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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

GEMINI PROGRAM MISSION REPORT

GEMINI IX-A

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MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

JULY 1966

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GEMINI FLIGHT HISTORY			
Mission	Description	Launch date	Major accomplishments
Gemini I	Unmanned 64 orbits	Apr. 8, 1964	Demonstrated structural integrity. Demonstrated launch vehicle systems performance.
Gemini II	Unmanned suborbital	Jan. 19, 1965	Demonstrated spacecraft systems performance.
Gemini III	Manned 3 orbits	Mar. 23, 1965	Demonstrated manned qualification of the Gemini spacecraft.
Gemini IV	Manned 4 days	June 3, 1965	Demonstrated spacecraft systems performance and crew capability for 4 days in space. Demonstrated EVA.
Gemini V	Manned 8 days	Aug. 21, 1965	Demonstrated long-duration flight. Demonstrated 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	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 controlled reentry to within 7 miles of planned landing point.
Gemini VI-A	Manned 1 day rendezvous	Dec. 15, 1965	Demonstrated on-time launch procedures, closed-loop rendezvous capability, and station keeping technique with spacecraft 7.
Gemini VIII	Manned 3 days rendezvous and dock, and EVA	Mar. 16, 1966	Demonstrated rendezvous and docking with GATV, controlled landing, emergency recovery, and multiple restart of GATV in orbit. Spacecraft mission terminated early because of an electrical short in the control system.
Gemini IX	Manned 3 days rendezvous and dock, and EVA (canceled after failure of TLV)	May 17, 1966	Demonstrated dual countdown procedures.
Gemini IX-A	Manned 3 days rendezvous and dock, and EVA	June 3, 1966	Demonstrated three rendezvous techniques, EVA with detailed work tasks, and precision landing capability.

GEMINI PROGRAM MISSION REPORT

GEMINI IX-A

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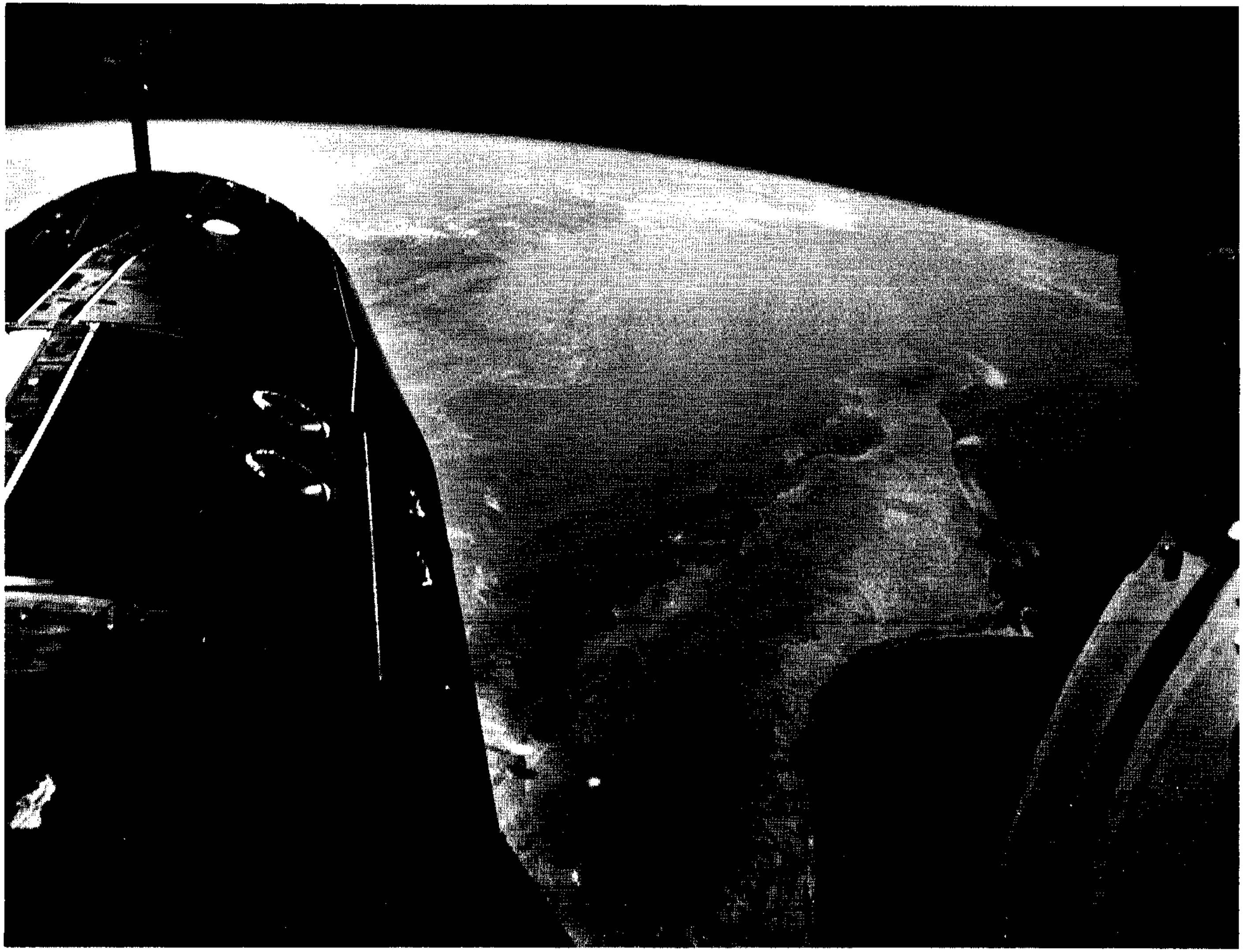

