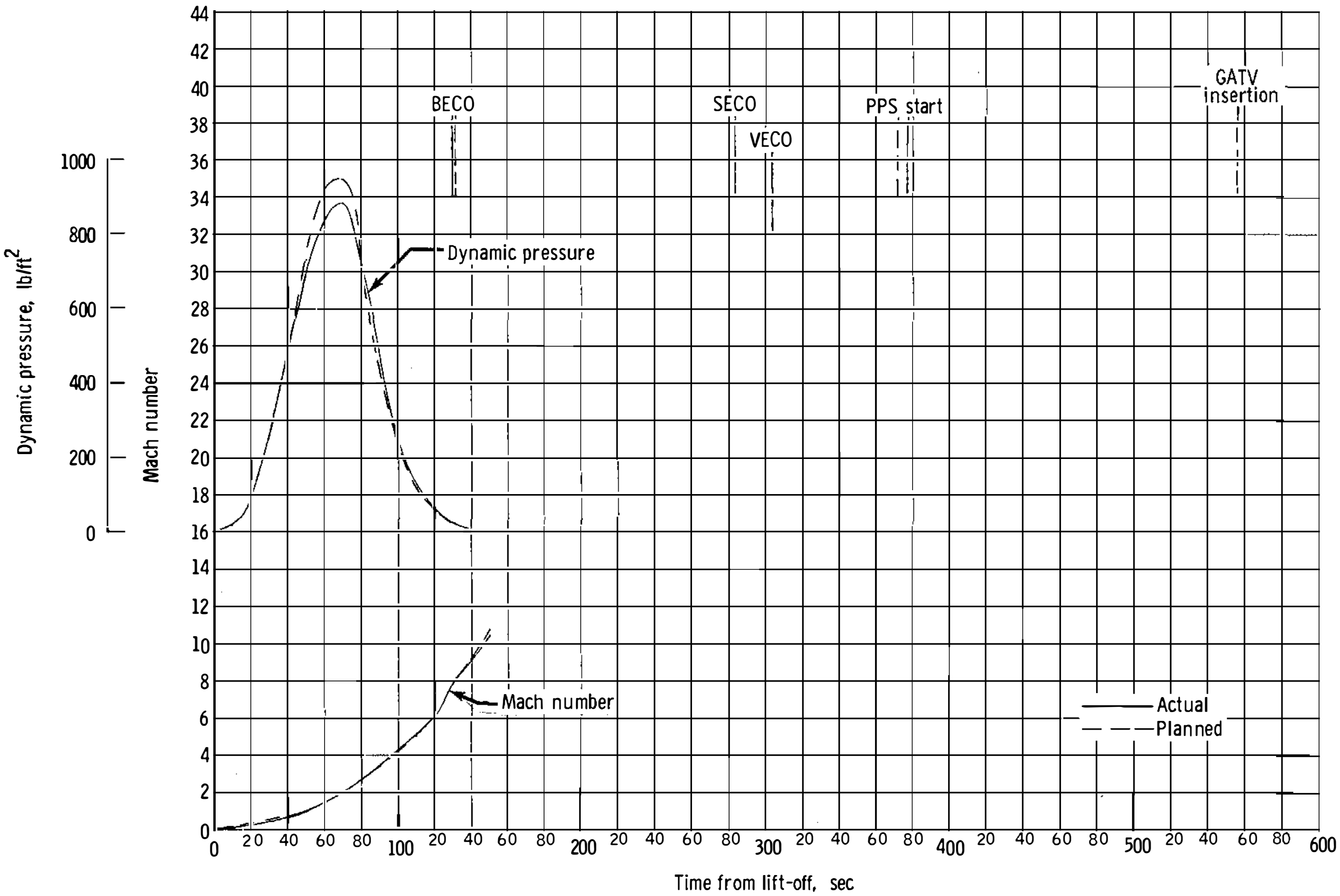

(d) Dynamic pressure and Mach number.

Figure 4.3-6.- Continued.

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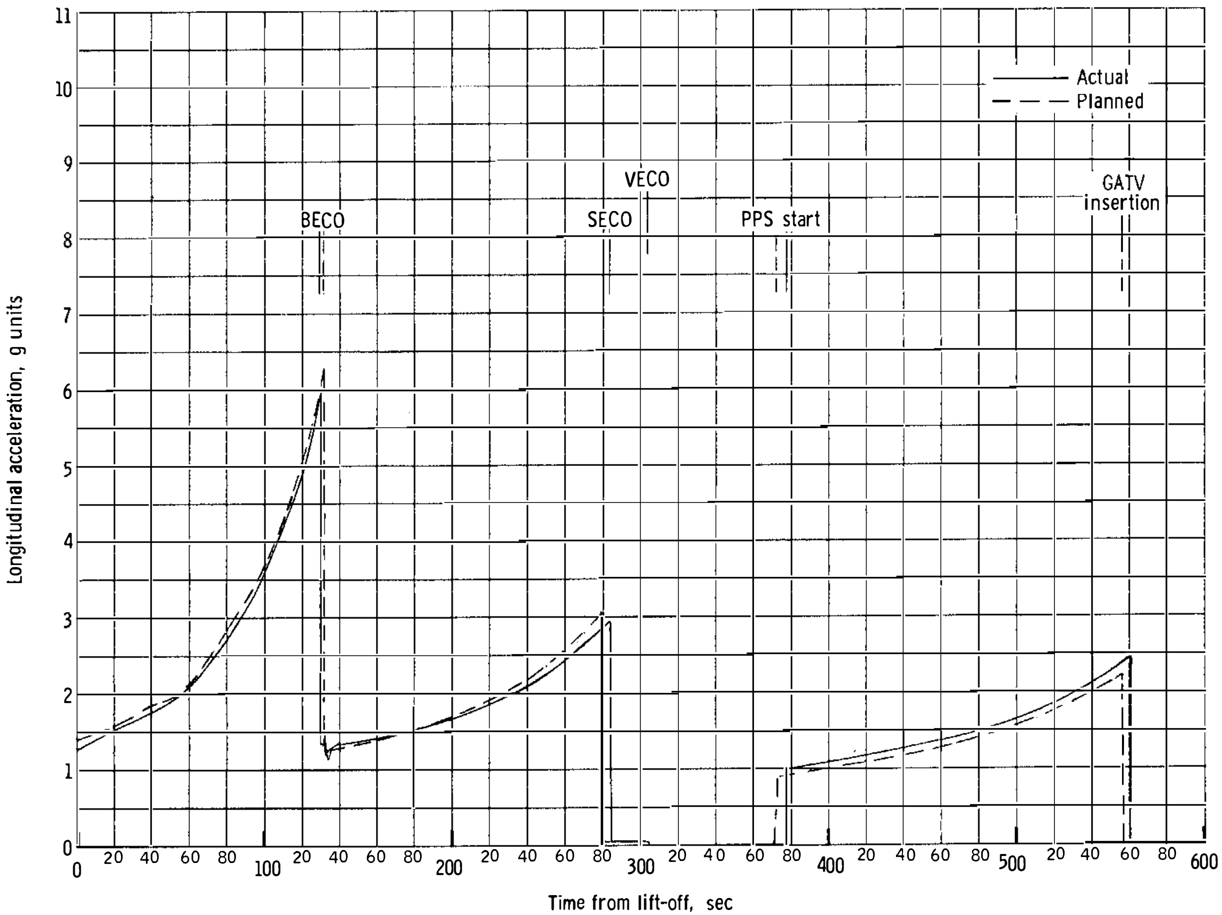

(e) Longitudinal acceleration.

Figure 4.3-6. - Concluded.

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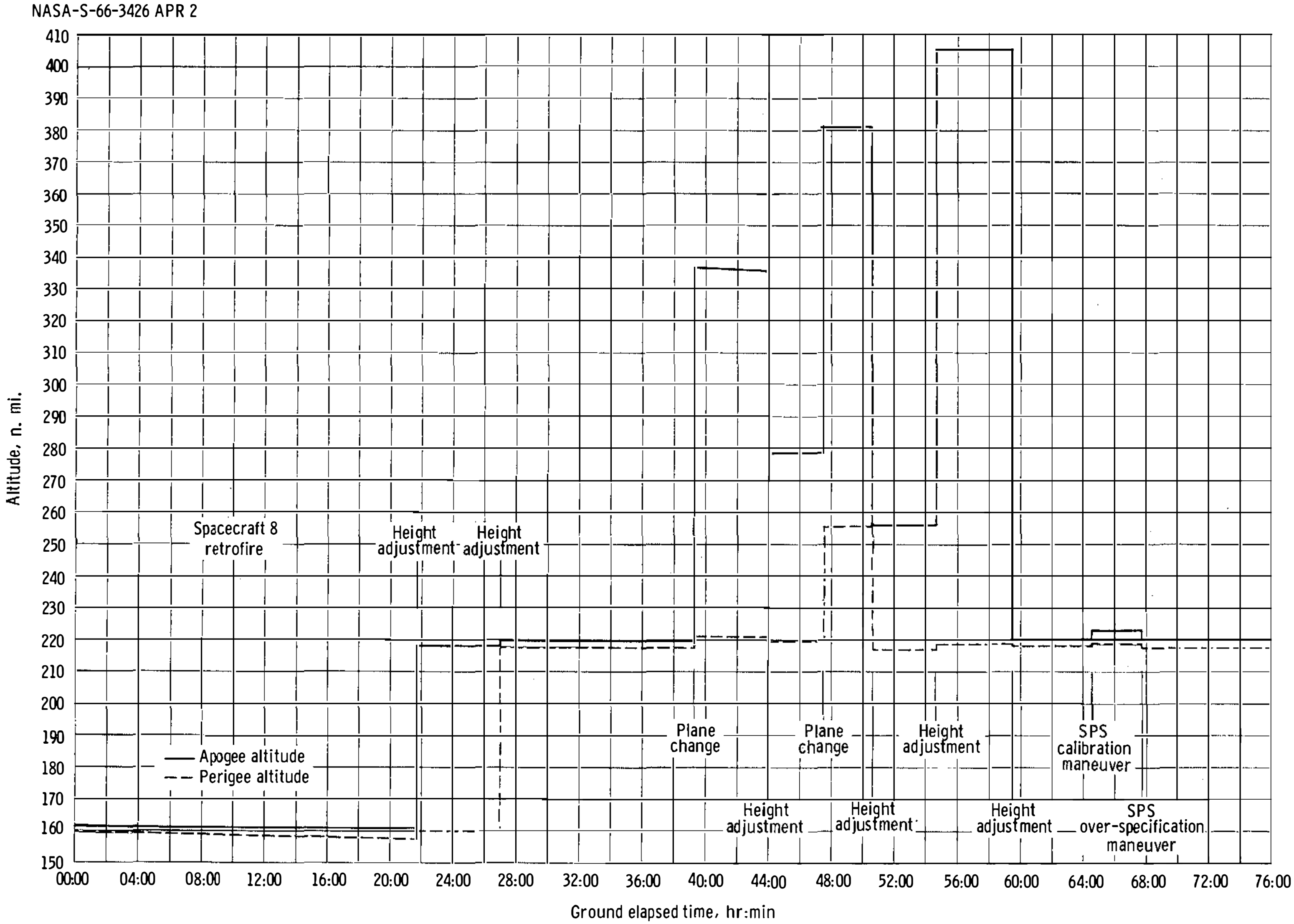

Figure 4.3-7. - GATV apogee and perigee altitude.

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5.0 VEHICLE PERFORMANCE

5.1 SPACECRAFT PERFORMANCE

5.1.1 Spacecraft Structure

The structure sustained the loading and environment of the mission satisfactorily. The Rendezvous and Recovery (R and R) Section of the spacecraft shows no signs of having been subjected to overstress, and minor abrasions on the Fiberite bumper ring are the only evidence of the dynamic structural disengagement of the spacecraft from the Gemini Agena Target Vehicle (GATV) during the control-system anomaly.

The mission was terminated before the planned bending-mode test of the docked configuration could be performed. The instrumentation for this test will be installed on Spacecraft 9, and this test is planned to be performed early in the Gemini IX mission. During launch and during a portion of the docked period, random excitation of the spacecraft accelerometers yielded data indicating that the measurement range and frequency response of the accelerometer system were satisfactory. The random data obtained indicate the frequency of the fundamental bending mode of the docked vehicles to be slightly higher than anticipated and within the envelope of stability conditions investigated.

The crew reported that, when preparing for retrofire, they had difficulty in mating and latching the centerline-stowage-compartment door. Postflight testing without the Extravehicular Life Support System (ELSS) or camera box in the compartment revealed no structural distortions that would require excessive forces to latch the door. Measurements with the cabin pressurized and unpressurized indicated minimal mismatch of the door to the structure, requiring a maximum of only 3 pounds to latch the door. It has been determined that 15 hours prior to the launch, the fit of the ELSS package was rejected as being too loose for launch vibration. As a result, the shear-pin fittings in the ELSS were adjusted so that the door preloaded the ELSS pack when closed. Because it is suspected that the deformations resulting from the pressurized cabin may have increased this preload and caused the difficulty, a test is being performed to examine this possibility and to establish a procedure for adjusting the fit of the ELSS pack in the stowage compartment.

After landing, the crew reported that water droplets were observed at the aft end of the right-hatch hinge. To establish whether the cause was sea-water leakage or internal moisture which had shifted as a result of the landing, a postflight leakage test of Spacecraft 8 was conducted at the contractor's plant. The leakage rate was 430 standard cc per minute, which is well within specification tolerances. No leakage was detected in the hatch area, but a small leak was found at the forward edge of the Environmental Control System (ECS) door at the point where a water stain was found during postflight inspection (section 12.6).

The spacecraft reentry aerodynamics and heating were nominal, with a maximum stagnation heating rate of 45.4 Btu/ft^2/sec and a total heat of 10 000 Btu/ft^2. The apparent stagnation point, as measured on the heat shield, was 13.40 inches below center, which compares closely with the same measurement made after previous lifting reentries. Time histories of the angle-of-attack and lift-to-drag ratios are shown in figure 5.1.1-1.

Gemini VIII had five patches of Velcro bonded to the external surface of the reentry assembly, extending forward in a line from the right hatch. These were to be used during extravehicular activity (EVA) to provide hand holds for the pilot when going from the spacecraft to the GATV. The two patches on the top of the Reentry Control System section survived reentry heating, although the nylon hooks were melted together, and some holes were burned through to the surface of the beryllium shingle. A patch on the cabin shingles and two on the top of the Rendezvous and Recovery Section burned completely off during reentry, leaving only a small amount of charred residue from the bonding agent. The Velcro patches were qualified for launch heating only and are not required to survive reentry heating.

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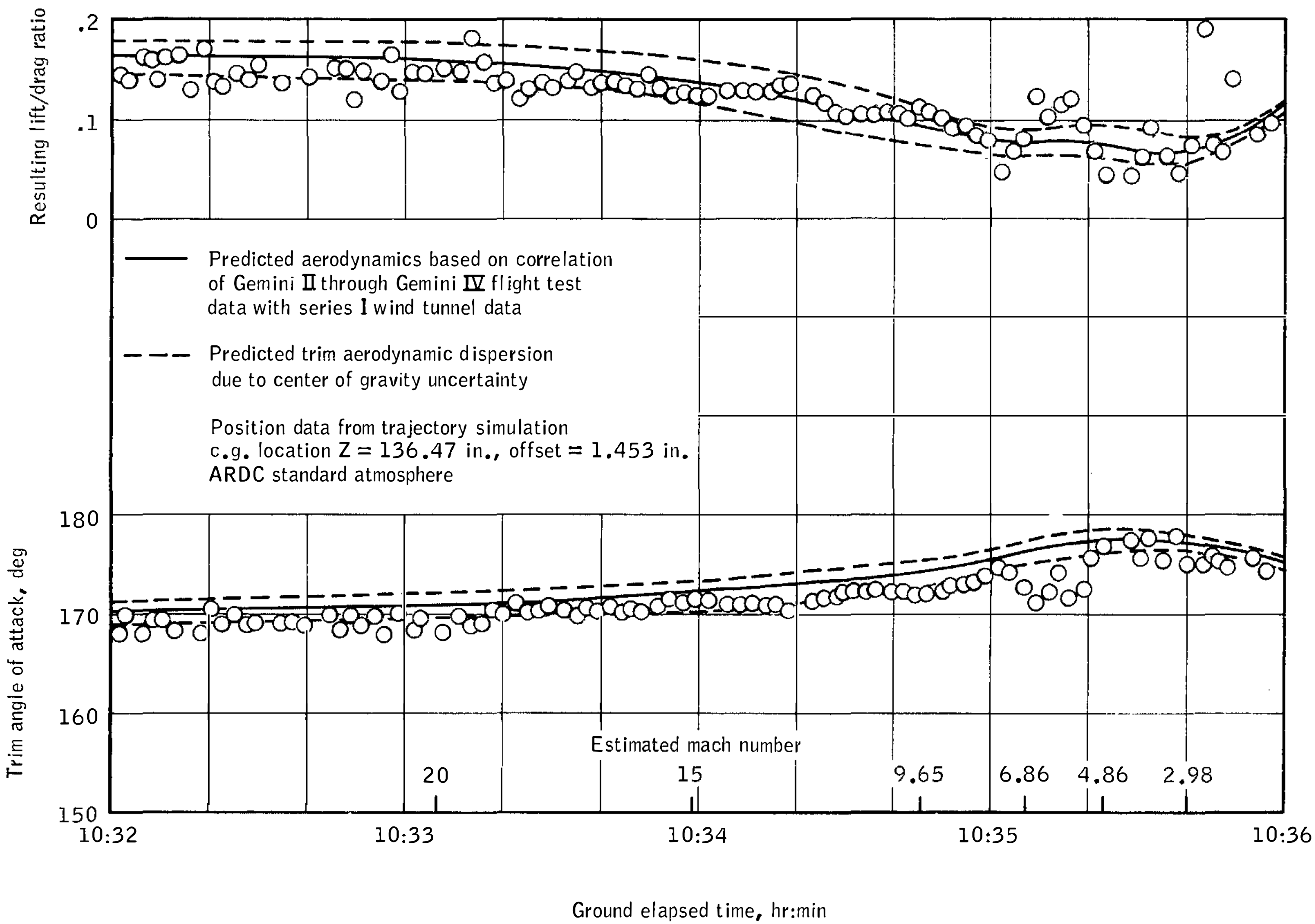

Figure 5.1.1-1. - Preliminary reentry angle of attack and lift-to-drag ratio.

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5.1.2 Communications Systems

All spacecraft communications equipment performed in a satisfactory manner and without evidence of malfunction. During the post-mission debriefings and data analyses, a few minor abnormalities were noted and investigated. During prelaunch tests, several hours before launch, it was found that the flight crew could talk to each other and to ground personnel at reduced volume with the voice-control-center keying-mode switch in the push-to-talk (PTT) position without operating the PTT switch. This did not cause any voice communication problem during the mission; however, it was believed to be an abnormal condition and therefore is under investigation. The condition has been duplicated in laboratory tests and found to exist in the Spacecraft 10 equipment.

Three tapes of good quality were recorded on the spacecraft voice tape recorder during the mission. All voice communications, both transmitted and received, were recorded during this mission. Communications blackout during reentry was predicted to be from 10 hours 30 minutes ground elapsed time (g.e.t.) to 10 hours 35 minutes g.e.t. Signal-strength records were not available to verify these times; however, the predicted and actual times on previous missions have agreed very closely. During this mission, as in previous missions, there were isolated instances of poor intelligiblity during air-to-ground voice communications, possibly caused by improper microphone positioning coupled with low speech intensity. This is judged to have been the reason because, in nearly all cases, the quality immediately improved after the ground personnel asked for a repeat transmission. There were also momentary instances of interference by high breath noise. Background noise, probably caused by air turbulence in the space suit, was intense during brief periods and seemed to vary with crew movement or possibly with suit or neck dam adjustment.

The many instances of superior voice quality, however, showed that the spacecraft equipment was adequate.

5.1.2.1 Ultrahigh frequency voice communications.- Ultrahigh frequency (UHF) voice communications were satisfactory and adequate for mission support during the time preceding retrofire. During the latter part of the anomaly period, communications with the crew were somewhat weak and distorted and some repeats were required. The spacecraft was still tumbling at this time and the adapter-mounted antenna was being used; therefore, the fading signal was probably caused by regions of high attenuation in the antenna pattern being intermittently displayed to the ground station as the spacecraft tumbled. Communications were understandable, even during this period, as evidenced by the fact that a complete air-ground voice transcript was prepared from tapes recorded at the Mission Control Center - Houston (MCC-H).

There were no voice communications with the spacecraft from the beginning of reentry blackout until after the pararescue swimmers were deployed, even though the crew tried several times to contact recovery forces. The spacecraft equipment was operating properly, as evidenced by the flight crew's report of one very good contact with unidentified recovery personnel after the swimmers were deployed. The lack of communications may be explained to some degree by the fact that only one rescue aircraft was in the immediate landing area and it was equipped with only one UHF transmitter-receiver. Because the spacecraft uses a frequency different from those in use by the swimmers and other recovery forces, the aircraft could not simultaneously communicate with the spacecraft and other recovery personnel and could have been tuned to a different frequency at the times the flight crew attempted contact.

5.1.2.2 High frequency voice communications.- The high frequency (HF) voice communications equipment is included in the Gemini spacecraft for emergency purposes during orbit and to aid in locating the spacecraft after landing. The equipment was not used until the postlanding mission phase. HF voice communications were attempted during the postlanding phase, but no contact was established. The crew reported reception of oriental music, which was also received by the California and Canton Island network stations. The HF direction-finding mode was successful, (see section 6.3.3), which is evidence of proper HF transmitter operation.

5.1.2.3 Radar transponders.- The radar transponder configuration was similar to that on Spacecraft 7, and consisted of two C-band transponders, one mounted in the adapter for orbital use and one in the reentry assembly for use during launch and reentry.

The operation of both transponders was very satisfactory, as evidenced by the excellent tracking information supplied by the network stations. There were no problems with spacecraft equipment. Beacon-sharing operations by ground radar were satisfactory. Because of the position of the spacecraft at the time of retrofire for the landing area in the western Pacific, there was no C-band tracking during reentry. The recovery ship reported skin-track radar contact after communications blackout at a range of 105 nautical miles.

5.1.2.4 Digital Command System.- The performance of the Digital Command System (DCS) was satisfactory throughout the mission. Flight-control personnel reported that all commands sent were validated. The DCS case temperature and power supply voltages were normal, and the received signal strength usually varied between -55 dBm and -65 dBm, a strong signal level.

5.1.2.5 Telemetry transmitters.- Nominal operation of all telemetry transmitters was indicated by the quantity and quality of data received. Several network signal-strength charts were reviewed and the signal levels were found to be more than adequate for good telemetry reception and tracking.

5.1.2.6 Antenna systems.- All antenna systems deployed and operated properly during the mission, as evidenced by Communications-System performance. The HF whip antenna installed on the adapter assembly was not extended in orbit. The HF whip antenna installed on the reentry assembly deployed, radiated, and retracted normally during postlanding operations. The UHF descent and recovery whip antennas deployed and operated properly.

5.1.2.7 Recovery aids.- All communications recovery aids operated normally. The UHF recovery beacon was turned on after spacecraft two-point suspension on the main parachute. Reception of beacon signals was reported by aircraft at distances up to 136 nautical miles. One UHF voice transmission was completed with unidentified recovery forces after the pararescue personnel were deployed, and the crew established one voice contact with the recovery aircraft using the swimmers' walkie-talkie radio after the hatches were opened.

The flashing light extended normally, but was not necessary and was not turned on by the crew. The external intercommunications jack, which was provided to permit communications between the rescue personnel and the crew prior to opening the hatches, was not used because the swimmers had not been provided with intercom equipment. The spacecraft was successfully located by means of direction-finding bearings using spacecraft HF transmissions in the HF-DF mode. Operation of spacecraft recovery aids is further described in section 6.3.3.

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5.1.3 Instrumentation and Recording System

The Instrumentation and Recording System performed satisfactorily during the mission with two anomalies being experienced:

(a) The transducer or associated wiring for measuring the Orbital Attitude and Maneuver System (OAMS) regulated helium pressure (parameter GC05) failed at 7:11:30 g.e.t.

(b) The telemetry readout of the cryogenic mass quantities (parameter CA09) was erratic until it was turned off during the fifth revolution.

5.1.3.1 Overall system performance.- A total of 265 parameters were monitored on this mission. Parameter GC05, the OAMS regulated helium pressure, failed at 7:11:30 g.e.t. Further discussion regarding this parameter is included in section 5.1.8. It can be concluded only that a random failure in the transducer or its associated wiring occurred, because the telemetry readout of the reserve-tank pressure did not change at the time of the indicated failure and the source pressure remained steady. The adapter equipment section with the transducer and PCM high-level multiplexer was not recovered, thus precluding any examination of the associated wiring.

Postflight testing is being conducted on the reentry-vehicle circuitry in search of the failure in the mass-quantity cryogenic indication, parameter CA09.

5.1.3.2 Delayed-time data quality.- The delayed-time data reception at the Mission Control Center - Cape Kennedy, and the Texas, Hawaii, and Antigua ground stations is summarized in table 5.1.3-I. This table represents computer-processed data for all delayed-data dumps actually made and for the data from the last orbit and reentry recovered from the onboard PCM tape recorder. The table shows that for the data processed, the usable data exceeded 98.43 percent; and for the onboard PCM recorder alone, the usable data recovered was 99.799 percent. The excessive data losses at Cape Kennedy are attributed to a low-angle pass on revolution 2; however, these data were recovered through the Texas ground station.

5.1.3.3 Real-time data quality.- The real-time data received at Cape Kennedy (CNV) and Hawaii ground stations are summarized in table 5.1.3-II. For all the ground stations listed, the usable data recovered exceeded 97.48 percent. All percentages were derived from computer-processed data edits.

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TABLE 5.1.3-I.- DELAYED-TIME DATA FROM SELECTED STATIONS

Station	Revolution	Total data received		Total losses		Usable data, percent
		Duration, hr:min:sec	Prime subframes	Prime subframes	Percent	
Cape Kennedy	Launch, 1, 2	02:32:37	91 567	5 232	5.713	94.287
Texas	2	01:14:01	44 405	446	1.000	99.000
Hawaii	3, 4, 5	03:52:15	139 354	636	0.456	99.544
Antigua	Launch, 1	01:03:50	38 302	122	0.319	99.681
Onboard recorder	5, 6, 7	03:03:26	110 062	221	0.201	99.799
Summation		11:46:09	423 690	6 657	1.571	98.429

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TABLE 5.1.3-II.- REAL-TIME DATA RECEIVED FROM SELECTED STATIONS

Station	Revolution	Total data received		Total losses		Usable data, percent
		Duration, min:sec	Total master frames	Master frames	Percent	
Cape Kennedy	Launch, 1/2, 2/3	17:50	42 793	312	0.729	99.271
Hawaii	3, 4, 5	21:31	51 648	2 064	3.996	96.004
Summation		39:21	94 441	2 376	2.515	97.485

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5.1.4 Environmental Control System

The Environmental Control System performance was satisfactory throughout the mission. All measured parameters were within the expected ranges of values through all phases of the flight.

5.1.4.1 Crewman comfort.- Crew comfort was good. The system was used with both suit compressors and the A-pump in each coolant loop operating. From approximately 1 hour after launch until just before retrofire, both crewmen had their helmets and gloves removed and wrist dams installed and the pilot also had his neck dam in place. Suit inlet temperatures were 49° F shortly after launch and increased gradually to approximately 54° F near the end of the mission. This increase was a direct result of the increasing coolant inlet temperature to the suit heat exchanger which rose from an initial 42° F to 47° F near the end of the mission. The radiator Vernatherm valve was controlling to approximately 40° F throughout the mission. The 5° F increase of temperature rise between the Vernatherm valve and the suit heat exchanger from launch until the end of the mission is apparently a result of thermal stabilization of the spacecraft, as it compares closely with trends on previous spacecraft. Cabin temperature started at 89° F and increased to 94° F by the end of the mission. This increase is attributed to the high electrical power load of the spacecraft. A review of data from previous spacecraft shows an increase in cabin temperature during periods of high spacecraft electrical load.

5.1.4.2 Gas entrainment.- The crew reported a considerable amount of gas entrainment in the drinking water. The design of the drinking-water storage system for this spacecraft precluded use of the vacuum servicing procedures used on previous spacecraft. When mated to the launch vehicle on the launch pad, the drinking-water storage tank was oriented such that the outlets of the tank were on the horizontal centerline of the tank. Servicing was accomplished by forcing the bladders against the outer wall of the tank to remove gas from the tank, and then backfilling with water. Servicing in this manner could easily trap gas between the bladder and the wall of the tank. An outlet will be added and the servicing procedures changed to allow vacuum servicing similar to that used on Spacecraft 6 and 7 for future spacecraft.

5.1.4.3 Primary oxygen system.- The primary oxygen system functioned as expected throughout the mission. Oxygen usage rate could not be determined because of a failure in the telemetry section of the quantity indicator.

5.1.4.4 Reentry.- The revised ventilation and cabin pressurization procedures for reentry were effective in preventing ingestion of irritating fumes into the suit circuit. Procedures were changed to actuate oxygen high rate at 27K-feet altitude and leave the snorkel inflow valve closed until after spacecraft landing.

5.1.4.5 Postlanding.- The postlanding suit configuration was helmets and gloves off. Neck and wrist dams were not installed. One crewman reported being warm and the other crewman was so uncomfortably warm that he disconnected his space suit from the suit circuit to eliminate the flow of warm gas over his body. The temperature of the gas entering the suit immediately after landing was warmer than normal because of heating of the gas by the hot spacecraft structure. Removing the hoses probably provided an improvement in apparent comfort because the flow of warm gas was terminated. However, only a few minutes should be required for the spacecraft structure to cool down so that the gas supplied to the space suit would be only a few degrees above ambient temperature. Postlanding cooling would have been improved by installation of the wrist dams, because the rate of gas flow through the space suit would have been increased. However, comfortable conditions may still not have been attained. Installation of the neck dam in addition to the wrist dam would have reduced the flowrate of gas through the suit and probably degraded the cooling.

5.1.5 Guidance and Control System

5.1.5.1 Summary.- The Guidance and Control System performed satisfactorily throughout the mission, except for a possible association with the attitude control anomaly. Table 5.1.5-I contains a summary of events significant to the system. Ascent (secondary), rendezvous, and reentry guidance was excellent with results close to nominal. The control system performed properly during the exacting station keeping and docking maneuvers. The available evidence indicates that the attitude control anomaly was not a failure of control-system components. The Auxiliary Tape Memory Unit (ATMU) was utilized for the first time to reload the onboard computer memory and successfully entered the touchdown-predict reentry program.

5.1.5.2 Inertial Guidance System performance evaluation.-

5.1.5.2.1 Ascent phase: The Inertial Guidance System (IGS) roll, pitch, and yaw steering command deviations are represented in figure 5.1.5-1. Superimposed on the IGS steering quantities are the steering signals indicated by the primary system, the Radio Guidance System (RGS), along with the IGS attitude-error limit lines for nominal steering signals. Analog time histories of predicted pitch and yaw attitude errors for winds at T - 5 hours are shown for the first 90 seconds of flight. The IGS responded as expected to the vehicle dynamics, as directed by the primary guidance, and gave all indications of excellent performance during the ascent guidance phase.

With the introduction of the variable launch azimuth and dog-legged trajectory into the Gemini flights, there has been some concern as to what the Flight Director Attitude Indicator (FDAI) should indicate after completion of the required launch-vehicle roll program. At T - 3 minutes the Inertial Measuring Unit (IMU) X-axis was oriented to a true heading of 96.4 degrees, 6.4 degrees south of East. (See fig. 5.1.5-2). Because the IMU is referenced to GLV axes which are aligned 5 degrees west of North on the launch pad, the reading displayed on the FDAI was 101.4 degrees at this time. After the programmed 14.95-degree roll-left maneuver was performed, the FDAI displayed 86.45 degrees (101.4 - 14.95). As noted in the figure, the GLV Y-axis (pitch axis) which was oriented 5 degrees west of North before launch, was then oriented 9.9 degrees east of North and the pitch plane or true launch azimuth was 99.9 degrees (14.95 - 5 + 90). To obtain the launch azimuth from the FDAI post-roll reading (86.45), it is necessary to add 13.45 degrees, the sum of the GLV Y-axis offset from North (5 degrees) and the GLV Z-axis offset from the IMU X-axis (8.45 degrees).

Both azimuth updates were received and properly utilized by the onboard computer. The significant misalignment of 43 arc-seconds that remained after the updates indicated that the ground-computed velocity-update values were less accurate than on previous flights.

If guidance switchover had occurred early in Stage II operation, the second-stage engine cutoff (SECO) conditions, prior to any incremental velocity adjust routine (IVAR) correction, would have been within the following deviations from nominal: +7 ft/sec in velocity, +200 feet in altitude, and +0.007 degree in flight-path angle. The low in-plane IMU navigation errors, coupled with the IGS delivering the SECO discrete signal within 25 milliseconds of the primary SECO discrete signal, substantiates the comparison between primary and secondary guidance. The 10 ft/sec separation ΔV and a subsequent IVAR correction would have resulted in a close-to-nominal trajectory.

Figure 5.1.5-3 shows the accelerations measured by the IMU during the period after SECO. As noted, the accelerations decreased to near zero at SECO + 18 seconds but then increased prior to the firing of the aft thrusters and separation so that approximately 0.4 ft/sec^2 existed at the start of the separation sequence.

On this mission it was planned that the IVAR solution would be applied according to the corrections indicated on the Incremental Velocity Indicators (IVI's), if the required velocity change was indicated to be between 5 and 30 ft/sec forward. If a value between 5 ft/sec aft and 5 ft/sec forward was indicated, then the minimum 5 ft/sec forward separation maneuver was to be performed. The IVAR was utilized as planned on this mission and resulted in an apogee of 146.7 nautical miles, 1.2 nautical miles higher than the nominal 145.5. Because of the slight overspeed condition existing at SECO, the pre-separation IVAR display was negative (4 ft/sec aft) and therefore, as planned, the minimum separation ΔV of 5 ft/sec forward was applied (6.2 ft/sec actual) and the effect of the small velocity error at SECO was minimized (the previously used standard 10 ft/sec separation ΔV would have raised apogee to approximately 149 nautical miles).

If the IVI's had been driven to zero, the resulting apogee would have been approximately 5 nautical miles lower than actually achieved, or approximately 3 nautical miles lower than targeted. The IGS out-of-plane navigation error was 12 ft/sec and of opposite sign to the 13 ft/sec RGS error. Therefore, the IVAR called for a 25 ft/sec out-of-plane correction. The perigee correction to be applied at apogee, as computed by the IVAR, was less than 0.1 ft/sec, which reflects the close-to-nominal perigee achieved.

The IVI display, as actually computed by the onboard IVAR, was reconstructed using IGS navigational and gimbal-angle data. The reconstruction agrees with the crew reading of 4 ft/sec aft and 25 ft/sec up just prior to spacecraft separation from the GLV. The 25 ft/sec is the out-of-plane velocity component which was displayed as an UP indication because the spacecraft was still rolled 90 degrees. A second reading was reported by the crew as 10 ft/sec aft, 18 ft/sec right, and 12 ft/sec up. The crew reported this to have been read sometime after separation. These readings would have been valid at about 405 seconds after lift-off or 39 seconds after separation. Since the onboard computer was switched from ascent to prelaunch mode at 405 seconds, this display would have remained the same until zeroed by the crew or by switching into the catchup mode. The computer was switched to catchup mode before the roll to heads-up attitude was completed; therefore, the out-of-plane velocity of 22 ft/sec was displayed in component form (18 right, 12 up). Following the roll to zero degrees, the out-of-plane velocity would have been displayed as 22 ft/sec right if the computer had remained in ascent mode. The values of the reconstructed IVAR parameters in the final computation cycle, as compared with the actual final values obtained from telemetry, are presented in table 5.1.5-II.

A preliminary estimate of IMU component errors was obtained by comparing ground tracking measurements with guidance position and velocity data.

The external tracking data used for comparisons were GE Mod III final data and Missile Trajectory Measurement (MISTRAM) data using the 100K-foot legs. From lift-off (LO) to LO + 270 seconds, the GE Mod III final data and MISTRAM data agree along the X (downrange) axis within 1 ft/sec, and within 2.5 ft/sec along the Z (crossrange) axis. After LO + 290 seconds the GE tracking data became noisy (see fig. 5.1.5-4). The velocity residuals along the Y (vertical) axis indicate a discrepancy between the GE Mod III and MISTRAM data, particularly after LO + 270 seconds. The rapid increase in the MISTRAM comparison residuals after LO + 270 seconds suggests a MISTRAM P-bias error, although the GE Mod III tracking may also have been in error. The accelerometer telemetry data acquired during ascent had no significant dropouts, and were excellent for analysis.

The velocity residuals obtained with MISTRAM were used to estimate a set of IMU component errors which induced velocity-error propagations along the X-axis and Z-axis as shown in figure 5.1.5-5. The residuals obtained using GE Mod III final data were used to estimate component errors which could account for the error along the Y (vertical) axis.

Figure 5.1.5-5 contains a history of preflight IMU component calibrations and the postflight deduced coefficients discussed herein.

The Z (crossrange) velocity error appears to have been caused by an azimuth error in platform orientation, because the Z-axis is approximately the out-of-plane axis during ascent, and the velocity-error trend has the shape of the downrange velocity. An azimuth misalignment of approximately 43 arc-seconds, which the RGS apparently failed to correct, and a g-sensitive drift (X-gyro spin-axis unbalance) of 0.72 deg/hr/g can account for most of the error along the Z-axis. This large g-sensitive-drift term could easily be a combination of other smaller gyro-drift terms which propagate along the Z-axis; however, it is difficult to determine each small drift term because they are highly interdependent.

The step change in velocity difference observed at first-stage engine cutoff (BECO), and the ramp-like trend of the X-axis velocity residuals from LO to LO + 257 seconds, indicate a timing error (fig. 5.1.5-4). A much smaller change was noted at SECO; therefore, it was concluded that there was a timing error in correlating the IGS and tracker time, compensated for by an IGS time-scale-factor error of 100 ppm. An accelerometer scale-factor error of 200 ppm was also determined to be a major contributor to the X-velocity error. The trend of the Y-velocity error was somewhat uncertain; however, a curve fit of the data was obtained, and the error sources are shown in table 5.1.5-III.

A summary of preliminary estimates of IMU component errors and the total velocity error induced by each error source during powered flight are given in table 5.1.5-III. In addition, sensor and tracking errors obtained from a preliminary Error Coefficient Recovery Program (ECRP) computer run are presented. The major error sources obtained from the ECRP agree very well with those obtained by a hand fit.

The present best estimates of the guidance position and velocity errors at injection are given in table 5.1.5-IV. These quantities were obtained from position and velocity comparisons using present best estimates of the tracker reference trajectory. In this table, the IMU error consists of sensor errors, while navigation errors result from various approximations within the airborne computer. An estimate of orbital injection parameters at SECO + 20 seconds, as determined from the IGS and other sources, is given in table 5.1.5-V.

5.1.5.2.2 Orbital phase: The IGS was utilized during this phase of the mission as a reference for ground-calculated translation maneuvers and to compute the velocity corrections required for the closed-loop portion of the rendezvous maneuver. The IMU was aligned several times between separation and retrofire with no apparent difficulty. Exact times and results cannot be determined because torque currents were not telemetered; however, representative pitch and roll errors during known alignment periods are listed in table 5.1.5-VI. A representative time history of these errors is contained in figure 5.1.5-6 for the preretrofire alignment, which was performed in both platform and pulse modes, and the figure also indicates relative performance in each mode.

A summary of major translation activity, as calculated from telemetered accelerometer data, is shown in table 5.1.5-VII. As a result of an accelerometer bias check made during the first revolution, small X and Z accelerometer bias updates were inserted after the height-adjust maneuver. The errors in bias prior to the update were not large enough to cause significant errors in calculation prior to that time. As noted in the table, the velocity changes obtained were within 0.4 ft/sec of those desired in all cases where an attempt was made to be precise.

In order to determine the desirability of reducing desired-velocity-change residuals in all axes, an analysis of this activity after the 61.5 ft/sec coelliptic maneuver was made and the results are presented in figure 5.1.5-7. The ΔV's accrued in each axis from each attitude-control thruster were summed with those from the translation thrusters and are plotted on the figure. Preflight-test thrust values for each thruster, telemetered firing times, and nominal prerendezvous spacecraft weight were used to calculate the ΔV's. The out-of-plane accumulation was observed to vary from -0.3 to +0.4 ft/sec from attitude control activity alone. No out-of-plane translation thrusters were operated. The vertical velocity varied from -0.1 to +0.6 ft/sec from a combination of pitch attitude thruster activity and three short firings from thrust chamber assembly (TCA) no. 16. The inplane accumulation was +0.9 ft/sec from the aft translation thrusters and from the canted TCA no. 16. Note that no ΔV's are accumulated in this axis from attitude thruster activity.

MDIU readouts taken during this period would have properly reflected the ΔV history plotted in the figure. The crew report of fluctuating readouts is therefore substantiated and reflects normal system operation.

The rendezvous radar was turned ON, in STANDBY, at 3 hours g.e.t. and switched to SEARCH about 5 minutes later. At 3 hours 27 minutes g.e.t., with the transponder operating, the dipole and spiral antennas were observed to be switching normally. Figure 5.1.5-8 contains a history of significant radar events during the rendezvous maneuver. The target was acquired intermittently on the dipole antenna, at 3:27:48 g.e.t., and the radar locked on solidly 3.5 minutes later, at a range of 181 nautical miles. The lock-on sequence was normal, with the first computer range readout (1095K feet) occurring at 3:26:06 g.e.t.

Normal tracking ensued until 3:39:45 g.e.t. when the radar signal strength dropped 8 dB for 4 seconds and the radar crystal current indicated that an automatic-frequency-control (AFC) sweep occurred. (See fig. 5.1.5-9 for a history of these parameters.) This transient was caused by a SPIRAL SELECT command being sent to the Gemini Agena Target Vehicle (GATV). Telemetry data indicate that the transponder switched to spiral antenna for 4 seconds, did not recognize a target, and then returned to the dipole antenna and locked on.

Tracking again continued normally until 5:01:32 g.e.t. At this point, at a range of 46 nautical miles and an elevation of 16 degrees, the radar signal strength dropped abruptly from -70 dBm to -85 dBm. Real-time GATV telemetry from Guaymas and the Rose Knot Victor indicated that the radar was locked on the dipole antenna. For approximately 30 minutes, until 5:30:45 g.e.t., the signal strength fluctuated as indicated in figure 5.1.5-9, with loss of lock occurring once at 5:04:07 g.e.t. At 5:21:30 g.e.t., 7 minutes after terminal phase initiation (TPI), the SPIRAL SELECT command was again sent at a relative elevation angle of approximately 35 degrees (55 degrees off the spiral axis). Under these conditions the radar should have locked on the spiral antenna. GATV telemetry data are not available for this period so it cannot be determined if the radar locked on the spiral or returned to the dipole antenna. At approximately 5 hours 34 minutes g.e.t., the time when the radar switched to the wide bandwidth amplifier, normal tracking resumed and continued until rendezvous was completed.

The abnormal fluctuations in signal strength are representative of those which would be expected from the relatively narrow beam width of the spiral antenna. Investigations are underway to determine the cause of these abnormal fluctuations. The erratic radar angle measurements reported by the crew occurred during this period and are attributed to the same cause. However, as indicated below, the radar-dependent calculations of the onboard computer were proper at this time, indicating that the information received was of a nature to be correctly processed by the computer. Figure 5.1.5-10 contains a time history of the residuals obtained from comparing rendezvous radar range, azimuth, and elevation with like quantities computed from ground tracking data. The residuals exhibit a cyclic variations which is caused by errors in the ground data, but give no indication of off-nominal radar performance.

The encoder was used during the predocked, docked, and postdocked phases of the mission and performed normally. Commands were sent via both the RF link and the hardline. The RF link was utilized during both predocking and postdocking periods. The messages were decoded and the corresponding actions were correctly initiated by the GATV programmer.

Time histories of radar temperature and pressure and transponder temperature are included in figure 5.1.5-11. The transponder temperature experienced a positive heat transient during the launch phase, then fluctuated normally between 45° and 65° F for the remainder of the mission.

The radar parameters were nominal throughout, except for a short period after docking when the system was left in STANDBY. During this period, the temperature rate of rise increased to 16°F/hr.

The rendezvous mode was selected at 3:34:00 g.e.t. (approximately 3 minutes after radar lock-on) for a rendezvous mode check. The first total-velocity-to-rendezvous (ΔV_T) calculation was 854 ft/sec, which was proper for the conditions at that time. The mode was re-initialized for the closed-loop phase at 3:53:00 g.e.t.

Figure 5.1.5-12 contains time histories of ΔV_T calculated in flight by the onboard computer and computed postflight from dynamic simulations using Real-Time Computer Complex (RTCC) and BET state vectors. Figure 5.1.5-13 contains radar range, azimuth, and elevation, and the three IMU gimbal angles. A comparison of these figures shows that the onboard computer calculation of ΔV_T was sensitive to off-bore-sight conditions. Variations in ΔV_T occurred at 4:19:00 g.e.t. and again at 4:44:00 g.e.t., when a pitch-down maneuver was initiated prior to a platform alignment. These variations are representative of those which can be expected when the angle of the boresight is significantly off or during rapid attitude changes when antenna servo lags exist. Preflight tests on this radar show the following errors for 10-degree off-boresight conditions:

Angle off boresight, deg	Elevation error, deg	Azimuth error, deg
+10	0.9	0.1
-10	0.5	0.5

Further comparison of these figures shows that the apparent bias between the simulated and onboard-computed values disappears after the platform alignment. Although the errors in the trajectory data shown in figure 5.1.5-10 preclude an accurate assessment of this bias, a 0.1 degree misalignment in the sensitive out-of-plane (yaw) axis would cause an offset of this order. The sensitivity to angular errors noted here clearly indicates the value of accurately tracking the FDI needles and the need for the best possible platform alignment at this time.

The TPI velocity calculated at 5:14:45 g.e.t. agreed well with the back-up value transmitted from the ground. The values were:

Condition	Onboard-computed ΔV, ft/sec	Ground-computed ΔV, ft/sec
Fore-aft	26 forward	32 forward
Right-left	8 left	5.7 left
Up-down	3 down	1.7 down
Total vector	27.5	32.6

The radar range during the final phase is shown in figure 5.1.5-14. The range was closing linearly prior to the braking maneuver and, if extrapolated to the nominal time of rendezvous, would have resulted in a miss distance of 1500 feet.

The Auxiliary Tape Memory Unit (ATMU) was installed and utilized for the first time on this mission. Although the early reentry forced cancellation of the extensive tests programmed for the ATMU, the major mission objectives were achieved and satisfactory performance of the unit was demonstrated.

The ATMU was turned on at 7:39:13.8 g.e.t. in preparation for loading the touchdown-predict reentry program (Module IV). Figure 5.1.5-15 contains a time history of the significant events during this period. The crew reported that the first attempt to load the module was unsuccessful, but that on the second attempt, the operation proceeded smoothly.

Module IV-A was successfully transferred within 4 minutes 30 seconds after powering up the ATMU and was verified in the next 5 minutes 30 seconds for a total time of 10 minutes 6 seconds for the automatic reprogram and verify operation. Approximately 30 minutes later, the redundantly stored touchdown-predict reentry Module IV-B was verified against the previously loaded Module IV-A. This search and verify operation required an additional 9 minutes 10 seconds. Approximately 95 short-duration thruster firings occurred during this period with no adverse effects. An operational test of these conditions was to have been conducted during the mission; therefore, an important secondary mission objective is considered to have been met.

The cause of the reported failure of the ATMU to operate on the first attempt has not been determined; however, the system was recycled and operated properly. Similar indications would result from an incorrect manual data insertion unit (MDIU) entry, or other procedural error, from failure of the computer to process the information entered, or from failure of the computer to receive or recognize the ATMU mode discretes. The sequence of events does not show that the computer running light went off as it should have when the ATMU was switched to AUTO following the MDIU insert, with ATMU power on. It cannot be determined from telemetry whether these events occurred; however, all subsequent performance was nominal. The ATMU case reached a maximum temperature of approximately 70° F during prelaunch operations, then stabilized at approximately 65° F during the orbital phase. No detectable loss in ATMU internal pressure was noted.

5.1.5.2.3 Retrofire - reentry phase: The flight crew reported before retrofire that the time-to-go to retrofire (T_R) was counting up when T_R was initially read out of the computer. This occurred because, at that time, the Time Reference System (TRS) was loaded with T_R for recovery area 45-1 which was in excess of 3600 minutes. This value overflowed this parameter in the MDIU subroutine. T_R is rescaled from 2^{22} in the TRS to 2^{16} in the MDIU subroutine. Therefore no value of T_R greater than 1092 minutes can be read out of the computer through the MDIU without causing an overflow. However, the T_R for recovery area 45-1 was still valid in the TRS and was counting down properly. After the T_R for recovery area 7-3 was updated in the TRS at the next network station, the value was properly displayed to the crew by the MDIU.

The IGS operated correctly throughout the retrofire and reentry phases of the flight. The total velocity change as a result of the firing of the retrorockets was 1.99 ft/sec higher than predicted (table 5.1.5-VII). The total footprint shift due to retrofire was 9.8 nautical miles as shown in figure 5.1.5-16.

From retrofire to an altitude of 400K feet, a 10-degree bank angle toward the south was flown as planned. At 10:26:53.239 g.e.t., the computer commanded a zero-degree bank angle which indicated proper spacecraft navigation to the 400K-foot level when compared with the time of 400K feet as computed on the ground by using IVI data acquired after retrofire. From the 400K foot level to guidance initiation, the back-up bank angle of 52 degrees toward the south was flown as planned. At 10:29:58.5 g.e.t. the spacecraft passed an acceleration level of 1.0 ft/sec^2 (density altitude-factor of 8.71237) and the computer began to calculate the bank commands necessary to guide the spacecraft to the desired target.

At 10:31:02.91 g.e.t., the flight crew started to fly the bank angles commanded by the onboard computer. From this time until guidance termination at 10:35:50.392 g.e.t., the commands from the computer were accurately flown by the flight crew. The time histories of bank command, actual bank angle, downrange error, and crossrange error are presented in figure 5.1.5-17. The computer properly terminated guidance at a density altitude factor of 4.609.

Table 5.1.5-VIII contains a comparison of the actual telemetry data with that reconstructed after the flight using the DCS update, gimbal angles, spacecraft body rates, and platform accelerometer outputs. This table indicates close agreement between the sets of data, and demonstrates the proper functioning of the computer in the reentry mode.

The IGS-computed spacecraft position at guidance termination (80 000 feet) was 1.4 nautical miles to the right of the desired track. The insert in figure 5.1.5-16 shows the relative position of the spacecraft at touchdown with respect to the planned target. No tracking data are available to accurately check the navigation accuracy of the onboard systems; however, the recovery aircraft reported the spacecraft in sight on the main parachute at an estimated distance of 3 miles.

5.1.5.3 Control system performance evaluation.-

5.1.5.3.1 Attitude Control and Maneuver Electronics (ACME): The attitude control system became active at LO + 339.6 seconds (2.1 seconds after SECO) when the Orbital Attitude and Maneuver System (OAMS) attitude control power was turned on. The ACME was in the rate-command mode at this time and thrusters 3, 4, 5, 6, and 8 were automatically fired in an attempt to null the small post-SECO rates of the combined vehicles. Direct mode was selected 1.1 seconds later, thus stopping the thruster firings. Separation from the GLV was nominal with thrusters 9 and 10 firing for 8 seconds. The spacecraft-separation switch was operated 2.8 seconds after the thruster-firing command, and rate-command mode was selected 2.0 seconds after separation. Normal, small transients in the rates were observed and were immediately nulled by the rate command system. The roll-to-heads-up maneuver was performed at 2 deg/sec after translation thrusting had ceased.

Attitude and translation control was nominal throughout the rendezvous phase. Translations were performed both in rate command and platform modes. Attitude thruster activity, counteracting small disturbance torques, was normal and attitudes were held within approximately ±1 degree in all cases. A time history of gimbal angles during the coelliptical maneuver is shown for reference in figure 5.1.5-18. Attitude control during radar boresight tracking was excellent, showing the capability to follow the radar angles to within ±0.30 degree. Also, as shown in the station-keeping and docking sequence films, the capability for very precise attitude and translation control was available and exercised.

At 7:00:26.7 g.e.t., approximately 27 minutes after docking, the telemetry signal from thruster 8 indicated ON for 4.9 seconds, OFF for 4.0 seconds, then ON for the remainder of the flight. The spacecraft/GATV combination was being controlled by the GATV Attitude Control System (ACS) at this time in Flight Control Mode 3. The system was gyrocompassing, in-plane, with geocentric (GEO) rate ON. The OAMS attitude control power was OFF, the ACME mode select switch was in PULSE, and the IMU was in ORB RATE. In this configuration, the ACME is incapable of transmitting valid firing commands to the thrusters.

Figure 5.1.5-19 contains the sequence of significant events as they occurred during the anomaly plotted in relation to spacecraft roll rate. As indicated, the initial telemetry firing indications from thruster 8 were correct, in that the dynamic response matched the disturbance which should have been present. The first corrective action was taken, with the ACME in pulse mode, 11.5 seconds after the anomaly occurred. This mode was ineffective due to the short firing times associated with pulsed operation; therefore, DIRECT and then RATE COMMAND were selected with more success. In fact, while in the rate-command mode, the rates were essentially reduced to zero. At 7:02:37.4 g.e.t., the dynamic responses indicate that thruster 8 stopped firing, although the telemetry indication remained ON. Low grade accelerations were present which were representative of those which can result from a thruster expelling oxidizer only. Accelerations of this order could also have been obtained from the GATV ACS (for which no telemetry data are available), but in a very unlikely set of conditions. During this period, several firing commands were sent to thruster no. 8 with no response. At 7:07:20.3 g.e.t., after an interval of

4 minutes 42.9 seconds, the original disturbance returned, indicating that thruster no. 8 was again operating at or near full thrust. From this time until the spacecraft was separated from the GATV at 7:15:12.3 g.e.t., the disturbance was present and, as seen in figure 5.1.5-20, was controllable in the direct mode. The pitch and yaw rates were held to low values during this period; however, the roll rate did exceed 10 deg/sec for a total of approximately 100 seconds in six 15-to-20 second intervals. Each time the roll rate exceeded 10 deg/sec, it was quickly brought back to near zero using the direct control mode, and did not exceed 20 deg/sec at any time prior to undocking. The status of the GATV ACS throughout this period is uncertain except for one data point at 7:12:38.6 g.e.t., but appears from combined-vehicle acceleration calculations to have been cycled ON and OFF several times. The selection of redundant ACME logic and secondary thruster valve-driver circuitry, as reported by the crew, cannot be corroborated because these functions were not telemetered; however, the data does indicate that ACME bias power was turned off momentarily at 7:13:38.8 g.e.t. There was no telemetry channel to indicate the utilization of the yaw/pitch roll-logic switch or the motorized fuel shut-off valves; however, by analyzing the combination of thruster firings in response to roll hand-controller commands, it was determined that the pitch logic was not selected for roll control during the anomaly period.

Separation from the GATV occurred at 7:15:12.3 g.e.t. with thrusters 11 and 12 firing for 6.6 seconds. Rates at this time were +3, -5, and -2 deg/sec in pitch, roll, and yaw, respectively. After separation, moderate hand-controller activity was present, although the direct mode was not sufficient to contain the roll rate. At 7:15:44.7 g.e.t., the ACME bias power was inadvertantly removed, disabling the control system, and the roll rate increased to 296 deg/sec over the next three minutes, due to the uncontrolled firing of thruster 8, although short periods of intermittent or degraded thruster 8 performance appeared to exist. It is clear that the crew was not aware that ACME bias power was off because significant hand-controller activity is evident during this period. As noted in figure 5.1.5-19, the RCS squib valves were actuated at 7:16:25.1 g.e.t., but no RCS thrusters were fired until 7:19:03.8 g.e.t., probably because the ACME-DIRECT switch was in the ACME position with the ACME bias power off. When the ACME-DIRECT switch was apparently placed in the DIRECT position, RCS control was normal. The disturbance torque from thruster 8 ceased at 7:18:15.7 g.e.t. when the OAMS attitude-thruster circuit breakers were opened. Control was regained using the RCS in DIRECT-DIRECT. Subsequent checks of the OAMS thruster 8 circuit breaker and the RCS using ACME modes indicated correct ACME performance; in addition, telemetry indications and fault characteristics lead to the conclusion that the malfunction probably was external to the control system. (See sections 5.1.7 and 5.1.8 for further discussions of the flight-control anomaly.)

The OAMS thrusters, with the exception of number 8, were utilized in platform and pulse modes for the preretrofire platform alignment with no difficulty. An RCS control mode check in rate-command and reentry rate-command modes was satisfactorily performed and the rate-command mode was utilized during retrofire (both rings) with minimal attitude errors resulting (1.5 degrees, 1.5 degrees, and 4.0 degrees, in pitch, yaw, and roll, respectively).

Following retrofire, the RCS A-ring was turned off and the control mode switched to PULSE. At 400K feet altitude the reentry rate command mode was energized and used for 6 minutes 15 seconds. During this time, at approximately the 3g level, the RCS A-ring was turned on and the B-ring turned off. Three minutes later, at drogue parachute deployment, the B-ring was turned on again and the control system was switched into the orbit rate-command mode. The system remained in this configuration with both RCS rings on until the spacecraft was powered down. The maximum rates observed prior to drogue parachute deployment were approximately 5 deg/sec in pitch and yaw, slightly less than observed on previous flights. The control parameters during a representative portion of the reentry phase are presented in figure 5.1.5-21. A separate plot comparing the roll-rate command with the roll rates achieved during the period of reentry which contained maximum roll rates is included as figure 5.1.5-22. These data indicate that the reentry rate-command system was responding properly to hand-controller inputs.

5.1.5.3.2 Horizon sensors: The horizon sensors, both primary and secondary, performed satisfactorily and the crew reported no difficulties. As on previous missions, losses of track were experienced during sunset periods as a result of sun interference. Numerous losses of track occurred during station keeping with the GATV, caused by the relative attitudes of the two vehicles with respect to the horizon. The primary sensor was turned off prior to docking, turned on at 7 hours 45 minutes g.e.t., turned off before retrofire at 9 hours 58 minutes g.e.t., and remained off for the remainder of the mission. The secondary sensor was turned on for evaluation, performed satisfactorily, turned off after the first 42 minutes of flight, and remained off for the remainder of the mission.

TABLE 5.1.5-I.- SPACECRAFT GUIDANCE AND CONTROL SUMMARY CHART

Time from lift-off, sec		Event	Component status					Remarks
Planned	Actual RGS		ACME	Computer	IMU	Horizon sensor	Radar	
0.00	0.00	Lift-off	IGS backup	Ascent	Free	Primary	Off	16:41:02.389 G.m.t.
8.48	8.48	Start roll program	IGS backup	Ascent	Free	Primary	Off	
20.48	20.47	Stop roll program	IGS backup	Ascent	Free	Primary	Off	
23.04	23.04	Start pitch program 1	IGS backup	Ascent	Free	Primary	Off	
88.32	88.24	Stop pitch program 1 Start pitch program 2	IGS backup	Ascent	Free	Primary	Off	
104.96	104.76	No. 1 gain change	IGS backup	Ascent	Free	Primary	Off	
105.00	105.00	No. 1 IGS update	IGS backup	Ascent	Free	Primary	Off	
119.04	118.87	Stop pitch program 2 Start pitch program 3	IGS backup	Ascent	Free	Primary	Off	
145.00	145.00	No. 2 IGS update	IGS backup	Ascent	Free	Primary	Off	
153.85	154.615	BECO	IGS backup	Ascent	Free	Primary	Off	
162.56	161.72	Stop pitch program 3	IGS backup	Ascent	Free	Primary	Off	
168.35	168.40	First guidance command	IGS backup	Ascent	Free	Primary	Off	
336.73	337.52	SECO	IGS backup	Ascent	Free	Primary	Off	
356.75	365.66	Spacecraft separation	Direct, then rate command	Ascent	Free	Primary	Off	

TABLE 5.1.5-I.- SPACECRAFT GUIDANCE AND CONTROL SUMMARY CHART - Continued

G.e.t., hr:min:sec		Event	Component status					Remarks
Planned	Actual RGS		ACME	Computer	IMU	Horizon sensor	Radar	
	00:06:05.7	Spacecraft—GLV separation	Direct, then rate command	Ascent	Free	Off	Off	1. Aft-firing thrusters 9 and 10 fire from 00:06:02.9 until 00:06:10.9 g.e.t. (Δt = 8.0 sec). 2. Roll to heads-up position begins at 06:19.1. Completed at 00:07:07 g.e.t.
	00:28:21	Horizon sensor check	Platform	Prelaunch	SEF	SEC		Used secondary sensor for 13 minutes with nominal performance.
00:50:00		Accelerometer bias check	Pulse	Catchup	Orbit rate	Primary	Off	
01:19:37		Platform alignment	Platform	Prelaunch	SEF	Primary	Off	One minute prior to height-adjust maneuver, alignment errors were +0.9° and -0.3° in pitch and roll.
01:34:37	01:34:36.2	Height-adjust maneuver	Platform	Catchup	Orbit rate	Primary	Off	Forward firing-thrusters 11 and 12 fired for 6.3 sec.
02:03:25		Platform alignment	Platform and pulse	Prelaunch	SEF	Primary	Off	One minute prior to phase-adjust maneuver, pitch and roll alignment errors were -0.4° and +0.2°.
02:18:25	02:18:25.6	Phase-adjust maneuver	Rate command	Catchup	Orbit rate	Primary	Off	Aft-firing thrusters 9 and 10 fired 68.4 sec.
02:45:50	02:45:52.8	Plane-change maneuver	Rate command	Catchup	Orbit rate	Primary	Off	Aft-firing thrusters 9 and 10 fired 35.7 sec. Yaw = 90°.
02:50:00		Platform alignment	Platform	Prelaunch	SEF	Primary	Off	Pitch and roll alignment errors 1 minute prior to height-adjust maneuver were -1.5° and +0.4°.
03:00:00	03:00:00	Radar to standby	Horscan	Prelaunch	SEF	Primary	Standby	
03:03:41	03:03:42.2	Height-adjust maneuver	Platform	Catchup	Orbit rate	Primary	Standby	
03:07	03:05	Radar on	Platform	Prelaunch	Orbit rate	Primary	On	
03:27:35		Platform alignment		Rendezvous	SEF	Primary	On	

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TABLE 5.1.5-I.- SPACECRAFT GUIDANCE AND CONTROL SUMMARY CHART - Continued

G.e.t., hr:min:sec		Event	Component status					Remarks
Planned	Actual RGS		ACME	Computer	IMU	Horizon sensor	Radar	
	03:31:18	Radar lock-on	Platform	Rendezvous	Orbit rate	Primary	On	Range = 180 nautical miles.
03:47:35	03:48:09.7	Circularization N_{SR} maneuver	Rate command	Catchup	Orbit rate	Primary	On	Aft-firing thrusters 9 and 10 fired 78.6 sec.
	05:13:56	Platform alignment	Pulse	Rendezvous	SEF	Primary	On	Pitch and roll alignment errors 1 minute prior to TPI were +0.8° and +0.8°.
	05:14:55.7	Terminal-phase initiation	Rate command	Rendezvous	Orbit rate	Primary	On	
	05:27:26.0	First correction maneuver	Rate command	Rendezvous	Orbit rate	Primary	On	
	05:39:19.9	Second correction maneuver	Rate command	Rendezvous	Orbit rate	Primary	On	
	05:43:08.9	Terminal phase finalization (TPF)	Rate command, pulse	Rendezvous	Orbit rate	Primary	On	TPF consisted of several maneuvers, the last of which was done in PULSE. All others in RATE COMMAND.
	05:58:57	Formation flying	Pulse, platform, direct, and rate command	Catchup	Orbit rate	Primary	On	
		Platform alignment		Catchup	BEF	Primary	Off	
06:25 to 06:35	06:33:16	Docking	Rate command		Orbit rate	Primary	Off	
	07:00:26.7	Thruster 8 fails ON	Pulse (OAMS off)	Prelaunch	Orbit rate	Off	Off	Thruster 8 indicated on for 4.9 sec, then off for 4.0 sec, then on continuously.
	07:15:12.3	Undocking	Direct		Orbit rate	Off	Off	Forward-firing thrusters fire for 6.6 sec. NOTE: See Section 5.1.5.3.1 and figure 5.1.5-15 for details during this period.

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TABLE 5.1.5-I.- SPACECRAFT GUIDANCE AND CONTROL SUMMARY CHART - Concluded

G.e.t., hr:min:sec		Event	Component status					Remarks
Planned	Actual RGS		ACME	Computer	IMU	Horizon sensor	Radar	
	07:25:30	Spacecraft rates stabilized	RCS-Direct	Prelaunch	Orbit rate	Off	Off	
	07:28:12	Check mode of OAMS thrusters	Pulse	Prelaunch	Orbit rate	Off	Off	Thruster 8 starts to fire when circuit breaker is closed for 1 sec.
	07:39:14	ATMU power on	Pulse	Prelaunch	Orbit rate	Off	Off	Module IV-A loaded in computer.
	09:01:40	Control mode check	Pulse	Prelaunch	Orbit rate	Off	Off	Checks pulse, direct, and rate command using RCS thrusters.
	09:16:55	Control mode check	Reentry rate command	Prelaunch	Orbit rate	Off	Off	Checks reentry rate command using OAMS thrusters.
	09:19:17	Horizon sensor on	Pulse	Prelaunch	Orbit rate	Primary	Off	Operation normal.
	09:20:00	Platform alignment	Pulse	Prelaunch	BEF	Off	Off	One minute prior to retrofire, alignment errors were +0.2° in pitch and roll.
	09:52:17	Control mode check	Reentry rate command	Prelaunch	Orbit rate	Primary	Off	Checks reentry rate command, direct, and rate command using RCS ring A thrusters.
10:04:47	10:04:46.6	Retrofire	Rate command	Reentry	Free	Primary	Off	
10:26:49	10:26:48.6	400K feet	Pulse	Reentry	Free	Off	Off	
	10:30:21.1	Change to reentry rate command	Reentry rate command	Reentry	Free	Off	Off	
10:36:24	10:36:46.9	Drogue deploy	Rate command	Reentry	Free	Off	Off	
	10:41:26	Landing	Rate command	Reentry	Free	Off	Off	

TABLE 5.1.5-II.- RESULTS OF INCREMENTAL VELOCITY ADJUST ROUTINE (IVAR)

	Actual	Reconstructed
Velocity to be applied at apogee, V_{gp}, ft/sec	0.102	0.098
Velocity to be applied at perigee, V_{ga}, ft/sec	-9.586	-9.584
Radial velocity, V_p, ft/sec	-0.398	-0.434
Inertial velocity, V, ft/sec	25 747.953	25 747.956
IVI fore-aft, $V_{X_{S/C}}$, ft/sec	-10.07	-9.48
IVI right-left, $V_{Y_{S/C}}$, ft/sec	18.65	18.07
IVI up-down, $V_{Z_{S/C}}$ ft/sec	-11.62	-13.47
Time to apogee, T_{AP}, sec	3071.78	3072.55

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TABLE 5.1.5-III.- ASCENT IGS AND TRACKING SYSTEM ERRORS

Error source	Specification value	Engineering estimates				Error coefficient Recovery Program estimates			
		Error	Velocity error, ft/sec			Error	Velocity error, ft/sec		
			X	Y	Z		X	Y	Z
Constant drift	-0.3 deg/hr	deg/hr				deg/hr			
X_p-gyro		0.08	0	N	-0.8	0	0	0	0
Y_p-gyro		-0.1	-0.1	-2.7	0	0.06 ± 0.4	N	1.6	0
Z_p-gyro		0	0	0	0	0.15 ± 1.0	N	0	1.3
g-sensitive drift	0.5 deg/hr/g	deg/hr/g							
X_p-gyro spin-axis unbalance		-0.72	0	-0.2	6.8				
Y_p-gyro spin-axis unbalance		N	0	0	0				
Z_p-gyro spin-axis unbalance		0.1	N	0	0.8				
X_p-gyro input-axis unbalance		-0.22	0.1	0	-2.7				
Y_p-gyro input-axis unbalance		0.12	0.11	3.2	0				
Z_p-gyro input-axis unbalance		+0.08	N	0	+2.7				

N = negligible

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TABLE 5.1.5-III.- ASCENT IGS AND TRACKING SYSTEM ERRORS - Continued

Error source	Specification value	Engineering estimates				Error coefficient Recovery Program estimates			
		Error	Velocity error, ft/sec			Error	Velocity error, ft/sec		
			X	Y	Z		X	Y	Z
Accelerometer bias	300 ppm	ppm				ppm			
X_p		44	0.18	-1.0	0				
Y_p		10	0	0	-0.1				
Z_p		-100	0	-1.1	0				
Accelerometer scale factor	360 ppm								
X_p		200	4.9	0	0	195 ± 60	4.8	0	0
Y_p		N	0	0	0	0	0	0	0
Z_p		-150	0	+1.0	0	330 ± 156	0	-2.2	0
Misalignments									
Azimuth misalignment	60 sec	43 sec	0	0	5.2	48.5 ± 18 sec	0	0	5.8
Pitch misalignment	100 sec	30 sec	0	-3.6	0	-3 ± 26 sec	0	-0.3	0
Time bias		0.029 sec	6.4	1.7	0	28 + 5 sec	6.2	1.6	0
IGS time scale factor	50 ppm	-100 ppm	-7.5	-2.0	0	-95 ± 49 ppm	-7.2	-1.9	0
Total velocity error			5.2	-4.7	11.8		3.8	-1.2	7.1

N = negligible

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TABLE 5.1.5-III.- ASCENT IGS AND TRACKING SYSTEM ERRORS - Concluded

External tracker errors						
System	Range bias, ft	P-bias, ft	Q-bias, ft	Azimuth, radians	Elevations, radians	Refraction, n units
GE Mod III (final)	-70 ± 20	N/A	N/A	N	N	10 ± 10
MISTRAM 100K	3 ± 2.5	2.2 ± 2.0	0	N/A	N/A	-30 ± 15

N = negligible

N/A = not applicable

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TABLE 5.1.5-IV.- ORBIT INJECTION PARAMETERS AT SECO + 20 SECONDS

Data source	Inertial velocity, ft/sec	Inertial flight-path angle, deg	Inertial velocity components (computer coordinates), ft/sec		
			X	Y	Z
Flight plan	25 728	-0.001	25 311	4610	34
IGS	25 740	-0.04	25 323	4620	8
Preliminary best-estimate trajectory	25 737	-0.03	25 318	4625	-4
MISTRAM 10K	25 736	-0.02	25 318	4620	-3
MISTRAM 100K	25 734	-0.02	25 318	4612	-4
GE Mod III/Final	25 737	-0.02	25 318	4624	-5
GE Mod III (real time)	25 745	-0.16			
MISTRAM IP	25 741	-0.13			

TABLE 5.1.5-V.- GUIDANCE ERRORS AT SECO + 20 SECONDS

Error	Position, ft			Velocity, ft/sec		
	X	Y	Z	X	Y	Z
IMU	900 ± 100	170 ± 100	1030 ± 100	5.2 ± 1.0	-4.5 ± 2.0	11.8 ± 2.0
Navigation	+20	-50	-15	-0.2	-0.4	-0.4
Total guidance	920 ± 100	120 ± 100	1015 ± 100	5.0 ± 1.0	-4.9 ± 2.0	11.4 ± 3.0

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TABLE 5.1.5-VI.- PLATFORM ALIGNMENT ACCURACY DURING MAJOR MANEUVERS

Maneuver	Time, g.e.t., hr:min:sec	Alignment accuracy (gimbal angle minus horizon sensor output)		Control mode at time of maneuver
		Pitch, deg	Roll, deg	
Height adjust	1:34:36.4	0.9	-0.3	Platform
Phase adjust	2:18:25.8	-0.4	0.2	Rate command
Vernier height adjust	3:03:42.2	-1.5	0.4	Platform
TPI	5:14:55.7	0.8	0.8	Rate command
Retrofire	10:04:46.6	0.2	0.2	Pulse, platform

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TABLE 5.1.5-VII.- TRANSLATION MANEUVERS

Event	Time, g.e.t., hr:min:sec	Components			Total ΔV, ft/sec	Planned ΔV, ft/sec
		ΔV_X, ft/sec	ΔV_Y, ft/sec	ΔV_Z, ft/sec		
Tail-off	0:05:37	82.34	21.19	4.17	85.13	
Separation	0:06:02.9	6.17	1.01	-0.22	6.25	5.0
Height adjust	1:34:36.4	-3.13	-0.19	0.11	3.14	2.9
Phase adjust	2:18:25.8	50.59	-0.33	-0.21	50.59	50.6
Plane change	2:45:52.8	-0.27	-0.29	-26.64	26.64	26.2
Vernier height adjust	33:03:42.2	2.27	0.19	-0.06	2.28	2.0
Coelliptic	3:48:09.7	57.39	22.33	0.06	61.58	61.2
TPI	5:14:55.7	22.32	-14.19	6.74	27.30	31.5
First correction	5:27:26.0	4.22	-14.40	-2.16	15.15	N/A
Second correction	5:39:19.9	-6.81	-3.13	-5.80	9.47	N/A
TPF	5:43:08.9	31.35	23.93	12.00	41.22	39.8
Retrofire	10:04:46.6	-292.66	113.81	-0.71	314.01	312.0

N/A = Not applicable

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TABLE 5.1.5-VIII.- COMPARISON OF COMPUTER TELEMETRY REENTRY PARAMETERS WITH POSTFLIGHT RECONSTRUCTION

Parameters	Time in mode = 1643.7 sec 400K ft			Time in mode = 2180.5 sec guidance termination		
	Telemetry	MAC	IBM	Telemetry	MAC	IBM
Radius vector, ft	21 300 862.0	21 302 957.0	21 301 040.0	20 973 800.0	20 978 217.0	20 974 004.0
Velocity, ft/sec	24 411.87	24 410.11	24 411.44	1918.38	1873.76	1917.49
Flight-path angle, deg. . .	-1.361	-1.356	-1.361	-32.996	-32.997	-32.990
Spacecraft heading, deg	88.56	88.56	88.56	110.20	110.19	110.20
Longitude, deg	102.99	102.98	102.99	136.00	135.96	136.00
Latitude, deg	28.82	28.82	28.82	25.08	25.09	25.08
Range to target, n. mi. . .	1772.59	1772.61	1772.52	1.39	2.30	1.36
Crossrange, n. mi.	7.73	7.70	7.70	1.25	1.35	1.27
Downrange, n. mi.	NA	NA	NA	-4.54	-2.47	-4.44
Predicted zero lift range, n. mi.	NA	NA	NA	3.97	3.80	3.97
Density altitude factor factor	NA	NA	NA	4.661	4.644	4.660
Bank command, deg	0.0	0.0	0.0	-90.0	-90.0	-90.0
Integration time, sec . . .	1327.496	1327.496	1327.497	1864.296	1864.296	1864.294

NA = Not available

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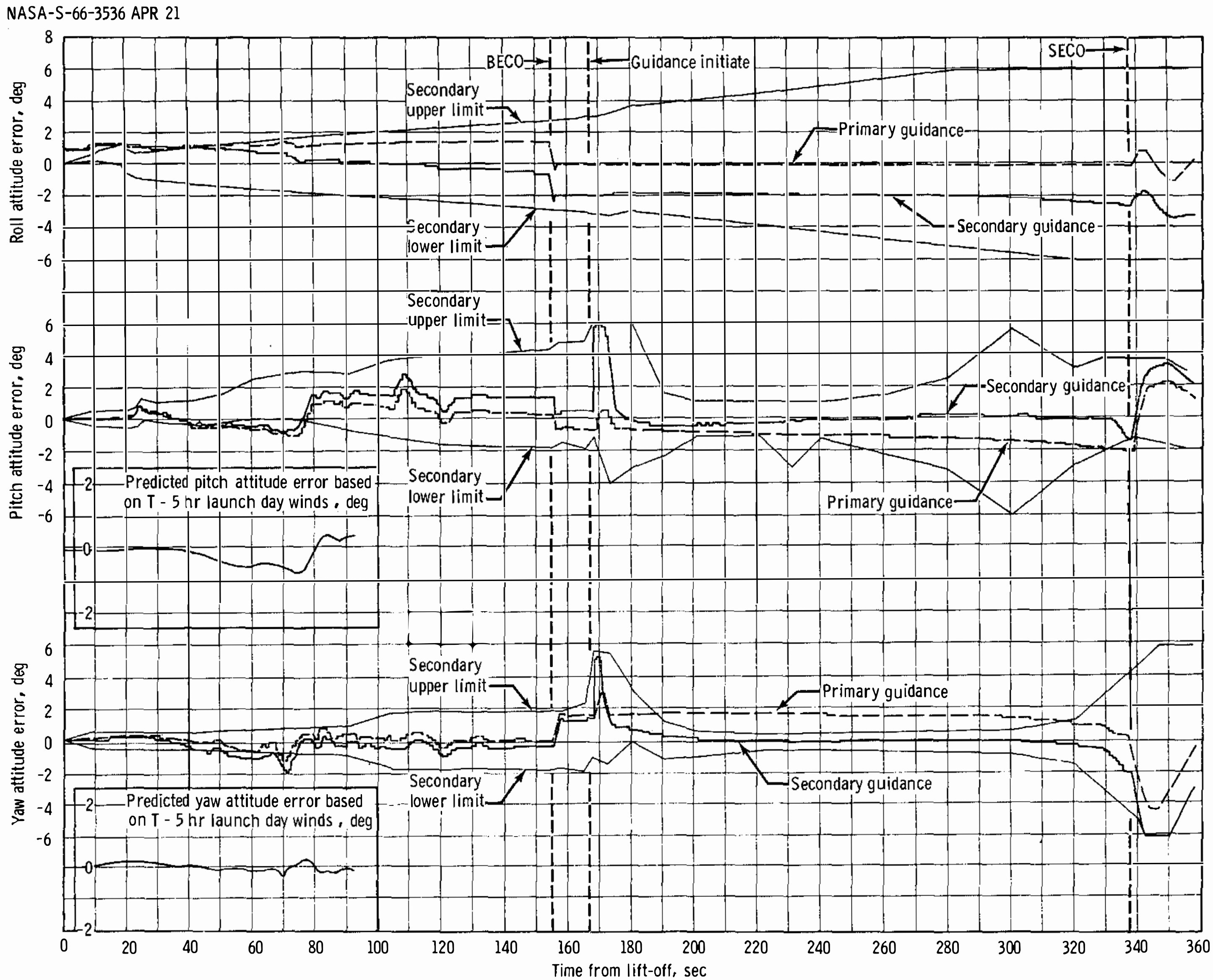

Figure 5.1.5-1. - Comparisons of launch vehicle and spacecraft steering errors

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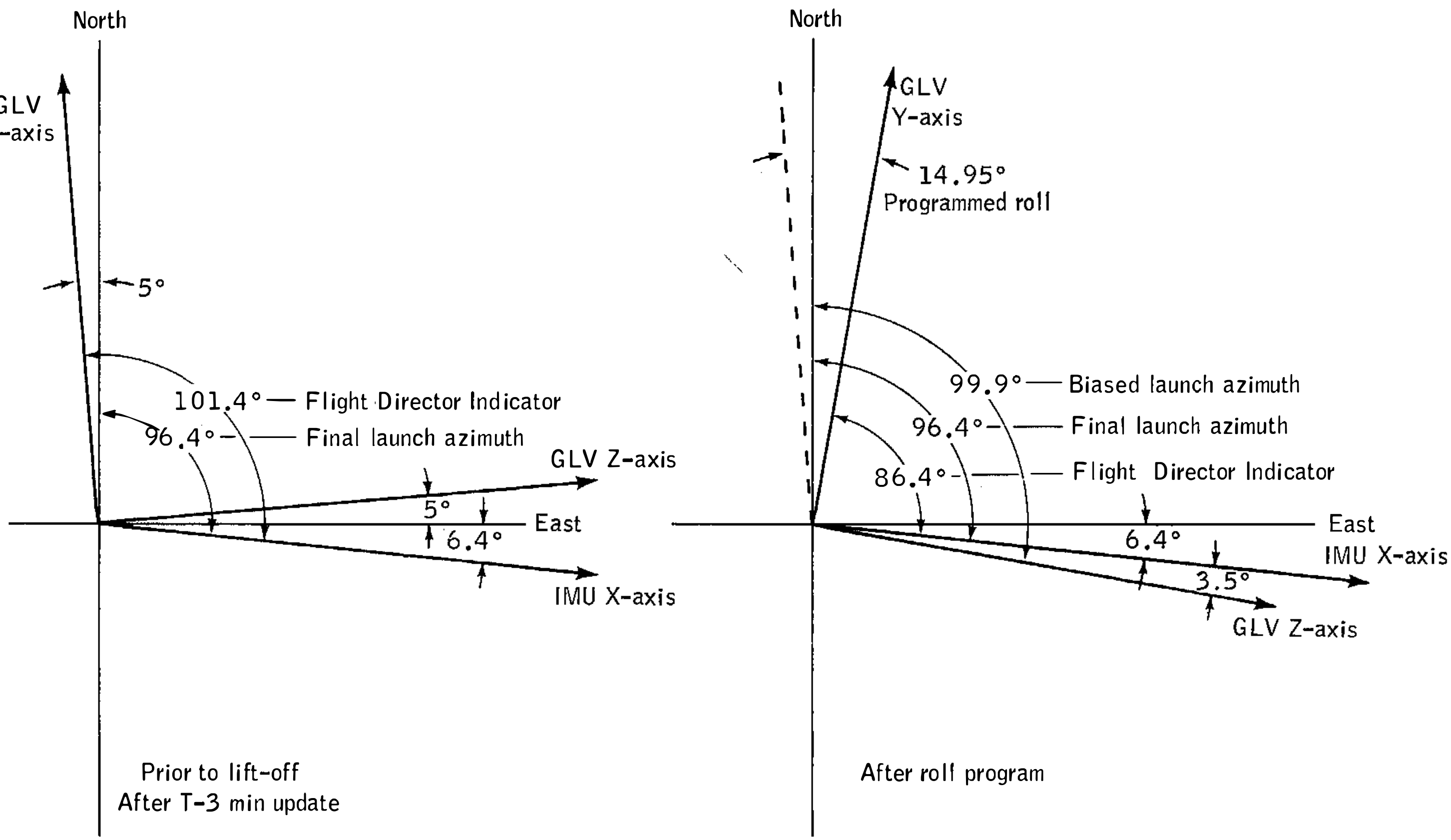

Figure 5.1.5-2. - Spacecraft IMU and GLV launch heading relation.

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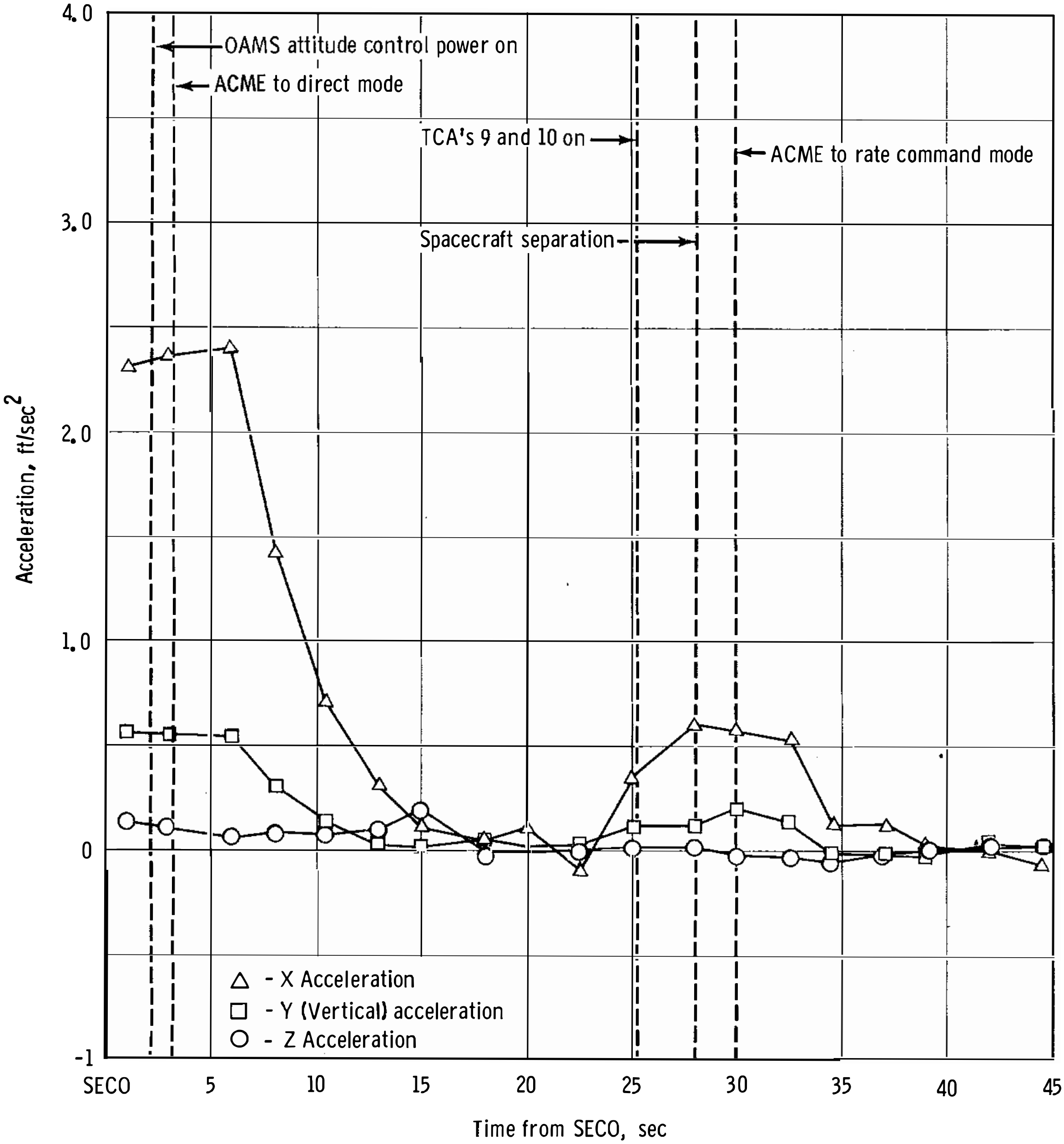

Figure 5.1.5-3. - Spacecraft acceleration measured after SECO.

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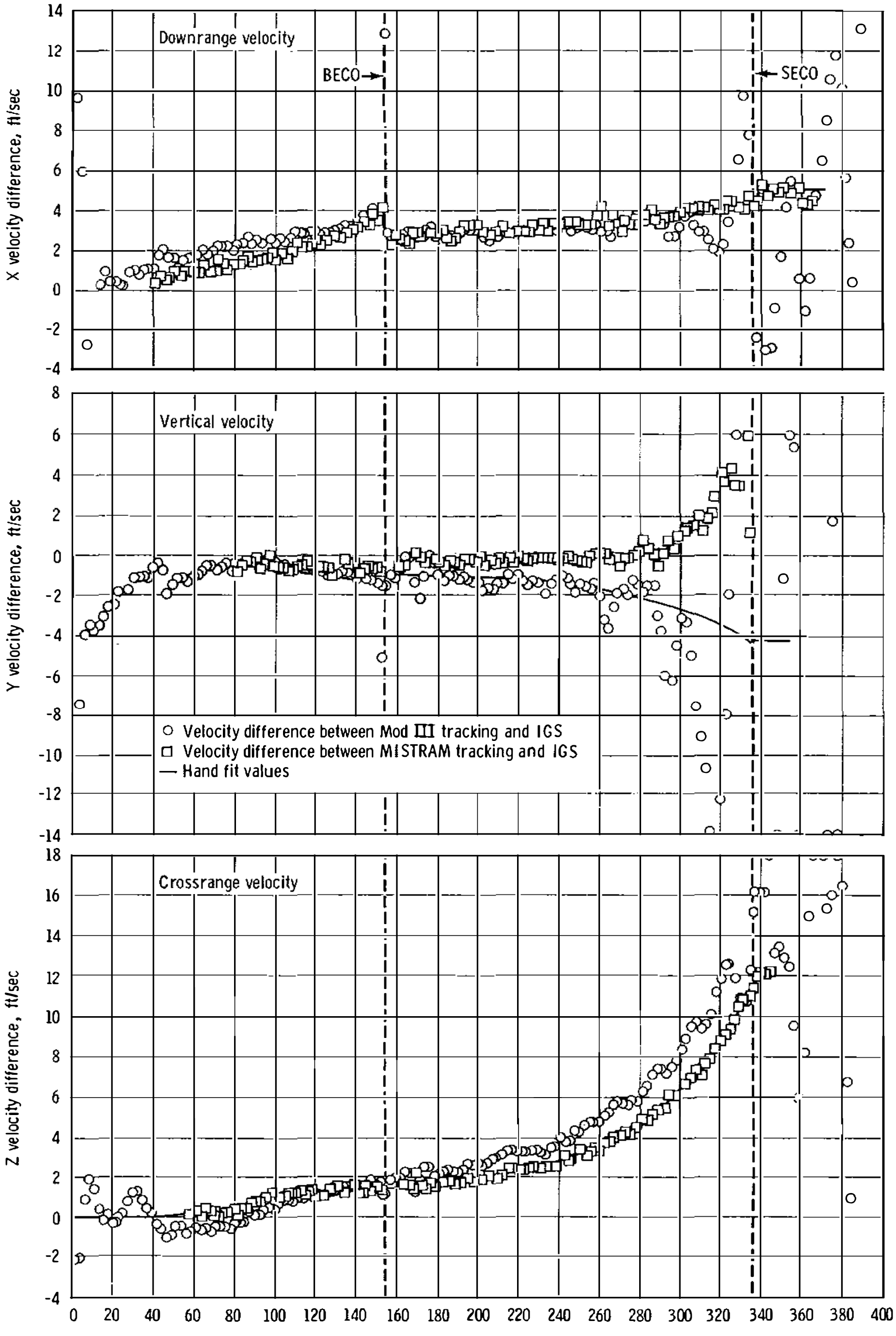

Figure 5.1.5-4. - Comparisons of spacecraft IGS and radar tracking velocities.

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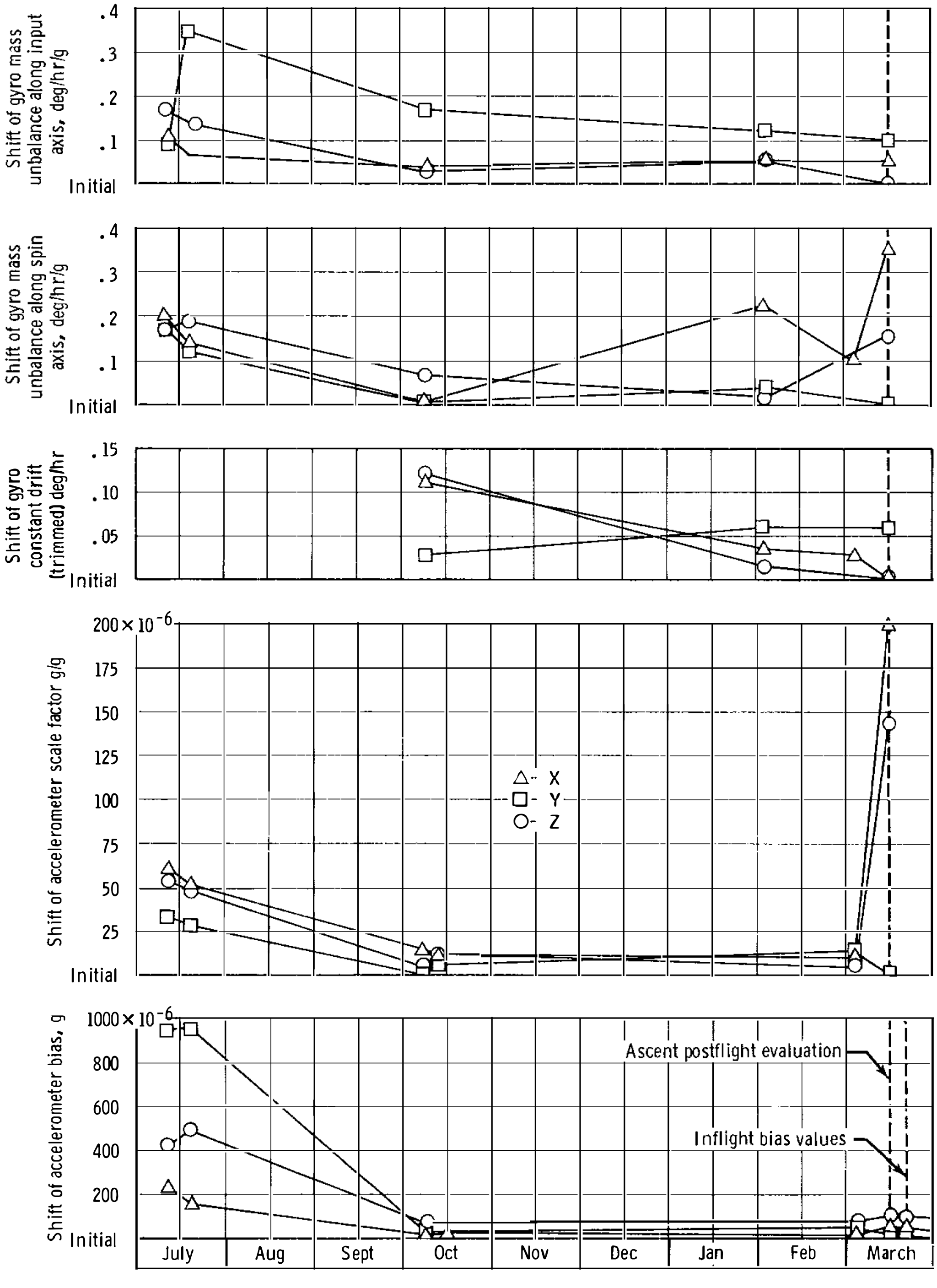

Figure 5.1.5-5. - IMU error coefficient history.

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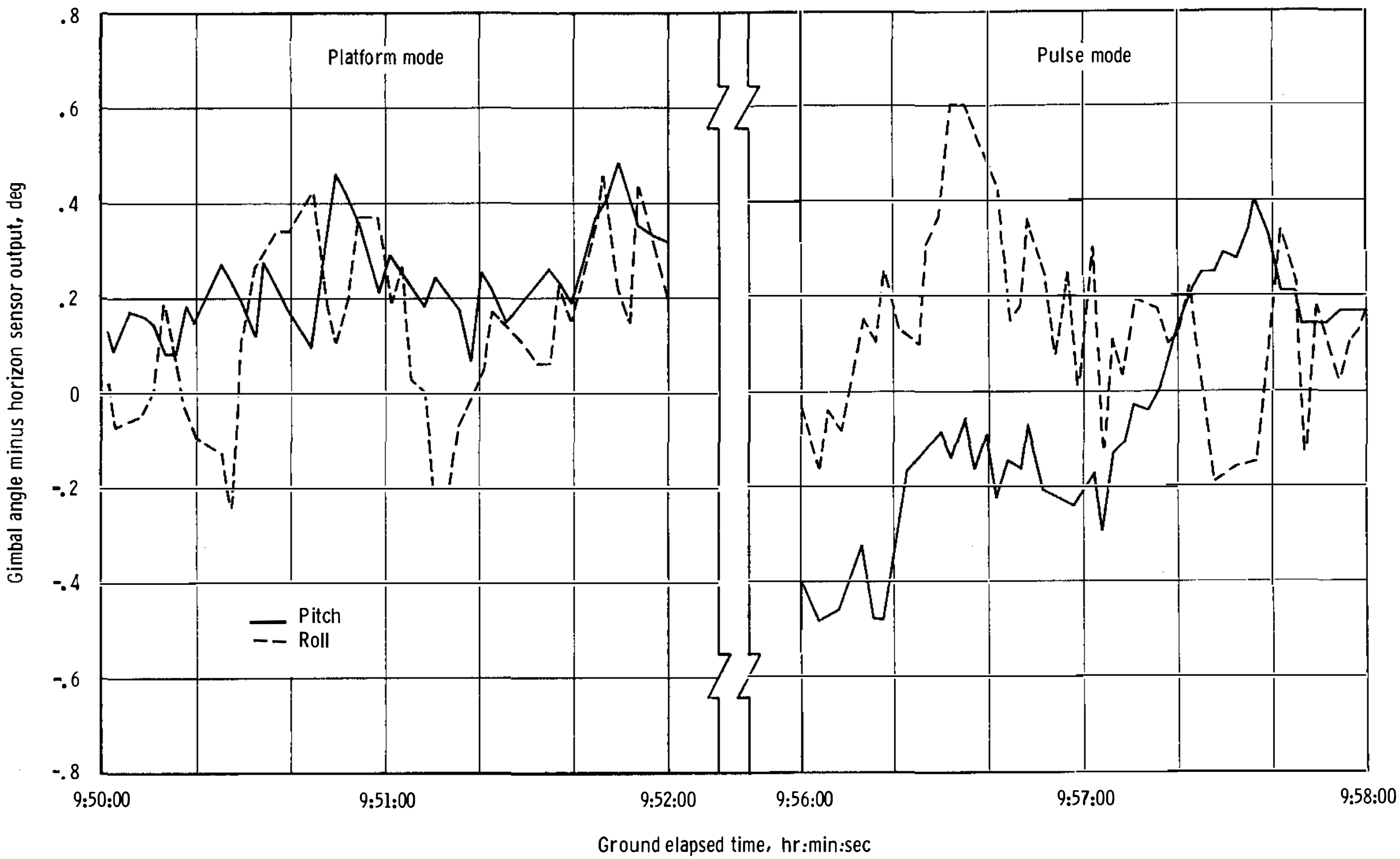

Figure 5.1.5-6.- Preretrofire platform alignment.

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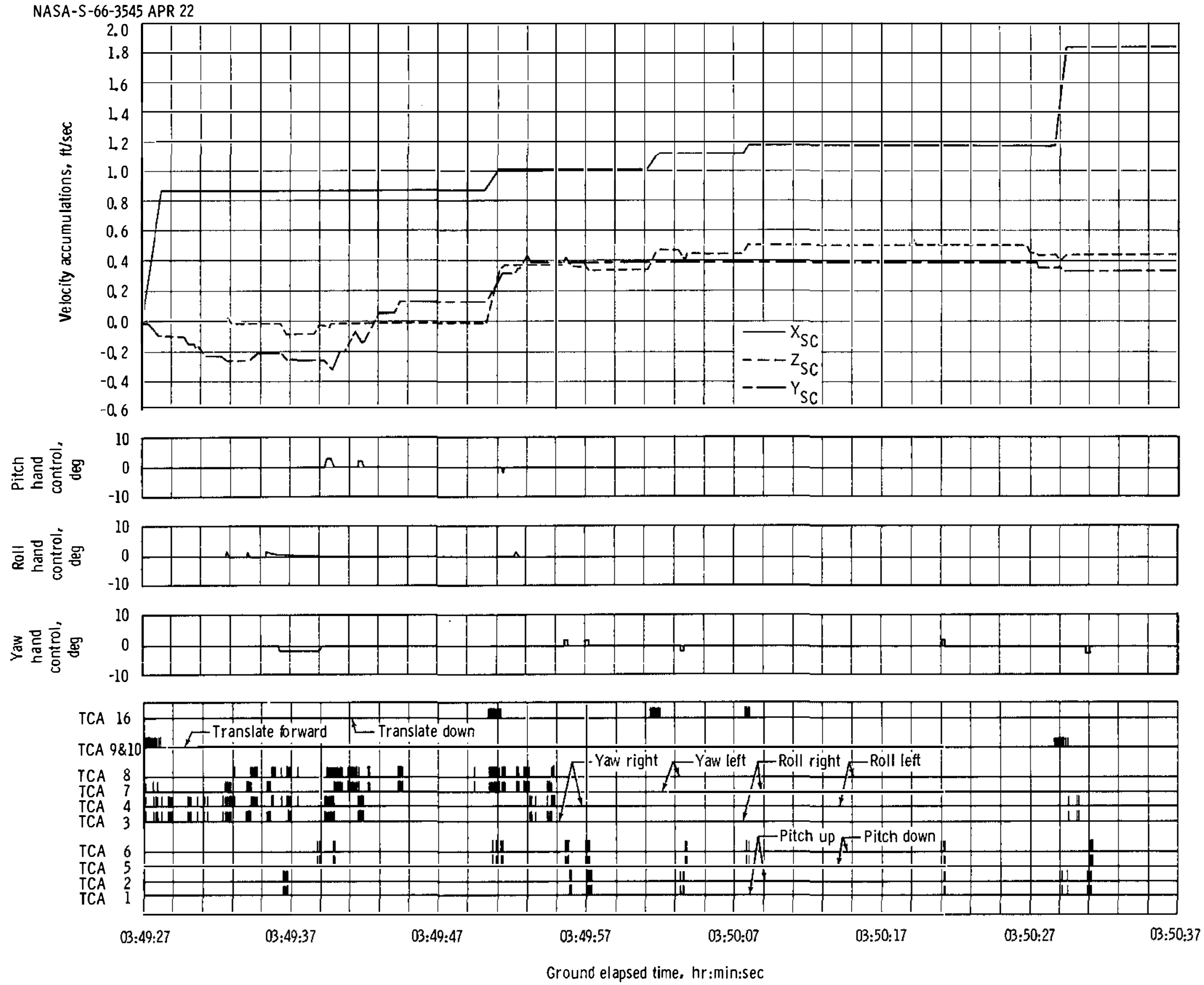

Figure 5.1.5-7. - Analysis of activity in reducing desired-velocity-change residuals.

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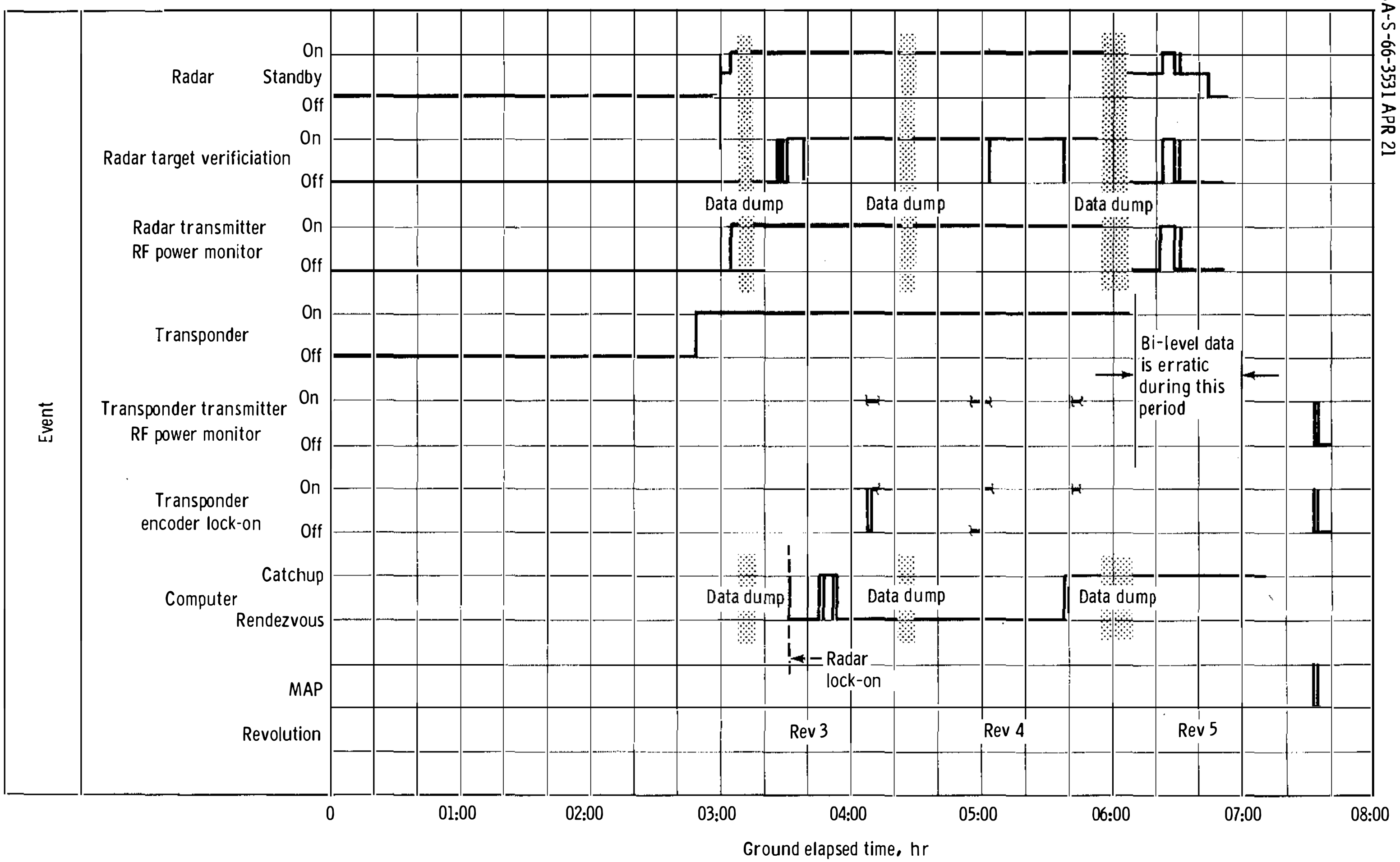

Figure 5.1.5-8. - Rendezvous radar system events.

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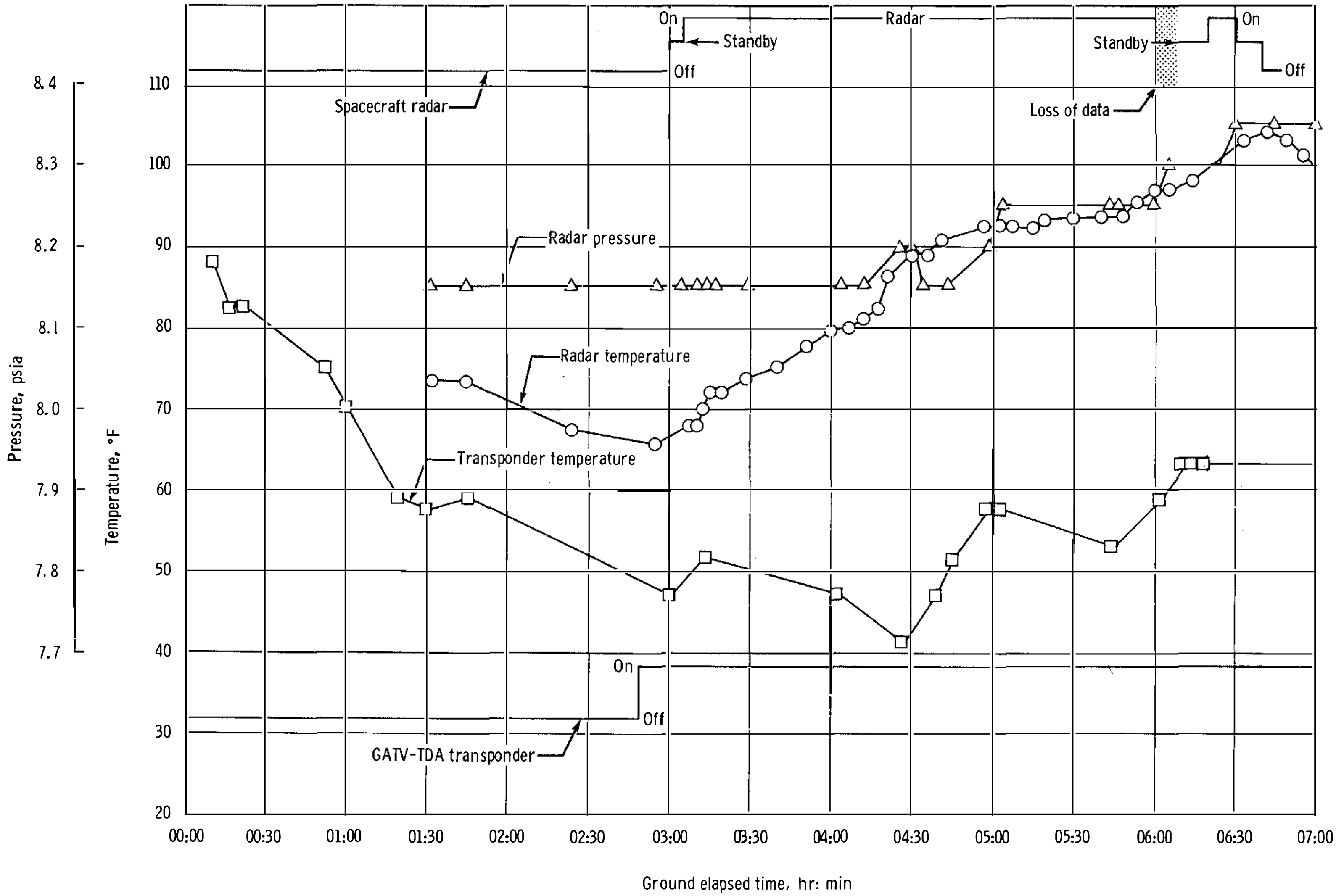

Figure 5.1.5-11. - Radar/transponder environmental parameters.

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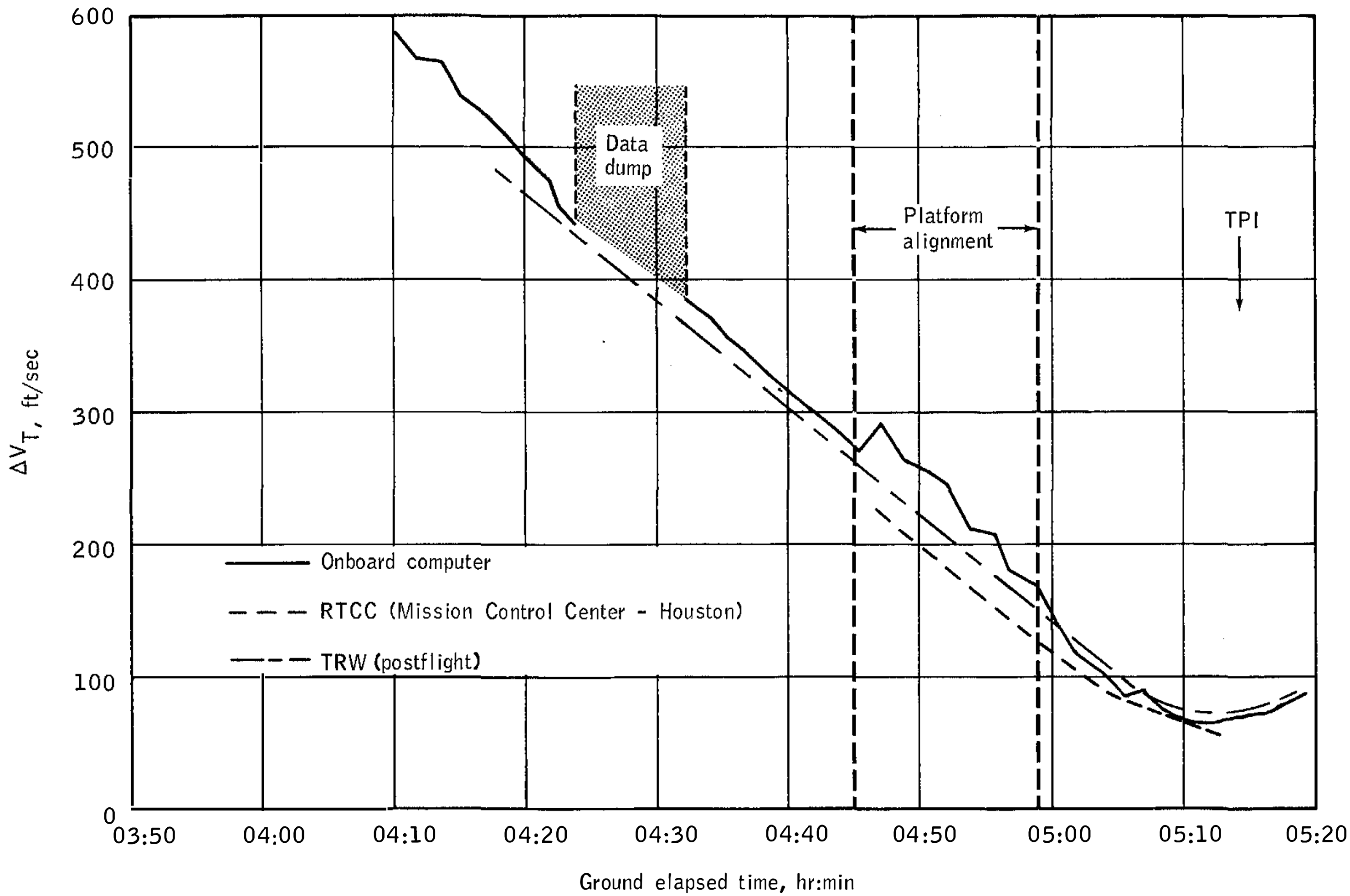

Figure 5.1.5-12. - Rendezvous total velocity compairson.

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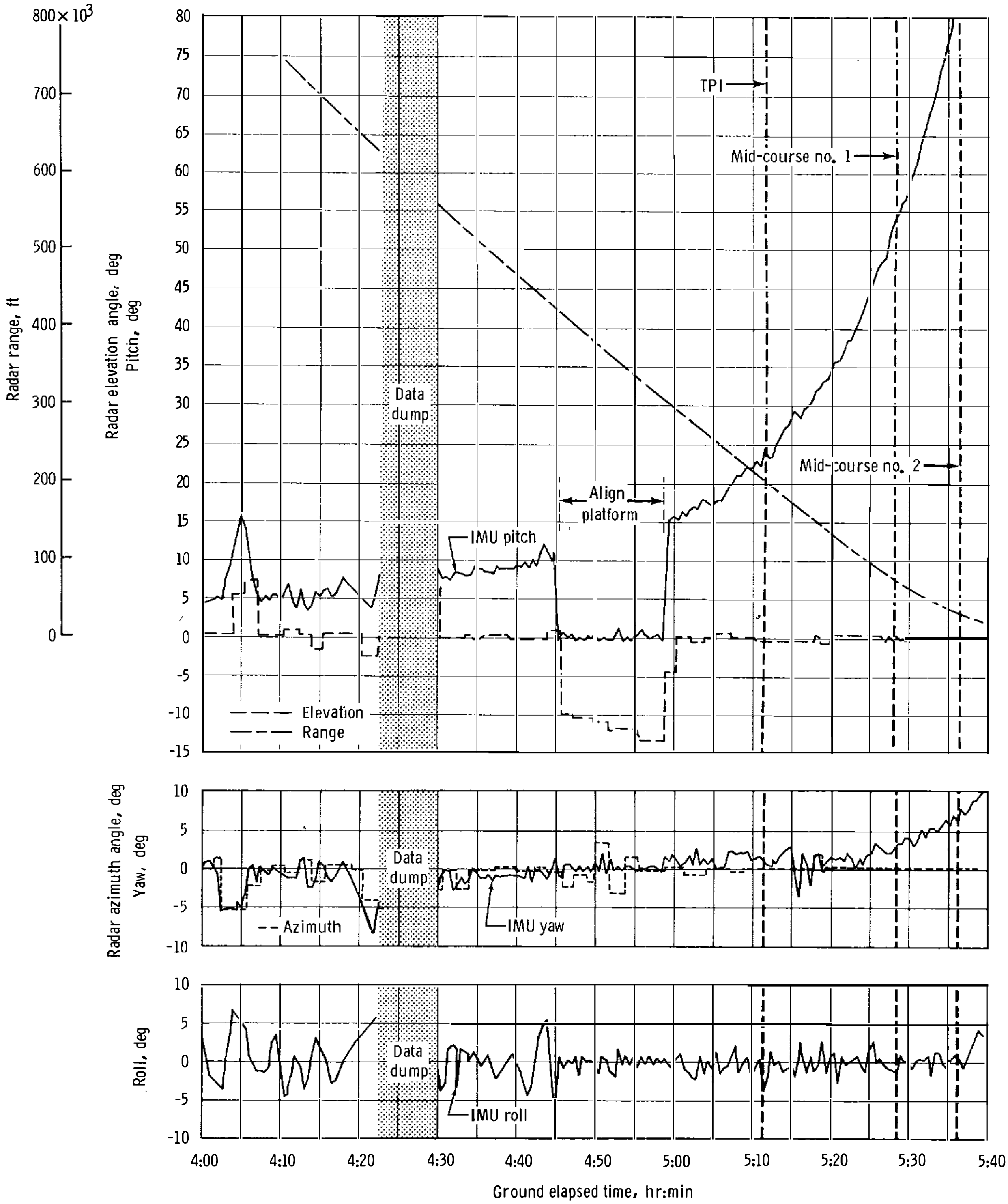

Figure 5.1.5-13. - Rendezvous phase time history.

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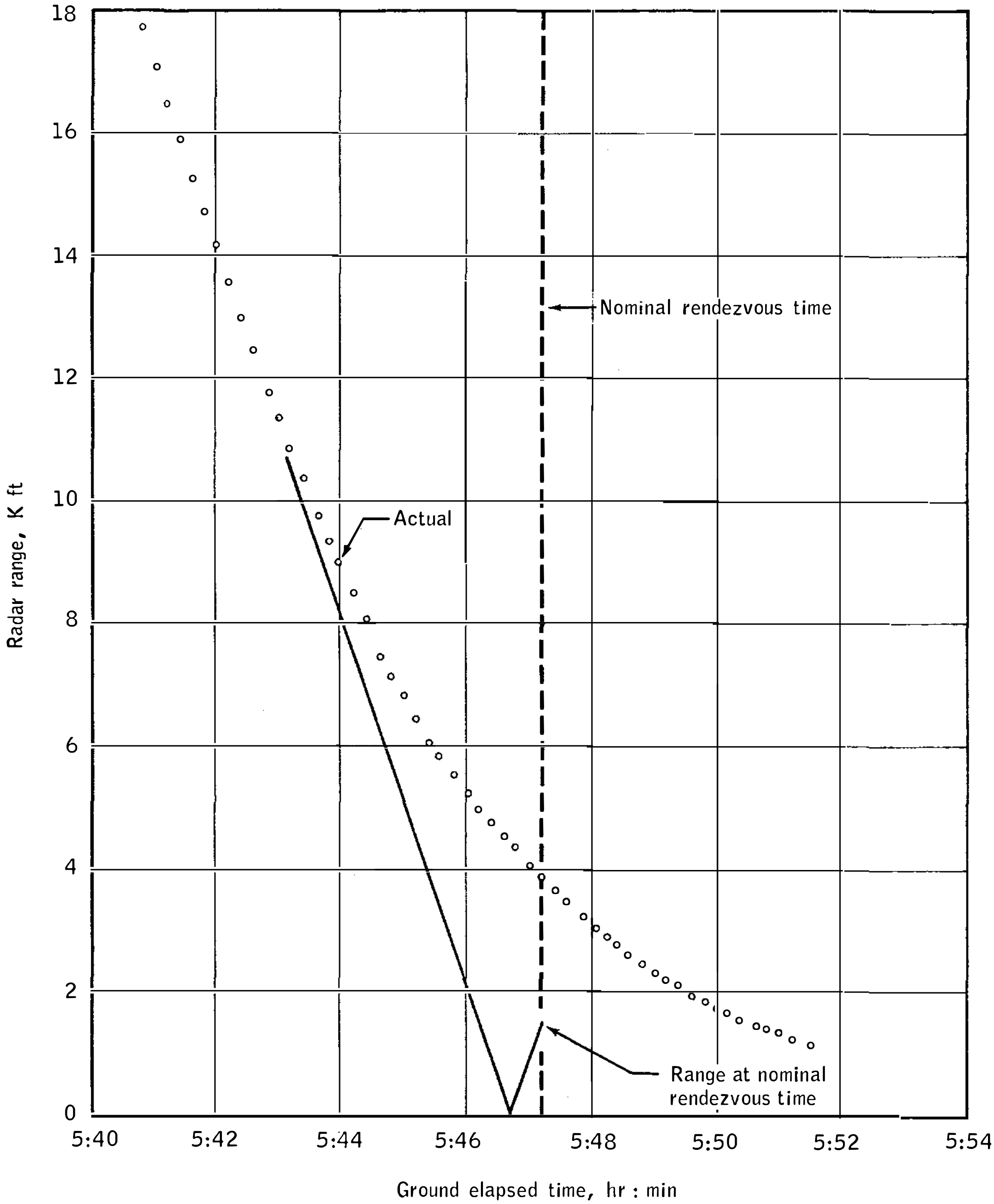

Figure 5.1.5-14. - Rendezvous approach phase.

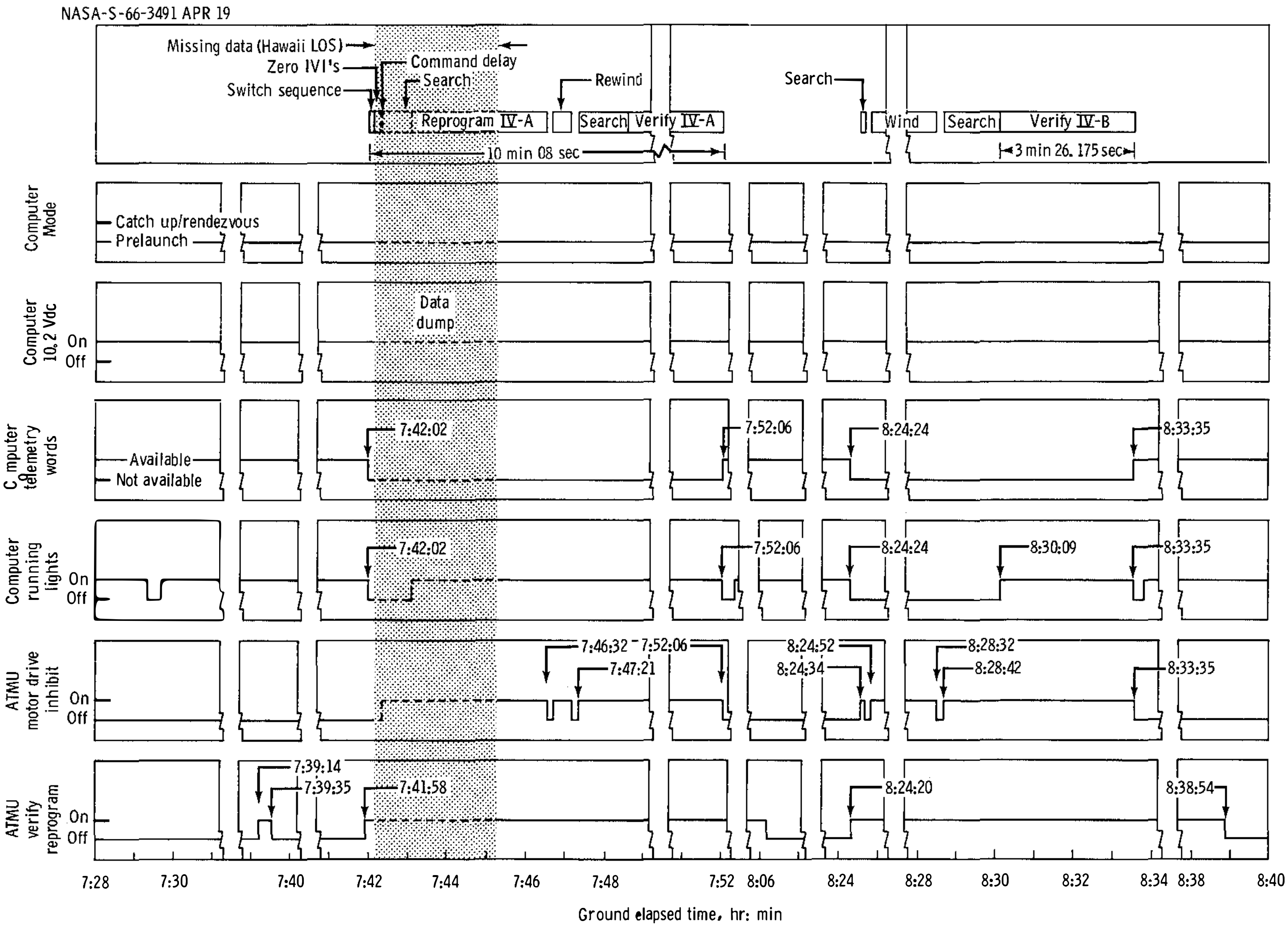

Figure 5.1.5-15. - Auxiliary Tape Memory Unit (ATMU) events, module IV reprogram and verify.

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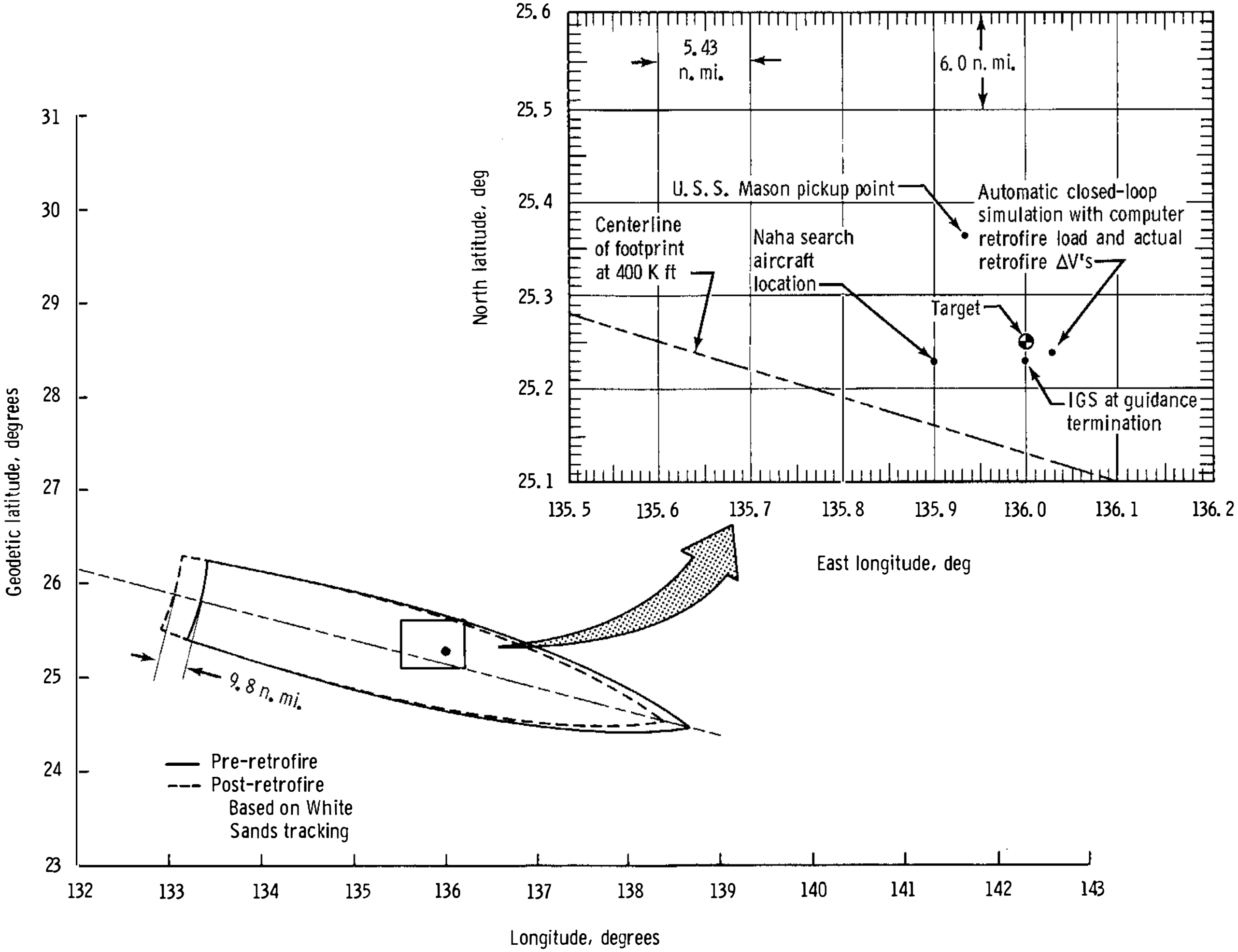

Figure 5.1.5-16. - Touchdown comparisons.

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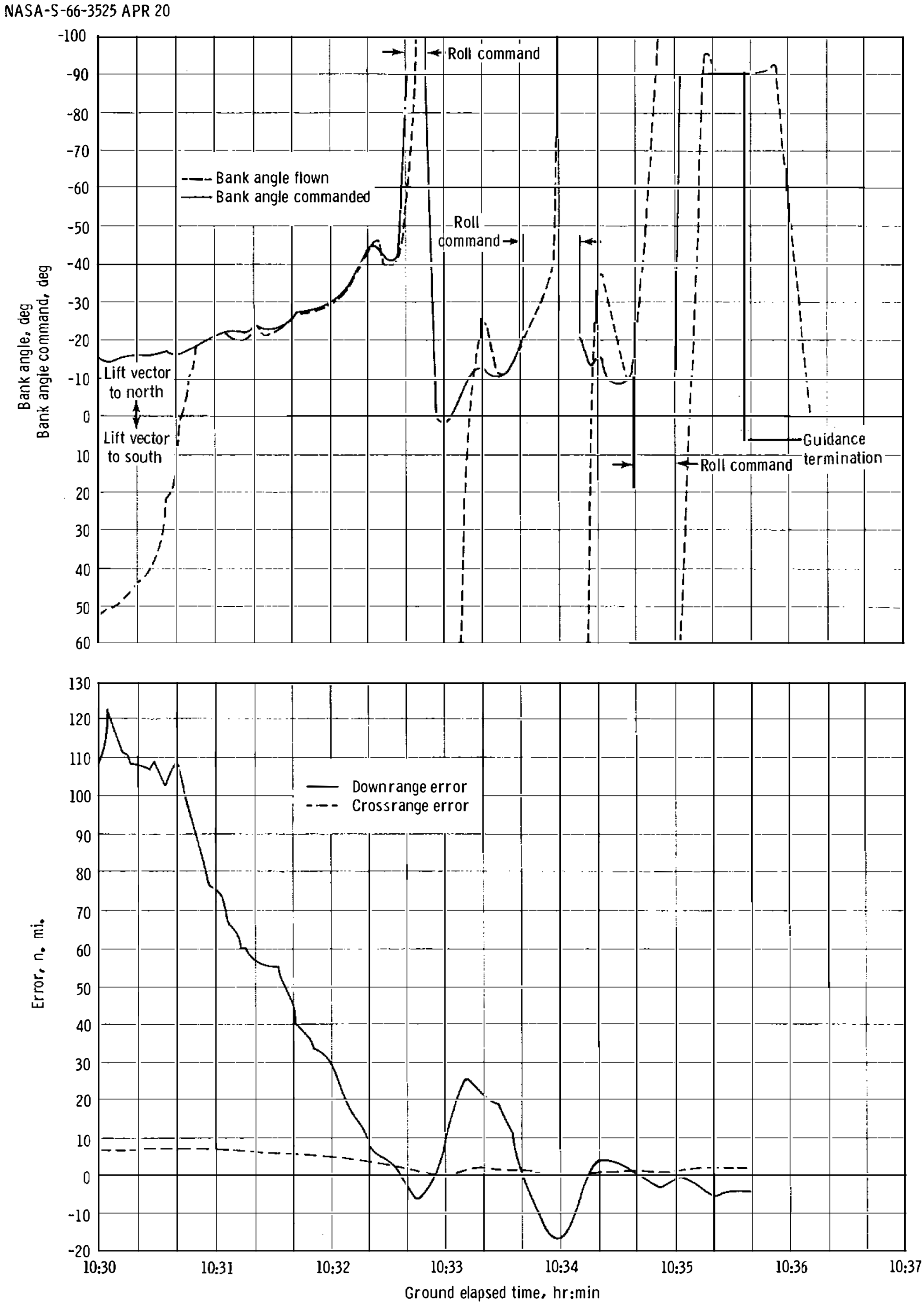

Figure 5.1.5-17. - Reentry guidance parameters.

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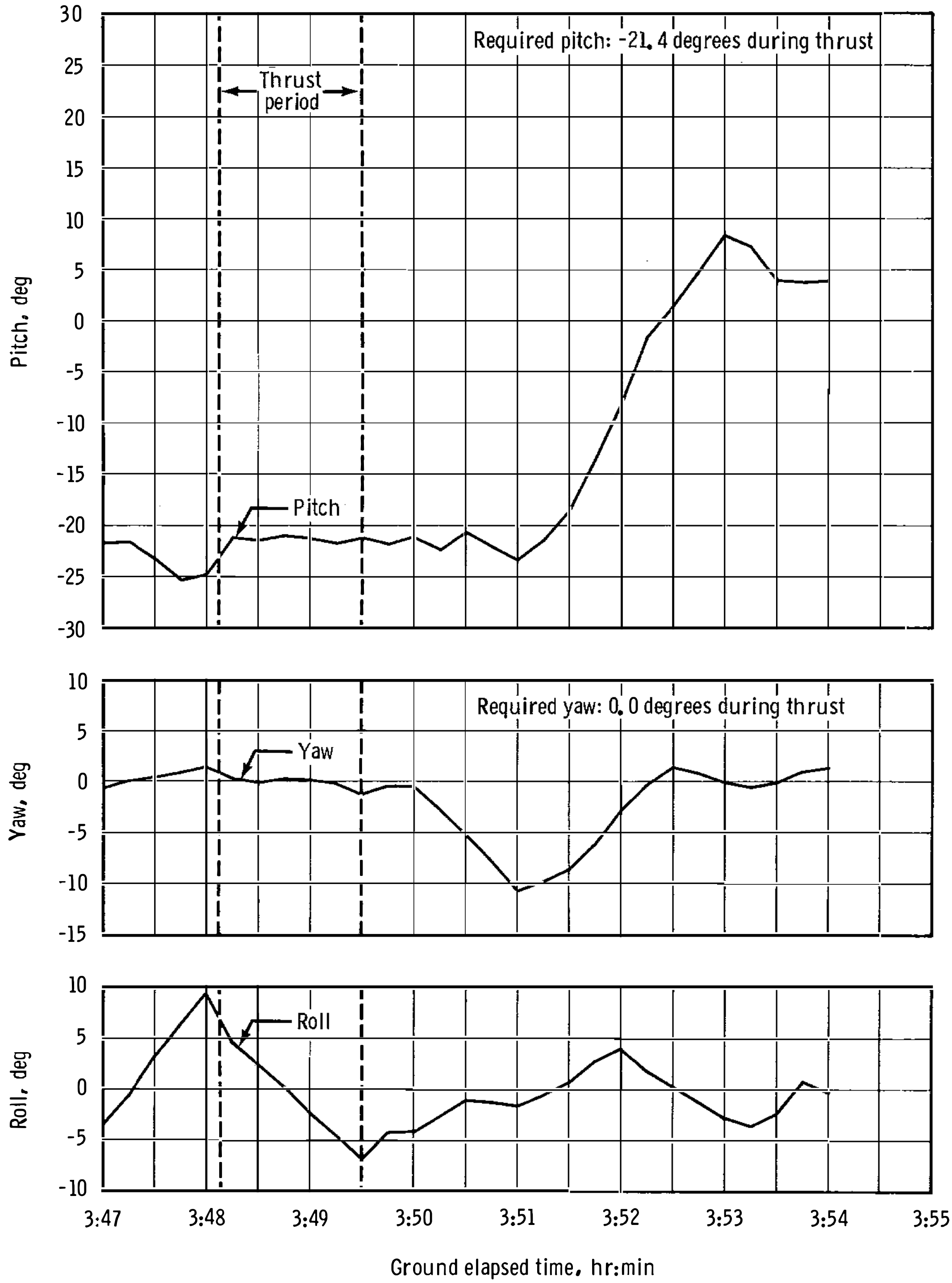

Figure 5.1.5-18. - Coelliptic maneuver attitudes.

5.1.6 Time Reference System

Analysis of available data indicates that throughout the mission all components of the Time Reference System performed according to specifications. The electronic timer began counting elapsed time approximately 6 milliseconds after lift-off. Maximum error during 36 240 seconds was approximately 100 milliseconds or 2.8 parts per million, which is well within the specification requirement of 10 parts per million at 25 ± 10° C. In addition, the electronic timer successfully initiated the auto-retrofire sequence at 36 286.6 seconds.

The event timer and the elapsed-time digital clock were used several times during the mission and were found to be correct when checked against other sources. The flight crew reported satisfactory operation of the G.m.t. battery-operated clock and the G.m.t. mechanical clock, but made no special accuracy checks. The clocks were not compared against an accurate clock during the recovery sequence. Satisfactory timing on tapes from the biomedical tape recorder and the on-board voice tape recorder indicates normal operation of the time correlation buffer.

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5.1.7 Electrical System

The Electrical System performed in a satisfactory manner except for the malfunction in the electrical distribution system which apparently caused the flight control anomaly. The load sharing between the fuel-cell sections was not as good as on previous missions, particularly in the first few revolutions; however, this was not unexpected because one section had an extended activated storage period prior to the flight. The flight-control problem which resulted in termination of the mission manifested itself in considerable fluctuation of the common-control-bus voltage. This fluctuation is considered to be normal because all thruster solenoids are powered from this bus.

5.1.7.1 Fuel Cell Power System.- The Fuel Cell Power System performed as required in delivering electrical power to the spacecraft systems. The spread in flight performance between the two fuel-cell sections and the resultant load sharing, detailed in sections 5.1.7.1.1 and 5.1.7.1.2, are consistent with laboratory test results. The lower performance of section 2 can be attributed primarily to the longer storage period which it experienced after initial activation. Modifications incorporated in the spacecraft since Spacecraft 7 appear to have been effective during the flight of Spacecraft 8 in allowing efficient purges and water-pressure control.

5.1.7.1.1 Fuel-cell section-activation history: Section 1 was activated for the first time on February 8, 1966, and section 2 on November 3, 1965, as part of the Gemini VII operation. Section 2 was removed from Spacecraft 7 until the possible effects of an over pressurization received during prelaunch preparations could be determined. Subsequent over-stress over-pressurization tests of similar hardware by the vendor, and leak rates of section 2 at previously recorded levels, confirmed that it had not been damaged. Both sections were activated for the second time during the midcount prelaunch activities of Spacecraft 8 on March 15, 1966.

The second activation of section 1 proceeded in a normal manner, without any unusual incidents. The second activation of stacks B and C of section 2, after accounting for expected reduced performance as a result of storage since first activation, also proceeded in a normal manner. However, a maximum of only 27.5 volts was attained by stack 2A after initial introduction of reactants. This compares with the normally exceeded 31.5-volt open-circuit specification voltage. Investigation of the difficulty showed that stack 2A was not at open circuit but was producing approximately 5 amperes, thus accounting for the unexpectedly low voltage. This current drain was corrected in approximately one hour by removing a short on an Aerospace Ground Equipment (AGE) wire. While the cause of the current drain on stack 2A was under investigation, its hydrogen inlet valve was shut, once for 2 to 3 minutes and again for 50 seconds, while purging stacks 2B and 2C. During the longer of these periods, stack 2A dropped to 19.5 volts, reflecting the effect of hydrogen starvation.

5.1.7.1.2 Fuel-cell section-performance variations: Figure 5.1.7-1 shows the performance of sections 1 and 2 during second activation, prelaunch standby, and the first and ninth hours of flight. The second activation of section 1 is consistent with that experienced on previously flown fuel cells. The decay in performance during the prelaunch standby period and the improving performance during the early flight hours were also observed on previous missions. The overall performance of section 2 was about as expected, considering the long time between first and second activations; however, a comparison of the stack data shows that the performance of stack 2A was lower than 2B and 2C. The lower performance of stack 2A was apparently caused by the hydrogen starvation or the out-of-sequence 5-ampere load which it experienced during second activation, or the combination of the two. Figure 5.1.7-2 shows the performance that section 2 would have achieved if stack 2A had performed in the same manner as stacks 2B and 2C. Comparison of the normal section 1 (figure 5.1.7-1) with the unaffected stacks B and C of the storage-degraded section 2 (figure 5.1.7-2) shows the apparent effect of fuel-cell storage after activation. By assuming a linear time dependance and no major manufacturing quality-control differences, the post-activation storage-degradation rate was approximately 0.6 volt per thousand hours of storage for between 10 and 20 amperes per section at second activation. These degration rates are approximately equal to those experienced with laboratory sections and about twice the degradation rate observed after second activation in the stack-storage test program. Unlike section 1, the performance of section 2 stayed constant during the initial flight hours.

5.1.7.1.3 Load sharing: Figure 5.1.7-3 shows the current supplied by each of the fuel-cell sections and the percent of the section current that each of the stacks supplied for the entire mission. The three stacks of Section I almost equally shared the total load of that section. From the low performance shown in figure 5.1.7-1, it is evident why section 2 supplied only 35 to 37 percent of the main-bus current. Similarly, the 26 to 30 percent of section 2 load carried by stack 2A is accounted for by the degraded performance discussed in paragraph 5.1.7.1.2 Stacks 2B and 2C shared the remaining section 2 current almost equally.

When the spacecraft main batteries were initially placed on the bus during prelaunch operations, they assumed approximately 50 percent of the spacecraft load. This sharing dropped to approximately 33 percent at one-half hour before launch and to 14 percent at 8 minutes after lift-off, just before the batteries were removed from the main bus.

When the batteries were placed back on the line in preparation for retrofire and reentry, they picked up only 11 percent of the main-bus load. Comparing this performance with the 14-percent and 22-percent load sharing maintained by the main batteries in the ascent phases of the Gemini V and VII missions, respectively, and considering the low performance of fuel-cell section 2, it appears that the performance of one or more of the batteries was also somewhat lower than normal. This indication has been further substantiated by the in-flight battery checks.

5.1.7.1.4 Differential-pressure indications: Two series of fuel-cell purges were conducted in flight, starting at approximately 3 hours 9 minutes g.e.t. and at 8 hours 19 minutes g.e.t. The flight crew reported observing the differential-pressure warning lights illuminate during three of the four hydrogen purges. The fourth ON condition was recorded in the bi-level telemetry data.

Figure 5.1.7-4 shows the analog hydrogen-to-oxygen differential pressures recorded during the second series of purges. These data indicate that the maximum differential-pressure increase occurred in the section being purged, and that a similar, but reduced, change occurred simultaneously in the other section. The oxygen-to-hydrogen differential-pressure increase in the section not being purged was a result of an open cross-over valve between the sections during the purge. This increase was small because of the additional lines connecting the two sections. The bi-level sensors that signal the warning lights are adjusted to actuate at differential pressures greater than approximately 1.4 psid and figure 5.1.7-4 shows a maximum differential pressure of only 0.72 psi during the hydrogen purges. The fact that the warning lights illuminated during these purge cycles is attributed to the pressure drop in the lines between the analog and bi-level sensor locations.

A similar, but lesser, effect of the differential-pressure sensor locations was observed during the oxygen purges when a decrease in the oxygen pressure was indicated during the oxygen purge. A decrease in oxygen pressure was also manifested as a decrease in oxygen-to-water pressure of approximately 0.1 psi upon initiation of the hydrogen purges. These indicated changes were not reflected by any change of gas pressure in the product-water storage tank.

All of the observed differential-pressure indications are consistent with ground test results, thus indicating normally functioning systems. These observations indicate that, at least for the two series of hydrogen purges, no restriction to the flow of gases occurred.

The observations also indicate that the water reference pressure was accurately maintained. This conclusion was further substantiated by the lack of any observed water-to-oxygen differential-pressure warning-light illumination during the launch phase of the flight.

5.1.7.2 Reactant supply system.- The reactant supply system performed as expected throughout the mission. The only anomaly was the inadvertent opening of the hydrogen-heater and oxygen-heater circuit breaker as discussed in section 5.1.7.3.

5.1.7.3 Power distribution system.- Although nominal power was delivered by the main bus throughout the mission, the following circuit breakers were found open at various times during the mission: (1) Auxiliary Tape Memory Unit (ATMU), (2) fuel-cell oxygen and hydrogen heaters, (3) RCS heaters, (4) antenna select relay, and (5) Orbital Attitude and Maneuver system (OAMS) control. Review of the data indicates that the circuit breaker for the fuel-cell oxygen and hydrogen heaters tripped at 5:49:07.3 g.e.t., but was probably inadvertently opened because there was no surge of main-bus current associated with the drop out.

The associated circuits and components for the oxygen and hydrogen heaters, the ATMU, and also the OAMS control circuits (powered by the common control bus) that are in the reentry assembly were investigated to determine their condition after flight and no discrepancies were found.

Postflight inspection of Spacecraft 8 revealed several blown fusistors in the pyrotechnic system. This has been observed on previous missions and is caused by a partial short circuit resulting from the normal slag formation in fired pyrotechnic devices.

5.1.7.3.1 Common-control-bus performance: Common-control-bus performance was satisfactory throughout the mission, although measured voltage levels were 0.50 to 0.75 volts lower than noted during previous missions. Figure 5.1.7-5 shows a time history of the common-control-bus voltage throughout the mission. For comparison, a simplified control-bus voltage-response plot for the Gemini VI-A mission (a comparable mission in terms of control-bus power demands) is also shown on the figure. In addition to being generally lower, the Spacecraft 8 control-bus voltage level also declined more rapidly during the period prior to the OAMS thruster malfunction than during the same time period of the Gemini VI-A mission. At 6:33:41 ground elapsed time (g.e.t.), 19 seconds after docking and rigidizing, there appears to have been a sharper decline in common-control-bus voltage. This was followed by the depressions characterizing the thruster malfunction period. Immediately following the shutdown of OAMS thruster 8 and after the flight crew regained control of the spacecraft with the Reentry Control System (RCS), the bus voltage recovered. The unusually low bus voltages recorded during the approximate 3 minutes of recorded postlanding data were accounted for by inadvertent firing of the RCS thrusters. This was caused by immersion of the attitude control electronics in salt water, which shorted the driver outputs to ground and energized the thruster solenoids.

The voltage transients resulting from thruster firings during the mission, particularly during the rendezvous maneuvers, were compared with those of other missions and found to be similar in magnitude and structure.

Postmission discharge of the batteries which supplied power to the common control bus showed that 9.7 amp-hrs, 9.8 amp-hrs, and 10.9 amp-hrs of usable power remained in squib batteries 1, 2, and 3, respectively. Similar discharge checks following the Gemini VI-A mission, which lasted approximately 15 hours longer than the Gemini VIII mission, showed 12.0 amp-hrs, 12.7 amp-hrs, and 12.6 amp-hrs remaining. Several factors evidently contributed to the larger ampere-hour usage. First, from OAMS and RCS propellant-usage data, Gemini VIII used approximately 163 pounds more than did Gemini VI-A, which indicates that Gemini VIII had considerably more thruster activity mostly as a result of propellant usage during the anomaly period. Second, the RCS thruster firings on the water account for some portion of the ampere-hour difference, the amount of which depends upon when the circuit breakers were opened.

5.1.7.4 Control system anomaly.- Figure 5.1.7-6 shows a more detailed plot of control-bus voltage from the initial inadvertent firing of OAMS thruster 8 to the eventual in-flight identification and correction of the problem. This period can be divided into seven parts:

(a) 7:00:26.7 to 7:02:37.4 g.e.t. — This period was characterized by voltage transients caused by thruster 8 first coming on, then going off, then staying on continuously, and by the counter thruster responses commanded by the flight crew.

(b) 7:02:37.4 to 7:07:20.3 g.e.t. — In this period, although telemetry was indicating thruster 8 to be on, spacecraft dynamics indicated that thruster 8 was not producing significant thrust; however, a low-grade spacecraft acceleration, representative of the thrust obtained when only the oxidizer valve is open, was present (see section 5.1.5). The average bus voltage should have recovered to the initial value of 25.35 volts at this time; therefore, the incomplete recovery of the bus voltage to only 25.20 volts supports the possibility of a single thruster solenoid being energized. It is important to note that during this 4 minutes 53 second period, thruster 8 was commanded on in several command modes, was indicated on continuously by telemetry, but apparently did not fire at any time.

(c) 7:07:20.3 to 7:15:44.7 g.e.t. — During this period, thruster 8 once more was on continuously and the bus voltage transients indicate the continued countering efforts by commanded thruster firings. At 7:15:12 g.e.t., thrusters 11 and 12 (forward-firing maneuver thrusters) were fired, separating the spacecraft and the Gemini Agena Target Vehicle (GATV).

(d) 7:15:44.7 to 7:18:15.7 g.e.t. — The RCS was activated during this period. Just prior to this operation it is a possibility that the motor valves were closed because, electrically, thruster 8 appears to have been on; however, spacecraft dynamics indicate it was not thrusting from 7:17:04 to 7:17:24 g.e.t. At 7:17:24 g.e.t., though not recorded, the motor valves would have to have been reopened, as spacecraft dynamics indicated that thruster 8 was thrusting. No electrical change was evident at that time.

(e) 7:18:15.7 to 7:19:03.8 g.e.t. — At the start of this period, the OAMS thruster circuit breakers for the solenoid-valve power were opened. Thruster 8 was off; this is evident in the telemetry records from the recovery of the bus voltage and from the spacecraft dynamics.

(f) 7:19:03.8 to 7:25:30 g.e.t. — In this period, the continuous set of voltage transients indicated the activity of the RCS thrusters when commanded by the flight crew in gaining control of the spacecraft. At 7:25:30 g.e.t., the rates were nulled in all axes.

(g) 7:25:30 to 7:28:30 g.e.t. — The flight crew reactivated the OAMS and found that thruster 8 would fire continuously when its circuit breaker was closed, even when the hand controller was in a neutral position. The voltage transient at 7:28:27 g.e.t. amounted to a depression of 1.25 volts in bus voltage when only the one thruster, no. 8, was firing.

The following four facts stand out from the preceding data:

(a) Telemetry indicated thruster 8 was on for 4.9 seconds and off for 4.0 seconds at the beginning of this sequence, then on for the remainder of this period.

(b) During the period from 7:02:37.4 to 7:07:20.3 g.e.t. when thruster 8 was not full on, it was commanded on several times without a successful reaction.

(c) The only times after the malfunction started when common-control-bus data, spacecraft-dynamics data, and telemetry bi-level data agree that thruster 8 was off was during the times when the thruster 8 circuit breaker was opened.

(d) There were periods of low-grade accelerations that were indicative of a single thruster valve opening. These occurred in periods (b) and (d) of figure 5.1.7-6.

From the above facts, it may be deduced that the failure was electrical rather than mechanical, and was complex in nature.

The circuits involved with the anomalous condition of the flight control system are shown in figure 5.1.7-7. The firing of the thrusters is normally accomplished by switching one end of each of the fuel and oxidizer solenoid coils to ground by means of transistor switches. The transistor switches are activated by logic circuits, commanded directly by the flight crew or automatically by the control system. Either primary or secondary transistor switching circuits may be selected by the crew.

From figure 5.1.7-7 it can be seen that the thruster will fire if,

(a) False inputs are sent to the valve drivers

(b) The valve drivers malfunction

(c) A low-resistance short exists in any of the wiring from the solenoids to the drivers

(d) A wire failure exists in the thruster solenoids.

Failure modes 1 and 2 can be eliminated for three reasons:

(a) The flight crew reported switching from the primary to the secondary drivers without a successful commanded response from thruster 8.

(b) A failure in this portion of the circuitry will not explain the low-grade accelerations characteristic of a single thruster valve operating.

(c) If it were possible to have a high-resistance short sufficient to drop out only one solenoid (2.0 volts across the solenoid), then the telemetry voltage would be greater than 15 volts. Hence, telemetry would have indicated off rather than on as it did during periods of low-grade accelerations.

It is evident from the above that the failure was in the solenoids or in the spacecraft wiring between the solenoids and the junction of the two solenoid ground returns.

Further isolation of the failure has met with little success. The fault is not a simple one; it must vary in resistance sufficiently to enable either or both thruster solenoids to fire and still meet the ON requirements of telemetry (less than 5 volts).

On Spacecraft 9 and subsequent spacecraft, the OAMS thrusters will be powered from a separate bus which will be armed and disarmed by a single switch. This will provide the crew with a rapid means of disabling all OAMS thrusters before dynamic rates have time to build up.

5.1.7.5 Sequential system.- The performance of the sequential system during the mission was nominal, as indicated in table 4.2-I.

At time of retrofire (T_R) - 256 seconds, the IND RETRO ATT light should have illuminated amber, thereby cueing the flight crew to position the spacecraft in the proper retrofire attitude. The crew reported that this light failed to illuminate. The circuitry and components involved with this apparent anomaly were checked during the postflight inspection of the spacecraft and found to be satisfactory.

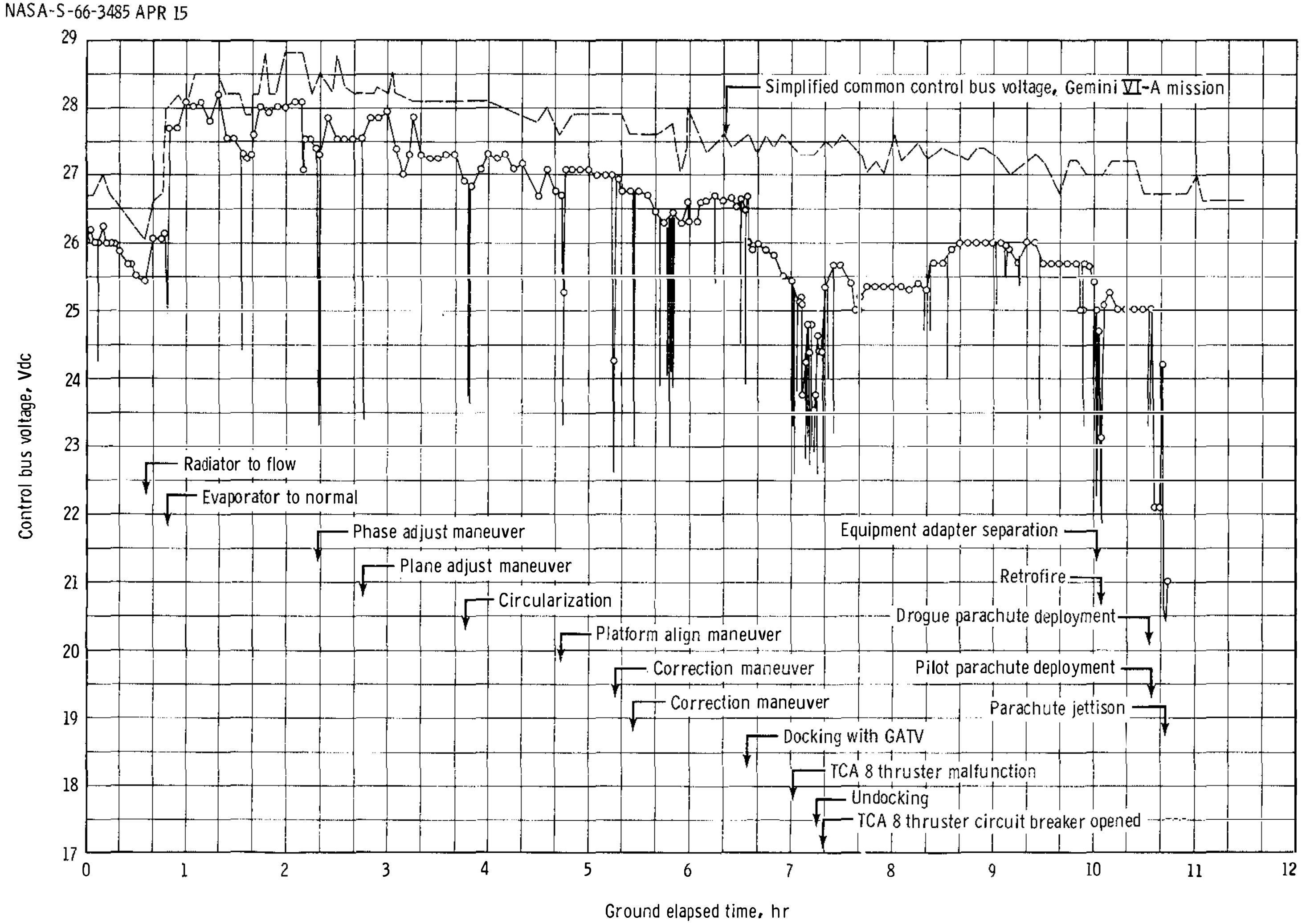

Figure 5.1.7-1. - Common control bus voltage performance.

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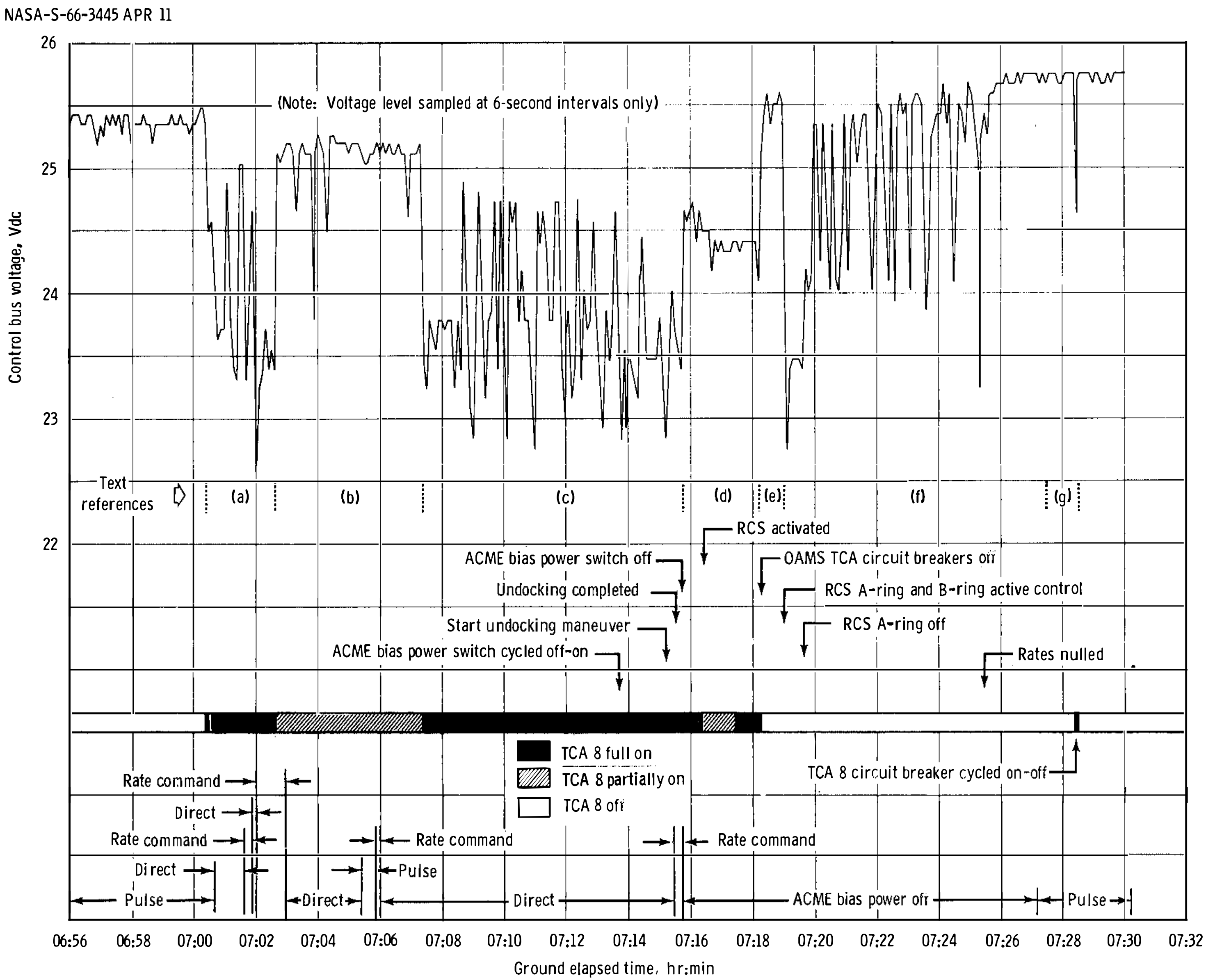

Figure 5.1.7-2. - Common control bus performance during OAMS thruster malfunction period.

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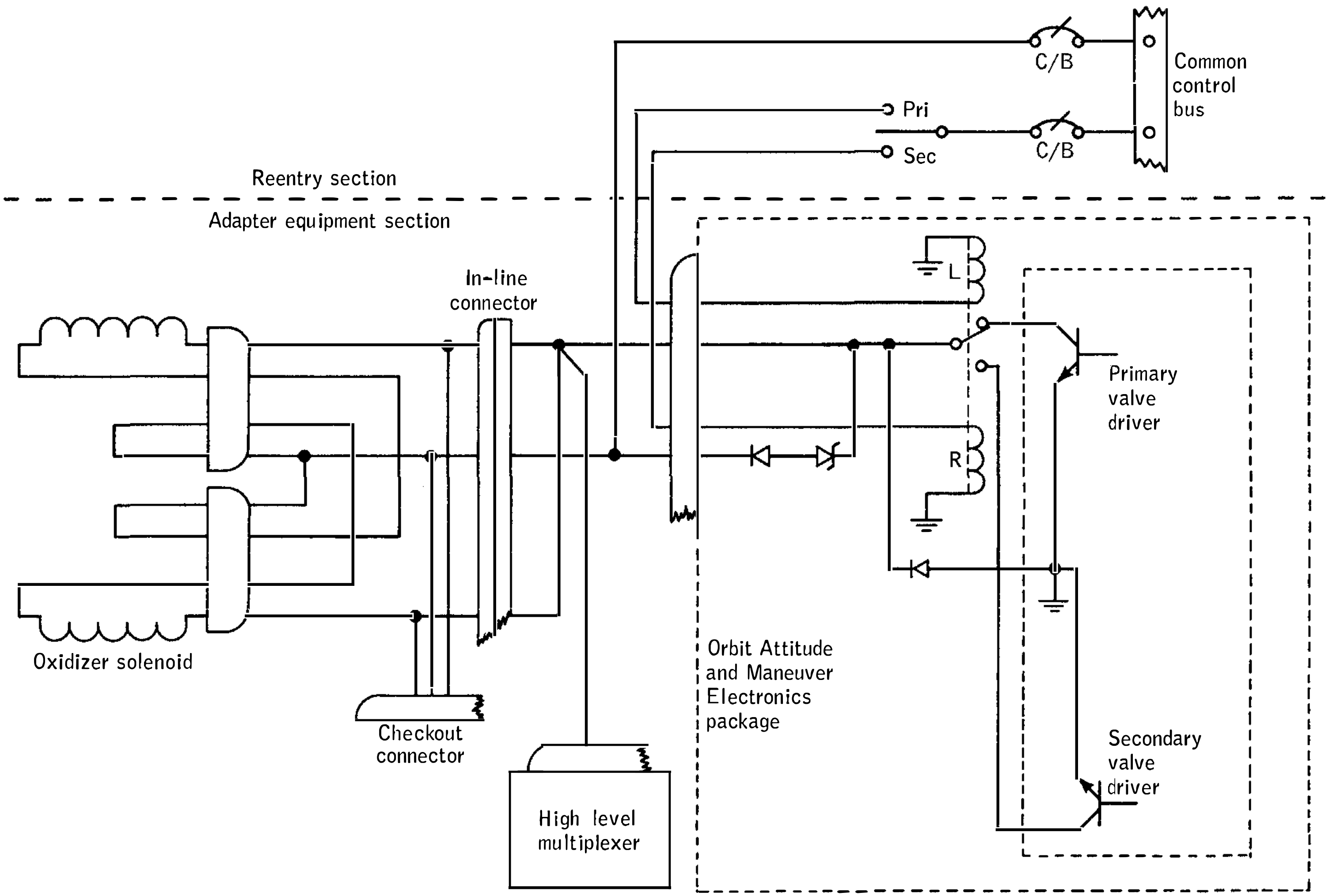

OAMS thruster 8

Figure 5.1.7-3. - Wiring schematic of critical area.

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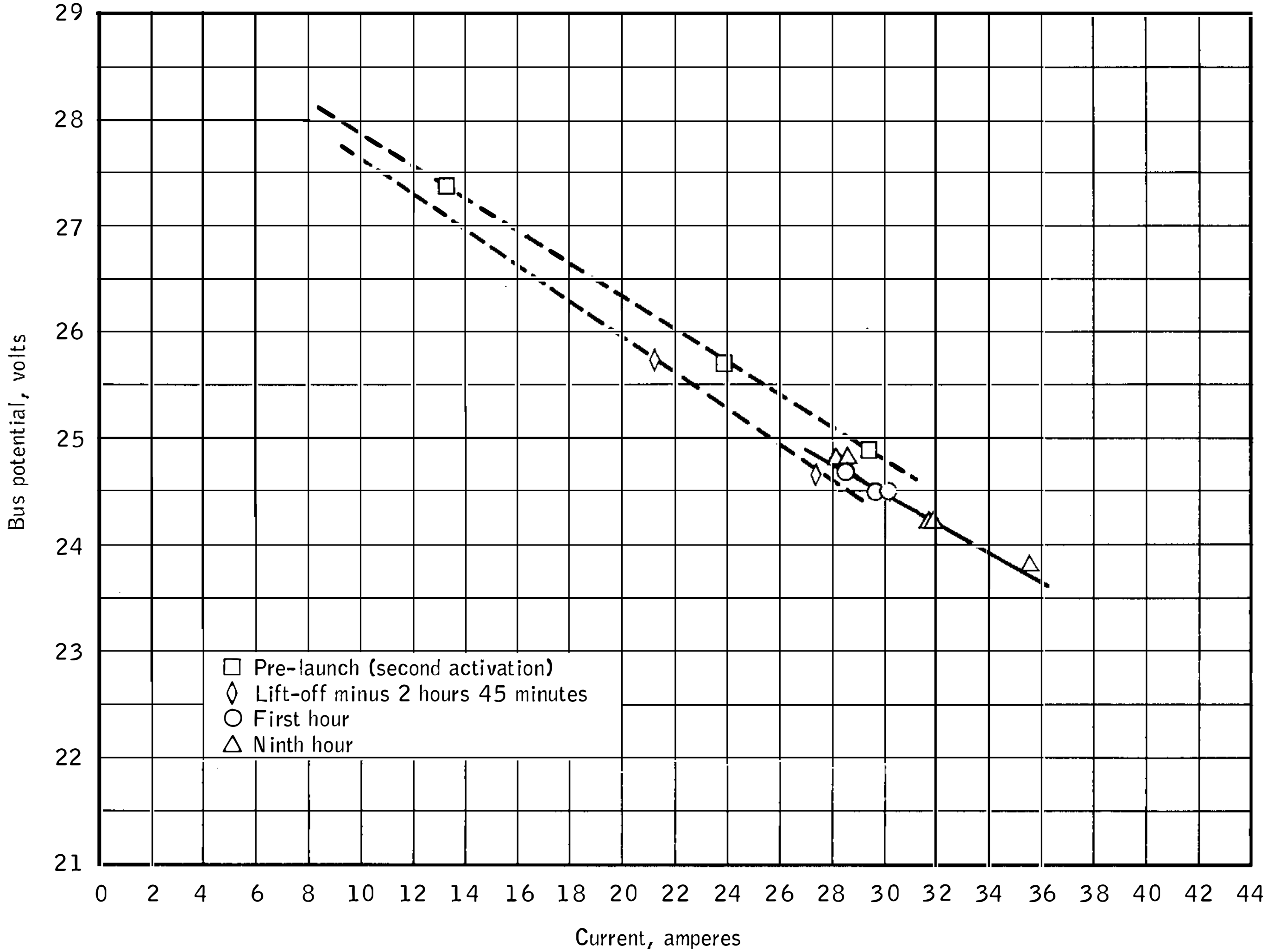

(a) Section 1

Figure 5.1.7-4. - Fuel cell performance.

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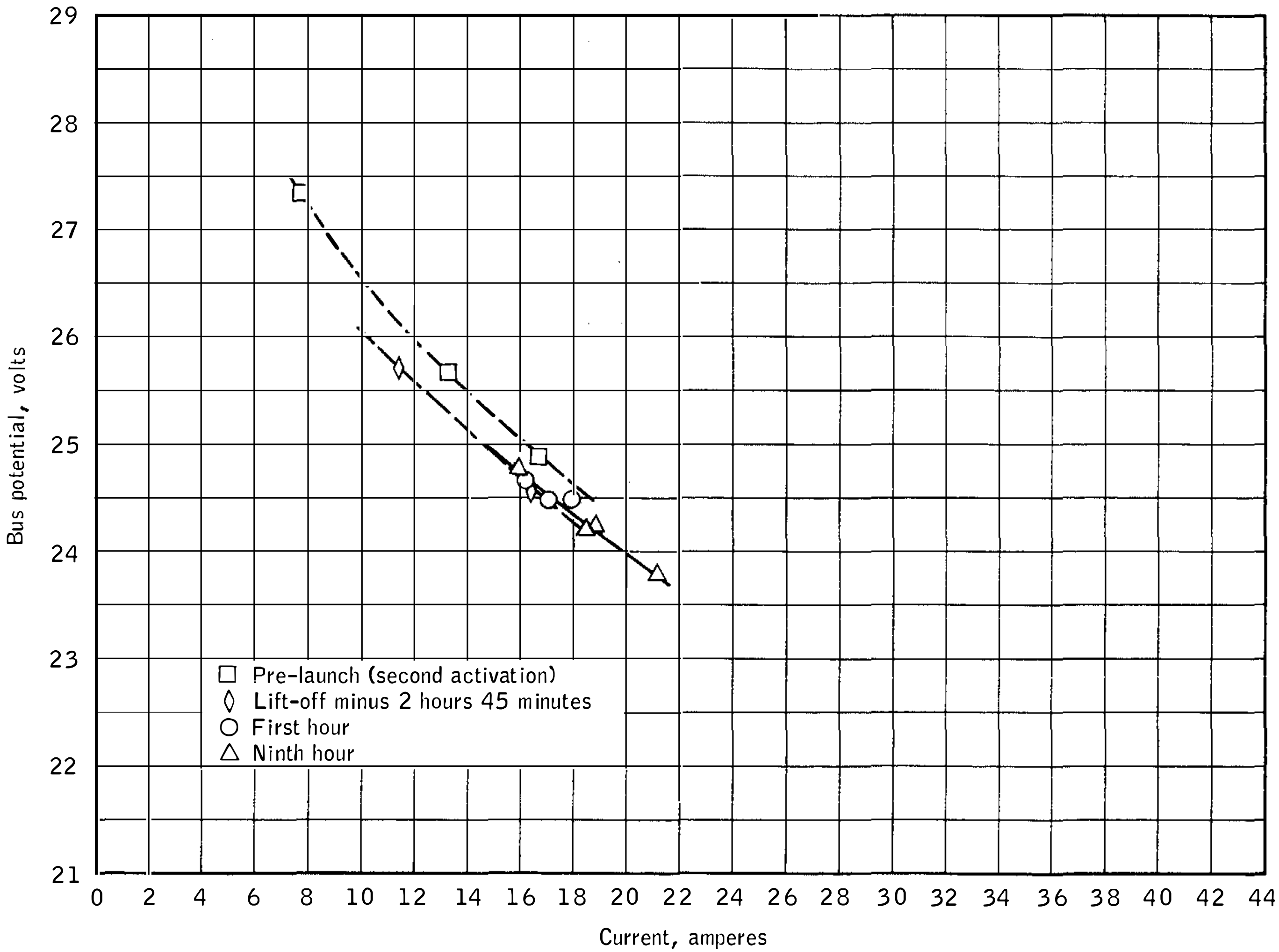

(b) Section 2

Figure 5.1.7-4. - Concluded.

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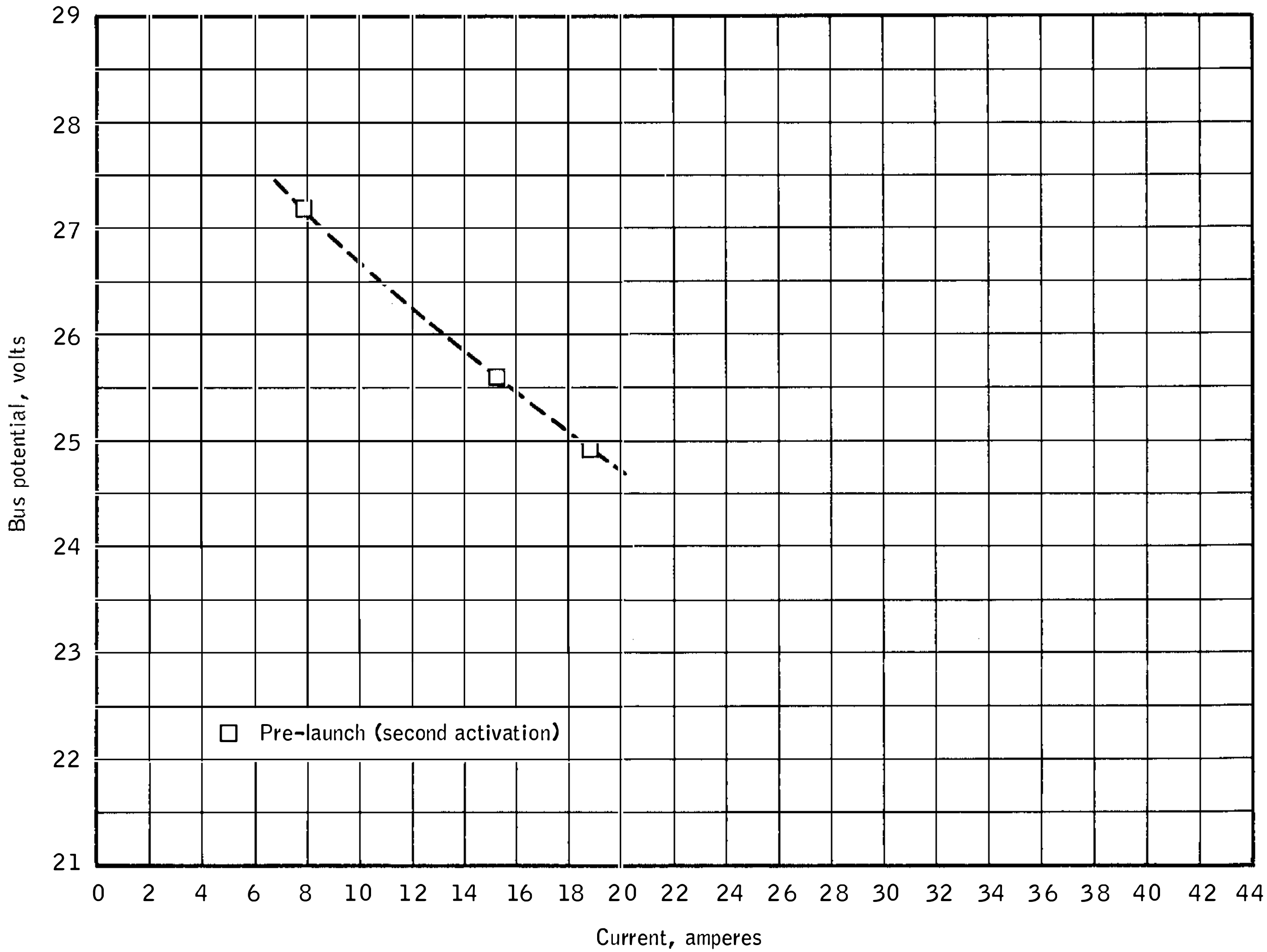

Figure 5.1.7-5. - Section performance, stacks 2B and 2C base.

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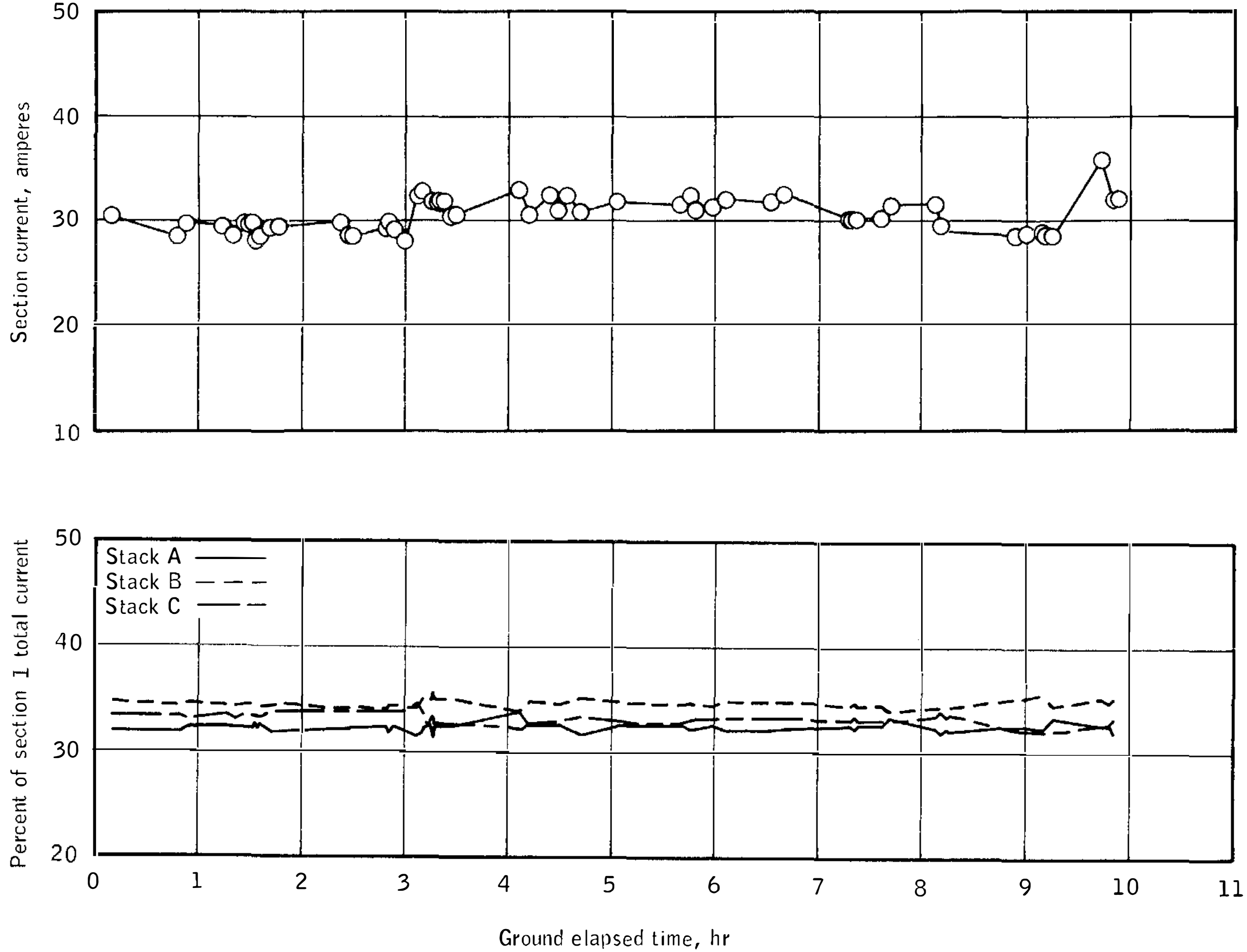

(a) Section 1.

Figure 5.1.7-6. - Fuel cell load-sharing characteristics.

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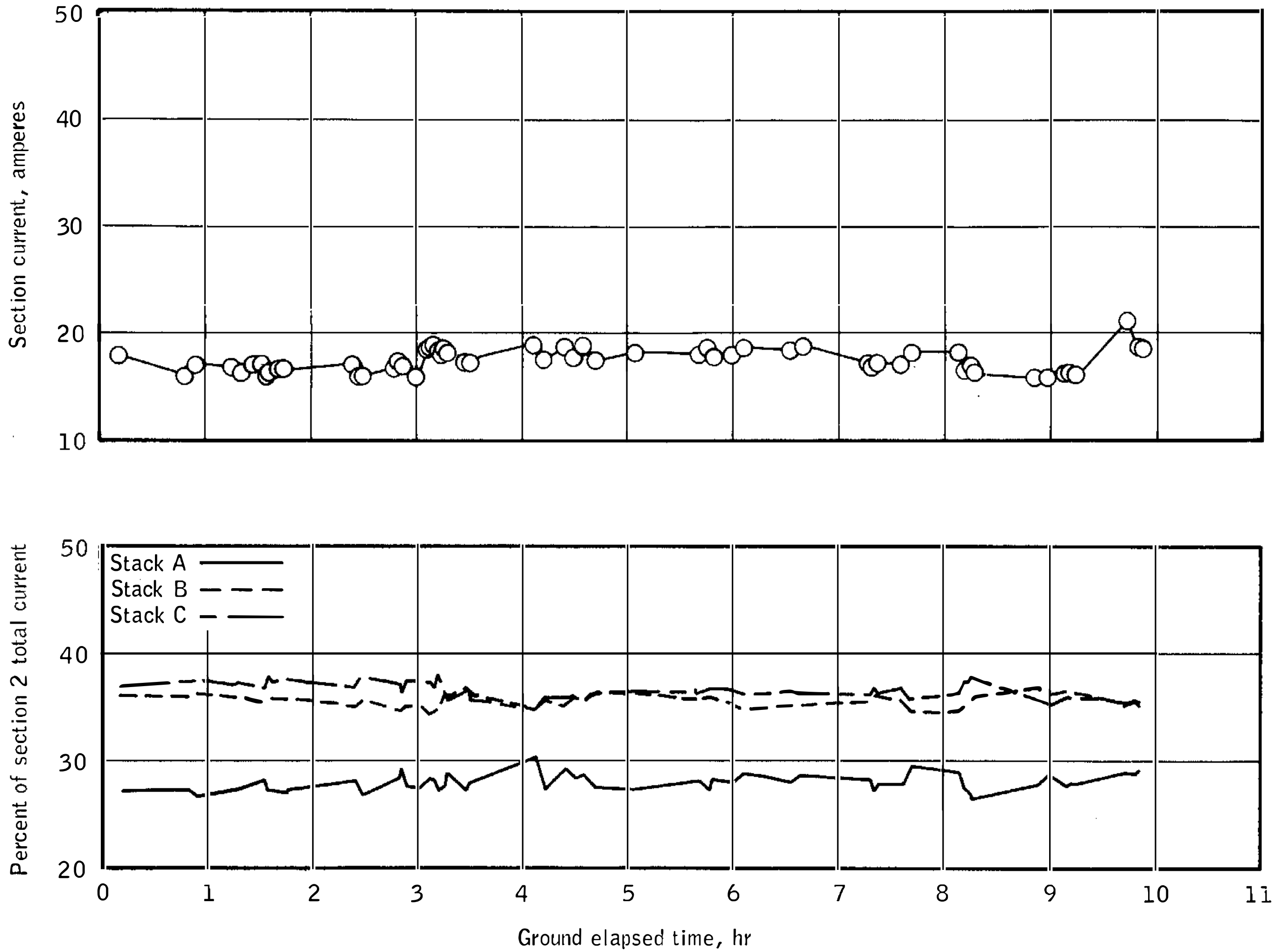

(b) Section 2.

Figure 5.1.7-6. - Concluded.

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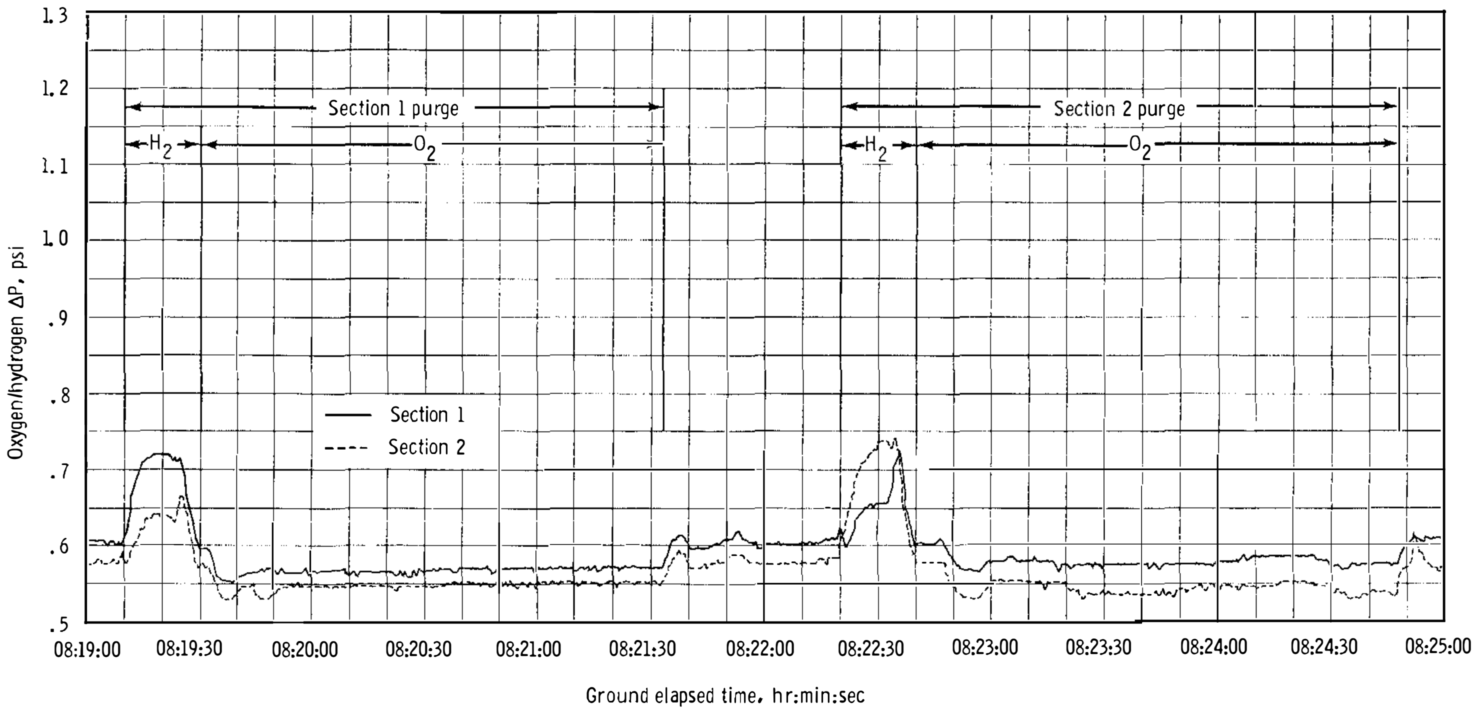

Figure 5.1.7-7. - Oxygen-hydrogen differential pressure during purge operations.

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5.1.8 Spacecraft Propulsion Systems

Performance of the spacecraft propulsion systems (the Orbital Attitude and Maneuver System (OAMS), the Reentry Control System (RCS), and the Retrograde Rocket System) was satisfactory, except for a possible association with the flight-control anomaly. The cause of the indicated loss of regulated pressure which occurred during the uncontrolled firing of thruster 8 is unknown. The period of degraded system performance reported by the crew after the rates were brought under control is attributed to the degraded performance known to exist when closing and reopening the motor valves.

5.1.8.1 Orbital Attitude and Maneuver System.-

5.1.8.1.1 Preflight: The quantities of fuel and oxidizer loaded, and the amount of helium pressure serviced, are presented in table 5.1.8-I. The quantity of fuel shown included 14.5 pounds loaded in the reserve fuel tank. These loadings constitute an available overall mixture ratio of 1.20 by weight. The same propellant servicing procedure as employed for Spacecraft 7, namely, withdrawal of the proper ullage (3 percent at 80° F) from tanks filled to capacity, was incorporated into the Spacecraft 8 servicing procedures. The composition of the oxidizer differed from that used on previous missions in that it contained 0.83 percent by weight of nitrous oxide which was added in order to curtail stress corrosion of the tank material. Otherwise, the fuel and oxidizer conformed to the normal military specifications.

The OAMS was activated approximately 30 minutes before lift-off and all parameters were within the expected limits. Static firings of all eight attitude engines were performed by the crew to vent gas from the propellant manifolds and to provide a final end-to-end verification of control-system operation. OAMS attitude engines 1 through 6 were each fired twice for an accumulated static-test firing time of 1.5 seconds for each engine. As a result of the test sequence, which started and finished with engines 7 and 8, these two engines were fired four times each for an accumulated time of 2.5 seconds each.

5.1.8.1.2 Flight: The OAMS maneuver engines exhibited satisfactory performance throughout the mission. Only the firing times of engines 9, 10, 11, and 12 were of sufficient duration to compute meaningful values of thrust. These engines produced 187.5 and 150.5 pounds of thrust for the 9 and 10, and 11 and 12 combinations, respectively, or 97 percent and 96.5 percent of that measured during the predelivery acceptance tests. The total number of maneuver engine starts and firing durations were as follows:

	Engine number					
	9, 10	11, 12	13	14	15	16
Total number of starts*	26	61	63	41	23	51
Firing duration, seconds	222	99	40	40	35	41

*Resolution of the telemetered data is 0.1 second, whereas the minimum possible pulse width is 0.02 second.

The planned and actual maneuver thruster firing times are compared in table 5.1.8-II. Several maneuvers required considerable activity of the radial thrusters 13 and 16 to obtain the desired incremental velocity. Three factors may have produced this condition:

(a) The spacecraft thrusters may not have been perfectly aligned.

(b) The spacecraft attitude may not have been maintained exactly during the firing.

(c) Additional thruster activity was required to remove any velocity imparted to the spacecraft by attitude engines fired during translation maneuvers.

Rather extensive attitude-engine firings were required to counter the disturbance torques produced by the maneuver engines. The primary cause of these disturbance torques can be attributed to the moment arm produced by an offset in the spacecraft center-of-gravity from the thrust vectors of the maneuver engines. The magnitudes of the resultant accelerations are shown in table 5.1.8-III for selected times during the mission. The table shows the forward and downward center-of-gravity shift which reduced the effect of the disturbance torques over the duration of the mission. This shift occurred as the OAMS propellant was consumed. At 2 hours 20 minutes g.e.t., the conditions for which data were obtained afforded a direct calculation of the magnitude of the offset. The longitudinal displacement of the thrust vector with respect to the center-of-gravity was determined to be 4.2 inches ahead of the center-of-gravity. The radial offset of the center-of-gravity was determined to be 1.5 inches above the longitudinal axis. These values agree with the preflight-calculated center-of-gravity location within the accuracy limitations of the data. The other factor causing the disturbances is attributed to engine misalignment within the spacecraft, but that effect is believed to be small.

Although these accelerations were within the control capability of the spacecraft attitude control system, they did require considerable corrective action from the attitude engines, resulting in larger engine firing times and consequently in higher-than-normal propellant consumption, which is discussed in a subsequent paragraph on propellant usage.

Injector temperature data, available only on thruster 10, showed a maximum temperature of 220° F at 4 hours 20 minutes g.e.t. This temperature followed a 78.2-second firing, and is considered normal.

The OAMS attitude thrusters exhibited satisfactory thrust levels prior to the spacecraft—GATV undocking. Specific thruster performance values are tabulated in table 5.1.8-IV. The total number of starts and firing duration of the eight attitude thrusters are as follows:

Thruster number	1	2	3	4	5	6	7	8
Total number of starts*	1670	1300	3560	3250	1540	1370	2640	1770
Firing duration, seconds	221	209	618	597	125	128	637	900

*Based on delayed-time data only.

Resolution of the telemetered data is 0.1 second whereas the minimum possible pulse width is 0.02 second.

Figure 5.1.5-19 shows that nominal thrust was being produced by thruster 8 at the beginning of the failure period, 7:00:26 g.e.t., and during the firing at 7:17:30 g.e.t., just prior to opening the circuit breaker. However, during this interval, accelerations were indicated to be less than nominal in a few instances. In one case, thruster 8 ceased to fire while the spacecraft and the GATV were still docked, (from 7:02:37 to 7:07:20 g.e.t.), but a small roll acceleration reflecting a 0.5-pound disturbance force was recorded. This force was approximately the same as that produced by oxidizer flow alone; however, the corresponding yaw accelerations appeared to be lower than that expected. The value in yaw was very small and in the same order of magnitude as the accuracy of the data. Varying accelerations after undocking but prior to opening the circuit breakers are presumed to result from intermittent thruster firings or from closing and opening the motor valves; however, the crew did not report operating the motor valves at this time. The thrust levels of thruster 8 are believed to have been nominal whenever both propellant valves were open. This is based on the nominal accelerations measured just prior to opening the circuit breaker and on OAMS thruster lifetime capabilities.

During troubleshooting of the OAMS after the spacecraft was stabilized, rates produced by attitude thrusters decreased to essentially zero until 7 hours 40 minutes g.e.t. when pitch thrusters 1 and 2 appeared to be producing some low-level thrust. By 7 hours 50 minutes g.e.t., pitch-control authority was fully restored, and at 9 hours 5 minutes g.e.t., the yaw thrusters appear to have been operating normally. These changes in thrust are attributed to the closing and opening of the motor valves. The precise total sequence of events cannot be obtained because motor-valve positions were not telemetered. After opening the valves, satisfactory pitch-thruster performance was restored prior to the restoration of the yaw-thruster performance because a greater amount of pitch control was first demanded. (Approximately 1.5 seconds were required to restore full control authority to pitch thrusters 1 and 2; 1.9 seconds to pitch thrusters 5 and 6; 0.7 second to yaw thrusters 3 and 4; and 0.9 second to yaw thruster 7.) A large number of pulses, ranging from 17 on thrusters 3 and 4 to 60 on thruster 2, were required to restore engine performance due to the use of the pulse mode. In this mode, a 20-millisecond signal is transmitted to fire the thrusters. The phenomenon associated with opening and closing the motor valves has been experienced previously and is under investigation to determine the cause.

The sequence of events during the failure period is presented in section 5.1.5.3. At the time of failure, thruster 8 had been off for 27 minutes. There was no apparent anomalous performance of this thruster prior to the firing that occurred at 7:00:26 g.e.t., nor was its duty cycle any more severe than that of the other engines.

The valves on thruster 8 opening unintentionally was probably caused by an electrical short to ground. The design of the control system is such that voltage is normally applied to one end of the solenoid coils and a firing command is effected by grounding the other end of the coils. As discussed in section 5.1.7, there were several locations in the spacecraft at which the fault could have occurred. One possible location is within the valve itself. However, from a review of the valve design, the acceptance test data of thruster 8, and the past history of the failure records of all Gemini valves during manufacturing, development, qualification, and reliability testing, the probability that the failure can be attributed to a short within the valve, other than from an isolated quality-type problem, is considered remote.

The regulator maintained 298 to 300 psia throughout the flight. No tendency to creep was observed. From 7:11:29.4 hours g.e.t. until adapter separation, the regulated pressure data indicated essentially zero pressure. This can only be attributed to a failure in the regulated pressure transducer or its associated circuitry. Satisfactory regulator performance has been verified by spacecraft angular accelerations, indicating correct propellant pressure at the injector, as well as from the F-package transducer which, at the time of the indicated failure, was sensing correct ullage pressure in the reserve fuel tank.

The total quantity of usable oxidizer and fuel was 411 and 340 pounds, respectively. When referenced to the preflight-determined mixture ratio of 1.05, 698 pounds of propellant would have been available to the crew. The propellant consumed during the mission is compared with the preflight planned usage rate in figure 5.1.8-1; also included are the mixture ratios used to establish the flight propellant quantities. The figure also shows the ground-computed values as determined from the general gaging equation during the flight and from the flight values read by the crew from the onboard propellant quantity indicator (PQI). The PQI value at activation was 101 percent, as compared to a preflight estimated value of 105 percent. This introduced an initial +4 percent correction factor in addition to corrections required for mixture ratio excursions from the fixed QPI gage reference of 1.05. When the readings obtained from the crew were corrected for the flight mixture ratio variations and decreased by 4 percent, the values correlated closely with the ground-computed values.

A comparison of the two measurements of propellant quantity, PQI and the gaging equation, shows good agreement. The propellant required through docking was somewhat greater than the flight-plan estimates. This was caused partly by the added real-time requirement of a plan-change and a vernier height-adjust maneuver, which consumed 27.6 pounds of maneuver propellant. Additional quantities were also consumed because the maneuver firing durations were greater than planned due to the post-maneuver corrections discussed previously. The lower flight mixture ratio realized up through docking, as compared with the preflight estimates, indicates that more attitude propellant was required than had been planned.

During the period 7:00:26 to 7:25:30 g.e.t., the attitude thrusters consumed 190 pounds of propellant, according to the results obtained from the gaging equation. From engine acceptance-test data measured by the manufacturer and the flight engine firing-duration data, 203 pounds were consumed by all attitude thrusters, which is in agreement with the gaging-equation results within the accuracy of the system.

At the time the equipment section was jettisoned, 563 pounds of propellant had been used, as determined by the general gaging equation. The actual overall mission mixture ratio was 0.90.

5.1.8.2 Reentry Control System.-

5.1.8.2.1 Preflight: The planned propellant loadings are compared with the actual loadings in table 5.1.8-I. The type of fuel and oxidizer loaded in the RCS was the same as that used in the OAMS (section 5.1.8.1.1).

5.1.8.2.2 Flight: The crew reported that they neither turned on the RCS heater nor noticed any heater warning lights during the mission. Throughout the orbital phase until RCS activation, the measured temperatures ranged between 72° and 87° F. Source-pressure leakage over the 24-day period from servicing to activation was negligible. The respective A-ring and B-ring source pressures just prior to system activation were 3080 psia at 86° F and 3110 psia at 76° F, which compares well with the serviced pressures of 3102 psia and 3046 psia corrected to flight temperatures at activation.

Activation of the RCS occurred at approximately 7:16:25 g.e.t. to enable the crew to control spacecraft rates following spacecraft GATV separation. Typical rates measured during operation of the RCS, presented in table 5.1.8-IV, show nominal performance of the system. Although the first RCS firing indication occurred at 7:18:15.2 g.e.t., when yaw-right and yaw-left B-ring engines (3, 4, 7, and 8) appear to have received an 8.9-second-duration firing signal, the first actual RCS firing command occurred at 7:19:03 g.e.t. with both A and B rings operational and normal system response was observed. ACME bias power had been off since 7:15:45 g.e.t., and there was no hand-controller movement. Also, the control system does not contain the logic which would provide yaw or roll, simultaneous left and right commands. The most reasonable explanation is that the two RCS B-ring yaw circuit breakers were inadvertantly cycled, thereby providing the false 8.9-second engine-firing indication.

After system activation, the A-ring and B-ring regulators, respectively, remained within a range of 296 (+2, -0) psia and 298 (+6, -0) psia. The minimum B-ring source pressurant temperature of 35° F reflected a high control-system demand rate. The 72° to 101° F oxidizer-feed temperature range encountered is well within the operational capability of the system.

The A-ring was turned off at 7:19:38 g.e.t. after 79.7 seconds of firing time accumulated over 4 pulses. The B-ring was then used to achieve control, with the command pilot using 126 pulses and an accumulated firing time of 306.4 seconds, until 7:31:25.7 g.e.t. when the

B-ring was turned off. A check of the B-ring system operation from 9:01:49 to 9:07:27 g.e.t. in pulse and orbit rate-command modes showed nominal performance. A final check of the A-ring operation in rate-command, pulse, direct, and reentry rate-command modes, performed from 9:52:19 to 9:54:07 g.e.t., also provided nominal data. Prior to retrofire, thruster 3B had accumulated 145.9 seconds in 61 pulses and thruster 7B had accumulated 143.5 seconds in 54 pulses. The total number of starts and firing duration of all eight attitude engines in the A and B rings were as follows:

A-ring

Engine number	1	2	3	4	5	6	7	8
Total number of starts*	105	100	160	130	95	95	180	160
Firing duration, seconds	31	31	90	51	28	28	98	54

B-ring

Engine number	1	2	3	4	5	6	7	8
Total number of starts*	100	100	280	186	150	150	280	190
Firing duration, seconds	15	11	168	35	11	11	168	34

*Resolution of the telemetered data is 0.1 second, whereas the minimum possible pulse width is 0.02 second.

Dual-ring operation in orbit rate command was used during the retrofire period. The A-ring was turned off at the end of retrofire, and pulse mode operation was selected when the rates induced from the retrorockets had been damped. The reentry rate-command mode was selected at the beginning of guidance. Operation was switched from the B-ring to the A-ring when the B-ring source pressure dropped below 1400 psia. Orbit rate command was selected at 10:36:41 g.e.t., when the drogue parachute was extended, and the B-ring was turned on shortly thereafter at 10:37:25.6 g.e.t. The crew reported that propellant was expended between approximately 30K feet (10:37:00 g.e.t.) and main parachute deployment at 10K feet (10:38:08 g.e.t.). Postflight deservicing verified that no propellant remained in the system.

5.1.8.3 Retrograde rocket system.- All four retrorockets fired nominally in the automatic sequence, following initiation of retrofire at 10:04:46.6 g.e.t. The performance of the retrograde rocket system is shown in table 5.1.8-V.

TABLE 5.1.8-I.- OAMS AND RCS SERVICING AND SYSTEM ACTIVATION DATA

System	Source pressurant data			Propellant servicing data						Usable mixture ratio	Propellant quantity indicator, percent
				Oxidizer			Fuel				
	Pressure serviced, psig (a)	Servicing date	Pressure after activation, psia (a)	Total quantity, lb	Unusable, lb	Servicing date	Total quantity, lb	Unusable, lb	Servicing date		
OAMS											
Planned	2920	-	2900	417.7	15.7	-	347.5	6.7	-	[b]1.05	1.05
Actual	2921	2-20-66 (24 days before launch)	2936	(420.04)	-	2-18-66 (26 days before launch)	346.6	-	2-19-66 (25 days before launch)	-	1.01
RCS, A-ring											
Planned	3015	-	2755	20.2	1.2	-	15.8	0.7	-	1.3	Not applicable to RCS
Actual	3015	2-20-66 (24 days before launch)		20.2	-	2-18-66 (26 days before launch)	15.8	-	2-19-66 (25 days before launch)		
RCS, B-ring											
Planned	3015	-	2755	20.2	1.2	-	15.8	0.7	-	1.3	
Actual	3012	2-20-66 (24 days before launch)		20.2	-	2-18-66 (26 days before launch)	15.8	-	2-19-66 (25 days before launch)		

[a]All gas pressures in this table are referenced to 70° F.

[b]Required to fullfill preflight mission planned objectives.

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TABLE 5.1.8-II.- OAMS MANEUVER ENGINE SUMMARY

Maneuver	Total firing time, seconds		Engine firing time, seconds						Engine start summary						Attitude engines required (a)	Attitude engine duty cycle, percent (b)
	Planned	Actual	9-10	11-12	13	14	15	16	9-10	11-12	13	14	15	16		
Separation	6.0	8.1	8.0	0	0.1	0	0	0	1	0	1	0	0	0	3-7, 4-8 7-8, 1-2	50, 10 20, 7
Height adjust	5.0	9.7	0	6.1	0	0	3	0.6	0	5	0	0	1	1	5-6, 7-8	54, 46
Phase adjust	68.0	72.3	57.2	4.3	4.7	4.1	0	2.0	1	5	5	4	0	2	3-7, 4-8 7-8, 1-2 5-6	50, 10 18, 50 12
Plane adjust	35.0	39.9	35.7	2.4	0	0.3	0	1.5	1	3	0	1	0	3	1-2, 5-6 3-7, 7-8	61, 8 50, 18
Vernier height adjust	2.0	4.9	3.4	1.1	0	0	0	0.4	1	2	0	0	0	1	1-2	75
Coelliptic	82.0	80.2	78.2	0	0	0	0	2.0	2	0	0	0	0	3	1-2, 5-6 3-7, 7-8	50, 8 50, 25
TPI	43.0	53.4	20.9	3.5	0	21.6	7.4	0	1	4	0	5	2	0	1-2, 3-7 5-6	50, 48 10
First correction, 82 degrees	-	35.3	13.4	0	5.8	0	16.1	0	1	0	2	0	3	0	1-2, 3-7	91, 75
Second correction, 34 degrees	-	29.4	0	0	12.4	0	17.0	0	0	0	1	0	1	0	3-7, 1-2	74, 78
Braking	67	99.0	0	64.9	0.1	10.9	0.1	23.0	0	11	0	7	1	11	5-6	20
Station keeping and docking	-	57.5	5.1	10.3	16.8	9.2	6.6	9.5	18	36	53	24	21	30	-	-
Post-failure	-	10.4	0.2	[c]6.6	0.2	0	3.0	0.4	1	[c]1	1	0	2	4	-	-

[a]All maneuver engines produced disturbance torques that required correction by the attitude engines (see table 5.1.8-III). These data identify the attitude engines fired during the main maneuver firing.

[b]Resolution of the telemetered data is 0.1 second whereas the minimum possible pulse width is 0.02 second.

[c]Spacecraft-GATV separation firing.

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TABLE 5.1.8-III.- SPACECRAFT ATTITUDE ACCELERATIONS

INDUCED BY OAMS MANEUVER-THRUSTER FIRINGS

Maneuver thruster	Pitch acceleration, deg/sec^2 (a)	G.e.t. hr:min	Roll acceleration, deg/sec^2 (a)	G.e.t. hr:min	Yaw acceleration, deg/sec^2 (a)	G.e.t hr:min
9 - 10	+0.72	0:06	-0.18	0:06	0.0	0:06
	+0.34	2:20	-0.20	2:45		
	+0.28	3:50			0.0	3:50
	+0.17	5:15				
11 - 12	-0.26	2:20	-0.02	2:20	0.0	2:20
	-0.10	5:46	-0.02	5:46	-0.02	5:46
13	+0.11	2:21	-0.35	2:21	+0.35	2:21
14	0.0	2:20	+0.27	2:20	-0.35	2:20
15	+0.30	5:15	-0.12	5:15	0.0	5:15
16	-0.38	2:21	0.0	2:21	0.0	2:21
	-0.26	5:50	+0.10	5:50	0.0	5:50

[a] Accuracy of acceleration data is ±0.03 deg/sec^2.

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TABLE 5.1.8-IV.- OAMS AND RCS ATTITUDE ENGINE PERFORMANCE [a]

Engine numbers	Thrust, lb		Angular acceleration, deg/sec^2	
	Preflight	Flight	Preflight	Flight
OAMS				
1-2	45.7	43	3.7	3.5
3-4	46.1	44	3.7	3.5
5-6	45.8	44	3.7	3.5
7-8	46.4	43	3.7	3.6
RCS A-ring				
1-2	46.9	44	3.5	3.3
3-4	47.1	44	3.5	3.4
5-6	47.0	44	3.5	3.4
4-8	46.9	47	1.7	1.7
RCS B-ring				
1-2	46.9	44	3.5	3.3
3-7	47.3	47	1.7	1.7
5-6	47.1	44	3.5	3.3
4-8	47.0	47	1.7	1.7

[a] Typical values determined at various times throughout the mission.

TABLE 5.1.8-V.- RETROGRADE ROCKET SYSTEM

(a) System performance

Parameter	Actual	Predicted	Deviation, percent
ΔV, ft/sec[a]			
Longitudinal	292.5	292.0	-0.17
Vertical	114.1	110.0	-3.6
Lateral	0.3	0.0	-
Total	314.0	312.0	-0.61
Corrected ΔV, ft/sec[b]	314.7	-	+0.23
Spacecraft preretrofire weight, lb	5738	5770	+0.54

(b) Individual motor performance

Parameter	1	2	3	4
Total impulse, lb-sec[c]	14 219	14 320	14 332	14 222
Specific impulse, $\frac{\text{lb-sec}}{\text{lb}}$[c]	253	255	255	253
Web burn time, seconds	5.3	5.2	5.2	5.6
Ignition time, g.e.t., hr:min:sec	10:04:46.6	10:04:57.4	10:04:52.2	10:05:03.3

[a]Read by the crew from the onboard computer.

[b]The corrected values are based on retrorocket acceptance-test data.

[c]Predelivery acceptance test data.

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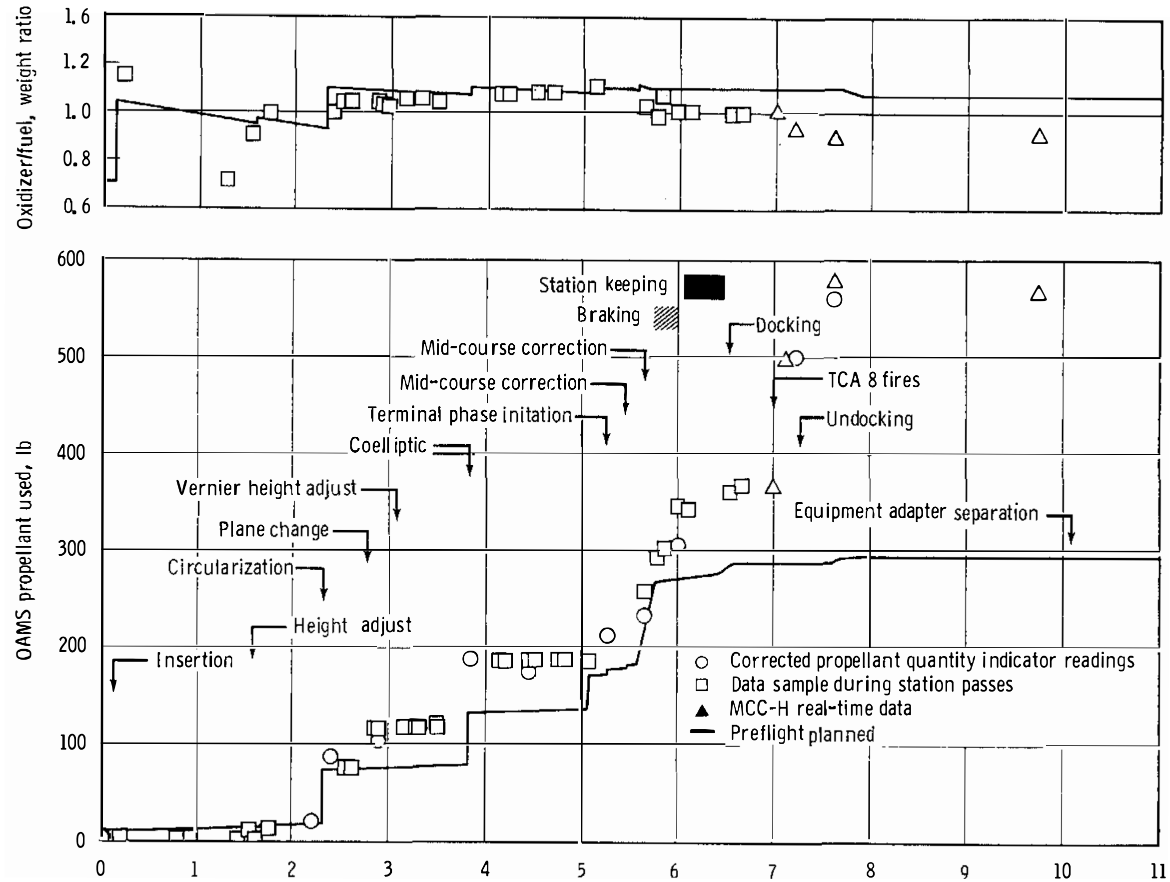

Figure 5.1.8-1. - OAMS propellant consumption.

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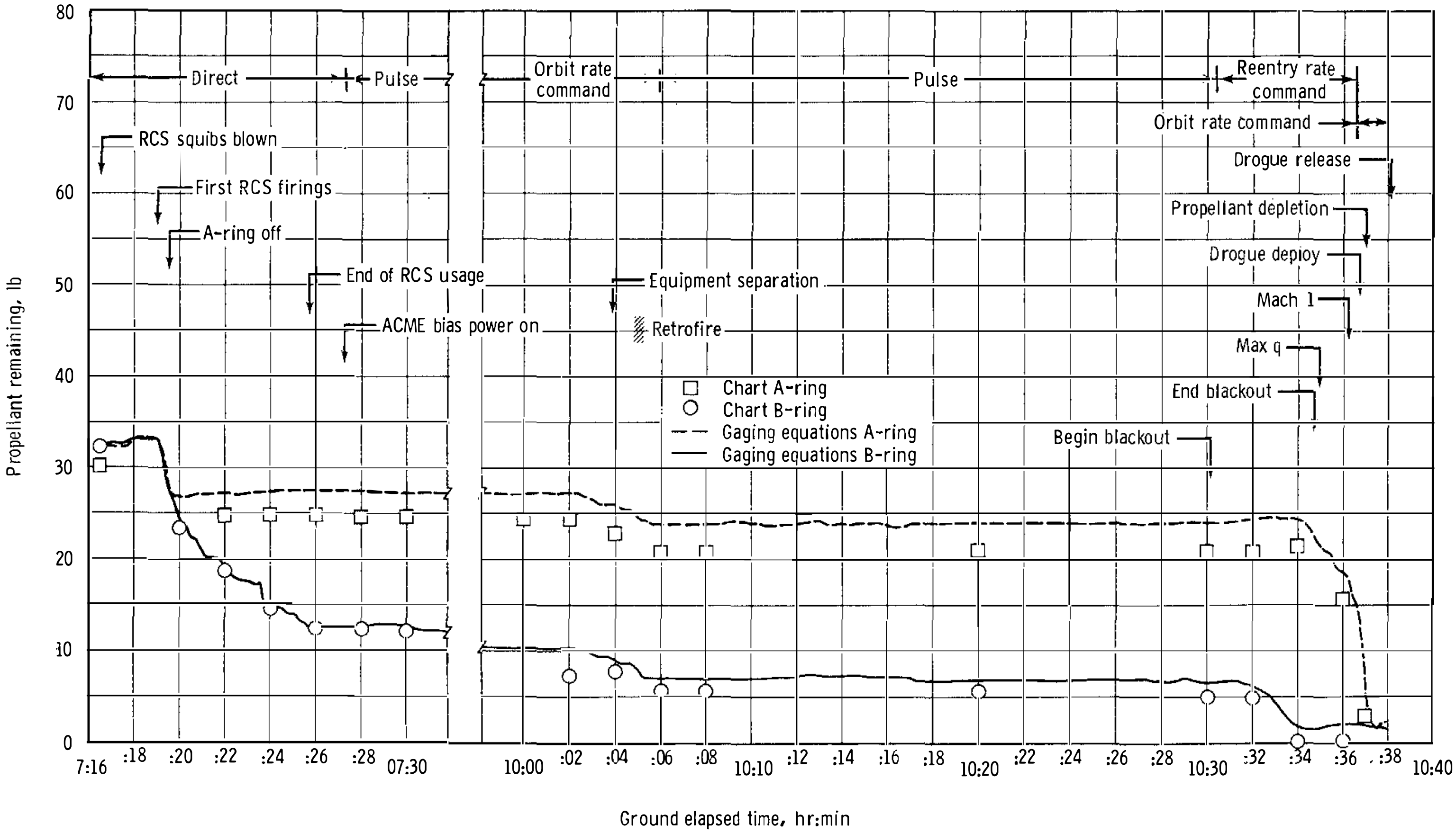

Figure 5.1.8-2. - RCS propellant consumption.

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5.1.9 Pyrotechnics

The pyrotechnic system performed all required functions in a satisfactory manner. A postflight examination of the spacecraft revealed loosened electrical connectors and bridgewire resistances of 5 to 15 ohms in the functioned devices. These occurrences have been observed on previous spacecraft, are considered to be normal, and do not represent any hazard to mission performance or crew safety.

For the first time in a Gemini mission, the crew elected to jettison the radar and horizon-scanner fairings later than usual to avoid the possibility of debris from the spacecraft--Gemini Launch Vehicle separation damaging the scanner unit. The fairings were jettisoned at 7 minutes 30 seconds ground elapsed time while the spacecraft was free of any body rates. When the jettison was performed, the crew noted that a body rate developed in the pitch-up yaw-right direction. Telemetry confirms these rates to be approximately 2.2 deg/sec in pitch and 0.7 deg/sec in yaw, which is in agreement with the anticipated energy developed by the jettisoning of the two fairings.

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5.1.10 Crew Station

5.1.10.1 Crew-station design and layout.- The design of the crew station was satisfactory for the portion of the mission that was accomplished. The principal changes to the crew station from the previous missions were related to the extravehicular equipment, which was not unstowed in this mission. There were a few anomalies in other items of equipment, and these are discussed below.

5.1.10.1.1 Equipment stowage: The basic equipment stowage provisions were satisfactory except for the centerline stowage compartment. When the door to the centerline stowage compartment was opened in flight, the lower shelf deflected upward approximately 5/8 inch. The crew experienced considerable difficulty in holding the shelf down while closing the door. The opposite condition occurred in the Gemini VI-A and VII missions: the lower shelf deflected downward when the door was opened. See section 5.1.1 for additional details.

The stowage provisions for the television monitor, carried in the right footwell for the D-15 experiment, were unsatisfactory for restowage in orbit. The retention strap was too short to be engaged readily, and it was difficult for the flight crew to hold the monitor in its stowage location while attaching the strap. Also, because the mounting arrangement was designed so that launch and reentry loads tended to move the monitor in a direction that would tend to loosen it, the integrity of the mount was dependent on the tightness of the strap.

The extravehicular visor for the pilot's helmet was stowed in the right footwell for launch. In the preparations for reentry, the pilot was unable to restow the helmet bag and visor in the same location forward of the television monitor. As a result, the crew stowed the visor behind the left seat for reentry. Reentry stowage of the visor had not been planned since it was to have been jettisoned in orbit after the extravehicular operation.

5.1.10.1.2 Cabin lighting: The cabin lighting was satisfactory for this mission. The red filter added to the right utility light was satisfactory for illuminating the GATV command-encoder control. The variable-intensity red post light added for illuminating the digital clock was satisfactory. A medium intensity setting on this light was used throughout the mission.

The crew reported a wide variation between the cabin interior and exterior light intensity during orbital daylight. They did not use the polaroid window filters but used their sunglasses for visual protection when looking at the sun-illuminated Gemini Agena Target Vehicle (GATV).

They were not bothered by reflections from the docking bar during day-side station keeping.

5.1.10.1.3 Crew furnishings: The ejection seats were not used except for support and restraint of the crew. At the time of crew ingress to the spacecraft just before launch, an epoxy-like substance was found in the Koch fitting on the pilot's left shoulder strap. This foreign substance prevented mating the components at the proper time in the launch countdown. There was approximately a 10-minute delay in closing the hatch while the substance was removed. No further difficulty was encountered and no hold was required.

The new location for the voice tape recorder on the left cabin wall was apparently satisfactory; however, the recorder was not removed from its holder during the mission.

The crew had both lap belts attached at the time of the control-system malfunction. They were held in the seats by the lap belts and were adequately restrained.

5.1.10.2 Displays and controls.-

5.1.10.2.1 Displays: The crew-station displays were satisfactory for the rendezvous mission. The command pilot found the added markings on the Flight Director Attitude Indicator to be very helpful in reading and controlling the spacecraft attitude. He reported being able to read pitch angles to less than 1/2 degree, and to control pitch attitude to less than 1 degree.

As reported in previous missions, the readability and location of the G.m.t. clock was poor. During the rendezvous the pilot used a stop watch mounted on Velcro on the right instrument panel to provide a readily accessible time display for the rendezvous backup procedures.

5.1.10.2.2 Controls: The attitude and maneuver hand controllers were satisfactory. During the Orbital Attitude and Maneuver System anomaly, the crew had to exercise care in the manner in which they observed the overhead circuit breakers because of the effects of the high rotational rates. Access to the undocking switches on the center panel was satisfactory.

5.1.10.3 Pilots' operational equipment.-

5.1.10.3.1 Still cameras: The 70-mm Hasselblad camera was used to obtain excellent photographs during the mission. Because the mission was terminated early, only 17 photographs were obtained and the super wide-angle Hasselblad camera was not used.

5.1.10.3.2 Sequence camera (16-mm): The 16-mm sequence camera mounted on the left window provided excellent coverage during the rendezvous, docking, and separation. Only two film magazines were exposed. The camera was operated without the circuit breaker and light assembly. The lens-setting and frame-rate scales were not visible while the camera was mounted on the window bracket and this made camera adjustment difficult.

5.1.10.3.3 Lightweight headsets: The pilots reported satisfactory performance of the lightweight headsets, except that the oral thermometer became detached during use. Postflight inspection showed that the Velcro patch used to hold the probe in place had come loose.

5.1.10.3.4 Optical sight: The light intensity of the optical sight was satisfactory except that the outer edges of the reticle faded out when the sight was dimmed. The voltage regulator for dimming the optical sight was not required. As reported after the Gemini VI-A mission, the optical-sight alignment varied in proportion to the tightness of the mounting knob. When the knob was tight, the sight alignment was within 1/2 degree of the radar-indicated boresight axis. There was no way to establish whether the small remaining error was in the sight or the radar. This variation in alignment had no apparent effect on the rendezvous operation.

5.1.10.4 Space suits and accessories.- There were no discrepancies in the space suits and accessories except for the life vests. The lack of identifying markings on the life vests caused inconvenience and delay when the crew was preparing for the early reentry. Several minutes were lost in identifying the left and right life-vest packages and determining which end was the top. After this delay, the life vests were donned satisfactorily.

5.1.10.5 Pilots' personal equipment.-

5.1.10.5.1 Food: The crew prepared only a few items of food. They reported that the rehydratable items were slow to reconstitute. In the postflight debriefing the crew indicated that they probably used less water than specified in the instructions on the food bag. In addition, the air entrained in the water would have reduced the amount of water actually introduced. The unused flight food was returned for evaluation. Rehydration of the same food items as used in flight with the proper amount of water was accomplished satisfactorily. It is believed that the slow rehydration of the food was due to the gas entrainment in the water which reduced the amount of water put in the food inflight. The crew also reported that several bite-sized items were broken apart. The tendency for the bite-sized items to stick together was also noted in the unused flight food. Postflight evaluation showed that particular care was required to avoid breaking the overwraps when separating the bites. Once an overwrap was broken, there was a high probability of crumbs being produced.

5.1.10.6 Bioinstrumentation.- The bioinstrumentation equipment performed satisfactorily during this mission, and satisfactory biomedical data were obtained on both pilots. The only discrepancy was the detachment of the Velcro that was there to hold the oral temperature probe onto the lightweight headset.

5.1.11 Landing System

All parachute landing-system events occurred when commanded by the flight crew and were within established tolerances. Figure 5.1.11-1 illustrates the major sequences with respect to the ground elapsed time and pressure altitude at which they occurred. These data correlate very well with the previous missions in which the landing-system sequence was actuated near the nominal drogue parachute deployment altitude of 50 000 feet. The stability of the spacecraft after drogue parachute deployment was similar to that reported on previous missions. The command pilot estimated the oscillations to be approximately ±20 degrees as read off the attitude indicator. This is within design limits of the fully inflated drogue parachute. During spacecraft pickup, the main parachute was lost at sea; however, the Rendezvous and Recovery Section was retrieved and examination of the drogue and pilot parachute assemblies revealed no damage. Examination of all other landing-system components confirmed satisfactory operation.

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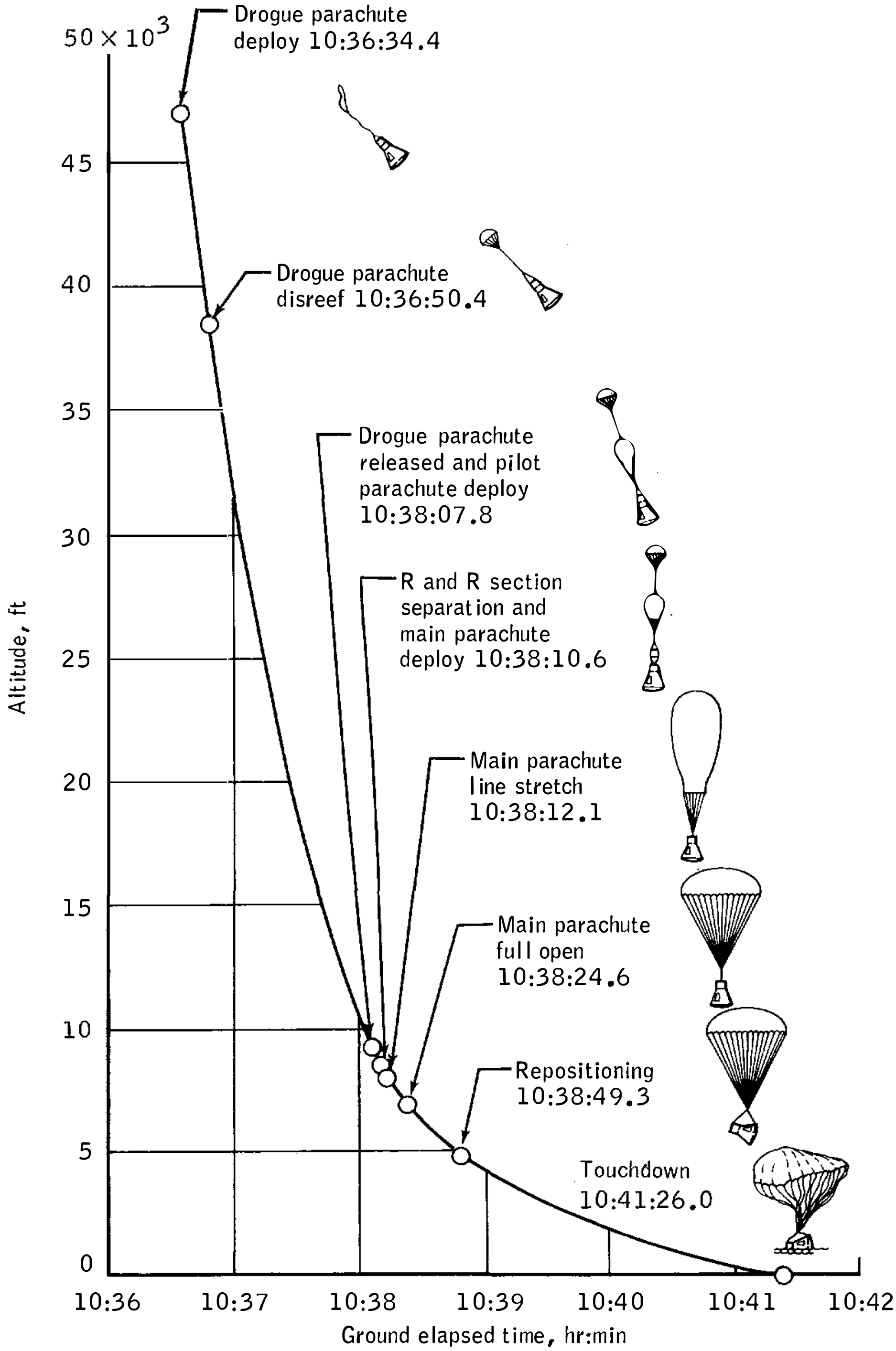

Figure 5.1.11-1. - Landing system performance.

5.1.12 Postlanding

All postlanding and recovery aids functioned properly. The UHF descent and recovery antennas extended when the spacecraft was repositioned to two-point suspension on the main parachute. The sea dye marker was automatically dispensed upon touchdown. The recovery hoist loop and flashing light were deployed when the main parachute was jettisoned by the flight crew. The HF antenna extended and retracted when commanded by the flight crew. All of these functions were verified by recovery crew communications, photographs, and recorded data. The operational effectiveness of the recovery aids is covered in the Communications and Recovery Operations sections of this report (sections 5.1.2 and 6.3).

The spacecraft was damaged in several areas during retrieval operations. Complete details are given in the Recovery Operations and Postflight Inspection sections of this report (sections 6.3 and 12.6).

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5.2 GEMINI LAUNCH VEHICLE PERFORMANCE

The Gemini Launch Vehicle (GLV) was launched on schedule after a countdown that involved no unplanned holds. All systems performed satisfactorily and a satisfactory orbital insertion of the spacecraft was achieved.

5.2.1 Airframe

Flight loads on the launch vehicle were well within its structural capability, showing little effect as a result of the increase in spacecraft weight over previous flights. The vibration and acceleration environments were comparable with those of previous flights.

5.2.1.1 Longitudinal oscillation.- Data indicate the occurrence of the same type of intermittent longitudinal oscillations (POGO) that have been experienced on nearly all previous Gemini launches. Maximum longitudinal oscillations at the spacecraft—launch vehicle interface occurred at lift-off (LO) + 135.5 seconds with an amplitude of +0.22g and a corresponding frequency of 12 cps.

Continuous low-frequency, low-amplitude longitudinal oscillations occurred during Stage II flight. These oscillations, the frequency of which varied from 2.7 to 6.5 cps, reached a maximum amplitude of ±0.45g at the spacecraft—launch vehicle interface at LO + 280 seconds. Although similar longitudinal oscillations have been experienced on previous flights, the amplitudes occurring on this mission were approximately two to three times greater. These oscillations, however, were not sensed by the flight crew and were inconsequential to overall structural loading.

5.2.1.2 Structural loads.- Ground winds gusting to 22 mph induced prelaunch lateral oscillations with a bending moment equal to 46.0 percent of the allowable wind-induced bending moment.

Estimated loads on the launch vehicle are shown in the following table. These data indicate that critical loading occurred, characteristically, at station 320 during the pre-BECO region of flight and reached 78.7 percent of design ultimate load.

Launch vehicle station, in	Maximum qα		Pre-BECO	
	Load, lb	Percent of design ultimate	Load, lb	Percent of design ultimate
276	57 980	58.0	49 130	49.1
320	149 170	43.2	271 340	78.7
935	491 740	67.8	451 780	62.3
1188	510 600	75.9	456 670	68.0

A comparison of Gemini VIII flight loads with previous flights is shown in the following table.

Mission	Launch-vehicle load (percent of design ultimate)	
	Station 935 (maximum qα)	Station 320 (pre-BECO)
Gemini I	66	76
Gemini II	64	80
Gemini III	63	78
Gemini IV	68	81
Gemini V	57	79
Gemini VI-A	61	83
Gemini VII	58	79
Gemini VIII	68	79

5.2.1.3 Post-SECO disturbance.- There were six indications of post-SECO disturbances in both the low-range and the high-range axial-accelerometer data. The times of occurrence and the acceleration levels are given in the following table; all of these occurrences were also noted on rate-gyro data.

Time from SECO, sec	Peak acceleration, g
3.33	3.04
3.79	2.53
4.43	2.19
5.35	1.01
7.12	0.02
[a]34.65	0.42

[a]Spacecraft separation was at SECO + 28.12 seconds.

The crew reported that they did not feel these disturbances. It is believed that the post-SECO disturbances were of sufficiently high frequency (approximately 80 cps) to be attenuated by the launch-vehicle and spacecraft structures; therefore, these disturbances were not felt by the crew.

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5.2.2 Propulsion

Performance of the Propulsion System was satisfactory during the flight. A comparison of preflight-predicted and postflight-reconstructed engine performance is shown in tables 5.2.2-I and 5.2.2-II, and indicates good agreement between predicted and actual performance.

5.2.2.1 Propellant loading and average inflight temperatures.- The following tables provide data on loaded propellant weight and average propellant temperature during flight.

PROPELLANT LOADING

Propellant	Weight, lb			
	Stage I		Stage II	
	Requested	Actual	Requested	Actual
Fuel	89 145	89 243	21 909	21 884
Oxidizer	172 155	172 237	38 491	38 685

AVERAGE PROPELLANT TEMPERATURE

Propellant	Temperature, °F			
	Stage I		Stage II	
	Predicted	Actual	Predicted	Actual
Fuel	44.0	43.2	40.0	41.9
Oxidizer	43.7	42.5	45.2	43.2

Satisfactory agreement between preflight and postflight values was achieved on all parameters.

5.2.2.2 Stage I performance.- Start transients of both Stage I engine subassemblies displayed no anomalies and were in the range of previous GLV and Titan II experience. Data indicate that the oxidizer-pressure-pressurant switch (OPPS) cycled at 1.61 seconds after engine ignition. The switch closed for 7 milliseconds, then opened for 7 milliseconds, then closed and remained closed. Pressure was rising through the switch-actuation pressure of 410 psia when this cycling occurred; however, the instrumentation sampling rate of 40 samples per second does not permit the determination of actual pressure. Since switch interrogation, for engine performance, is performed at 2.2 seconds after engine ignition, well after the start transient, no problem is anticipated in this area on future vehicles. Engine performance during steady-state operation was normal and close to that predicted, as shown in table 5.2-I. Engine shutdown was initiated by fuel exhaustion with approximately 107 pounds of usable oxidizer remaining.

5.2.2.3 Stage II performance.- Performance of the Stage II Propulsion System was close to that predicted, as shown in table 5.2-II. A somewhat slow start of subassembly 3 was indicated by a slow rise in the thrust-chamber pressure. Transient flow rates to the engine were nominal during start, and throughout Stage II flight.

Stage II shutdown was initiated by a command from the Radio Guidance System and was followed by a shutdown-transient total impulse of 35 544 lb-seconds. The predicted shutdown total impulse was 36 100 (±7000) lb-seconds.

5.2.2.4 Performance margin.- Real-time calculations performed during the countdown led to a prediction that the nominal payload capability would exceed the spacecraft weight by 398 pounds. Minimum capability, based on propellant temperature readings just prior to lift-off, was predicted to be -215 pounds. Postflight-reconstructed vehicle performance shows that the achieved vehicle performance was 8830 pounds, or 81 pounds in excess of the nominal preflight-predicted capability. The reconstructed burning time margin was +1.34 seconds.

TABLE 5.2.2-I.- PRELIMINARY STAGE I ENGINE PERFORMANCE

Parameter	Preflight prediction	Postflight reconstruction	Percent difference
Thrust[a], lb	437 453	433 952	-0.80
Thrust (flight average), lb	462 508	461 080	-0.31
Specific impulse[a], lb-sec/lb	261.42	261.42	0.00
Specific impulse (flight average), lb-sec/lb	278.14	278.44	+0.11
Engine mixture ratio[a]	1.9519	1.9416	-0.53
Engine mixture ratio (average between sensors)	1.9443	1.9280	-0.82
Oxidizer flow rate[a], lb/sec	1106.13	1095.33	-0.98
Oxidizer flow rate, (average between sensors), lb/sec	1097.78	1090.04	-0.71
Fuel flow rate[a], lb/sec	567.22	564.65	-0.45
Fuel flow rate, (average between sensors), lb/sec	565.11	565.87	+0.13
Burn time (87FS1 to 87FS2), sec	157.22	157.92	+0.45

[a]Standard inlet conditions

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TABLE 5.2.2-II.- PRELIMINARY STAGE II ENGINE PERFORMANCE

Parameter	Preflight prediction	Postflight reconstruction	Percent difference
Thrust[a], lb	101 542	104 122	+2.54
Thrust (flight average)[b], lb	102 613	102 735	+0.12
Specific impulse[a], lb-sec/lb	311.02	312.06	+0.33
Specific impulse (engine flight average)[b], lb-sec/lb	314.49	314.06	-0.14
Engine mixture ratio[a]	1.8071	1.7657	-2.29
Engine mixture ratio (average between sensors)	1.7680	1.7912	+1.31
Oxidizer flow rate[a], lb/sec	210.34	213.19	+1.36
Oxidizer flow rate, (average between sensors), lb/sec	208.57	210.12	+0.74
Fuel flow rate[a], lb/sec	116.14	120.48	+3.74
Fuel flow rate, (average between sensors), lb/sec	117.72	117.05	-0.57
Burn time (91FS1 to 91FS2), sec	182.90	182.92	-0.01

[a]Standard inlet conditions

[b]Includes roll control nozzle thrust

5.2.3 Flight Control System

Analysis of the flight-control performance revealed satisfactory operation of the primary and secondary systems from lift-off to spacecraft separation. No flight-control hardware anomalies were encountered. The flight was accomplished in the primary mode. Switchover from primary to secondary guidance and control could have been successfully accomplished at any time during powered flight.

5.2.3.1 Stage I flight.- Ignition transients were normal. The peak actuator travel and rate-gyro disturbances recorded during the ignition and holddown period are listed in table 5.2.3-I. The combination of thrust misalignment and engine misalignment at full thrust initiated a roll transient of 2.4 deg/sec at LO + 0.1 second. Proper flight control response damped out this transient in 1.8 seconds. A clockwise roll bias of 0.84 degree was introduced at lift-off by an engine misalignment of 0.17 degree. The open-loop roll and pitch programs were performed as planned and were nominal in rates and duration. All Three Axis Reference System (TARS) discretes were executed within the nominal times and are listed in table 5.2.3-II. Rate and attitude responses of the primary and secondary system correlated well throughout Stage I flight.

The attitude dispersions during the programmed Stage I flight were caused primarily by drift of the TARS gyros or a high wind profile, or a combination of the two. Figure 5.1.5-1 shows the dispersions between the primary and secondary flight-control systems. Table 5.2.3-III lists the maximum rates and attitude errors encountered during Stage I flight.

5.2.3.2 Staging sequence.- The maximum rates and attitude errors were normal during the staging sequence. Maximum rate indications during staging were:

Axis	Primary gyros		Secondary gyros	
	Maximum rates, deg/sec	Time from BECO, sec	Maximum rates, deg/sec	Time from BECO, sec
Pitch	+1.09 -2.53	0.573 .718	+1.37 -2.36	0.708 0.686
Yaw	+2.07 -1.37	.716 .706	+2.65 -2.02	0.718 0.706
Roll	+0.69 -4.16	1.760 .264	+0.60 -4.16	0.002 0.264

Maximum attitude errors from Stage I to Stage II configuration were:

Axis	Attitude errors, deg	Time from BECO, sec
Pitch	+0.406 -0.771	0.417 2.067
Yaw	+1.70 -0.025	2.667 0.017
Roll	+1.438 -0.416	0.067 1.167

5.2.3.3 Stage II flight.-

5.2.3.3.1 Response to radio guidance commands: Radio guidance enable was initiated by the TARS timer at LO + 161.65 seconds. The first pitch command was for 10-percent pitchdown at LO + 168.41 seconds and was followed immediately by a 100-percent pitch-down command for approximately 4.0 seconds. After the initial pitch maneuver, small pitch commands, varying between 6 percent and 8 percent, were continuously transmitted to the vehicle until 330 seconds after lift-off. At that time, a 13-percent pitch-down command was transmitted for approximately 5 seconds. The second-stage cutoff command was transmitted to the vehicle at LO + 337.516 seconds and second-stage engine cutoff (SECO) occurred 0.020 second later.

Response to the first yaw command at LO + 168.41 seconds (a 100-percent command of approximately 1.5-second duration) was an approximate 0.05-degree yaw-left shift. After the termination of this yaw-left command, the transmitted commands were less than 0.02 deg/sec throughout Stage II flight.

Small vehicle disturbances were noted between LO + 245 seconds and LO + 320 seconds. These disturbances created rates of approximately 0.1 deg/sec peak-to-peak in pitch.

5.2.3.3.2 Post-SECO and separation phase: Vehicle rates between SECO and spacecraft separation were normal. The maximum rates experienced between SECO and spacecraft separation are listed in table 5.2.3-IV. Spacecraft separation was accomplished at SECO plus 28.12 seconds.

TABLE 5.2.3-I.- TRANSIENTS DURING STAGE I HOLDDOWN PERIOD

Actuator	Maximum travel		
	Maximum during ignition		Maximum during holddown null check, in.
	Travel, in.	Time from T - 0, sec	
Pitch, 1_1	-0.076 +0.004	-2.517 -3.21	-0.016 +0.004
Yaw/roll, 2_1	+0.196 -0.028	-2.467 -2.367	-0.008 +0.012
Yaw/roll, 3_1	+0.230 -0.017	-2.467 -2.367	+0.006 -0.014
Pitch, 4_1	-0.015 -0.005	-2.417 -2.517	+0.015 -0.005

Axis	Maximum rates			
	Stage I gyro, deg/sec		Stage II gyro, deg/sec	
	Primary	Secondary	Primary	Secondary
Pitch	-0.32 +0.17	-0.20 +0.20	-0.23 +0.25	-0.10 +0.28
Yaw	-0.18 +0.19	-0.24 +0.15	-0.19 +0.29	-0.18 +0.20
Roll	-0.28 +0.29	-0.31 +0.29	-0.28 +0.40	-0.34 +0.44

TABLE 5.2.3-II.- TARS ROLL AND PITCH PROGRAMS

Program	Actual time, LO + sec	Planned time, LO + sec	Rate gyro (average), deg/sec	Torquer monitor, deg/sec	Nominal rates, deg/sec
Roll - start	8.48	8.48	1.25	1.25	1.25
- stop	20.47	20.48	1.25	1.25	
Pitch - step 1					
- start	23.04	23.04	-0.70	-0.694	-0.709
Pitch - step 2					
- start	88.24	88.32	-0.51	-0.500	-0.516
Pitch - step 3					
- start	118.87	119.04	-0.23	-0.250	-0.235
- stop	161.72	162.56			

TABLE 5.2.3-III.- MAXIMUM RATES AND ATTITUDE ERRORS DURING STAGE I FLIGHT

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Axis	Maximum rates, deg/sec				Time from lift-off, sec			
	Primary gyros		Secondary gyros		Primary gyros		Secondary gyros	
	Stage I	Stage II	Stage I	Stage II	Stage I	Stage II	Stage I	Stage II
Pitch	+0.24 -0.99	-0.25 -0.96	+0.30 -0.79	+0.28 -0.90	0.333 65.383	0.983 49.033	105.980 25.083	0.983 33.683
Yaw	+0.67 -0.56	+0.79 -0.48	+0.75 -0.64	+0.79 -0.49	71.980 67.383	72.883 67.383	72.930 67.730	81.733 68.633
Roll	+2.42 -1.54	+3.72 -1.94	+2.48 -1.62	+3.75 -1.76	0.033 9.083	0.033 9.183	0.083 9.080	0.033 9.083

Axis	Attitude errors, deg		Time from lift-off, sec	
	TARS	IGS	TARS	IGS
Pitch	+1.89 -1.26	+2.75 -0.95	108.0 69.5	108.5 70.5
Yaw	+0.89 -1.36	+0.45 -1.85	82.8 70.8	83.0 70.5
Roll	+1.31 –	+1.30 -0.70	108.7 –	20.0 150.0

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TABLE 5.2.3-IV.- VEHICLE RATES BETWEEN SECO AND SPACECRAFT SEPARATION

Axis	Rate, deg/sec
Pitch	
Maximum positive rate at SECO + 1.6 sec	+0.93
Maximum negative rate at SECO + 0.067 sec	-0.35
Rate at SECO + 20 sec	-0.13
Rate at spacecraft separation (SECO + 28.12 sec) . . .	-0.04
Yaw	
Maximum positive rate at SECO + 10.69 sec	+0.39
Maximum negative rate at SECO + 0.897 sec	-0.97
Rate at SECO + 20 sec	+0.29
Rate at spacecraft separation (SECO + 28.12 sec) . . .	+0.29
Roll	
Maximum positive rate at SECO + 1.50 sec	+0.39
Maximum negative rate at SECO + 7.45 sec	-0.38
Rate at SECO + 20 sec	+0.19
Rate at spacecraft separation (SECO + 28.12 sec) . . .	0.00

5.2.4 Hydraulic System

The hydraulic system operated satisfactorily throughout the flight. Significant parameters, reflecting each system's performance, are presented in the following table.

Hydraulic event	Stage I		Stage II system, psia
	Primary system, psia	Secondary system, psia	
Starting transient (maximum)	3370	3340	3910
Starting transient (minimum)	2420		
Steady state	3050	2900	2930
BECO/SECO	2790	2810	2830

During Hydraulic System pressurization with the electric motor-driven pump at T - 375 minutes, the selector valve failed to switch from the secondary system to the primary system. The airborne hydraulic system was not considered affected because both the electric pump and the selector valve are not used in flight. Complete details of the anomaly are discussed in section 5.2.10.

The minimum pressure observed during the Stage I primary-system start transient reflects demands made upon the system by a lift-off roll transient.

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5.2.5 Guidance System

Performance of the Stage I and Stage II guidance systems was satisfactory throughout powered flight and resulted in placing the spacecraft in an acceptable orbit.

5.2.5.1 Programmed guidance.- Programmed guidance as shown by actual and nominal data in table 5.2.3-II is considered within acceptable limits. The trajectory was nominal and the errors at BECO, compared with the no-wind prelaunch nominal trajectory, were 40 ft/sec low in velocity, 2244 feet low in altitude, and 0.12 degree low in flight-path angle.

5.2.5.2 Radio guidance.- The Radio Guidance System (RGS) acquired the pulse beacon of the vehicle, tracked in the monopulse automatic mode, and was locked on continuously from lift-off to 44.0 seconds after SECO. There was a 44.7-second period of intermittent lock until final loss-of-signal at 88.7 seconds after SECO. Track was maintained to an elevation angle of 1.5 degrees above the horizon. The average received signal strength at the central station during Stage II operation was satisfactory. Rate lock was continuous from LO + 29.0 seconds to LO + 392.8 seconds (55.3 seconds after SECO). Rate lock was maintained to an elevation angle of 2.0 degrees above the horizon.

Pitch steering commands were issued, as planned, by the airborne decoder at LO + 168.41 seconds. An initial 10-percent pitch-down steering command (0.2 deg/sec) was given for 0.5 second, followed by the characteristic 100-percent pitch-down steering command (2.0 deg/sec) for 4.0 seconds. Pitch steering at guidance initiate was indicative of a nominal first-stage trajectory. The steering gradually returned during the following 12.09 seconds to relatively small pitch-down commands slowly varying from 0.10 to 0.14 deg/sec. At LO + 250 seconds, because of noisy tracking data, the rates became oscillatory. This particular phenomenon is a normal characteristic of tracking data when the ground guidance system is being influenced by atmospheric effects. Past experience has shown the noise to increase as the tracking elevation angle decreases. As a result, the commands varied between 0.1 to 0.18 deg/sec pitch down until 7.5 seconds before SECO. The pitch-down commands then gradually increased to 0.49 deg/sec, at which time guidance was terminated (SECO - 2.5 seconds). During this final increase, a phase difference was noted in the steering commands from the RGS and the Inertial Guidance System (IGS). That is, the RGS commanded pitch down while the IGS commanded pitch up. This phenomenon that appeared in the pitch steering commands near SECO is attributed to low-frequency tropospheric effects. These effects are not predictable, and cannot be corrected by smoothing in the guidance system, as were the effects of the high-frequency noise experienced in the Gemini V mission. Analysis has shown that, on the Gemini VIII mission, the major contributor to the errors at SECO plus 20 seconds was these tropospheric effects.

Yaw steering was initiated at LO + 168.41 seconds. The commands were indicative of the small dog-legged trajectory (-0.12 degree wedge angle) executed during the second-stage flight. The philosophy behind the dog-legged trajectory, executed on this flight through means of the closed-loop guidance, was to remove the out-of-plane wedge angle (i.e., position error) that existed between the in-orbit target vehicle and the GLV at lift-off. The wedge angle was calculated through means of a prelaunch targeting procedure, which used the target vehicle's real-time ephemeris data to compute the properly biased launch azimuth. The targeting procedure was limited to handle all out-of-plane errors up to a wedge angle of 0.55 degree, although the actual flight setting (finalized at T - 60 minutes before GLV lift-off) was dependent on the prelaunch-determined (T - 2 hours before lift-off) GLV performance. As a result, yaw-left commands of 100 percent (2.0 deg/sec) were sent for a duration of 1.5 seconds. Approximately 7 seconds later, the steering had gradually returned to yaw-left commands of less than 0.02 deg/sec and remained under this value until LO + 330 seconds. At that time, yaw-right commands of up to 0.05 deg/sec were issued until termination of guidance. At SECO + 20 seconds, the yaw velocity was -10.8 ft/sec and the yaw position was -21 187 feet, as compared with the planned values of 0.5 ft/sec and -3921 feet (prelaunch guidance residuals due to insertion targeting accuracies).

SECO occurred at LO + 337.536 seconds at an elevation angle of 6.85 degrees above the horizon. The SECO + 20 second conditions were well within the 3-sigma limits. Table 4.3-I shows a comparison of the actual values with the planned values. The SECO + 20 second errors were attributable to shutdown timing at SECO, TARS gyro drifts, winds, roll-engine misalignment, and noise and biases in the guidance data.

The yaw-position and yaw-velocity errors at SECO resulted in a requirement for the spacecraft to make a 26.2 ft/sec out-of-plane maneuver in the second revolution (see section 4.0). Although the errors were within tolerance, studies are in progress to define a means to further minimize them. After the end of tailoff at SECO + 20 seconds, vehicle rates were 0.13 deg/sec down, 0.29 deg/sec right, and 0.19 deg/sec clockwise, looking forward.

The computing system, in conjunction with the RGS ground and airborne systems, completed all prelaunch and launch operations in a normal and satisfactory manner. The prelaunch transmission and verification of the targeting ephemeris data between the Real-Time Computer Complex at Houston and RGS computing system was also satisfactory. The spacecraft Inertial Guidance System (IGS) ascent updates from the ground computer, transmitted over the spacecraft Digital Command System, were as follows:

Update reference time from lift-off, sec	Update transmission time from lift-off, sec	Value, ft/sec
100.00	105.357	64.88
140.00	145.357	-120.07

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5.2.6 Electrical System

The Electrical System operation was satisfactory from prelaunch power transfer to spacecraft separation. A review of voltage and current levels on the Instrumentation Power Supply (IPS) and the Auxiliary Power Supply (APS) indicated nominal system performance.

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5.2.7 Instrumentation System

5.2.7.1 Ground.- All measurements planned for use performed satisfactorily throughout the countdown and launch. There were 81 landline measurements programmed for use in the launch. No anomalies were experienced and data recovery was 100 percent. The umbilical-release sequence was as planned and was complete in 0.825 second.

5.2.7.2 Airborne.- There were 191 measurements scheduled for use in this launch. The normal data loss at staging lasted 330 milliseconds and no anomalies or unexpected data loss was encountered.

Review of signal-strength records disclosed a complete absence of the signal-strength attenuation seen on the two previous flights. This performance indicates that the modification to the telemetry antenna was instrumental in eliminating the attenuation of the telemetry signal strength.

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5.2.8 Malfunction Detection System

Performance of the Malfunction Detection System (MDS) during pre-flight checkout and flight was satisfactory. Flight data indicated that all MDS hardware functioned properly. MDS parameters are shown in table 5.2.8-I.

5.2.8.1 Engine MDS.- Actuations of the malfunction-detection thrust-chamber pressure switches (Stage I) (MDTCPS) and the malfunction-detection fuel-injector pressure switch (MDFJPS) were as follows:

Switch	Condition	Actuation time from lift-off, sec	Pressure, psia
Subassembly 1 MDTCPS	Make	-2.358	602
	Break	+154.591	585
Subassembly 1 MDTCPS	Make	-2.348	617
	Break	+154.601	545
Subassembly 1 MDFJPS	Make	+155.291	NA
	Break	+337.692	NA

5.2.8.2 Airframe MDS.- The MDS rate-switch package performed properly throughout the flight. No vehicle overrates occurred from lift-off through spacecraft separation.

5.2.8.3 Tank pressure indications.- All tank pressure indicators performed acceptably throughout flight. Both IPS and APS Stage II oxidizer-tank pressure gages on the spacecraft instrument panel were out of calibration during launch vehicle—spacecraft simulated flight tests and during launch. Because there were no abort requirements based on these indications and because the Mission Control Center-Houston (MCC-H) readout of the launch-vehicle transducer was within specification, the discrepancy was waived for flight. The flight crew reported that the response of the spacecraft Stage II oxidizer gages was consistent with the expected flight profile, except that they read low and were offscale low during late Stage II flight. All MCC-H indications of tank pressures were near nominal throughout flight. All A and B sensors, including the Stage II oxidizer sensors, agreed within specification throughout flight.

TABLE 5.2.8-I.- MALFUNCTION DETECTION SYSTEM SWITCHOVER PARAMETERS

	Switchover setting	Maximum or positive (a)	Time from lift-off, sec	Minimum or negative (b)	Time from lift-off, sec
Stage I primary hydraulics	Shuttle spring (1500 psia equiv.)	3356 psi	-2.17	2393 psi	-2.42
Stage I tandem actuators					
No. 1 subassembly 2 pitch	±4.0 deg	1.26 deg	70.7	0.54 deg	82.8
No. 2 subassembly 2 yaw/roll	±4.0 deg	0.42 deg	82.7	1.29 deg	70.7
No. 3 subassembly 1 yaw/roll	±4.0 deg	0.77 deg	70.7	0.86 deg	82.6
No. 4 subassembly 1 pitch	±4.0 deg	0.54 deg	83.0 and 92.7	1.21 deg	69.6
Stage I pitch rate	+2.5 deg/sec -3.0 deg/sec	0.13	106.1	1.00	111.0
Stage I yaw rate	±2.5 deg/sec	0.67	72.4 and 81.9	0.56	67.6
Stage I roll rate	±20 deg/sec	2.42	0.03	2.43	154.9
Stage II pitch rate	±10 deg/sec	0.03	163.5	2.11	171.6
Stage II yaw rate	±10 deg/sec	1.47	156.0	1.85	170.9
Stage II roll rate	±20 deg/sec	0.49	156.3	0.47	155.7

[a]Positive indicates pitch up, yaw right, and roll clockwise.

[b]Negative indicates pitch down, yaw left, and roll counterclockwise.

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5.2.9 Range Safety and Ordnance Systems

The performance of all range-safety and ordnance items was satisfactory.

5.2.9.1 Flight Termination System.- Both GLV command receivers received adequate signal for proper operation throughout powered flight and beyond spacecraft separation.

The following command facilities were used:

Time from lift-off, seconds	Facility
0 to 67	Cape Kennedy - 600W transmitter and single helix antenna
67 to 120	Cape Kennedy - 10kW transmitter and quad helix antenna
120 to 259	Bermuda - 10kW transmitter and steerable antenna
259 to 434	Grand Turk - 10kW transmitter and steerable antenna
434 to 722	Antigua - 10kW transmitter and steerable antenna

5.2.9.2 Range safety tracking system.- Missile Trajectory Measurement (MISTRAM) system I was used as the primary source for impact prediction and provided accurate information through insertion.

5.2.9.3 Ordnance.- The performance of all ordnance items was satisfactory.

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5.2.10 Prelaunch Operations

Propellant loading operations were delayed by spacecraft prelaunch operations and loading started approximately 1 hour late. Loading was complete by 4:22 a.m. e.s.t., approximately 40 minutes ahead of schedule. Total flow time was 3 hours and 33 minutes. This was the first vehicle to be loaded with the new propellant flowmeters which did not have the automatic temperature compensators nor the gear-change adapters. The flowmeters operated as a volumetric (gallons) instead of a weight (pounds) measuring system. During the loading, it was discovered that one of the two parallel flowmeters for the Stage I oxidizer tank had malfunctioned, indicated by an out-of-specification limit at the high-light check point. Consequently, the second flowmeter was used alone to complete the loading.

At 5:30 a.m. e.s.t. (T - 365 minutes), during preparations to perform the flight-control gain checks, the hydraulic-system selector valve failed to respond to a command to pressurize the primary hydraulic system. The valve stopped between the secondary and primary selection point and permitted the electric-driven pump to operate in an unloaded condition. This valve was cycled numerous times in an attempt to repeat the malfunction. However, the valve worked properly each time. It was therefore decided to continue the count. Each subsequent time in the count that this valve was operated, particular attention was paid to its response, but no further difficulty was encountered. It is significant to note that both the electric-driven pump and the selector valve are used only during the ground tests and have no airborne function, although they are airborne equipment.

At 7:35 a.m. e.s.t. (T - 240 minutes), the range countdown was initiated and proceeded to the scheduled 6-minute hold at T - 3 minutes without any unscheduled holds. Only one incident which required additional verification occurred during the range countdown. At T - 62 minutes, during the second guidance command test, the recorder traces reflected an out-of-limit condition. This recorder was being used to verify proper response of the vehicle to guidance commands by monitoring signal levels of the gyros. Subsequent tests indicated that a recorder malfunction may have caused the out-of-specification indication. At T - 35 minutes, a retest was made using the launch-vehicle telemetry and the Hangar T ground-station recorders. A review of these test data revealed that the airborne system was satisfactory, and at T - 12 minutes the system was declared ready for launch.

Postlaunch checks at the GLV contractor's test facilities and at Launch Complex 19 revealed that variations in the recorder power supply could cause trace anomalies similar to those occurring in the launch countdown. Further investigation is continuing.

The scheduled 6-minute hold at T - 3 minutes lasted 5 minutes and 54 seconds and the launch was accomplished as required, at 11:41:02 a.m. e.s.t. A review of films taken during the launch disclosed the spacecraft upper-umbilical connector (spacecraft station Z 156.6) failed to release by the drop-weight system at lift-off. The normal drop-weight release system performed as planned by releasing the weight on receipt of the Missile Operational Countdown System (MOCS) "blow bolt" signal, but the umbilical connector failed to release. As the vehicle lifted from the pad, connector release was accomplished by the static lanyard secured between the spacecraft and the umbilical tower. An investigation of the drop-weight system after the launch revealed the following:

(a) The drop weight was released at the proper time and produced a momentary impulse in the lower lanyard rather than in both the lower and upper one simultaneously. The downward force on the lower lanyard was sufficient only to shear the retaining cup, as verified by the dead facing of the spacecraft half of the umbilical, but not at the proper angle to pull the ground half of the umbilical free of the spacecraft.

(b) The ground half of the umbilical was finally pulled free from the spacecraft by the static lanyard (backup system).

(c) After lift-off the drop weight was lifted out of the guide tube causing damage to the pulley, which indicated improper rigging. Apparently this damage, in conjunction with a piece of butyl tape that jammed in the pulley, impeded the normal travel of the lanyards, thereby softening the impact of the drop weight on the crushable honeycomb at the bottom of the guide tube. On previous launches, the entire honeycomb was compressed about 3/8 inch, whereas on this launch, only a slight impression (that of the bolthead on the bottom of the drop-weight) was made in the honeycomb.

It appears that misrigging of the upper and lower lanyards to the drop weight caused the lower lanyard to exert the major force, thus resulting in an improper pull angle which prevented separation of the plug. The lower lanyard rigging will be changed to provide more slack and insure a positive initial pull by the upper lanyard.

Pad damage was minimal and comparable to that of previous launches. The launch vehicle for Gemini IX was erected on March 24, 1966, eight days after the launch of Gemini VIII.

5.3 SPACECRAFT—GEMINI LAUNCH VEHICLE INTERFACE PERFORMANCE

The various aspects of the spacecraft—Gemini Launch Vehicle interface, as defined in reference 14, performed within established specification limits. The performance of the electrical and mechanical interfacing systems was obtained from launch-vehicle and spacecraft instrumentation and also from crew observation.

The electrical circuitry performed as anticipated. There was no indication of electrical shorting during the spacecraft—launch vehicle separation event. The Malfunction Detection System (MDS) performed satisfactorily. Spacecraft Inertial Guidance System (IGS) steering commands to the launch vehicle were in agreement with the GLV Radio Guidance System, as validated by the GLV telemetry.

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5.4 GEMINI AGENA TARGET VEHICLE PERFORMANCE

The Gemini Agena Target Vehicle (GATV) performed satisfactorily well beyond the required 5-day lifetime. On ascent, a near-perfect 161-nautical-mile circular orbit was attained. During rendezvous and docking, the vehicle responded properly and was very stable.

After spacecraft reentry, the GATV was thoroughly exercised to evaluate the Propulsion System performance and to gain experience and confidence in overall vehicle operation.

All systems functioned throughout the mission. A total of 18 firings were accomplished in orbit — eight Secondary Propulsion System (SPS) Unit I firings, two SPS Unit II firings, and eight Primary Propulsion System (PPS) maneuvers. Vehicle electrical power lasted approximately eight and one-half days. During this period, over 5100 commands were sent, accepted, and properly executed by the command system.

An anomaly was noted during a plane-change maneuver which resulted in the vehicle being unexpectedly translated into a considerably higher orbit. The problem was analyzed and determined to be the result of a center-of-gravity (c.g.) offset from the vehicle centerline in conjunction with a slow-responding control system. The slow-responding control system was incorporated into the GATV to provide the necessary docked stability of the spacecraft—GATV combination. The orbit was later adjusted and the vehicle was parked as planned, in very close to the desired 220-nautical-mile circular orbit. After loss of vehicle electrical power, radar tracking data showed that the vehicle was remaining essentially stable and maintaining very close to orbit rate in pitch.

The performance of the vehicle and its systems, including a discussion of the period covering the docked control problem, is discussed in detail in the following sections.

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5.4.1 Airframe

Structural integrity of the GATV was satisfactorily maintained throughout the launch and orbital phases of flight.

5.4.1.1 Launch phase.- Temperature measurements on the shroud indicated that the maximum temperature reached was 240° F at lift-off (LO) + 130 seconds, corresponding to an altitude of approximately 190 000 feet. The maximum temperature measured on the Target Docking Adapter (TDA) was 270° F at LO + 120 seconds.

Lateral accelerometers on the TDA did not provide valid data during the Mach 1, maximum-q region of flight. Vibration was in excess of the ±1.5g capability of the instruments for approximately 40 seconds. Lateral accelerometers on the aft section indicated 3.4g rms in the Y-axis and 2.2g rms in the Z-axis. Peak axial accelerations were 6.0g at booster engine cutoff (BECO) and 2.84g at sustainer engine cutoff (SECO).

Estimated structural loads are given in the following table. These data indicate that critical GATV loading occurred at station 322 during the maximum qα region of flight.

Station	Maximum qα		Pre-BECO	
	Load, lb	Percent of design ultimate	Load, lb	Percent of design ultimate
248	34 900	46.5	7 570	10.1
322	65 600	50.8	36 700	28.5

5.4.1.2 Separation.- The Target Launch Vehicle (TLV)-GATV separation monitor indicated an average separation velocity of 48 in/sec. This velocity compares with the separation velocity of 40 in/sec recorded during the Gemini VI mission. On this flight, as on Gemini VI, the data provided by the separation monitor were incomplete because of an instrumentation problem (see section 5.4.7).

5.4.1.3 Ascent maneuver.- During the ascent maneuver, there were no abnormal vibrations or accelerations indicated. This period included main-engine ignition, horizon-sensor cover jettison, and shroud separation. All measured temperatures were close to the predicted values. The aft-section temperatures started increasing at main-engine ignition (LO + 376 seconds) with peaks ranging from 124° F for the SPS-module bulkhead temperatures to 260° F for the aft-bulkhead temperatures. These peaks occurred at about main-engine cutoff (LO + 560 seconds). After main-engine cutoff, the temperatures decreased to orbital temperatures.

5.4.1.4 Docking phase.- Docking and undocking operations were indicated by accelerometer data to be quite smooth. During docking, the two lateral TDA accelerometers indicated a disturbance of less than one g peak-to-peak at 06:33:16 spacecraft ground elapsed time (g.e.t.). The longitudinal accelerometer showed nothing at this time but a disturbance was indicated at 06:33:18 g.e.t. Undocking is evidenced at 07:15:11 g.e.t. when these accelerometers again indicated a disturbance of less than one g peak-to-peak. The longitudinal accelerometer produced no significant data during the periods of docking and undocking.

5.4.1.5 Orbital phase.- Accelerations during the orbital phase were reviewed only during the times the Propulsion System was in operation. Lateral accelerometers indicated only low-level vibrations during SPS operation, and 2.69g rms during PPS operation. Axial accelerations during SPS operation were not detectable, but during PPS operation the axial acceleration rose sharply, indicating ignition, then steadily increased as the firing continued. Due to the decreasing weight of the vehicle, these values increased from 0.95g at the start of the ascent PPS firing to approximately 3.6g during the last orbital firing.

The range of airframe temperatures measured during the orbital phase of the mission are indicated in the following table, and are compared with the predicted ranges.

Structural component	Minimum temperature, °F		Maximum temperature, °F	
	Predicted	Actual	Predicted	Actual
TDA	--	20	--	100
Aft bulkhead	28	30	162	120
SPS aft bulkhead	27	10	152	120
Shear panels	31	40	137	90
Radiation shields	-28	10	152	160

These temperatures are outside the predicted range in some cases, but the predictions were always more conservative than the values obtained. Also, a summary of measured radiation-shield temperatures for a typical revolution is shown in figure 5.4.1-1 for comparison with predicted temperatures. These measurements are somewhat higher than predicted, but the cooling trend through the darkness period is obvious.

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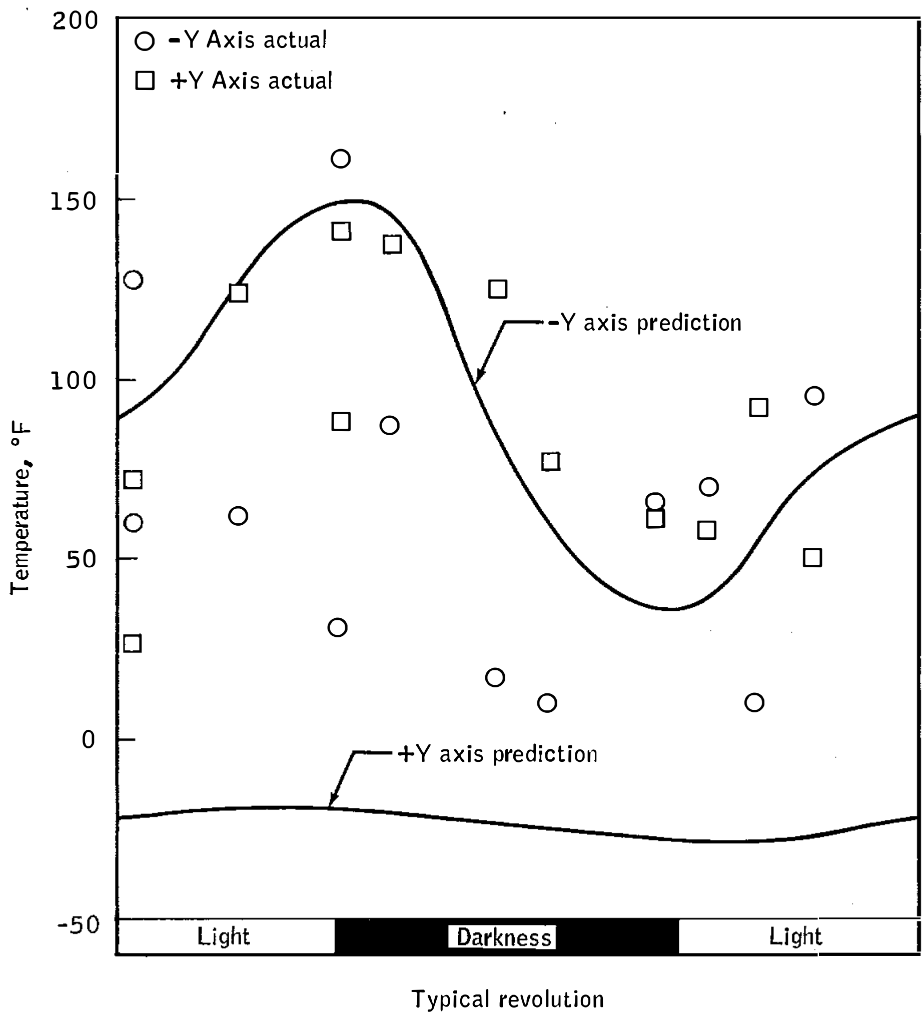

Figure 5.4.1-1. - Radiation shield temperatures (orbital).

5.4.2 Propulsion System

5.4.2.1 Summary.- The performance of the GATV Propulsion System was excellent. A total of 20 firings (9 PPS, 9 SPS Unit I 16-pound thrusters, and 2 SPS Unit II 200-pound thrusters) were made. It should be noted that the eight in-orbit firings of the PPS represent a major increase in demonstrated large-engine flight restarts. Total thrust times of 221, 340, and 72 seconds were achieved for the respective engines. Start transients, steady-state performance, and shutdown impulses were within acceptable limits. Approximately 600 pounds of PPS propellants and 180 pounds of SPS propellants remain in the vehicle.

5.4.2.2 Design.- Section 3.4.2.1 and figure 3.4-1 outline the details of the design changes incorporated in the Propulsion System since the flight of GATV 5002 (Gemini VI mission). In addition, the normal sequence of engine events given in that section can be compared to the actual flight performance.

5.4.2.3 Prelaunch.- All components of the propulsion systems were serviced as required and within prescribed limits prior to launch. Specific propulsion system parameters at the time of launch are noted in tables 5.4.2-I and 5.4.2-II.

5.4.2.4 Launch phase.- All GATV propulsion parameters were nominal during the TLV boost period.

5.4.2.5 GATV ascent firing.- The ascent firing of the GATV engines was excellent insofar as SPS and PPS performance were concerned. A ΔV of 8246.5 ft/sec was planned and achieved. The SPS start valves were opened at LO + 343.0 seconds and, as controlled by the D-timer, SPS ignition was 16 seconds later. SPS firing duration was 20 seconds. Unit I performance was very good with average thrust chamber pressure (P_C) values of 81 and 83 psi for the right (+y) and left (-y) modules, respectively. Thrust-chamber skin temperatures and thrust-chamber pressures (P_C) verified normal operation (figs. 5.4.2-1 and 5.4.2-2). During the ascent phase, vehicle weight was approximately 18 000 pounds, and the 20-second SPS firing imparted approximately 1.14 ft/sec to the vehicle for main-tank propellant orientation.

The PPS fire signal for the ascent firing was at LO + 375.978 seconds and main-engine ignition occurred at LO + 377.082 seconds (fig. 5.4.2-3). Data indicate a nominal first-firing performance, in that the actual thrusting period (from 75-percent P_C to velocity meter cutoff) was 183.33 seconds as compared to a predicted 183.50 seconds. The start transients were as predicted, and there was no evidence of a hard or abnormal start. In fact, the start dynamics appeared to be mild, within the limitations of the telemetry data to determine the start characteristics. Steady-state operation was achieved within 20 seconds and the average P_C value was 518 psi. The steady-state average turbine speed was 24 890 rpm, compared with the expected value of 24 700 rpm. Engine shutdown was at LO + 560.402 seconds. The engine tailoff, as measured by velocity meter readings, was 2980 lb-seconds, which is a normal value. Details of the ascent and subsequent Propulsion System operations are shown in tables 5.4.2-III, 5.4.2-IV, and 5.4.2-V, and in parts (a) through (i) of figure 5.4.2-3.

During the ascent PPS operation at approximately LO + 460 seconds, a vehicle roll-rate transient peaking at 0.3 deg/sec was seen but was countered by the Attitude Control System (ACS) until the end of the first maneuver. At this time the apparent driving force ceased. This activity is believed to have been caused by a slight thermal distortion of the turbine exhaust duct as has been observed on other Agena flights. This roll torque is not considered a mission problem except for the small amount of control-gas usage.

5.4.2.6 Pressurization system.- Operation of the pressurization system during the first PPS thrust period, which is the only active pressurization period of an Agena flight, was as planned and no anomalies were noted. A record of tank pressures during the ascent maneuver is shown in figure 5.4.2-4. The initial firing of the pyrotechnic valve, allowing source pressure to enter the propellant tanks, occurred at LO + 376.4 seconds; and the second valve, which isolates the oxidizer tank from the pressurization system and from the fuel tank, was fired at 314 seconds after PPS ignition. As is normal, the temperature of the source tank decreased rapidly during the pressurization period and reached a value of -50° F at LO + 570.0 seconds.

The propellant feed system, which includes the lines and connectors for filling the main tanks and the lines and propellant isolation valves (PIV) feeding the PPS, operated satisfactorily. For the ascent firing, the PIV's are open at launch. The PIV's opened as planned with each in-orbit firing. An average time of 1.50 and 0.90 seconds for oxidizer and fuel, respectively, was required between the PIV ACTUATE electrical signal and the time of full pump-inlet pressure. Fully-open PIV times were about 2.2 seconds for oxidizer and 0.95 seconds for fuel. Significant temperature drops were not observed at the pump inlets during the postfire venting process of the PIV's, and in no case was an undesirable effect indicated by the subsequent engine-start transients. The time between firings varied from 3 hours 8 minutes to 12 hours 13 minutes.

5.4.2.7 Primary Propulsion System orbital operations.- Propellant tank temperatures and start-tank temperatures varied in flight as shown in figure 5.4.2-5 and were generally within expected ranges. The propellants remained above +40° F in all cases. The contractor had predicted bulk propellant temperatures of 39° to 58° F. As discussed previously and tabulated in tables 5.4-III through 5.4-V, the PPS was fired eight times in orbit. All propulsion parameters were within acceptable limits and main-engine performance was very repeatable. Based on the available telemetry data, all starts were satisfactory and there were no rough or hard-start tendencies. Oxidizer preflow (table 5.4-III) appears to have been in the range of 6 to 8 pounds. Steady-state performance for all engine firings was nominal. It should be noted that because of the bootstrapping time required for the turbopump, a true steady-state condition does not occur on short firings of less than 20 seconds. In addition, the start transient greatly affects the averaging values of very short firings of 3 seconds or less. Shutdown tail-off impulses were very consistent, with an average value of 2815 lb-seconds. PPS thermal data during the first few days of the mission indicated the expected minor solar-heating cyclic behavior. Most values were as anticipated (fig. 5.4.2-5). The measured PPS nozzle-extension skin-temperature variation was somewhat greater than expected and indicated that a low temperature of -120° F was reached in the area of this transducer. This temperature is below the design-limit temperature of -70° F. An investigation will be conducted to determine the validity of the measurement as well as the impact of this temperature (if valid) on the structure of the nozzle extension.

5.4.2.8 Secondary Propulsion System orbital operations.- The SPS Unit I and Unit II engines were utilized in orbit for a total of 310 and 72 seconds of operation, respectively. In the eight Unit I and two Unit II firings, no propulsion problems were observed and both modules performed as planned. Tables 5.4-IV and 5.4-V summarize the SPS function in flight. Preceding each main-engine operation, the SPS Unit I engines fired for either 20 or 70 seconds, depending on whether PPS start-mode A or C was selected. In every case the chamber pressure and thrust-chamber skin-temperature measurements indicated performance within specification (fig. 5.4.2-6). Six A-type starts and three C-type starts were made. No B-type starts (36 seconds of Unit I operation) were made during this mission. A decision was made to use short A-type SPS firings with PPS starts because of an apparent excessive use of ACS gas during SPS Unit I firings. This excessive use of gas is under investigation.

Operation of the Unit II engines during 21-second and 51-second firings was as expected, and it appeared that there was no excessive ACS gas usage during these firings.

Heating of the SPS +Y right module by the PPS turbine exhaust duct and heating of the PPS nozzle extension by the SPS were noted in several cases. In observing the engine thermal transients, certain cases of higher-than-expected skin temperatures on the left Unit I module were noted during the thrust periods. This item is also under investigation. However, the unit stayed well within allowable thermal limits. During the coast periods, the temperatures of SPS components were quite stable. The engine bi-propellant valves did not encounter excessive heating due to postfire heat soak back. As noted on the propulsion temperature transient plots (fig. 5.4.2-7), a significant shift in all aft-rack heating rates was noted while the GATV was in a highly elliptical orbit.

TABLE 5.4.2-I.- PPS PRELAUNCH PARAMETERS

Helium-sphere pressure, psia	2600
Helium-sphere temperature, °F	61
Fuel-tank pressure, psig	40
Fuel-tank temperature (estimated bulk temperature), °F	44.9
Weight of fuel loaded, lb	3818
Oxidizer-tank pressure, psig	30
Oxidizer-tank temperature (estimated bulk temperature), °F	46.0
Weight of oxidizer loaded, lb	9702
Start-tank pressure (fuel), psia	1001
Start-tank pressure (oxidizer), psia	990
Start-tank temperature (fuel), °F	47
Start-tank temperature (oxidizer), °F	46
Fuel-pump inlet temperature, °F	51
Oxidizer-pump inlet temperature, °F	51

TABLE 5.4.2-II.- SPS PRELAUNCH PARAMETERS

Parameter	Engine location	
	+Y	-Y
Nitrogen pressure, psig	4031	4117
Nitrogen manifold pressure, psig	177.1	182.1
Fuel manifold pressure, psig	180.4	182.1
Oxidizer manifold pressure, psig	184.2	185.0
Fuel manifold temperature, °F	62.6	66.0
Oxidizer manifold temperature, °F	62.6	62.6
Weight of fuel loaded, lb	79.12	79.18
Weight of oxidizer loaded, lb	88.75	88.38

TABLE 5.4.2-III.- PRIMARY PROPULSION SYSTEM DATA[a]

Start-transients data					
PPS maneuver number[b]	Ascent	1	2	3	4
Network station	ETR	CRO	HAW	CYI	TEX
Pump inlet temperature (o/f) 0.5 sec prior to FS, °F	51.2/52.5	56.1/61.5	57.4/60.2	53.9/53.9	69.7/80.2
Pump inlet temperature (o/f) 1.5 sec after FS, °F . . .	47.6/48.7	56.1/56.3	53.7/57.6	53.9/53.9	64.0/80.2
Pump inlet temperature (o/f) 5.0 sec after FS, °F . . .	46.4/46.2	56.7/61.5	53.7/58.9	53.9/53.9	64.0/69.7
Firing duration (75% P_c to B-108 cutoff), sec	183.317	1.174	1.147	19.250	0.785
Time of PPS FS, g.e.t. . . .	[c]375.978	21:43:55.532	27:04:43.139	39:16:45.638	44:02:30.949
Time, FS to FGGV open, sec	0.055	0.089	0.090	0.091	0.067
Time, FS to OGGV open, sec	0.055	0.078	0.050	0.048	0.056
Time, FS to TMP rise, sec	0.243	0.278	0.241	0.231	0.246
Time, FS to oxid valve open, sec	0.442	0.414	0.383	0.353	0.454
Time, FS to OMP or OFP switch make, sec	0.893	0.856	0.863	0.859	0.822
Time, FS to both switches make, sec	0.914	0.873	0.895	0.887	0.849
Time, FS to fuel valve open, sec	1.019	0.939	1.003	0.972	0.919
FVAP at time of fuel valve open, psia	[d]560	[d]443	[d]422	[d]430	[d]543
FVIP at time of fuel valve open, psia	854.3	878.7	854.3	854.3	903
Time, FS to ignition, sec	1.082	0.983	1.028	1.020	0.956

[a]As taken from tabulated data. Not corrected for transducer delays.

[b]List of abbreviations follows table.

[c]Time from GAATV lift-off, sec.

[d]Extrapolated data.

TABLE 5.4.2-III.- PRIMARY PROPULSION SYSTEM DATA[a] - Continued

Start-transients data				
PPS maneuver number[b]	5	6	7	8
Network station	ASC	ANT	TAN	RKV
Pump inlet temperature (o/f) 0.5 sec prior to FS, °F	65.4/72.6	69.7/81.8	71.2/78.6	66.8/71.7
Pump inlet temperature (o/f) 1.5 sec after FS, °F	60.0/66.9	62.7/71.2	64.0/71.7	60.2/65.5
Pump inlet temperature (o/f) 5.0 sec after FS, °F	60.0/61.5	62.7/65.5	58.7/65.5	54.9/60.2
Firing duration (75% P_c to B-108 cutoff), sec	8.054	2.499	2.150	2.625
Time of PPS FS, g.e.t.	47:39:38.955	50:47:11.707	54:39:27.578	59:28:19.340
Time, FS to FGGV open, sec	0.088	0.073	[e]0.093	0.089
Time, FS to OGGV open, sec	0.076	0.042	[e]0.082	0.042
Time, FS to TMP rise, sec	0.238	0.231	[e]0.230	0.278
Time, FS to oxid valve open, sec . . .	0.349	0.315	(e)	0.414
Time, FS to OMP or OFP switch make, sec	0.865	-	0.839	0.875
Time, FS to both switches make, sec	0.910	0.848	0.875	0.912
Time, FS to fuel valve open, sec . . .	0.999	0.934	0.984	0.980
FVAP at time of fuel valve open, psia	[d]422	[d]487	[d,e]434	[d]426
FVIP at time of fuel valve open, psia	854.3	866.5	[e]866.5	854.3
Time, FS to ignition, sec	1.023	0.973	0.991	1.018
Time to 75% thrust, sec	1.040	0.982	1.019	1.045
Time to OVIP recharge start, sec . . .	0.889	0.844	[e]0.832	0.838
Time to FVIP recharge start, sec . . .	0.743	0.834	[e]0.781	0.850
Prefire FSP, psig	47.9	44.7	(e)	45.5
Prefire OSP, psig	26.4	25.6	(e)	24.8

[a]As taken from tabulated data. Not corrected for transducer delays.

[b]List of abbreviations follows table.

[d]Extrapolated data.

[e]Telemetry data in question.

TABLE 5.4.2-III.- PRIMARY PROPULSION SYSTEM DATA[a] - Continued

Start-transients data - Continued					
PPS maneuver number[b]	Ascent	1	2	3	4
Network station	ETR	CRO	HAW	CYI	TEX
Time, FS to 75% thrust, sec	1.107	0.996	1.040	1.038	0.965
Time, FS to OVIP recharge start, sec	0.838	0.860	0.871	0.861	0.900
Time, FS to FVIP recharge start, sec	0.778	0.740	0.716	0.777	0.786
Prefire FSP, psig	53.6	51.1	51.1	51.1	46.3
Prefire OSP, psig	44.0	30.4	29.6	28.8	27.2
Prefire FVIP, psia	1001	1100.3	1038.6	1038.6	1173.5
Prefire OVIP, psia	989	1100.3	1013.9	1013.9	1161.4
TMP_c peak/steady state average, psig	519/455	567/420	539/403	577/455	603/421
P_c average, psia	518	492	487	513	490
Estimated oxidizer preflow (±1 lb), lb	8.6	6.4	7.8	7.6	6.7
Time of PPS SS, g.e.t. . . .	[c]560.402	21:43:57.702	27:04:45.314	39:17:05.908	44:02:32.699
Postfire data					
Time, SS to P_c decay, sec	0.029	0.087	0.056	0.027	0.027
Time, SS to TMP_c decay, sec	0.142	0.139	0.157	0.125	0.121
Time, SS to FGGV close, sec	0.213	0.139	0.199	0.134	-
Time, SS to OGGV close, sec	0.140	0.127	0.186	0.123	0.119
Time, SS to oxidizer valve close, sec	1.271	1.071	-	1.348	1.031
Postfire OVIP, psia	1124	1026	1013	1112	1001
Postfire FVIP, psia	1100	1051	1151	1149	1038
Postfire OSP, psig	17.6	29.6	29.6	25.6	26.4
Postfire FSP, psig	35.9	50.3	51.1	46.3	47.9

[a]As taken from tabulated data. Not corrected for transducer delays.

[b]List of abbreviations follows table.

[c]Time from GAATV lift-off, sec.

TABLE 5.4.2-III.- PRIMARY PROPULSION SYSTEM DATA[a] - Concluded

Start-transients data - Continued				
PPS maneuver number[b]	5	6	7	8
Network station	ASC	ANT	TAN	RKV
Prefire FVIP, psia	1051.0	1124.8	1075.6	1051.0
Prefire OVIP, psia	1013.9	1100.3	1038.6	1013.9
TMP_c peak/steady state average, psig	543/445	597/416	567/410	579/415
P_c average, psia	505	495	496	498
Estimated oxidizer preflow (±1 lb), lb	7.4	7.8	7.3	7.0
Time of PPS SS, g.e.t.	47:39:48.039	50:47:15.198	54:39:30.747	59:28:23.010
Postfire data				
Time, SS to P_c decay, sec	0.043	0.042	0.036	0.027
Time, SS to TMP_c decay, sec	0.140	0.115	0.134	0.129
Time, SS to FGGV close, sec	0.151	0.124	0.112	0.139
Time, SS to OGGV close, sec	0.140	0.082	0.101	0.127
Time, SS to oxidizer valve close, sec	1.394	1.182	1.263	1.352
Postfire OVIP, psia	1100.3	1063.3	[e]1038	1038.6
Postfire FVIP, psia	1124.8	1087.9	[e]1063	1087.9
Postfire OSP, psig	24.8	24.8	24.0	22.4
Postfire FSP, psig	45.5	44.7	45.5	39.9

[a]As taken from tabulated data. Not corrected for transducer delays.

[b]List of abbreviations follows table.

[e]Telemetry data in question.

LIST OF ABBREVIATIONS FOR TABLE 5.4.2-III

FGGV	Fuel gas generator valve
FS	Fire signal
FSP	Fuel suction pressure
FVAP	Fuel valve actuation pressure
FVIP	Fuel venturi inlet pressure
g.e.t.	Ground elapsed time
o/f	Oxidizer/fuel
OFP	Oxidizer feed pressure
OGGV	Oxidized gas generator valve
OMP	Oxidizer manifold pressure
OSP	Oxidizer suction pressure
OVIP	Oxidizer venturi inlet pressure
PPS	Primary Propulsion System
P_c	Chamber pressure
SS	Shutdown signal
TMP	Turbine manifold pressure

TABLE 5.4.2-IV.- SPS UNIT I PERFORMANCE

Number	Ascent	1	2	3	4	5	6	7	8
Start time, g.e.t.	[a]0:05:58.0	21:42:47.4	27:03:35.0	39:16:25.6	44:01:22.81	47:39:18.9	50:46:51.7	54:39:07.52	59:27:59.3
90 percent P_c time, sec									
+Y	0.431	0.274	0.274	0.368	0.280	0.274	0.305	0.280	0.243
-Y	0.271	0.209	0.240	0.209	0.215	0.208	0.209	0.210	0.239
P_c average, psia									
+Y	81.5	76.9	79.2	79.9	77.0	80.1	78.2	79.9	79.0
-Y	81.9	77.4	81.0	79.5	78.1	81.0	78.9	79.3	81.1
Tank pressure, psia									
+Y	214.0	202	209.2	209.2	203.0	211.6	206.8	209.0	209
-Y	222.8	206	218.1	211.0	210.0	218.1	213.3	215.5	218
Cutoff time, g.e.t. . .	[a]0:06:17.9	21:43:57.6	27:04:45.1	39:16:47.6	44:02:32.8	47:39:40.9	50:47:13.7	54:39:29.6	59:28:21.4
Propellant temperature, °F									
Oxidizer +Y	63	71	71	71	76	72	72	69	69
Oxidizer -Y	63	63	59	64	61	66	66	70	63
Fuel +Y	63	76	77	79	88	77	77	60	72
Fuel -Y	66	66	64	66	85	70	71	80	66

[a]Times are minutes and seconds from GAATV lift-off.

TABLE 5.4.2-V.- SPS UNIT II PERFORMANCE

Start time, g.e.t.	64:30:46.79	67:38:47.90
90 percent P_c time, sec		
+Y	0.125	0.150
-Y	0.125	0.115
P_c average, psia		
+Y	94.0	93.8
-Y	93.6	93.3
Tank pressure, psia		
+Y	197.0	197.1
-Y	199.5	199.0
Cutoff time, g.e.t.	64:31:07.79	67:39:38.90
Propellant temperature, °F		
Oxidizer +Y	45	66
Oxidizer -Y	66	64
Fuel +Y	68	66
Fuel -Y	68	66

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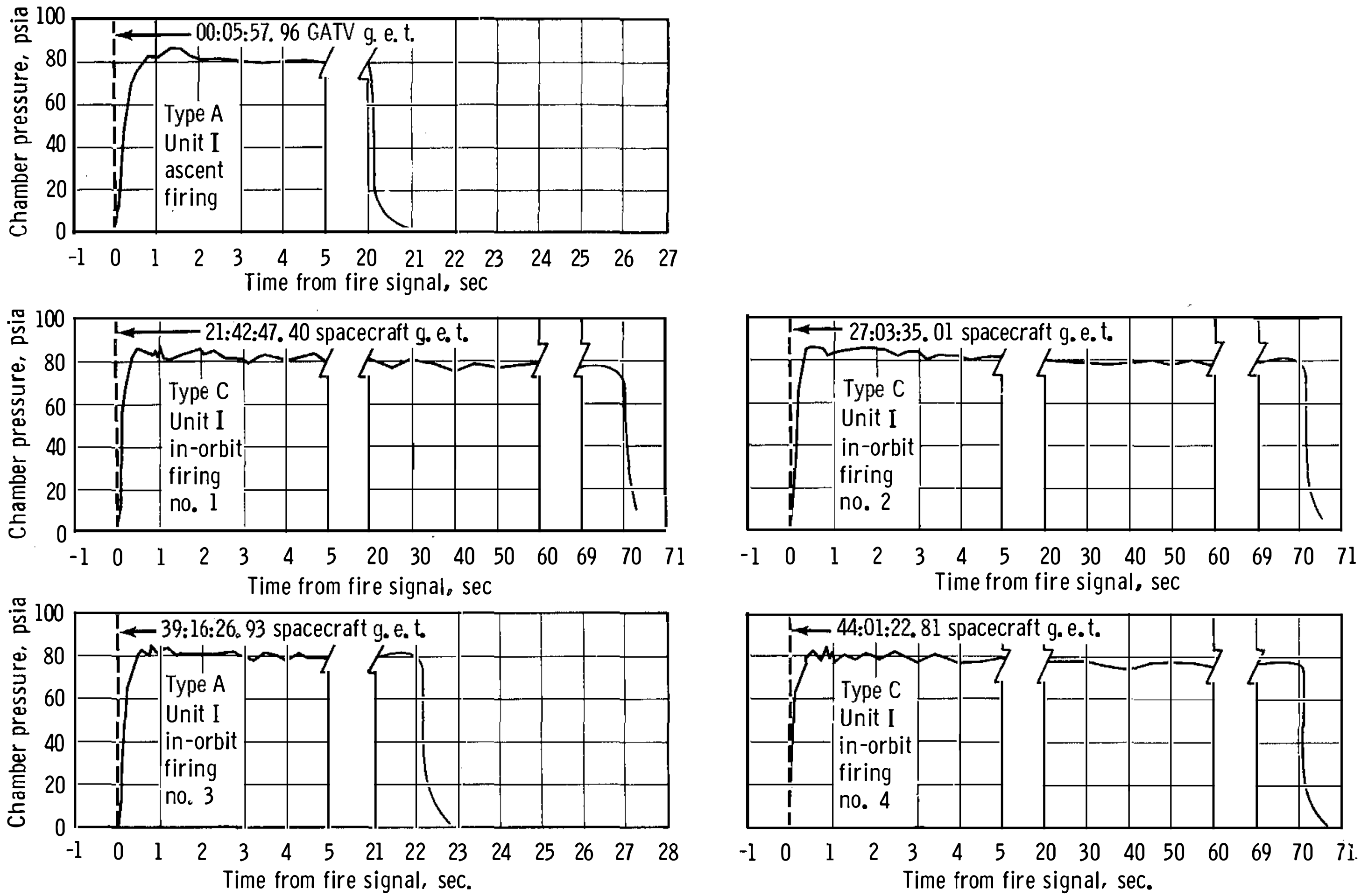

(a) Unit I firings ascent through no. 4.

Figure 5.4.2-1. - SPS +Y chamber pressure traces.

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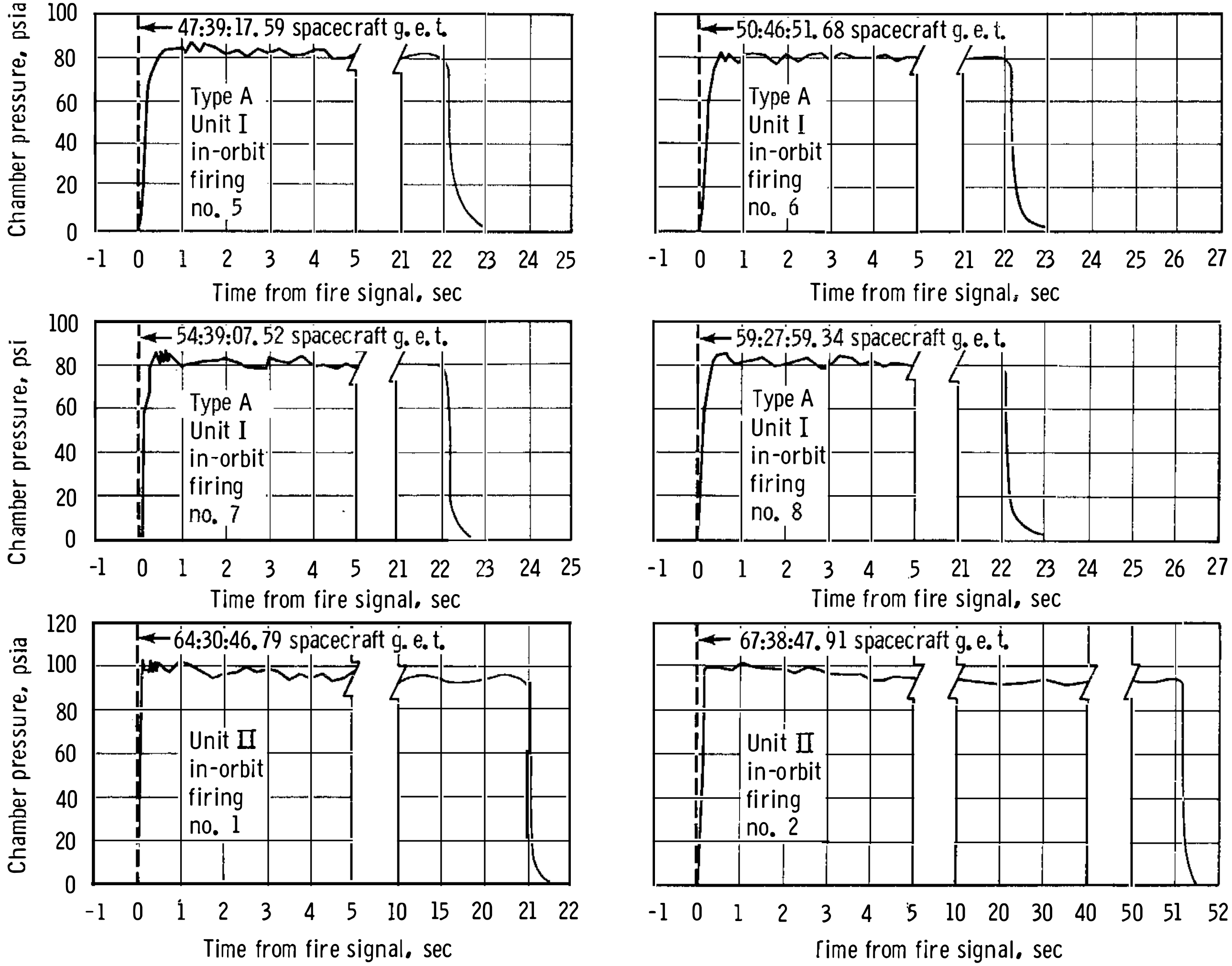

(b) Unit I engine firings no. 5 through no. 8, and unit II engine firings no. 1 and no. 2.

Figure 5.4.2-1. - Concluded.

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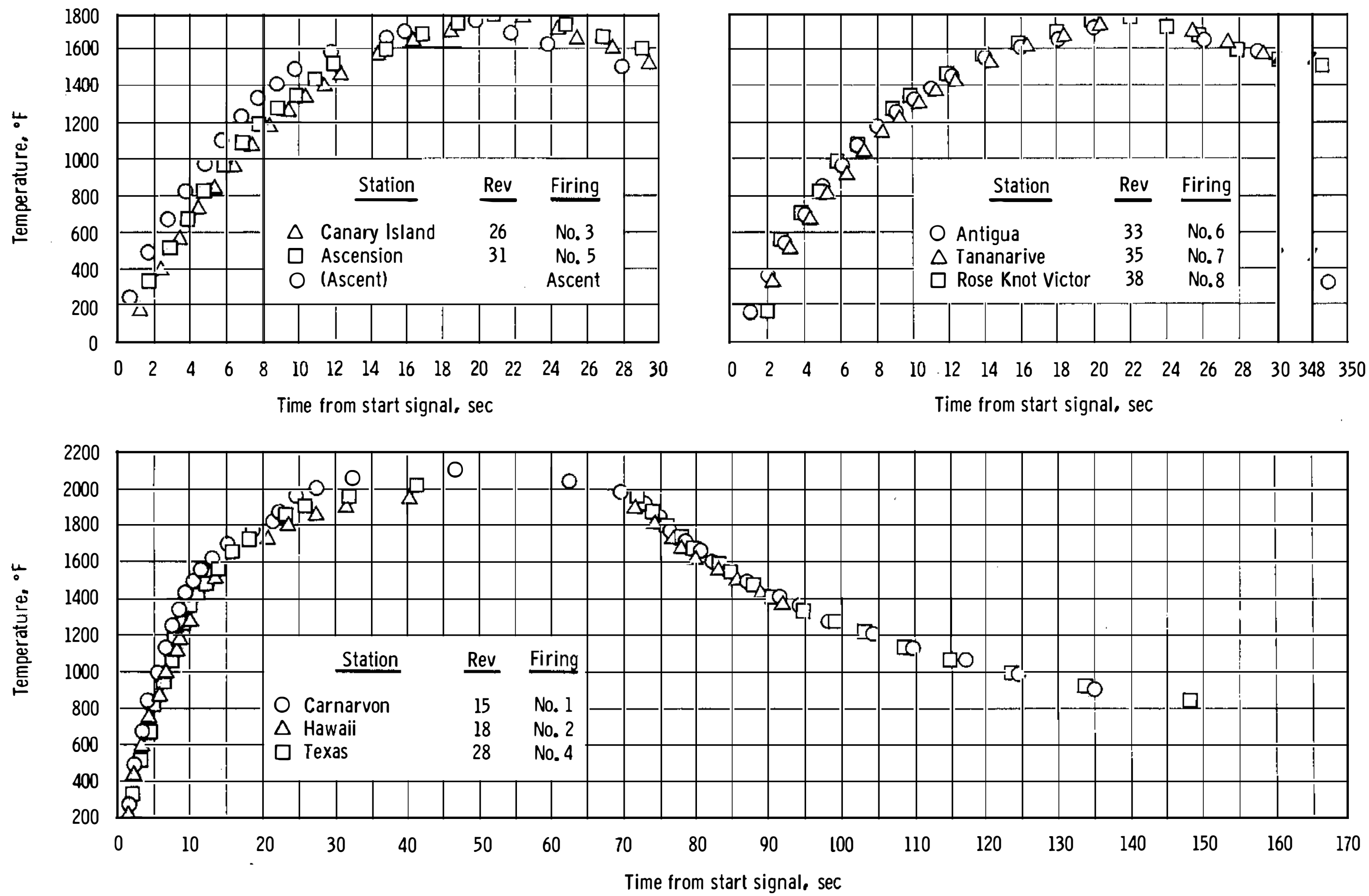

(a) +Y Unit I

Figure 5.4.2-2. - SPS skin temperature.

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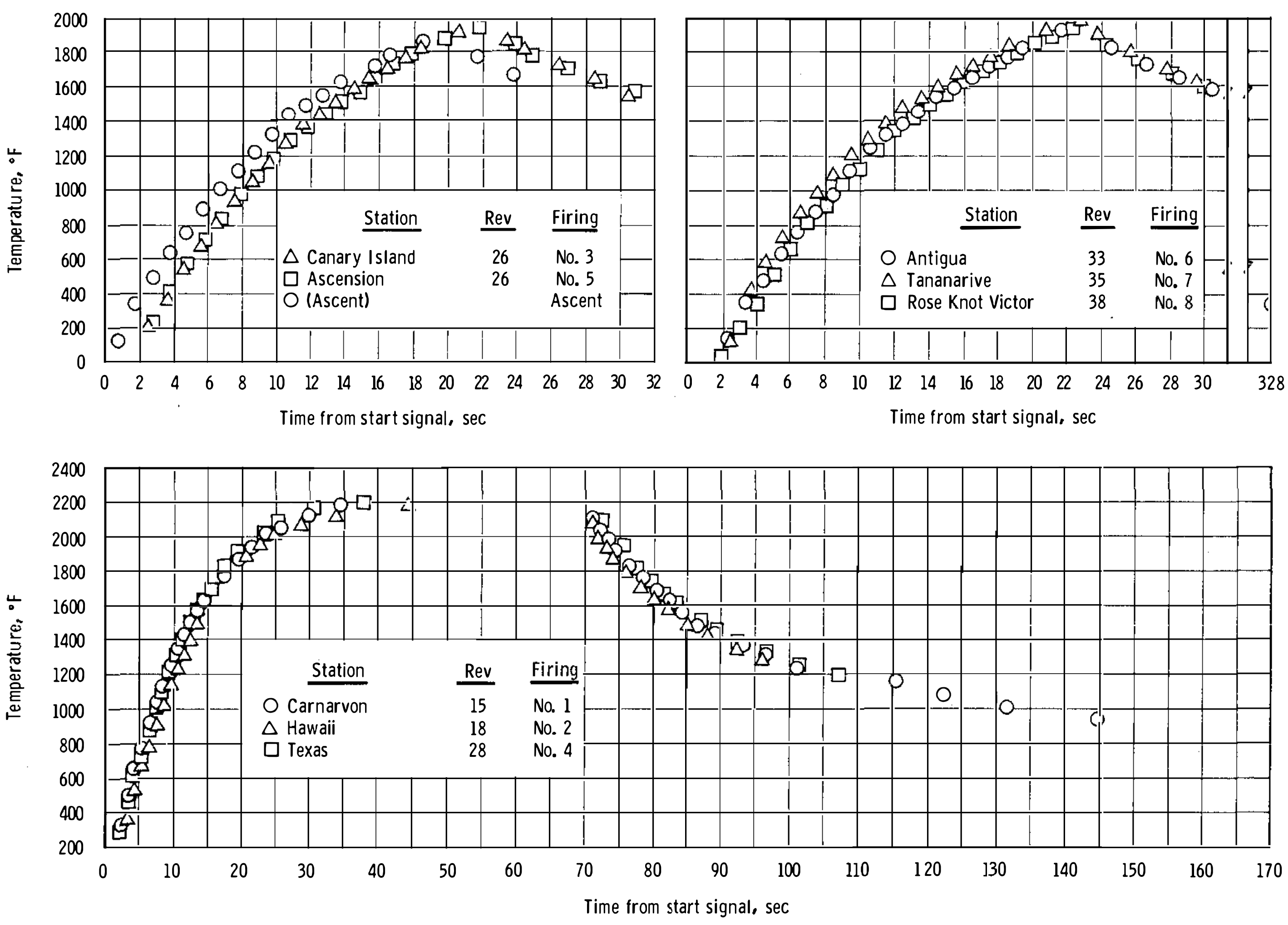

(b) -Y Unit I

Figure 5.4.2-2. - Continued.

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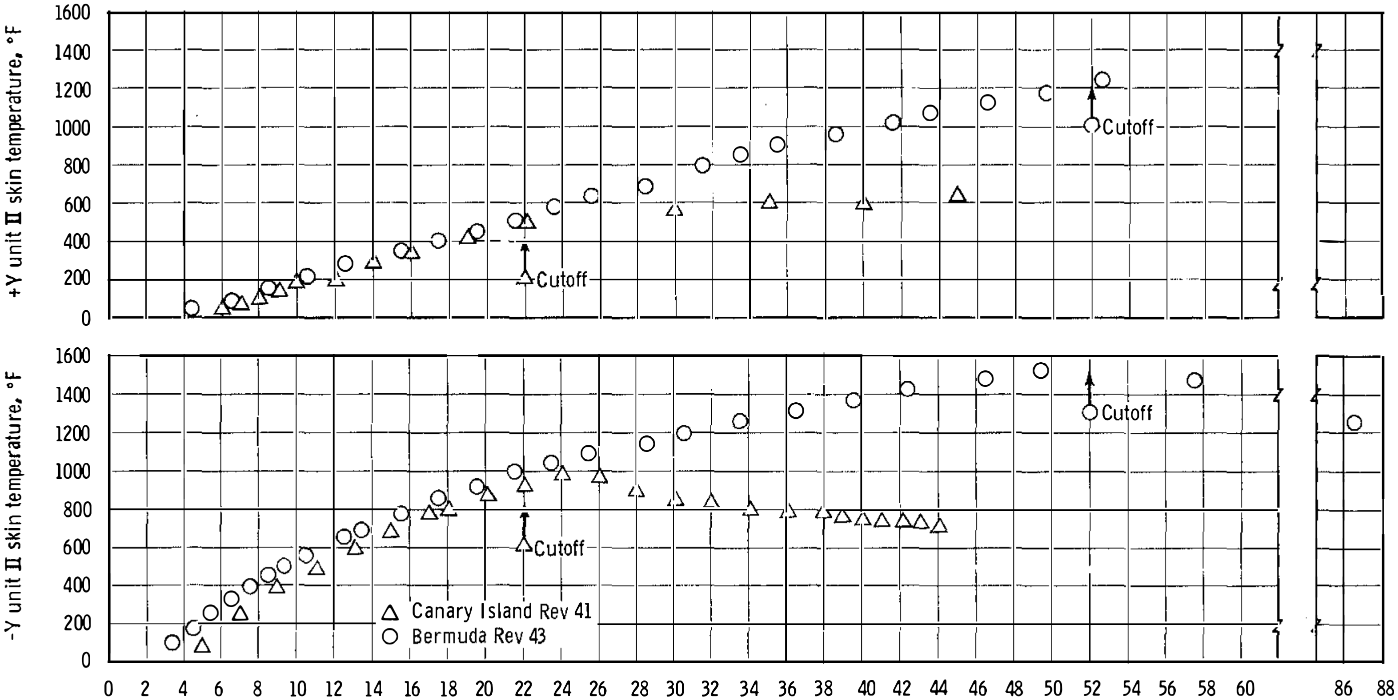

(c) Unit II

Figure 5.4.2-2. - Concluded.

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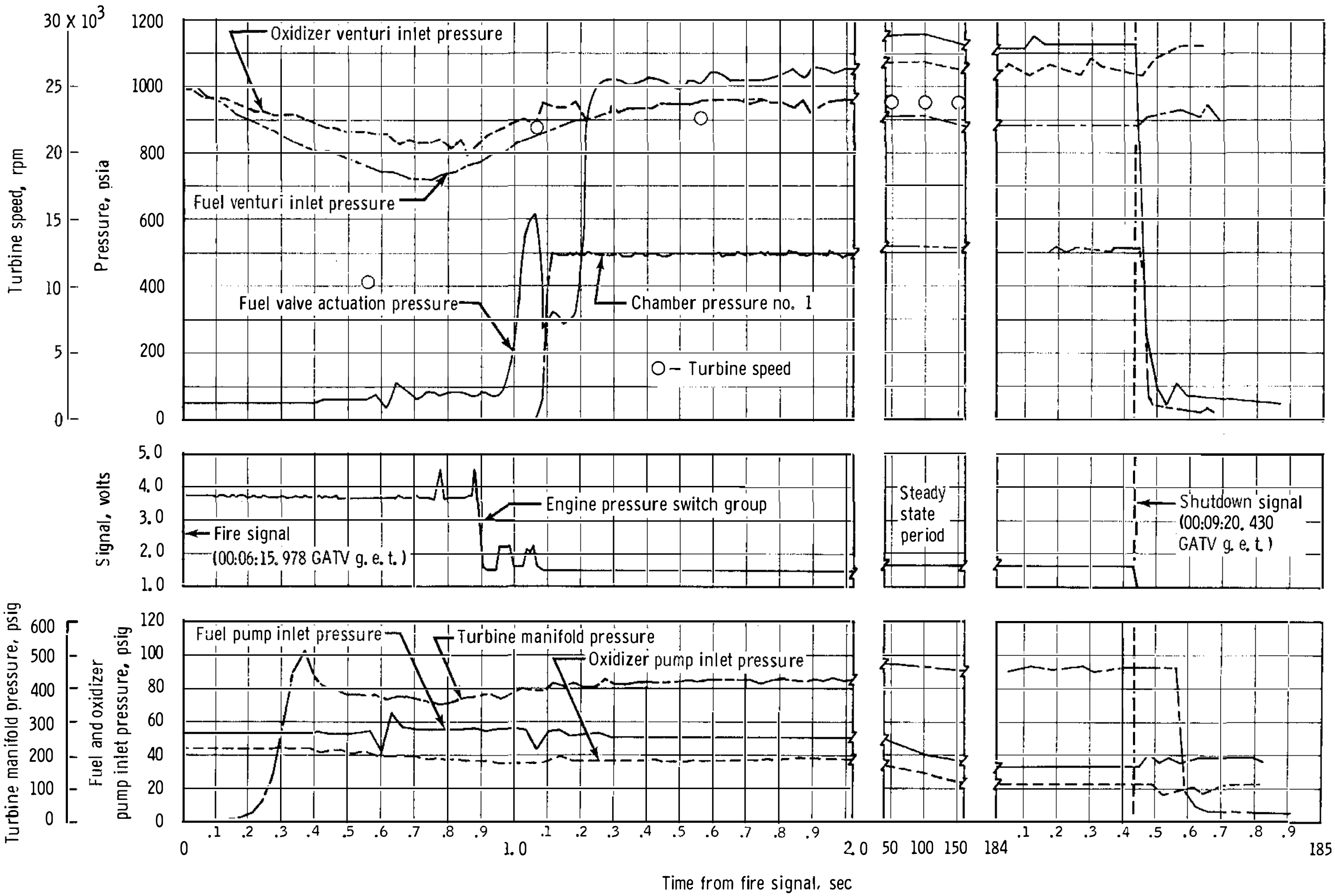

(a) Ascent maneuver

Figure 5.4.2-3. - PPS Performance transients.

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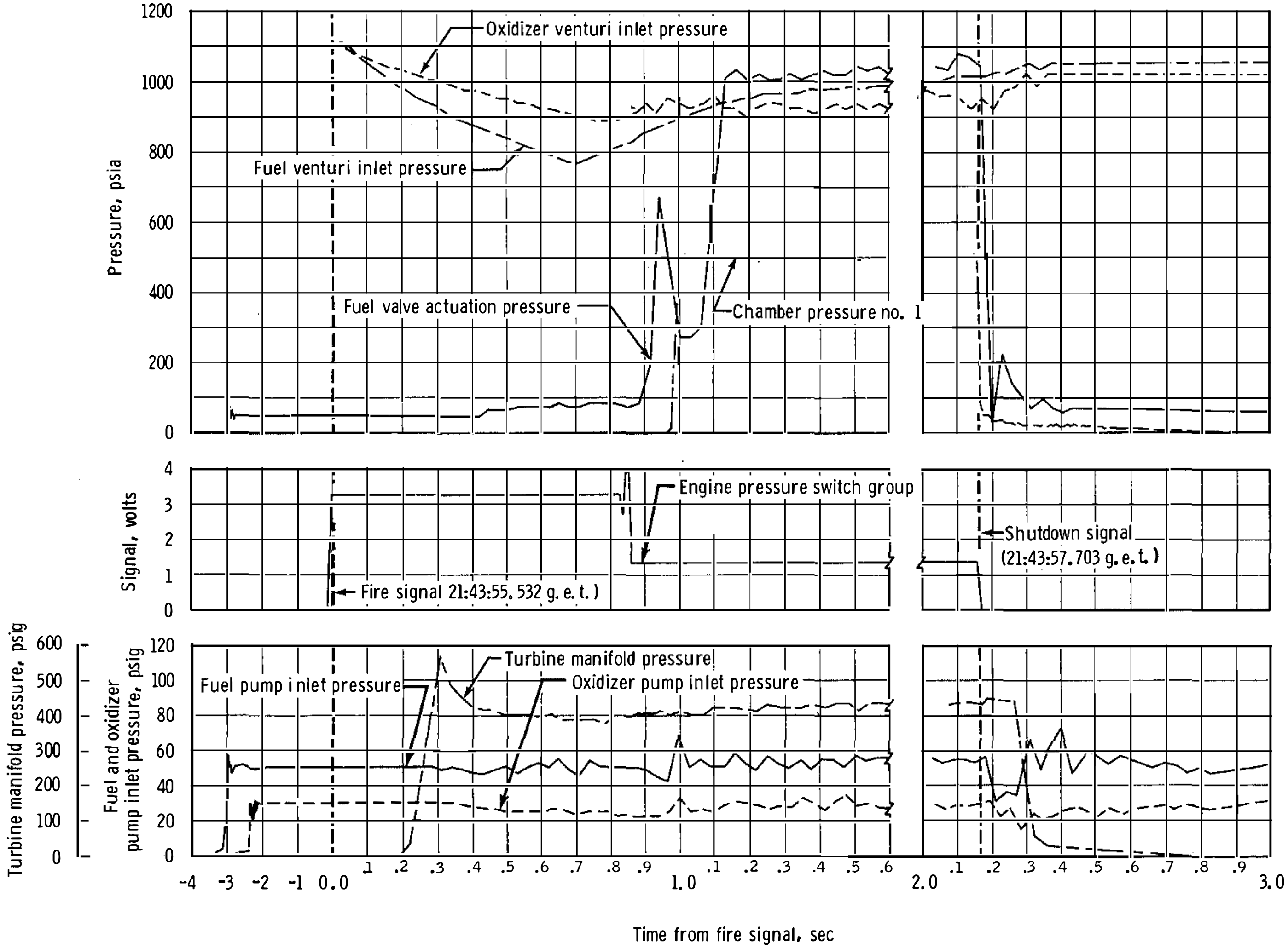

(b) In orbit maneuver no. 1

Figure 5.4.2-3. - Continued.

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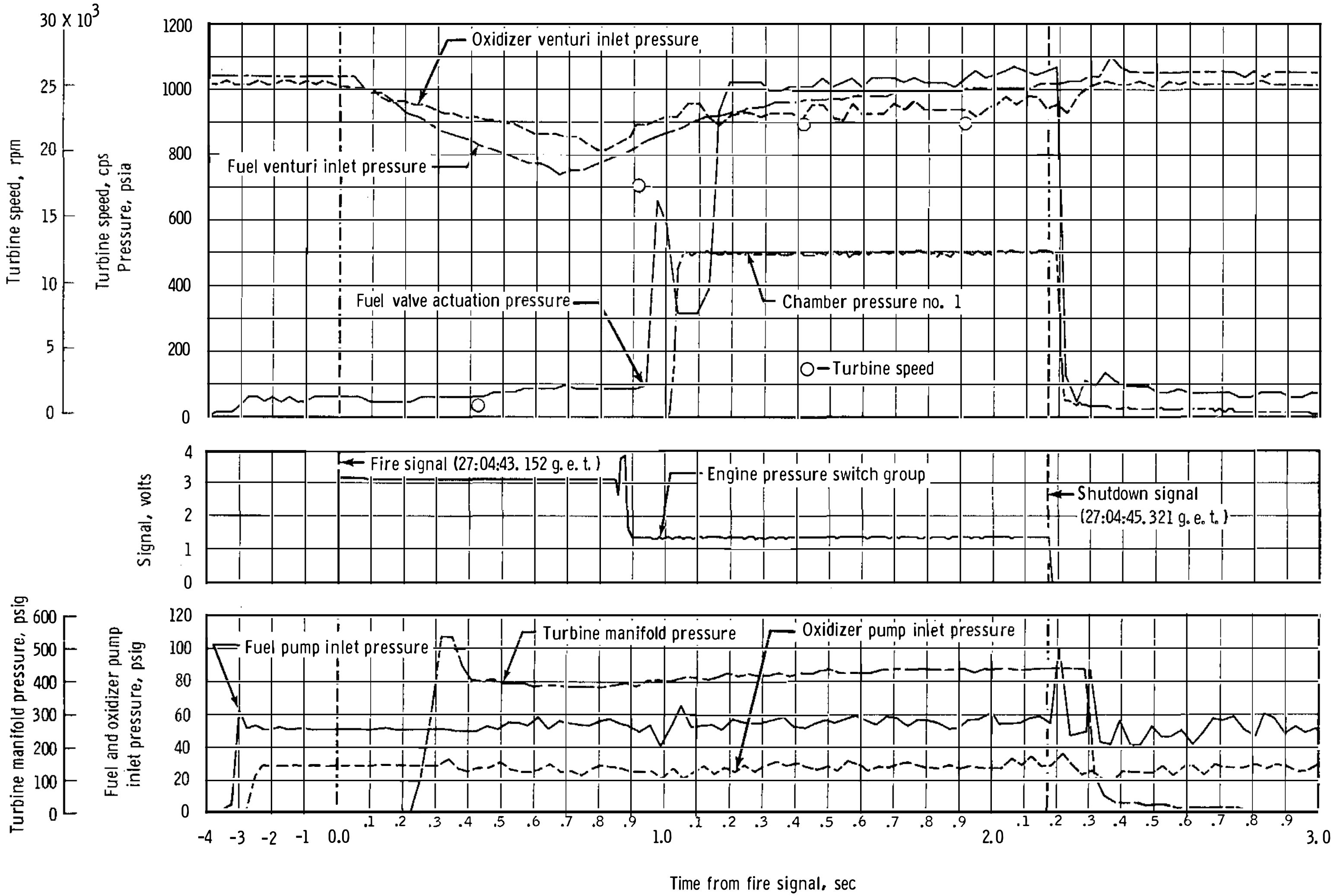

(c) In orbit maneuver no. 2

Figure 5. 4. 2-3. - Continued.

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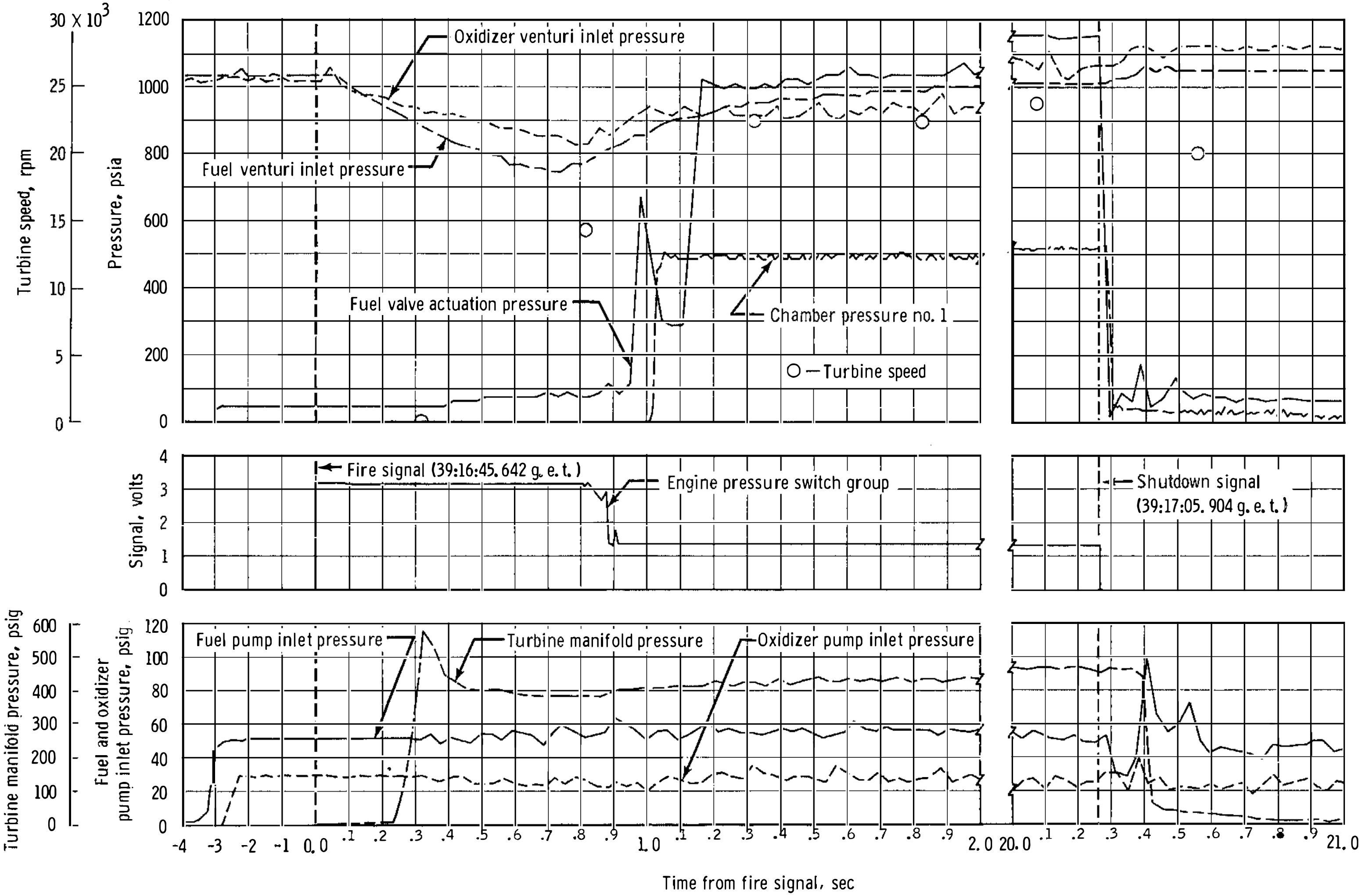

(d) In orbit maneuver no. 3.

Figure 5.4.2-3. - Continued.

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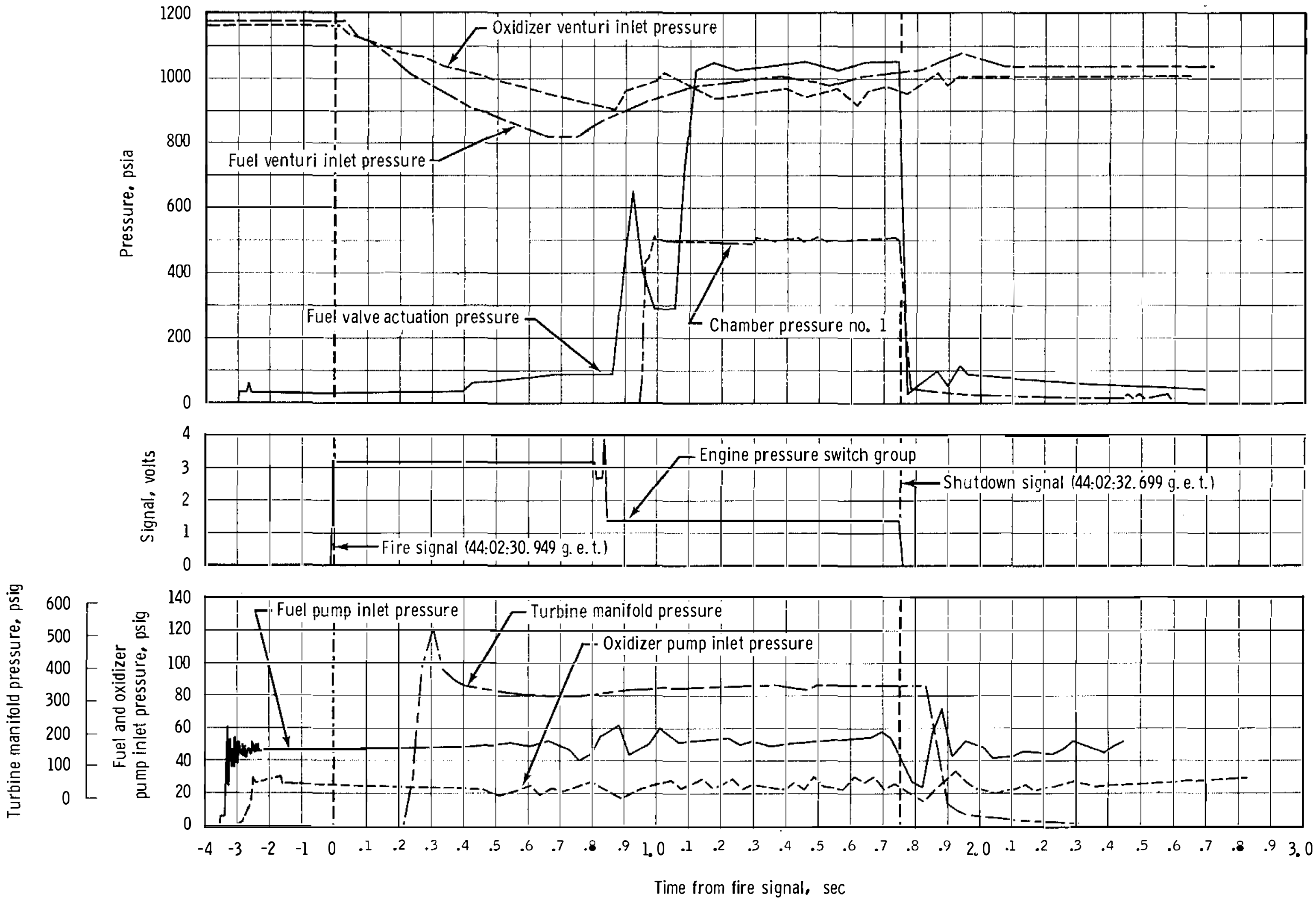

(e) In orbit maneuver no. 4

Figure 5.4.2-3. - Continued.

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(f) In orbit maneuver no. 5

Figure 5.4.2-3. - Continued.

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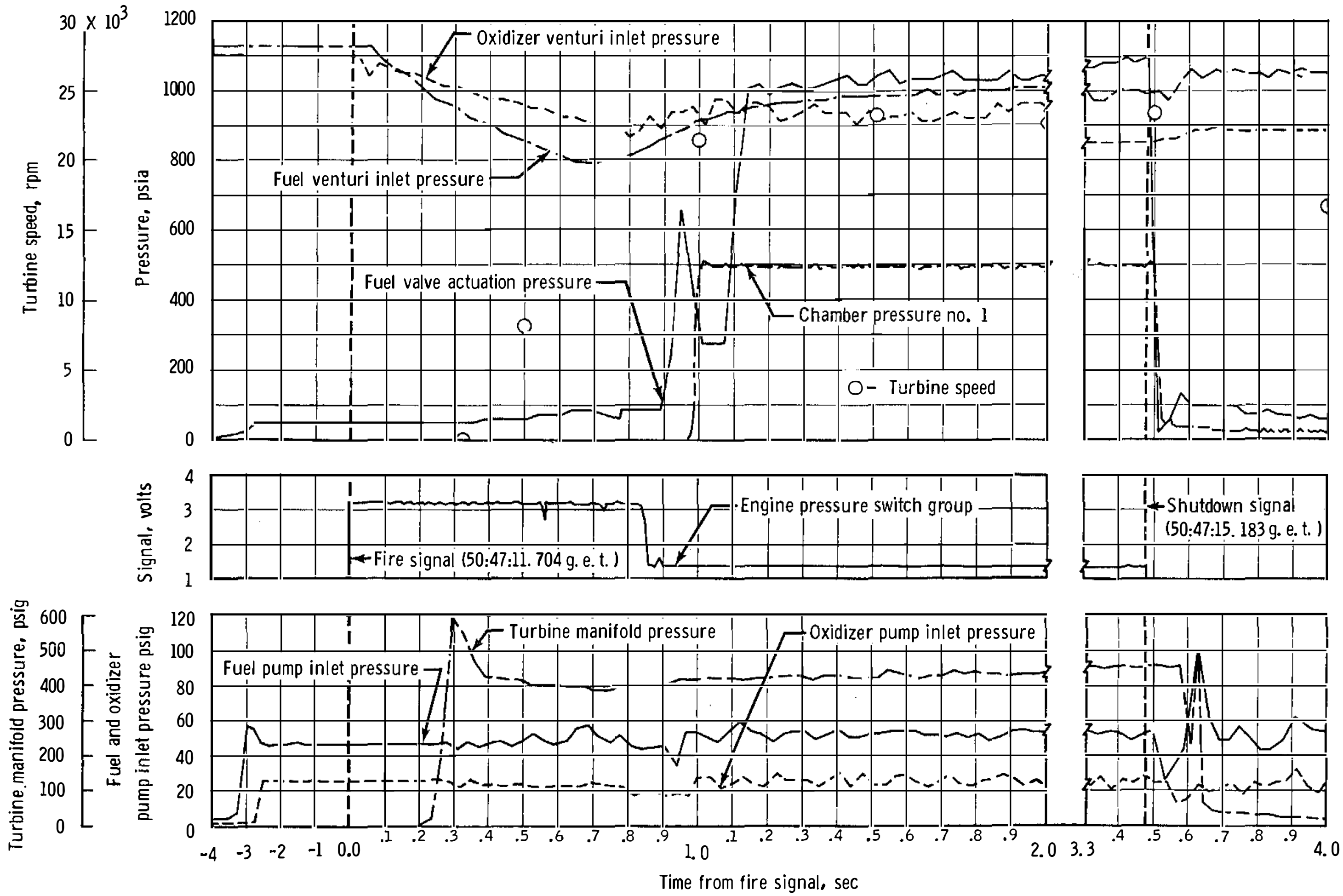

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(g) In orbit maneuver no. 6

Figure 5.4.2-3. - Continued.

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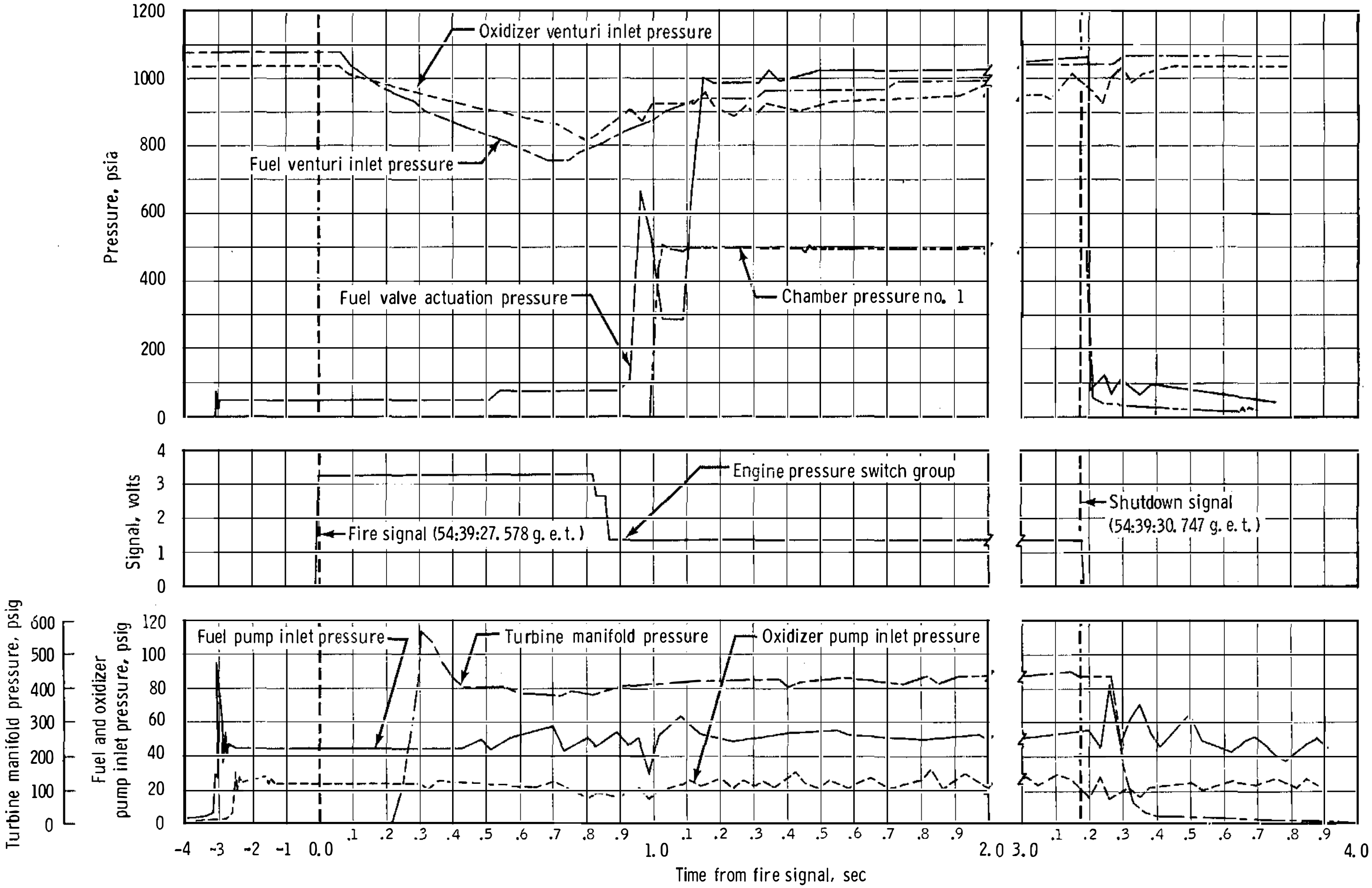

(h) In orbit maneuver no. 7.

Figure 5.4.2-3. - Continued.

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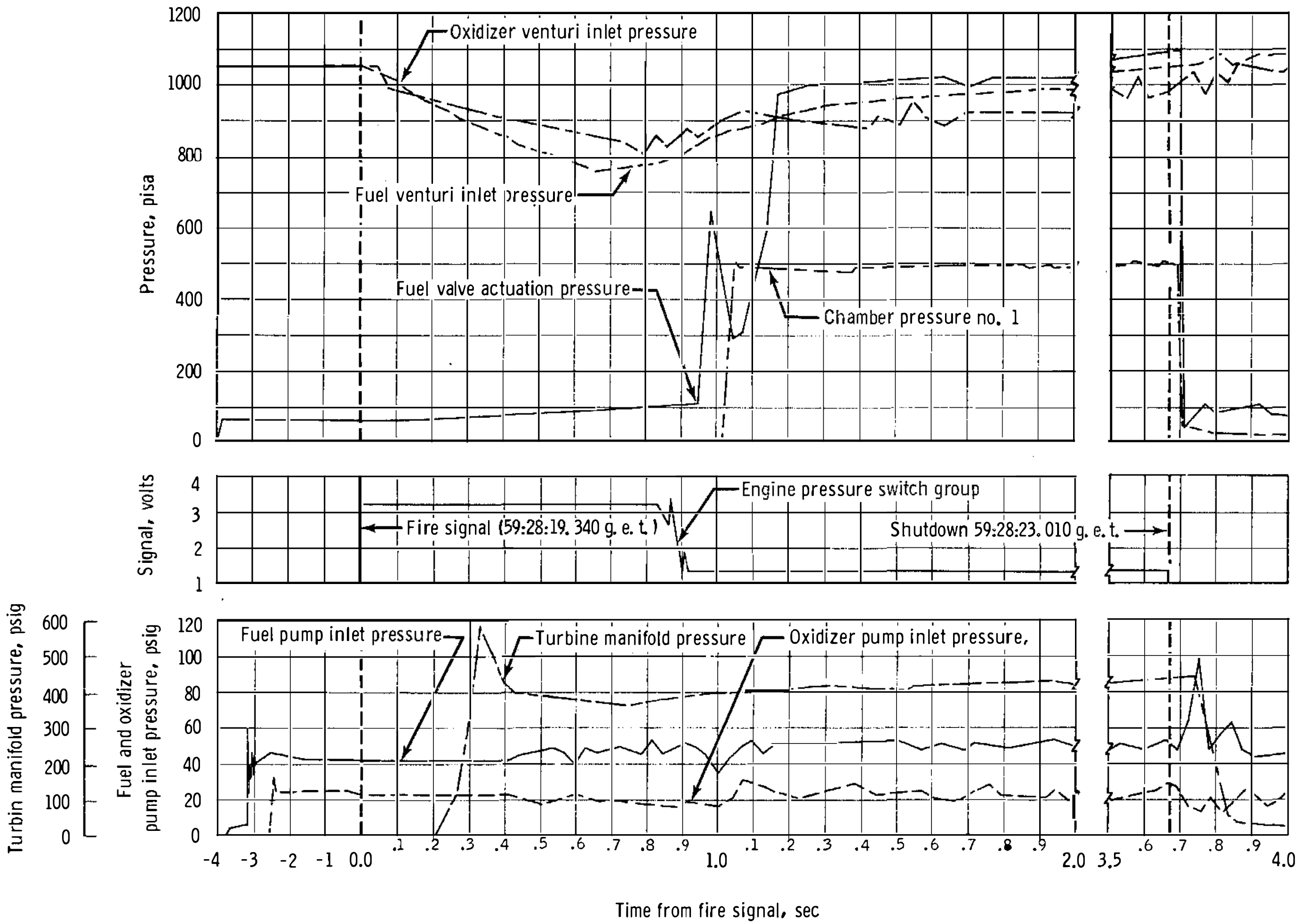

(i) In orbit maneuver no. 8

Figure 5.4.2-3. - Concluded.

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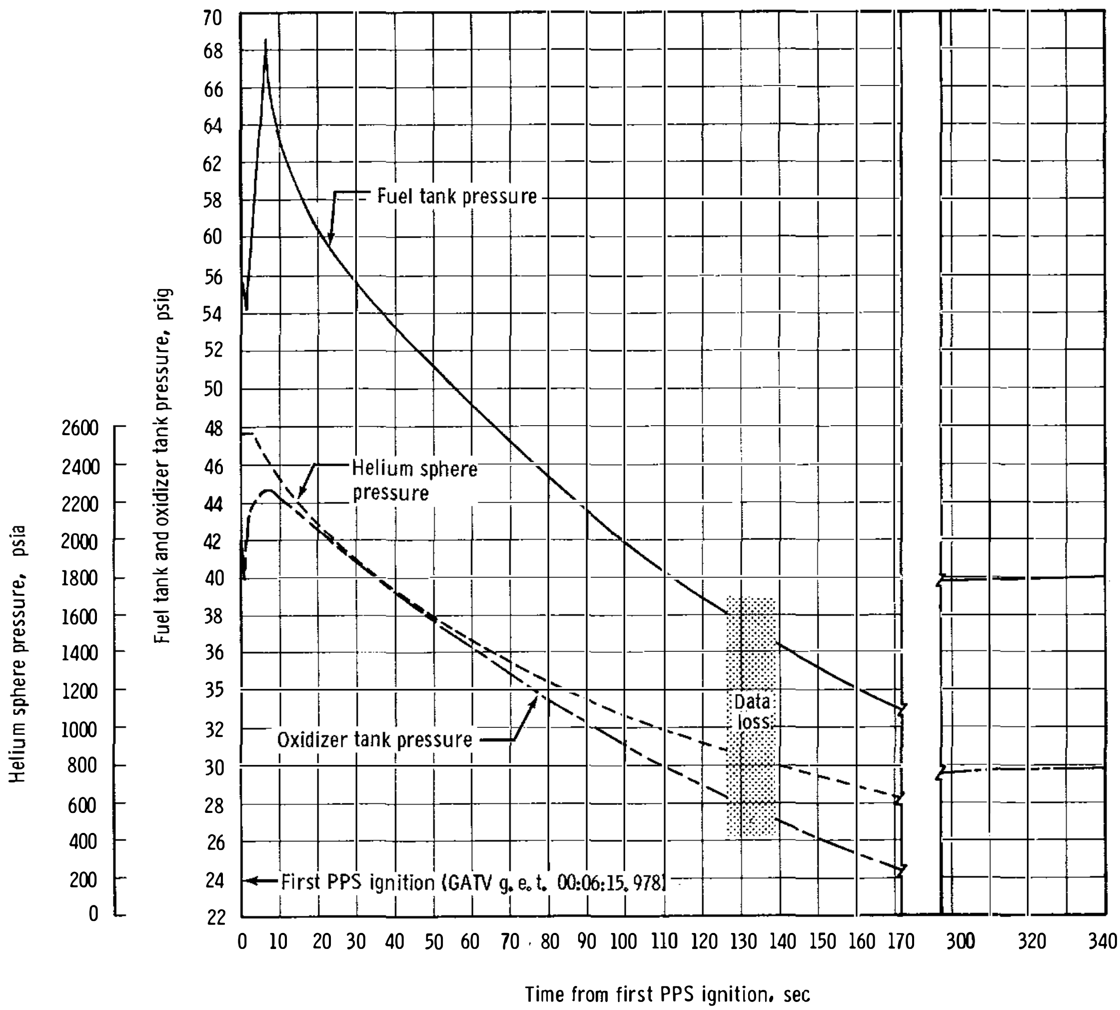

Ascent maneuver

Figure 5.4.2-4. - PPS tank pressure profile.

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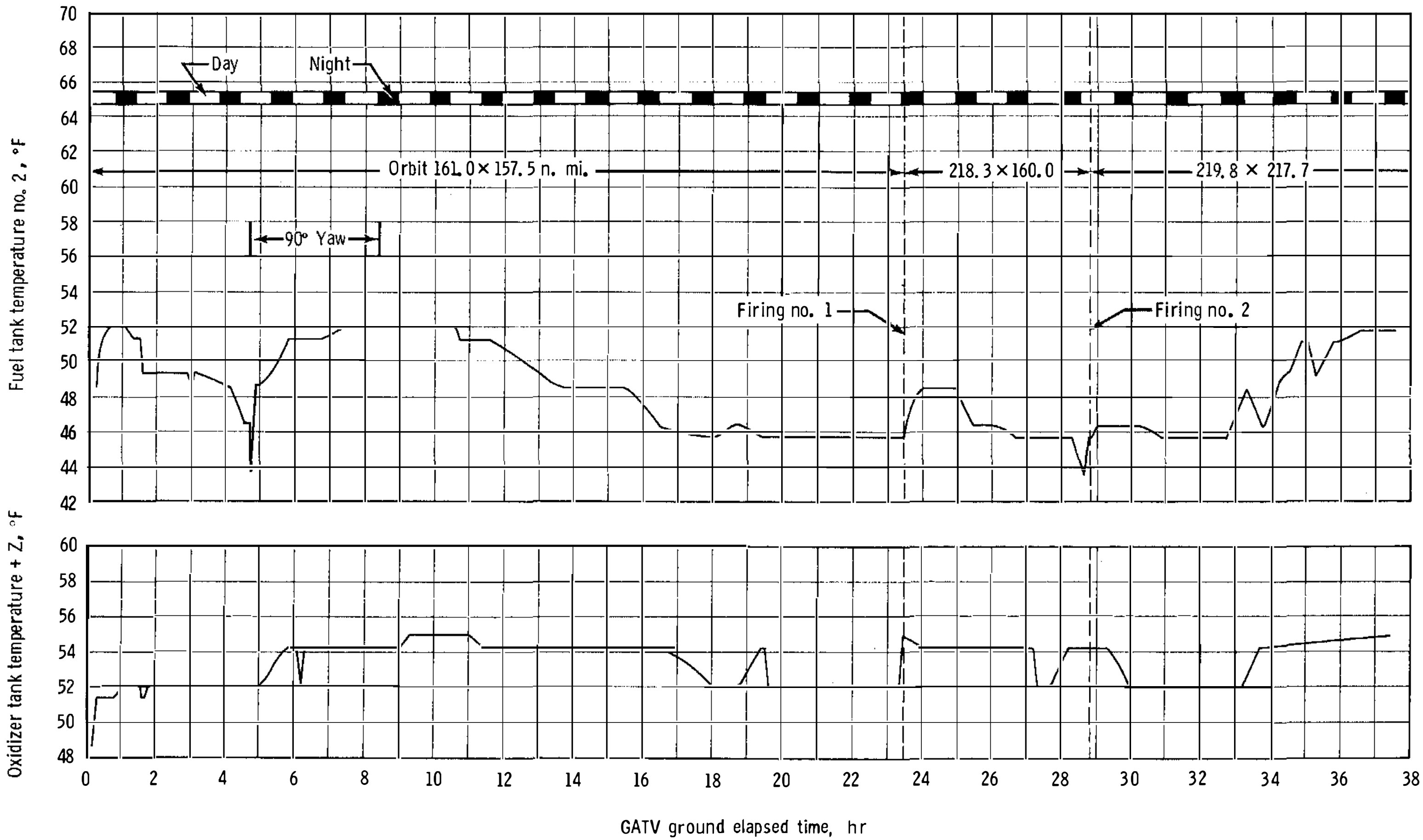

(a) Main tanks, 0 through 38 hours GATV g.e.t.

Figure 5.4.2-5. - PPS thermal history.

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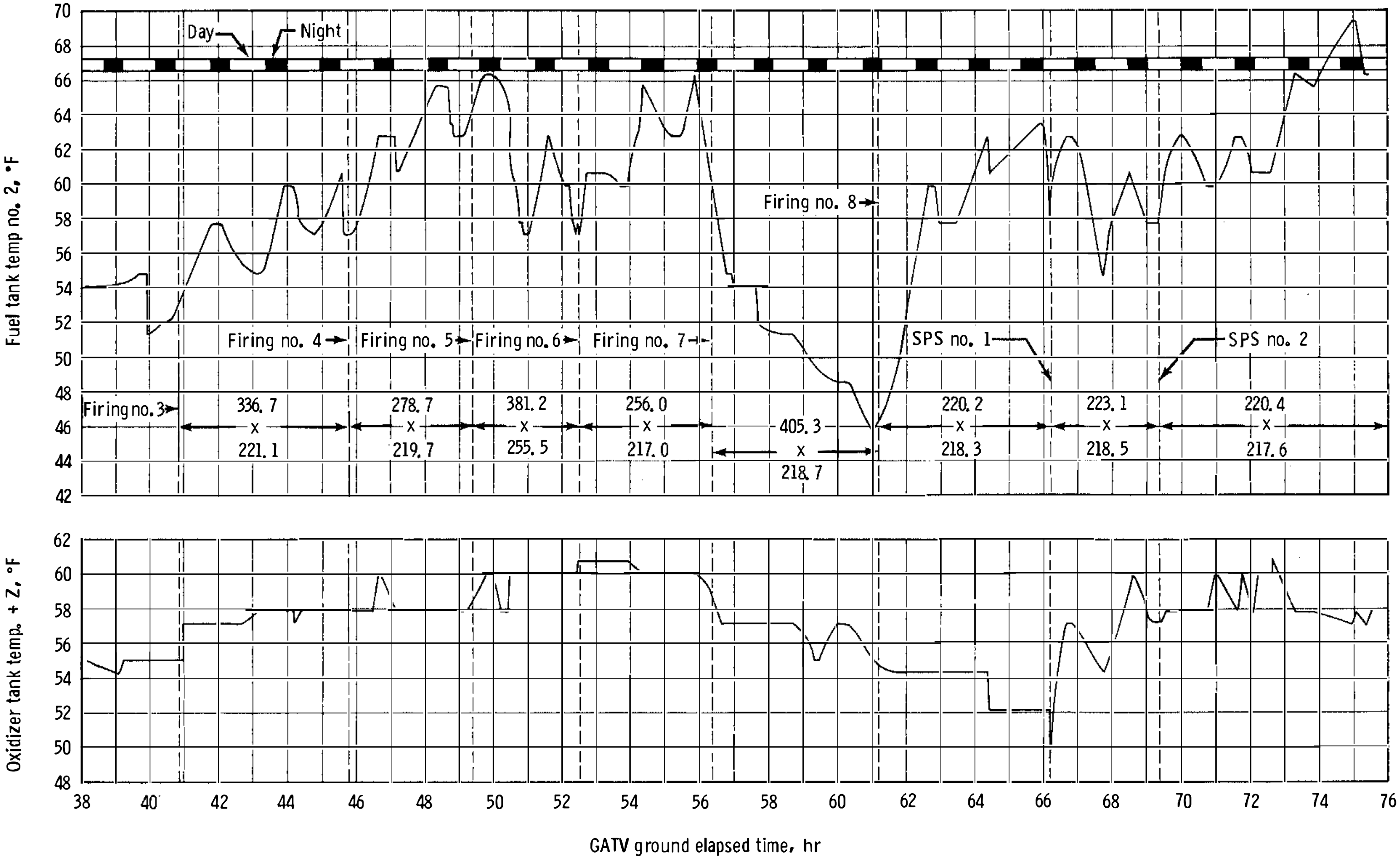

(b) Main tanks, 38 through 76 hours GATV g.e.t.

Figure 5.4.2-5. - Continued.

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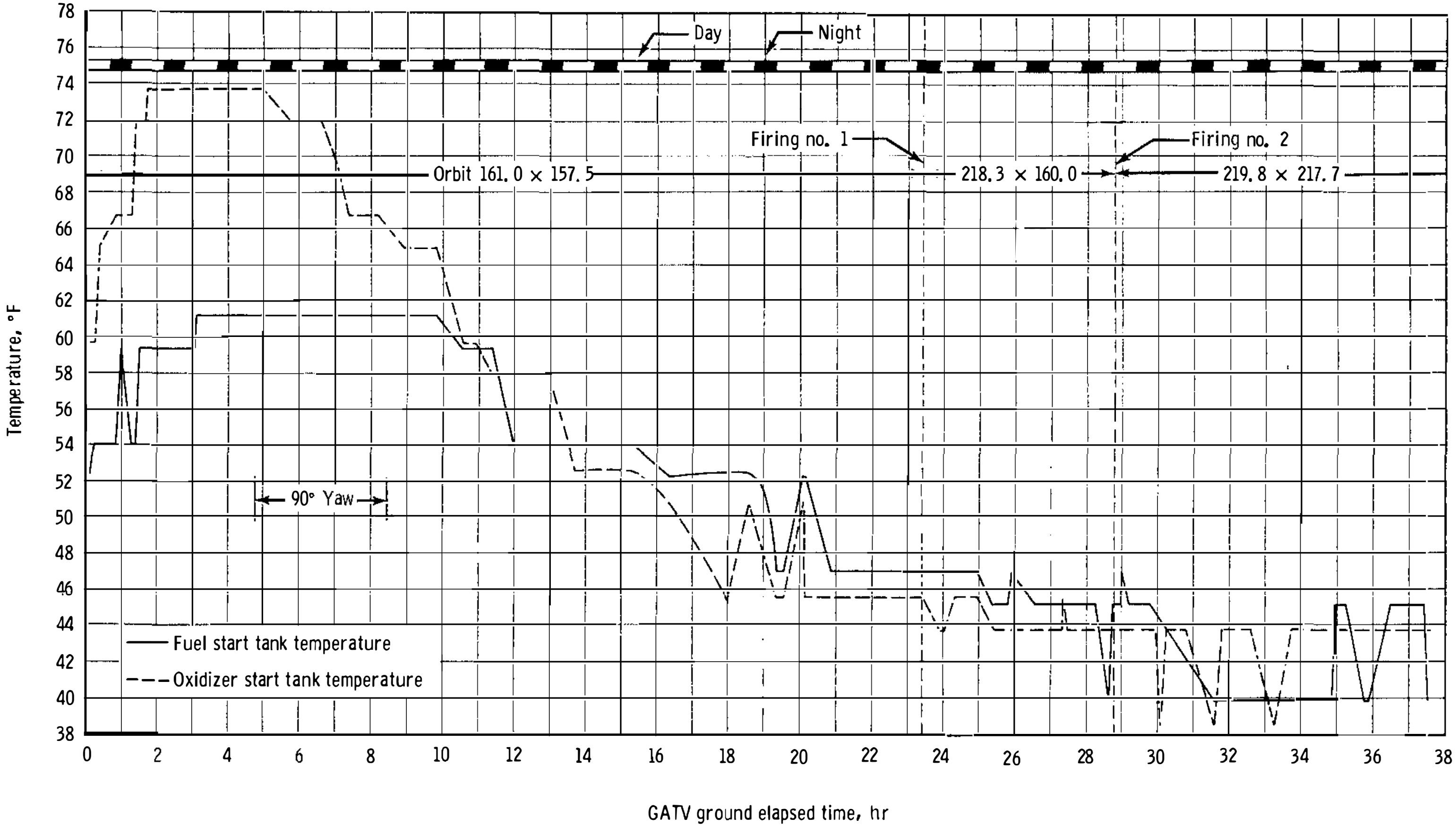

(c) Start tanks 0 to 38 hours

Figure 5.4.2-5. - Continued.

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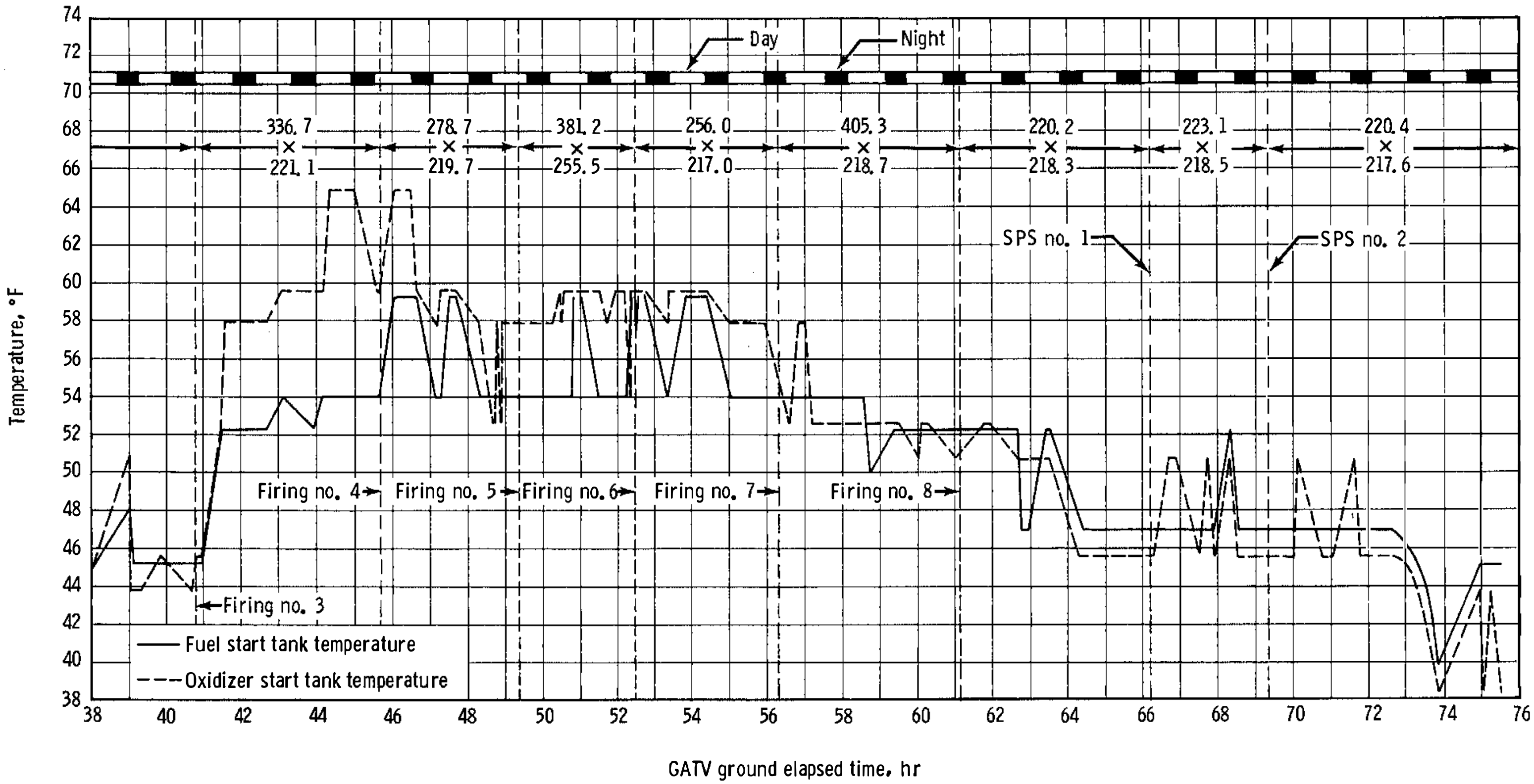

(d) Start tanks 38 to 76 hours

Figure 5.4.2-5. - Concluded.

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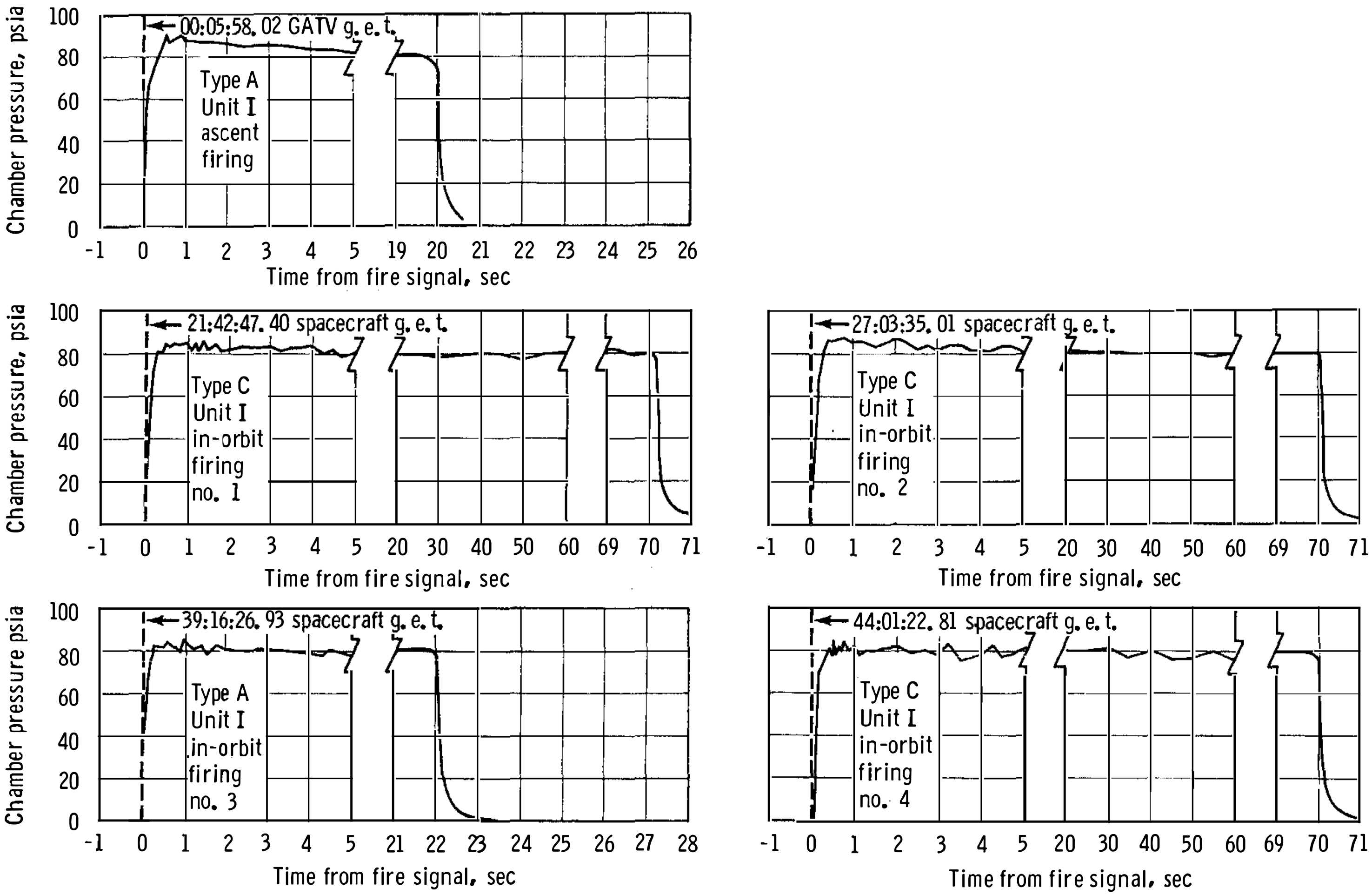

(a) Unit I engine firings, ascent through no. 4

Figure 5.4.2-6. - SPS -Y chamber pressure traces.

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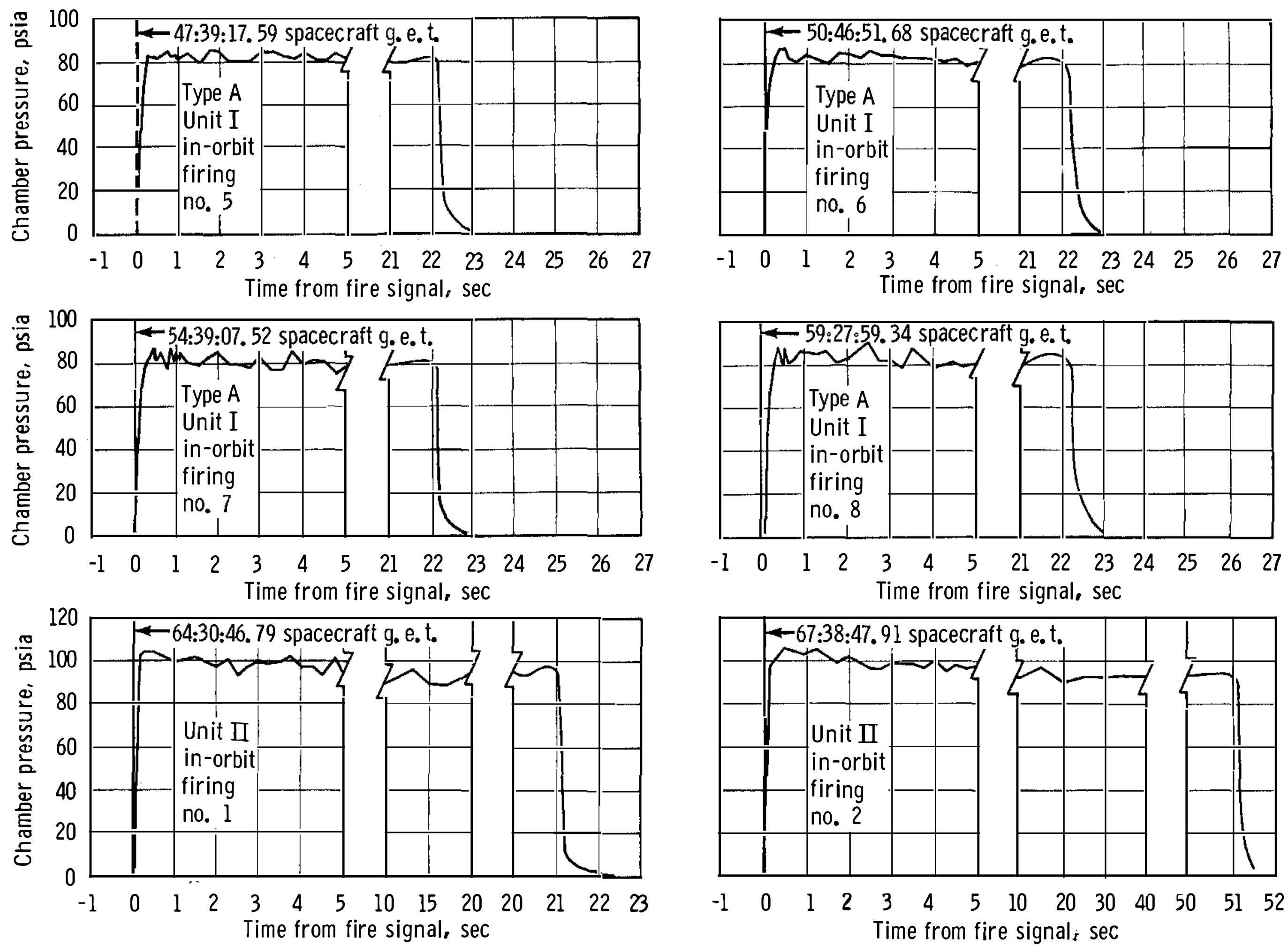

(b) Unit I engine firings no. 5 through no. 8, and unit II engine firings no. 1 and no. 2.

Figure 5.4.2-6. - Concluded.

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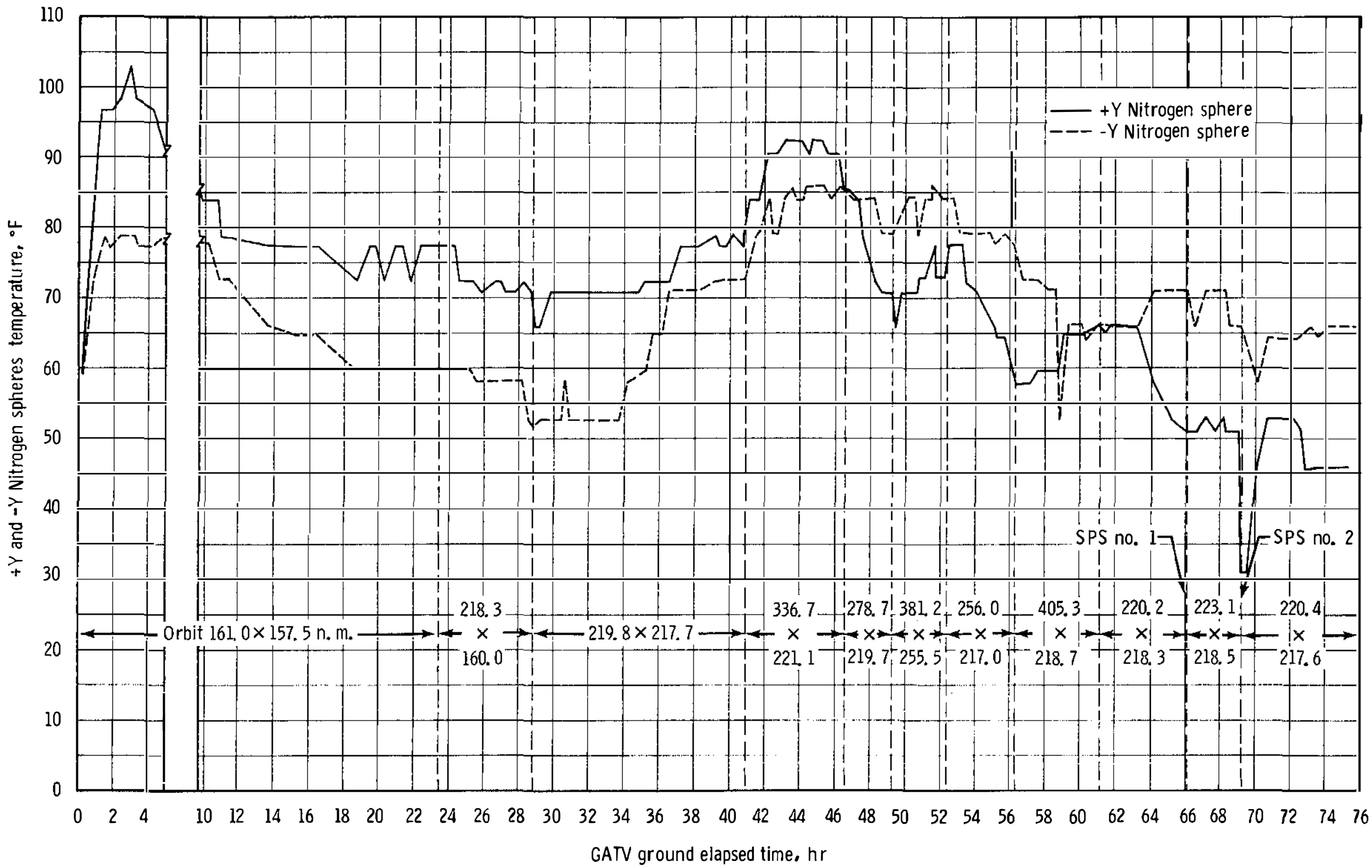

(a) Nitrogen spheres.

Figure 5.4.2-7. - SPS thermal history.

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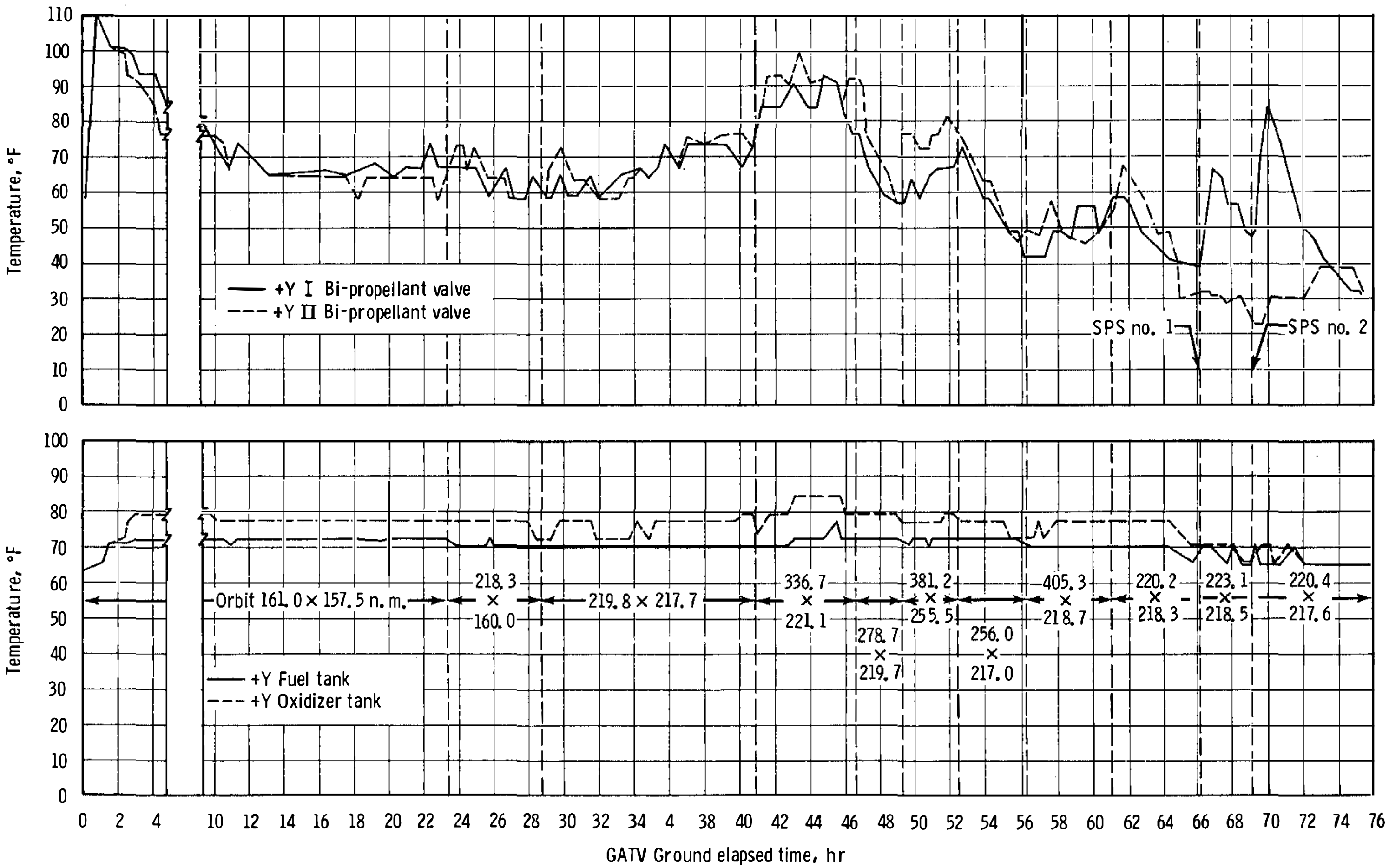

(b) Propellant tanks and valves, +Y module.

Figure 5.4.2-7. - Continued.

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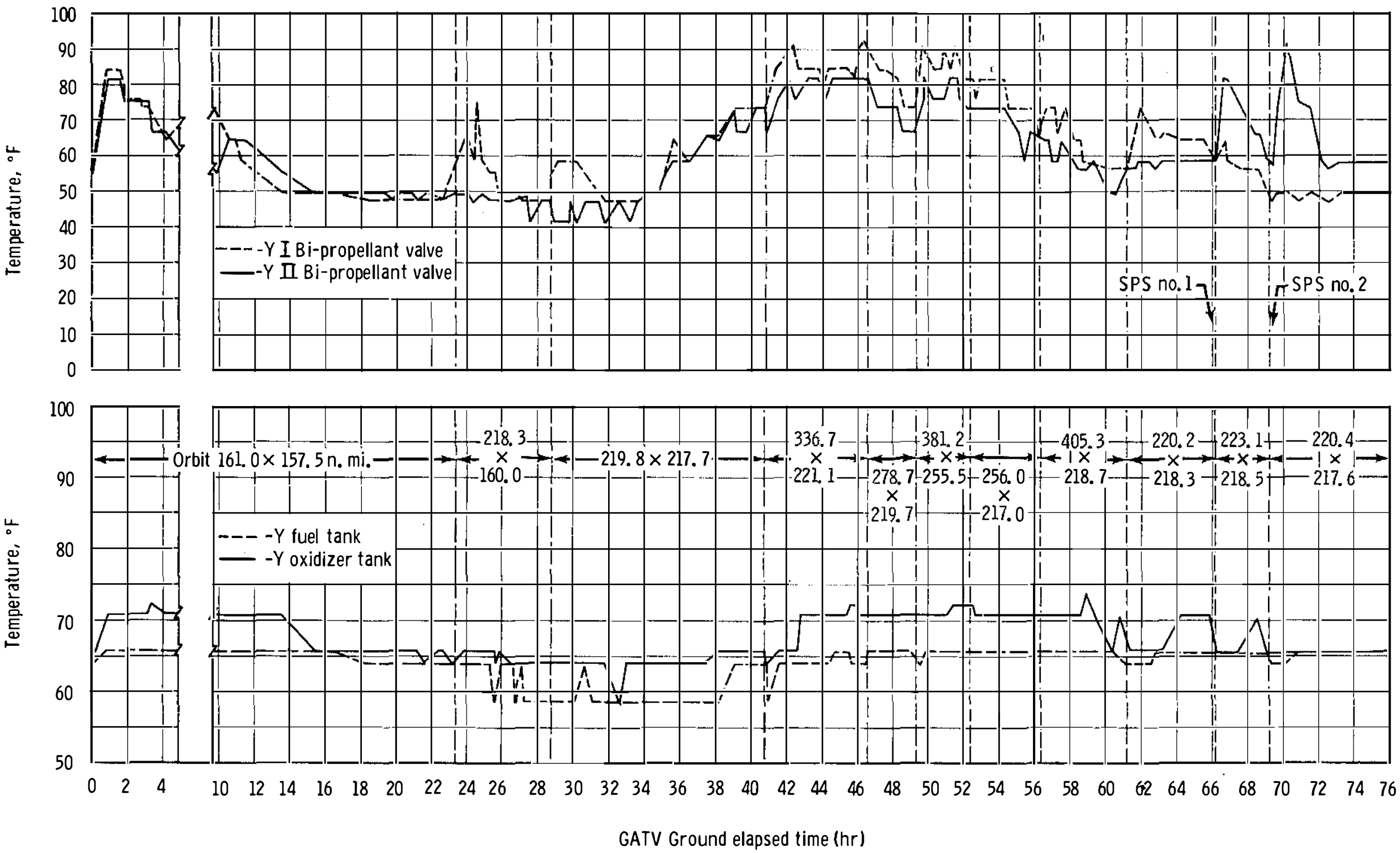

(c) Propellant tanks and valves, -Y module

Figure 5.4.2-7. - Concluded.

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5.4.3 Communications and Command System

The performance of the Communications and Command (C and C) System was excellent throughout the flight. The Command System used UHF, L-band radar, and hardline commands and the performance of each was without flaw. The telemetry and tracking systems functioned very well.

5.4.3.1 Command system.- The command system functioned as expected in using the UHF-RF link to and from the ground stations, the L-band radar RF link to and from the spacecraft, and the hardline link to and from the spacecraft in the docked configuration. The spacecraft real-time commands (RTC's) to the GATV were consistently followed by message acceptance pulses (MAP's), and all transmitted ground commands were also followed by MAP's from the GATV. Retransmission of ground commands was not required at any time.

The velocity-meter counter was loaded and verified from the ground command stations. Early in the flight, a minor problem in loading the velocity meter was found to be caused by incorrect timing for inserting the loads. When the ground station personnel increased the time between messages by a few hundredths of a second, all velocity-meter loads were then received and correctly entered. The GATV received and verified approximately 2400 commands in the 3 1/2 days that the network was operational. In addition, the flight crew sent approximately 45 RTC's the first day. Approximately 5100 commands were transmitted and verified during the total 8-day period that the GATV had electrical power. All commands were received, verified, and executed satisfactorily.

5.4.3.2 Tracking system.- The C-band and S-band transponders operated as expected throughout the flight. The temperature of the C-band transponder stabilized at 135° F and the temperature of the S-band transponder reached a maximum of 157° F and then stabilized at a temperature of 137° F. The upper temperature design limit for both transponders is 165° F.

5.4.3.3 Telemetry system.- The telemetry system operated satisfactorily during the entire flight. All temperatures, voltages, and status bits were within specifications. The tape recorder (which stores data for 20 minutes before being erased) was running during the spacecraft anomaly, but was not turned off until the first 13 minutes of the anomaly data had been erased. The crew should be commended for remembering to turn off the tape recorder during this busy period. During this time period, the events from the ground stations (fig. 5.4.3-1) coincide with the events from the tape-recorder data (fig. 5.4.3-2). During the latter part of the 8-day flight, the tape recorder operated continuously for approximately 36 hours in the record mode except during dump periods. Playback data were good.

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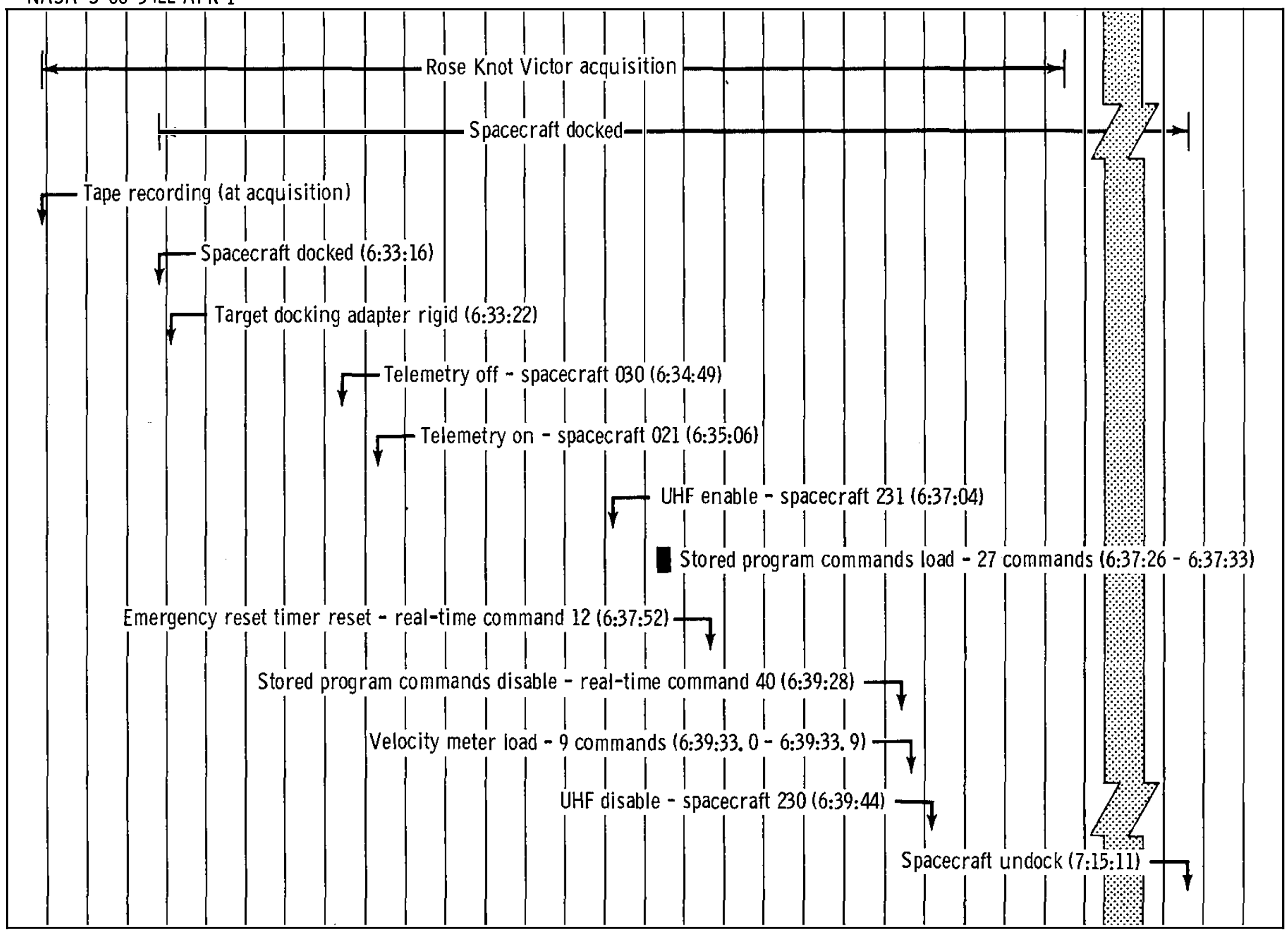

Figure 5.4.3-1. - GATV real-time telemetry data.

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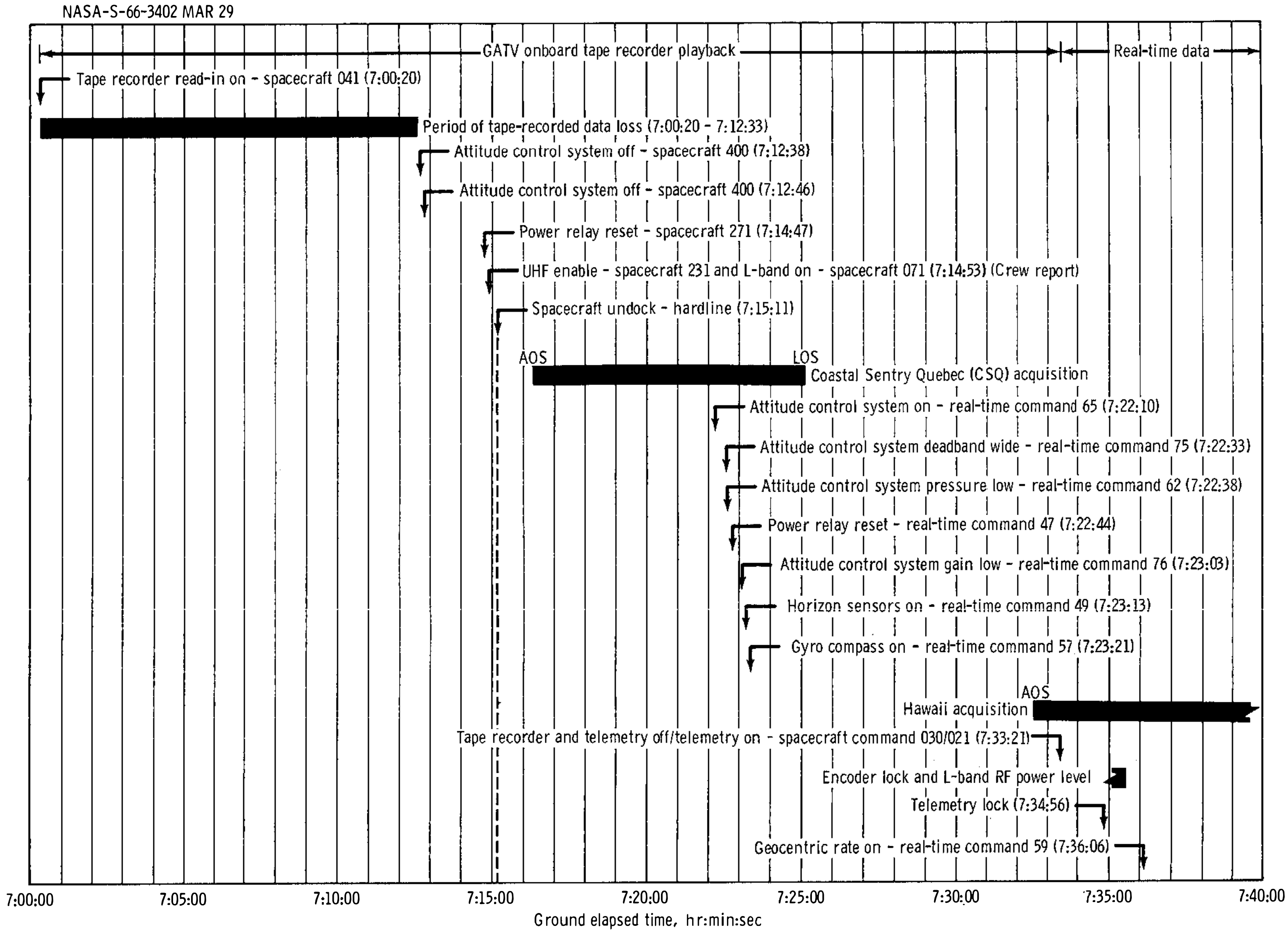

Figure 5.4.3-2. - GATV tape recorder dumped data.

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5.4.4 Hydraulic and Pneumatic Systems

5.4.4.1 Hydraulic System.- The Hydraulic System operated normally throughout each of the nine PPS burns. During Hydraulic System operation, the pump discharge pressure increased normally from zero to about 2800 psig and occasionally reached as high as 3000 psig during a maneuver. After each period of operation, the pump discharge pressure decreased to zero within a 2-second period after engine cutoff. During the flight, hydraulic-reservoir pressure was normal and varied from 50 to 80 psig, increasing to the upper value during system operation.

5.4.4.2 Pneumatics.-

5.4.4.2.1 Propellant tank pressurization system: Prior to lift-off, the propellant tanks were pressurized to 30 and 40 psig for the oxidizer and fuel tanks, respectively. By the time of initiation of the first firing of the PPS, at 377.5 seconds, these pressures had increased to 44.1 and 55.5 psig due to the change in reference pressure from atmosphere to nearly zero at 450K feet altitude. Shortly after the opening of the pyrotechnically operated helium valve to the propellant tanks, the tank pressures started fluctuating slightly, dropping to about 24 and 33 psig for the oxidizer and fuel tanks, respectively, at PPS cutoff. After the ascent firing, the pressures increased to 29.4 psi in the oxidizer tank and 52.3 psi in the fuel tank. The helium supply-tank pressure dropped from about 2560 psia at PPS engine ignition to 625 psia at engine cutoff following the ascent PPS maneuver.

During the PPS maneuver in the eighteenth revolution, the propellant-tank pressures remained practically constant, decreasing from 28.5 to 28.0 psig for the oxidizer tank and from 51.3 to 50.8 psig for the fuel tank. By the end of the sixth PPS maneuver, these pressures had decreased to 24.5 and 46.0 psig, respectively. By the forty-third revolution the pressures were reading 23.6 and 44.5 psig, respectively. Throughout the flight, the propellant tank pressures remained within the expected levels.

5.4.4.2.2 Attitude Control System: The ACS was activated at LO + 310.58 seconds, shortly after separation of the GATV from the TLV. The pressure in the three nitrogen supply tanks remained nearly constant at 3290 psia from lift-off through separation. This pressure was somewhat higher than expected at lift-off because of a temperature rise of the tanks, but did not represent a problem or a hazardous condition. At SPS ignition (LO + 358 sec), the pressure had dropped to about 3190 psia. At PPS ignition (LO + 377.5 sec) a further pressure decrease to 3160 psia had occurred, where it remained through PPS cutoff.

Over Guaymas during revolution 3, the first GATV yaw maneuver was performed to orient the vehicle at 90 degrees with the TDA north for rendezvous. The maneuver was initiated with the ACS in the high-pressure mode with control gas regulator no. 1 pressure indicating 100 psig, and control gas regulator pressure no. 2 indicating 5 psig, which are normal.

After about 5 seconds, the no. 1 regulator pressure dropped to 25 psig for a short time, and then leveled off at 12 psig for the remainder of the maneuver. The pressure change appeared similar to a normal switching to the low-pressure mode; however, no command had been sent to switch to the low-pressure mode and no MAP's were received to indicate a spurious command being accepted. Numerous yaw maneuvers were performed after this and the problem never occurred again. An investigation indicates a possible temporary short, but it is believed that contamination in the regulator most probably caused the anomaly.

During the sixth revolution, due to the spacecraft control anomaly, the pressure decreased to 1080 psia. The gas temperature and pressure remained practically constant from that time through the eighteenth revolution.

During the thirty-first revolution at the start of the fifth maneuver, the pressure had dropped to 210 psia. By the end of revolution 43, as calculated on a mass basis, approximately 6 percent of the attitude control gas remained. The gas temperature of 92° F differed very little from the launch temperature and had negligible effect on the mass calculations.

5.4.5 Guidance and Control System

The Guidance and Control System operated as designed. The system placed the GATV into an acceptable orbit of 161.4 nautical miles by 160.7 nautical miles with an inclination angle of 28.9 degrees. However, during the post-docking out-of-plane firings of the PPS, an unexpected positive in-plane velocity error occurred. This placed the vehicle into a much larger orbit than predicted, although the resulting inclination was within 0.02 degree of that required. The in-plane maneuvers were accomplished correctly and after all of the eight PPS and two SPS in-orbit firings were completed, the vehicle was placed in the planned 220-nautical-mile circular orbit with the ACS maintaining attitude control.

5.4.5.1 Ascent guidance sequence.- The sequence of events for the GATV guidance system during ascent started at LO + 282.08 seconds with the start of the sequence timer. At LO + 303.96 seconds, SLV-3 vernier engine cutoff (VECO) and the uncaging of the GATV gyros were confirmed. Separation was at LO + 308.50 seconds and the GATV ACS was enabled at LO + 310.58 seconds. The -1.5 deg/sec pitch rate was initiated at LO + 342.96 seconds to place the GATV in the proper attitude for the ascent maneuver. The pitch and yaw ACS was disabled at LO + 375.95 seconds in readiness for engine start.

After the Propulsion System start sequence was initiated, the hydraulic pressure buildup was coincident with turbine speed and was nominal (fig. 5.4.5-1). Pitch channel performance was nominal. There was no pitch gas activity after the thrust-initiate signal. The initial actuator setting was at +0.41 degree. The actuator dynamic response was nominal and the thrust vector reached the proper position in 7 seconds, and it maintained this position during the entire maneuver. The pitch gyro showed an error of +0.3 degree before engine thrust but settled out around its null position after 30 seconds of engine firing and remained there until velocity meter cutoff (VMCO).

The yaw channel exhibited a larger-than-normal yaw-gyro position error at engine thrust (fig. 5.4.5-1). The yaw-actuator offset before the firing was +0.79 degree. Coincident with turbine spin-up at LO + 377 seconds, the yaw gyro indicated an acceleration of 2.5 deg/sec and rapidly reached the 5-degree telemetry saturation point. The yaw actuator reached its nominal offset center-of-gravity position of -0.75 degree in 7 seconds. It remained close to this position and reached -1.0 degree at the end of the maneuver. The yaw gyro also reached a nominal position of +0.8 degree during the maneuver but increased to +1.4 degrees at the end of the maneuver due to the center-of-gravity shift as propellants were consumed. The oscillations of the yaw gyro towards the end of the maneuver were an indication of fuel sloshing.

Roll channel performance was nominal (fig. 5.4.5-1). At separation a normal roll rate was noted which was being brought within the roll deadband of ±0.8 degree prior to engine ignition. There was normal gas-valve activity throughout the firing period. At turbine spin-up, the vehicle rolled from -0.5 to +1.8 degrees, as shown by the horizon-sensor trace. This roll torque was damped out entirely with the gas jets by LO + 400 seconds. A roll-right torque was noted at LO + 460 seconds but was damped out by the roll gas jets by LO + 480 seconds. This roll has been noted on other Agena flights and has been attributed to the heating of the turbine exhaust duct, which causes it to move and change the thrust component of the exhaust gases.

The engine was cut off properly by the velocity meter at LO + 560.4 seconds, followed by activation of pitch and yaw pneumatics which started returning the vehicle to zero degrees in yaw.

5.4.5.2 In-orbit attitude maneuvers.- While in orbit, the GATV was maneuvered with the ACS in both pitch and yaw. The pitch maneuvering was done by applying a -3.99 deg/sec geocentric rate continuously to the pitch gyro. This rate was used to keep the yaw axis of the vehicle perpendicular to the local horizontal and functioned satisfactorily every time it was commanded on. Numerous attitude maneuvers were made in yaw. The GATV was first maneuvered from its insertion attitude of 0,0,0 degrees to 0,0,(-90) degrees by ground command. After docking, the vehicle was maneuvered back to 0,0,0 by a crew command. The maneuver was performed satisfactorily using the 1.5-deg/sec yaw rate and required 55 seconds for completion.

The GATV was extremely stable during the docking phase. No perturbations were noted during the initial contact or during the rigidizing sequence.

While docked, the vehicles were stabilized using the GATV ACS in flight control mode 6 (ACS deadband wide, ACS pressure high, ACS gain high/docked, horizon-sensor gains high, and hydraulic gain/docked). A difference between the indicated spacecraft attitude and the commanded GATV attitude was reported by the crew and indicated in the data. This discrepancy is discussed in section 5.1.5. The combined vehicle was very stable until the spacecraft anomaly occurred 27 minutes after docking. During the docked portion of the anomaly, the GATV ACS correctly attempted to null the yaw and roll rates.

After undocking at 7:15:06 g.e.t., the GATV ACS was off and the vehicle was in an unknown attitude and experiencing rates in yaw, pitch, and roll. Over the next command station (Coastal Sentry Quebec), a real-time command was sent which commanded the ACS to ON in flight control mode 1 (deadband wide, ACS pressure and gain low). This is the normal orbital coast mode. Within one revolution, the GATV was completely stabilized within the deadbands at an attitude of 0,0,0 degrees.

The vehicle was maneuvered to -93.8 degrees in yaw for the third orbital maneuver, after which it was turned to -180 degrees (engine forward). Both maneuvers were performed satisfactorily and telemetry confirmed that the vehicle was in the proper orientation. Various other yaw maneuvers from +90-degree to -180-degree headings were also made and operation was normal.

A +90-degree yaw maneuver was also made without using a fixed yaw-rate input. For this maneuver the geocentric rate and the gyrocompassing-loop signals were used to turn the vehicle from +90 to 180 degrees. The vehicle responded perfectly and took about 5 minutes to yaw around and about 7 minutes to stabilize at the new attitude. This method is about five times slower than the method using the fixed yaw rate but a much smaller amount of control gas is required.

A gyro-drift test was made by turning the horizon sensor and geocentric rates off and observing the difference between gyro position and scanner output readings approximately one and one-half hours later. The pitch gyro drifted 1.3 degrees and the roll gyro drifted 0.4 degree in this time. This is well within the specification values of 6 deg/hr and 1 deg/hr, respectively.

5.4.5.3 In-orbit propulsion guidance.- A typical operation of the in-orbit propulsion guidance for in-plane maneuvers is illustrated by the second PPS orbital firing at 27:04:43 g.e.t. This was a circularization maneuver from an elliptical orbit and resulted in an orbit of 220.5 by 219.9 nautical miles after a 2.0-second firing that provided a velocity increment of 106 ft/sec. Prior to the firing, the GATV attitude was 0,0,0 degrees (TDA forward with roll and pitch vehicle axes perpendicular to the local vertical) and the vehicle was in flight control mode 3 (ACS pressure high, ACS deadband narrow, and ACS gain high/undocked). This control mode is standard for all PPS undocked firings. Turbine speed started to increase from zero at 27:04:43.6 g.e.t., and hydraulic pressure rose to the normal value of 2700 psig about 3.5 seconds later. During the period from 27:04:45 to 27:04:53, about 8 seconds, the ±5-degree yaw-gyro telemetry channel was saturated (fig. 5.4.5-2). The yaw actuator exhibited a -2.0-degree initial transient after turbine spin-up and returned to -0.2 degree one-half second later and then moved to -1.3 degrees at the end of the firing. The pitch actuator returned to a position of +0.2 degree during the firing, corresponding to a pitch-gyro position error of +0.3 degree. During the firing, the pitch gyro error increased to 2.0 degrees. The roll-axis turbine spin-up torque was normal and was not damped out until after the end of the maneuver (fig. 5.4.5-2). The firing was terminated normally by the velocity-meter cutoff signal at 27:04:45.3 g.e.t. Deadband operation was verified as 0.3 degree in roll, 0.2 degree in pitch, and 0.2 degree in yaw. Both static and dynamic control gains looked normal during the firing, and horizon-sensor operation was proper.

The first plane-change maneuver (PPS orbital firing no. 3) called for the resultant orbital parameters to remain at 220.5 by 219.9 and for the inclination angle of the orbit to change from 28.89 to 30.60 degrees. Vehicle attitude was 0,0,(-90) degrees (TDA north) and the vehicle was in flight control mode 3. At 39:16:07.8 g.e.t., the geocentric rate of +3.99 deg/min was removed from the roll gyro. At 39:16:08.8, the vehicle was yawed to -93.8 degrees. PPS start sequence A was completed at 39:16:46.69 g.e.t. (75-percent PPS P_c). Thrust time was 19.236 seconds from the time the PPS engine reached 75-percent P_c to engine cutoff. The velocity meter shutdown the engine as expected.

The desired velocity-to-be-gained was 1600 ft/sec. The achieved velocity gained was 1601.2 ft/sec due to additional tailoff impulse. The resulting inclination angle was 30.62 degrees, which was within 0.02 degree of that desired. However, the resulting orbital parameters were 338.4 by 221.1 nautical miles, which indicated an in-plane velocity error of 188 ft/sec.

Analysis of the gyro and hydraulic actuator data showed that the yaw-gyro output was beyond the 5-degree telemetry limit for 10.3 seconds (fig. 5.4.5-3). In addition, because of the stops built into the gyro at ±10 degrees it is very probable that the gyro also saturated during this period. At the same time, the yaw-actuator position varied from -0.35 degree to -1.59 degrees and then to -1.10 degrees within 4 seconds of initiating the maneuver. At the end of the maneuver, the actuator position was indicated to be -1.08 degrees. Hydraulic-pressure buildup during this period appears to have been correct and the initial negative actuator spike showed normal response to pressure buildup and turbine spin-up. Finally, after 75-percent thrust buildup, the yaw-gyro output was increasing at a rate of approximately 8.5 deg/sec. Thus a large vehicle attitude dispersion in right yaw occurred during the maneuver which was not corrected by the control system.

Operation of the pitch and roll channels was nominal (fig. 5.4.5-3). The roll gyro indicated that turbine-speed buildup in a clockwise direction was as expected. Control-gas jet operation also was verified as proper. In addition, horizon-sensor gains were verified as operating within limits.

VMCO of the PPS was normal and the ACS immediately started correcting yaw position error as the vehicle was returned to the 0,0, (-90)-degree attitude. Gyrocompassing was turned on and the vehicle stabilized to narrow deadband limits of ±0.8 degree in roll and ±0.25 degree in pitch and yaw.

PPS orbital maneuver no. 5 was the second plane-change maneuver and was required to shift the GATV back to an inclination of 28.87 degrees. Because the velocity angular error from the first plane-change maneuver was about -8.0 degrees, this plane-change maneuver was biased by this much. That is, the vehicle was placed at -101.8 degrees for the maneuver (the heading of the first maneuver -8.0 degrees or -101.8 degrees). A 789 ft/sec maneuver was initiated by a stored-program command at 47:39:37. The vehicle was in flight control mode 3. Operation in the pitch and roll axes was nominal (fig. 5.4.5-4). The yaw-axis gyro again reached telemetry saturation soon after 75-percent thrust was reached and the yaw actuator quickly went to -1.8 degrees and had recovered only to -1.3 degrees by the end of the maneuver (fig. 5.4.5-4). This was the same type of dispersion as noted on the previous plane-change maneuver except that the amount of yaw attitude error had increased. This was confirmed by the resulting orbit of 383.9 by 257.8 nautical miles which indicated a yaw in-plane velocity error of 239 ft/sec. The thrust was terminated by the velocity meter at 47:39:46.7 g.e.t. and the ACS immediately started to return the vehicle yaw axis back to the correct narrow deadband limit of ±0.25 degree.

Subsequent maneuvers with the PPS and SPS were used to correct the orbit altitude and inclination to the final and correct circular orbit of 220 nautical miles and inclination angle of 28.87 degrees.

A postflight analog-computer simulation was conducted by the GATV contractor to examine the vehicle characteristics during a PPS maneuver in order to investigate the yaw error and determine the cause. The analog simulation was developed in a manner to correspond to the flight conditions that existed during the out-of-plane PPS maneuvers. The results revealed that the system had operated as designed; however, the large c.g. offset, in conjunction with the low dynamic gains and the long time constant of the lead-lag circuitry, was responsible for the large yaw transient. The modified lead-lag circuitry was incorporated in the GATV Flight Control System to stabilize the vehicle at the low frequency of the first bending mode of the GATV—spacecraft combination when firing the PPS in the docked configuration. Additional simulations were made with the pitch and yaw c.g. offsets reduced to near zero and these showed a maximum gyro excursion of less than 3 degrees. This would keep the yaw velocity errors well within the desired limits and hold any change in apogee to less than 26 nautical miles for a plane-change maneuver as large as 3.6 degrees. Thus it appears that the large initial deviation in yaw can be reduced to acceptable values by elimination of the large vehicle c.g. offset.

The dynamic response of the hydraulic system relative to changes in c.g. is dependent upon the parameters outlined in the control-system block diagram shown in figure 5.4.5-5. The input to the yaw control loop creates a signal to drive the actuator so that no error exists at the summing junction. Due to the low dynamic gains and the slow response of the lead-lag transfer function, large vehicle errors are created before the actuator aligns the engine through the center-of-gravity.

Because the analog simulation showed a high sensitivity to c.g. offset errors in the GATV Flight Control System, the c.g. shift associated with the difference in vehicle weight of 1800 pounds for the two out-of-plane maneuvers, in combination with output limiting of the yaw gyro, probably accounts for the 27-degree yaw-velocity-vector error made by the second plane-change maneuver.

5.4.5.4 Miscellaneous comments.- The attitude-gas usage during the entire mission is shown in figure 5.4.5-6. Approximately 60 pounds of attitude gas were expended during the docked anomaly period. Between 4 and 7 pounds of attitude gas were expended for each PPS or SPS maneuver. After all ten in-orbit maneuvers were completed, approximately 8 pounds of gas remained. This remaining gas continued to stabilize the vehicle for 135 hours until loss of electrical power. The gyro-speed monitor indicated nominal operation of the gyros throughout the mission. The velocity meter operated properly and was used to terminate all PPS firings. The two SPS firings were not cut off by the velocity meter, but by the backup stored-program command. This was probably due to miscalculation of vehicle weight. A much longer SPS firing would have been required to obtain the desired velocity. The velocity-meter electronics-oven temperatures stayed within acceptable limits of 168° to 172° F.

The horizon-sensor head temperatures varied from 62° to 85° F, which was well within their operating range. The internal temperature of the inertial reference package remained within a nominal range of 144° to 147° F during the mission.

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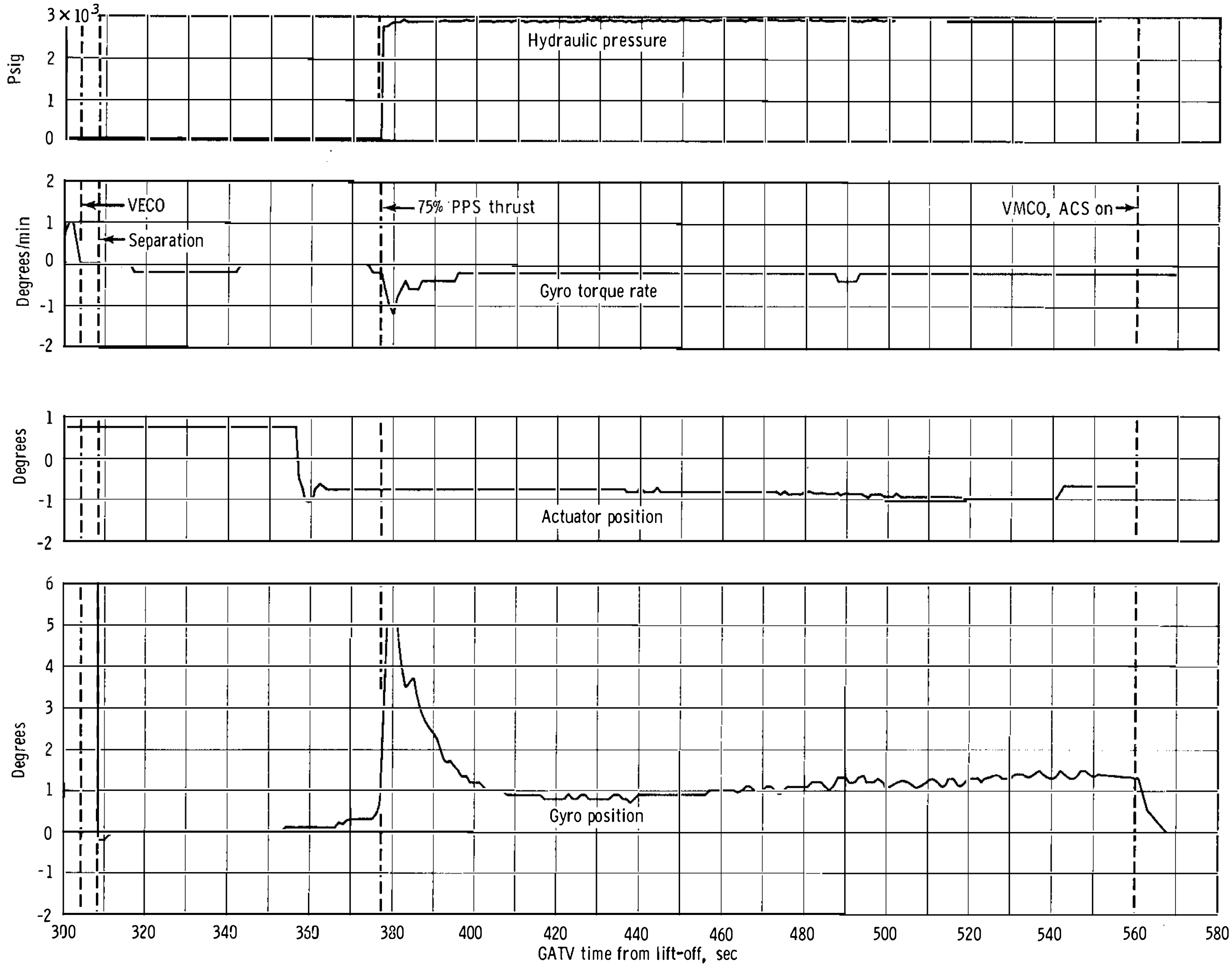
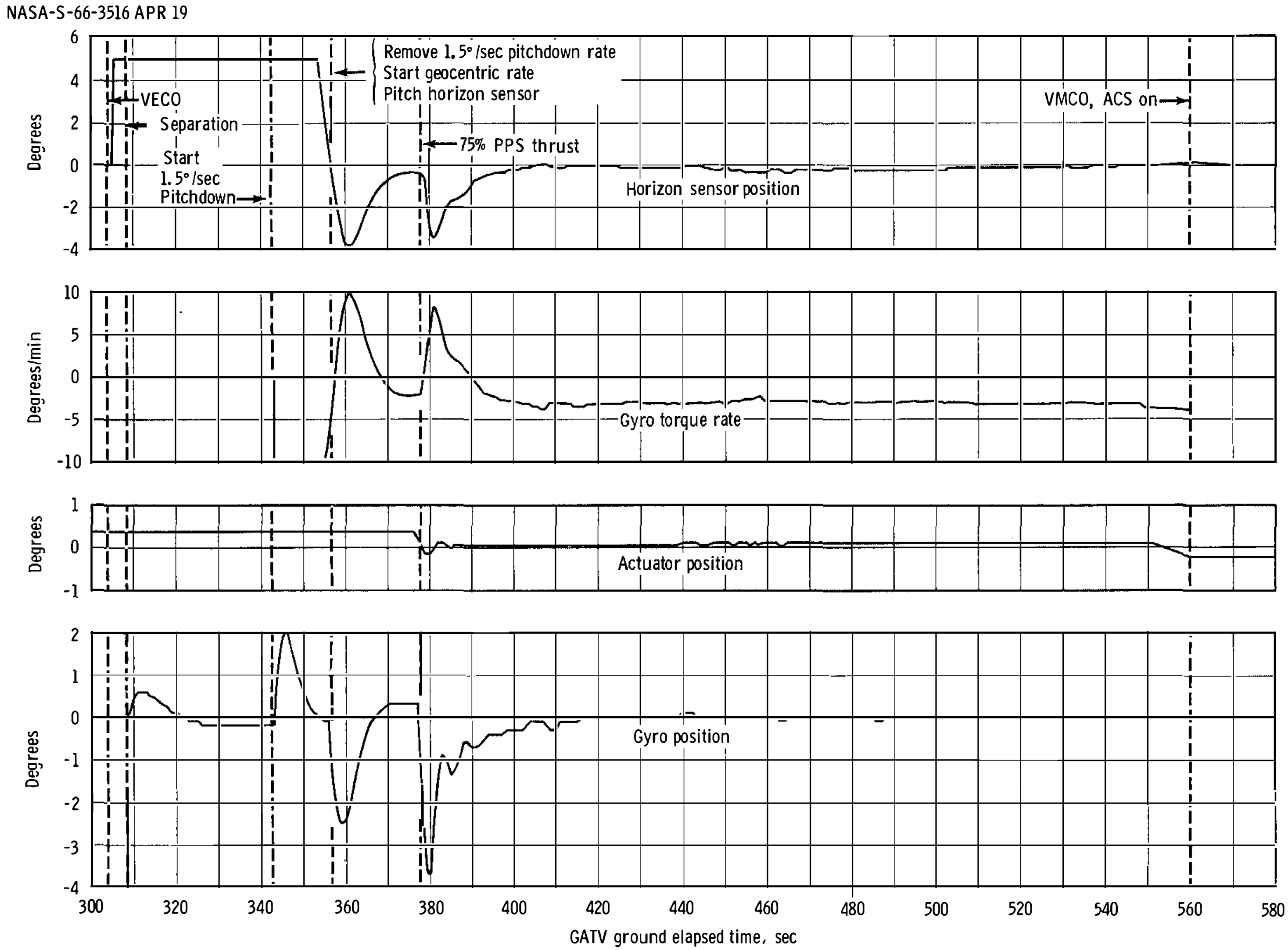


(a) Yaw axis

Figure 5.4.5-1. - Ascent guidance performance.



(b) Pitch axis.

Figure 5.4.5-1. - Continued.

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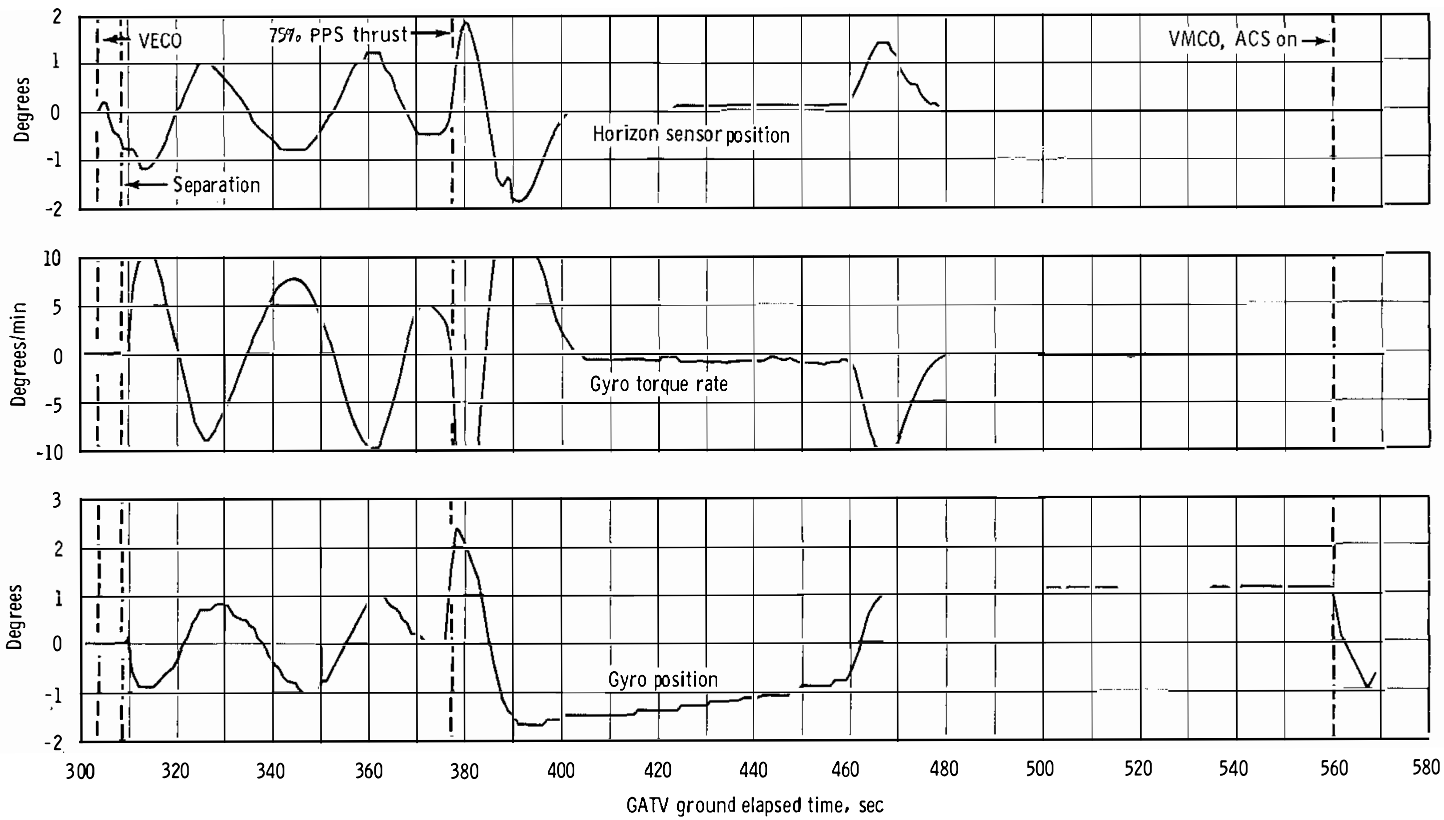
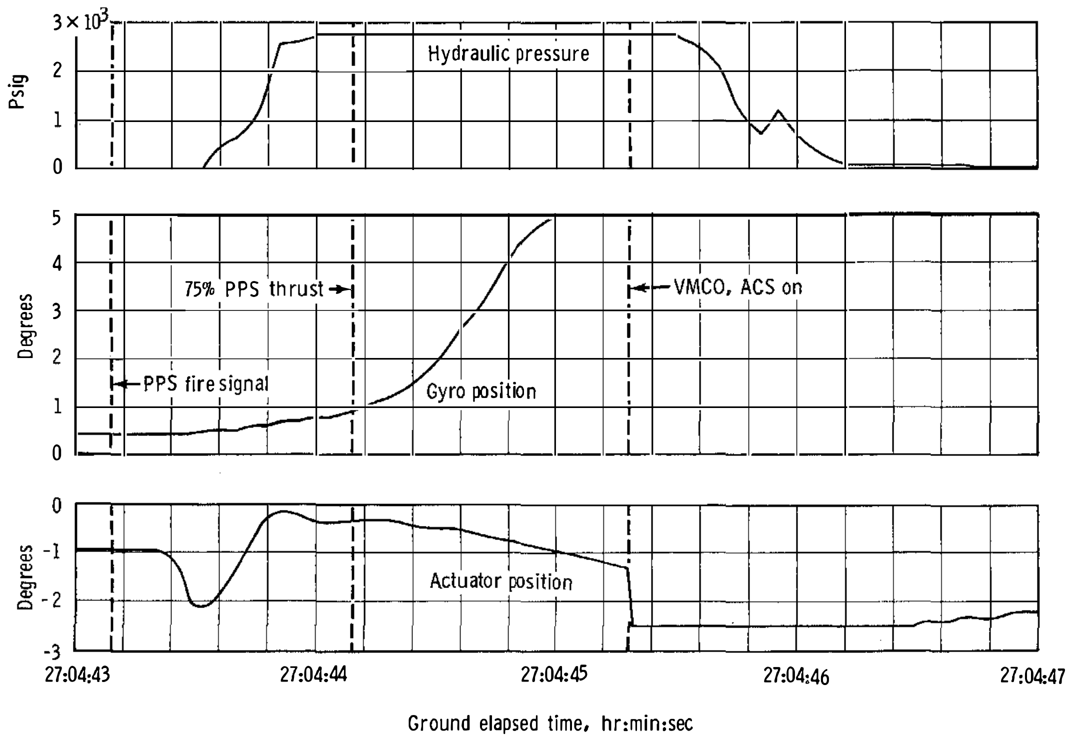


(c) Roll axis

Figure 5.4.5-1. - Concluded.



(a) Yaw axis

Figure 5.4.5-2. - Guidance performance (revolution 18 maneuver).

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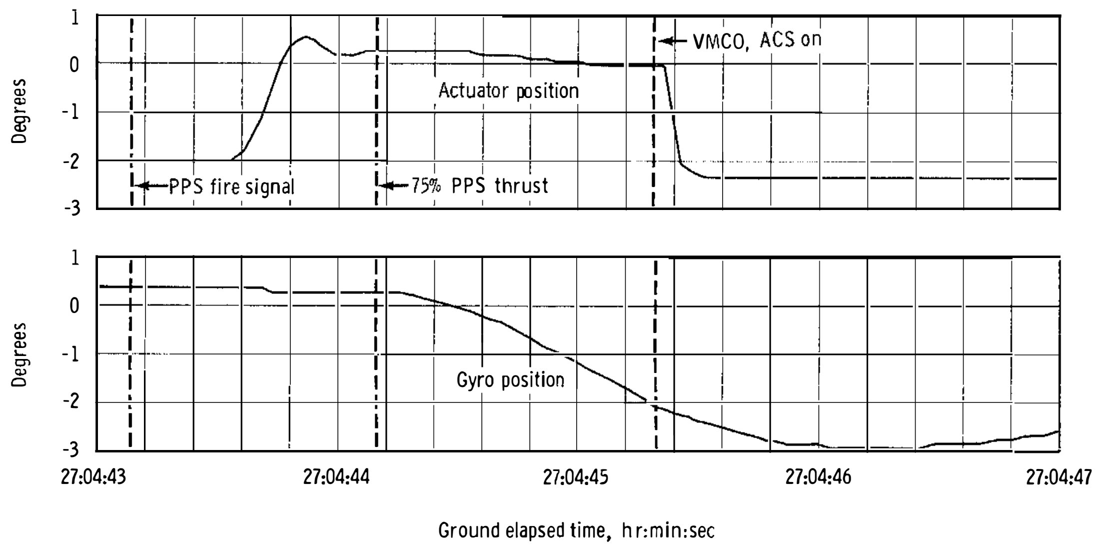

(b) Pitch axis

Figure 5.4.5-2. - Continued.

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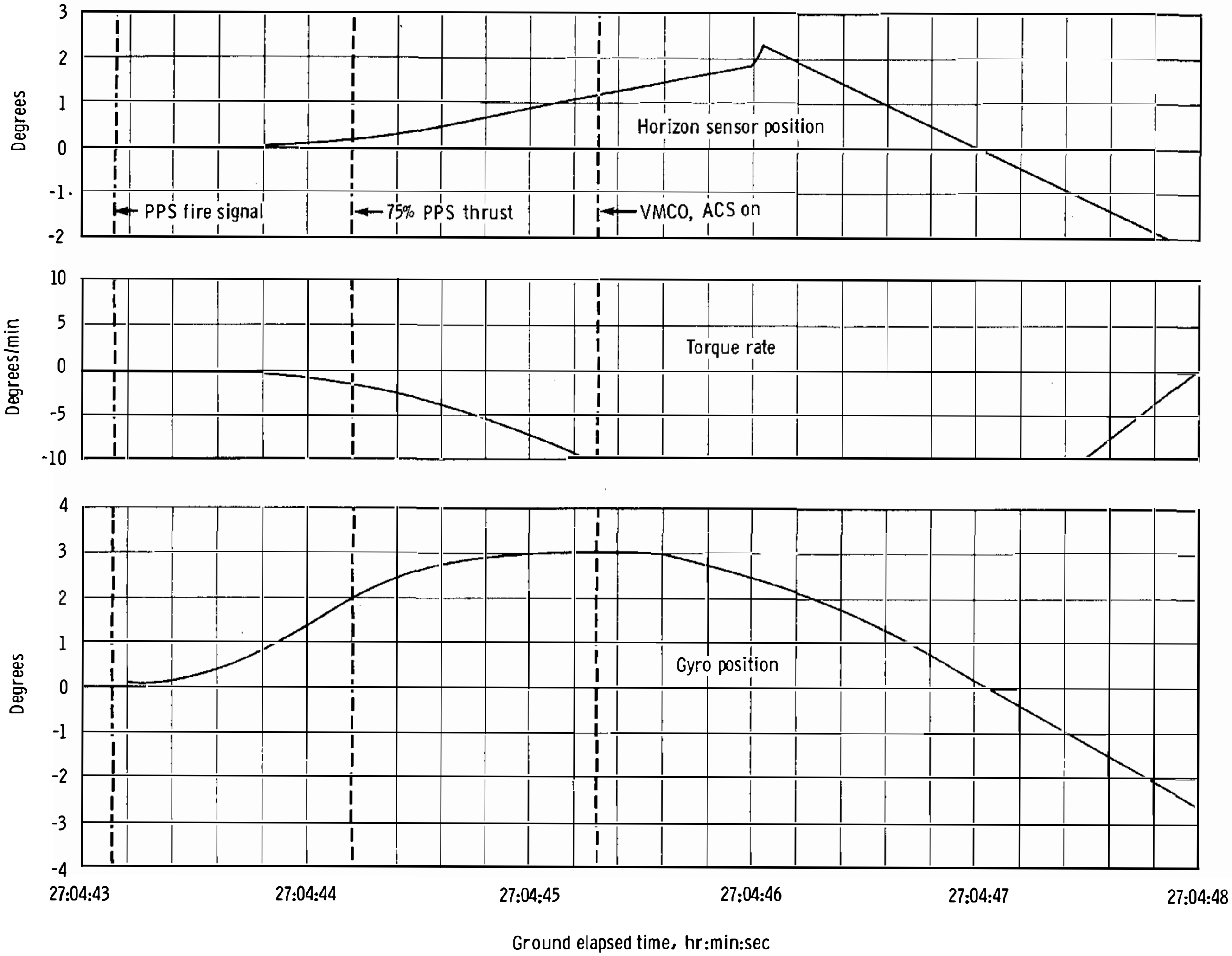

(c) Roll axis

Figure 5.4.5-2. - Concluded.

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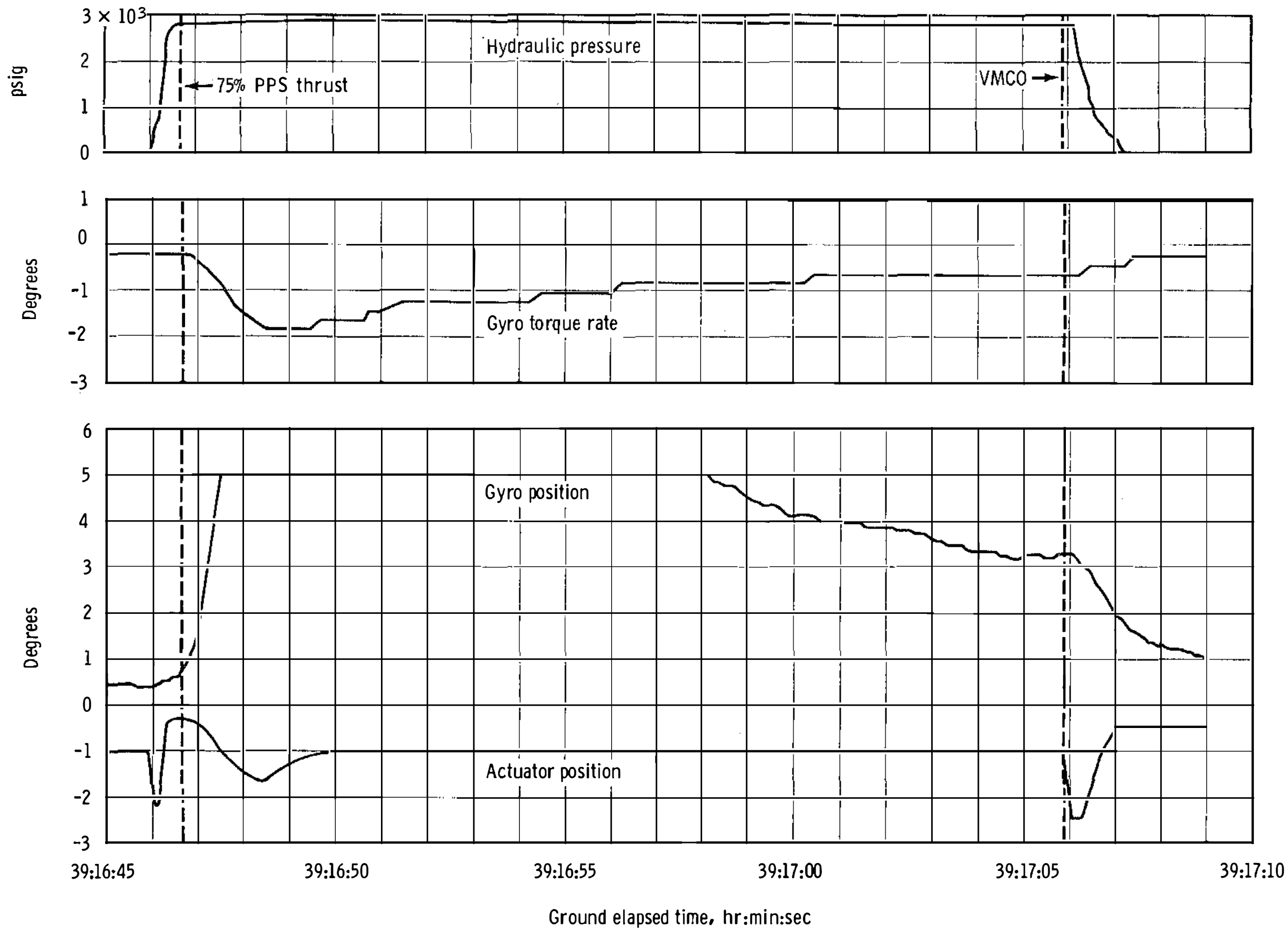

(a) Yaw axis

Figure 5.4.5-3. - Guidance performance (revolution 26 maneuver).

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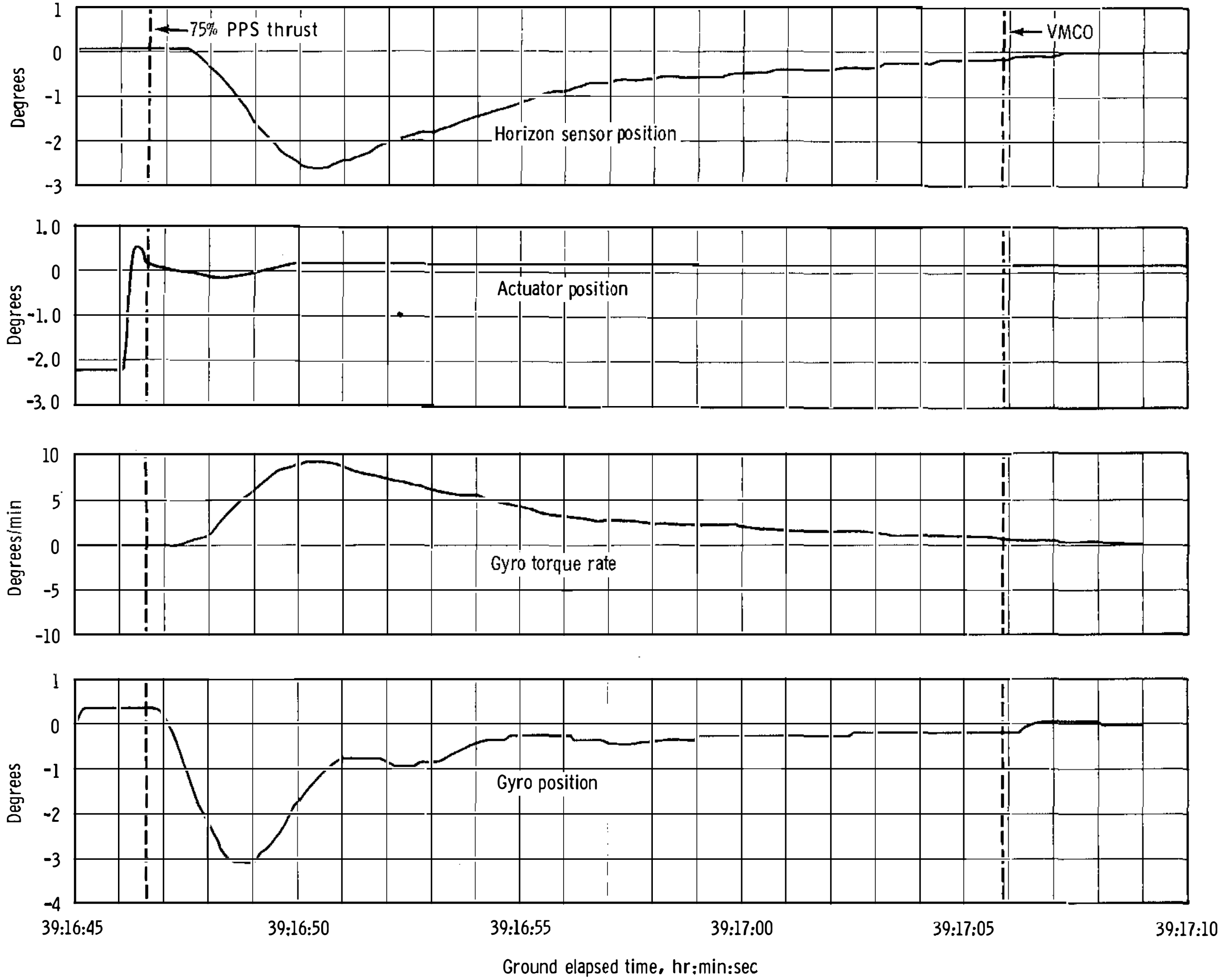

(b) Pitch axis

Figure 5.4.5-3. - Continued

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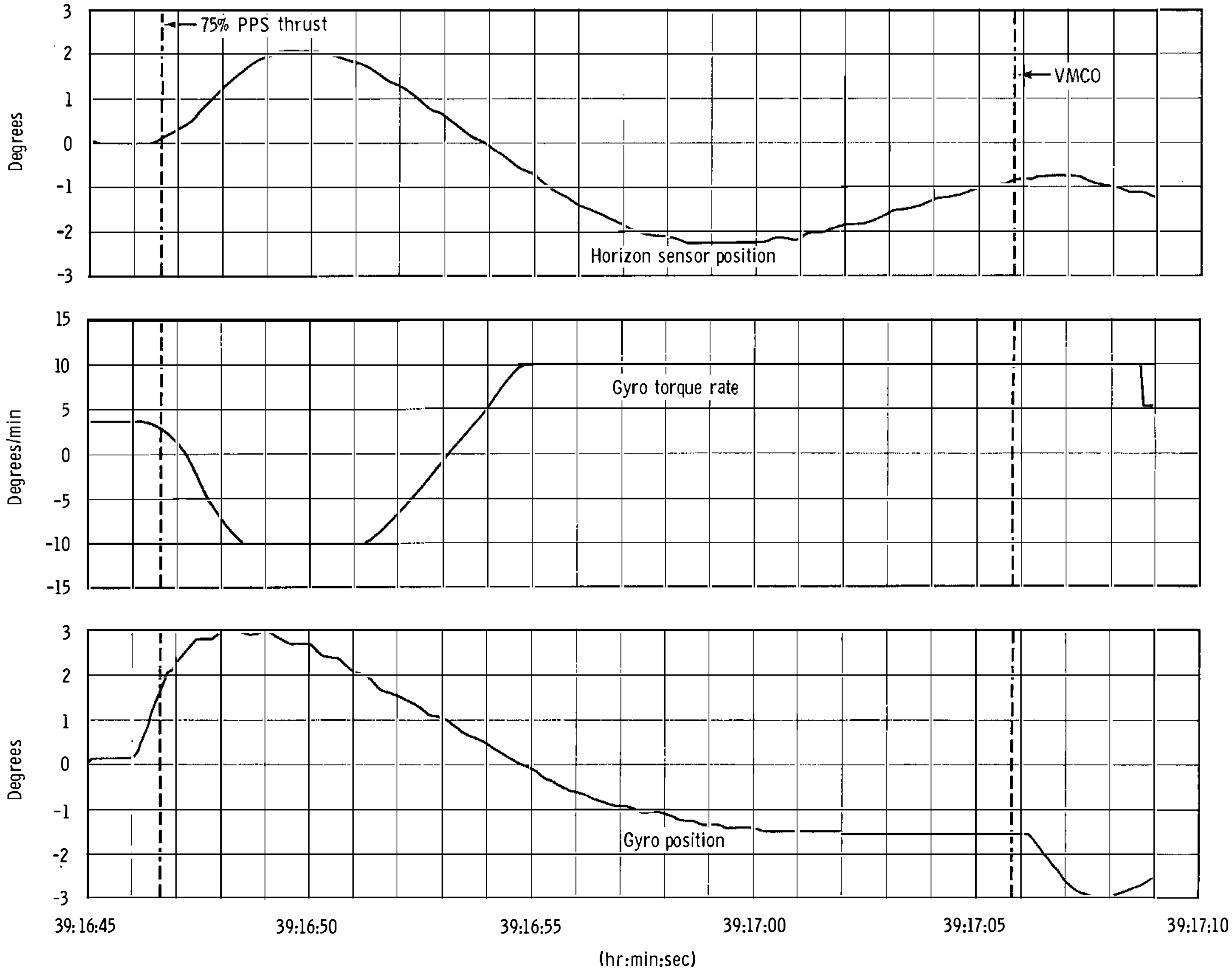

(c) roll axis

Figure 5.4.5-3. -Concluded.

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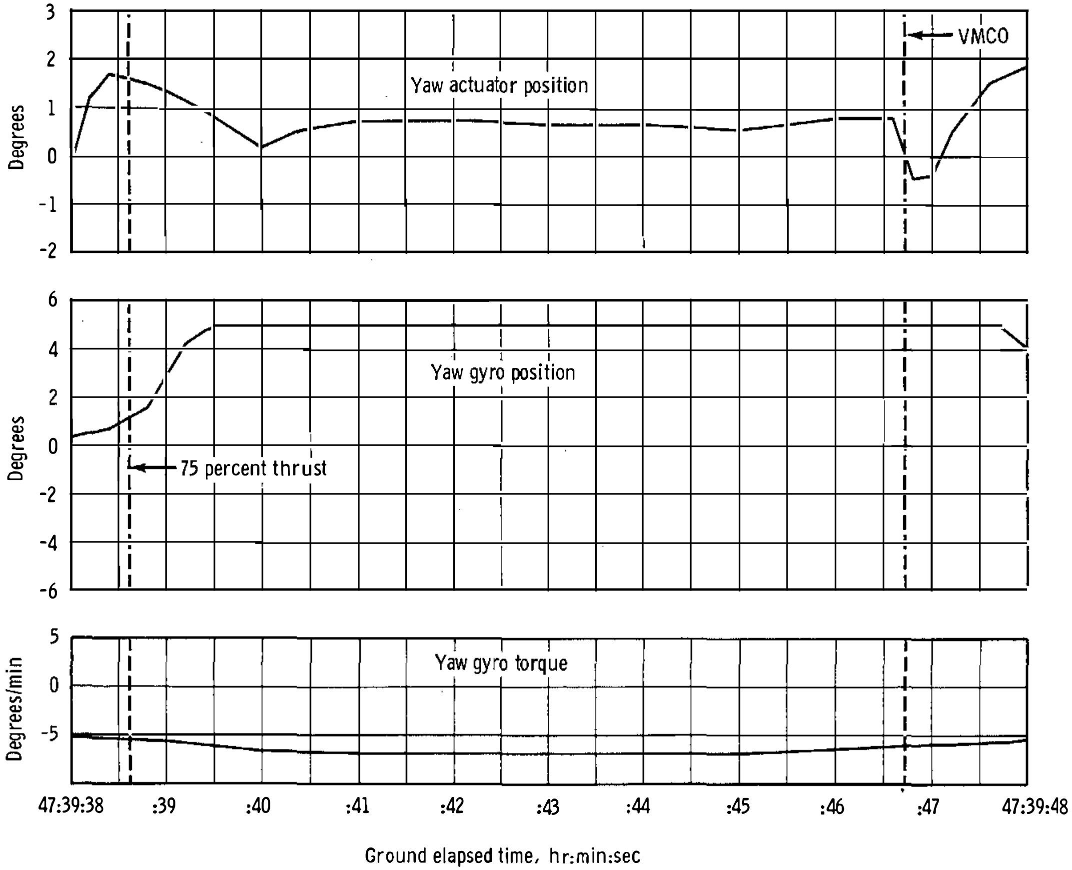

(a) Yaw axis

Figure 5.4.5-4. - Guidance performance (revolution 31 maneuver).

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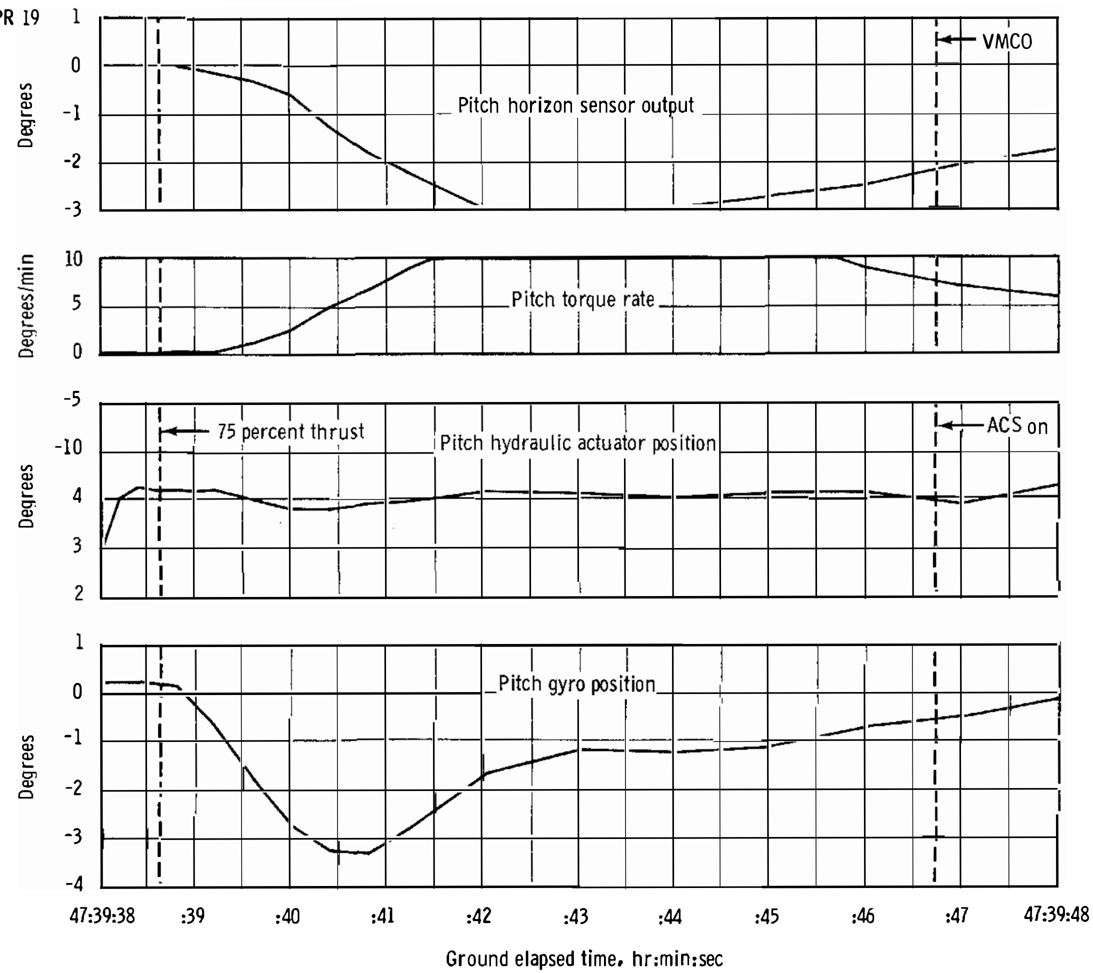

(b) Pitch axis

Figure 5.4.5-4. - Continued.

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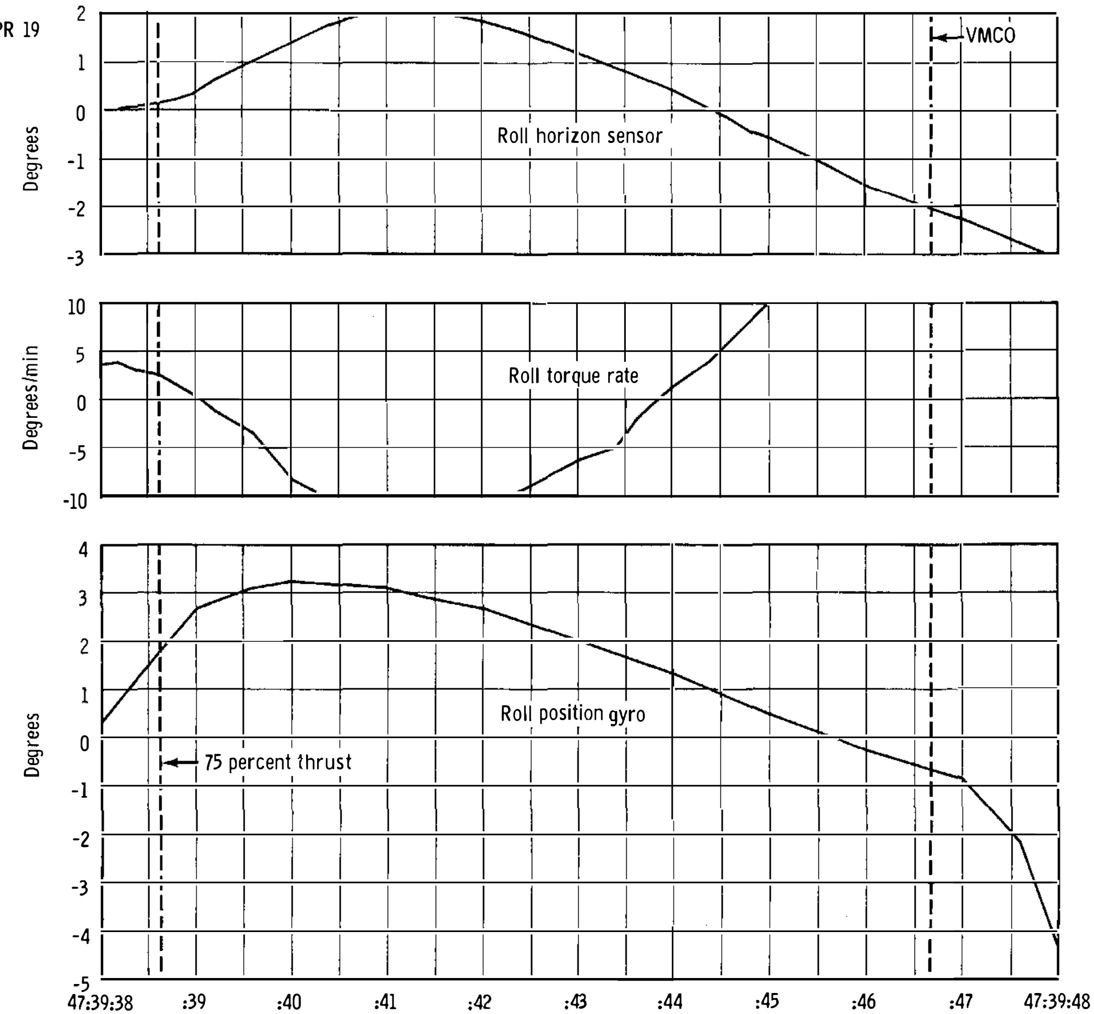

(c) Roll axis

Figure 5.4.5-4. - Concluded

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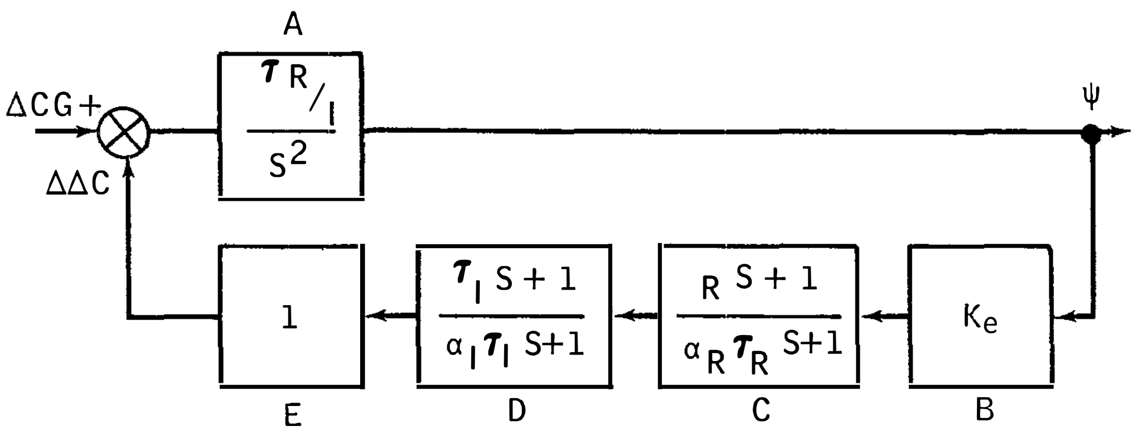

ΔCG	=	Change in vehicle center of gravity
ΔΔC	=	Change in feedback command
ψ	=	Gyro angle
A	=	Gyro response
B	=	Hydraulic static gain 0.8 degree engine/degree vehicle
C	=	Lead network, $\frac{2\ S+1}{0.5\ S+1}$
D	=	Lag network, $\frac{7\ S+1}{100\ S+1}$
E	=	Hydraulic servo dynamics, approximately one
S	=	Laplacian operator

Figure 5.4.5-5. - GATV hydraulic channel response.

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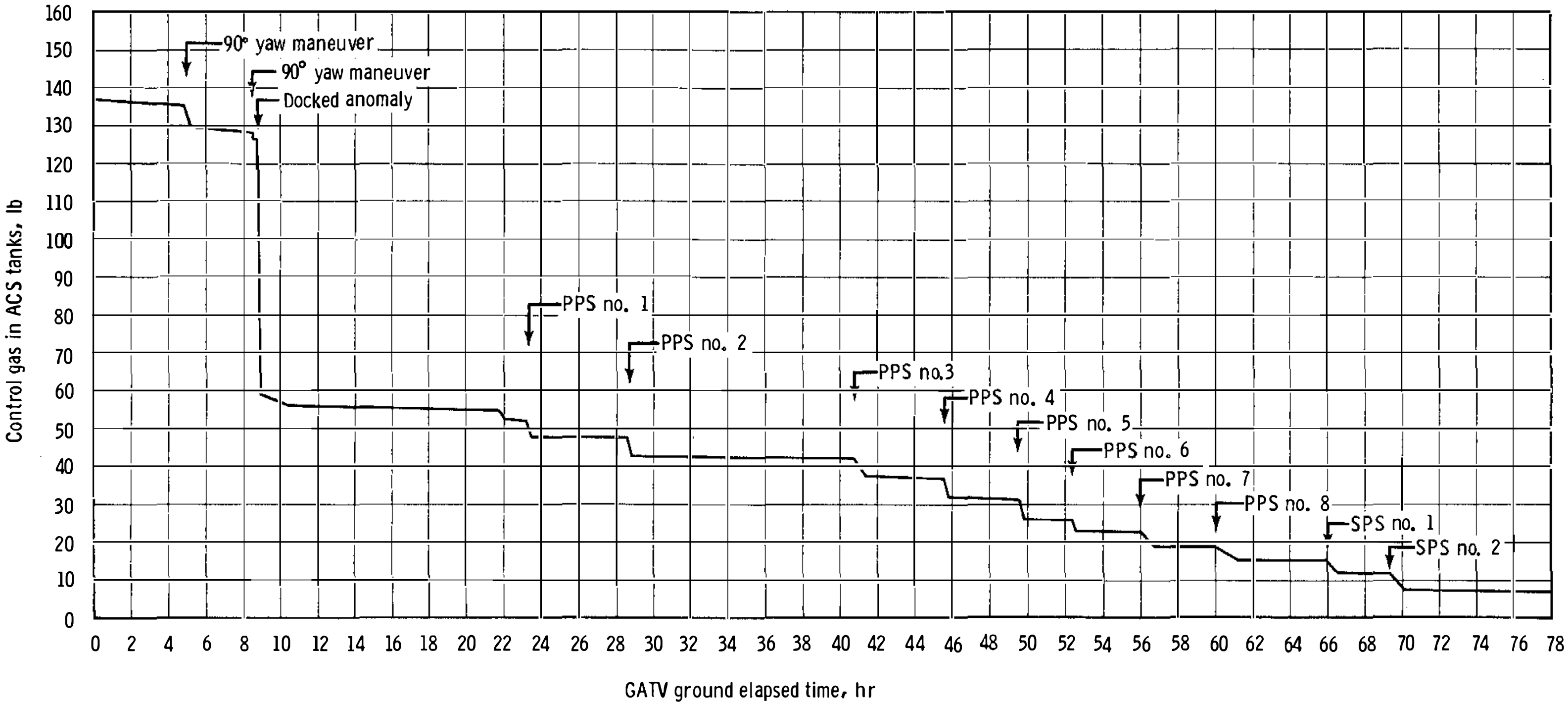

Figure 5.4.5-6. - GATV control gas usage.

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5.4.6 Electrical System

The Electrical System performed normally in all respects throughout the mission and to power depletion. The electrical parameters showing high, low, and mean current, voltage, and temperatures are shown in table 5.4.6-I.

5.4.6.1 Main-bus power.- The main-bus unregulated voltage followed the predicted discharge characteristics for the six primary batteries. A nominal 25-volt potential was maintained at an average load of 13.5 amperes. Fluctuations of load profile reflected the expected systems' functions throughout the mission. The capacity of the batteries was estimated to be 28 000 ampere-hours at launch. Battery power was depleted to 2710 ampere-hours (22 volts) by the end of revolution 122 with a complete loss of power (18 volts) estimated to have occurred sometime between revolution 131 and 132. These figures confirm the estimated battery capacity.

5.4.6.2 Regulated power.- All regulated dc voltages and the 400 cps, 3-phase, regulated ac voltage remained within specified limits.

5.4.6.3 Component temperatures.- The temperature indications of all Electrical System components (batteries, regulators, and inverter) were nominal and approximated the predicted values.

TABLE 5.4.6-I.- GATV ELECTRICAL PARAMETERS

Electrical telemetry parameter	Low	High	Mean
Unregulated bus, volts	22.0	29.0	24.7
Regulator no. 2 60W, volts	28.2	28.5	28.3
Regulator no. 1 60W, volts	28.2	28.3	28.3
Unregulated current, amperes	9.7	25.0	13.5
Regulator no. 1 20W, volts	-28.5	-28.8	-28.7
Battery temperature, °F			
No. 1	53.3	91.6	66.8
No. 2	59.1	74.5	68.7
No. 3	59.1	66.8	61.0
No. 4	53.3	59.1	56.5
No. 5	51.4	59.1	53.3
No. 6	45.6	59.1	51.4
Inverter temperature, °F	70.3	104.0	87.6
400 cps, phase AB, volts	115	115	115.0
400 cps, phase BC, volts	114	115	114.7
Structural current, amperes	0.41	*7.9	1.03
Regulator no. 2, 20W, volts	28.2	28.5	28.4
Pyrotechnic bus, volts	23.1	29.7	25.6
Regulator no. 2 temperature, °F	61.0	84.2	68.7
Regulator no. 1 temperature, °F	66.8	98.6	91.6

*High value observed during rigidize-motor operation; not indicative of actual value of current.

5.4.7 Instrumentation System

The Instrumentation System provided for the monitoring of 153 analog and 25 step-function (tell-tale) parameters. All instrumentation parameters were operative at lift-off and only two parameters (TDA accelerometer, A523, and nozzle external skin temperature, B-184) failed to provide good data during the mission. One additional parameter (TLV-GATV separation monitor, A14) provided degraded, but adequate, monitor signals.

The TDA accelerometer no. 1 (A523) mounted in the GATV Z-axis (yaw), experienced a period of intermittent operation from LO + 149.9 through LO + 201.8 seconds. The data obtained from all other periods appeared normal. Data from other vehicle accelerometers and vehicle events indicated that this was an isolated occurrence related only to the TDA Z-axis accelerometer.

The PPS nozzle-extension external skin temperature no. 1 (B184) provided erroneous data from the start of the PPS ascent maneuver to the end of the mission. This thermocouple was mounted on the edge of the nozzle extension of the PPS within the plume region of the SPS Unit I, +Y-axis. The primary purpose of the measurement was to analyze the thermal shock caused by PPS ignition. The secondary purpose was to measure the temperature of the nozzle extension during the operation of SPS Unit I. This nozzle-extension skin temperature indicated erroneous data during the cooling period after the PPS ascent maneuver. The rate of cooling was greater than that measured on parameter B185, which was also mounted on the nozzle extension. On later PPS maneuvers, the indicated temperatures of the suspected thermocouple (B184) rose and fell only with SPS Unit I initiation and termination. The temperature indication did not continue to increase with PPS ignition, as was expected. This indicated that the thermocouple junction was no longer bonded to the PPS nozzle extension, but was still within the plume of the SPS Unit I. After the PPS ascent maneuver, engine data from this parameter were considered inaccurate and erroneous. Thermocouple bonding techniques are being reviewed to preclude future failures of this type.

The TLV-GATV separation monitor A14 failed to provide the correct signals for separation and separation rate. This monitor normally reflects three successive voltage steps which establish the times of 3 steps of separation travel from which the rate of separation may be calculated. The first voltage increase establishes the time for 10 inches of vehicle separation. Two additional voltage increases which follow are associated with the additional travel of two increments of 30 inches each, from which separation rates may be calculated. The initial signal established the degree of separation noted between LO + 308.890 and LO + 308.952 seconds. The time of separation was also confirmed by other vehicle events and instrumentation. The initial voltage monitor level and the following voltage level were incorrect, and the third voltage level was impossible to read. The first voltage increase was 4.29 volts rather than 1.25 volts, and the second increase was to the telemetry full-scale voltage of 5.0 volts rather than approximately 2.5 volts. The third voltage level was expected to be approximately 3.75 volts, but was apparently off scale of the channel.

A similar output of this monitor was observed on the Gemini VI mission with the malfunction attributed to a shorted capacitor in the monitoring circuit. Gemini VIII data indicate that an instrumentation problem does exist. Post-mission testing has established that the actuation-switch lever arm resonates at a vibration frequency upon activation and presents erroneous switch closures to the monitor amplifier. Steps will be taken to eliminate this resonance in future vehicles.

5.4.8 Range Safety

Performance of the Range Safety System was satisfactory.

5.4.8.1 Flight termination system.- Both command receivers received adequate signal to execute commands throughout the ascent phase. No commands were sent and no spurious commands were received.

The following command sites were used:

LO to LO + 310 sec . . . . Cape Kennedy, high power

LO + 310 sec to LO + 500 sec . . . Grand Turk Island, high power

LO + 500 sec to LO + 650 sec . . . Antigua, high power

5.4.8.2 Track system.- The C-band transponder was used by various radars to provide input position data for the Instantaneous Impact Predictor (IIP) Computer. System performance was satisfactory.

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5.5 TARGET LAUNCH VEHICLE PERFORMANCE

The performance of the Target Launch Vehicle (TLV), an Atlas SLV-3, was satisfactory. The vehicle boosted the Gemini Agena Target Vehicle (GATV) to the required velocity and position for subsequent insertion into the planned orbit. The TLV also provided the required discrete signals to the GATV for system operation after staging, and for separation from the TLV.

The Gemini Atlas-Agena Target Vehicle (GAATV) was launched from Complex 14, Air Force Eastern Test Range, at 15:00:03.127 G.m.t. on March 16, 1966. There were no holds or difficulties encountered during the countdown.

5.5.1 Airframe

Structural integrity of the TLV airframe was satisfactorily maintained throughout the flight. The 5-cps longitudinal oscillation normally encountered after lift-off reached a maximum amplitude of 0.44g peak-to-peak at approximately lift-off (LO) + 7 seconds and was damped by LO + 20 seconds. This oscillation is excited during release of the launcher hold-down arms.

Axial-accelerometer data indicate peak accelerations at booster engine cutoff (BECO) and sustainer engine cutoff (SECO) of 5.95g and 2.90g, respectively. The expected accelerations were 6.28g and 3.07g. These differences are the result of the slightly-earlier-than-planned booster cutoff.

The engine-compartment thermal environment was normal, as indicated by data from five temperature transducers located in various areas in the thrust section. The maximum temperature was recorded near BECO and reached 100° F in the area of the sustainer fuel pump. The minimum temperature recorded during the boost phase was 43° F. This minimum occurred at LO + 65 seconds on the sustainer instrumentation panel.

Booster-section jettison, at BECO + 3 seconds, and GATV separation, at vernier engine cutoff (VECO) + 5 seconds, were normal. Gyro and accelerometer data indicate normal transients and vehicle disturbances at these times.

5.5.2 Propulsion System

5.5.2.1 Propulsion system.- Operation of the Propulsion System was satisfactory. A comparison of actual computed thrust with the predicted thrust levels is shown in the following table.

TLV Engine Performance

Engine		Pounds of thrust			
		Lift-off	BECO	SECO	VECO
Booster	Predicted	330 085	379 370	NA	NA
	Actual	324 440	375 840	NA	NA
Sustainer	Predicted	56 870	80 430	79 690	NA
	Actual	56 100	80 100	78 700	NA
Vernier	Predicted	1 150	1 405	1 040	1 050
	Actual	1 150	1 445	1 080	890

NA - Not applicable

The engines started at LO - 1.79 seconds and ignition, thrust rise, and thrust levels were normal prior to launch. The booster engines were cut off by a flight-control autopilot command at LO + 129.79 seconds. The sustainer engine operation was terminated upon command at LO + 283.68 seconds. The sustainer shutdown characteristics were as expected, and the vernier system transitioned to tank-fed operation satisfactorily. Vernier engine operation under tank-fed conditions was normal, with VECO command at LO + 303.936 seconds. A summary of the cutoff relay activations and the start-of-thrust-decay times for all engines is shown in the following table:

Event	Engine relay box activation, LO + seconds	Start of thrust decay, LO + seconds
BECO	129.794	129.875
SECO	283.678	283.726
VECO	303.936	304.059

The environmental temperature measurements reflected normal radiation heating during the sustainer phase of flight and indicated no evidence of cryogenic leaks, as were indicated during the Gemini VI TLV flight (SLV-3 5301).

5.5.2.2 Propellant utilization.- The propellant utilization system, consisting of a 6-point sensor system, a computer-comparator, and controls to the propellant utilization valve (main fuel valve to sustainer engine) operated properly. The system sensed levels in the liquid-oxygen and fuel tanks at six discrete points during flight and commanded the valve so as to end the flight with the optimum ratio of propellants remaining.

Propellant residuals at SECO were calculated from instrumented head-pressure ports in the liquid-oxygen and fuel tanks.

The liquid-oxygen head-pressure port uncovered immediately before SECO and the fuel head-pressure data were extrapolated to determine an uncovering time of 0.5 second after SECO. Usable propellant residuals based on these data are shown in the following table:

	Liquid oxygen, lb	Fuel, lb	Time to theoretical liquid-oxygen depletion, sec	Excess fuel at theoretical liquid-oxygen depletion, lb
Predicted	859	493	4.70	109
Actual	1036	843	5.59	403

These data indicate the fuel excess to be very close to the 3-sigma dispersion of 410 pounds for SLV-3 vehicles.

5.5.2.3 Propellant loading.- The tanking procedure was modified such that instead of loading the fuel tank to the 100-percent tanking probe plus 10 to 15 gallons, the vehicle was loaded on the 100-percent probe and 30 gallons were then drained. This change in procedure was a result of a suspected overfill problem during a preflight tanking operation.

5.5.3 Flight Control System

The performance of the Flight Control System was satisfactory. Vehicle transients at lift-off were moderate, as indicated by initial engine movement at LO + 0.73 seconds, and were quickly damped following autopilot activation at 42-inch motion. The lift-off roll transient reached only 0.17 degree in the counterclockwise direction at a peak rate of 0.78 deg/sec. Engine position shifts at booster-section jettison were normal. Gyro data provided indications that the roll and pitch program maneuvers were properly executed.

The usual rigid-body oscillations were observed as the vehicle passed through the region of maximum dynamic pressure. Maximum booster-engine positive-pitch deflections to counteract the effects of aerodynamic loading occurred at approximately LO + 63 seconds with an average deflection of 1.0 degree.

The programmer enabled guidance steering at 80.0 seconds; however, no steering commands were required during the boost-phase steering period. Spurious small-amplitude steering commands were noted on the pitch torque-amplifier output and in the pitch and yaw rate-gyro data after LO + 120 seconds. These commands occurred as a result of intermittent guidance-system lock.

Low-amplitude oscillations were observed between LO + 70 seconds and BECO, with a frequency that increased from 1.6 to 2.3 cps during that period. The oscillations were similar to those observed on previous SLV-3 vehicles, including 5301, and are attributed to sloshing of the GATV propellants.

The guidance-initiated staging discrete signal was indicated at the programmer input at LO + 129.65 seconds and the resultant switching sequence was successfully executed. Vehicle transients associated with BECO and booster-section jettison were normal and were quickly damped by the autopilot. The vehicle first bending mode occurred in the yaw plane between BECO and booster-section jettison. The zero-to-peak amplitude sensed by the rate gyros was 0.23 deg/sec at a frequency of 4.3 cps. Following booster-section jettison, the first bending mode occurred predominantly in the pitch plane with a zero-to-peak amplitude on the rate gyros of 0.21 deg/sec at a frequency of 5.2 cps and was damped out in approximately 3 seconds.

Proper system response was exhibited to all guidance steering commands. The initial steering commands during the sustainer phase resulted in low-amplitude rigid-body oscillations which were damped to negligible values by LO + 210 seconds.

The guidance SECO discrete was indicated at the programmer at LO + 283.67 seconds. The vernier attitude-correction steering commands were executed with no resulting control oscillations.

The guidance VECO and TLV-GATV separation discrete commands occurred at LO + 303.93 seconds and LO + 308.30 seconds, respectively. Gyro and axial-accelerometer data exhibited normal characteristics for these events. Displacement gyro errors and associated rates at VECO, at which time the GATV gyros were uncaged, are listed in the following table:

Axis	Displacement error, deg	Rate, deg/sec
Pitch	0.00	0.01 up
Yaw	0.01 left	0.00
Roll	0.07 CW	0.11 CCW

Rate-gyro and axial-accelerometer data, including the regions around staging and TLV-GATV separation, were reviewed and no abnormal disturbances or unusual indications were evident.

The usual effects on the SLV-3 as a result of GATV ignition were observed on the TLV rate-gyro data at approximately LO + 377 seconds.

5.5.4 Pneumatic and Hydraulic Systems

5.5.4.1 Pneumatic system.- Operation of the pneumatic system was satisfactory. The tank pressurization system properly regulated the main liquid-oxygen and fuel-tank ullage pressures during the boost phase of flight and the control system provided pressure for sustainer and vernier propulsion control.

Liquid-oxygen and fuel-tank pressures were stable at 28.6 psig and 64.5 psig, respectively, at lift-off, and at 29.2 psig and 65.5 psig at BECO. The differential pressure across the propellant-tank intermediate bulkhead was normal, being measured as 13.7 psid (fuel-tank pressure minus liquid-oxygen head pressure plus ullage pressure) at lift-off, 20 psid at BECO, and 20.5 psid at VECO. The minimum bulkhead differential pressure experienced during flight was 10.6 psid, at LO + 1.8 seconds.

During the boost phase, 86.7 pounds of the 148.6 pounds of helium aboard were used to pressurize the propellant tanks. The source pressure to the propellant-tank pressure regulators was 2970 psig at lift-off, and 1560 psig at VECO.

One minor problem was encountered during the countdown, at approximately T - 177 minutes, when the helium pressure supply to the airborne bottles indicated pressures to 3300 psi, compared to a normal 3000. It was determined that a pressure switch in the loading system (aerospace ground equipment) had failed to operate. The helium loading was switched to manual control and no countdown hold was required.

5.5.4.2 Hydraulic system.- The booster and sustainer/vernier hydraulic system pressures were adequate to support the demands of the systems throughout the countdown and flight.

At engine start, normal hydraulic pressure transients were indicated, followed by stabilization of system pressures to 3070 psig in the booster system and 3050 psig in the sustainer/vernier system. These pressures were satisfactorily maintained until the respective engine cutoffs. After SECO and cessation of sustainer pump output, the sustainer/vernier system reverted to vernier-solo accumulator operation. The vernier system pressure was 1500 psig at VECO. All return system pressures were normal.

5.5.5 Guidance System

The TLV was guided by the Mod III Radio Guidance System (RGS), which performed satisfactorily throughout the countdown and powered flight. This was accomplished by both the ground and airborne systems properly sending and decoding the required steering commands and discrete signals.

5.5.5.1 Programmed guidance.- Stage I programmed guidance, as indicated by rate-gyro output from the autopilot, executed the planned roll and pitch maneuvers successfully (refer to section 5.5.4).

5.5.5.2 Radio guidance.-

5.5.5.2.1 Booster steering: The radio-guidance ground stations acquired the pulse beacon of the TLV at LO + 58.6 seconds. Subsequently, lock-on was continuous until beyond LO + 350 seconds, except for the normal dropout during booster-section jettison, and the interval from LO + 120 to 124 seconds when lock-on was intermittent. Rate lock-on was acquired at LO + 56 seconds and, except for the normal dropout during booster-section jettison, was continuous until LO + 380 seconds, at which time tracking was terminated.

Booster steering, implemented to steer out Stage I dispersions as a function of look-angle constraints, was enabled by the TLV Flight Control System at LO + 80 seconds, as planned. However, no corrections were required during Stage I and, therefore, no steering commands were generated. Telemetered decoder-output data, however, indicated minor spurious pitch-down and yaw-left commands at approximately LO + 123 seconds. These commands, executed by the Flight Control System, were of low magnitude (approximately 4.0 percent) and were not unexpected during periods of intermittent lock. This condition was investigated as a result of its occurrences on previous SLV-3 flights, with the conclusion that it would not present any potential problem to the overall vehicle performance on future flights. BECO (as indicated at the programmer input) occurred at LO + 129.65 seconds at an elevation angle of 35.49 degrees. The errors at BECO were 96 ft/sec low in velocity, 6147 feet low in altitude, and 0.46 degree low in flight-path angle (refer to table 4.3-V).

5.5.5.2.2 Sustainer steering: Sustainer steering was initiated at LO + 145 seconds with a 70-percent yaw-left command of 1/2-second duration and an 85-percent pitch-up command of 1-second duration. The yaw commands were issued, as expected, to provide the preplanned dog-leg maneuver. The purpose of the dog-leg maneuver was to increase the Gemini Launch Vehicle (GLV) window and to provide the GLV with two second-day launch opportunities had they been required. Steering commands were less than 5 percent for the remainder of sustainer phase. SECO occurred at LO + 283.67 seconds.

VECO (as indicated at the programmer input) occurred at LO + 303.93 seconds at an elevation angle of 14.23 degrees. The VECO conditions were well within the 3-sigma limits. The initial velocity was nominal, the vertical velocity was 3.7 ft/sec low, and the yaw velocity was 0.5 ft/sec right. The following table compares the actual conditions of the achieved coast ellipse with those of the real-time filtered inflight desired conditions (i.e., real-time error analysis). The vernier corrections were transmitted at LO + 284 seconds and consisted of a 0.8-degree pitch-down attitude change and a 0.5-degree yaw-right attitude change.

VECO condition	Filtered inflight	
	Desired	Actual
Time from lift-off, sec	305.0	303.93
Space-fixed velocity, ft/sec	17 562.4	17 562.4
Vertical velocity, ft/sec	2 816.7	2 813.0
Yaw velocity, ft/sec	0.0	+0.5

5.5.6 Electrical System

Operation of the Electrical System was satisfactory during countdown operations and throughout flight. All electrical parameters were within tolerance. There were no evidences of unusual transients or anomalies.

5.5.7 Instrumentation System

5.5.7.1 Telemetry.- The TLV telemetry system operated satisfactorily during the flight. One lightweight telemetry package was used to monitor 114 parameters, distributed on 9 continuous and 5 commutated channels. All but two of these measurements provided good quality data. These measurements were GATV adapter surface temperatures, LA59T and LA57T; LT59T was invalid throughout the flight and LA57T yielded satisfactory data during only a portion of the flight.

The usual telemetry dropout was evidenced at booster-section jettison through the period from LO + 133.11 seconds to LO + 133.42 seconds.

5.5.7.2 Landline.- The landline instrumentation system carried a total of 47 analog and 54 discrete vehicle measurements. All 101 measurements provided satisfactory information until planned disconnect at lift-off.

5.5.8 Range Safety System

Operation of the Range Safety System was satisfactory. No range-safety functions were required or transmitted and no spurious range-safety commands were generated. Range-safety plots and telemetry readouts in Central Control were normal during the flight.

Radio frequency (RF) signal strength received at command receiver 1 indicated that adequate signal margins were available for proper operation of the RF command link at all times during the flight.

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5.6 GEMINI ATLAS-AGENA TARGET VEHICLE INTERFACE PERFORMANCE

Performance of the Gemini Atlas-Agena Target Vehicle (GAATV) interface was satisfactory throughout ascent and separation in accordance with reference 15. No structural problems were encountered and a normal separation occurred at 308.3 seconds after lift-off. Proper velocity was achieved and no excess pitch or roll motions were imparted to the Gemini Agena Target Vehicle (GATV) by the Target Launch Vehicle (TLV). Correct signals, such as sequence-timer start, uncage gyros, and separation, were transmitted to the GATV at the proper times. No flight-termination-system interface operation was required and no false operation occurred. More detailed discussions of these items are included in the report sections concerned with the appropriate systems of the TLV and the GATV.

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5.7 SPACECRAFT-GEMINI AGENA TARGET VEHICLE INTERFACE PERFORMANCE

Performance of the spacecraft-Gemini Agena Target Vehicle (GATV) interface was satisfactory throughout the flight and all systems functioned within the specification requirements (ref. 16). The performance of the electrical, mechanical, and command-system interfaces was determined from crew observations and from instrumentation data derived from the various systems.

All interfacing functions, including the GATV status display panel, mooring-drive system, L-band command link, acquisition lights, and approach lights, performed normally throughout the flight. The jettison of the aerodynamic shroud was normal and occurred at LO + 386.7 seconds. Target Docking Adapter (TDA) skin-temperature and accelerometer data are discussed in section 5.4.1.

The GATV lower acquisition light, which had been modified for this mission, was acquired at a range of 45 miles by the flight crew. Estimated brightness at that range was equivalent to that of a sixth-magnitude star. The GATV running lights were not visible to the crew until the spacecraft was within 200 feet. Close inspection during station keeping revealed that the aft green light was not on. The lights could not be used during docking because both the forward and aft green lights are required for alignment when docking is performed by the command pilot. The overall performance of the running lights was not fully evaluated on this mission because the acquisition lights were used for visual tracking during rendezvous, and all but a few minutes of the final approach and station keeping occurred in daylight. The TDA approach lights, together with the spacecraft docking light, provided sufficient illumination of both vehicles for attitude reference and docking.

All lights and gages on the GATV status display panel operated satisfactorily except the DOCK light which was dim and difficult to read. The apparent cause was failure of one of its two lamps. The crew reported difficulty in reading the panel at 50 to 75 feet. At that range the lights were readable only through the 6-power telescope on the hand-held sextant. The gages were not readable until docking was completed and the two vehicles rigidized. It was also reported that the gage dials were partially obscured by contamination or film on the cover glass.

The mooring-drive system operated normally during docking. Automatic rigidizing was completed 6.9 seconds after spacecraft engagement of the docking-cone latches. Spacecraft-to-GATV contact was estimated to be at 1-inch left of center with very little angular misalignment, and at a velocity of approximately 3/4 ft/sec. During spacecraft engagement and rigidizing, the TDA accelerometer indicated less than one g peak-to-peak in the horizontal (Y) and vertical (Z) axes and less than one-half g in the longitudinal (X) axis. The crew reported no visual evidence of electrical discharge at time of contact.

Initiation of the undocking sequence was accomplished by actuation of the recently added UNDOCK switch. Unrigidizing and separation occurred 3 seconds after switch engagement. Combined vehicle rates just prior to separation were:

Axis	Spacecraft rates, deg/sec	GATV rates, deg/sec
Pitch	+3	-3
Yaw	-2.5	-2.5
Roll	-5	+5

Post-separation telemetry did not indicate that the TDA latches had reset. This was attributed to the low voltage input to the TDA instrumentation relay caused by the GATV status display panel remaining in the dim condition. Subsequent cycling of the mooring-drive system with the status panel on BRIGHT provided the proper indication of latch reset.

6.0 MISSION SUPPORT PERFORMANCE

6.1 FLIGHT CONTROL

The Gemini VIII mission was controlled from the Mission Control Center in Houston (MCC-H). This portion of the report is based on real-time observations and may not agree with the detailed postflight analysis and evaluations in other sections of this report.

After the spacecraft recovery operations were completed, a modified three-shift operation was adopted to give real-time experience to less experienced personnel. During this latter phase of the mission, controller manning was required only in the following areas: Flight Director, Assistant Flight Director, Operations and Procedures, Network, Flight Dynamics, and Gemini Agena Target Vehicle (GATV).

6.1.1 Premission Operations

6.1.1.1 Premission activities.- The flight-control teams at MCC-H participated in vehicle-compatibility, launch-complex, and data-flow tests, and conducted the normal network simulations and systems tests. These activities began January 27, 1966, and continued through March 13, 1966.

6.1.1.2 Documentation.- Documentation for the mission was generally adequate and only the normal amount of updating was required after deployment of the remote-site controllers. Because the time span of the mission was compressed after the spacecraft recovery, the GATV solo-phase mission planning was accomplished in real time.

6.1.1.3 MCC/network flight-control operations.- The network went on mission status March 3, 1966, and flight controllers started deployment to the remote sites on February 27, 1966. Between March 3, 1966, and March 19, 1966, four Instrumentation Support Instructions (ISI's) were issued for telemetry calibration curve updates, and 23 ISI's were distributed to change the remote-site data-processor programs, causing some errors. Each ISI contained several changes.

6.1.1.4 Prelaunch.-

6.1.1.4.1 Gemini Atlas-Agena Target Vehicle countdown: The Gemini Atlas-Agena Target Vehicle (GAATV) countdown proceeded smoothly, running slightly ahead of schedule during most of the initial tests. Except for a slightly high structural current (which presented no in-orbit problems) on the GATV, no systems problems were noted during the countdown. Two minor telemetry problems were noted. The X-axis accelerometer indicated 13g high, and incorrect calibration data were supplied for the velocity-meter temperature measurements. During the GAATV trajectory run with the impact predictor (IP) 3600 computer, the crossrange nominal plot did not correspond to actual data sent from the IP 3600. Also, a one-velocity bit excursion was experienced and this was traced to a multiple parity error on the IP 3600 tape. This was considered to be a non-operational problem and the run was reported as successful.

6.1.1.4.2 Gemini Space Vehicle countdown: The terminal count was picked up by MCC-H at T - 680 minutes and proceeded normally. During the trajectory run, at T - 260 minutes, a problem was discovered in plotboard 4 in that it would not initialize; however, this problem was corrected prior to lift-off. Also, during the T - 260 minute run, an erratic update cycle was experienced during the first 60 seconds because of subchannel problems with the Real-Time Computer Complex (RTCC).

The only other problem noted in the terminal countdown occurred when low-speed messages sent during a Computation and Data Flow Integrated Subsystems (CADFISS) run were allowed to flow into the normal RTCC telemetry processor. This did not result in a serious problem, and proper procedures should preclude a recurrence.

GATV trajectory data from the Canary Island station defined a requirement for a spacecraft lift-off time of 16:41:03 G.m.t. Final recommended lift-off time based on GATV trajectory data from the Carnarvon and Woomera stations was 16:41:02 G.m.t. on a launch azimuth of 99.9 degrees. Other spacecraft launch windows associated with the GATV trajectory data were as follows:

Latest time for lift-off, G.m.t.	Spacecraft rendezvous apogee number
16:41:35	4 (upper orbit docking initiate)
16:43:23	5 (upper orbit docking initiate)
16:45:11	6 (upper orbit docking initiate)
16:47:14	6 (lower orbit docking initiate)

6.1.2 Powered Flight

6.1.2.1 Gemini Atlas-Agena Target Vehicle powered flight.- At GAATV lift-off, noisy high-speed data from the IP precluded updates on trajectory displays until T + 20 seconds. The first-stage GAATV trajectory was slightly low, with a maximum inertial flight-path angle of 28.8 degrees as compared to a nominal value of 30.2 degrees.

At staging, inertial velocity was nominal and flight-path angle was 0.6 degree lower than nominal. After staging, the high-speed trajectory data began to reflect considerable noise. The crossrange versus downrange distance plot was not usable, because the nominal data was plotted incorrectly.

GAATV sustainer engine cut-off (SECO) conditions were very close to nominal, putting the GATV coast-ellipse plotboard trajectory exactly on the nominal trace. Again, trajectory and sequential data were noisy for the early portion of the GATV Primary Propulsion System (PPS) thrust, with the trajectory nominal.

A tabulation of GATV insertion cut-off conditions follows:

	IP(RAW)	Bermuda
Mission recommendation	GO	GO
Velocity ratio V/V_R	1.000	1.000
Velocity (V), ft/sec	25 369	25 358
Flight-path angle (γ), deg	-0.01	+0.02
Altitude (h), n. mi.	161.0	161.0
Inclination (i), deg	28.9	28.9

The resultant orbit based on the transferred Bermuda insertion vector was 156.3 by 161.6 nautical miles. Subsequent low-speed tracking data through Bermuda on revolution 3 showed the orbit to be 159.8 by 161.4 nautical miles.

A later review of the ascent data revealed that at the start of PPS insertion thrust, a positive yaw transient greater than the telemetry transducer range (±5 degrees) was experienced. After the initial yaw transient, a steady-state offset of approximately +0.77 degree was obtained at lift-off (LO) + 425 seconds. The steady-state offset gradually increased to 0.93 degree at LO + 452 seconds, +1.65 degrees at LO + 471 seconds, +1.21 degrees at LO + 517 seconds, and +1.59 degrees at LO + 544 seconds just prior to PPS shutdown. The reason for the initial yaw transient was unknown during the mission period. Postflight analysis revealed the cause as a center-of-gravity offset (see section 5.4).

6.1.2.2 Gemini Space Vehicle powered flight.- The Gemini Space Vehicle lift-off occurred at 16:41:02.389 G.m.t. The flight-path angle in Stage I flight was approximately 1.5 degrees below the calculated nominal at its maximum, but had returned to nominal at staging. The RTCC-computed cut-off parameters were:

Source	Velocity, ft/sec	Flight path angle, deg	Altitude, n. mi.	Wedge angle, deg
GE/B	25 745	-0.16	86.9	0.07
IP (smooth)	25 741	-0.13	86.8	0.08
IP (raw)	25 685	-0.08	86.8	0.06
Bermuda	25 743	-0.05	86.8	0.08

During lift-off, a variation was noted in the indicated Environmental Control System (ECS) oxygen quantity (fig. 6.1-1). This variation was discounted because the cryogenic tank pressure did not vary. At LO + 109 seconds, parameter CA09 dropped (see fig. 6.1-1), reached zero at LO + 118 seconds, and returned to normal after staging. A second drop to zero started more slowly at LO + 210 seconds, bottomed out at LO + 232 seconds, and returned to normal after insertion. No further difficulties with the cryogenic gaging system were encountered.

6.1.3 Orbital

The GE/Burroughs insertion vector was transferred to the orbit phase and predicted an initial orbit of 85.4 by 155.6 nautical miles. Post-insertion tracking gave an orbit of 86.7 by 147.0 nautical miles and indicated a required plane-change maneuver of approximately 30 ft/sec.

Due to the slightly negative flight-path angle at spacecraft insertion, the line of apsides did not coincide with the prelaunch established maneuver line. Because of this misalignment, which would require a negative pitch angle of about 20 degrees, the radial velocity component of the coelliptical maneuver was predicted to be approximately 22 ft/sec. Personnel in the Auxiliary Computer Room (ACR) ran a study to optimize the maneuver line and found that by optimizing, the total change-in-velocity (ΔV) cost would remain approximately the same. Optimizing would change the coelliptical pitch angle to -2 degrees and would bring the coelliptical and terminal-phase-initiate (TPI) maneuvers closer together. It was agreed with the Flight Director to leave the maneuver line as it was established in prelaunch.

The final update of the height-adjust maneuver was based on Carnarvon tracking of both vehicles, and was uplinked to the crew over Hawaii as $\Delta V = 2.0$ ft/sec to be executed at 01:34:37 g.e.t.

The crew report of the height-adjust maneuver indicated that it was executed on time; however, they experienced problems in nulling the residuals. Tracking over the United States after the maneuver indicated an orbit of 86.7 by 145.3 nautical miles. Because of GATV S-band beacon heating limitations, the S-band beacon was turned off. Accelerometer bias updates from the Air Force Eastern Test Range (ETR) were loaded into the spacecraft by the Digital Command System at the completion of the height-adjust maneuver. The updated values for bias were valid and very accurate throughout the remainder of the flight, although the crew did question their accuracy at the end of revolution 2.

The initial update of the phase-adjust maneuver was passed to the crew over Antigua during revolution 2. The values passed were a ΔV of 49.3 ft/sec at 02:18:26 g.e.t. The final phase-adjust update was passed to the crew over the Ascension station as a ΔV of 50.6 ft/sec at 02:18:25 g.e.t. The crew reported that the maneuver was executed on time with all residuals nulled.

Over Carnarvon during revolution 2, the crew was given the initial update of the plane-change maneuver. This update required a ΔV of 26.2 ft/sec at 02:45:50 g.e.t., and was based on tracking by the Ascension station prior to the phase-adjust maneuver.

Carnarvon revolution 2 tracking (immediately following the phase-adjust maneuver) indicated that the phase angle between the two vehicles at the coelliptical maneuver point would be 0.09 degrees greater than desired, resulting in the TPI time being 3 minutes 42 seconds earlier than desired. Carnarvon also indicated that an additional plane-change maneuver of 4.4 ft/sec would be required after the planned one. The altitude differential between the target-vehicle orbit and the spacecraft orbit at the coelliptical maneuver point was predicted to be 15.0 nautical miles.

Tracking from Hawaii on revolution 2, after the plane-change maneuver, indicated a phase lag of 0.22 degree greater than desired and an altitude differential of 16.0 nautical miles at the coelliptical maneuver point. The Hawaii track also showed the two vehicles to be coplanar. The TPI time was predicted to be 8 minutes 23 seconds earlier than desired.

Based on the Hawaii tracking data, an additional height-adjust maneuver was scheduled over the States. This maneuver was passed to the crew as a ΔV of 2.0 ft/sec at 03:03:41 g.e.t. With this maneuver, conditions at the coelliptical maneuver were predicted to be an altitude differential (Δh) of 15 nautical miles, phasing such that TPI would be 1 minute 30 seconds early, with the vehicles in coplanar orbits.

White Sands tracking data after the second height-adjust maneuver predicted a Δh of 14.8 nautical miles and TPI 1 minute 30 seconds early. A preliminary coelliptical maneuver, with a ΔV of 61.6 ft/sec at 03:47:34 g.e.t., was passed to the crew over Antigua. The final update of the coelliptical maneuver, based on revolution 3 over Antigua, spacecraft tracking, and on revolution 3 GATV tracking by Eglin Air Force Base, was given to the crew over the Rose Knot Victor. This tracking data indicated a Δh of 14.6 nautical miles and a TPI time of 4 minutes 22 seconds late at the coelliptical maneuver.

The two-impulse processor was used to compute the terminal-phase backup maneuver in both the ACR and RTCC. Both ACR and RTCC ran a two-impulse solution using Pretoria C-band spacecraft revolution 3 vectors and Eglin C-band GATV revolution 3 vectors (pre-coelliptical maneuver data), and both solutions were in close agreement. Resulting conditions at TPI were as follows:

Time initiated	9 minutes 1 second late
Out of plane	1.7 ft/sec
Δh	14.5 n. mi.
ΔV total	31 ft/sec

The second two-impulse solution was computed using Hawaii C-band spacecraft revolution 3 and Eglin C-band GATV revolution 3 vectors (post-coelliptical data). Again both ACR and RTCC were in close agreement. The RTCC solution was passed to the crew over Texas. The resulting conditions at TPI were as follows:

Time initiated	8 minutes 14 seconds late
Out of plane	5.8 ft/sec left
Δh	14.5 n. mi.
ΔV total	32.6 ft/sec

A third two-impulse solution was run using California C-band spacecraft revolution 3 and Guaymas S-band revolution 4 vectors which confirmed the Hawaii solution except that the out of plane decreased to 3.7 ft/sec left.

The terminal-phase-final (TPF) maneuver was predicted to require a 40 ft/sec change in velocity.

It appears that Carnarvon tracking in the second revolution indicated that the phase-adjust maneuver was too small, and that TPI was going to occur approximately 4 minutes early. Hawaii tracking in the second revolution indicated that the Δh was 16 nautical miles, and that TPI would occur approximately 8 minutes early. Hawaii also recommended a 1.8 ft/sec height adjust, which would have the effect of delaying the predicted TPI about 7 minutes. However, tracking over the United States indicated that the phase-adjust was larger than it should have been by approximately 2 ft/sec; also, the second height-adjust that was made over California resulted in TPI occurring 9 minutes late.

The reason for the variations in predicted TPI time can be attributed to ground radar velocity errors, and possibly any extended nulling of desired-velocity-change residuals after the radar ephemeris was in process. This problem is under study; however, it is believed that the tracking radars were functioning properly and the results reflect the accuracy of single pass data combined with the procedures used.

Over Hawaii, revolution 3, the GATV C-band transponder was turned off and the S-band transponder was turned on. This was done to avoid possible interference between the two C-band transponders (spacecraft and GATV).

During the fourth revolution, a tank-pressure decay was noted in both of the fuel-cell reactant supply system tanks. The beginning of the parallel trend was noted at the Coastal Sentry Quebec and Hawaii during revolution 4, and confirmation was planned for Rose Knot Victor during revolution 5. At Rose Knot Victor acquisition of signal (AOS), however, both tanks were normal in pressure and the conclusion was that the fuel-cell oxygen and hydrogen heaters circuit breaker had opened early during revolution 4. At Tananarive on revolution 5, the crew verified that they had found the circuit breaker open and reset it between Hawaii and the Rose Knot Victor, thus restoring the tank heaters to normal operation.

The crew reported to the Rose Knot Victor on revolution 5 that they were docked with the GATV. The vehicle weights were thereafter combined in the RTCC program, and the crew was requested to turn the spacecraft C-band transponder off and turn both GATV transponders on.

At Coastal Sentry Quebec AOS on revolution 5, the crew reported that they were having a serious attitude-control problem. At that time they were undocked from the GATV and had armed the Reentry Control System (RCS). At approximately mid-pass, the crew reported that they were slowly regaining control of the spacecraft using RCS DIRECT-DIRECT. By Hawaii acquisition on the same revolution, the crew reported that the spacecraft was stabilized and telemetry indicated that they had used approximately two-thirds of the RCS propellant. (EDITOR'S NOTE: Postflight calculations showed that they had 25 pounds of propellant in the A-ring and 9 pounds in the B-ring just prior to retrofire, or approximately one-half of the total RCS propellant. The real-time indication of 4 pounds in the A-ring was caused by low gas temperature resulting from adiabatic cooling during the anomaly.) Over Hawaii, the crew reported that they had no control with the Orbital Attitude and Maneuver System (OAMS), and they also reported that they had no RCS ACME control, but that RCS DIRECT-DIRECT was functioning normally. When the crew reported control-system problems, all planned maneuvers were removed from the summary maneuver table, the RTCC-program vehicle weight was changed to that of the spacecraft, it was requested that the spacecraft reentry C-band transponder be turned on during the Hawaii pass, and based on Hawaii data, MCC-H made the decision for early mission termination. This decision was based on data which showed RCS propellant remaining in both rings to be less than half the amount loaded. Also, both rings of the RCS had been activated and significant propellant had been used. Mission rules required termination of the mission under these conditions.

In order to determine if any possible recontact problems would exist with the GATV after retrofire, the crew was asked to give their estimation of the location of the GATV. Remote sites also were asked to determine which vehicle was leading and the relative vehicle altitudes. It was determined that there was no danger of recontact after retrofire. The Flight Director was notified of the availability of the West Pacific landing area (zone 3) during revolution 6 or 7. Reentry lighting conditions for both revolutions were given to the Flight Director, who announced during the Hawaii revolution 5 pass that the reentry would be in zone 3, revolution 7. The RTCC and ACR were updated with new spacecraft weights, taking into account the RCS fuel already used. The ACR was requested to compute retrofire times without a spacecraft maneuver to provide separation from the GATV, while the RTCC was requested to compute times with a separation maneuver. A preretrofire onboard-computer update load for area 3 revolution 7, including a separation maneuver, was sent to the Rose Knot Victor and the Coastal Sentry Quebec. Open-loop zero-lift reentry times were also available in the event the crew was unable to load the data from the Auxiliary Tape Memory Unit (ATMU) reentry module. During the revolution 6 pass, the Rose Knot Victor updated the spacecraft computer with the time-to-retrofire (T_R) and reentry load, including a separation maneuver. It was determined between the Rose Knot Victor pass and the Coastal Sentry Quebec pass on revolution 6 that a separation maneuver was not needed for a safe retrofire, and the correct preretrofire load was sent to the Coastal Sentry Quebec and to Hawaii.

At the Rose Knot Victor on revolution 6, the crew stated that a complete check of the control system showed that thruster no. 8 was failed open at the time of the attitude control problem. Just prior to anticipated Rose Knot Victor loss-of-signal and after the computer had been updated for a 7-3 reentry, a time-of-equipment-reset (T_x) command was transmitted by the spacecraft communicator, who intended to send the updated T_R command to the Time Reference System (TRS). All spacecraft communications switches were in the manual position, thus preventing the T_x command from controlling equipment operation. At the Coastal Sentry Quebec on revolution 6, the crew reported that ATMU Module IV-A had been loaded into the onboard computer and that they had verified it with ATMU Module IV-B. There was a report from the crew that T_R was counting up; flight control personnel had no explanation for this in real time. The T_R, as calculated in the RTCC, was again modified and this new updated T_R was transmitted from the Coastal Sentry Quebec, after which all T_R indications in the spacecraft were normal. (Section 5.1.5 contains a discussion of this occurrence.)

At Hawaii, the crew was given the remaining backup quantities necessary for reentry. The crew also confirmed that their preretrofire update was correct as displayed by the Manual Data Insertion Unit (MDIU) readout, and at Hawaii they verified that the rate-command mode in RCS was now operational but that the reentry rate-command mode had not been verified. They also reported at this time that they had regained OAMS control and that they had adequate OAMS capability with which to align the platform. At Ascension, on revolution 7, the ground passed a recommended procedure for RCS usage during reentry. The procedure was to retrofire using dual-ring rate command, go to B-ring PULSE until 400K feet, and REENTRY RATE COMMAND thereafter, using B-ring until it was depleted before turning on the A-ring. Over Kano during revolution 7, T_R was in syncronization, but the crew reported that the T_R - 256 seconds telelight illumination did not occur.

The onboard telemetry tape-recorder data for the anomaly period was transmitted to the ground over Hawaii on revolution 5. A single playback of this data was made on site. From that playback, the following preliminary analysis and conclusions were made. The attitude control problem occurred at approximately 7 hours g.e.t. This was very near Tananarive loss-of-signal (LOS) on revolution 5. At that time the spacecraft and GATV were docked. Thruster 8 came on and created a yaw-left and roll-left torque. The possibility that the problem may have been caused by an electrical short circuit was discussed in the MCC-H. When the docked spacecraft—GATV combination started to yaw and roll, the GATV Attitude Control System (ACS) attempted to hold the combination stable, but did not have sufficient thrust. The crew commanded the GATV ACS off, with a resulting increase in angular acceleration and rate. The spacecraft OAMS was turned on and different attitude modes were tried in an attempt to control the docked combination. Several times the crew did get the rates down to very low levels while they were still docked. Although the telemetry event indicated that the thruster was on continuously, it was not clear from the on-site playback whether thrust was being continuously supplied by thruster 8. At approximately 7 hours 11 minutes g.e.t., the OAMS-regulated pressure dropped sharply to zero. This was concluded to be a transducer failure, since the OAMS source pressure continued to decrease beyond this point at the same rate as previously noted. At approximately 7 hour 13 minutes g.e.t., significant rates in all axes were noted; and at approximately 7 hours 15 minutes g.e.t., the crew undocked from the GATV after again attaining some stabilization. After undocking, angular rates became very large in a short period of time. The RCS was armed at approximately 7 hours 17 minutes, and all OAMS thruster circuit breakers were turned off. RCS DIRECT-ACME was inoperable due to some portion of the ACME being powered down, and control of the spacecraft was regained using RCS DIRECT-DIRECT. The ground calculation made from the Rose Knot Victor data on revolution 5 indicated that prior to the problem there were 157 pounds of fuel and 226 pounds of oxidizer remaining in the OAMS. ACR off-line calculations of OAMS fuel remaining after spacecraft control was regained indicated that there were 43 pounds of fuel and 144 pounds of oxidizer remaining. OAMS regulated pressure was assumed to be 300 psia for this calculation.

6.1.4 Reentry

Retrofire occurred on time at 02:45:49 G.m.t. (10:04:47 g.e.t.) March 17, 1966. IVI readings and a report that all four retrorockets had fired was the last voice transmission received from the spacecraft prior to blackout. No telemetry data were available during reentry.

6.1.5 GATV Orbital

The complete GATV mission profile is shown on figure 6.1-2, including the vehicle heading, flight-control modes, Primary Propulsion System (PPS) and Secondary Propulsion System (SPS) operations, and special tests. Table 6.1-I explains the flight-control modes.

Prior to the spacecraft docking, all GATV systems appeared normal. The Target Docking Adapter (TDA) was unrigidized over Carnarvon during revolution 1. The L-band beacon was turned on, the beacon boom antenna extended, and the status-display panel and approach lights were turned on over Hawaii on revolution 3. The GATV was yawed to a TDA-north attitude over Texas on revolution 3 in preparation for docking. The S-band beacon was turned off because of a slightly high temperature. The only anomaly noted during this yaw was the ACS control-gas regulated pressure, which dropped to 15 psi during the yaw; minimum expected pressure was 75 psi. The vehicle yaw appeared normal in spite of the low gas pressure, and no other adverse effects were noted. During revolution 5, when attempting to verify the uplinked stored-program-commands (SPC) for the docked yaw maneuver and loading of the velocity meter by the automatic mode, problems were encountered which later were attributed to the ground equipment.

Docking occurred over the Rose Knot Victor on revolution 5. ACS control gas required for docking was 2 pounds, as estimated from ACS control-gas pressure drop (preflight estimates indicated 2.5 pounds required for docking and undocking). The attitude gas usage for the GATV during the Gemini VIII mission is shown in table 6.1-II. At Rose Knot Victor LOS, the GATV was very stable with all systems operating normally.

Undocking was accomplished just prior to Coastal Sentry Quebec AOS during the period of spacecraft attitude-control problems. GATV attitudes were beyond the range of telemetry measurements for some time after spacecraft separation. The exact GATV attitudes at Coastal Sentry Quebec AOS were unknown since the ACS was turned off by the flight crew prior to undocking. The GATV was returned to flight control mode 1 from flight control mode 6; it then returned to TDA-forward stable flight within 30 minutes. Following GATV attitude stabilization, all flight-plan activities were discontinued pending spacecraft recovery. The majority of GATV activities following spacecraft landing and preceding the second PPS operation at 21 hours 42 seconds g.e.t. were composed of remote-site Digital Command System (DCS) checks to isolate the problems encountered in verifying SPC loads and loading the velocity meter (VM). The problem was found in a broken wire in a connector cable at the Rose Knot Victor and an improperly completed engineering instruction at Hawaii. Two minor anomalies were noted in the vehicle data during this period. The GATV pitch attitude remained at -2 degrees, occasionally moving to -1.8 degrees, then back to -2 degrees; this condition could be caused by a slight leak in the no. 2 attitude-control thruster. After operating the vehicle in flight control mode 3, the vehicle resumed normal slow limit cycling back and forth across the deadband (possibly operation of thruster 2 during operation in flight control mode 3 caused the valve to reseat, or seal). The second item was the abnormally long time period required for ACS control-gas regulated-pressure drop to a low pressure after completion of flight control mode 3 operations (low-pressure command was verified on telemetry subframe C).

Eight orbital firings were performed by the GATV PPS. The firings ranged from 0.85-second minimum impulse to a 19.6-second plane change, with the majority of the firings between 1 and 3 seconds. Of the eight firings, five utilized the short 22-second A ullage sequence. The start C 70-second ullage sequence was used for the other three PPS firings. The PPS performance appeared to be normal during all of the eight firings.

During the large out-of-plane PPS firing of 19.6 seconds, it became apparent that vehicle attitude was considerably off its intended yaw heading, resulting in a large in-plane velocity component. This same heading offset was again noted on the second out-of-plane PPS firing, the inclination-adjust maneuver, and once again resulted in a large in-plane velocity component. Analog records of all previous firings were reviewed. It was concluded that some sort of failure had occurred in the yaw hydraulics gain circuitry which had resulted in a reduction of the gain of the yaw-gyro error signal being applied to the engine yaw actuator of approximately 4 to 1. It was recommended that any remaining PPS firings be made in the docked-hydraulic-gains mode, which essentially doubles the gain. It was also decided at this time not to make any more out-of-plane maneuvers. An in-plane retrograde maneuver was planned to lower the apogee to 220 nautical miles. The results were near perfect. The yaw offset was again noted but the firing was short and the effect of slight yaw-heading errors had much less effect on the resulting orbit when the maneuver was in plane. On the basis of the success of this in-plane maneuver, two more in-plane maneuvers were planned, a dwell-initiate and a dwell-terminate maneuver, in order to deplete some of the propellants and to achieve a 220-nautical-mile circular orbit. These two PPS firings were performed and were very successful and accurate, although the yaw offset was noted during each firing. It should be noted that the yaw-hydraulics-gain problem was the only major system problem noted during the mission. The time of the firing and resultant orbit for each PPS and SPS operation can be found in the mission profile charts (fig. 6.1-2).

Because of the excessive control-gas usage during PPS operations, only 15 pounds of ACS control gas remained at the time the first SPS firing was to be initiated. As the SPS Unit II engines had not been previously operated, actual control-gas usage rates during SPS Unit II operation were uncertain (preflight estimate was 0.04 lb/sec). Also, the uncertainty of the ACS control gas remaining that was introduced by telemetry-system specification tolerances established 6 pounds as the lower limit for flight-planning activities. Based upon the above information, approximately 9 pounds of ACS control gas were available for SPS operations. The first SPS operation was planned for 20 seconds; this firing was intended to provide the first actual SPS in-orbit operation and verification of control-gas usage rates. The normal 6 minutes of gyrocompassing were eliminated to allow more accurate measurement of control-gas usage rates during the maneuver. Predicted ACS control-gas usage was 1.8 pounds (1 pound for yaw, and 0.8 pound for SPS Unit II operation). The first SPS Unit II operation occurred over the Canary Islands on revolution 41. This firing was performed using flight control mode 7 to reduce velocity-vector errors due to center-of-gravity (c.g.) offset. Control-gas usage during the firing was 2 pounds as compared with the predicted 1.8 pounds, providing confidence in the premission prediction rates and the capabilities to perform additional firings to SPS depletion.

Over the Eastern Test Range (ETR) on revolution 42, the second SPS Unit II operation was performed at the existing heading of +90 degrees. This firing was also performed with docked gains to reduce thrust vector errors due to c.g. offset. The predicted ACS control-gas consumption was 1.86 pounds. The firing appeared nominal except that 5 pounds of control gas were expended. Because of the high usage rate during the second SPS firing and the small amount of ACS control gas remaining, additional SPS operations were deleted and the remaining control gas reserved for guidance tests and attitude stabilization for the remainder of the mission. The GATV orbit after this final

SPS firing was 220 by 222 nautical miles with a 28.867-degree inclination angle.

Besides the PPS and SPS tests, several additional tests were performed with the GATV. These tests were as follows:

(a) Antenna-switching test: Over Carnarvon on revolution 41, the antenna was switched from the orbit antenna to the ascent antenna and left there for one revolution. This test was performed to determine telemetry-system capabilities using the ascent antenna. No appreciable change in signal strength was noted.

(b) Undocked orbit-coast operation in flight control mode 10: The purpose of this test was to evaluate the ACS gas consumption and gyrocompassing in an unusual flight-control mode. The vehicle was configured for ACS gain high/docked, wide deadband, low ACS pressure, and high horizon-sensor gains. (See table 6.1-II for control-gas usage during operation in various flight-control modes.)

(c) TDA rigidizing and unrigidizing sequences: The TDA was cycled through the rigidizing and unrigidizing sequence twice over the Coastal Sentry Quebec. The purpose of these tests was to exercise the TDA, to measure current rise and voltage drop during the sequence, and to verify that the latch-reset mechanism was functioning correctly. All TDA functions were normal. Subsequently, the TDA was unrigidized and rigidized a total of 25 to 30 times.

(d) Velocity meter loading: The purpose of the velocity-meter loading tests was to isolate the cause of the remote-site difficulties in loading the VM with the required 16 commands. Over the Coastal Sentry Quebec on revolution 39, the VM was loaded manually with all zeros (except the index bit). After verifying a correct VM word of all zeros and an index bit on one, the DCS was used to attempt the automatic loading of a VM word of all ones. The velocity meter word after this attempt was incorrect, indicating that not all of the DCS commands were accepted by the velocity meter. This problem was caused by the lack of a delay between the transmission of each command and has been corrected by providing a 90-millisecond delay between the transmission of each command of the VM load.

(e) Recovery from unusual attitude: The object of the recovery-from-unusual-attitude test was to obtain data on horizon-sensor performance and guidance-system response in recovering from an unusual attitude. The intent of the test was successfully accomplished by the vehicle perturbations following the spacecraft anomaly. The GATV stabilized within 30 minutes after the ACS was turned on over the Coastal Sentry Quebec on revolution 6.

(f) PPS start sequence A: The purpose of this test was to determine whether the PPS would start with a shorter SPS ullage orientation period than that normally used. This test was successfully performed five times.

(g) PPS minimum-impulse operation: The purpose of the PPS minimum-impulse operation was to determine the minimum PPS operating time, thus providing additional capability for small orbital maneuvers. This test was performed successfully over Texas on revolution 29.

(h) SPS operation without gyrocompassing period: An SPS operation was performed without a gyrocompassing period prior to the SPS firing. The purpose of this test was to determine gas-usage rates during SPS operation. The predicted ACS control-gas usage rates were verified with pressure and temperatures during the firing. After the mission, these usage rates will be defined from ACS thruster activity.

(i) Memory-readout interface tests: Numerous vehicle memory-readout interface tests with remote sites were performed, resulting in telemetry subframe B memory readouts. The purpose of these tests was to check the remote-site memory readout capabilities. A great deal of difficulty was encountered. The problem was traced to a ground hardware problem and is under investigation.

(j) Remote-site velocity-meter loading tests: Multiple remote-site velocity-meter loading tests, both automatic and manual, were performed. The results were as follows:

(1) Automatic - Negative results for the majority of the tests. This problem is under investigation and is believed to be a ground hardware problem.

(2) Manual loading - Positive results.

(k) L-band transponder temperature tests: The purpose of the L-band transponder temperature test was to determine temperature rise on the L-band faceplate temperature affected by leaving the L-band transponder on for indefinite periods of time. Data showed that no significant temperature rise was encountered.

(l) Yaw using gyrocompassing: The purpose of the test was to determine the capability, time, and control gas required to accomplish a yaw maneuver utilizing the gyrocompassing signal rather than the yaw on/off sequence. The test was accomplished over the United States on revolution 44. With the vehicle at a +90-degree heading, the gyrocompassing circuitry was configured for a heading of 180 degrees. Errors sensed by the horizon sensor to yaw gyrocompassing circuitry caused the vehicle to yaw to 180 degrees. This yaw maneuver was exceedingly smooth, vehicle pitch and roll positions did not exceed deadbands, and the control-gas usage was too small to be measured from pressure and temperature indications. Approximately 7.5 minutes were required to complete the maneuver.

(m) Gyro drift test: The purpose of this test was to determine the drift rate of the gyros. The GATV Guidance System was inertially referenced on revolution 44 by removing horizon sensors and geocentric rate. The difference between the horizon sensor output and the gyro position at precisely the end of one orbit measures the drift rate of the gyros. The roll-gyro and pitch-gyro drift were approximately 0.5 and 1.3 degrees, respectively.

MCC-H GATV support was terminated at 19:20:21 G.m.t., March 19, 1966, during revolution 47. At this time, 579 real-time commands and 1885 stored-program commands had been transmitted to the vehicle. The consumables used, up to this time, were as follows:

Consumables	Quantity used
Electrical power, amp-hr	980
ACS control gas, lb	133
PPS and SPS propellants	See table 6.1-III

TABLE 6.1-I.- GATV ATTITUDE FLIGHT-CONTROL MODES

Flight-control function	Undocked			Docked		
	Flight control mode 1	Flight control mode 2	Flight control mode 3	Flight control mode 6	Flight control mode 7	Flight control mode 10
ACS pressure	Low	Low	High	High	High	Special
ACS deadband	Wide	Narrow	Narrow	Wide	Narrow	--
ACS gain	Low	Low	High undocked	High docked	High docked	Combinations
Hydraulic gain	Undocked	Undocked	Undocked	Docked	Docked	--
Horizon sensor	Low	High	High	High	High	--

TABLE 6.1-II.- CONTROL-GAS USAGE

Maneuver	Control gas used, lb
PPS insertion with a 13-sec pitch maneuver	5.0
Yaw no. 1 (undocked) (0 to -90 deg heading)	1.5
Docking (flight control mode 6)	2.0
Yaw no. 2 (docked)	Data lost
Undocking (flight control mode 6)	58.0
Flight control mode 10, ACS gain Hi-DKD, wide deadband, low pressure	To small to measure
Flight control mode 3 for 4 minutes	1.3
PPS no. 1	7.0
PPS no. 2	4.5
Yaw no. 3 (undocked) (0 to -90 deg heading)	2.0
Yaw no. 4 and no. 5 plus PPS no. 3 (-90 to -93.6 to -90 deg heading)	2.5
Yaw no. 6 plus PPS no. 4 (-90 to 180 deg)	5.0
Yaw no. 7, no. 8, and no. 9 plus PPS no. 5 (180 to -90.9 to -90 deg)	6.0
Yaw no. 10 plus PPS no. 6 (-90 to 180 deg)	3.0
Yaw no. 11 plus PPS no. 7 (180 to 0 deg)	4.0
Yaw no. 12 plus PPS no. 8 (0 to 180 deg)	4.5
Yaw no. 13 plus SPS Unit II no. 1 (180 to +90)	3.0
SPS Unit II no. 2	5.0
Yaw no. 14 gyrocompassing yaw (-90 to 180 deg heading)	0

TABLE 6.1-III.- AGENA PROPULSION OPERATIONS

	Burn number										
	Insertion	1	2	3	4	5	6	7	8	9	10
Firing time, sec	183.3	1.21	1.19	19.6	0.850	7.38	2.47	2.2	2.76	21.0	51.0
G.m.t. of maneuver, hr:min:sec	March 16 15:06:01	March 17 14:23:49	March 17 19:44:37	March 18 07:57:29	March 18 12:42:25	March 18 16:20:20	March 18 19:27:54	March 18 23:20:10	March 19 04:09:01	March 19 09:11:49	March 19 12:19:50
G.e.t. of thrust, hr:min:sec[a]	[b]358 sec	21:42:47	27:03:35	39:16:27	44:01:23	47:39:18	50:46:52	54:39:08	59:27:59	64:30:47	67:38:48
Type of maneuver	Ascent 20 sec Unit I	PPSC	PPSC	PPSA	PPSC	PPSA	PPSA	PPSA	PPSA	SPS II	SPS II
ΔV required, ft/sec	8234.8	104.4	104.0	1600.0	96.0	789.0	272.0	247.7	309.1	63.0	152.7
PPS fuel consumed, lb	2853.22	22.66	22.36	406.76	17.66	145.57	41.73	37.82	39.46	0.0	0.0
PPS oxidizer consumed, lb	7399.72	104.53	103.76	835.15	90.12	420.89	153.99	143.94	148.8	0.0	0.0
SPS oxidizer consumed, lb	1.328	4.648	4.648	1.461	4.648	1.461	1.461	1.461	1.461	16.64	42.432
SPS fuel consumed, lb	1.208	4.228	4.228	1.328	4.228	1.328	1.328	1.328	1.328	14.44	36.822
PPS oxidizer remaining, lb	2284.28	2179.57	2075.74	1234.80	1144.28	719.99	566.00	422.1	274.1	274.1	274.1
PPS fuel remaining, lb	958.58	935.90	913.44	607.131	586.42	461.56	419.83	381.99	321.82	321.82	321.82
SPS oxidizer remaining, lb	175.79	171.142	166.494	165.033	160.385	158.924	157.463	156.002	154.541	137.901	95.469
SPS fuel remaining, lb	157.104	152.876	148.648	147.320	143.092	141.764	140.436	139.108	137.780	123.340	86.518
Burn time remaining, sec											
PPS	57.14	54.53	51.03	29.93	27.43	18.53	14.58	10.88	6.62	6.62	6.62
SPS I	2648.0	2578.0	2508.0	2486.0	2416.0	2394.0	2372.0	2350.0	2328.0	2076.11	1433.79
SPS II	210.41	204.85	199.29	197.54	191.98	190.22	188.46	186.70	184.94	164.94	113.94

[a]Times given are the initiation time of SPS ullage maneuver prior to PPS maneuver.

[b]Time from GAATV lift-off.

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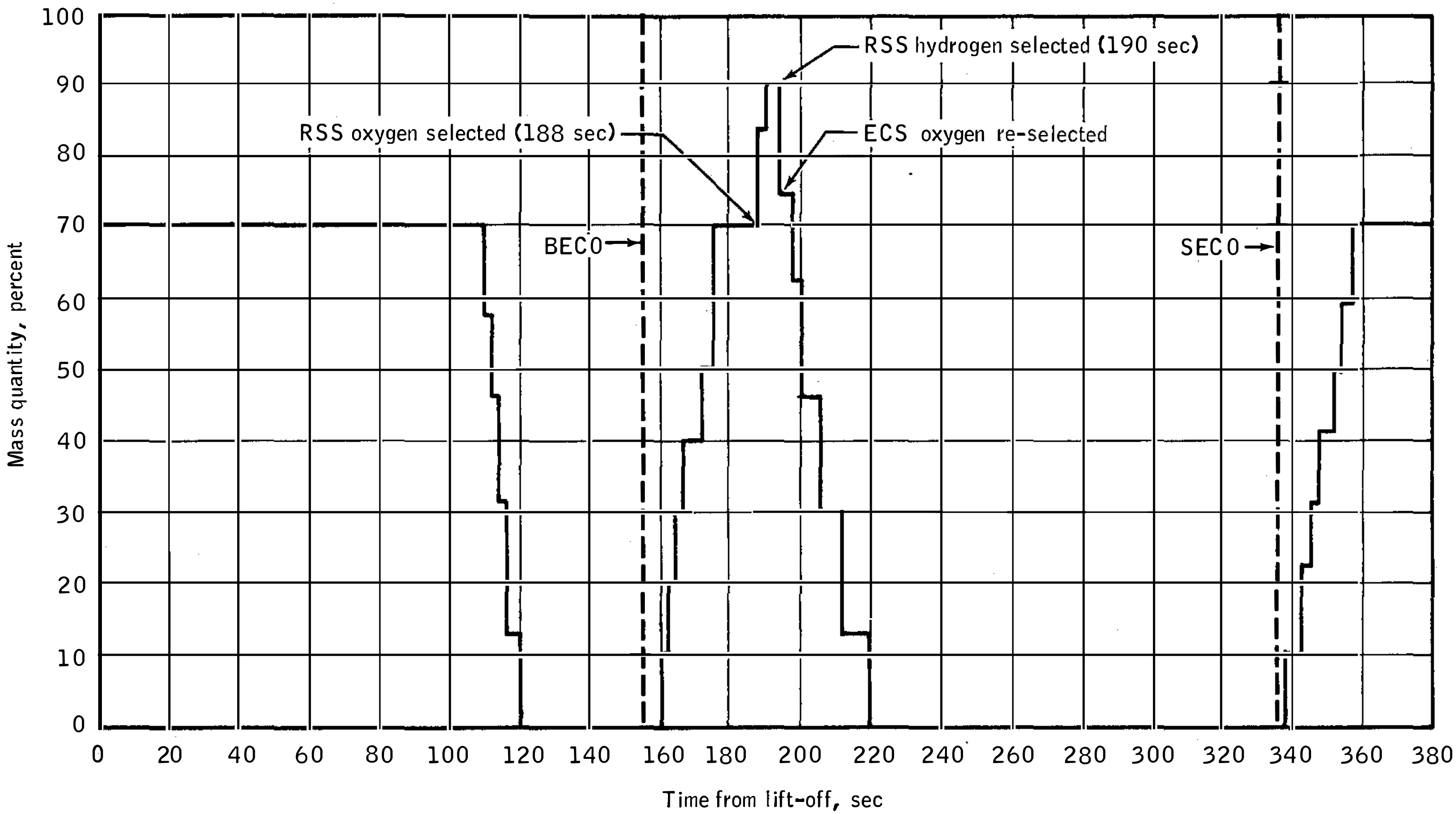

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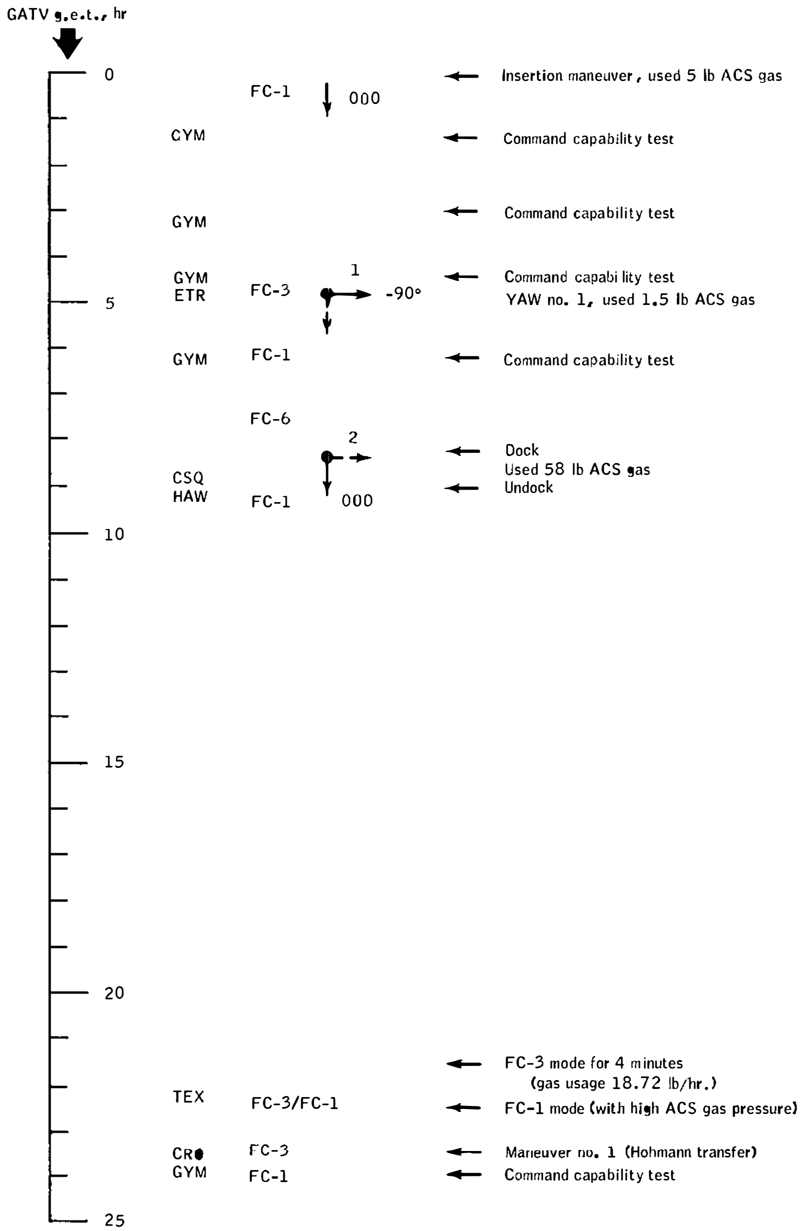
Figure 6.1-1. - Quantity variations in fuel-cell reactant supply system (RSS) during powered flight.

NASA-S-66-3474 APR 15



(a) 0 to 25 hr.

Figure 6.1-2. - GATV summary flight plan.

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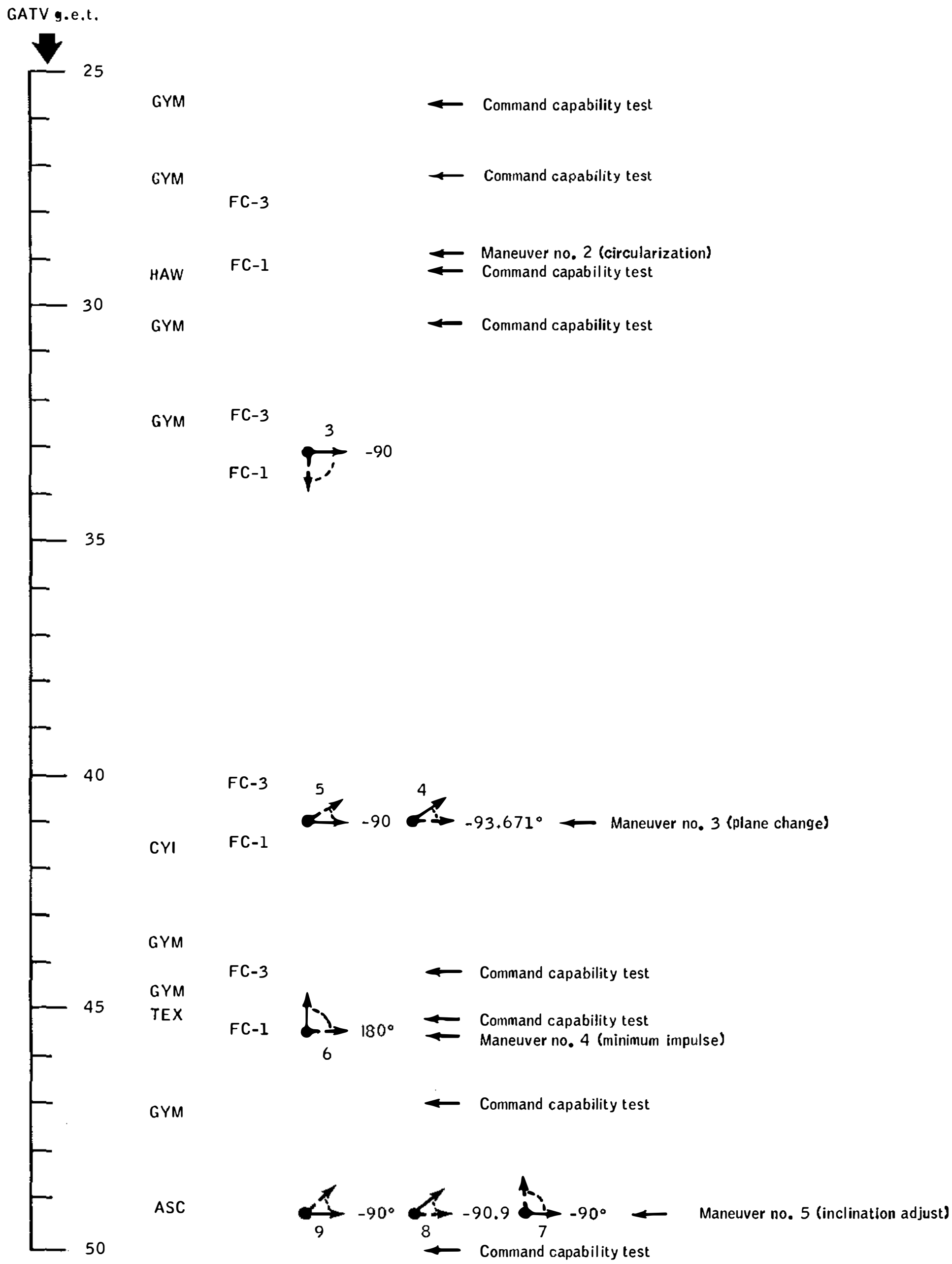
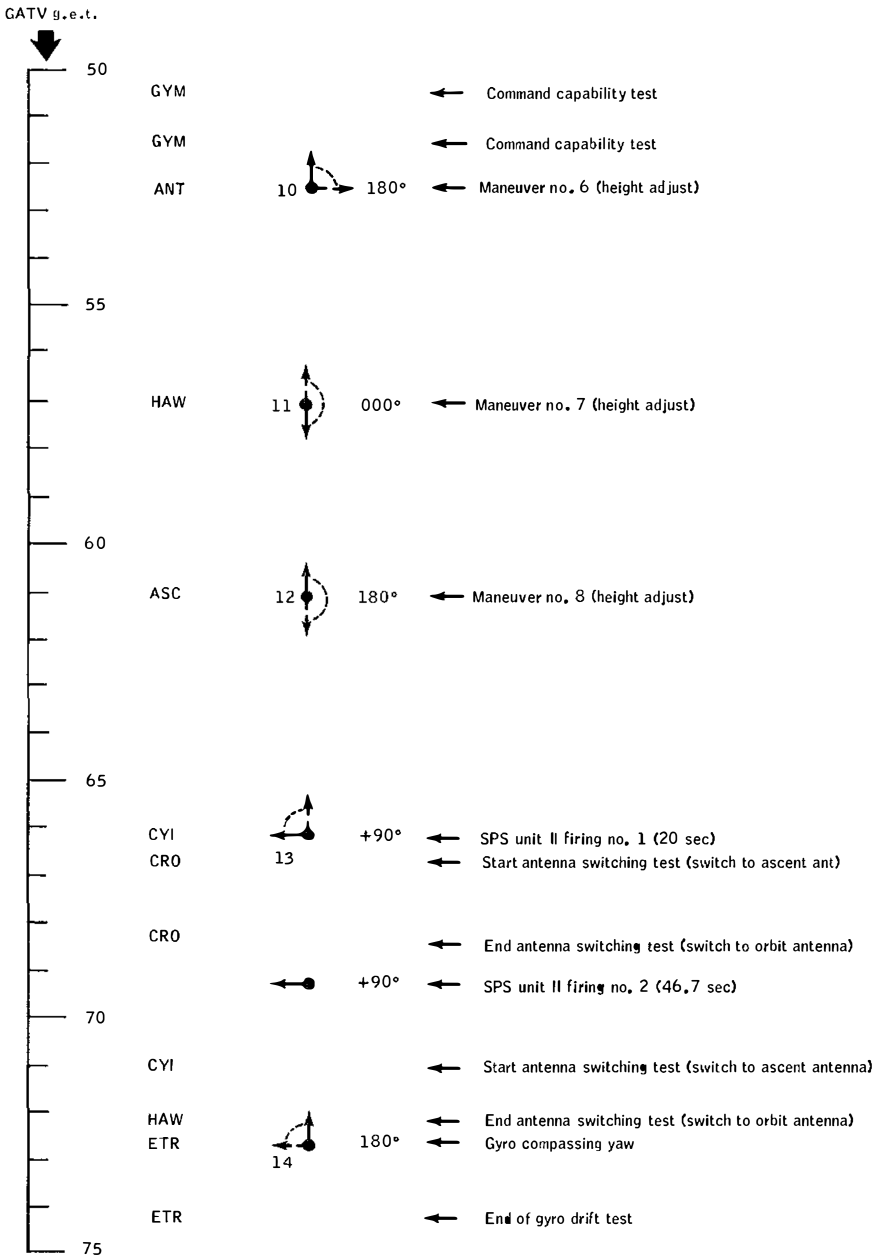
NASA-S-66-3475 APR 15



(b) 25 to 50 hr

Figure 6.1-2. - Continued.

NASA-S-66-3476 APR 15



(c) 50 to 75 hr

Figure 6.1-2.- Concluded.

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6.2 NETWORK PERFORMANCE

The network was placed on mission status for Gemini VIII on March 3, 1966. The Gemini Atlas-Agena Target Vehicle (GAATV) lift-off was at 15:00:03 G.m.t. March 16, 1966. The Gemini Space Vehicle lift-off was at 16:41:02 G.m.t. March 16, 1966. Spacecraft landing occurred at 03:23:35 G.m.t. March 17, 1966. The Gemini Agena Target Vehicle (GATV) was left in a near-circular parking orbit of approximately 220 nautical miles.

6.2.1 Mission Control Center-Houston (MCC-H) and Remote Facilities

The network configuration and the general support required from each ground station are indicated in table 6.2-I. Figure 4.3-1 shows the world-wide network stations. In addition, approximately 15 aircraft provided supplementary photographic, weather, telemetry, and voice-relay support in the launch and reentry areas. Certain North American Air Defense Command (NORAD) radars provided tracking of the Gemini Launch Vehicle (GLV) and spacecraft.

6.2.2 Network Facilities

Performance of the network is reported on a negative basis by system and site. All performance not detailed in this report was satisfactory.

6.2.2.1 Remote sites.-

6.2.2.1.1 Telemetry: The telemetry ground stations supporting the mission had no equipment problems of major importance. Several incidents such as receiver tuning that was too critical, a broken wire in the telemetry output buffer (TOB), and a defective TOB module caused data losses and dropouts. The premission brief-systems-tests/detailed-systems-tests (BST/DST) are being rewritten to include a check of the program under all signal conditions, thus insuring early detection and correction of these particular problems for future missions. The biomedical data from the Antigua station was intermittently poor, with a double electrocardiogram pulse being generated. This problem is currently under study.

6.2.2.1.2 Radar: Prior to the mission, the Hawaii Verlort radar was physically moved approximately 1100 feet to make room for the installation of the unified S-band system. Although some confusion resulted before the mission, the computers at the Manned Spacecraft Center and at the Goddard Space Flight Center were properly programmed prior to lift-off, and there was no loss of mission support.

During Computation and Data-Flow Integrated Subsystems (CADFISS) tests, a test bit is inserted in the radar data and is removed for flight by manual switching. The switching was inadvertently omitted, and the GATV real-time radar data at the Carnarvon station was lost for revolution 13. Procedures are being modified to reduce this possibility for future missions.

Radar support during the mission was very satisfactory. Problems were solved very quickly and efficiently as they developed. Several unique situations did occur during the mission. A teletype routing problem delayed data from the Hawaii station on spacecraft revolution 6 and from the Pretoria station on GATV revolution 17. The Woomera and California stations sent in third-range interval data which could not be accepted by the Real-Time Computer Complex (RTCC) since it was not configured to accommodate such long ranges. Several sites reported difficulty in tracking the GATV during revolution 45 due to poor signal strength. This resulted from the vehicle being placed in a nose-up attitude over Pretoria on revolution 45. The unusually high apogee of the GATV during revolution 14 caused an overlap of radar track between the Hawaii and California stations and between the Bermuda and Texas stations, in addition to the normal overlap between California, Guaymas, Texas, White Sands, Eglin, Bermuda, and the Air Force Eastern Test Range radars. A new beacon-sharing procedure was developed and successfully used for the remainder of the mission. Considerable interest was expressed both prior to and during the mission regarding the capability of Manned Space Flight Network (MSFN) radars to skin-track the GATV. The missile precision instrumentation radar system radars at MILA (Kennedy Space Center), Patrick Air Force Base, and Grand Bahama Island did skin-track the GATV during the active phase of the mission. Indications are that only FPQ-6/TPQ-18 type radars will be able to consistently acquire and track the GATV in skin mode. The FPS-16 radars at White Sands and Eglin Air Force Base may be able to skin-track the GATV on certain favorable passes. The MSFN radars continued to track the GATV after termination of the active mission period. The GATV beacons were expected to operate until battery depletion about March 24, 1966. During this period, the GATV became essentially a calibration satellite for network tracking radars.

6.2.2.1.3 Acquisition aids and timing: The performance of the acquisition aids and of the timing system was excellent throughout the mission. A total of 10 seconds of spacecraft data was lost at the Coastal Sentry Quebec due to a blown fuse.

6.2.2.1.4 Command: In the command area, several problems occurred in the FRW-2 transmitters; however, mission support was not affected because backup systems were available in all cases. A minor GATV message-acceptance-pulse (MAP) change was the only equipment modification required during the mission; the last four bits of the GATV eight-bit MAP were unstable and the ground MAP equipment was modified to ignore these bits.

Several sites experienced problems in automatically transmitting GATV velocity meter (VM) loads and receiving positive comparisons. In all cases the VM was successfully transmitted manually. After evaluating all available data, it was determined that the one-second automatic loading time in the VM register was marginal. This time restriction does not apply to manually transmitted loads. After spacecraft recovery, tests were conducted with the GATV using the Digital Command System (DCS) at the Texas station. The transmission time of the VM load was lengthened by modifying the DCS. Preliminary results showed a large improvement in the number of valid loads transmitted. Further investigations are underway at the present time.

6.2.2.1.5 Missile Trajectory Measurement (MISTRAM) System: The MISTRAM System supported the launches with no significant problems.

6.2.2.2 Computing.-

6.2.2.2.1 Manned Spacecraft Center (MSC) computing: The RTCC received high-speed data from the Impact Predictor (IP) and Burroughs/General Electric (B/GE) complexes via the launch trajectory data systems (LTDS) for both the GAATV and Gemini Space Vehicle launches. Data quality was good and both launches were nominal. Computer problems experienced during orbital operations are covered elsewhere in this report; however, it is worthy of note that the Mission Control Center at Houston, Texas (MCC-H) received the required real-time computer support at all times.

The RTCC received no data during the reentry phase of the mission. The nominal landing point for an area 3 revolution 7 reentry is at 25 degrees 15 minutes north latitude and 136 degrees 00 minutes east longitude. Based upon preretrofire data, nominal retrofire data, and the nominal retrofire time and sequence, the landing point was computed to be at latitude 25 degrees 13 minutes north, and longitude 136 degrees 05 minutes east.

6.2.2.2.2 Remote-site data processors (RSDP): The RSDP equipment and the telemetry on-line monitoring, compression, and transmission (TOMCAT-1) programs were operational for the mission except for some printout scaling and several engineering unit conversions. These were documented and sent to all sites in RSDP status messages prior to lift-off.

During GATV insertion, the Bermuda station had a 30-second drop-out of telemetry to MCC-H. Investigation revealed that the telemetry station was out of synchronization. Corrective action was taken and BDA supported effectively during the remainder of the mission.

The VM loading problem described in section 6.2.2.1.4 involved considerable time and effort from RSDP personnel. In addition, during revolutions 11 and 12, GATV data were lost from the Air Force Eastern Test Range downrange stations. After investigation it was determined that MCC-H could not accept GATV data without the Gemini synchronization counter in the output buffer stepping correctly. An interim corrective procedure was established at affected stations whereby the Gemini simulator was used at the same time that GATV line data were being processed. Changes to the MCC-H telemetry stations which will correct this situation permanently are being studied.

The GATV maneuver program was not operational when it arrived at the remote sites. Several changes were made in an attempt to correct the program; however, at mission termination it was still not operational. Additional effort is currently being expended to make the program operational.

6.2.2.2.3 Goddard Space Flight Center (GSFC) computing: The Goddard real-time system (GRTS) supported the mission without incident. The GRTS was used to generate nominal pointing data for the spacecraft, the GATV, and the Gemini Launch Vehicle (GLV). The GRTS was also used for testing the network during the F - 6 day network simulation as well as the F - 0 day terminal countdown.

The GRTS accepted high-speed data from the IP and B/GE complexes via the Launch Monitor Subsystem for both the GAATV and Gemini Space Vehicle launches. Parameters resulting from launch-phase computations were transmitted to the Mission Control Center at Cape Kennedy (MCC-K).

The predicted impact point of the GLV, as computed by GSFC, was at latitude 6.24 degrees north and longitude 110.69 degrees west. Time of reentry was computed to be 22:28 G.m.t., March 17, 1966. The landing point of Spacecraft 8 was computed to be 25.25 degrees north latitude and 136.00 degrees east longitude.

Upon termination of active mission support, the GSFC computers began to actively monitor the orbital flight of the GATV. This operation continued until the GATV batteries were depleted. Pointing data was generated and transmitted to the tracking network once every 24 hours.

6.2.2.3 Communications.-

6.2.2.3.1 Ground communications: Communications to all stations were generally better than for previous missions. With the exception of the Range Tracker, outages were few and quickly corrected. Normal propagation problems were encountered with an increase in both number and severity being observed toward the end of the mission. This condition had been predicted, based on solar activity.

Special efforts were made during spacecraft revolutions 6 and 7 to insure that voice and data transmissions would be in the best possible condition. This particular time was an unfavorable period at the Ascension site due to deterioration of day frequencies and below-peak efficiency of transitional night frequencies. The Cape Kennedy communications technician provided special backup radio circuits which utilized separate frequency assignments. In addition, Houston Recovery requested a voice circuit via NASA communications (NASCOM) facilities to Hawaii. This circuit, along with several Houston-Hawaii voice circuits from Department of Defense (DOD) resources, constituted voice communications channels to the deployed recovery forces.

6.2.2.3.2 Air-to-ground: Spacecraft communications were very good during the entire mission. The Texas station had a blown fuse in the primary UHF transmitter during revolutions 1 and 2; however, the standby transmitter was used with no loss of support.

6.2.2.3.3 Frequency interference: The California station reported radio frequency interference (RFI) on the HF air-to-ground frequency. Interference was moderate and in the form of oriental music. It was later determined that the source was Radio Peking. The California station also reported interference on the spacecraft real-time telemetry frequency. The source was found to be a National Guard transmitter. Appropriate action was taken. Cape Kennedy reported interference in the HF band. Again, the source was quickly identified and silenced. The Hawaii station reported RFI in the spacecraft telemetry band. Appropriate action was taken and the interference ceased.

TABLE 6.2-I.- GEMINI VIII NETWORK CONFIGURATION

Systems / [a]Stations	Acquisition aid	Air-to-ground remoting	C-band radar	Digital Command System	Data routing and error detection	Downrange uplink	Flight controller, air-to-ground	Flight controller, manned	Gemini launch data system	GLV telemetry	High-speed radar data	High-speed telemetry data	SPANDAR	Radio frequency command	Remote-site data-processor summary	S-band radar	Delayed time telemetry	Recovery antenna telemetry	R and R telemetry	Real-time telemetry display	Teletype	Voice (SCAMA)
MCC-H		X		(X)			X				X				X		X			X	X	X
MCC-K	X			X	X				X	X	X	X					X	X	X		X	X
A/C		X																	X			
ANT	X	X	X			X						X		X			X	X	X	O		
ASC		X	X															X	X			
BDA	X	X	X			X					X	X		X		X	X		X	O	X	X
CAL	X	X	X													X					X	X
[b]CNV			X			X				X				X					X	X		
CRO	X		X	X			X	X						X	X	X	X		X	X	X	X
CSQ	X			X			X	X						X	X		X		X	X	X	X
CTN	X	X																	X		X	X
CYI	X		X	X			X	X						X	X	X	X		X	X	X	X
EGL	X		X																		X	X
GBI	X	X	X			X				X		X		X		X	X	X	X	O		
GTK	X	X	X			X				X		X		X		X	X		X	O		
GYM	X						X	X							X	X	X		X	X	X	X
HAW	X		X	X			X	X						X	X	X	X		X	X	X	X
KNO	X	X																	X		X	X
MLA			X																			
PAT			X																			
PRE			X																			
RKV	X			X			X	X						X	X		X		X	X	X	X
RTK	X	X	X																X			
TAN	X	X																	X		X	X
TEX	X	X		X		X					X	X		X		X	X		X	(*)	X	X
WHS	X		X																		X	X
WLP													X								X	X
[c]WOM	X		X																		X	X

[a]Location of stations is shown on figure 4.3-1(a).

[b]Wind profile measurements in support of planned recovery operations.

[c]Non-interference basis.

Legend:

(X) Master Digital Command System
O Remoting
(*) Real-time and remoting

6.3 RECOVERY OPERATIONS

6.3.1 Recovery Force Deployment

As in previous Gemini missions, recovery plans and procedures were devised for the rapid location and safe retrieval of the spacecraft and flight crew following any conceivable landing situation. For planning purposes, Gemini landing areas are divided into planned landing areas and contingency landing areas. The planned landing areas are further divided into the launch-site landing area, launch-abort (powered flight) landing areas, secondary landing areas, and the primary or nominal end-of-mission landing area. A landing outside one of these planned landing areas is considered to be a contingency landing.

Department of Defense (DOD) forces provide support in all of these various landing areas. The level of support provided is commensurate with the probability of a landing in a particular area and also with any special problems associated with such a landing. Table 6.3-I contains a summary of those forces committed for Gemini VIII recovery support.

The planned landing areas in which support forces are prepositioned for search, on-scene assistance, and retrieval are located and defined as follows:

(a) Launch-site landing area is that area where a landing would occur following an abort during the late portions of the countdown or during early powered flight. This area extends approximately 40 nautical miles seaward from Cape Kennedy and 3 nautical miles west from Launch Complex 19. Recovery forces deployed in this area for the Gemini VIII mission are shown in figure 6.3-1.

(b) Launch-abort (powered flight) landing areas are areas within the boundaries formed by the most northern and southern launch azimuths, the seaward extremity of the launch-site landing area, and the west coast of Africa. A landing within these boundaries would occur following an abort above 45 000 feet and prior to spacecraft orbital insertion. Recovery-force deployment in these areas is shown in figure 6.3-2.

The secondary landing areas are located in four zones placed around the world in the West Atlantic, East Atlantic, West Pacific, and mid-Pacific. Landing areas were designated within these zones each time the ground track crossed the zone. The positions of these areas thus provide landing areas periodically throughout the flight and prior to the nominal end-of-mission. It is this type of landing area that was used in the West Pacific following the inflight emergency aboard Spacecraft 8. Typical recovery support in these areas (figs. 6.3-3 to 6.3-5) is a destroyer equipped with a retrieval crane and search/rescue aircraft on alert at nearby air bases.

The fourth type of planned landing area is the primary landing area where the spacecraft would land following a nominal mission. For Gemini VIII, this area was located in the West Atlantic, zone 1, and because of its higher probability of use, the recovery support deployed consisted of the LPH4 aircraft carrier U.S.S Boxer, helicopters, tracking aircraft, and search/rescue aircraft. Support provided for this area is shown in figure 6.3-4.

The contingency forces consisted of aircraft deployed to staging bases around the world (fig. 6.3-5) so that they could reach any point along the ground track within 18 hours of notification of a spacecraft landing. When possible, preselected contingency aiming points are designated near recovery zones or along contingency lines (fig. 6.3-5) to take advantage of the nearby location of recovery forces.

6.3.2 Location and Retrieval

The flight crew initially reported the difficulties sustained in spacecraft attitude control during the pass over the tracking ship Coastal Sentry Quebec near the West Pacific landing area 5-3 (revolution 5 – landing zone 3). Following this report, forces in the West Pacific zone were alerted for a possible landing in that area. A short time later the decision was made to terminate the mission in landing area 7-3. Recovery forces were notified of this decision and proceeded toward the aiming point at 25°15' N. latitude, 136°00' E. longitude. The sequence of recovery events was as follows:

Time, hr:min		Event
G.m.t.	g.e.t.	
March 17 00:08	7:27	Aircraft at Okinawa and Tachikawa alerted for possible spacecraft landing in West Pacific area.
00:24	7:43	U.S.S. Leonard Mason ordered to proceed at best speed to 7-3 aiming point (25°15' N., 136°00' E.)
00:33	7:52	Naha Rescue 1 (HC-54) airborne.
00:47	8:06	Naha Rescue 2 (HC-54) airborne.

Time, hr:min		Event
G.m.t.	g.e.t.	
01:15	8:34	Rescue 2 aborted with fire.
01:59	9:18	Second Rescue 2 airborne.
02:45	10:04	Spacecraft retrofire.
03:06	10:25	Naha Rescue 1 was on station at the aiming point. Naha Search 1 (HU-16) was on station 100 nautical miles uprange from the aiming point. U.S.S. Mason was approximately 115 nautical miles north of the aiming point with an estimated time of arrival of 13 hours 28 minutes g.e.t.
03:17	10:36	U.S.S. Mason reported radar contact with the spacecraft at a range of 105 nautical miles.
03:20	10:39	Naha Rescue 1 sighted spacecraft on main parachute at a range of 3 nautical miles.
03:21	10:40	U.S.S. Mason report of weak signals on spacecraft voice frequency (296.8 mc) received at Mission Control Center - Houston.
03:22	10:41	Naha Rescue 1 reported spacecraft landing and flotation attitude normal.
03:26	10:45	Naha Rescue 1 report of landing position as 25°14' N., 135°50' E., received at Kunia Control Center.
03:35	10:54	Pararescueman deployed to spacecraft. HF DF network reported fix on spacecraft as 25°24' N., 136°00' E. ± 120 nautical miles.
04:11	11:30	Naha Search 1 report that spacecraft flotation collar in place received at Kunia Control Center.

Time, hr:min		Event
G.m.t.	g.e.t.	
04:26	11:45	Naha Rescue 1 report that flight crew in good condition received at Kunia Control Center.
04:29	11:48	Report of spacecraft hatches open received at Kunia Control Center.
05:31	12:50	R and R Section found 100 yards from spacecraft and marked by smoke.
06:05	13:24	U.S.S. Mason reported visual contact with spacecraft.
06:28	13:47	Flight crew boarded U.S.S. Mason (fig 6.3-6).
06:37	13:56	Spacecraft secured onboard U.S.S. Mason. Pickup point was reported by the U.S.S. Mason as 25°22' N., 135°56' E. (fig 6.3-7). (Apparent difference between pickup point and landing point is probably due to small navigation errors in determining ship and aircraft positions. With low wind velocity it is difficult to attribute the difference between points to the drift of the spacecraft while waiting arrival of the ship.)
06:56	14:15	R and R Section secured onboard U.S.S. Mason.
07:15	14:34	U.S.S. Mason reported estimated time of arrival at Okinawa as 23:00 G.m.t. on March 17 to Kunia. Condition of flight crew reported as good by doctor onboard U.S.S. Mason.
March 18 00:10	31:29	U.S.S. Mason arrived at Okinawa to offload spacecraft and flight crew.

The time delay from spacecraft sighting (10 hr 39 min g.e.t.) to the first report of the flight crew's condition (11 hr 45 min g.e.t.) was caused by three factors:

(a) There was a lack of communication between recovery forces and the flight crew on the spacecraft voice frequency (296.8 mc). It is believed this problem resulted because the one radio onboard the aircraft tunable to the spacecraft frequency was also being used to communicate with the pararescuemen on a different frequency.

(b) The pararescuemen in this landing area were not equipped with the swimmer/spacecraft interphone.

(c) There is an inherent communications delay of the voice relay link among the flight crew, pararescuemen, Naha Rescue 1, U.S.S. Mason, Kunia Control Center, and the Mission Control Center — Houston.

6.3.3 Recovery Aids

6.3.3.1 UHF recovery beacon (243.0 mc).- Signals from the spacecraft recovery beacon were received by the following aircraft.

Aircraft	Initial time of contact, G.m.t.	Altitude, ft	Range, n. mi.	Receiver	Mode
Rescue 1 (HC-54)	03:19	9 000	3	SPP	CW
Rescue 3 (HC-130H)	04:25	20 000	136	ARD-17	CW
Search 1 (HU-16)	03:23	7 000	100	ITT	CW Pulse

Rescue 1 was approximately 3 nautical miles from the spacecraft during descent on the main parachute.

6.3.3.2 HF Transmitter (15.016 mc).- Signals from the spacecraft HF transmitter were received by thirteen stations of the DOD HF/DF networks. Three reports included the azimuths to the spacecraft and a computed spacecraft position. These three reports also included a possible radius of error. The reports were as follows:

Time of fix, G.m.t., March 17, 1966	Reported position of spacecraft, degrees and minutes	Radius of error, n. mi.
03:26	25-00N 135-30E	120
03:35	25-24N 136-00E	20
03:42	25-23N 135-56E	19

No recovery forces reported HF flight-crew voice reception.

The HF antenna was retracted prior to shipboard retrieval.

6.3.3.3 UHF voice transmitter (296.8 mc).- The recovery ship U.S.S. Mason reported a weak, unintelligible signal on 296.8 mc and this was the only report of UHF voice reception by the recovery forces.

6.3.3.4 UHF survival radio (243.0 mc).- The UHF survival radio was not used.

6.3.3.5 Flashing light.- The flashing light erected properly but was not activated by the flight crew. At landing, the door that covers the light was still connected at the hinge but did not impede light erection.

6.3.3.6 Fluorescein sea marker.- The sea dye marker diffusion was normal and was sighted at a range of 2650 yards by the recovery ship. It was sighted at a range of 3 to 10 nautical miles by five of the recovery aircraft. The spacecraft was still releasing dye at spacecraft pickup time, approximately 3 hours after landing.

6.3.3.7 Swimmer interphone.- The pararescuemen deployed to the spacecraft were not carrying the interphone so this system was not used.

6.3.4 Postretrieval Procedures

The spacecraft was powered down and the pyrotechnics were safed by the flight crew prior to retrieval. The flight crew egressed from the spacecraft and boarded the retrieval ship by means of a Jacob's ladder. The spacecraft was retrieved with the ship's davit crane and placed on the spacecraft cradle. Due to the rocking motion of the ship, the davit crane hold-off ring was left on the spacecraft for additional stability; consequently, the hatches were left closed until the recovery ship reached Okinawa.

Observations of the spacecraft at retrieval were as follows:

(a) The HF antenna was retracted. Recovery and UHF descent antennas were normal (erected).

(b) The flashing light and recovery loop were erected. The light was not flashing.

(c) Both windows were fogged.

(d) The RCS shingle heating effect appeared normal.

(e) The main-parachute riser hold-off ring was slightly damaged during retrieval.

(f) The main-parachute riser was not fully released from the forward bridle disconnect.

(g) The interior of the spacecraft was clean, neat, and dry. A slight burning smell was noticed around the spacecraft.

The Rendezvous and Recovery (R and R) Section was recovered with the drogue and pilot parachutes still attached. The R and R Section appeared to be in good condition (fig. 6.3-8).

The onboard films and voice tapes were removed by the flight crew and hand-carried to Cape Kennedy for postflight debriefings.

On March 18, 1966, the flight crew departed the destroyer, U.S.S. Leonard F. Mason, at Naha Port, went by helicopter to Kadena Air Base, and boarded a plane for Cape Kennedy.

The spacecraft was off-loaded at Okinawa and taken by truck to Naha Air Facility where deactivation procedures were begun.

6.3.5 Spacecraft 8 Reentry Control System Deactivation

A portion of the spacecraft postretrieval procedure was the deactivation of the Reentry Control System (RCS) at Naha Air Base, Naha, Okinawa. The primary reason for deactivation of the RCS at Naha was to safe the system prior to transporting the spacecraft aboard a USAF C-130 to the spacecraft contractor's facility in St. Louis, Missouri.

In order that the RCS be as free from hypergolic propellants as possible, rings A and B of the RCS were completely flushed with Freon-MF and methyl alcohol. Freon-MF was used in the oxidizer system and methyl alcohol in the fuel system; in addition, a nitrogen gas purge was used in both systems. This brought the system propellant parts-per-million (ppm) count to less than 25.

Following delivery of the spacecraft to St. Louis, the RCS was vacuum dryed in an altitude chamber and a postflight analysis was conducted.

The landing safing team (LST) consisted of NASA and spacecraft contractor engineers and technicians. This team was responsible for deactivating the RCS according to the procedures of reference 17.

When the LST arrived at Naha Air Base on March 19, 1966, the spacecraft had already been unloaded from the destroyer U.S.S. Leonard F. Mason. Preliminary examination of the spacecraft revealed that one shingle covering the RCS was broken during pick-up; however, the plumbing of the RCS was intact. The remaining shingles from around the RCS were removed, the cylindrical section was flushed with water, and all arrangements were made to begin actual deactivation procedures the following morning, March 20, 1966. Throughout the operation normal safety procedures were observed, and there was no visual indication of toxic vapors from any of the 16 RCS thrust chamber assemblies.

Before the pressurant in each ring was relieved to atmospheric pressure, the LST obtained pressure readings of source pressure from test point 1 on the A-package of both rings and of regulated lock-up pressure from test point 6 on the B-package of both rings. A 1/4-inch-inside-diameter flexible hose, 4 feet in length, from test point 1 to a calibrated 300 psi precision pressure gage was used for this operation. Source pressure readings of 1070 psig (ambient dry bulb temperature of 68° F) were obtained from both the A-ring and B-ring. A regulator lock-up pressure reading of 300 psig was obtained from both the A-ring and the B-ring. The pressure in each ring was then relieved to atmospheric pressure. Immediately following the source pressurant draining operation, the pressurant upstream of the propellant bladders and downstream of the system B-package check valves was relieved through test points 4 and 6 by venting through separate propellant scrubber units.

At no time prior to the flushing operation did a propellant solenoid valve leak vapors which would have indicated that the valve was partially stuck open. All the RCS valves appeared to function normally. No problems were encountered during the deactivation of the spacecraft.

In accordance with reference 17, any system propellants remaining after flight were to be collected for analysis. Flush-fluid samples and nitrogen-purge gas samples from each ring were also to be collected for analysis. Insufficient samples of fuel and oxidizer were obtained for analysis.

TABLE 6.3-I.- RECOVERY SUPPORT

Landing area	Access time, hr:min		Support
	Aircraft	Ship	
Launch site area:			
Pad	00:05		4 LARC (amphibious vehicle) 1 LCU (large landing craft) with spacecraft retrieval capabilities
Land	00:10		2 LVTR (amphibious vehicle) with spacecraft retrieval capabilities
Water (if flight crew ejects)	00:02		3 M-113 (tracked land vehicles)
Water (if flight crew is in spacecraft)	00:15		4 CH-3C (helicopters) (3 with rescue teams) 1 MSO (mine sweepers) with salvage capabilities 1 boat (50 ft) with water salvage team
Launch abort:			
A_1	4:00	12:00	1 LPH (aircraft carrier) with onboard helicopter capabilities, 4 DD (destroyers), 1 AO (oiler), and 6 aircraft on station (3 HC-97 and 3 HC-130)
A_2	4:00	35:00	
B	4:00	2:00	
C	4:00	15:00	
D	4:00	24:00	
Primary:			
West Atlantic	1:00	4:00	1 LPH (aircraft carrier) from area A, station 3 3 HC-130H (search and rescue) 5 JC-130 (3 telemetry and 2 communications relay) 6 SH-3A helicopters (3 location, 2 swimmer, and 1 photo) 3 P3-A (on-scene commander)
Secondary landing areas:			
West Atlantic (Zone 1)	30-min strip alert	6:00	1 LPH (carrier) from station 3
East Atlantic (Zone 2)		6:00	1 DD (destroyer) 1 AO (oiler)
West Pacific (Zone 3)		6:00	2 DD (destroyers) (rotating on station)
Mid-Pacific (Zone 4)		5:00	1 DD (destroyer)[a]
Contingency			29 aircraft on strip alert at staging bases
Total			11 ships, 10 helicopters, 39 aircraft

[a]In addition, an oiler (AO) was assigned to the area for logistic purposes.

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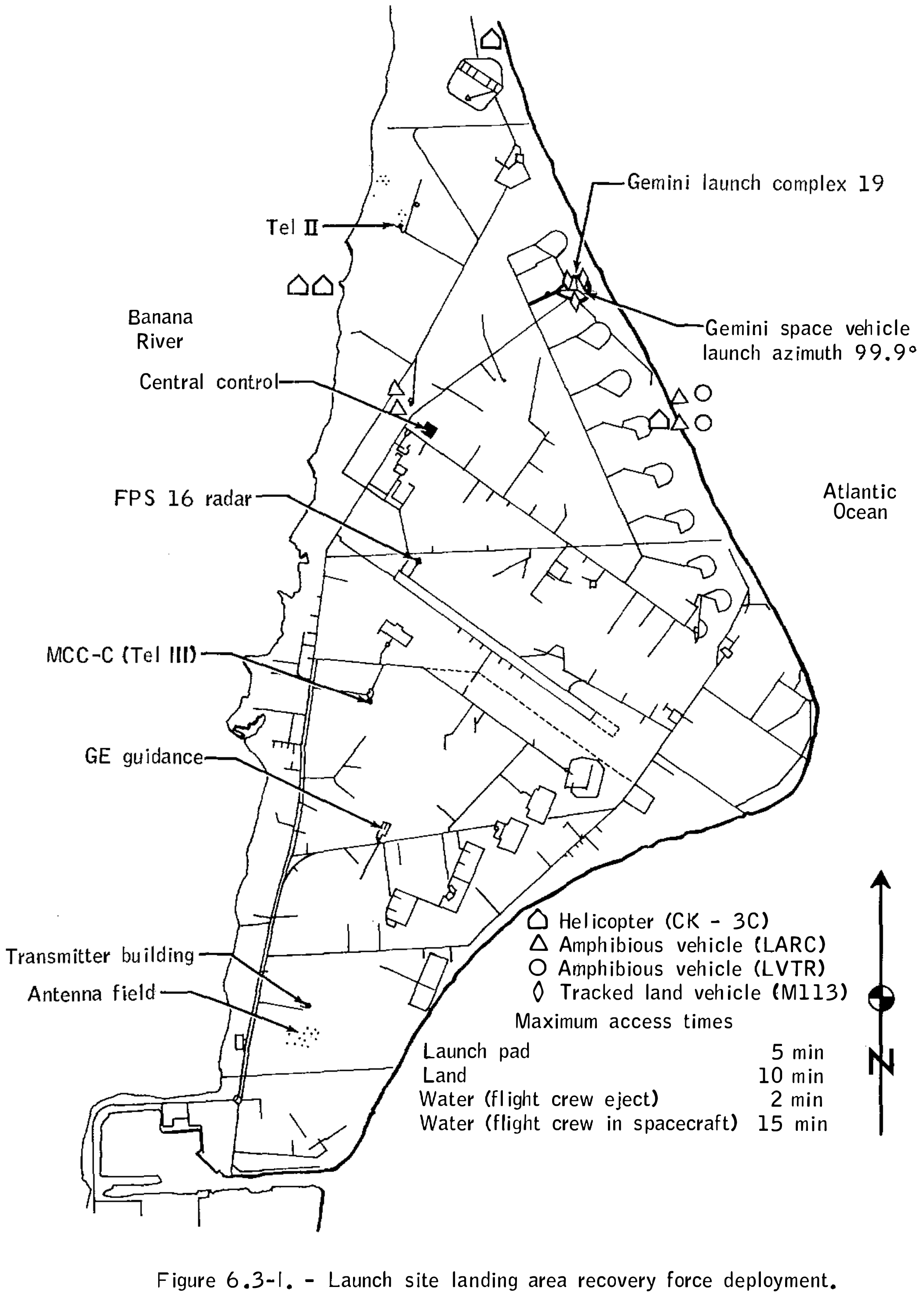

Figure 6.3-1. - Launch site landing area recovery force deployment.

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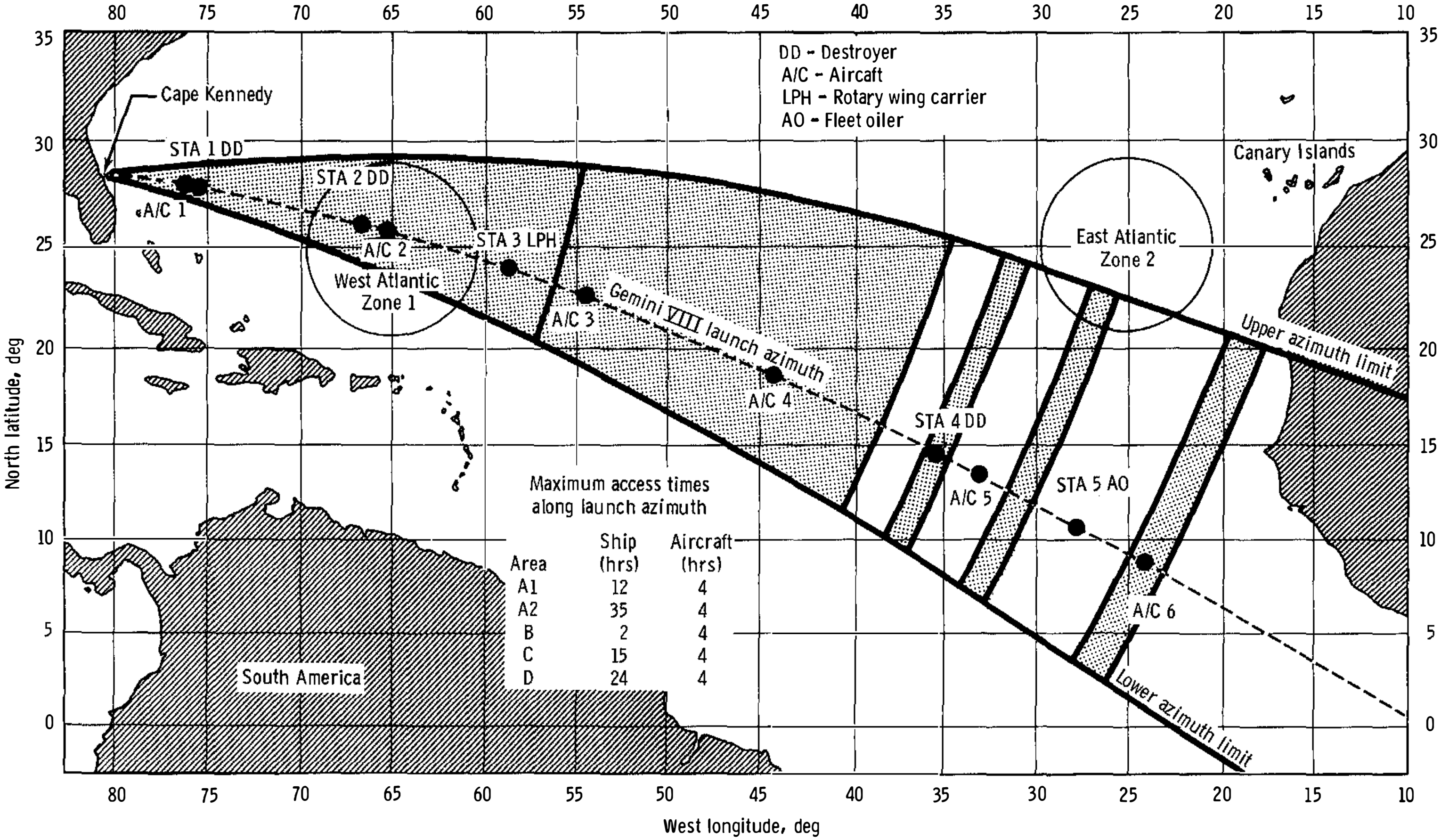

Figure 6.3-2. - Gemini VIII launch abort areas and recovery ship and aircraft deployment.

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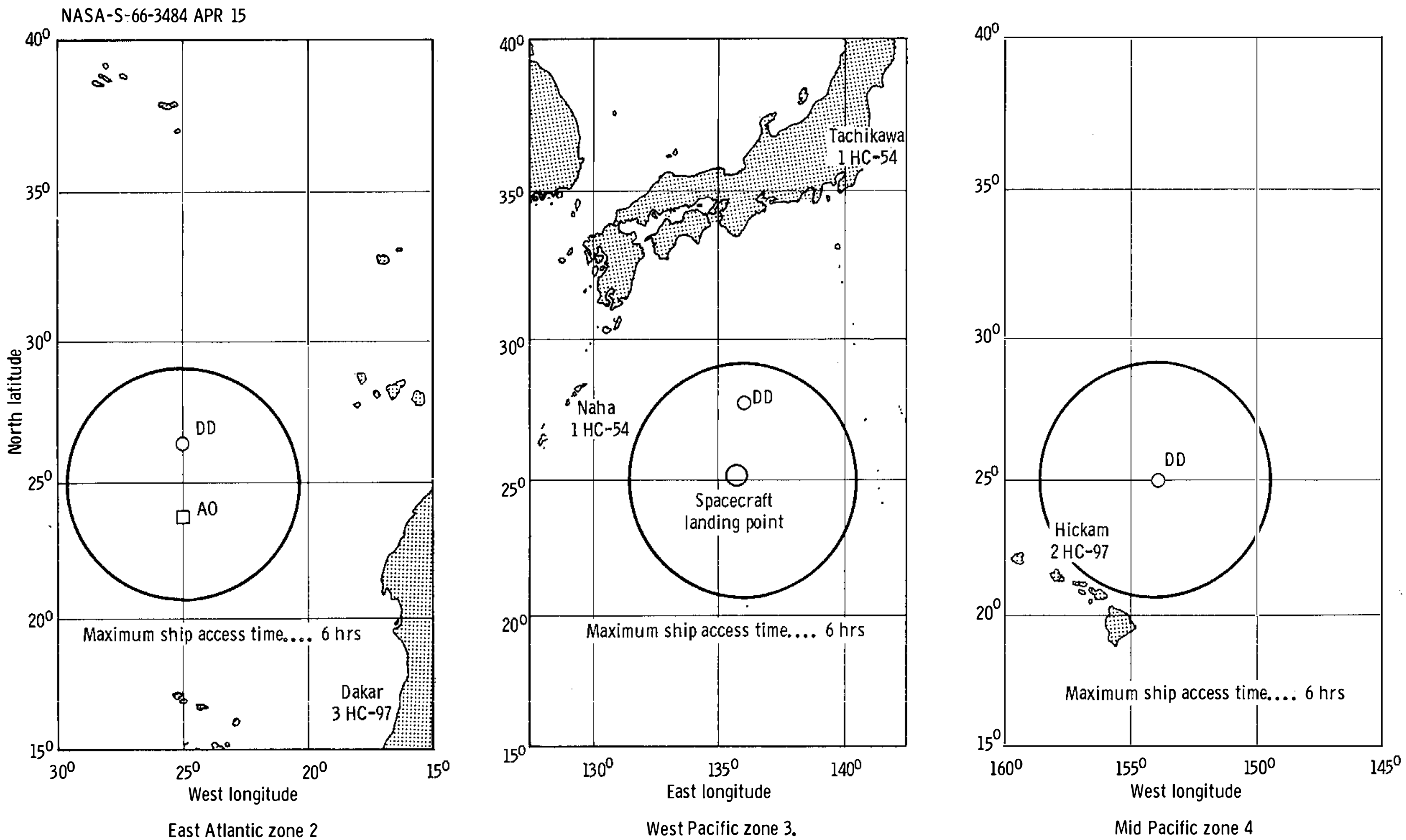

When it became apparent a West Pacific landing would occur the following additional aircraft were released from normal search and rescue duty to aid in locating the spacecraft.

Base	Aircraft
Tachikawa	1 HC-130H
	1 HC-54
Naha	3 HO-16

Figure 6.3-3.- Gemini VIII landing zone force deployment.

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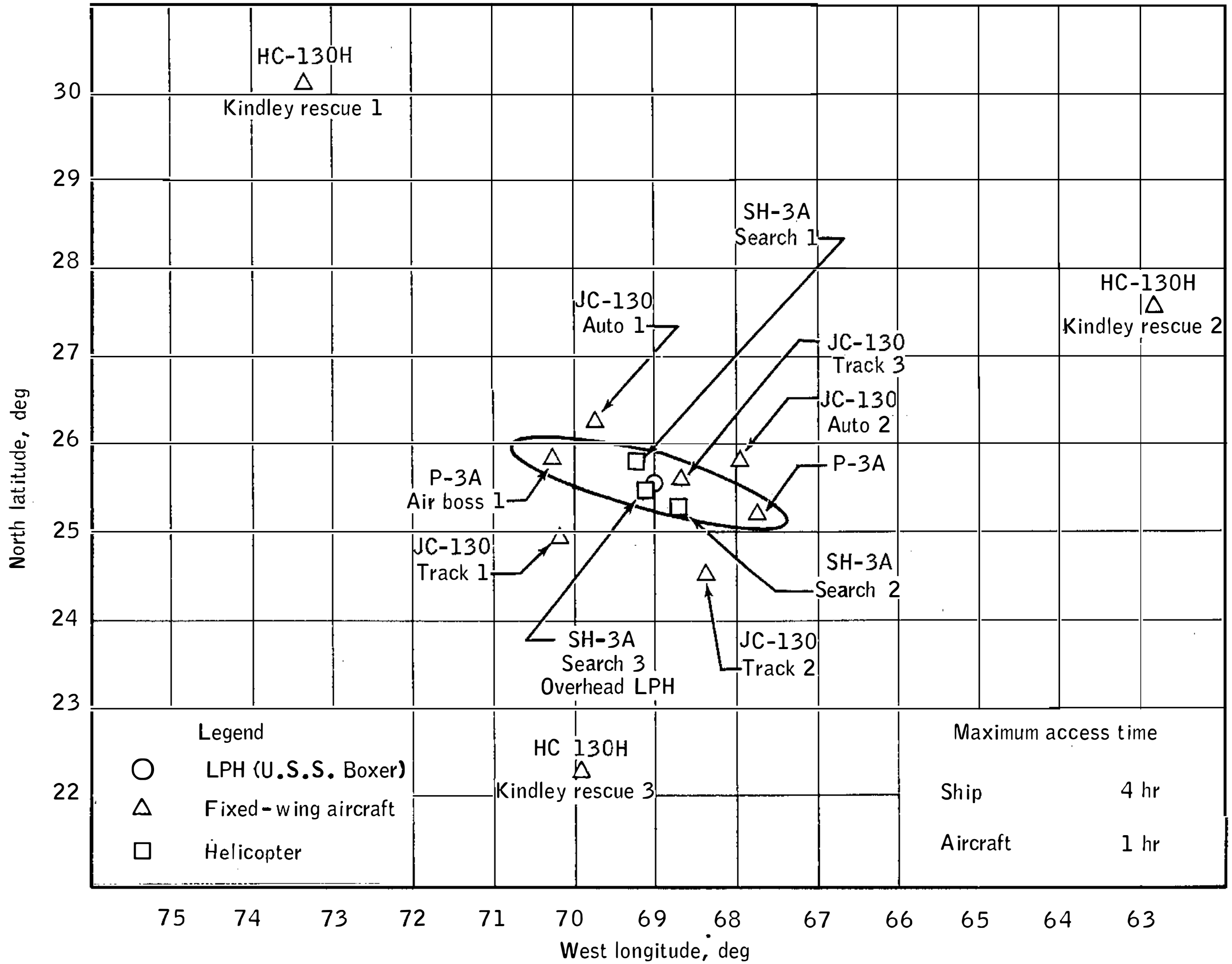

Figure 6.3-4 Primary landing area recovery force deployment.

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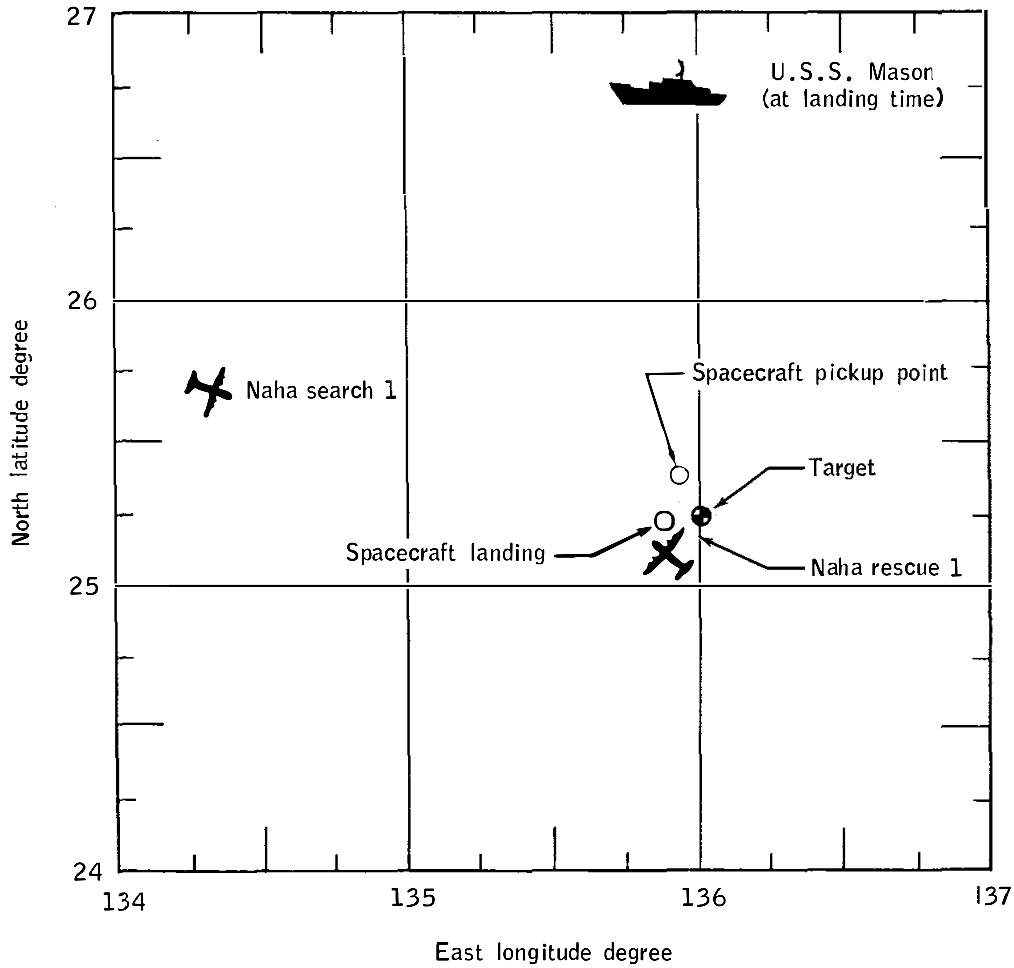

Figure 6.3-7. - Spacecraft landing area information.

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Figure 6.3-8. - R and R Section.

7.0 FLIGHT CREW

7.1 FLIGHT CREW PERFORMANCE

7.1.1 Crew Activities

The flight crew accomplished a well-executed, closed-loop rendezvous with the target vehicle, and, after a short period of station keeping, they successfully accomplished the docking of their spacecraft with the Gemini Agena Target Vehicle (GATV) during the fourth revolution (M=4). This accomplishment met the primary and secondary objectives of the mission in relation to the rendezvous and docking phase. Station keeping, performed prior to the docking phase, and the docking task appeared easy and were less difficult than had been experienced during training simulations. The early termination of the mission, because of a spacecraft control-system malfunction, prevented accomplishment of the scheduled experiments and extravehicular activities (EVA). The flight plan activities which were accomplished are shown in figure 7.1-1, Summary Flight Plan.

7.1.1.1 Prelaunch through insertion.- After the crew entered the spacecraft, adequate time was available to complete all required prelaunch functions. Launch-vehicle engine ignition was smooth and lift-off was very apparent to the crew. Crew performance during powered flight was good and all required cockpit activities and confirmation of events were accomplished accurately and on time. After second-stage engine cutoff (SECO), computer readouts were conducted by the pilot and the separation maneuver was started on time. Because the pilot did not hear the spacecraft-separate MARK given by the command pilot, there was a short delay between the start of the thruster firing and spacecraft separation; however, separation from the Gemini Launch Vehicle (GLV) was clean. Shortly afterward, the crew received an insertion GO from the ground. After the normal debris from the spacecraft—GLV separation had cleared, the nose and horizon-scanner fairings were jettisoned, and this imparted an unexpected moment to the spacecraft. The insertion checklist was then completed and all systems were found to be in a normal condition.

7.1.1.2 Rendezvous.- The rendezvous activities consisted primarily of the following:

(a) A series of translation maneuvers to obtain the desired relative position and velocity from which the spacecraft guidance system could compute the remaining maneuvers for transferring to a rendezvous course with the target vehicle

(b) Terminal rendezvous maneuvers, including monitoring the computer solutions for the terminal phase initiate (TPI), selecting the proper time for TPI, applying the TPI maneuver, executing mid-course corrections, controlling the line-of-sight drift, and braking

(c) Station keeping with the GATV

(d) Docking.

Each of these major phases is discussed separately below.

7.1.1.2.1 Translation maneuvers: As in the Gemini VI-A mission, there were five mid-course orbit-adjust maneuvers:

(a) Height adjust

(b) Phase adjust

(c) Plane adjust

(d) Vernier height adjust

(e) Circularization.

The height adjust was a horizontal, in-plane maneuver applied at first perigee after insertion to correct the apogee to 146 nautical miles. It was a retrograde maneuver of 2.9 ft/sec applied at 1:34:37 ground elapsed time (g.e.t.) with the spacecraft at 0,0,0-degree attitude. The forward-firing thrusters were used for a thrust time of approximately 5 seconds. The attitude control was in platform mode. The platform was aligned and switched to ORB RATE prior to the maneuver. After completion of initial thrusting, the crew experienced some difficulty in nulling the residual desired-velocity changes. It was noted that the accelerometer data would vary approximately 0.2 ft/sec between readings taken 4 seconds apart and with no applied maneuver thrust. The maneuver was accomplished accurately, on time, and with nominal fuel consumption.

The phase adjust was a horizontal in-plane maneuver performed at the second apogee to raise perigee so that the spacecraft would reach TPI at the correct time. The platform was aligned before the maneuver using pulse attitude-control mode to minimize the alignment time. The maneuver was initiated at 2:18:25 g.e.t. at 0,0,0-degree attitude with rate-command attitude control and aft-firing thrusters for a period of 1 minute 8 seconds. The velocity change was 50.6 ft/sec. The crew again encountered some difficulty in reducing the residuals to 0.2 ft/sec, as in the height-adjust maneuver. However, the maneuver was well executed and the desired results were obtained.

The plane-change maneuver was initiated at 2:45:50 g.e.t., 25 minutes before the end of the second revolution, and resulted in a horizontal velocity change of 26.2 ft/sec, 90 degrees to the right of the orbit path. The platform was aligned for about 15 minutes after which the maneuver was performed with the aft-firing thrusters and with the control system in the rate-command control mode.

Near the second perigee, a 2 ft/sec posigrade, vernier height-adjust maneuver was requested by the ground. The information was received shortly before the requested time of the maneuver and there was no time for platform alignment or pointing-command inputs to the computer. The maneuver was performed with a 3-second thrust from the aft-firing thrusters, starting at 3:03:41 g.e.t. in 0,0,0-degree attitude and in the rate-command control mode. Residuals could not be nulled because the maneuver was performed without the aid of the computer.

Shortly after the vernier height adjustment, a solid radar lock-on was established at a range of 180 nautical miles. At this point the computer was switched to the rendezvous mode for the rendezvous test. This test exercised the closed-loop mode of the guidance system by collecting samples of radar and platform data and displaying the two-impulse rendezvous velocity requirements on the Incremental Velocity Indicator (IVI). The crew then compared these data with the nominal values on charts to verify the performance of the guidance system prior to the actual rendezvous maneuvers. Based on eight data points, the results of the rendezvous test indicated satisfactory performance of the closed-loop computer mode.

Because of the range at which radar lock-on occurred, the rendezvous test was completed just before the circularization maneuver. In fact, the circularization maneuver was applied 36 seconds late, at 3:48:11 g.e.t. Although the effect of the delay on the mission was insignificant, the crew recommended that the rendezvous test not be performed on subsequent missions, as the same data are obtained after the circularization maneuver. The in-plane circularization maneuver was performed at a pitch attitude of 21 degrees down for 1 minute 22 seconds, and resulted in a velocity change of 61.2 ft/sec. The rate-command control mode was used. This maneuver was also well executed and placed the spacecraft in the proper orbit and phase in relation to the target vehicle for the terminal phase of rendezvous.

7.1.1.2.2 Terminal-phase maneuver: The computer was switched from CATCHUP to RENDEZVOUS 5 minutes 40 seconds after the circularization maneuver. The range, range rate, and pitch angle were recorded every 100 seconds as planned. Range and angle were plotted on the onboard polar graph and it was observed by the pilot that the difference in altitude between the spacecraft and GATV orbits was about 1.5 nautical miles less than nominal. Radar angular track appeared to be somewhat erratic, which probably introduced some scatter into the polar plot. When the target elevation angle reached 10 degrees, the platform was aligned for 13 minutes in the pulse control mode. After this alignment, the spacecraft was again controlled to permit radar boresight on the GATV, and the crew monitored the elevation angle in anticipation of reaching TPI.

The transfer maneuver was applied 1 hour 26 minutes 10 seconds after the circularization maneuver. The closed-loop solution was used for the transfer maneuver as well as for both mid-course corrections; however, backup solutions were also obtained.

The trend of the backup solutions generally agreed with the closed-loop solutions except for the up/down correction at transfer and gave the crew confidence that the closed-loop solutions were correct. The crew believed this discrepancy to be caused by the cyclic inaccuracy which occurred in the radar angle information at ranges between 45 and 25 nautical miles.

Translation inputs required to control the line-of-sight drift were relatively minor, except for the out-of-plane drift. Near the end of the braking maneuvers, a total of 18 ft/sec had been recorded in the left/right window of the IVI. The first reduction in closing rate was applied at a range of 1.7 nautical miles to reduce range rate from 44 to 36 ft/sec. Several additional braking maneuvers were applied until a stable station-keeping position was reached at a range of 150 feet. Crew performance throughout the rendezvous and braking maneuvers was very good from the standpoint of performing the maneuvers, computing backup solutions, and making the correct decision each time to continue with the closed-loop solution.

7.1.1.2.3 Station keeping and docking: Station keeping began at 5:56:56 g.e.t., or about 42 minutes after TPI, at a range of 150 feet. This range was soon closed to 50 feet. There were no difficulties with station keeping in any of the control modes and the crew was able to observe the GATV closely and feel confident of its suitability for docking. Station keeping was performed for a relatively short time because darkness was rapidly approaching and the crew desired to dock under daylight conditions. The crew commanded GATV flight control mode 6 for the actual docking, closed the distance to about 3 feet, held station at this range to perform final verification of the GATV status, and waited for telemetry confirmation from the Rose Knot Victor (RKV) network station. During station keeping and docking, the crew demonstrated good judgment and sound engineering pilot techniques.

7.1.1.3 Operational checks and experiments.- The scheduled operational checks from lift-off to 6 hours 50 minutes g.e.t. were completed according to the flight plan. The remaining operational checks, with the exception of the Auxiliary Tape Memory Unit (ATMU) exercise, were not performed because of the early termination of the flight.

Three experiments were initiated before the flight was terminated. Only two of the three produced useful results. Refer to section 8.0 for additional information on experiments.

7.1.1.3.1 Platform alignments: The platform alignments were accomplished using the Orbital Attitude and Maneuver System (OAMS) attitude control in the platform mode, except for the alignments before the phase-adjust maneuver, TPI, and the final alignment for retrofire. These alignments were done manually in the pulse mode because it was felt that a more accurate alignment would be obtained in the time available for the alignments.

7.1.1.3.2 General-purpose photography: The objectives of general-purpose photography were met until the early termination of the flight. The crew recorded the docking with the boresighted 16-mm camera; also, 70-mm photographs were taken at nearly-equal time intervals during the final rendezvous and docking phase. The photographic data content and the quality of the photographs were excellent.

7.1.1.3.3 Auxiliary Tape Memory Unit exercise: The planned ATMU exercise was not performed because of the termination of the flight. However, because the reentry math flow was in an ATMU module and had to be loaded before reentry, Module IV-A was loaded and verified automatically and was reverified with Module IV-B. The loading, together with the thruster firing, fulfilled most of the objectives of the ATMU exercise.

7.1.1.3.4 Experiment M-5, Bioassay of Body Fluids: The equipment for Experiment M-4 was not unstowed during the flight, and the urine-collection devices (UCD's) were collected from the flight crew by the medical officer on the recovery ship.

7.1.1.3.5 Experiment S-3, Frog Egg Growth: The two chambers of frog eggs on the right side of the spacecraft were fixed according to the flight plan, unit 1 at 00:40:10 g.e.t., and unit 2 at 2:25:07 g.e.t. The left unit no. 1 was "fixed" after landing at 13:02:50 g.e.t. Because the flight was terminated early, the experiment was only 50-percent completed.

7.1.1.3.6 Experiment S-9, Nuclear Emulsion: The experiment was activated at 23 minutes into the flight, and telemetry data were received which indicated that the experiment was performing as expected. The experiment package was mounted on the spacecraft adapter. Because the flight terminated before the scheduled EVA, the package could not be retrieved and consequently the desired information was lost.

7.1.1.4 Control systems.- Until approximately 7 hours g.e.t., all control systems operated as expected and the crew was able to exercise precise control of the spacecraft.

The pulse mode was adequate for station keeping, provided the maneuver thrusters were operated for only short periods. Platform mode required very little attention during station keeping and was considered a good control mode when other tasks required complete concentration. The crew selected RATE COMMAND for docking, docked, shut off the OAMS attitude-control power and the horizon scanner, and switched to PULSE in accordance with the post-docked checklist. The spacecraft–GATV combination was very stable after docking and after performing a 90-degree yaw maneuver.

At 7:00:26.7 g.e.t., with OAMS attitude control power off, OAMS thruster no. 8 fired continuously for 4.9 seconds, was inactive for 4 seconds, and then began thrusting again. A few seconds later, the pilot noticed a 5 deg/sec roll rate on the Flight Director Attitude Indicator at this time and also noted a roll attitude of about 30 degrees. He immediately informed the command pilot who took steps to gain control of the vehicle. Neither crewman felt any sensations of rolling or heard any thruster noise even though they had their helmets off. The spacecraft was in darkness during this period and had just experienced loss-of-signal (LOS) from Tananarive.

The pilot sent command ACS-OFF to turn off the GATV ACS and the crew did not notice any change in the situation. He also shut off the GATV horizon sensors and geocentric rate. The rates continued to increase and the crew activated the spacecraft control system to control them. The OAMS attitude-control power was turned on at 7:00:38 and the direct mode was selected about 2 seconds later. The roll rate had increased to approximately 15 deg/sec. The spacecraft–GATV combination was quickly stabilized but when the hand controller was released, the rates built up again in yaw and roll.

The command pilot nulled the rates several times and then switched the spacecraft to RATE COMMAND at 7:01:36.4 g.e.t. The roll rates of the combination were reduced to zero but telemetry later showed that OAMS thrusters 3, 4, 7, and 8 were firing continuously. However, there were no onboard indications of which thrusters were firing. The yaw thrusters were cancelling each other and telemetry later showed that the GATV was able to damp the rates while the spacecraft was in RATE COMMAND. The crew believed that they were controlling the rates at that time.

Pulse mode was selected at 7:01:50.6 g.e.t., and the rates built up in all three axes, but primarily in roll. Thruster pulses had no noticeable effect because of the short duration. The crew again selected RATE COMMAND at 7:01:58 g.e.t. and the pitch and yaw were nulled, but roll was only held at a constant rate with thrusters 3, 4, 7, and 8 on.

Thruster 8 apparently stopped firing at 7:02:37.4 g.e.t. and the roll rate was stopped immediately. The docked vehicle combination remained essentially stable for approximately 5 minutes and during that time the crew attempted to determine the trouble. At 7:02:54.6 g.e.t. the crew selected the direct mode and slight yaw and roll rates developed, possibly caused by oxidizer bleeding from thruster 8. The crew effectively damped the rates in direct mode and maintained control when they switched to PULSE at 7:05:25.2 g.e.t. The direct mode was again selected and control was maintained, with thruster 8 apparently not firing during this period.

At 7:07:20.3 g.e.t., thruster 8 again began firing, producing rates primarily in the roll and yaw axes; however, the crew was able to maintain the rates at relatively low levels. About 10 seconds later, the crew sent ACS-OFF with no apparent change to the rates and were unable to determine the cause of the divergence. Sometime later, the ACS was cycled back on and then turned off at 7:12:38.6 g.e.t. but again there was no change and no clue to the cause of the control problem. At this time the crew seriously suspected that the problem was in the spacecraft, even though the unexpected rates had first occurred with OAMS power off. The crew cycled the Attitude Control and Maneuver Electronics (ACME) bias power off and on rapidly at 7:13:38.6 g.e.t. with no apparent result. The propellant motor valves were shut off, and when there was still no apparent effect, they were returned to ON. Attitude driver logic was also switched and the crew believes that they switched the roll logic to the pitch thrusters; however, there were no indications of pitch-thruster activity in direct combination with any roll commands. None of these actions had any effect and the crew decided to separate from the GATV in order to isolate the problem to one vehicle or the other. The rates were damped to what the crew determined to be a safe level, and a normal undocking was accomplished at 7:15:12.3 g.e.t. The rates just prior to separation were 3 deg/sec in pitch, 5 deg/sec in roll, and 2 deg/sec in yaw. However, the roll rate rapidly diverged to 30 deg/sec by 17 seconds after separation and the crew switched the ACME to RATE COMMAND, with some reduction in roll rate.

At 7:15:44.7 g.e.t. the telemetry indicates that ACME bias power was interrupted and that a rapid increase in spacecraft roll rate began with thruster 8 the only thruster firing. (After the flight, the crew did not specifically report that they had turned ACME bias power off during this period and did not recall doing so when questioned. They stated that some switches on the overhead circuit-breaker panel were found in the OFF position after the spacecraft had been stabilized, and had to be reset to get control with ACME.)

After this, the rates were reaching an uncomfortable level with no apparent means for the crew to gain control of the situation and they were also beginning to feel the onset of vertigo; consequently, at 7:16:25.1 g.e.t. they activated the Reentry Control System (RCS). Less than 2-minutes later, the OAMS circuit breakers were opened, and this stopped thruster 8 from thrusting. On first activation of the RCS, there was no response due to the ACME bias power being off. Less than 1 minute after the OAMS was deactivated, the crew switched to DIRECT-DIRECT and started reducing rates with both RCS rings. About 30 seconds later, the A-ring was turned off because the rates were being reduced and the spacecraft was under control. About 6 minutes later, the spacecraft rates were reduced to zero and the crew started to control the spacecraft in pulse mode.

The spacecraft remained stable and, starting at 7:28:12.5 g.e.t., the crew checked the thrusters one at a time. Approximately 14 seconds later, the thruster 8 circuit breaker was closed momentarily and thrust resulted. Having isolated the malfunction, the crew utilized the OAMS to control and align the spacecraft for retrofire, using the remaining OAMS attitude thrusters and conserving the RCS propellant for reentry.

The OAMS thrust output seemed degraded for a short period following the OAMS power-up. However, the crew commented that the thrust output improved with time and was adequate for attitude control and orientation for retrofire. The RCS had approximately 32 pounds of propellant remaining and this was sufficient for the crew to maintain control in the pulse and reentry rate-command modes through drogue parachute deployment.

7.1.1.5 Retrofire and reentry.- Shortly after the flight crew regained control of the spacecraft, a decision was made to reenter in area 6-3 or 7-3. Later, it was decided to reenter in area 7-3 to permit ample crew preparation time for stowage and for completion of preretrofire requirements. Immediately after the decision to reenter, stowage was initiated and proceeded quite smoothly, with the exception of the difficulties in stowing the Experiment D-15 television monitor and the EVA visor and in closing the centerline stowage container. The platform was aligned using the OAMS with thruster 8 inoperative.

The crew was not given notification in sufficient time to prepare for the first Digital Command System (DCS) update for retrofire because the ground personnel desired to get the load sent and verified prior to the imminent loss-of-signal at the Rose Knot Victor. Premission planning had established a mission rule that the crew would be notified in sufficient time to place the computer in PRELAUNCH and inform the ground personnel that the computer was ready for the update. Because this procedure was not followed, the crew did not know whether or not the computer was in the prelaunch mode at the time of the update. After the update, the pilot commanded the computer to display the time-to-go to retrofire and found that it was counting up instead of down. The crew then thought that the counting up was being displayed because the computer may not have been in the correct mode to receive the update. (Subsequent postflight analysis revealed it was in prelaunch mode. See section 5.1.5 for an explanation of the time-to-go to retrofire display.)

A second update was sent from the Coastal Sentry Quebec, and the crew checked the Manual Data Insertion Unit (MDIU) quantities and determined that they had a good update in the computer and the Time Reference System (TRS) for reentry. The crew had previously inserted and verified Module IV-A and IV-B into the ATMU and had found it necessary to cycle the computer on and off during this operation.

The indicate-retroattitude sequence light illuminated only after the telelight switch was depressed. Time-of-retrofire (T_R) - 1 minute events were reported by the crew to be nominal, and they also heard the ground countdown from T_R - 10 to T_R - 2 seconds before loss-of-signal from the Kano network station. The retrorockets fired automatically and exactly on time, with the pilot backing up this event by depressing the manual retrofire switch at T_R + 1 second. The spacecraft attitude was held very close to nominal during the retrofire maneuver by utilizing both RCS rings in the rate-command control mode. Spacecraft attitude was maintained with reference to the Flight Director Indicator (FDI) because retrofire occurred on the night side.

At completion of retrofire the crew read out velocity changes on the IVI to the ground as 292 aft, zero left/right, and 114 down, which were very close to nominal. The computer readout verified the IVI velocities. The crew crosschecked these velocities with the required reentry bank angle from the onboard bank-angle charts, which indicated a reentry bank angle of 52-degrees left for the resultant conditions after the retrofire maneuver.

The retropackage and docking bar were jettisoned at the proper time. The spacecraft was positioned to the proper reentry attitude

(180 degrees) using the RCS B-ring in the pulse control mode. After entering the day side, the proper pitch attitude was maintained by using the horizon as a reference. At approximately 400K feet, the spacecraft was rolled to 52-degrees left bank. Guidance initiate occurred on time at approximately 290K feet. The downrange-error indication deflected to 90 miles and held this position. The R_N minus R_P (miles down range to zero-lift initiate) was predicted by the ground to be 77 miles. A cross check of the onboard charts with the downrange error indications in the FDI verified that they coincided within 50 miles, which was evidence that the computer guidance was providing proper steering information; therefore, the crew elected to fly a closed-loop reentry.

During the initial reentry maneuvering, the spacecraft was flown with sufficient roll to null the downrange and crossrange error indicators. Upon nulling the indicators, a 15-degree roll rate was commanded and maintained until after peak reentry acceleration. The pulse mode of control was utilized as long as possible during the reentry to conserve fuel. As the acceleration began to increase significantly, the reentry rate-command mode was selected to provide adequate control over the spacecraft. Just prior to the peak acceleration, the command pilot switched from the RCS B-ring to the A-ring to conserve the remaining B-ring propellant for controlling the spacecraft during the critical period between drogue parachute deployment and disreefing. Some fuel from both rings was still available at drogue deployment.

The crew considered the reentry rate-command control mode satisfactory for flying the reentry. The use of this mode, rather than the rate-command mode, provided significant fuel savings. The crew was somewhat concerned over the fact that a full 15-deg/sec roll rate could not be achieved and that they could not completely null the indicated roll error.

Communications with the ground were lost at retrofire; however, the crew was quite confident concerning the landing area due to the close coincidence of computer guidance steering and the onboard backup reentry charts. At the termination of guidance the crew read out the landing coordinates of the point as 25.05-degrees north latitude and 136.09-degrees east longitude, which was very close to nominal. Crew comments concerning visual observations (retropackage, ion sheath, window coating) during reentry were very similar to those reported by previous crews.

7.1.1.6 Landing and recovery.- The drogue parachute was deployed at 50K feet with some increase in oscillations (±20 degrees) prior to disreef. The remaining propellant was expended at this time, using the rate-command control mode. Main parachute deployment and two-point suspension were nominal; however, the crew reported the landing shock to be considerably more severe than expected.

Postlanding communications consisted of one period of radio communication with the rescue aircraft approximately 30 minutes after landing. Shortly thereafter, the crew observed the pararescuemen descend into the water; however, flotation-collar attachment took an unusually long time because of the heavy sea state. The crew completed their postflight checks without difficulty but were quite uncomfortable due to the sea condition. Subsequent to attachment of the flotation collar, the hatches were opened and the crew became more comfortable as they awaited pickup by the destroyer, the U.S.S. Leonard F. Mason. Approximately 3 hours after landing, the U.S.S. Mason came along side and attached a line to the spacecraft. The crew egressed from the left hatch with some difficulty due to the fairly severe bobbing caused by swells of 12 to 15 feet. The main parachute, which had been attached to the spacecraft, was lost during this operation.

7.1.1.7 Mission training and training evaluation.- Flight-crew training was accomplished as outlined in the Mission Training Plan. The command pilot, in addition, had completed extensive training as a result of his participation as backup command pilot for the Gemini V mission. Table 7.1-1 contains a summary of the crew training for the Gemini VIII mission.

The mission, due to the inclusion of rendezvous, re-rendezvous, docking, and extravehicular activities, together with flight experiments, required that the flight crew participate in a wide variety of training activities and simulations in preparation for the flight. The crew was required to complete a very intensive and demanding work schedule to meet the anticipated launch date.

The Rendezvous Simulator and the Gemini Mission Simulator were utilized for crew rendezvous training and procedures development. Docking practice was accomplished on the Translation and Docking Trainer with additional docking and GATV flight-plan maneuvers being accomplished on the Gemini Mission Simulator during the final phase of training at Cape Kennedy. The early availability of an operational visual display for subsequent crews will greatly increase the training value of the Gemini Mission Simulator for this type of mission.

The performance of the crew during the mission indicated that they had been well trained in the accomplishment of the mission objectives. Crew reaction and performance during and after the control-system malfunction indicated that they were able to recovery from an emergency situation and function satisfactorily and accurately during the terminal phase of the flight.

TABLE 7.1-I.- CREW TRAINING SUMMARY

Activity	Training time, hr	
	Command pilot	Pilot
Gemini systems briefing	74	79
Operational briefings	79	78
Gemini Mission Simulator	125	123
Dynamic Crew Procedures Trainer	10	5
Translation and Docking Trainer	16	21
Rendezvous simulation	51	51
Extravehicular-activities training	44	84
Egress training	6	10
Planetarium	11	18
Spacecraft Systems Tests (SST)	76	84

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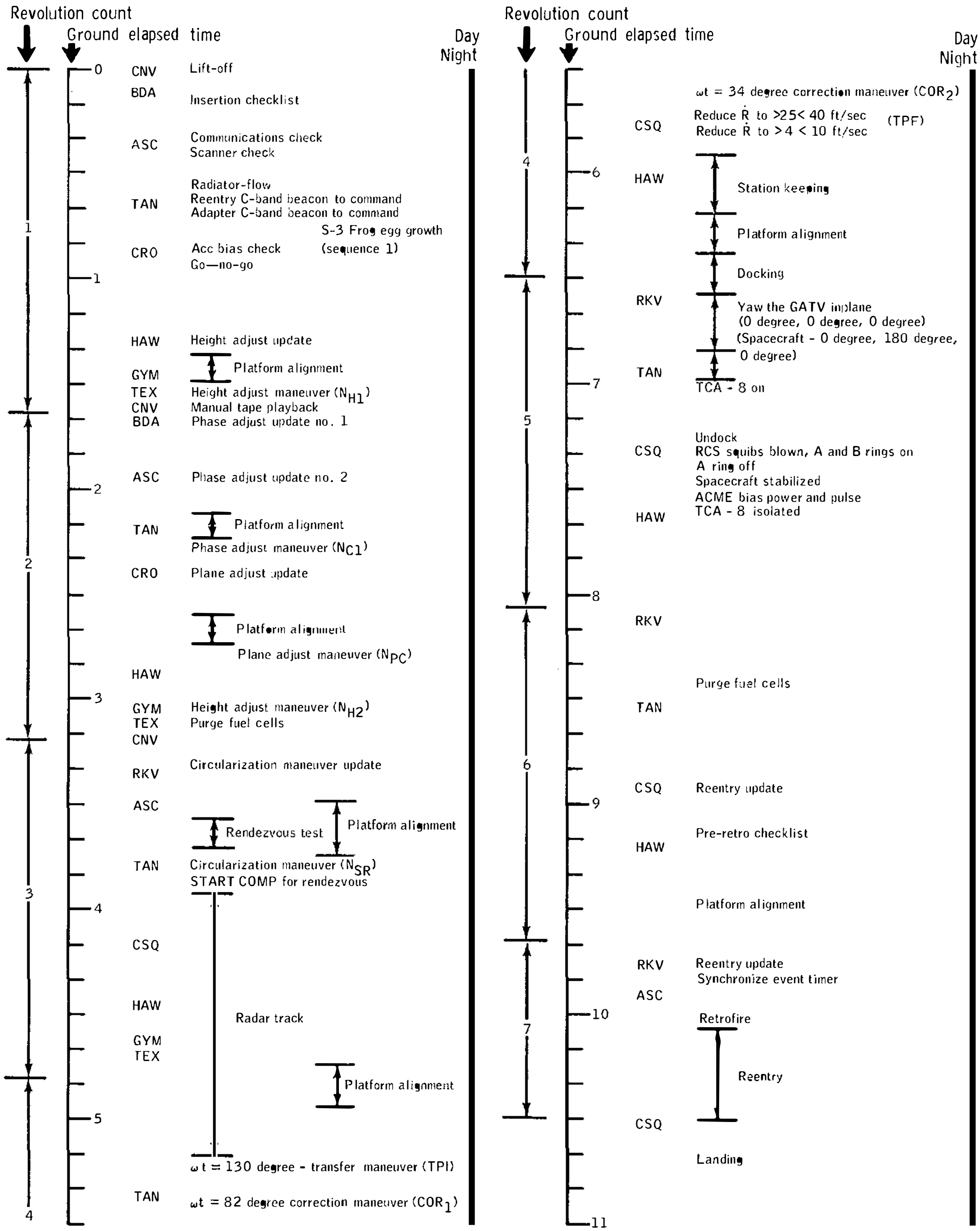

0 to 11 Hours g. e. t.

Figure 7. 1. 1-1. - Summary flight plan.

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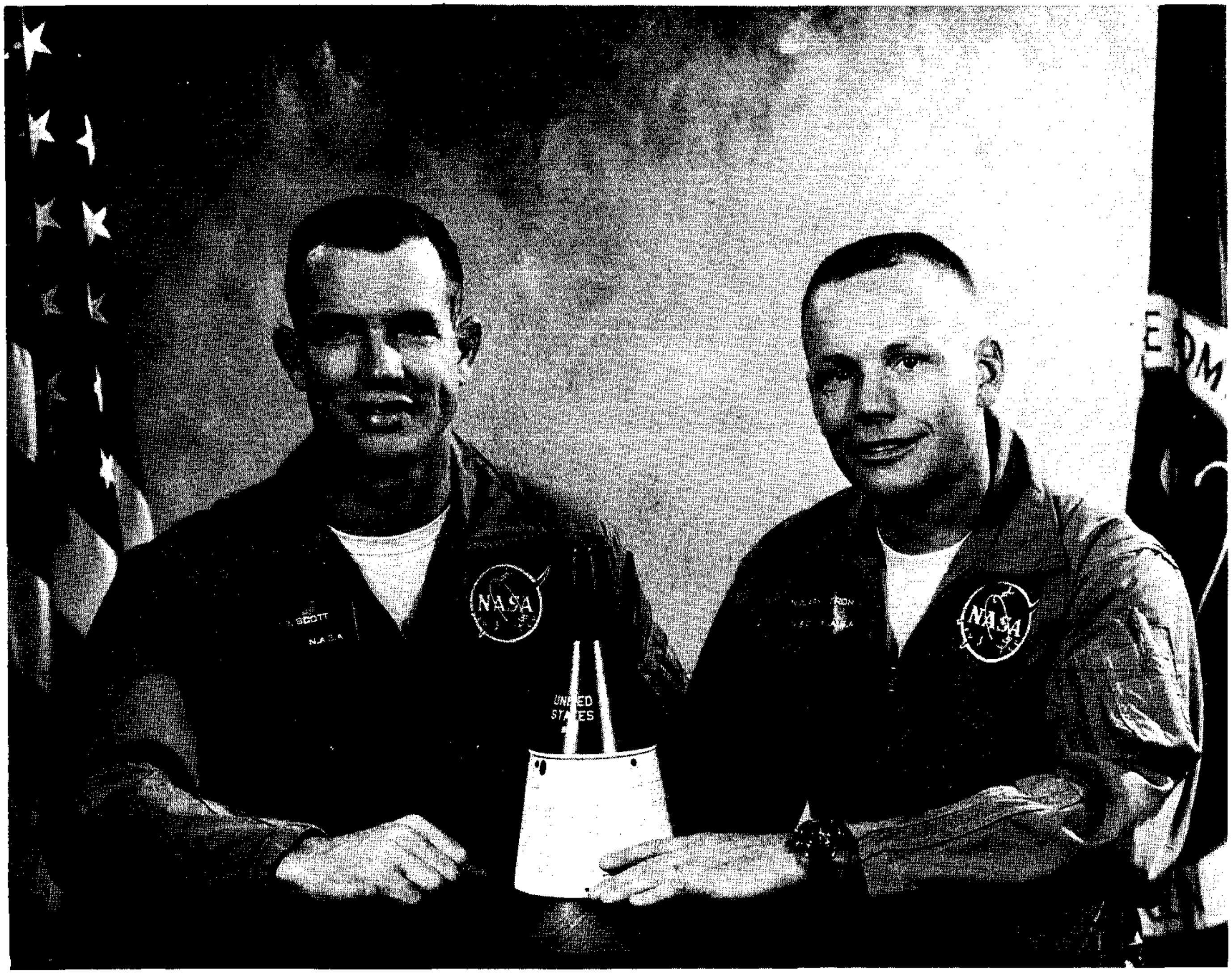

Astronaut David R. Scott, pilot, and Astronaut Neil A. Armstrong, command pilot.

7.1.2 Gemini VIII Pilots' Report

7.1.2.1 Prelaunch.- In order to achieve all the planned objectives of the first day, it was necessary to restrict the planned rendezvous sequence to 9 hours or less. The launch window for rendezvous in six or less orbits was limited to approximately a 5-minute period. In order to maximize the possibility of launching within this window, crew insertion was scheduled for T - 115 minutes. This time was ample for crew activities required prior to launch. Only two incidents required additional time to that scheduled in the count: the left Koch fitting on the right-hand ejection seat was inoperative due to a spillage of adhesive material into the mechanism, and a launch-vehicle programmer sequence test had to be repeated late in the count. The adhesive material was satisfactorily removed by the backup flight crew, and the sequence test was completed without requiring a hold.

Information concerning the Gemini Atlas-Agena Target Vehicle (GAATV) launch and orbital elements was forwarded to the flight crew by the Spacecraft Test Conductor and was appreciated. Small Gemini Launch Vehicle (GLV) oscillations due to erector lowering, sequence tests, and wind gusts could be observed on the spacecraft rate indicators. Communications throughout the count were satisfactory.

7.1.2.2 Powered flight.- Acceleration, sound, and vibrational changes provided a definite lift-off signal. The roll program started at lift-off (LO) + 9.5 seconds and was completed at an indicated 93 degrees. The crew expected an indication of 97 degrees rather than 93 degrees. The preflight change from 97 to 93 degrees was available at T - 3 minutes and should have been forwarded to the crew. The pitch program began at the correct time. Some mild vibration was noted after LO + 20 seconds but disappeared at approximately the time that supersonic speed was achieved. Subsequent powered flight was smooth. Two small tabs near the nose, one forward of each window, were observed to be oscillating throughout the flight within the sensible atmosphere. No longitudinal oscillations (POGO) were detected by the crew.

The staging sequence was very smooth. An exhaust-gas fireball was observed to extend in front of the spacecraft at Stage II ignition. Some residue appeared to accumulate on the windows at this time.

Closed-loop radio guidance was initiated on time. Lofting of the GLV was considerably less than had been expected, but yaw steering appeared normal. All spacecraft systems appeared satisfactory throughout launch, although the Environmental Control System (ECS) oxygen pressure was slightly above normal. No fuel-cell differential-pressure warning lights were observed to illuminate. A ratio of instantaneous-velocity to desired-velocity-at-SECO of 0.8 was reported by Mission Control Center-Houston (MCC-H) at LO + 307 seconds. Second-stage engine cutoff (SECO) occurred approximately at the planned time.

7.1.2.3 Insertion.- Residual rates at SECO were negligible. Separation was accompanied by a substantial amount of debris diverging radially from the spacecraft. A 7-second separation thrust changed the fore/aft Incremental Velocity Indicator (IVI) reading from 4 ft/sec aft before the thrust to 10 ft/sec aft after the thrust. The total velocity from the computer increased from 25 726 to 25 748 ft/sec. This 22 ft/sec increase was attributed to the 6 ft/sec separation maneuver and the 16 ft/sec tail-off. The out-of-plane error was indicated to be 18 ft/sec to the right.

Fairing jettison was accompanied by a surprisingly strong yaw-right and pitch-up moment. Release of both the nose fairing and the horizon-scanner cover were observed visually. The insertion checklist was completed at 00:11:00 g.e.t. A platform alignment was performed in the platform control mode and the spacecraft tended toward the left side of the yaw deadband. Thruster activity was predominantly restricted to yaw-right thrusters 3 and 4. This activity was necessary to compensate for the normal yaw-left moments produced by the launch-cooler evaporator exhaust.

7.1.2.4 Pre-transfer maneuvers.- The mid-course rendezvous maneuvers performed prior to terminal phase initiation (TPI) are shown in the following table:

Maneuver	G.e.t., hr:min:sec	ΔV, ft/sec	Direction	Control mode	Propellant quantity remaining after maneuver, percent
Height adjust	01:34:37	2.9	Retrograde	Platform	98
Phase adjust	02:18:25	50.6	Posigrade	Rate command	88
Plane change	02:45:50	26.2	Southeast	Rate command	-
Vernier height adjust	03:03:41	2.0	Posigrade	Rate command	-
Coelliptic	03:48:11	61.2	Posigrade, down, 21°	Rate command	75

With the exception of the second height adjust, the platform was aligned for 15 minutes prior to each maneuver. It occurred to the crew during this period that precise platform alignment was probably unnecessary for small maneuvers of less than 10 ft/sec, because the small errors that might be accumulated with only a short alignment could be corrected during subsequent maneuvers.

An excessive amount of time was required to null residual desired-velocity changes after each maneuver because responses in the computer readouts were slow and somewhat inconsistent below 0.5 ft/sec after small correction maneuvers, particularly in the right-left direction. In nulling these residuals, a more effective procedure would have been to null the MDIU address in the maneuver direction only, and, because of the associated small effects on the trajectory, reduce the other two components to approximately 1 ft/sec by using only the incremental velocity indicators.

Solid radar lock-on was obtained at a range reading of 179.11 nautical miles, after a short period of intermittent lock. Data were recorded continuously from the coelliptic maneuver to TPI; however, boresight was maintained to within only 2 degrees following the coelliptic maneuver, until after the platform alignment. Range-rate data between the coelliptic maneuver and TPI were questionable because of a 3-ft/sec scatter between sampling points. Initial total transfer-velocity computations also varied more than expected. Radar angle indicators remained steady until a range of approximately 45 nautical miles had been reached, although radar tracking resulted in the target vehicle staying approximately one-half degree left-of-center and one-half degree above-the-center in the optical sight. Between the 45 and 25 nautical-mile ranges, the radar boresight, relative to the optical boresight, varied as much as several degrees in a random fashion. Radar tracking was continued, however, because of the greater visual concentration required to maintain optical track. The GATV dipole antenna was utilized until a relative range of 20 nautical miles was reached, at which point the spiral antenna was selected.

The platform was aligned for 13 minutes prior to TPI, during the period between elevation angles to the target of 10 and 14 degrees. The elevation angle to the target was monitored by using computer address 84 (sine of radar elevation angle).

7.1.2.5 Terminal phase.- The terminal-phase-initiation maneuver was based on the following cues:

(a) Five minutes 30 seconds after the pitch gimbal angle exceeded 21.4 degrees, which was 1 hour 26 minutes 10 seconds after the coelliptical maneuver, only 48 seconds from the ground-computed time of 1 hour 25 minutes 38 seconds

(b) Comparison of closed-loop, backup-chart, and ground-computed ΔV required for TPI

(c) Minimization of closed-loop total transfer ΔV required

(d) Range at the last data point prior to TPI (32.46 n. mi.) compared to the ground-computed range at the same point (32.5 n. mi.)

(e) TPI ΔV and range, based on a polar plot of relative position from the coelliptical maneuver to TPI, compared with the data in (a) through (d).

Consideration of the available cues and the apparent anomalies, such as the small inconsistencies of the radar-angle indications and the smaller-than-planned difference in altitude between the Spacecraft 8 orbit and the GATV orbit, resulted in the selection of the onboard closed-loop solution for the TPI maneuver.

The transfer maneuver was monitored for computer, platform, and radar malfunctions according to onboard procedures and charts. The two planned closed-loop mid-course corrections were performed between TPI and terminal phase finalization (TPF). Four backup mid-course corrections were calculated, but not utilized because of the excellent performance of the radar and onboard-computer combination. The performance was determined by comparing the onboard-computed ΔV's with that computed from backup charts and also by comparing the actual relative trajectory with the nominal trajectory on the polar plot (fig. 7.1.2-1). Comparative ΔV's are shown in tables 7.1.2-I, 7.1.2-II, and 7.1.2-III. At the completion of the last mid-course correction, the range was 3.5 nautical miles, the pitch angle was 100 degrees, the GATV was sunlit and appeared as a cylinder, line-of-sight rates were negligible, the relative trajectory was close to nominal, and 65-percent of the propellant quantity remained. At this point, the TPF or braking maneuver was initiated.

Braking maneuvers were performed in increments based on visual cues and continuous readouts of onboard range and range rate. Because of the optimum relative position combined with low line-of-sight rates and the late time of arrival, a higher range rate was maintained during closing than had been planned. However, the braking was smooth, easily controlled, and at no time was there any question of other than a successful rendezvous. The first braking maneuver was 8 ft/sec aft, performed at a range of 1.72 nautical miles, 44 ft/sec closing velocity, and a pitch angle of 116 degrees. Eight subsequent maneuvers culminated in station keeping at 150 feet along the local horizontal and in a blunt-end-forward (BEF) attitude, 42 minutes after TPI, with 55-percent propellant remaining. The size and shape of the stabilized GATV provided excellent visual cues throughout the braking maneuver.

7.1.2.6 Station keeping.- Station keeping was performed in pulse, rate-command, and platform control modes. If maneuvering thrusters are operated for short periods only, no moments are created which cannot be readily removed with a few pulses in pulse mode. The platform mode was a very good mode for station keeping, and the operation required very little attention. A 10-to-15 minute BEF platform alignment was conducted in both platform and pulse modes using small impulses from the maneuvering thrusters to maintain a constant relative position to the GATV. Station-keeping range was generally maintained at approximately 50 feet. At this range, the GATV status-display-panel lights and gages could not be adequately observed. However, all lights, with the exception of the docking light, could be observed by using the 6-power magnification of the sextant.

7.1.2.7 Docking.- Docking was performed with the GATV configured to flight control mode 6, a tight-deadband mode. Flight control mode 1, the coasting mode used prior to docking, also appeared to be satisfactory.

The spacecraft was stopped approximately 3 feet from the Target Docking Adapter (TDA) to inspect the status display panel, spacecraft latches, and docking-cone configuration. No discrepancies were noted.

With the spacecraft in the rate-command control mode, an approach to the TDA was initiated. Contact occurred with less than 2-inches linear displacement and very little angular misalignment at a contact velocity of approximately 3/4 ft/sec. No electrostatic discharge was noted at contact. No GATV reaction was apparent. Entry of the spacecraft nose into the docking cone was very smooth. The latches appeared to engage immediately and the cone began to retract. A STOP-RIGID signal was sent by the crew immediately upon illumination of the RIGID light. STOP-ARM switch cycling was accompanied by illumination of the ARM light, indicating proper hard-line command capability.

The docking maneuver was performed over the Rose Knot Victor to assure maximum data collection. This placed the spacecraft near the terminator with the TDA pointing north, giving the appearance of a night docking through the left window. The docking light was on and illumination of the GATV was considered satisfactory.

7.1.2.8 GATV yaw maneuver.- The command sequence directing the GATV to yaw the spacecraft—GATV combination through a 90-degree attitude change was performed. The yaw rate was slightly greater than the expected rate of 1.5 deg/sec, and the 90-degree yaw attitude change was completed in 55 seconds. Yaw-rate initiation and termination were crisp, but smooth. Pitch and roll were held quite small during the maneuver; however, the spacecraft Inertial Guidance System (IGS) did indicate an 8-degree pitch-down attitude at the completion of the yaw maneuver.

7.1.2.9 Control system problem.- At approximately 7 hours g.e.t., the two spacecraft were configured for the platform-parallelism test, which was to have provided a comparison of the spacecraft and GATV attitude reference systems. The GATV Attitude Control System (ACS) was active, and the TDA L-band transponder was off. The spacecraft attitude-control power switch and maneuver-control switches were off. The radar was off, and the control mode switch was in PULSE.

Shortly after sending encoder command 041 (recorder ON), roll and yaw rates were observed to be developing. No visual or audible evidence of spacecraft thruster firing was noted, and the divergence was attributed to the GATV.

Commands were sent to de-energize the GATV ACS, geocentric rate, and horizon sensors, and the spacecraft Orbital Attitude and Maneuver System (OAMS) was activated.

The rates were reduced to near zero, but began to increase upon release of the hand controller. The ACS was commanded on to determine if GATV thruster action would help reduce the angular rates. No improvement was noted and the ACS was again commanded off. Plumes from a

GATV pitch thruster were visually observed, however, during a period when the ACS was thought to be inactivated.

After a period of relatively stable operation, the rates once again began to increase. The spacecraft was switched to secondary bias power, secondary logics, and secondary drivers in an attempt to eliminate possible spacecraft control-system discrepancies. No improvement being observed, a conventional troubleshooting approach with the OAMS completely de-energized was attempted, but subsequently abandoned because of the existing rates.

An undocking was performed when the rates were determined to be low enough to preclude any recontact problems. Approximately a 3 ft/sec velocity change was used to effect separation of the two vehicles.

Angular rates continued to rise, verifying a spacecraft control-system problem. The hand controller appeared to be inactive. The Reentry Control System (RCS) was armed and, after trying ACME-DIRECT and then turning off all OAMS control switches and circuit breakers, was found to be operative in DIRECT-DIRECT. Angular rates were reduced to small values with the RCS B-ring. Inspection of the OAMS revealed that the no. 8 thruster had failed open. Some open Attitude Control and Maneuver Electronics (ACME) circuit breakers probably accounted for the inoperative hand controller noted earlier. All yaw thrusters other than number 8 were inoperative. Pitch and roll control were maintained by using the pitch thrusters.

7.1.2.10 Preretrofire.- Prior to retrofire, the spacecraft was stowed essentially in the launch configuration. Television-monitor stowage required excessive time and effort because of the design of the installation. The extravehicular activity (EVA) visor had to be stowed outboard of the top of the left seat because no reentry stowage location had been provided. As in previous flights, the center stowage box was difficult to close because the door pins and holes did not align.

The new Auxiliary Tape Memory Unit (ATMU) was utilized to transfer the reentry program to the computer. Module IV-A was automatically loaded and verified, and the load was then verified automatically with Module IV-B. Occasional attitude thruster activity occurred during the process with no apparent effect.

Two attempts were made by the ground personnel to transmit the time-to-go to retrofire and preretrofire command load. The first load was unsatisfactory, as the time-to-go to retrofire was negative and counting up after the Digital Command System (DCS) light had been reset; no explanation for this situation was received. The second update was satisfactory and all Manual Data Insertion Unit (MDIU) readouts agreed with the transmission received from the ground controllers. Backup bank angles, times, recovery call signs, and weather were received expeditiously prior to retrofire.

7.1.2.11 Retrofire.- All four retrorockets fired on time using the AUTO sequence. The rate-command control mode was used and retrorocket-thrust misalignments appeared to be small. The retrofire velocity vector, as indicated by the IVI, was 292 ft/sec aft, 000 ft/sec out-of-plane, and 114 ft/sec down. No change in these values was noted as the retro adapter was jettisoned, and the following changes in velocity were read out of the computer: 292.5 ft/sec aft, 0.3 ft/sec right, and 114.1 ft/sec down.

7.1.2.12 Reentry.- The reentry rate-command and pulse modes were selected for reentry to minimize fuel consumption because, prior to retrofire, the propellant quantity was indicated to be 4 pounds in the B-ring and 23.5 pounds in the A-ring.

The following control-mode sequence was used:

Period	Ring	Control mode
Retrofire	A and B	Rate command
Retrofire to 400K feet	B	Pulse
400K feet to 3g	B	Reentry rate command
3g to drogue parachute deploy	A	Reentry rate command
Drogue parachute deploy to fuel depletion	A and B	Rate command

As an altitude of 400K feet was approached, the spacecraft was rolled to a bank angle of 52-degrees left, the crew-computed lift-vector orientation required to reach the target in case of a guidance-system malfunction. The computer indication of 400K feet occurred precisely at the ground-predicted time, adding credence to the computation.

Three minutes fifteen seconds after the 400K feet indication, guidance initiate occurred exactly on schedule. Downrange error was 90 nautical miles, comparing satisfactorily with the ground prediction of 77 ± 50 nautical miles. A 52-degree left bank angle was maintained for 1 minute while the downrange and crossrange errors were monitored. Indicated oscillations were less than 5 miles, compared with the expected ±40 miles.

At the end of the monitoring period, the spacecraft was rolled from 52-degrees left to approximately 30-degrees right, zeroing the roll command. Crossrange error indicated the aim point to be 3 miles north of the flight path.

When the errors were nulled, a full-deflection command (15 deg/sec) was applied. Actual roll rates achieved appeared to be 8 to 12 deg/sec. The roll command logic is difficult to interpret and is felt to compromise the ability to accurately control the spacecraft during the high-acceleration portion of the trajectory. The roll-rate reversals require excessive fuel consumption. It was apparent that very little lift was available after the peak acceleration of 6g.

The drogue parachute was actuated at 50K feet and was accompanied by oscillations of ±20 degrees. The rate-command mode appeared to have little effect on stabilizing or destabilizing the spacecraft.

At 27K feet, oxygen high rate was actuated and the recirculation valve opened to the 45-degree position. Suit temperatures were warm but satisfactory. The cabin was found to be filled with acrid fumes upon opening the visors but, because the visors were immediately closed, very little entered the suits.

The main parachute was actuated at 10K feet. After using a cockpit mirror to verify a water landing, the spacecraft was oriented to the landing attitude. Cabin repressurization was actuated at 2000 feet but was ineffective in raising cabin pressure. The water landing was more severe than expected and was accompanied by substantial spacecraft attitude changes, with both windows being completely immersed.

7.1.2.13 Recovery.- Immediately after landing, voice transmissions on UHF and HF were initiated to the recovery forces. However, the only reception on the spacecraft frequencies was oriental music on HF. Approximately 15 minutes later, a C-54 rescue plane was observed passing overhead at approximately 800 feet. Ten minutes later, the first of three pararescumen were observed descending toward the water. The only UHF contact with the rescue aircraft was achieved about 30 minutes after landing; however, it was clear and all necessary recovery information was received.

Odors in the cabin were strong throughout the recovery period. The flotation collar was not secured and the hatches opened until approximately 1 hour 15 minutes after landing. This required more time than anticipated because of the sea state (3-to-5 foot waves with 10-to-15 foot swells).

The spacecraft and crew were recovered by the destroyer, U.S.S. Leonard F. Mason, 3 hours after landing. Egress was directly from the spacecraft to the destroyer ladder.

7.1.2.14 Systems operation.-

7.1.2.14.1 Platform: During alignment across the terminator, some scanner-ignore signals and spurious thruster firings were observed, as had been reported on previous flights. The new marking scheme on the attitude indicator was believed to be a great improvement.

7.1.2.14.2 Environmental control: The suit heat exchanger control was maintained at MAX COOL, resulting in satisfactory suit-inlet temperatures of approximately 50° F. The two suit fans were left on throughout the flight. Cabin temperature varied from 80° to 90° F and was marginally satisfactory. Coolant loops were operated on the high-flow A-pumps. The drinking-water supply was filled with gas bubbles, and the water-gun discharge appeared like a foam.

7.1.2.14.3 Electrical: The fuel cells operated well but shared the load in an unexpected manner. At insertion, section 1 was carrying 27 amperes and section 2 was carrying only 16 amperes. At preretrofire, the values were 30.4 and 15.0, indicating an increasing split. This monitoring capability was enhanced by the availability of a main-bus ammeter. Purges were performed at 3:05 and 8:25 g.e.t. Differential-pressure warning lights were illuminated during both section 2 hydrogen purges and the second section 1 hydrogen purge.

Main-battery voltages 1, 2, 3, and 4 were low during the preretrofire check list; they were indicated as 21.3, 21.5, 22.0, and 22.1 V dc, respectively. The antenna-select circuit breaker was discovered tripped during the count and the ATMU and hydrogen-heater circuit breakers were found tripped during flight. Several ACME circuit breakers were found opened or tripped after the control-system problem.

7.1.2.14.4 Computer: When in catchup mode, the values in addresses 80, 81, and 82 (desired-velocity-change displays) would vary with time up to several tenths of a foot per second without thruster activity, making it impossible to accurately remove residual velocities.

When in rendezvous mode, during the pre-transfer rendezvous calculations, the total velocity required to rendezvous, as read on the IVI's and address 70, did not vary smoothly with decreasing range as expected. On three occasions, the value momentarily increased from its previous value before decreasing again.

7.1.2.14.5 Food: The time required for preparing food was excessive for the planned mission. This was due not only to the packaging concept, but also to the inadequacy of the reconstitution process when even more water and time were allowed than required according to instructions. In addition, the bite-size food produced more crumbs than had been expected and crumb control required extra time. Germacide pills were not used due to the lack of available time.

7.1.2.15 Experiments and operational checks.- Experiments S-3 and S-9 were activated as planned; S-9 was not recovered due to early termination of the flight.

An accelerometer bias check was performed over Carnarvon on revolution 1. Subsequent difficulties in removing maneuver residuals indicated either an inaccurate calibration or ground-bias update, or an onboard problem in measurement or computation of spacecraft maneuver accelerations.

The hand-held sextant was not used quantitatively; however, several star and GATV observations illustrated its practicality as a navigational instrument and as a device to measure range and range rate at ranges and range rates less than approximately 10 000 ft and 25 ft/sec. In addition, the 6-power magnification of the eyepiece was useful in evaluating the GATV status display panel at distances less than approximately 80 feet.

The radar test prior to the coelliptical maneuver was only partially successful due to radar lock-on occurring relatively near the maneuver. This test seems to be of little use because the same evaluation can be performed by 19 minutes after the maneuver, with the time prior to the maneuver then being better utilized in insuring a precise coelliptical maneuver.

7.1.2.16 Visual sightings.- The most significant visual sightings during the flight consisted of the GATV, stars, and horizon relative to day-night cycles. In general, stars can be observed approximately 4 minutes prior to spacecraft sunset (about the point at which the spacecraft crosses the terminator), and the horizon is completely lost at this time. A well-defined airglow horizon becomes visible about 4 minutes after sunset. The stars remain visible until approximately 4 minutes after sunrise.

The first visual contact with the GATV occurred at 76 miles relative range, in reflected sunlight, about 20 minutes prior to spacecraft sunset. Stars were observed at the same time in the vicinity of the target, and slowly disappeared until only the GATV was visible at 56 miles, around 12 minutes later. At a range of 45 miles, visual contact with the GATV transitioned abruptly from reflected sunlight to the acquisition lights, which were comparable to a sixth-magnitude star. This was at approximately spacecraft sunset.

Subsequent fading of the background and appearance of the GATV in reflected sunlight occurred rapidly at 4 minutes after the next sunrise. The range was 3.8 miles and the GATV appeared as a bright cylindrical object. The brilliance of this scene cannot be overemphasized.

Other visual sightings consisted of thruster-firing reflections at night, ground details, contrails, and a large number of particles drifting rearward along the flight path across the nose of the spacecraft at daybreak prior to TPF. Similar particles had been observed previously, but always moving parallel to the spacecraft nose with the spacecraft in the small-end-forward (SEF) attitude.

TABLE 7.1.2-I.- COMPARISON OF SOLUTIONS FOR

TERMINAL-PHASE-INITIATION MANEUVER

Terminal-phase-initiation factors	Ground computations	Closed-loop computations	Backup computations
Time from coelliptical maneuver, hr:min:sec	1:25:38	1:26:39	1:26:10
Forward/aft, ΔV, ft/sec	32 forward	25 forward	34 forward
Up/down, ΔV, ft/sec	1.7 up	3 up	25 down
Left/right, ΔV, ft/sec	5.7 up	8 up	-

TABLE 7.1.2-II.- COMPARISON OF SOLUTIONS FOR FIRST

MID-COURSE CORRECTION MANEUVER

First mid-course correction factors	Backup computation	Backup computation	Closed-loop computation
Time from TPI, min:sec	2:30	8:30	11:40
Forward/aft, ΔV, ft/sec	4.5 aft	4 forward	12 forward
Up/down, ΔV, ft/sec	10 down	2.5 up	6 up
Left/right, ΔV, ft/sec	-	-	1 right

TABLE 7.1.2-III.- COMPARISON OF SOLUTIONS FOR SECOND

MID-COURSE CORRECTION MANEUVER

Second mid-course correction factors	Backup computation	Backup computation	Closed-loop computation
Time from TPI, min:sec	14:30	20:30	23:40
Forward/aft, ΔV, ft/sec	3 aft	1 aft	4 forward
Up/down, ΔV, ft/sec	2.5 up	4 up	7 up
Left/right, ΔV, ft/sec	-	-	5 right

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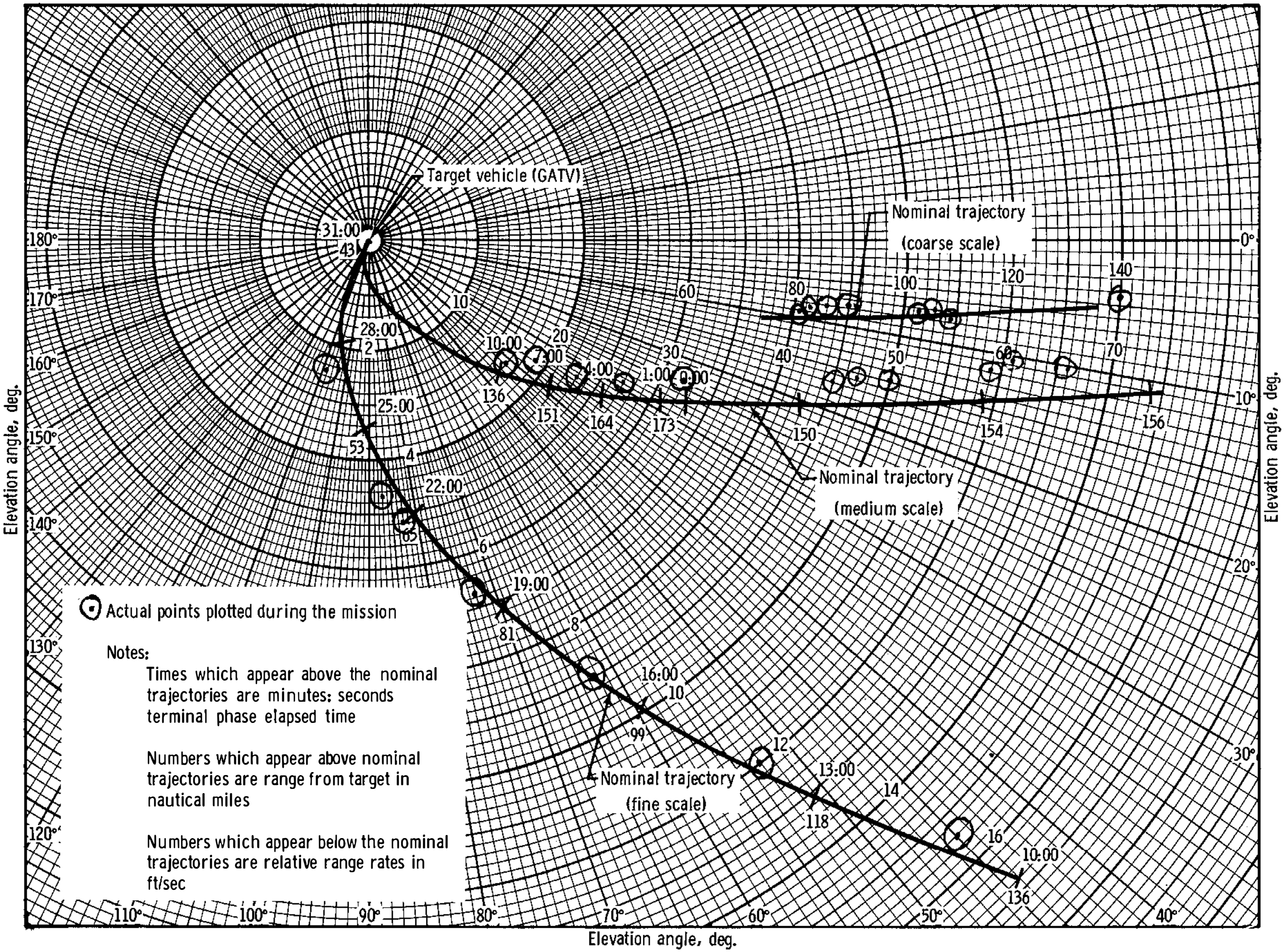

Figure 7.1.2-1. - Onboard target-centered coordinate plot of rendezvous.

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7.2 AEROMEDICAL

Gemini VIII was the first in a series of missions which include rendezvous, docking, and extravehicular activities in a relatively short and busy flight. The medical emphasis in this flight was shifted to operational medical support and biomedical monitoring only as required for mission safety. However, as a by-product of these operational procedures, a considerable amount of information was gained. A failure in the Orbital Attitude and Maneuver System produced physiological and psychological stresses in excess of those expected for the planned mission. The Gemini bioinstrumentation system, along with spacecraft data, provide an indication of the degree of stress and some of the physiological responses to this stress seen under emergency conditions. These data are presented in the following sections.

7.2.1 Preflight

7.2.1.1 Medical histories.- Clinical background data from the flight crew were obtained from their military health records, records of medical examinations conducted at the time of their selection as astronauts, and their annual medical examinations since selection. In addition, a considerable volume of data was collected during simulated flights and spacecraft systems tests. These data were reviewed during preflight activities and compared with the inflight and postflight data. Of particular interest was the pilot's response to treadmill studies performed during his pre-selection physical. The prolonged extravehicular activities planned for this mission were expected to require an unusual amount of physical stamina. These studies indicated that the pilot was capable of strenuous physical exertion without ill effects and provided physiological information for comparison with data received during the planned extravehicular activities.

Also of particular interest were the command pilot's tilt-table responses during his pre-selection physical. These studies, accompanied by a breath-holding procedure, produced bradycardia and a 3-second to 5-second period of syncope. This reaction was considered to be a normal variant and was therefore not disqualifying. These data, along with a preflight tilt study, alerted the medical support personnel to the possibility of syncope during the postflight tilts. This history also pointed to the need for a thorough briefing on the possibility of, and methods of self-protection against, postural hypotension during the recovery phase of the mission.

7.2.1.2 Preflight activities.- Medical support for the mission began at the initial spacecraft stowage review, shortly after crew selection. After review of the mission objectives and proposed flight plan, it was considered timely to delete the requirement for onboard blood-pressure measurements. This decision was in line with the Chief of Center Medical Program's objectives to improve crew comfort and the convenience of these operationally oriented missions without compromising medical data necessary for crew safety. Table 7.2-I lists the other preflight activities of medical significance.

7.2.1.2.1 Diet: After the crew moved to the Kennedy Space Center, their diet was specially prepared in the astronauts quarters. Although the crew continued to eat a normal, unrestricted diet, both the facility and the diet were closely monitored by medical personnel. On March 10, 1966, the prime crew were started on a low-residue diet which continued until launch. In order to decrease the necessity for defecation during flight, the crew were given a mild laxative, Bisacodil. Prior tests had indicated that one-half the normal recommended dose was sufficient for each of the crew members. The pilot took this medication on the night of March 10, 1966, and the command pilot took it on the night of March 11, 1966, with expected results and without side effects. Due to a 24-hour delay in the launch schedule, this medication was repeated on the night of March 13, 1966, again with expected results and no side effects.

7.2.1.2.2 Physical fitness: Both crewmen habitually maintained an excellent state of physical fitness. Although the preflight activities required a large amount of their time, both crewmen made a special effort to maintain a satisfactory level of physical fitness during the preflight period of this mission. Even prior to selection as a member of the Gemini VIII crew, the pilot ran several miles each morning. He continued this practice with few exceptions after moving to the launch site and was in an exceptionally good state of physical fitness. The command pilot, while not as muscular as the pilot, maintained a state of physical conditioning which was considered completely adequate for his role in the mission.

7.2.1.2.3 Drug and sensitivity test: The prime and backup crewmen were tested for adverse effects or sensitivity to each medication which was included in the onboard medical kit and for sensitivity to all items used in bioinstrumentation. No adverse effects were noted.

7.2.1.2.4 Physical examinations: The prime crewmen were given a physical examination by a specialist in internal medicine on March 6, 1966. A comprehensive medical examination was given to the prime crew on March 10, 1966, by the two crew flight surgeons, and specialists in ophthalmology and otolaryngology. The results of these physical examinations are entered as a part of the crewmen's health record. There were no abnormal findings, except the command pilot had signs and symptoms of a mild upper-respiratory infection. This was under treatment by the crew flight surgeon. On launch morning, a brief physical examination was given by the crew flight surgeon. The command pilot's previous signs and symptoms had almost completely disappeared and both crewmen were considered ready for flight.

7.2.1.2.5 Laboratory studies: In support of the M-5 Experiment (see section 8), there were two 48-hour urine collections preflight. These collections were completed on the mornings of March 7 and March 11, 1966. The required amount of blood was drawn at these times. The results of these determinations are shown in tables 7.2-II through 7.2-IV. Due to the remote landing area, hematology and certain blood chemistries were not possible and these determinations had to be omitted from this report.

7.2.1.2.6 Tilt-table studies: Preflight tilt-table studies are entered as a part of the crewmen's health record. Due to the emergency landing in a secondary area, no postflight tilt studies could be performed. Therefore, tilt studies are deleted from this report.

7.2.1.3 Prelaunch preparation.- Prelaunch preparations proceeded essentially as planned and are listed in table 7.2-V.

7.2.2 Inflight

This section includes events from lift-off to spacecraft landing, an elapsed time of approximately 10 hours 41 minutes.

7.2.2.1 Physiological monitoring.- Physiological data obtained from the Gemini bioinstrumentation system and certain environmental parameters were monitored by physicians at the Mission Control Center-Houston (MCC-H) and at remote network tracking sites. The electrocardiogram and pneumogram tracings on each crewman were relayed to MCC-H over the voice data lines either during a pass over the station or immediately after the pass. The quality of analog data received at MCC-H was satisfactory for clinical analysis.

7.2.2.1.1 Electrocardiograms: The rates and patterns of the electrocardiogram on each crewman remained within normal and expected limits. During the flight a detailed analysis of the electrocardiograms for rates, patterns, and intervals was made during each pass by the remote-site physicians and/or the physicians at MCC-H. A rate history of data received at MCC-H during the pass was obtained by use of a digital cardiotachometer and graphic printout method. The rates were also transformed into graphs by Aeromedical Staff Support Room personnel at MCC-H and further analyzed for trends or significant findings. The electrocardiogram (EKG) and pneumograph of each crewman were also recorded on the onboard biomedical tape recorders. Significant periods of these records were reviewed during the postflight analysis. Figure 7.2-1 shows the rates received at each station during the pass. These are displayed as the average, high, and low rates during the various station passes. Figure 7.2-2 shows data obtained from the biomedical tape recorder shortly after the flight. The heart rates of the crewmen are compared with the approximate rates of roll during the spacecraft control-system problem.

7.2.2.1.2 Respiration: The respiratory rates, as measured by the impedance pneumogram, were within the expected normal range and are shown in figures 7.2-1 and 7.2-2.

7.2.2.1.3 Oral temperature: Oral temperature probes were attached to the earpiece of the helmets and to the lightweight headsets. Due to the early termination of this flight, no oral temperatures were obtained.

7.2.2.2 Medical observations.-

7.2.2.2.1 Lift-off and powered flight: The crew experienced no difficulty in reading their instruments or communicating during powered flight. There was no longitudinal-oscillation (POGO) effect; however, there was a slight vibration noticed approximately 20 to 40 seconds after lift-off. The physical effects of g-forces encountered were less than anticipated by the crewmen. There were no unusual sensations described concerning the insertion into orbit and the associated transition to weightless flight.

7.2.2.2.2 Environment: During this relatively short and busy flight, the cabin environment remained warm, with temperatures steadily increasing to over 90° F. However, the suit inlet temperatures were around 50° F. With both suit fans on and the control at full cold, the crew were comfortable. Most of the flight was performed with the helmets and gloves off.

7.2.2.2.3 Food and water: Three meals of Gemini flight food per crewman per day were stowed aboard the spacecraft. The meals provided a daily average of 2748 calories for the command pilot and 2787 calories for the pilot. In addition, one snack (651 calories) was provided for each crew member. Because of the extremely busy flight plan during the early rendezvous phase of the mission, no time was allotted in the flight plan for eating until after rendezvous and docking. Two meals per crewman were stowed in an easily accessible area of the spacecraft. The crew planned to eat bite-sized portions of the menu and to reconstitute juices and other items for use whenever they could find the time to eat. Two meal packages were opened during the entire flight. Although no log of food and water was required or reported, it is estimated that the command pilot consumed between 400 and 600 calories and the pilot consumed between 600 and 800 calories. Water intake was considered adequate by the crewmen. They felt subjectively that some dehydration occurred after landing. There was no way to determine their state of hydration at the time of reentry.

7.2.2.2.4 Waste: Neither crewman removed the launch-day urine-collection device during this mission. The command pilot urinated once during the flight and found there was some leakage, approximately 20 cc, during reentry. The pilot did not urinate until after recovery.

7.2.2.2.5 Vision: The crew reported a reflection in their face plates under some lighting conditions which decreased their visual acuity to some extent. They also reported a definite coating on the spacecraft window. They did, however, observe that objects on the ground could be identified readily. They saw aircraft contrails over Los Angeles, ground fires in Africa, lightning and thunderheads over the South China Seas, and during reentry they were able to get a good view of the Himalayas. The crew felt that more stars were visible at night from the spacecraft than from the ground. Due to the unexpected termination of the flight, they were unable to quantitate this observation. They stated that visual acuity outside the window was affected considerably by the cockpit lighting. Dark adaptation seemed to be normal; however, as might be expected, they reported that lighting inside the cockpit had to be very dim to permit this adaptation when the white cockpit lights were in use. The command pilot used white lights on his side and the pilot used red lights. The pilot considered that in using the rendezvous charts, his effective vision was better with a dim red light than with a dim white light. The command pilot was concerned with transferring back and forth from the optical sight to the target to the radar information on the panel and believed that the very dim white light facilitated this transfer.

The command pilot was able to make the first visual contact with the Gemini Agena Target Vehicle (GATV) at a range of approximately 76 miles. At 4 hours 40 minutes ground elapsed time (g.e.t.), the GATV was definitely identified as a very bright, cylindrical object at a range of 55 miles. From the first visual sighting, the GATV was tracked visually for approximately 20 minutes on the day side of the orbit. Then, with the aid of the acquisition lights, it was tracked visually throughout the 35-minute night side. At sunrise the spacecraft was approximately 3.8 miles from the GATV.

After station keeping in close proximity to the GATV during the daylight side, docking occurred at 6 hours 33 minutes 16 seconds g.e.t., shortly after sunset. This gave the crew a wide range of visual experience with nearby space objects under various lighting conditions. In the sunlight, the GATV was brilliantly illuminated. Both crewmen reported they could look at the GATV without squinting, but they felt more comfortable with sunglasses. The crew experienced no difficulty interpreting visual cues and were able to accomplish close formation flying and docking with the GATV with relative ease. During docking, the spacecraft and GATV were so oriented that the command pilot's window was completely in darkness and the pilot's window was completely in daylight.

7.2.2.2.6 Orientation: Twenty-seven minutes after docking with the the GATV, a thruster failure occurred in the spacecraft Orbital Attitude and Maneuver System which caused the vehicles to roll and yaw at unexpected rates. This occurred during the darkness period of the orbit. Spacecraft lights were turned up and the crew were busy with other tasks, so they did not have visual or auditory clues to indicate they were starting to roll and yaw. The first indication of unusual spacecraft motion was a visual reference to the rate and attitude indicators on the instrument panel. Spacecraft motions were in three axes; yaw, pitch, and roll. Pitch and yaw rates did not exceed 20 deg/sec; however, the roll rate increased to approximately 300 deg/sec. The time history of these roll rates in relation to the pilot's and command pilot's heart rates is shown in figure 7.2-2.

The crew stated that they were not disoriented at any time during this period. There was no nausea, no pain or occular discomfort, no nasal congestion, and no sense that they were being thrown in any particular direction. During the period of maximum roll rates, they did notice that their sense of orientation was being disturbed. This was analogous to a high roll rate in an aircraft. They noted that by moving their heads in a particular direction they could detect the onset of this phenomenon. If they held their head position unchanged, they could avoid any disorientation. This was particularly noted in looking at the overhead circuit-breaker panel. They could hold their heads back against the head rest, turn slightly, and see the circuit breakers with relative ease. However, if they attempted to look at the circuit breakers in the normal fashion by bending forward and twisting their head to the appropriate side, they would get into a disorientation problem. This was noted quickly, and all unnecessary head motions were avoided. The crew reported that the visual reference to the horizon after sunrise, at 7 hours 8 minutes g.e.t., was comparable to seeing the ground go around when in a spin. This is not an unusual phenomenon for experienced pilots and was considered to be helpful in orienting themselves.

It is interesting to note that these crewmen reported no symptoms which could be attributed to the centrifugal force involved. In computing these forces, it was found that the center-of-gravity of the crewmen was approximately 14 inches from the center-of-gravity of the spacecraft in the longitudinal axis. Including the geometry of the seats in the spacecraft, it was determined that these forces would be a composite of the forces acting downward on the legs, laterally on the torso, and upwards on the head. The resultant vector was approximately 70 degrees from each crewman's vertical axis. The magnitude of the centrifugal force under these conditions is considered to be less than 2g as computed from available data. At no time were the centrifugal forces considered to be a problem by the crew.

7.2.2.2.7 Retrofire and reentry: The sensation of acceleration during retrofire was essentially the same as has been reported by previous crews. The crewmen believed that they could determine which retrorocket was firing by the lateral excursions associated with each retrorocket firing. The g-forces during reentry were as expected. There was no difficulty in breathing or in controlling the spacecraft. The crew were properly braced for the change in spacecraft attitude from single-point to two-point suspension and experienced no difficulty at that time. There were no symptoms referable to postural hypotension during descent.

7.2.3 Postflight

This portion of the report includes aeromedical observations from the time of spacecraft landing until the crew returned to the Kennedy Space Center. These data were obtained from limited clinical and laboratory examinations performed onboard the recovery ship; from medical observations of the crew at Tripler General Hospital, Hawaii; and from a limited medical examination of the ear, nose, and throat, and a medical debriefing upon return to Cape Kennedy. Postflight deviations from normal were limited to the following:

(a) Slight crew fatigue

(b) Nausea and diaphoresis prior to crew recovery

(c) Subjective dehydration

(d) Hemoconcentration.

7.2.3.1 Recovery medical activities.- Recovery medical activities planned for this and other short-duration Gemini rendezvous missions are to be reduced in scope. Previous Mercury and Gemini flights have provided the background experience necessary to anticipate the operational medical support required. This recovery, the first in a secondary landing area, indicates that these requirements were met.

7.2.3.1.1 Planned recovery medical procedures: The postflight medical evaluation was scheduled to be less detailed than that following the previous Project Mercury and Gemini flights. Routine tilt-table studies were scheduled the same as on previous missions, twice on recovery day and daily thereafter until crew-member responses returned to preflight values. Laboratory procedures planned were to be limited to routine chest roentgenograms, complete blood count, erythrocyte sedimentation rate, erythrocyte osmotic fragility test, and a urinalysis. Blood and urine specimens were to be collected for Experiment M-5. The postflight medical examination was also to be less comprehensive than those performed following previous flights, with special emphasis to be placed on the cardiovascular system; therefore, only the internist-cardiologist member of the medical evaluation team was deployed to the primary recovery ship. Examinations of additional systems were to be performed as indicated by the NASA physician and/or the Department of Defence (DOD) members of the Recovery Medical Team.

As in all previous Gemini missions, the primary recovery ship, an aircraft carrier, was located in the western Atlantic recovery zone 1; however, any of the smaller ships in the recovery force were available to affect retrieval of the spacecraft and its crew should it become necessary to land in other than the primary landing zone. Medical personnel, who have been indoctrinated in recovery medical procedures, are deployed onboard all of the smaller recovery ships pre-positioned in each of the four recovery zones. Termination of this mission earlier than planned due to inflight control problems resulted in reentry into the West Pacific landing zone 3 during revolution 7 (7-3 landing area). This landing area was supported by a destroyer, the U.S.S. Leonard F. Mason. The medical personnel onboard consisted of a Navy physician and hospital corpsman, as well as the medical technician from the DOD Medical Recovery Force who had been deployed to the ship following a pre-mission briefing and indoctrination session.

7.2.3.1.2 Narrative: Spacecraft landing forces were greater than the crew had anticipated. This was attributed by the crew to a combination of factors such as the spacecraft oscillation on the parachute, wave height, and the ocean swell. Following landing at approximately 10 hours 41 minutes g.e.t., the crew remained suited with the spacecraft hatches closed until the flotation collar was attached to the spacecraft by pararescue personnel. This took approximately 45 minutes and required an unusual effort on the part of pararescue personnel. The spacecraft had been sighted in the air prior to landing. Pararescue personnel were deployed into the landing area promptly; however, rough seas and motion sickness somewhat compromised their efforts. Nauseating odors from the heat shield and residual fumes from the Reentry Control System, combined with an uncomfortably hot spacecraft and a relatively rough sea state, caused considerable discomfort to the crew. Symptoms included nausea with minimal vomiting, profuse sweating, and subjective dehydration of both crew members. The total time spent in the spacecraft was approximately 3 hours. The crew egressed just prior to the spacecraft being hoisted aboard the recovery ship. Both crew members egressed from the left hatch, with the right hatch closed. The crew climbed aboard the recovery ship by means of a ladder and assistance from the ship's personnel.

The crew experienced no symptoms upon standing on the deck and being welcomed aboard the recovery ship 3 hours 5 minutes after landing (06:28 G.m.t.). Upon advice of the NASA Medical Director, both crew members had taken one 25-mg meclizine hydrochloride tablet, an anti-motion sickness drug, just prior to retrofire. The crew believed that this medication reduced their symptoms of nausea. Immediately after arriving onboard the ship, the crew proceeded to the ship's wardroom where the postflight medical evaluation was begun. At no time did either crew member exhibit evidence of disorientation, instability, or postural hypotension.

7.2.3.2 Examinations.- The medical examination was conducted by the Navy medical officer, assisted by the DOD recovery medical technician, who had been briefed and deployed for this purpose. Recovery medical procedures were carried out in accordance with Section III of the DOD Overall Medical Support Plan for Gemini Operations. Tilt-table studies and special laboratory procedures were omitted by direction of the Medical Director.

Medical observations began with the doffing of the space suits. The suits were removed by the recovery forces medical technician. Both crew members were thirsty but appeared only minimally dehydrated on clinical examination. The undergarmets were soaked with perspiration. The command pilot had some urine staining of the underwear, which occurred during reentry when the urine-collection device allowed approximately 20 cc of spillage. Except for minimal erythemia at the sensor sites, the skin of both crew members was normal during the inspection. Both crew members were tired, but showed no unusual evidence of fatigue. Both appeared to be in good physical condition; however, the pilot showed less evidence of the effects of sea sickness than did the command pilot. There were no other significant abnormalities.

No tilt-table studies were attempted. Due to ship's motion, it was difficult to accomplish simple procedures such as measuring the blood pressure and drawing blood samples. It was not possible to record an accurate body weight; and laboratory procedures, with the exception of partial urinalysis, were impossible. The laboratory results which are available are included in tables 7.2-II through 7.2-IV. The medical evaluation was completed approximately 2 hours 22 minutes after recovery.

After the crew had had a short, sound nap, the second blood specimens were obtained at 6 hours after recovery. Both crew members ate and went back to sleep until 6:00 a.m. local time (21:00 G.m.t.) the following morning, March 17, 1966. The recovery ship docked at Naha, Okinawa, Ryukyu Islands, at approximately 9:10 a.m. hours local time the next day. Shortly thereafter, a NASA team (including a NASA physician) came onboard. The crew and the NASA team departed the recovery ship at approximately 11:00 a.m. hours local time, March 18, and were flown to Hawaii.

They arrived at Hickam Air Force Base shortly after midnight local time, March 18, and were admitted to Tripler General Hospital for observation only. Although there were no medical examinations at Tripler, intake and output records were kept, and electrocardiograms were performed on both crew members the following morning. After discharge from the hospital, the crew returned to Hickam Air Force Base and were flown to Cape Kennedy, Florida.

Further examination, including caloric studies by a specialist in otolaryngology, were performed in conjunction with the medical debriefing. This examination again indicated no abnormalities.

TABLE 7.2-I.- SIGNIFICANT PREFLIGHT MEDICAL ACTIVITIES

Date	Activity	Medical study or support
December 4 through 10, 1965	Spacecraft test in altitude chamber	Prime and back-up crew examinations before and after each test. Biosensors used during each test.
February 3 through 13, 1966	Extravehicular-Life-Support-System test in altitude chamber	Prime and backup-pilot examinations before and after each test. Biosensors used during each test.
February 16, 1966	Joint combined system test	Back-up crew suited and sensored.
February 16, 1966	Tilt-table studies	Biosensors used.
March 6, 1966	Physical examination and tilt-table studies	Internist examination including use of biosensors.
March 7, 1966	Complete 48-hour urine collection and laboratory studies	Prime crew. Medical support to M-5 Experiment.
March 8, 1966	Physical examination, tilt-table studies, and laboratory studies	Back-up crew.
March 9, 1966	Simultaneous launch demonstration	Prime crew suited and sensored.
March 10, 1966	Simulated flight and EVA bio-medical test	Back-up crew suited with back-up pilot sensored.
March 11, 1966	Physical examination, tilt-table studies, laboratory studies, and complete 48-hour urine collection	Specialist examination including use of biosensors and medical support to M-5 Experiment.
March 15, 1966	Prelaunch physical examination	Prime crew examined by crew flight surgeons.

TABLE 7.2-II.- URINALYSIS

(a) Command Pilot

Determination	Preflight		Postflight
	March 7, 1966	March 11, 1966	March 17, 1966
Time (Local)	07:30	07:00	Recovery + 8 hours
Volume, cc	255	270	152
Color, appearance . . .	Yellow, clear	Yellow, clear	-
Reaction	Acid	Acid	pH 7
Specific gravity	1.030	1.025	-
Albumin	Negative	Negative	1+
Sugar	Negative	Negative	Negative
Bile	-	Negative	-
Microscopic	Rare epithelial cells; 0-2 wbc/hpf	0-2 wbc/hpf, few bacteria	-

(b) Pilot

Determination	Preflight		Postflight
	March 7, 1966	March 11, 1966	March 18, 1966
Time (Local)	06:30	06:45	Recovery + 15 hours
Volume, cc	255	225	510
Color, appearance . . .	Yellow, clear	Yellow, clear	-
Reaction	Acid	Acid	pH 7
Specific gravity	1.025	1.026	1.030
Albumin	Negative	Negative	1+
Sugar	Negative	Negative	Negative
Bile	-	-	-
Acetone		-	
Microscopic	Rare epithelial cells; 0-1 wbc/hpf mucous		-

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TABLE 7.2-III.- URINE CHEMISTRIES

(a) Command Pilot

Determinations Date, 1966 Time, e.s.t.	March 5 18:30	March 6 00:50	March 6 07:00	March 6 12:15	March 6 18:45	March 6 22:30	March 7 07:30	March 9 08:50 - 11:15	March 9 14:15
Total volume, ml	455	420	280	235	355	175	255	505	420
Glucose quality	Negative	Negative	Negative	Negative	Negative	-	-	Negative	Negative
Protein quality	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative
Specific gravity	1.028	1.028	1.023	1.029	1.029	1.027	1.030	1.014	1.012
Osmolality, mOs/kg	880	817	876	873	835	946	985	370	396
pH (paper)	6.5	6.5	5.0	7	7	5	5	7	6
Creatinine, g/vol	0.80	0.55	0.49	0.42	0.50	0.37	0.69	0.26	0.24
Creatine, g/vol	0.10	0.042	0.050	0.045	0.025	0.035	0.077	0.030	0.034
Urea nitrogen, g/vol	5.07	4.07	3.51	2.63	3.98	2.71	4.79	1.77	1.74
Total nitrogen, g/vol	5.92	4.75	4.00	3.10	4.35	3.10	5.38	1.87	1.92
Hydroxyproline, mg/vol	13.7	11.8	10.1	7.05	6.04	8.40	17.3	7.07	7.14
Uric acid, g/vol	0.38	0.30	0.17	0.20	0.26	0.14	0.22	0.18	0.11
α-Amino acid N, mg/vol	76	45	36	29	41	27	42	35	27
Sodium, mEq/vol	72	86	44	29	61	29	24	45	39
Potassium, MEq/vol	71	31	12	40	34	4.9	6.3	19	16
Chloride, mEq/vol	73	69	40	39	55	23	14	46	39
Magnesium, mEq/vol	2.7	2.5	2.3	1.9	2.7	2.3	2.6	1.6	0.97
Calcium, mEq/vol	4.2	4.1	3.1	2.6	4.7	3.3	3.0	2.6	1.6
Calcium, mg/vol	84	82	62	52	94	66	60	52	31
Phosphate, g/vol	0.51	0.41	0.25	0.12	0.25	0.14	0.43	0.078	0.064
17 hydroxy-corticosteroids	4.2	1.7	1.2	2.4	2.5	1.3	2.1	1.8	1.6

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TABLE 7.2-III.- URINE CHEMISTRIES - Continued

(a) Command Pilot

Determinations Date, 1966 Time, e.s.t.	March 9 18:30	March 9 22:30	March 10 07:30	March 10 12:30	March 10 15:45	March 10 23:40	March 11 07:00	March 11 12:45	March 17 *R+8 hrs
Total volume, ml	240	225	275	305	52	440	270	135	152
Glucose quality	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative
Protein quality	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative
Specific gravity	1.023	1.029	1.029	1.020	1.024	1.023	1.023	1.027	1.023
Osmolality, mOs/kg	892	973	991	895	897	785	879	1001	797
pH (paper)	6	5	5	6	6	6	6	5	5
Creatinine, g/vol	0.41	0.48	0.65	0.45	0.095	0.56	0.75	0.32	2.12
Creatine, g/vol	0.058	0.041	0.039	0.037	0.012	0.053	0.054	0.030	0.22
Urea nitrogen, g/vol . . .	2.59	3.13	5.28	3.17	0.61	3.59	3.82	1.85	14.4
Total nitrogen, g/vol	2.95	3.52	5.45	3.71	0.71	4.55	4.28	2.02	-
Hydroxyproline, mg/vol	8.16	10.8	12.1	6.10	1.35	10.6	18.9	5.40	-
Uric acid, G/vol	0.14	0.14	0.15	0.19	0.033	0.19	0.14	0.095	0.66
α-Amino acid N, mg/vol	36	42	50	40	7.2	44	43	21	-
Sodium mEq/vol	48	44	33	55	8.8	92	36	23	17
Potassium, MEq/vol	16	7.2	6.6	27	3.2	18	4.9	14	85
Chloride, mEq/vol	42	36	25	64	9.1	84	31	30	31
Magnesium, mEq/vol	1.9	3.2	4.6	2.0	0.43	2.4	5.5	1.3	3.9
Calcium, mEq/vol	2.8	4.5	6.0	4.1	0.85	4.4	6.1	1.5	4.5
Calcium, mg/vol	56	90	120	82	17	88	122	30	90
Phosphate, g/vol	0.28	0.32	0.39	0.16	0.037	0.25	0.26	0.10	-
17 hydroxy-corticosteroids	2.0	0.62	2.0	2.6	QNS	2.2	1.5	QNS	-

*R = Recovery

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TABLE 7.2-III.- URINE CHEMISTRIES - Continued

(b) Pilot

Determinations Date, 1966 Time, e.s.t.	March 5 13:00	March 5 21:30	March 6 00:30	March 6 07:00	March 6 14:00	March 6 18:45	March 6 23:30	March 7 06:30	March 9 08:50 - 13:10
Total volume, ml	230	350	135	225	440	455	275	255	455
Glucose, quality	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative
Protein, quality	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative
Specific gravity	1.033	-	1.034	1.034	1.030	1.025	1.022	1.025	1.009
Osmolality, mOs/kg	1013	1026	970	1010	954	635	710	913	285
pH (paper)	5	5	7	5	7	7	6	5	7
Creatinine, g/vol	0.56	0.81	0.29	0.59	0.71	0.34	0.37	0.69	0.18
Creatine, g/vol	0.10	0.20	0.038	0.17	0.16	0.027	0.050	0.051	0.023
Urea nitrogen, g/vol . . .	3.41	5.25	1.81	3.83	4.73	2.89	2.98	4.16	1.37
Total nitrogen, g/vol	3.93	6.16	2.09	4.28	5.37	3.19	3.30	4.67	1.46
Hydroxyproline, mg/vol	13.8	2.17	6.75	10.4	15.8	10.9	11.0	16.8	3.64
Uric acid, g/vol	0.26	0.33	0.13	0.23	0.41	0.24	0.14	0.20	0.10
α-Amino acid N, mg/vol	47	64	30	42	83	60	42	52	26
Sodium, mEq/vol	32	57	28	26	93	87	31	19	31
Potassium, MEq/vol	32	30	7.3	12	67	19	6.6	11	13
Chloride, mEq/vol	42	57	17	27	87	59	30	19	27
Magnesium, mEq/vol	2.0	3.0	1.4	2.6	3.2	2.4	2.5	2.6	0.73
Calcium, mEq/vol	2.4	2.3	1.0	2.1	3.2	2.4	2.0	1.2	1.0
Calcium, mg/vol	48	46	20	42	64	48	40	24	20
Phosphate, g/vol	0.13	0.46	0.14	0.23	0.31	0.27	0.20	0.40	0.038
17 hydroxy-corticosteroids	1.2	2.0	0.52	0.50	1.8	1.0	0.65	1.1	0.80

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TABLE 7.2-III.- URINE CHEMISTRIES - Concluded

(b) Pilot

Determinations Date, 1966 Time, e.s.t.	March 9 15:30	March 9 20:00	March 9 23:00	March 10 06:45	March 10 14:45	March 10 21:45	March 11 06:45	March 17 *R+30 min	March 18 *R+15 hrs
Total volume, ml	485	235	80	265	425	445	205	505	510
Glucose, quality	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative
Protein, quality	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative
Specific gravity	1.014	1.025	1.026	1.022	1.024	1.018	1.027	1.022	1.024
Osmolality, mOs/kg	450	892	890	875	957	666	990	789	815
pH (paper)	7	6	5	5	6	6	6	6	7
Creatinine, g/vol	0.40	0.48	0.23	0.66	0.86	0.52	0.70	1.74	1.68
Creatine, g/vol	0.058	0.033	0.022	0.12	0.13	0.062	0.045	0.18	0.24
Urea nitrogen, g/vol . . .	2.64	2.96	1.14	4.00	5.36	3.54	3.44	10.8	12.3
Total nitrogen, g/vol	-	3.17	1.26	4.78	6.13	3.78	3.95	12.5	14.6
Hydroxyproline, mg/vol	8.73	9.40	3.84	12.7	-	-	-	-	-
Uric acid, g/vol	0.15	0.13	0.037	0.15	0.32	0.27	0.15	0.90	1.20
α-Amino acid N, mg/vol	43	37	16	45	68	45	42	-	-
Sodium, mEq/vol	45	41	8.8	19	85	77	26	151	103
Potassium, MEq/vol	19	10	3.7	8.5	37	21	7.0	61	91
Chloride, mEq/vol	44	39	9.2	19	75	58	25	134	77
Magnesium, mEq/vol	1.6	2.6	1.3	3.6	4.0	3.1	3.7	6.5	6.1
Calcium, mEq/vol	1.6	2.5	0.91	2.9	3.9	2.3	2.4	6.3	2.9
Calcium, mg/vol	32	50	18	58	78	46	48	126	58
Phosphate, g/vol	0.096	0.18	0.075	0.26	0.40	0.37	0.21	-	-
17 hydroxy-corticosteroids	1.4	0.97	QNS	1.6	2.9	1.7	1.2	-	-

*R = Recovery

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TABLE 7.2-IV.- BLOOD CHEMISTRIES

(a) Command Pilot

Determination	Preflight		Postflight	
	March 7, 1966 07:00	March 11, 1966 07:00	March 17, 1966 Recovery + 45 min	March 17, 1966 Recovery + 6 hrs 30 min
Color	Normal	Normal	Moderate hemolysis	Normal
Appearance	Slight precipitation	Slight precipitation	Precipitation	Very slight precipitation
Sodium mEq/l	143	150	146	143
Potassium, mEq/l	4.5	4.6	5.0	4.3
Calcium, mEq/l	4.2	4.5	4.5	3.9
Calcium, mg percent	8.4	9.0	9.0	7.8
Magnesium, mEq/l	2.2	2.2	2.2	2.1
Chloride, mEq/l	103	104	97	92
Phosphate, mg percent	3.14	3.6	3.8	3.7
Glucose, mg percent	134	102	128	101
Blood urea, N mg percent	20	19	24	20
Total protein, gm percent	7.4	7.9	7.9	6.6
Albumin, gm percent	4.5	4.5	4.7	4.1
Uric Acid, mg percent	7.3	6.6	7.5	6.4
Cholesterol, mg percent	226	234	258	226
Total bilirubin, mg percent	0.5	0.3	-	-
Direct bilirubin, mg percent . . .	0.1	0.1	-	-
Alkaline Phosphatase, (BL units)	1.7	1.7	-	-

TABLE 7.2-IV.- BLOOD CHEMISTRIES - Concluded

(b) Pilot

Determination	Preflight		Postflight	
	March 7, 1966 07:00	March 11, 1966 07:00	March 17, 1966 Recovery + 45 min	March 17, 1966 Recovery + 6 hr 30 min
Color	Normal	Normal	Normal	Normal
Appearance	Slight precipitation	Precipitation	Precipitation	Precipitation
Sodium, mEq/l	149	145	149	141
Potassium, mEq/l	4.8	4.4	4.7	4.4
Calcium, mEq/l	4.5	4.7	4.7	4.4
Calcium, mg percent	9.0	9.4	9.4	8.8
Magnesium, mEq/l	2.2	2.2	2.3	2.4
Chloride, mEq/l	102	104	99	99
Phosphate, mg percent	3.41	4.0	4.2	4.0
Glucose, mg percent	85	98	97	85
Blood urea, N mg percent	16	15	19	18
Total protein, gm percent	7.5	7.7	8.5	6.6
Albumin, gm percent	4.6	4.3	4.8	4.1
Uric acid, mg percent	5.7	5.4	5.5	5.0
Cholesterol, mg percent	185	183	175	190
Total bilirubin, mg percent	0.6	0.5	-	-
Direct bilirubin, mg percent	0.2	0.1	-	-
Alkaline Phosphatase (BL units)	1.8	1.3	-	-

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TABLE 7.2-V.- LAUNCH MORNING ACTIVITIES, MARCH 16, 1966

Time, a.m. e.s.t.	Activity
06:30	Crew awake
07:25	Medical examination
07:40	Breakfast
08:30	Began sensoring
08:41	Began suiting
09:16	Began suit purge
09:29	Depart suiting area
09:38	Ingress into spacecraft
10:00	GAATV lift-off
11:41	Gemini Space Vehicle lift-off

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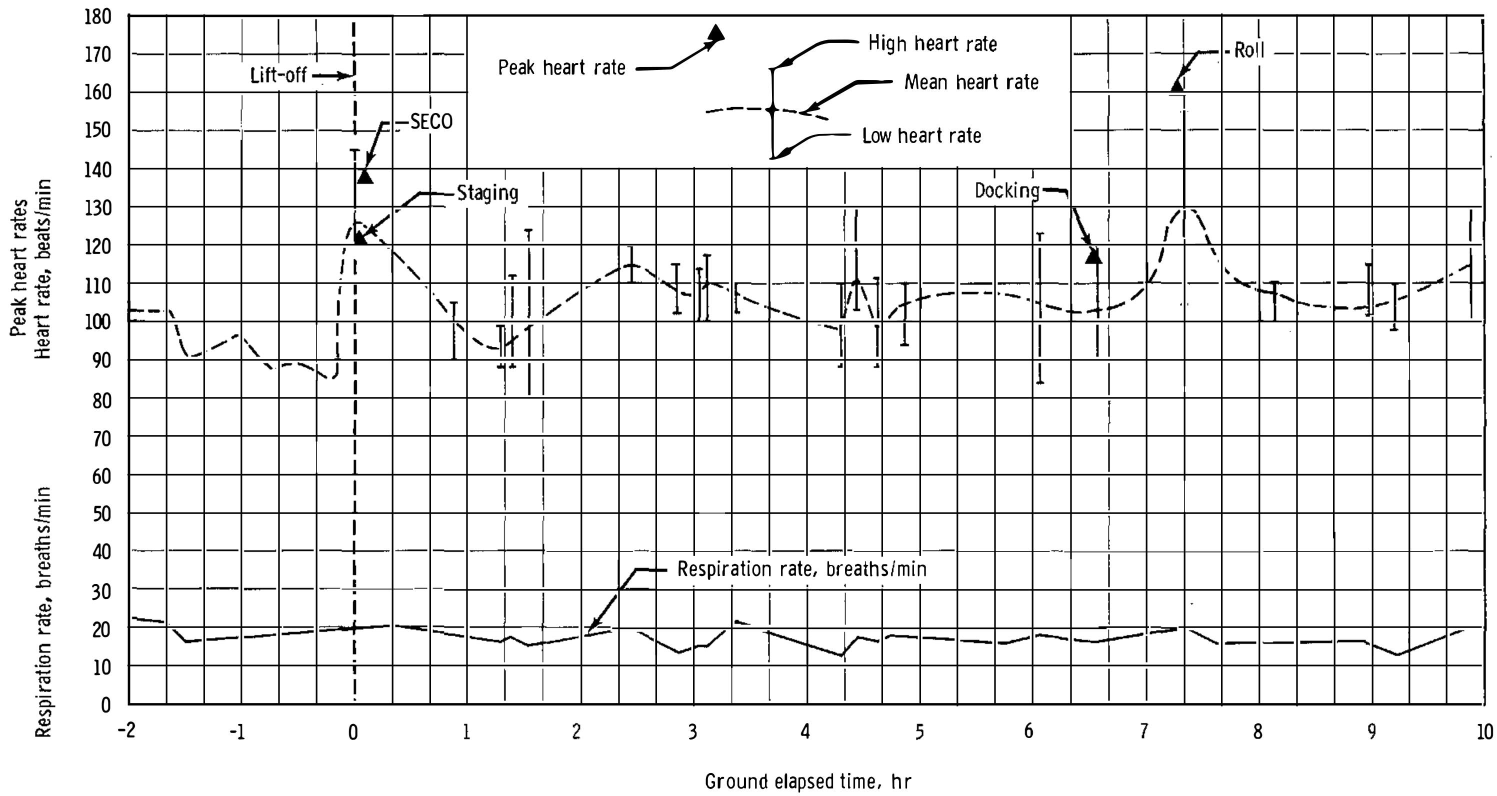

(a) Command pilot.

Figure 7.2-1. - Physiological measurements.

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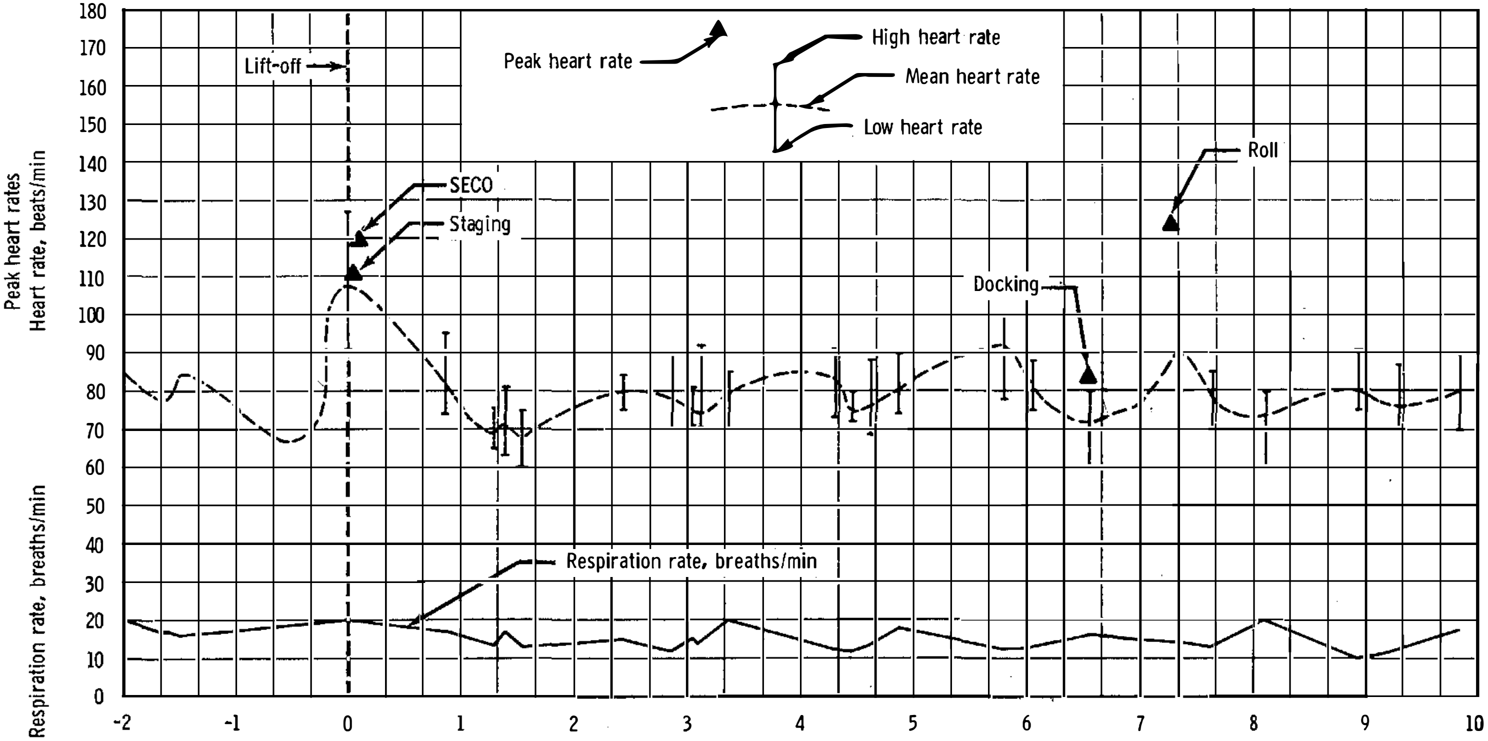

(b) Pilot.

Figure 7.2-1. - Concluded.

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8.0 EXPERIMENTS

Ten scientific, medical, and technological experiments, as listed in table 8.0-I, were planned for the Gemini VIII mission. The purpose of these experiments was to extend man's knowledge of space and to further develop the ability to sustain life in the space environment.

Because the duration of the Gemini VIII mission was only 10 hours instead of the planned 3 days, none of the experiment objectives were fully achieved.

TABLE 8.0-I.- EXPERIMENTS

Experiment number	Experiment title	Principal experimenter	Sponsor
D-3	Mass Determination	Deputy for Technology Headquarters, Air Force Space Systems Division, Los Angeles, California	Department of Defense
D-14	UHF/VHF Polarization	U.S. Naval Research Laboratory, Washington, D.C.	Department of Defense
D-15	Night Image Intensification	U.S. Naval Air Development Center, Johnsville, Pennsylvania	Department of Defense
D-16	Power Tool Evaluation	Air Force Aero Propulsion Laboratory, Wright-Patterson Air Force Base, Dayton, Ohio	Department of Defense
M-5	Bioassays Body Fluids	Space Medicine Branch, Crew Systems Division, NASA-MSC, Houston, Texas	NASA Office of Manned Space Flight
S-1	Zodiacal Light Photography	School of Physics, Institute of Technology, University of Minnesota, Minneapolis, Minnesota	Office of Space Sciences
S-3	Frog Egg Growth	Ames Research Center, Moffett Field, California	Office of Space Sciences
S-7	Cloud Top Spectrometer	National Weather Satellite Center, U.S. Weather Bureau, Suitland, Maryland	Office of Space Sciences
S-9	Nuclear Emulsion	Naval Research Laboratory, Washington, D.C. Goddard Space Flight Center, Greenbelt, Maryland	Office of Space Sciences
S-10	Agena Micrometeorite	Dudley University, Albany, New York	Office of Space Sciences

8.1 EXPERIMENT D-3, MASS DETERMINATION

8.1.1 Objective

The objective of this experiment was to test the technique and accuracy of a direct-contact method of determining the mass of an orbiting object.

The method would have involved accelerating the Gemini Agena Target Vehicle (GATV) by pushing it with the spacecraft. The mass of the GATV would be calculated from the resultant acceleration, spacecraft mass, and thrust level.

8.1.2 Equipment

No special spacecraft or GATV equipment was needed for this experiment.

8.1.3 Procedure

The experiment would have been evaluated by utilizing two independent methods: (1) the flight-crew method (inflight calculations performed by the flight crew), and (2) telemetered method (calculations performed on the ground utilizing telemetered data).

The flight crew would have performed the before-docking portion of the experiment by thrusting the spacecraft for 7 seconds using the aft-firing thrusters. The delta velocity (incremental velocity read from the onboard computer) and delta time (thrusting time over which the delta velocity is measured) with an updated spacecraft mass was to be used to compute the maneuvering thrust:

$$F = Ma = M_G \frac{\Delta V}{\Delta t} \tag{1}$$

where

F = thrust in pounds

M_G = mass of Gemini spacecraft in slugs

V = forward velocity in ft/sec

t = time in seconds.

The after-docking portion of the experiment would have been performed by thrusting the rigidized spacecraft—GATV combination for 25 seconds using the spacecraft aft-firing thrusters. The delta velocity and delta time was to have been taken from the last 7 seconds of the 25-second burn. With the spacecraft mass (M_G) and the maneuvering thrust (F) (equation 1) the mass of the GATV could be computed:

$$M_A = F \frac{\Delta t}{\Delta V} - M_G \qquad (2)$$

where

M_A = mass of the GATV in slugs

The before-docking maneuvering thrust and the after-docking GATV mass would also have been computed on the ground, utilizing telemetered information.

8.1.4 Results

Difficulties encountered with the spacecraft forced termination of the mission prior to any attempt of this experiment.

8.2 EXPERIMENT D-14, UHF/VHF POLARIZATION

8.2.1 Objective

This experiment was to measure the electron content of the ionosphere below the spacecraft by means of a dual-frequency Faraday rotation system utilizing two satellite-borne transmitters operating near 130 and 400 Mc. The principal purpose was to measure the inhomogeneities in the electron content which exist along the orbital path and to gain insight into the structure of the low ionosphere and its temporal variation. The geophysical and temporal correlation analyses which were to have been conducted would have aided in the prediction of the frequency and magnitude of ionospheric disturbances which might have occurred.

8.2.2 Equipment

The D-14 equipment consisted of a continuous-wave (CW) transmitter chassis, diplexer monopole antenna, and a dipole antenna boom, all located in the spacecraft adapter assembly.

8.2.3 Procedures

Each time the spacecraft approached the radio horizon of the ground station at Hawaii and the ground station at Antigua, the flight crew would have been required to position the spacecraft so that the antenna pointed toward the center of the earth. The antenna boom would have been extended prior to transmitting data. During each pass over Hawaii and Antigua, the flight crew would have maneuvered the spacecraft so as to maintain the antenna pointing toward the center of the earth as accurately as possible. After passing beyond the radio horizon or the line-of-sight to the station, the flight crew would have turned off the transmitters.

8.2.4 Results

Difficulties encountered with the spacecraft forced termination of the mission prior to any attempt of this experiment.

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8.3 EXPERIMENT D-15, NIGHT IMAGE INTENSIFICATION

8.3.1 Objective

The objective of this experiment was to aid in the development of a system for night surveillance of the sea and terrestrial features. The system would have been used for night viewing of various objects and for observation of airglow, sea state, and weather data. A three-way comparison would have been made of each scene: (1) one flight crewman looking directly at the scene, (2) the other crewman looking at a television viewing monitor, and (3) by later examining the televised scene as recorded on photographic film.

8.3.2 Equipment

The equipment for this experiment consisted of a television camera, camera control, viewing monitor, recording monitor and photographic camera, and monitor electronics and equipment control. The television camera and camera control were located in the spacecraft adapter assembly and were not recovered.

8.3.3 Procedures

This experiment called for spacecraft flight attitudes such that both the flight crew and the television camera viewed the same earth scene simultaneously. This required that the spacecraft longitudinal axis be approximately normal to the surface of the earth for each of the experiment tasks. In some cases it would have been necessary for the crew to orient the spacecraft in an attitude which would enable a specific target to be acquired in the television camera's field-of-view as the spacecraft approached the zenith of the target. Upon acquiring the target, the flight crew would have controlled the spacecraft's angular rate in order to track the target and record the scene for a period of approximately 60 seconds. Other tasks required only that the spacecraft longitudinal axis be aligned normal to the surface of the earth and also scanned from this attitude to an attitude where the horizon would have been just visible.

8.3.4 Results

Difficulties encountered with the spacecraft forced termination of the mission prior to any attempt of this experiment.

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8.4 EXPERIMENT D-16, POWER TOOL EVALUATION

8.4.1 Objective

The objective of this experiment was to investigate man's capability to perform work under true space conditions. Tests were performed in a KC-135 airplane flying a zero-g trajectory to determine the capability of an unrestrained man to perform work tasks with conventional tools. These tests confirmed beliefs that, due to weightless and resultant frictionless conditions, attempts to transmit torques and forces as tool outputs would be returned to the operator as reactive forces. In attempts to overcome the reactive forces on the operator, two basic methods have been under study: (1) physical restraint attachments such as handholds, belts, and harnesses, to restrain the reactive forces on the man, and (2) tools which internally balance the reactive forces to which the operator would otherwise be subjected.

It is believed that the second method mentioned is the better of the two approaches. A minimum-reaction power tool has been developed and tested, and has proven to be satisfactory. This tool was to have been used in Experiment D-16.

8.4.2 Equipment

The equipment for this experiment consisted of a space power tool, power-tool battery, hand wrench, and a tool restraint box in the spacecraft adapter assembly, plus a knee tether stowed in the crew compartment.

8.4.3 Procedures

The pilot would have egressed from the spacecraft and moved to the tool work panel located on the retroadapter. He would have then attached himself to the work site with the knee tether, removed the minimum-reaction power tool from the restraint box, and performed the specific work tasks on the prearranged work panel. Upon completion of the work tasks, he would have returned to the spacecraft cabin.

8.4.4 Results

Difficulties encountered with the spacecraft forced termination of the mission prior to any attempt of this experiment.

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8.5 EXPERIMENT M-5, BIOASSAYS BODY FLUIDS

8.5.1 Objective

The objective of this experiment was to use hormonal assays to determine the reaction of the flight crew to the stress requirements of space flight. Before and after the flight, two or three daily plasma samples and time urine samples were to be obtained. Urine samples were to be collected during the flight and stored along with a preservative. The crew would record the time and volume of each sample.

8.5.2 Equipment

During flight, urine would be sampled with a urine-sampling and volume-measuring system, which consisted of a valve with a tritiated water injector, a mixing bag, and 24 sample bags.

8.5.3 Procedures

Prior to urination, a precise volume of tritiated water was to be injected into the lines of the valve by a positive displacement pump incorporated into the valve. Urine would wash the tritium into the mixing bag. A sample of the urine containing tritium would then be transferred through the valve from the mixing bag to a sample bag. The sample bag would then be removed and stored. The total volume of each voiding would then be determined postflight by measuring the dilution of the tritium isotope.

8.5.4 Results

The M-5 experiment equipment was not unstowed during this mission, but certain samples were received that will be useful for future analysis and evaluation.

Two postflight blood samples were received from each flight crew member. A used urine-collection device (UCD) was recovered from the command pilot; the pilot did not use his UCD. Two postflight urine samples were received from the pilot and one from the command pilot.

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8.6 EXPERIMENT S-1, ZODIACAL LIGHT PHOTOGRAPHY

8.6.1 Objective

The objective of Experiment S-1 was to obtain photographs of the Zodiacal light, the airglow, and the gegenschein. Long exposures are required to photograph these dim-light phenomena.

8.6.2 Equipment

The experiment equipment consisted of a modified 35-mm camera with mounting brackets to position it in the cabin window.

8.6.3 Procedures

The spacecraft was to have been placed in the proper attitude for pictures which was to have been blunt-end forward (BEF) with the crew looking back along the orbit or, more specifically, looking approximately West at the point where the sun sets. Zero to 10 degrees pitch down would have been acceptable, from where a 40-to-50 degree yaw to the left, or toward South, would have placed the desired portion of the sky in the field of the camera.

The camera was to have been taken from the stowed position and mounted in the cabin window. The camera included an electronic device to program the exposure according to a predetermined sequence. This sequence would have started automatically at sunset. After completion of photography, the camera was to have been removed from the mount and restowed.

8.6.4 Results

Difficulties encountered with the spacecraft forced termination of the mission prior to any attempt of this experiment.

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8.7 EXPERIMENT S-3, FROG EGG GROWTH

8.7.1 Objective

The objectives of Experiment S-3 were to determine the effect of weightlessness on the ability of the fertilized frog egg to divide normally and to differentiate and form a normal embryo.

8.7.2 Equipment

The experiment was contained in two identical packages, one of which was mounted on each hatch of the spacecraft. Each package had four chambers containing frog eggs in water, with a partitioned section containing a fixative (5-percent formalin). Each package was insulated and contained temperature-control systems for both heating and cooling in order to maintain an experiment temperature of close to 70° F. Electrical power was obtained from the spacecraft Electrical System. The experiment was actuated by handles provided on the outside of each package. These handles and a switch for the heating element were manipulated by the adjacent flight crewman, either on ground command or according to a predetermined schedule. Identical hardware was used for control experiments on the ground.

8.7.3 Procedure

Eggs were obtained from several dozen female frogs (Rana pipiens) by injection of frog pituitary glands about 48-hours prelaunch, in order to induce ovulation. The best of these eggs (from two females) were selected for flight and fertilized by immersion in a sperm suspension made by macerating frog testes in pond water. The fertilized eggs were then removed to a 43° F cold room and placed in about 10 cc of pond water in the experimental chambers. The fixative was placed behind leak-proof partitions in the chamber. Each chamber received from 5 to 8 eggs, so that a total of 52 eggs were carried in the spacecraft. Two sets of controls were set up in identical hardware on the ground. The first was to run simultaneously with the flight, and the second was delayed about 2 hours so that changes in temperature experienced by the flight experiment could be duplicated on the ground more precisely than in the simultaneous control. Since telemetered temperatures were not received instantaneously, such a delayed control was necessary.

The flight experiment was placed in the spacecraft about 4 hours before launch. By keeping the fertilized eggs at about 43° F until this time, the first division of the eggs was retarded. It was hoped that this pre-cooling of the eggs would be sufficient to retard first cleavage until the zero-g phase of the flight. At approximately 40 minutes ground elapsed time (g.e.t.), the pilot was to turn the first handle on the right-hand experiment package, which would inject the fixative into the egg chamber, killing the eggs in that chamber and preserving them for microscopic study on recovery. A second handle was to be turned at 2 hours 10 minutes g.e.t., which would fix the remaining two chambers at about the eight-cell stage. Two chambers in the left-hand package were to be fixed at the end of the 3-day flight, just before reentry. The last two chambers were to remain unfixed and those embryos returned alive. All eggs and embryos were to be studied upon recovery for gross morphological abnormalities in cleavage planes and differentiation. Histological examination and electron microscopy were also anticipated.

8.7.4 Results

Although the cabin temperatures were considerably above the predicted 70° F, the temperature control system on the experiment packages was sufficient to retard first cleavage until the zero-g phase of the flight. Thus, the first fixation, at 40 minutes g.e.t., was successful in stopping development between first and second cleavage. The flight crew were also able to perform the second activation at 2 hours 25 minutes g.e.t. (15 minutes late) which was at about the eight-cell stage of development. Because of difficulties with the spacecraft, the flight was terminated after about 10 hours and the remainder of the experiment could not be accomplished. Thus, only the first half of the experiment was completed successfully. The fixed eggs in the first four chambers appeared identical in all respects when compared to the controls. The cleavage planes appeared normal and to have been proceeding on schedule. Histological and electron microscope study may show some abnormalities but this is not anticipated. The absence of a gravitational field does not appear to have any effect on the ability of the frog egg to divide normally during its early stages, when such an effect would be most likely to occur because of the large density gradient in these cells.

8.7.5 Conclusions

In spite of the fact that the frog egg is known to orient itself with respect to gravity during its very early development, a gravitational field is apparently not necessary for the egg to divide normally. Whether this independence from gravity applies to differentiation and morphological changes in later stages was not demonstrated because of the short duration of the flight. Whether the egg will divide normally if it is fertilized in zero-g, so that the egg never has a chance to become oriented with respect to gravity, is also unanswered at this time. It is hoped that these two very important questions can be answered in later flights.

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8.8 EXPERIMENT S-7, CLOUD TOP SPECTROMETER

8.8.1 Objective

The objective of this experiment was to use a simple hand-held spectrograph to investigate the possibility of using satellites to measure cloud-top altitudes.

8.8.2 Equipment

The equipment consisted of a spectrograph fitted with a 35-mm camera body.

8.8.3 Procedures

The spectrometer would have been removed from stowage and the shutter released. This would waste one frame of film but it would have placed the shutter mechanism in its proper position. The entrance aperture of the spectrometer was located 4 inches to the left of the view finder. The exposure times for the spectrograph were 1/4 and 1/8 of a second. One exposure would have been made of sunlight being reflected from a 6-inch by 6-inch card.

For each picture a voice report would have been made giving:

(a) The ground elapsed time

(b) A brief description of cloud formation (cirrus, stratus, etc.)

(c) An estimate of the azimuth angle from the North or from the sun

(d) An estimate of the angle of depression between horizon and the cloud.

8.8.4 Results

Difficulties encountered with the spectracraft forced termination of the mission prior to any attempt of this experiment.

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8.9 EXPERIMENT S-9, NUCLEAR EMULSION

8.9.1 Objective

The objective of this experiment was to contribute new knowledge to the fields of space science, astrophysics, and high-energy-particle physics. Cosmic rays provide a means for investigating nuclear interactions and electromagnetic acceleration and transmission mechanisms within the galaxy, and possibly beyond.

8.9.2 Equipment

The experiment equipment consisted of a nuclear emulsion package which was stowed in the spacecraft retrograde adapter section during launch and orbit.

8.9.3 Procedures

A major procedural requirement in the conductance of this experiment would have been to keep the spacecraft attitude in the proper orientation; however, attitude needed to be held only within ±10 degrees. The horizon-scanner mode of attitude control would have been sufficient for this accuracy. It would also have been necessary to orient the spacecraft so that the top face of the emulsion package laid in a plane which was normal (±10 degrees) to the earth's average magnetic field vector ($\bar{f}$) anytime the spacecraft was in the vicinity of the South Atlantic magnetic anomaly. This orientation will be referred to as the anomaly orientation.

Operations performed or to have been performed by the flight crew were as follows:

(a) The hinged cover, used to protect the experiment during launch, was opened remotely.

(b) The experiment was switched from OFF to mode 1 operation at a specified time after insertion into orbit. Further instructions for turning the experiment on and off were to have been provided as the mission plan developed.

(c) The spacecraft was to have been put into anomaly orientation each time it passed through the South Atlantic anomaly.

(d) The mode 1 operation was monitored by real-time telemetry. No mode 2 operation was planned unless mode 1 malfunctioned.

(e) If the crew found it no longer possible to maintain the exposure orientation, they were to have moved the switch to the mode 2 position, left it there for at least 15 seconds, and then returned it to the OFF position. This operation would have advanced the stack to the next background position. When exposure orientation was again possible for a period of at least 30 minutes, the switch was to have been returned to the mode 2 position for 15 seconds, and then reset to the OFF position, again moving the package to the next data position.

(f) The crew was requested to report the times at which all of the preceding actions were taken.

(g) During the planned EVA, the emulsion package would have been removed from the retroadapter and placed in the insulated container in the cabin.

8.9.4 Results

Telemetry channels were functioning satisfactorily prior to lift-off. At 00:23:00 g.e.t., the experiment was turned on. At 01:40:00 g.e.t., telemetry was indicating proper translations of the moving stack. At 03:10:00 g.e.t., telemetry indicated that the stack was still stepping properly and had completed approximately 200 of the 2000 steps. Controlled temperatures of this experiment were satisfactorily maintained between 40° and 46° F. At 06:19:00 g.e.t., telemetry indicated that the stack had moved through about 17.8 percent (360 steps) of its full travel (2000 steps) and was still functioning according to design, and that the temperature control was satisfactory.

Difficulties encountered with the spacecraft forced termination of the mission prior to EVA, and, as a result, the S-9 experiment was not recovered.

8.10 EXPERIMENT S-10, AGENA MICROMETEORITE COLLECTION

8.10.1 Objective

The objective of this experiment was to expose specially prepared and polished surfaces to the small-particle flux of the upper atmosphere and near-earth space environment, in an effort to gain useful knowledge of the impact and cratering properties of these small particles in space.

8.10.2 Equipment

The equipment consisted of a micrometeorite collector located on the Gemini Agena Target Vehicle (GATV).

8.10.3 Procedures

During EVA, the micrometeorite unit, located on the GATV, would have been opened to expose the collecting surface. If an attempt to rendezvous with the Gemini VIII GATV during the Gemini X flight had not been planned, the micrometeorite unit could have been retrieved, placed in a plastic bag, and stowed onboard the Gemini VIII spacecraft for reentry.

8.10.4 Results

Difficulties encountered with the spacecraft precluded any EVA or full experiment deployment. The experiment package remains on the GATV for possible recovery during future missions.

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9.0 CONCLUSIONS

The overall performance of the two launch vehicles, the Gemini-Agena Target Vehicle, the flight crew, and mission support was satisfactory for all phases of the mission that were accomplished. The spacecraft performance was very satisfactory during launch, rendezvous, docking, and reentry; however, about one-half hour after docking, an anomaly occurred in the circuitry for the yaw-left/roll-left thruster of the Orbital Attitude and Maneuver System that finally required activation of the Reentry Control System to regain control of the spacecraft. With less than one-half of the Reentry Control System fuel remaining after this incident, a decision was made to terminate the mission and land in one of the early planned landing areas. The performance of the Gemini Agena Target Vehicle propulsions systems was satisfactory and eight restarts of the Primary Propulsion System were successfully accomplished. The flight contributed to the knowledge of manned space flight, especially in the area of rendezvous, docking, and controlled reentry operations. The mission demonstrated adequate performance of the flight crew and of the ground operations personnel and associated equipment under emergency conditions.

The following conclusions were obtained from data evaluation and crew observations of the Gemini VIII mission.

1. The Target Launch Vehicle operated satisfactorily and placed the Gemini Agena Target Vehicle in the required coast-ellipse trajectory for a nominal insertion into orbit.

2. Performance of the Gemini Launch Vehicle with the modified GEMSIP injector on the second-stage engine was satisfactory in placing the spacecraft in an acceptable orbit for a nominal rendezvous with the orbiting target vehicle.

3. Voice communications were excellent throughout the Gemini VIII mission. The difficulty that the crew had in contacting recovery forces is attributed to the fact that the one recovery aircraft near the spacecraft carried a single UHF transceiver. The necessity for communications with the pararescuemen and with other elements of the recovery forces on UHF prevented continuous monitoring of the spacecraft transmitting frequency.

4. The Fuel Cell Power System operated satisfactorily. The difference in load sharing between the two sections may be attributed to the early first activation of section 2, the procedures used during the second activation of section 2, or both.

5. The uncontrolled firing of the yaw-left/roll-left thruster in the Orbital Attitude and Maneuver System resulted from a short circuit to ground at some point between the positive side of the solenoid coils of the thrust-chamber valves and the common contact of the relay that selects primary or secondary valve drivers in the attitude control electronics.

6. Although a substantial portion of the Reentry Control System propellants were used for spacecraft stabilization during the Orbital Attitude and Maneuver System anomaly, Reentry Control System propellant depletion did not occur with the control system in the reentry rate-command mode until after the drogue parachute had been deployed and had disreefed. This confirms that this control mode can be used to perform accurate reentries with low fuel usage.

7. Docking of the Gemini spacecraft with the Gemini Agena Target Vehicle proved to be a relatively simple task. The stability of the docked and rigidized vehicles for the 27-minute period prior to the spacecraft control problem proved to be excellent.

8. The performance of the Gemini Agena Target Vehicle was satisfactory for this mission. The performance of the Gemini Agena Target Vehicle propulsion systems was nominal for the eleven firings. The multiple-restart capability of the Primary Propulsion System was demonstrated to be satisfactory. The excellent performance of the Communications and Command System was also demonstrated by the correct execution of over 5100 commands without a malfunction.

9. The yaw velocity errors sustained during the Primary Propulsion System maneuver of the Gemini Agena Target Vehicle were caused by an offset center-of-gravity and low dynamic gains in conjunction with a long time constant in the lead-lag circuits of the control system. This error resulted in a varying amount of unexpected out-of-plane velocity components.

10. A very accurate reentry was made into the Western Pacific landing area, affording immediate on-scene assistance from a recovery aircraft.

11. The world-wide recovery forces demonstrated outstanding capability and provided excellent support when faced with the unexpected recovery of the spacecraft and crew in a secondary landing area.

10.0 RECOMMENDATIONS

The following recommendations are made as a result of engineering analyses and crew observations of the Gemini VIII mission.

1. A complete vacuum fill of the drinking-water system should be utilized.

2. The crew should maintain a flight log of the exact time they find open circuit breakers, malfunction lights, switches found in unexpected positions, and similar unexpected events. This will enable a more detailed postflight evaluation of any anomalies.

3. The spacecraft should be modified so that the crew can easily remove all power from the Orbital Attitude and Maneuver System at the onset of unexpected or unexplainable rates.

4. Procedures should be reviewed, and revised if necessary to prevent the spacecraft from becoming uncontrolled as a result of an incident such as occurred on Gemini VIII. At the onset of any unexplainable rate and where circumstances permit, all power should be removed from the control system and an orderly troubleshooting procedure followed. A study should be conducted to determine the best control mode to be used under circumstances where rates must be brought under control as quickly as possible.

5. Emphasis should be placed on simplifying restowage of equipment during the preretrofire period, especially those items which are heavy or bulky. This should include development of backup procedures for restowage of materials which, under normal circumstances, would be jettisoned during extravehicular activity.

6. The rendezvous radar test prior to the coelliptic maneuver should not be performed because it interferes with preparation for the maneuver and the required information is obtained from normal radar operation between the coelliptic maneuver and terminal phase initiate.

7. The postlanding checklist should be reviewed and revised to call out all items to be accomplished, rather than items not to be accomplished.

8. The suit harnesses and attaching life vests should be adequately coded or marked to enable the quickest possible installation prior to retrofire in case of a need for an early termination of the mission.

9. The Gemini Agena Target Vehicle Primary Propulsion System start-sequence B should be used for future operations to reduce Attitude Control System gas usage.

10. Methods should be investigated for reducing the time required for unstowing and preparing food.

11. The terminal angle of the roll program, as indicated on the Flight Director Indicator, should be incorporated in the T - 3 minute information to the crew.

12. The procedures used on Gemini VI-A and VIII to null the residual desired velocity changes should be simplified and should include only the significant axes.

13. An investigation should be conducted concerning the use of a directed vent as an integral part of the suit neck dam in order to prevent ballooning while the helmet is removed.

14. A thorough study and subsequent testing should be implemented to insure the capability to close and latch the centerline stowage door.

15. A study should be conducted to determine if the present procedure of aligning the platform before each rendezvous maneuver is necessary.

16. Recovery personnel should establish communications with the flight crew as soon as practical after spacecraft landing and should report the crew's status to the Recovery Control Center in the Mission Control-Houston as soon as possible.

11.0 REFERENCES

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2. Gemini Mission Evaluation Team: Gemini Program Mission Report GT-2, Gemini 2. MSC-G-R-65-1, NASA Manned Spacecraft Center, Feb. 1965.

3. Gemini Mission Evaluation Team: Gemini Program Mission Report GT-3, Gemini 3. MSC-G-R-65-2, NASA Manned Spacecraft Center, April 1965.

4. Gemini Mission Evaluation Team: Gemini Program Mission Report, Gemini IV. MSC-G-R-65-3, NASA Manned Spacecraft Center, June 1965.

5. Gemini Mission Evaluation Team: Gemini Program Mission Report, Gemini V. MSC-G-R-65-4, NASA Manned Spacecraft Center, August 1965.

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9. McDonnell Aircraft Corp.: Gemini Spacecraft Number 8 Performance/Configuration Specification. Report A900-8. (Revision E to be published in the as-launched configuration on or about May 1, 1966.)

10. NASA Manned Spacecraft Center: Project Gemini Orbital and Rendezvous Report for Gemini VIII. MSC Internal Note 66-FM-8, Feb. 8, 1966.

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12. Lockheed Missiles and Space Co.: Pre-injection Trajectory and Data for Gemini 8. Enclosure (A) to LMSC/A794338.

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14. NASA Manned Spacecraft Center: Gemini Spacecraft/Launch Vehicle Interface Specification and Control Document. Report ISCD-1, Nov. 22, 1963. (As revised)

15. Lockheed Missiles and Space Co.: SLV-3/Agena Interface Specification. IMSC No. 1417158, Aug. 10, 1965.

16. NASA Manned Spacecraft Center: Gemini Agena Interface Specification and Control Document. Report ISCD-2, April 20, 1965. (As revised)

17. McDonnell Aircraft Corp.: Gemini Spacecraft Postflight RCS Deactivation. SEDR F-399, March 14, 1966.

18. McDonnell Aircraft Corp.: Postflight Evaluation Procedures for Spacecraft 8. SEDR F499-8, March 2, 1966.

12.0 APPENDIX

12.1 VEHICLE HISTORIES

12.1.1 Spacecraft Histories

The spacecraft history at the contractor's facility in St. Louis, Missouri, is shown in figures 12.1-1 and 12.1-2. The spacecraft history at Cape Kennedy, Florida, is shown in figures 12.1-3 and 12.1-4. Figures 12.1-1 and 12.1-3 are summaries of activities with emphasis on spacecraft systems testing and prelaunch preparation. Figures 12.1-2 and 12.1-4 are summaries of significant, concurrent problem areas.

12.1.2 Gemini Launch Vehicle Histories

The Gemini Launch Vehicle (GLV) history and significant manufacturing activities at the contractor's facilities in Denver, Colorado, and in Baltimore, Maryland, are presented in figure 12.1-5. The GLV history at Cape Kennedy, Florida, is presented in figure 12.1-6. This figure also includes problem areas which were concurrent with GLV normal launch-preparation activities.

12.1.3 Gemini Agena Target Vehicle and Target Docking Adapter

Histories at the contractor's facility for the Gemini Agena Target Vehicle (GATV) at Sunnyvale, California, and at the contractor's facility for the Target Docking Adapter (TDA) at St. Louis, Missouri, are shown in figures 12.1-7 and 12.1-8, and at Cape Kennedy in figures 12.1-9 and 12.1-10. Figures 12.1-7 and 12.1-8 show significant manufacturing activities and concurrent problem areas. Figure 12.1-9 is a summary of activities with emphasis on GATV and TDA testing and prelaunch preparation. Figure 12.1-10 is a summary of GATV and TDA concurrent problem areas.

12.1.4 Target Launch Vehicle

Target Launch Vehicle (TLV) histories at the contractor's facility in San Diego, California, are shown in figure 12.1-11, and at Cape Kennedy, Florida, in figure 12.1-12. Both figures include systems testing and concurrent problems.

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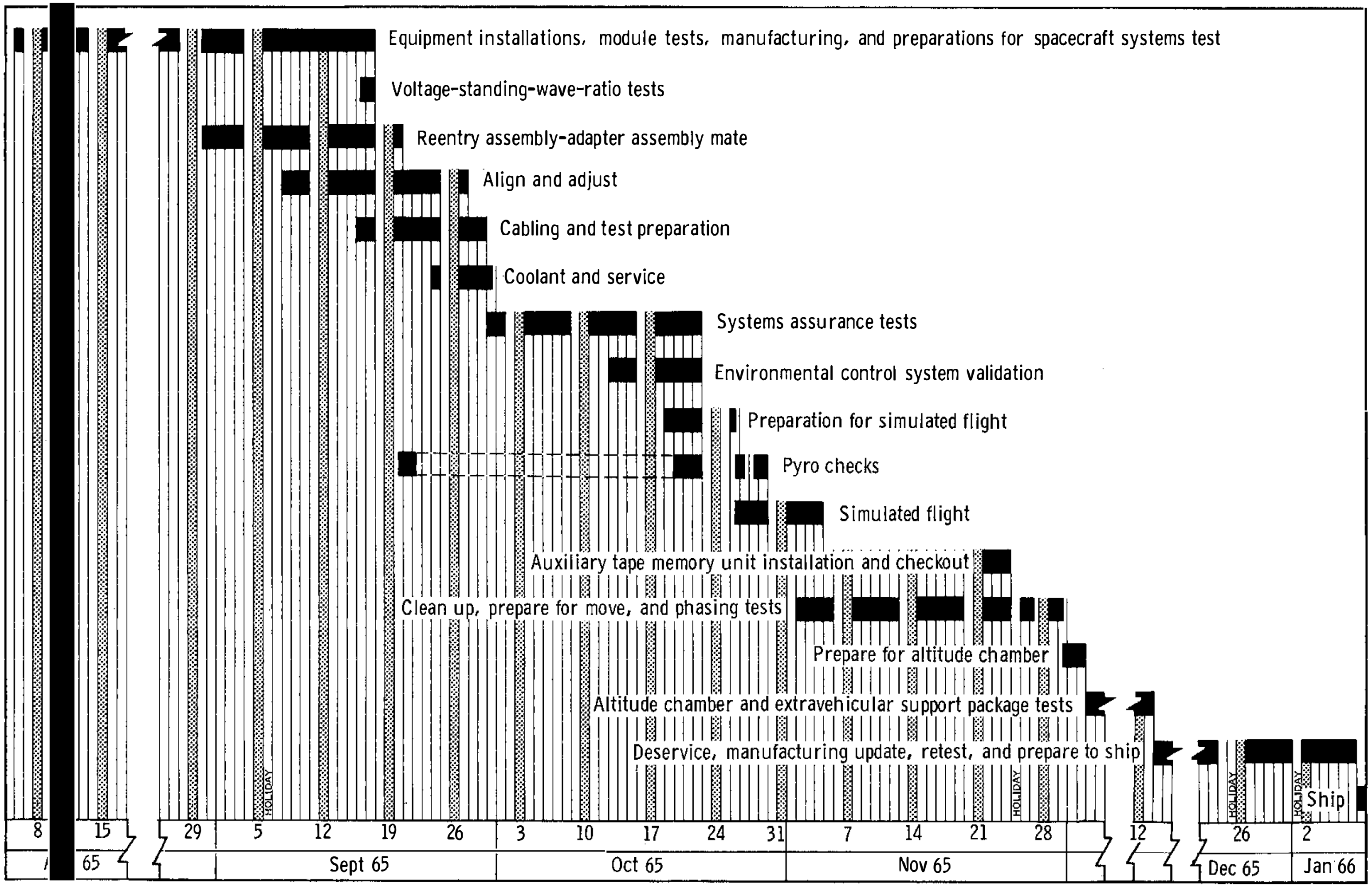

Figure 12. 1-1. - Spacecraft 8 test history at contractor facility.

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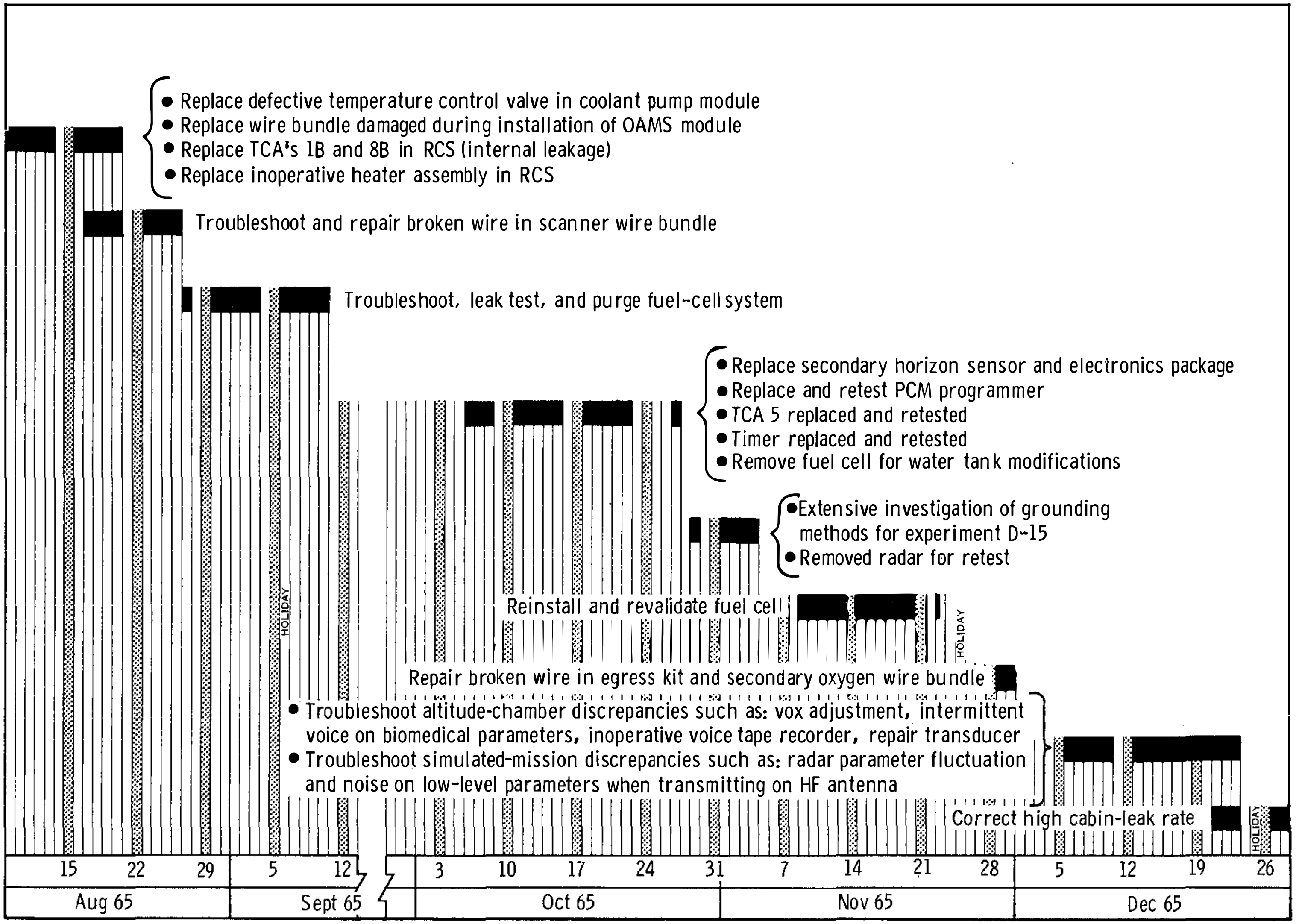

Figure 12.1-2. - Spacecraft 8 significant problems at contractor facility.

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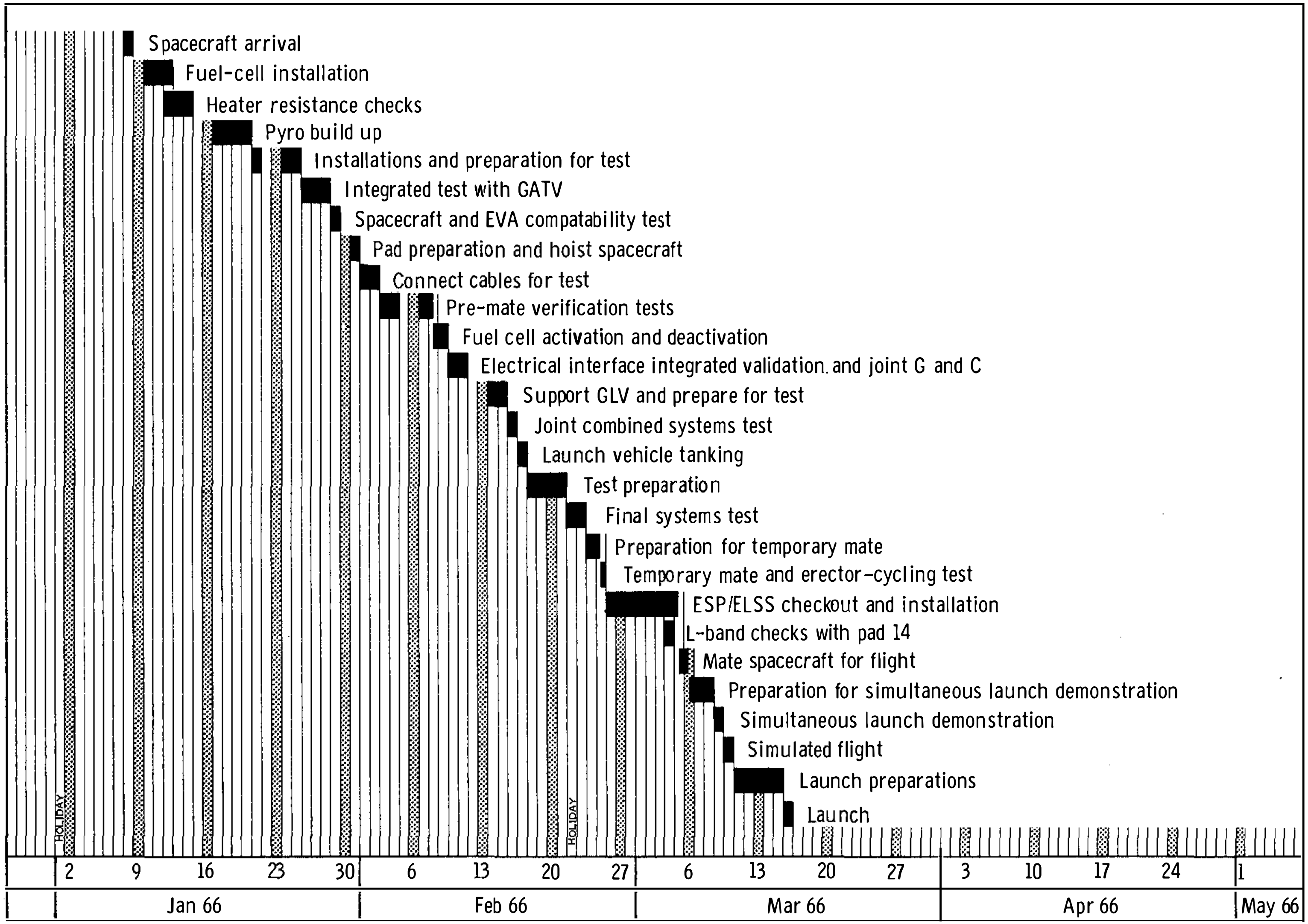

Figure 12.1-3. - Spacecraft 8 test history at Cape Kennedy.

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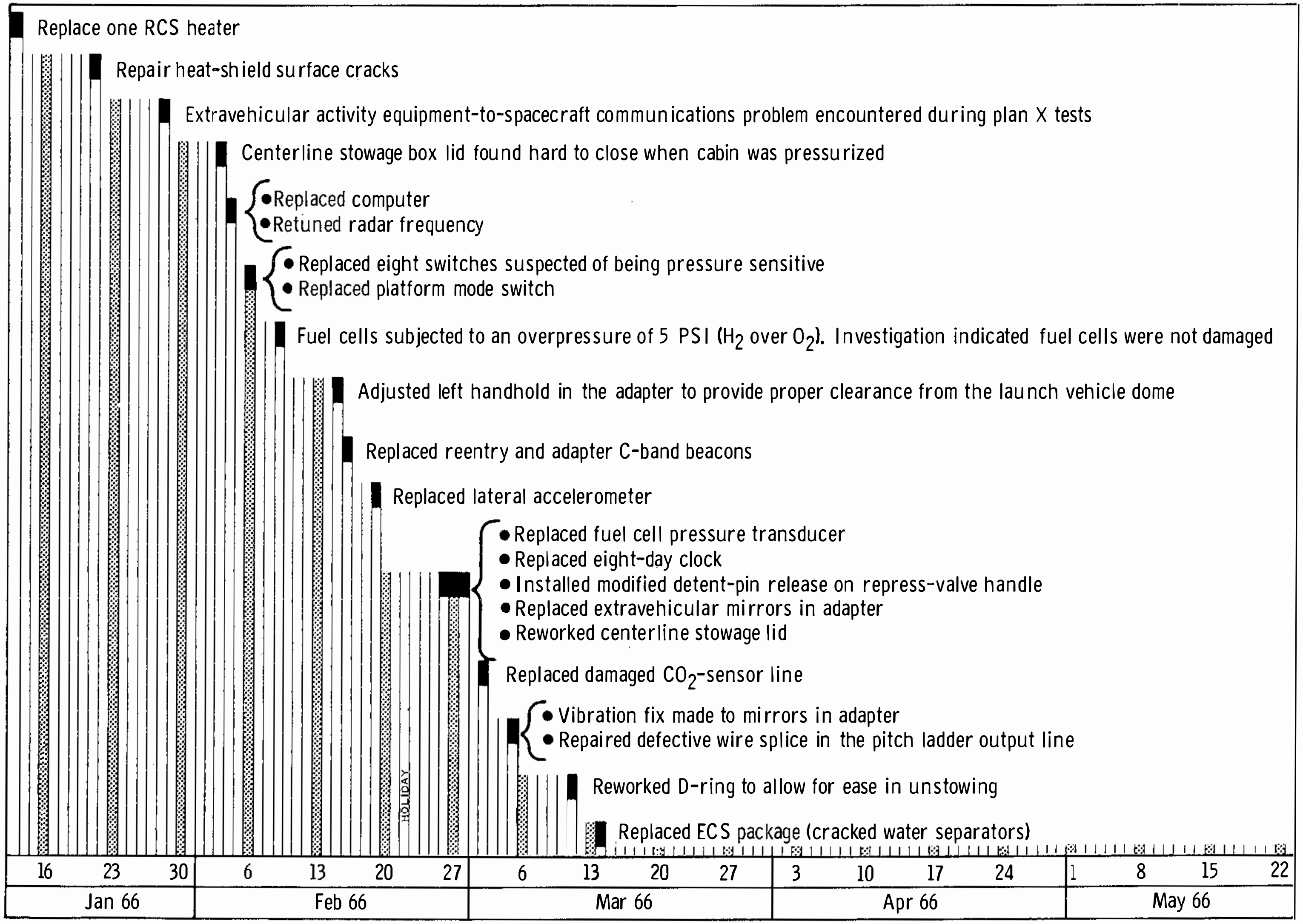

Figure 12. 1-4. - Spacecraft 8 significant problems at Cape Kennedy.

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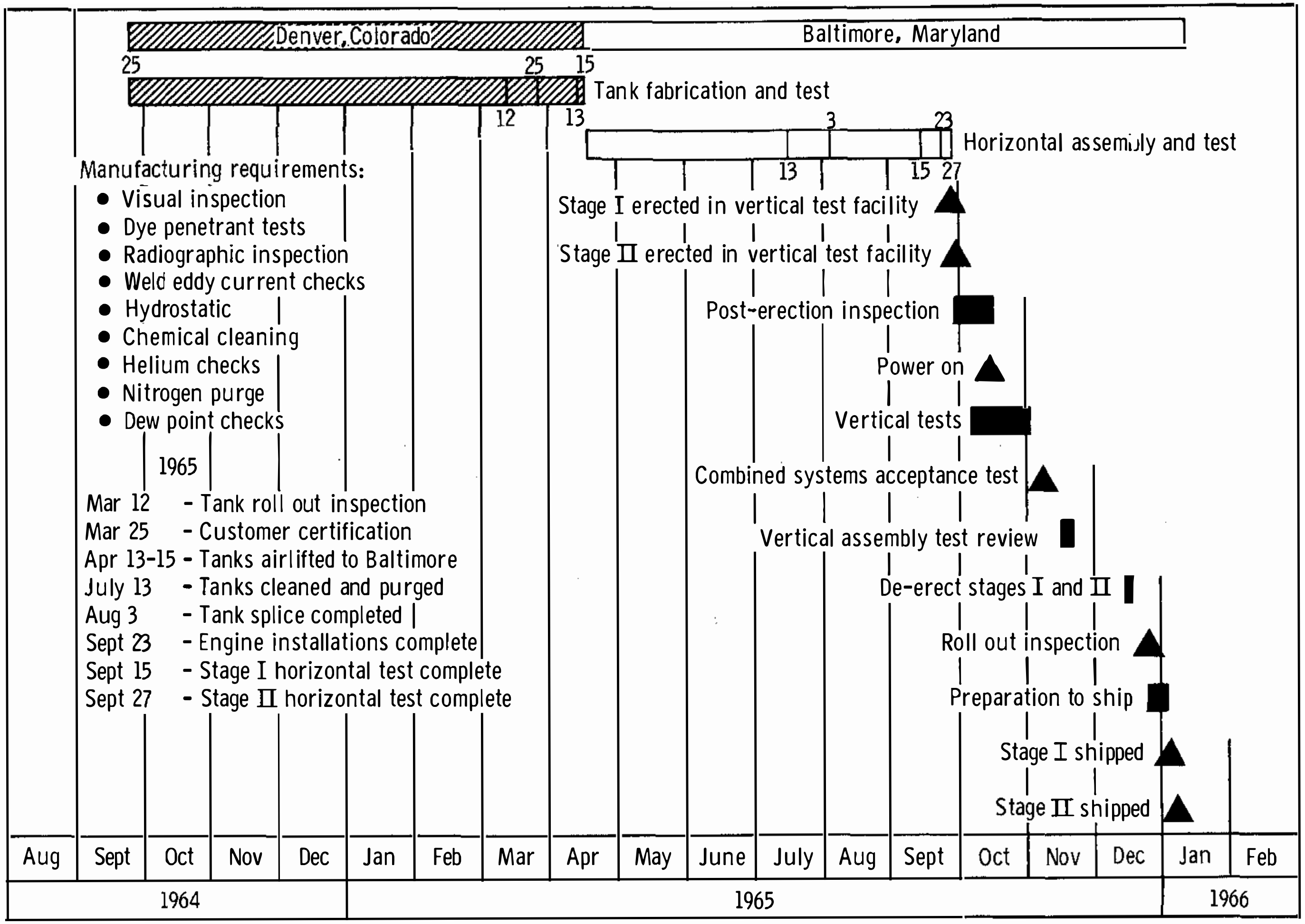

Figure 12.1-5. - GLV-8 history at Denver and Baltimore.

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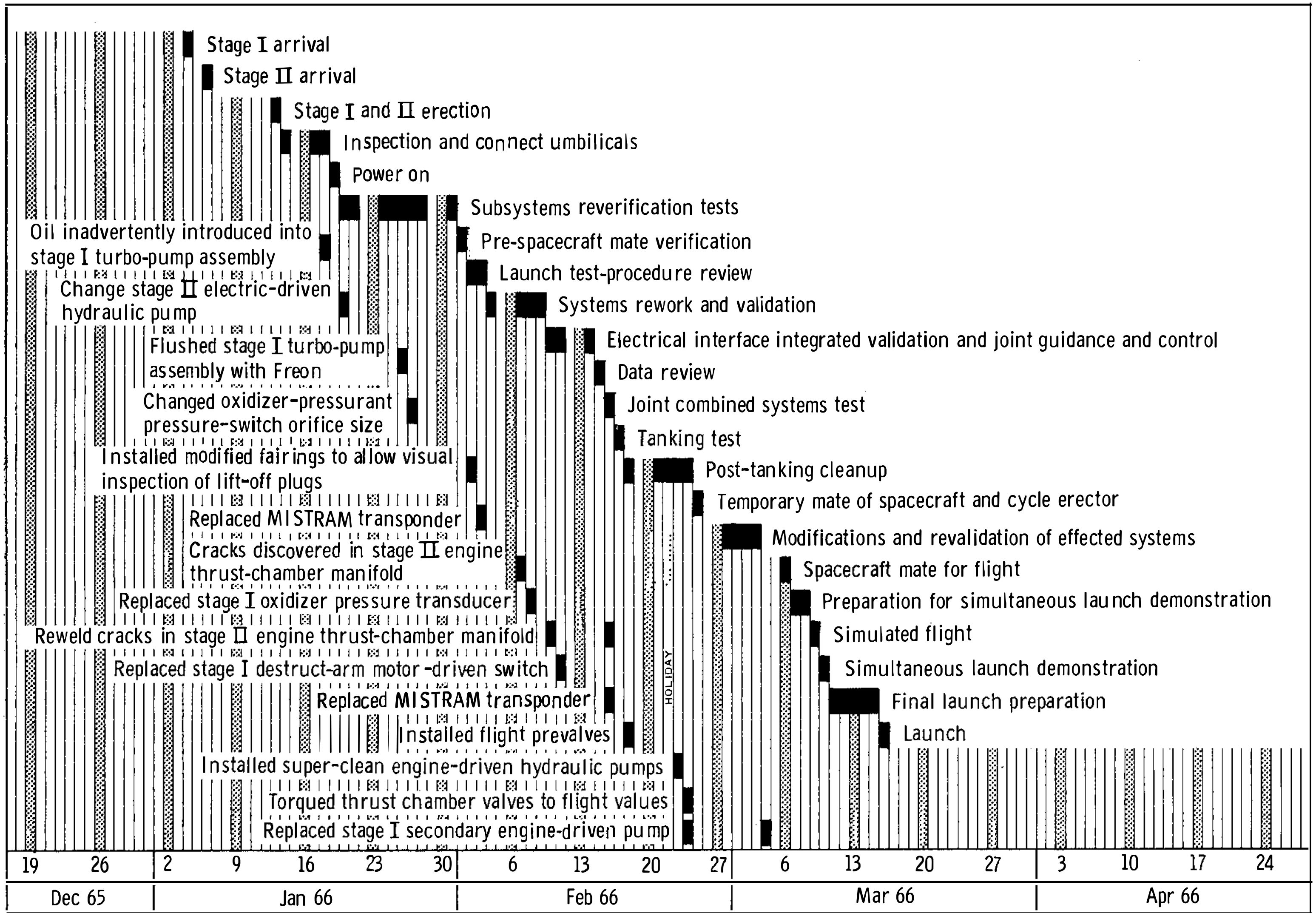

Figure 12.1-6. - GLV-8 History at Cape Kennedy.

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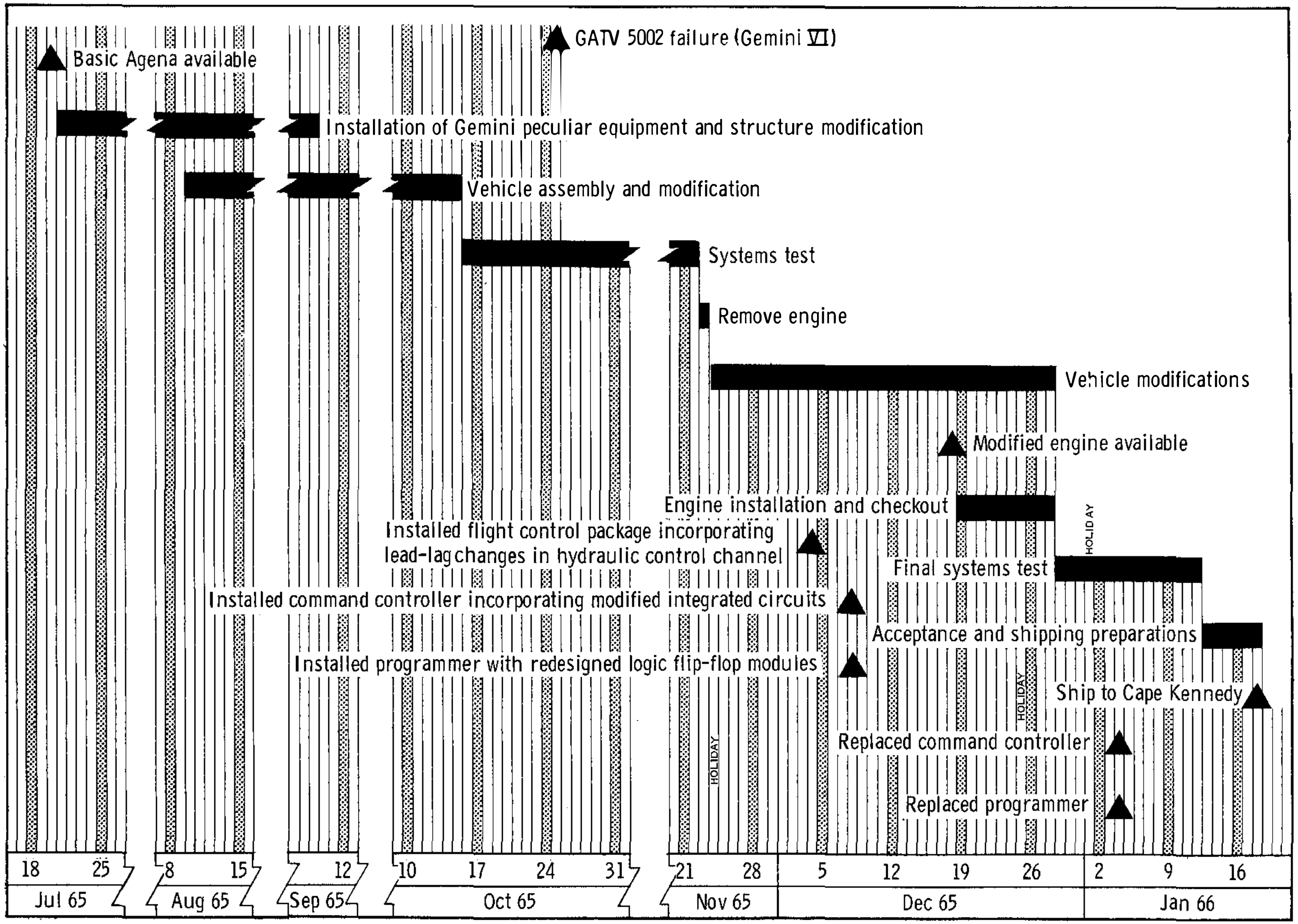

Figure 12.1-7. - GATV 5003 history at contractor facility.

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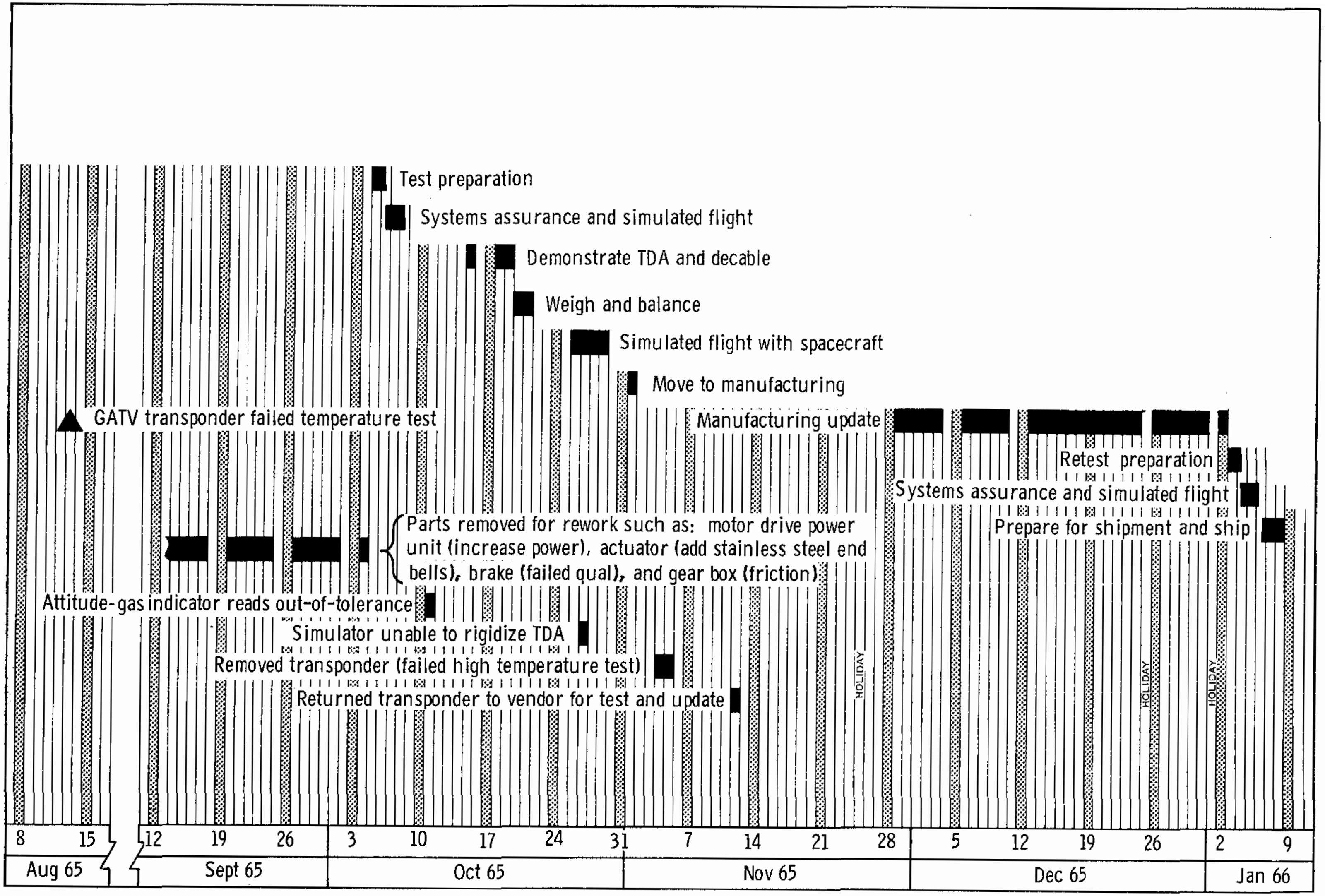

Figure 12.1-8. - TDA 3 test history and significant problem areas at contractor facility.

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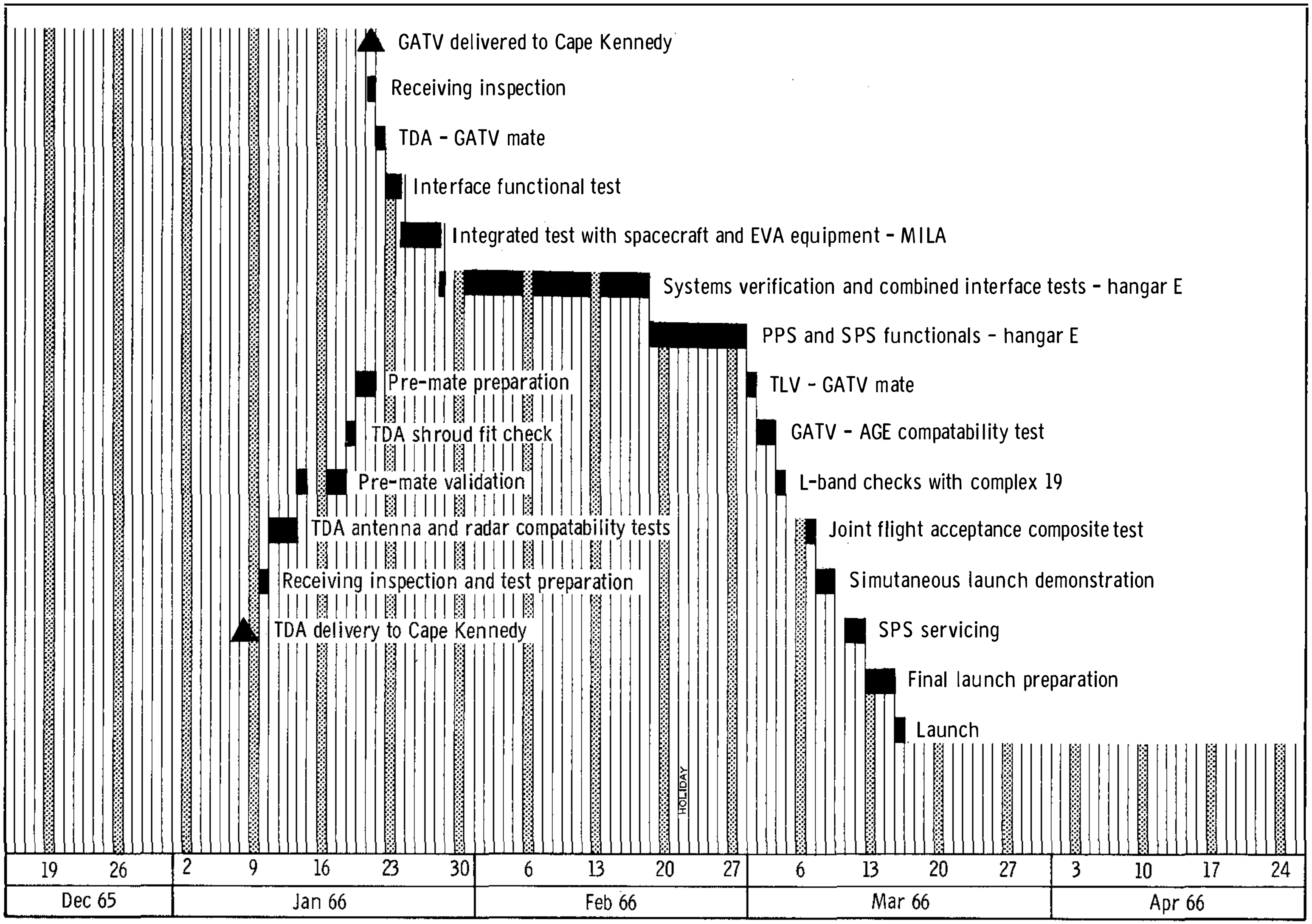

Figure 12.1-9. - GATV 5003 and TDA 3 test history at Cape Kennedy.

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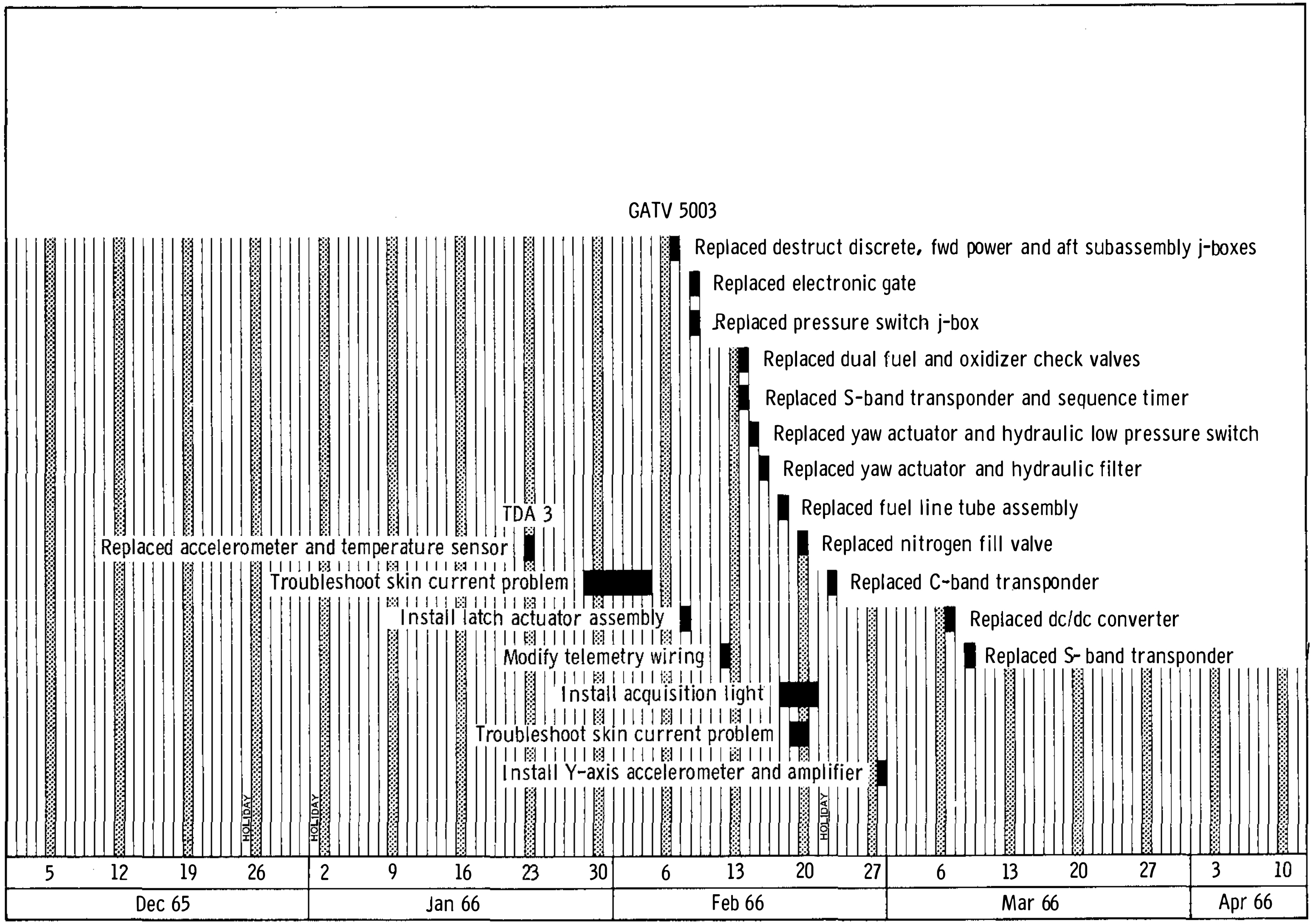

Figure 12. 1-10. - GATV 5003 and TDA 3 problems at Cape Kennedy.

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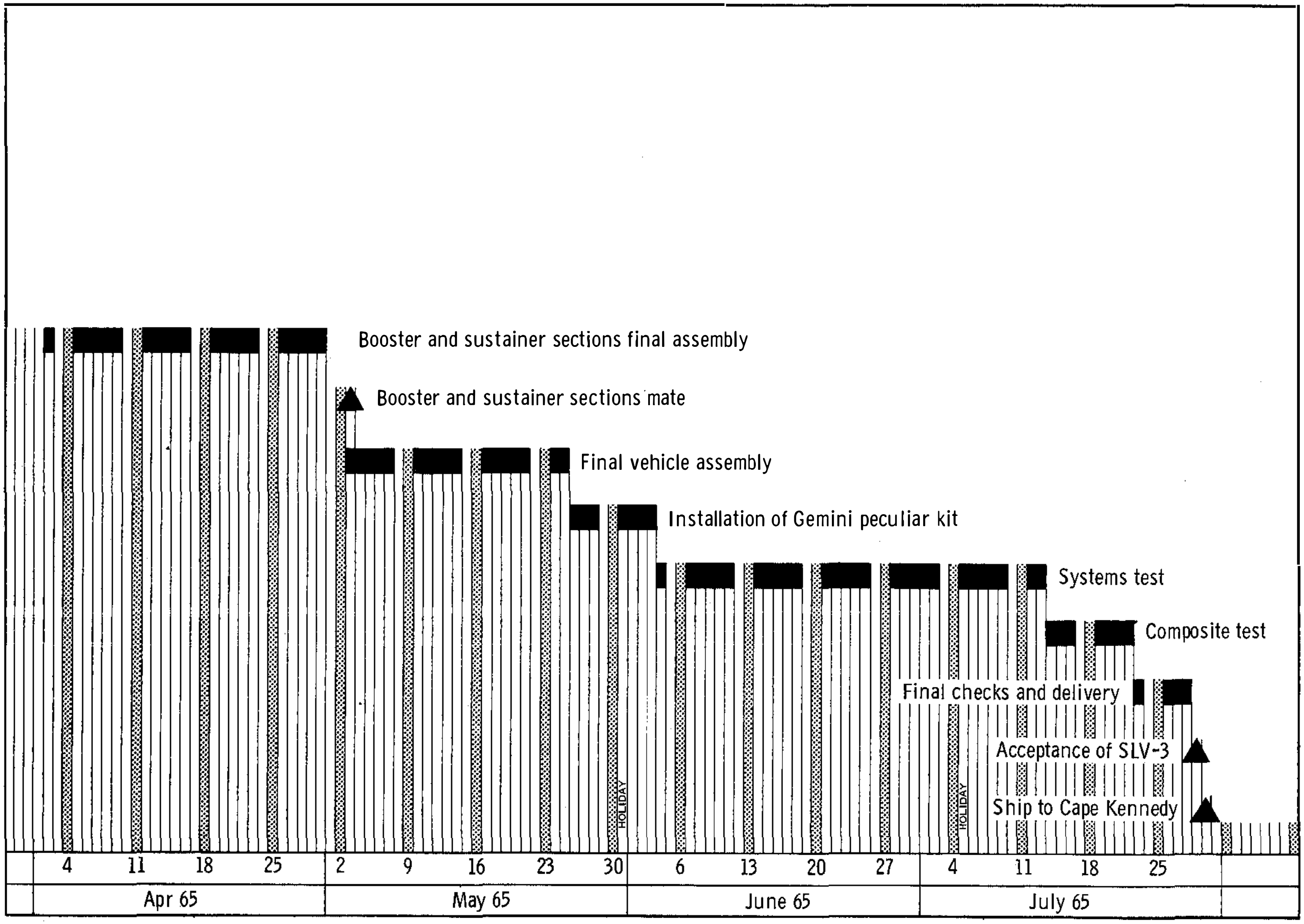

Figure 12.1-11. - SLV 5302 history at contractor facility.

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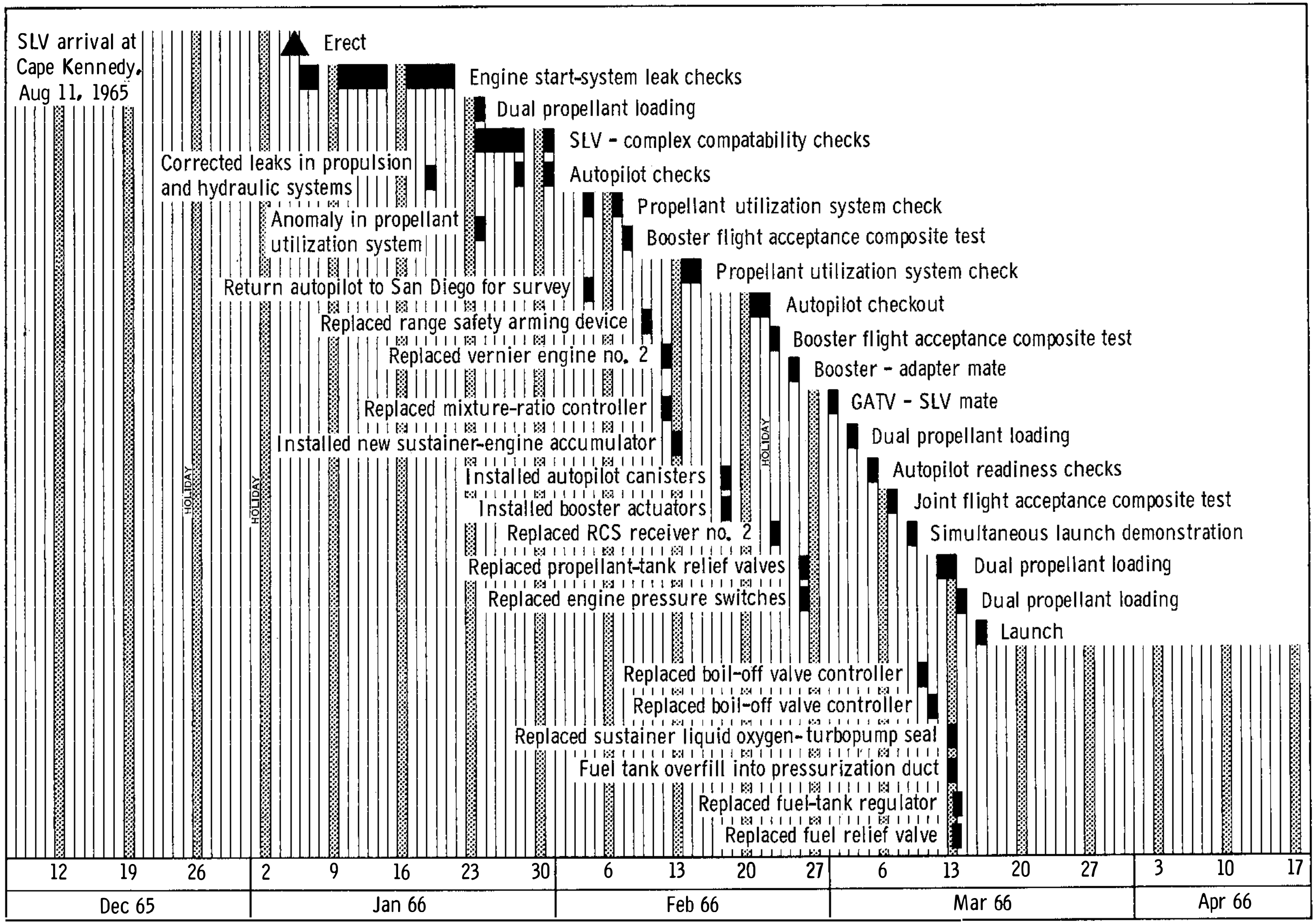

Figure 12.1-12. - SLV 5302 history at Cape Kennedy.

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12.2 WEATHER CONDITIONS

The weather conditions in the launch area at Cape Kennedy were satisfactory for all operations on the day of the launch, March 16, 1966. Surface weather observations in the launch area at 11:41 a.m. e.s.t. were as follows:

Cloud coverage	Low clouds, 3/10 covered; 3300 feet, scattered clouds; high, thin, broken clouds, 6/10 covered
Wind direction, deg from North	350
Wind velocity, knots	18
Visibility, miles	10
Pressure, in. Hg	30.09
Temperature, °F	70
Dew point, °F	59
Relative humidity, percent	68

The weather observations taken at 06:20 G.m.t., March 17, 1966, onboard the U.S.S. Leonard F. Mason located at latitude 25°22' north, longitude 135°56' east were as follows:

Cloud coverage	7/10 covered at 7000 feet
Wind direction, deg from North	275
Wind velocity, knots	2
Visibility, miles	15
Temperature, °F	71
Dew point, °F	58
Relative humidity, percent	65
Sea temperature, °F	86
Sea state	5-ft waves at 6-second period

Table 12.2-I presents the launch-area atmospheric conditions near the time of lift-off. Table 12.2-II provides weather data in the vicinity of Okinawa at 00:00 G.m.t., March 17, 1966. Figure 12.2-1 presents the launch-area wind direction and velocity plotted against altitude.

TABLE 12.2-I.- LAUNCH AREA ATMOSPHERIC CONDITIONS
AT 15:11 G.m.t., MARCH 16, 1966

Altitude, ft (a)	Temperature °F (a)	Pressure, lb/ft^2 (a)	Density, $slugs/ft^3$ (a)
0×10^3	68.0	2122.5	2325.6×10^{-6}
5	48.2	1772.9	2027.8
10	37.4	1471.9	1725.7
15	23.0	1216.3	1470.1
20	5.0	998.0	1255.1
25	-14.8	812.2	1066.0
30	-38.2	654.3	904.5
35	-54.4	521.9	752.2
40	-61.6	413.1	606.5
45	-72.4	325.5	491.6
50	-81.4	255.4	393.6
55	-88.6	199.0	312.5
60	-83.2	154.7	240.0
65	-81.4	121.3	187.0
70	-74.2	95.0	143.5
75	-68.8	74.7	111.7
80	-67.0	58.8	87.5
85	-58.0	46.5	67.9
90	-56.2	36.9	53.3
95	-47.2	29.4	41.7
100	-36.4	23.6	32.5
105	-29.2	19.0	25.6
110	-22.0	15.2	20.3
115	-20.2	12.5	16.6
120	-11.2	10.2	13.2

[a]The accuracy of the readings is indicated at the end of the table.

TABLE 12.2-I.- LAUNCH AREA ATMOSPHERIC CONDITIONS AT 15:11 G.m.t., MARCH 16, 1966 - Concluded

Altitude, ft (a)	Temperature, °F (a)	Pressure, lb/ft^2 (a)	Density, $slugs/ft^3$ (a)
125×10^3	-9.4	8.3	10.7×10^{-6}
130	-0.4	6.8	8.6
135	7.2	5.5	6.9
140	8.6	4.5	5.6
145	14.0	3.7	4.6
150	24.8	3.1	3.7
155	23.0	2.5	3.0
160	21.2	2.1	2.5
165	24.8	1.7	2.1
170	23.0	1.4	1.7
175	19.4	1.2	1.4
180	17.6	.9	1.2
185	19.4	.8	.9
190	14.0	.6	.8
195	1.4	.5	.6
200	-7.6	.4	.5
205	-16.6	.3	.4
210	-27.4	.2	.3
215	-36.4	.2	.3
220	-45.4	.1	.2
225	-52.6	.1	.2

[a]The accuracy of the readings is shown in the following table:

Altitude, ft	Temperature error, °F	Pressure rms error, percent	Density rms error, percent
0 to 60×10^3	1	1	0.5
60 to 120	1	1	.8
120 to 165	4	1.5	1.0
165 to 200	6	1.5	1.5
200 to 225	9	1.5	2.5

TABLE 12.2-II.- REENTRY AREA (OKINAWA) ATMOSPHERIC CONDITIONS

AT 00:00 G.m.t., MARCH 17, 1966

Altitude, ft	Temperature, °F (a)	Pressure, lb/ft^2 (a)	Density, slugs/ft^3 (a)	Wind speed, knots	Wind direction, deg from North
0.35×10^3	69.8	2088.5	2296×10^{-6}	27	18
4.93	57.2	1175.2	2000	4	15
10.25	42.8	1462.0	1695	54	21
19.06	15.8	1044.3	1280	50	47
24.64	-2.2	835.4	1065	50	66
31.46	-25.6	626.6	842	50	87
35.60	-43.6	522.1	732	51	96
40.42	-65.2	417.7	617	51	101
46.30	-90.4	313.2	494	51	89
54.07	-108.4	208.9	346	54	75
60.86	-101.2	146.2	238	68	31

[a]The accuracy of the readings is shown in the following table:

Temperature error, °F	1
Pressure rms error, percent	1
Density rms error, percent	0.5

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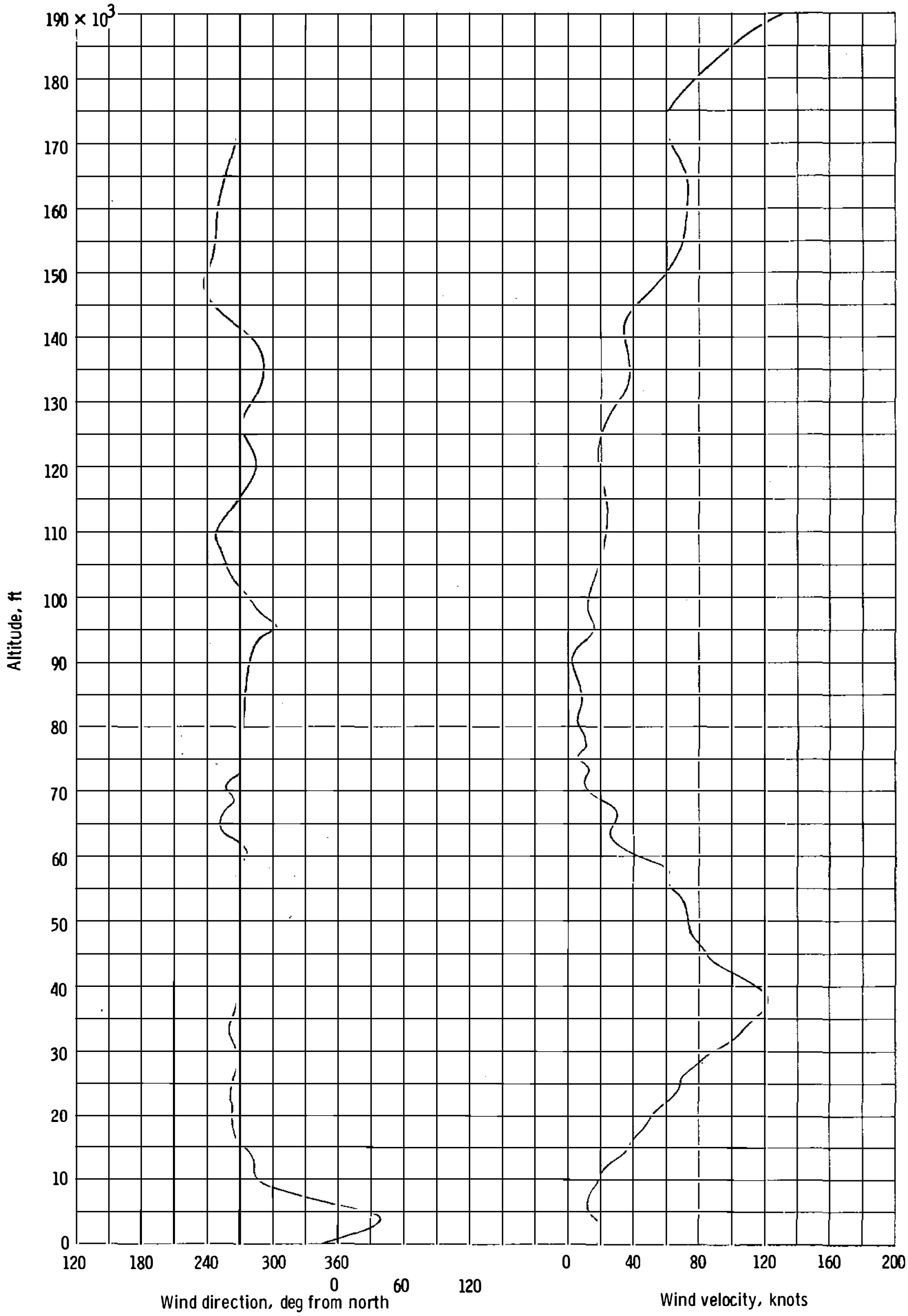

Figure 12.2-1. - Variation of wind direction and velocity with altitude for the launch area at 15:11 G. m. t., March 16, 1966.

12.3 FLIGHT SAFETY REVIEWS

During the following review meetings, the spacecraft, target vehicle, launch vehicles, extravehicular activity (EVA) equipment, and all supporting elements were determined to be in readiness for the Gemini VIII mission.

12.3.1 Spacecraft Readiness Review

The Flight Readiness Review of the spacecraft was held March 1, 1966. The following action items were to be completed prior to the launch:

(a) Perform additional verification firing of the Extravehicular-Support-Package (ESP) separation cartridge.

(b) Document and evaluate the degradation in pyrotechnic time delays being experienced at Kennedy Space Center (KSC).

(c) Identify all reuse-for-flight hardware by placing a letter R after the part number.

(d) Inspect heat-shield cracks to determine any change in configuration after cabin-pressure tests.

(e) Perform failure analysis on the suspected and replaced secondary A-pump circuit breaker and on the replaced fuel-cell hydrogen-to-oxygen differential-pressure transducer.

(f) Inspect and functionally test all quick disconnects that are to be actuated by the flight crew during the mission.

(g) Provide center-of-gravity calculations with and without the Extravehicular Life Support System (ELSS) and other significant stowage items.

(h) Verify rigging and measure closing forces of both hatches with the flight seals installed.

(i) Perform an end-to-end test of the flight ELSS and the ESP prior to spacecraft—launch vehicle mate.

12.3.2 Extravehicular Activity Equipment Review

On March 5, 1966, a review of the extravehicular activity (EVA) equipment was conducted at the Kennedy Space Center by the Gemini Program Office. Action items resulting from this review were as follows:

(a) Complete qualification testing and installation of the modified pressure-suit relief valve.

(b) Complete a failure-mode analysis of the emergency-oxygen regulator, prior to launch.

(c) Complete manned altitude-chamber testing of the EVA equipment.

(d) Conduct a nondestructive functional test of the rejected Hand-Held Maneuvering Unit which had exhibited trigger binding.

(e) Perform 100-percent microscopic inspection of all Microdot connectors and maintain rigid quality control on these connectors prior to launch.

(f) Complete vibration and altitude-chamber qualification testing on ELSS and ESP with heaters installed.

12.3.3 Design Certification Review

The Design Certification Review Board was convened in Washington, D.C., on March 6 and 7, 1966, and found the Gemini Agena Target Vehicle (GATV) satisfactory for flight. This decision was reached after consideration of the reports resulting from the investigation of the Gemini VI GATV incident October 25, 1965, and pending completion of the following items:

(a) Satisfactorily complete phases I and II of the test evaluation, and implement the phase III test plan at Arnold Engineering Development Center.

(b) Report on the fuel-contamination test procedures and the results of the GATV preflight fueling.

(c) Analyze the low temperature exhibited by the Primary Propulsion System.

(d) Evaluate the gas-generator fuel valve.

On March 6 and 7, 1966, a Certification Review was also held for the other elements of the Gemini VIII mission. These were the Gemini Launch Vehicle, the Target Launch Vehicle (TLV), the spacecraft, and the EVA equipment. Action items were remanded to the responsible organization for completion prior to the Mission Briefing.

12.3.4 Mission Briefing

The Mission Director conducted the Mission Briefing at the Kennedy Space Center on March 12, 1966. With the exception of a liquid-oxygen leak in the TLV, all elements were found to be in readiness to support the mission. A seal was replaced in the turbine and final dual tanking and leak checks were performed to clear this item.

12.3.5 Flight Safety Review Board

The Air Force Space System Division Flight Safety Review Board met at the Air Force Eastern Test Range on March 15, 1966. After insuring that all open items had been satisfactorily resolved, the board recommended to the Mission Director that the Gemini Launch Vehicle and the Gemini Atlas-Agena Target Vehicle be committed to flight. All ground and airborne systems were declared ready to accomplish the mission.

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12.4 SUPPLEMENTAL REPORTS

Supplemental reports for the Gemini VIII mission are listed in table 12.4-I. The format of these reports will conform to the external distribution format of NASA or that of the external organization preparing the report. Each report will be identified on the cover and the title page as being a Gemini VIII supplemental report. Before publication, the supplemental reports will be reviewed by the cognizant Senior Editor, the Chief Editor, and the Mission Evaluation Team Manager, and will be approved by the Gemini Program Manager. Distribution of the supplemental reports will be the same as that of this Gemini Program Mission Report.

TABLE 12.4-I.- GEMINI VIII SUPPLEMENTAL REPORTS

Number	Report Title	Responsible organization	Completion due date
1	Launch Vehicle Flight Evaluation Report — NASA Mission Gemini/Titan GT-8	Aerospace Corp.	May 16, 1966
2	Launch Vehicle No. 8 Flight Evaluation	Martin Co.	April 30, 1966
3	Manned Space Flight Network Performance Analysis for GT-8 Mission	Goddard Space Flight Center	May 16, 1966
4	Gemini GT-8 IGS Evaluation Trajectory Reconstruction	TRW Systems	April 30, 1966
5	GT-8 Inertial Guidance System and Computer Analysis	International Business Machines Corp.	April 30, 1966
6	Gemini Agena Target Vehicle 5003 Systems Test Evaluation	Lockheed Missiles and Space Co.	April 30, 1966
7	Atlas SLV-3 Space Launch Vehicle Flight Evaluation Report SLV-3 5302	General Dynamics Corp.	April 30, 1966

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12.5 DATA AVAILABILITY

Tables 12.5-I through 12.5-III list the mission data available at the NASA Manned Spacecraft Center. The trajectory and telemetry data will be on file in the Central Metric Data File of the Computation and Analysis Division. The photographic data will be on file at the Photographic Technology Laboratory.

TABLE 12.5-I.- SUMMARY OF INSTRUMENTATION DATA AVAILABILITY

Data description	
Paper recordings Spacecraft telemetry measurements and selected parameters for revolutions 1, 2, 3, 4, 5, 6, 7, reentry, and selected real-time site passes GLV telemetry measurements (launch) Telemetry signal-strength recordings MCC-H plotboards (Confidential) Range safety plotboards (Confidential) Radar data IP-3600 trajectory data (Confidential) MISTRAM (Confidential) Natural coordinate system Final reduced C-band (launch phase - Confidential) Natural coordinate system Final reduced Trajectory data processed at MSC and GSFC Voice transcripts Air-to-ground Onboard recorder (Confidential) Technical debriefing (Confidential) GLV reduced telemetry data (Confidential) Engineering units versus time plots Spacecraft reduced telemetry data Engineering units versus time Ascent phase Parameter tabulation (bandpass) Selected time history tabulations Orbital phase Parameter tabulations (statistical) for revolutions 1 and 3	Orbital phase - Continued Time history tabulation for revolutions 5, 6, and 7 Time history tabulations of selected parameters for selected times for revolutions 1, 2, 3, and 4 Time history plots for selected parameters and selected times for revolutions 1 - 7 Band-pass tabulations for selected parameters for revolutions 1, 2, and 3 Reentry phase Plots and tabulations of all system parameters Mod III RGS versus IGS velocity comparison (Confidential) Event tabulations Sequence-of-event tabulations versus time (including thruster firings) for ascent, reentry, and revolutions 1, 2, 3, 4, 5, 6, and 7, and for selected real-time passes for revolutions 1, 2, 4, and 5 Special computations Ascent phase IGS computer-word flow tag corrections (Confidential) Special aerodynamic and guidance-parameter calculations (Confidential) Steering-deviation calculation (Confidential) MISTRAM versus IGS velocity comparison (Confidential) Orbital phase Horizon sensor and gimbal angle comparison for revolutions 1, 2, 3, 4, 5, 7 and selected real-time site passes OAMS propellant-remaining computations for revolutions 1, 2, 3, 4, 5, 6, and 7

TABLE 12.5-I.- SUMMARY OF INSTRUMENTATION DATA AVAILABILITY - Concluded

Data description

Orbital phase - Continued

OAMS thruster-activity computations for revolutions 2, 3, 4, and 5

OAMS thruster-valve program for revolutions 1, 2, 3, 5, 6, and 7

Reentry phase

RCS propellant-remaining and thruster-activity computations

Lift over drag and auxiliary computations

True attitude angles (pitch, roll, and yaw) computed from telemetered gimbal angles.

Guidance and control and aerodynamic data combined plots.

Paper recordings

GATV telemetry measurements

MCC-K real-time passes for revolutions 1 through 45

SLV-3 telemetry measurements (launch)

GATV telemetry measurements (launch)

MCC-H and Range Safety plotboards

Radar data

IP-3600 trajectory data (Confidential)

C-band overlapping trajectory (Confidential)

Final reduced, coordinate systems 2 and 3

Trajectory data processed at MSC

GATV reduced telemetry data

Ascent phase

Engineering units versus time

Parameter tabulations (bandpass)
Time history plots and tabulations (selected parameters for selected intervals)

Digital parameter tabulations

Turbine speed and velocity meter readout
Programmer memory readout
Bi-level events

Orbital phase

All Primary Propulsion System (PPS) and Secondary Propulsion System (SPS) firings and GATV maneuvers including docking and undocking.

Engineering units versus time

Parameter tabulations (bandpass)
Time history plots and tabulations (Selected parameters for selected intervals during engine firings)

Digital parameter tabulations

Turbine speed and velocity meter readout
Programmer memory readout
Bi-level events

Data from selected sites from revolution 1 through 120 before and after all GATV PPS and SPS firings and maneuvers and during selected programmer memory loading and readout intervals.

Engineering units versus time

Parameter tabulations (bandpass)

Digital parameter tabulations

Programmer memory readout
Bi-level events

Special computations

Orbital phase

Sunrise - sunset computations

TABLE 12.5-II.- SUMMARY OF PHOTOGRAPHIC DATA AVAILABILITY

Category	Number of still photographs	Motion picture film, feet
Launch and prelaunch		
GAATV	1	[a]2 506
GLV and spacecraft	4	[a]12 927
Recovery		
Spacecraft in water	3	600
Loading of spacecraft on destroyer	14	800
Inspection of spacecraft	8	
Okinawa		300
General activities	28	
Inspection of spacecraft	97	
Postflight inspection	86	
Onboard spacecraft		
16-mm sequential camera	[b]15	161
70-mm still camera	19	

[a]Engineering sequential film only.

[b]Individual 16-mm frames

TABLE 12.5-III.- LAUNCH PHASE ENGINEERING SEQUENTIAL CAMERA DATA AVAILABILITY

(a) Spacecraft and GLV

Sequential film coverage, item	Size, mm	Location	Presentation	Total length of film, ft
1.2-1	16	50-foot tower, 19-7	GLV possible fuel leakage	384
1.2-2	16	50-foot tower, 19-9	GLV possible fuel leakage	383
1.2-3	16	50-foot tower, 19-4A	GLV possible fuel leakage	396
1.2-4	16	50-foot tower, 19-7	Surveillance of launch complex	1216
1.2-5	16	50-foot tower, 19-9	Surveillance of launch complex	1163
1.2-6	16	50-foot tower, 19-4A	Surveillance of launch complex	1175
1.2-7	16	50-foot tower, 19-4A	Surveillance of launch complex	1152
1.2-8	16	50-foot tower, 19-4A	Surveillance of launch complex	1215
1.2-9	16	50-foot tower, 19-1	GLV launch	170
1.2-10	16	50-foot tower, 19-5	GLV launch	170
1.2-11	16	50-foot tower, 19-7A	GLV launch	180
1.2-12	16	50-foot tower, 19-2	Spacecraft launch	80
1.2-13	16	50-foot tower, 19-7A	Spacecraft launch	78
1.2-14	16	Umbilical tower, second level	GLV Stage II umbilical	127
1.2-15	16	50-foot tower, 19-7A	GLV, engine observation	130
1.2-16	16	East launcher	GLV, possible fuel leakage	125
1.2-17	16	West launcher	GLV, possible fuel leakage	140
1.2-18	16	North launcher	GLV, engine observation	120
1.2-19	16	South launcher	GLV, engine observation	115
1.2-20	16	Umbilical tower, first level	GLV, umbilical disconnect	75
1.2-21	16	Umbilical tower, second level	GLV, umbilical disconnect	120
1.2-22	16	Umbilical tower, fourth level	GLV, umbilical disconnect	150
1.2-23	16	Umbilical tower, fifth level	GLV, umbilical disconnect	139
1.2-24	16	Umbilical tower, sixth level	GLV, umbilical disconnect	134
1.2-25	16	Umbilical tower, sixth level	GLV, umbilical disconnect	208

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TABLE 12.5-III.- LAUNCH PHASE ENGINEERING SEQUENTIAL CAMERA DATA AVAILABILITY - Continued

(a) Spacecraft and GLV

Sequential film coverage, item	Size, mm	Location	Presentation	Total length of film, ft
1.2-26	16	Umbilical tower, top level, no. 1	GLV, upper umbilical disconnect	145
1.2-27	16	Umbilical tower, top level, no. 2	J-bars and lanyard observation	125
1.2-28	16	50-foot tower, east side	Spacecraft umbilical	182
1.2-29	70	South of Pad 19	GLV and spacecraft launch	32
1.2-30	70	West of Pad 19	GLV and spacecraft launch	36
1.2-31	16	North of Pad 19	Tracking	390
1.2-32	16	West of Pad 19	Tracking	370
1.2-33	16	South of Pad 19	Tracking	370
1.2-34	16	South of Pad 19	Tracking	270
1.2-35	16	South of Pad 19	Tracking	300
1.2-36	35	South of Pad 19	Tracking	260
1.2-37	35	South of Pad 19	Tracking	276
1.2-38	35	Northwest of Pad 19	Tracking	282
1.2-39	70	Northwest of Pad 19	Tracking	120
1.2-40	70	Cocoa Beach, Florida	Tracking, ROTI	128
1.2-41	70	Melbourne Beach, Florida	Tracking, ROTI	16
1.2-44	35	C-54 Aircraft	Tracking	280

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TABLE 12.5-III.- LAUNCH PHASE ENGINEERING SEQUENTIAL CAMERA DATA AVAILABILITY - Concluded

(b) GAATV

Sequential film coverage, item	Size, mm	Location	Presentation	Total length of film, ft
1.2-4	16	East of Pad 14	TLV engine observation	100
1.2-5	16	West of Pad 14	TLV engine observation	150
1.2-6	16	Northwest of Pad 14	TLV engine observation	110
1.2-7	16	Ramp, south of Pad 14	TLV engine observation	105
1.2-8	16	West of Pad 14	TLV launch	70
1.2-9	16	Northwest of Pad 14	TLV launch	60
1.2-10	16	Northwest of Pad 14	TLV vernier-engine heat shield	170
1.2-11	16	Southeast of Pad 14	TLV vernier-engine heat shield	100
1.2-12	16	Umbilical tower, 79-feet level	TLV upper umbilical	80
1.2-13	16	Umbilical tower, 72-feet level	TLV lower umbilical	100
1.2-14	16	Southwest of Pad 14	Umbilical tower	90
1.2-15	70	Southwest of Pad 14	Umbilical tower	40
1.2-16	16	Northwest of Pad 14	Tracking	279
1.2-17	16	South-southwest of Pad 14	Tracking	291
1.2-18	35	West of Pad 14	Tracking	240
1.2-19	35	Patrick Air Force Base	Tracking, IGOR	147
1.2-20	70	Northwest of Pad 14	Tracking	128
1.2-21	70	Cocoa Beach, Florida	Tracking, ROTI	120
1.2-22	70	Melbourne Beach, Florida	Tracking, ROTI	126

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12.6 POSTFLIGHT INSPECTION

The postflight inspection of the Spacecraft 8 reentry assembly was conducted in accordance with reference 18 and with approved Spacecraft Test Requests (STR's) at the contractor's facility in St. Louis, Missouri, from March 22, 1966, to April 22, 1966. The Rendezvous and Recovery (R and R) Section was returned with the spacecraft to the contractor's facility, and the drogue and pilot parachutes were returned to Cape Kennedy for damage charting. Several items of equipment were removed onboard the recovery ship for return to the Manned Spacecraft Center (MSC); however, most crew-station stowed items were removed in accordance with STR 8000 upon receipt of the spacecraft in St. Louis and dispatched to MSC by special courier aircraft. The reentry assembly was received in fair condition in St. Louis. The roll bar on the forward end of the Reentry Control System (RCS) section was damaged and the outer perimeter of the heat shield had portions of the char layer missing. The retaining nut plates of the left-hand hatch window frame had been chiseled off and the left window removed and returned as a loose piece. The upper left beryllium RCS shingle was broken.

The following list itemizes the discrepancies noted during the detailed inspection of the reentry assembly:

(a) As on previous spacecraft, residue was found on the exterior surface of both hatch windows.

(b) A plug from the right upper adapter interconnect fairing, which contains the urine-dump heating element, was hanging loose by the heating element wires.

(c) One dome of the rendezvous radar was indented.

(d) The lower docking-latch door location (BY) was retracted.

(e) The actuator rod for the right hatch was scored.

(f) Severe corrosion and electrical shorting as a result of being immersed in sea water was noted within the Attitude Control Electronics (ACE) package.

(g) One rate-gyro case was slightly indented.

(h) The terminal end of a ground wire was broken.

(i) A cold-plate coolant line was indented.

(j) A squib battery connector contained moisture, residue, and corrosion.

(k) A small water stain was found at the forward edge of the Environmental Control System (ECS) door.

(l) Five fuses in the electrical fuse blocks were blown.

(m) An electrical interface connector between the RCS section and the cabin section contained corrosion.

(n) A Communications System coaxial cable connector to the switch in the Z160 bulkhead area was loose.

(o) The power-connector-plug potting on the HF antenna case was loose in the connector.

(p) A crack was found in the heat-shield Fiberite ring.

(q) A relay in a relay panel on the Z160 bulkhead was slightly dented.

(r) A foreign substance was found in the left-shoulder Koch fitting of the right ejection seat.

(s) Out-of-tolerance hand-controller loads were encountered during postflight tests of the hand controller.

12.6.1 Spacecraft Systems

12.6.1.1 Structure.- The overall appearance of the spacecraft was good. The appearance of the heat shield was normal, except for a small crack in the peripheral Fiberite ring. The heat damage to the lower-right adapter interconnect fairing was heavier than in the past. The thermal insulation blankets on the lower side of the reentry assembly were scorched more than noted on previous spacecraft. The stagnation point was located 13.4 inches below the horizontal centerline and 0.4 inch to the left of the vertical centerline. The heat shield was removed and dried with the reentry assembly and R and R Section. The wet weight of the heat shield was 322.43 pounds without the insulation blankets. The dry weight of the heat shield in the same configuration was 308.24 pounds.

Residue similar to that found on the windows of previous spacecraft was noted, and an investigation to determine the composition is being performed (STR 8002). The lower centerline docking-latch door was retracted because interference from the insulation block over the release hook had prevented the door from releasing. Examination of the ECS door and well-interior area indicated a small leak at the forward edge of the ECS door. A torque of 250 inch-pounds applied at the external hatch sockets was required to open each hatch. The R and R Section was given a comprehensive inspection (STR 8003). The measurements of loads and dimensions to obtain information for developing hatch-seal installation procedures were accomplished (STR 8017). The cabin was pressurized to 5.1 psid. Measurements of cabin leakage, centerline-stowage-box deflections, and hatch closing forces were made (STR 8023). Five heat-shield plugs and a portion of the Fiberite ring containing a crack were removed for inspection and analysis (STR 8505A). The travel of the hatch-actuator lock-release lever was measured to determine the optimum position (STR 8512).

12.6.1.2 Environmental Control System.- Drinking-water samples were taken and dispositioned for analysis in accordance with reference 18. The total water removed was 13 pounds 6 ounces. The lithium-hydroxide cartridge was removed from the ECS package and weighed. The cartridge weighed 100.77 pounds with a center-of-gravity 8.22 inches from the bottom of the cartridge. The cartridge was dispositioned for reuse (STR 8015). The secondary oxygen system was deserviced in accordance with reference 18 and no residual pressure remained in the bottles.

The ECS handles were actuated in accordance with reference 18 and the maximum handle force recorded was 25 pounds on the inlet snorkel handle. The cryogenic gaging system was investigated (STR 8027). A synethetic rubber plug from the right-hand upper adapter interconnect fairing, containing the urine-dump heating element, was hanging loose by the element wires. This plug is cemented in place during manufacture and was apparently released by the high temperatures experienced during reentry.

12.6.1.3 Communications System.- The external appearance of all communications equipment was good. A small amount of corrosion was evident on the coaxial-cable switches and connectors. The HF whip antenna was retracted and appeared to have operated normally.

A coaxial-cable connector to a switch on the Z160 small pressure bulkhead was loose. The power-connector-plug potting on the HF antenna case was loose in the connector.

The crew's helmets, communications harnesses, microphones, voice tape recorder, and light-weight headsets were returned to the contractor's facility and the tests outlined in STR's 8018 and 8019 were completed. Upon completion of the tests, the voice tape recorder was returned to Kennedy Space Center for further checks (STR 8019).

12.6.1.4 Guidance and Control System.- The wiring and control switches in the Attitude Control System were checked for continuities (STR 8503A). In addition, a bench checkout for continuity was performed on the ACE package, and the hand controller was removed and a pre-installation acceptance test was performed. The removal of the ACE package cover revealed excessive internal corrosion from sea-water immersion and evidence of shorting. Out-of-tolerance handle forces were encountered during bench testing of the hand controller.

One dome of the rendezvous radar was oil-canned inward. (This occurred on Gemini VI-A and apparently resulted from differential pressures experienced during reentry.) The radar was removed and externally inspected. Severe corrosion due to sea water was noted on the mounting flange and external case of the radar. The radar was packaged and will be retained in storage with the spacecraft. The cover of the ACE package was replaced and the package was prepared for storage with the spacecraft. The rate-gyro package (52-87700-33 serial no. E452) had a slight dent in the outer case. The Inertial Measurement Unit (IMU) system, Attitude Control Maneuver Electronics (ACME), rate gyros and inverters, horizon-sensor electronics, computer, and Auxiliary Control Power Unit (ACPU) were removed and sent to the vendors (STR's 8007, 8008, 8010, 8011, and 8012).

12.6.1.5 Pyrotechnic system.- Pyrotechnic resistance checks were performed on all electrically initiated pyrotechnic devices in the reentry assembly in accordance with reference 18. Four pyrotechnic devices indicated resistance readings and were removed for visual inspection. All four devices had detonated.

The postflight visual inspection of the wire bundle guillotines, parachute-bridle release mechanisms, and other pyrotechnics disclosed that all appeared to have functioned normally. Inspection of the hatch actuators revealed slight axial scoring on the right actuator rod.

The electrical connectors to the mild-detonating-fuse (MDF) detonators on the left and right sides of the Z192 bulkhead had the bayonet pins sheared off and were hanging loose from the cartridges. This condition has been noted on previous spacecraft and is considered acceptable. Both of the MDF detonators had high-order detonation.

The hatch-actuator breeches, rocket catapults, seat pyrotechnic devices, and other unfired pyrotechnic devices were removed for storage and subsequent disposition in accordance with reference 18.

12.6.1.6 Instrumentation and Recording System.- The pulse code modulation (PCM) tape recorder was removed from the spacecraft at St. Louis and sent to the contractor for tape removal and storage. The PCM programmer, instrumentation package 2, high-level multiplexer, and low-level multiplexer were removed at St. Louis and sent to the vendor (STR's 8013 and 8014). The dc-to-dc converters were removed and sent to the contractor for evaluation and reuse (STR 8500). The bio-medical tape recorders were removed in the spacecraft recovery area and returned to the MSC.

12.6.1.7 Electrical System.- The main and squib batteries were removed and discharged in accordance with reference 18. The following table lists the ampere-hours remaining in each battery after flight when discharged to the level of 20 volts with the battery still delivering the currents specified in reference 18.

Main battery	Serial number	Discharge, A-h	Squib battery	Serial number	Discharge, A-h
1	155	36.8	1	96	10.7
2	161	33.8	2	97	9.8
3	162	30.0	3	98	9.7
4	164	34.3			

The main and squib batteries were recharged and placed in bonded storage for future ground test use. The current leakage caused by salt-water immersion was checked and recorded in reference 18.

The fuse-block status check was performed, in accordance with reference 18, and the following fuses had been blown:

Fuse block	Pin no.	Fuse no.
XF-F	1	4-33
XF-F	3	4-51
XF-F	4	4-52
XF-M	3	4-26
XF-AE	4	13-13

The inspection of the aerospace ground equipment (AGE) test points was performed in accordance with reference 18, and 24 of the 31 test points contained corrosion, residue, or water. Results of each AGE test point inspection are contained in reference 18.

The terminal was broken off ground wire 258Z located behind access door 28 in the Z160 bulkhead area. The connector for squib battery no. 1 contained moisture and a small amount of residue and pin corrosion. The interface connector on wire bundle 209C between the RCS section and cabin section in the area of the Z160 bulkhead had corrosion in the female portion of the connector. Wires N93B22 and N94B22 from the RCS section to the cabin section were not routed through a connector and had to be cut to remove the RCS section. These wires were routed to the electrical striker plates in the R and R Section for transfer of signals to the GATV. The K3-59 RCS abort relay, located on the RCS-and-scanner-cover relay panel on the Z160 bulkhead, was slightly dented. An investigation to determine the possibility of a common-control-bus intermittent short was conducted (STR 8024).

A test was performed to determine if the circuit breakers were faulty for the oxygen and hydrogen heaters, Auxiliary Tape Memory Unit, antenna select, and Orbital Attitude and Maneuver System (OAMS) control circuit breakers (STR 8025). The investigation of the non-illumination of the amber IND RETRO ATT light at T_R - 256 seconds as reported by the crew was conducted (STR 8508). The reported anomaly of low main-battery voltages prior to adapter equipment section separation was investigated (STR 8509A).

An inspection of the electrical-wire-bundle clamp area for evidence of chafing, cutting, or abrasion was conducted (STR 8515). No evidence of damage to the electrical wire bundles was found in the examination of ten clamp areas.

12.6.1.8 Crew-station furnishings and equipment.- The appearance of the cabin interior was good. The switch positions and instrument readings were recorded and cabin photographs were taken immediately upon arrival of the spacecraft at St. Louis.

The command pilot's lap belt was twisted in the adjustment buckle and this may account for his comment of not being able to get the lap belt tight. An unknown substance was found on the left-shoulder Koch fitting of the right ejection seat. A sample of the substance was removed for analysis (STR 8028). An investigation was conducted to determine the out-of-calibration condition of the Stage II Malfunction Detection System (MDS) tank-pressure indicator (STR 8026).

The ejection seats were removed and deactivated in accordance with reference 18. The backboard contours, pelvic blocks, egress-kit contours, and lap belts were placed in government-furnished-equipment (GFE) bonded storage at the contractor's plant in St. Louis. The seat ballast was shipped to the Kennedy Space Center (KSC) for reuse. The GFE components contained in the survival kit were shipped to the MSC. The ejection seats, minus the above equipment, were shipped to the MSC for use on the Dynamic Crew Procedures Simulator (STR 8001).

12.6.1.9 Propulsion System.- The RCS thrust chamber assemblies appeared normal. The RCS was deactivated at Naha, Okinawa, and purge-gas samples were sent to Patrick Air Force Base, Florida, for analysis. Results of the purge-gas analysis are contained in reference 18. No propellants were obtained from either the A-ring or B-ring for analysis. Thrust chamber assembly 3A was removed and sent to the KSC for analysis (STR 8030). Thruster chamber assembly 5B was removed and sent to the contractor in St. Louis for analysis (STR 8513).

12.6.1.10 Landing System.- The drogue and pilot parachutes were returned to Cape Kennedy for washing, drying, and damage charting. The parachutes will be returned to the MSC for further analysis (STR 8004). Calibration tests of the static pressure system and altimeter were conducted (STR 8029). No anomalous readings were found. Visual examination of the R and R Section revealed that the apex line cutter and pilot-parachute mortar had not been actuated. This is normal for a nominal parachute recovery.

12.6.1.11 Postlanding recovery aids.- The flashing recovery light and the hoist-loop doors appeared to have functioned normally. An analysis was conducted to determine the amount of sea dye marker remaining (STR 8020).

12.6.1.12 Experiments.- The experiments equipment located in the crew-station area was removed and disposed of in accordance with STR 8000. The majority of the equipment was removed at the contractor's facility in St. Louis and dispatched by special courier aircraft to the MSC. The contractor conducted a circuit review of the D-15 experiment equipment to determine if it had any possible relationship to the anomalies which occurred during the Gemini VIII mission. After this review, the D-15 experiment equipment was removed from the right landing-gear well and shipped to the MSC.

12.6.2 Continuing Evaluation

The following is a list of the STR's that have been approved for the postflight evaluation of reported spacecraft anomalies.

STR no.	System	Purpose
8002	Structure	To determine the composition and origin of the residue on the windows emphasizing time-of-flight associated effects due to any materials freed during docking with GATV.
8003	Structure	To determine the operational environment of the spacecraft—GATV docking interface based on mechanical condition of R and R Section structure.
8004	Landing System	To conduct an evaluation of parachute materials exposed to the space environment.
8018	Voice communications	To investigate an anomaly which occurred during prelaunch testing of the Communication System.
8023	Structure	To determine if water leaked into the cabin as a result of forces exerted on the spacecraft at landing. Also, to determine cause of difficulty in closing centerline-stowage-box door.
8024	Electrical	To determine the possibility of a common-control-bus intermittent short.
8025	Electrical	To determine if the circuitry and circuit breakers for the oxygen and hydrogen heaters, Auxiliary Tape Memory Unit, antenna select, and OAMS control were functioning properly.
8026	Crew station	To determine the cause of out-of-calibration readings from Stage II Malfunction Detection System propellant-tank pressure indicator.
8027	Environmental Control System	To investigate the cause of a flight anomaly in the cryogenic gaging system.
8028	Ejection seat	To determine source and type of substance found on right-seat Koch fitting during final countdown.
8029	Landing System	To conduct calibration tests of static pressure system and altimeter.

STR no.	System	Purpose
8030	Propulsion System	To evaluate the quality of the potting and wiring in the electrical-connector riser arm of thrust-chamber-assembly solenoid valves.
8502	Pyrotechnics	To investigate out-of-tolerance resistance readings encountered during postflight testing of pyrotechnics.
8503	Guidance and Control System	To verify wiring and control switches in the attitude control system.
8505A	Structure	To investigate effects of reentry on the heat-shield areas which exhibited separations and cracks.
8508	Electrical	To investigate the non-illumination of the amber IND RETRO ATT light at T_R - 256 seconds.
8509	Electrical	To measure resistance in the battery test circuits from battery connector to the test voltage monitor point and spacecraft ground as a result of low voltage reported prior to adapter equipment section separation.

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