View of Baja California taken by the pilot while standing in the open hatch.

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1.0 MISSION SUMMARY

Gemini IX-A was the seventh manned mission and the third rendezvous mission of the Gemini program. The Target Launch Vehicle/Augmented Target Docking Adapter was launched from Complex 14, Cape Kennedy, Florida, at 10:00:02 a.m. e.s.t. on June 1, 1966. The Gemini Space Vehicle, with Astronaut Thomas P. Stafford as the Command Pilot and Astronaut Eugene A. Cernan as the Pilot, was launched from Complex 19, Cape Kennedy, Florida, at 8:39:33 a.m. e.s.t. on June 3, 1966, at the exact second for a nominal rendezvous during the third revolution. The flight was successfully concluded on June 6, 1966, with the recovery of the flight crew in the spacecraft. They were hoisted aboard the prime recovery ship (U.S.S. Wasp) approximately 53 minutes after a very accurate landing within sight of the recovery ship. The crew completed the flight in good physical condition and demonstrated full control of the spacecraft and competent management of all aspects of the mission.

After completing rendezvous, the docking portion of a primary objective to rendezvous and dock could not be accomplished because the ascent shroud was still attached to the Target Docking Adapter. Later in the flight a primary objective to conduct extravehicular activities was completed when the pilot performed extravehicular activities for over two hours; however, evaluation of the Astronaut Maneuvering Unit, a secondary objective, was not completed due to fogging of the pilot's visor. Secondary objectives that were accomplished were – to rendezvous during the third revolution, to conduct systems evaluations, to accomplish an equi-period rendezvous, to accomplish a rendezvous from above, and to demonstrate controlled reentry. A secondary objective to conduct experiments was not completely accomplished in that the Agena Micrometeorite Collection Experiment, S-10, could not be completed because extravehicular activities were not conducted while near the target vehicle.

The launch of the Target Launch Vehicle/Augmented Target Docking Adapter was very satisfactory for the Gemini IX-A mission. The countdown was completed with no holds, and, as a result of a nominal lift-off and launch phase, the Augmented Target Docking Adapter was inserted into a near-circular orbit having an apogee of 161.5 nautical miles and a perigee of 158.5 nautical miles, referenced to the Fischer ellipsoid earth model of 1960.

The Gemini Space Vehicle was to have lifted off approximately 1 hour 40 minutes after the target vehicle; however, the launch had to be postponed when the T-3 minute launch-azimuth update to the spacecraft computer could not be transmitted to the spacecraft because of ground equipment problems. The launch was rescheduled for June 3, 1966, and took place on that day, within one-half second of the desired time. The performance of the Gemini Launch Vehicle was satisfactory in all respects. The first-stage flight was normal, with all planned events occurring within the required limits, and staging and second-stage flight were completed normally. The spacecraft was inserted into an orbit with a perigee of 85.7 nautical miles and an apogee of 144.0 nautical miles, referenced to the Fischer ellipsoid earth model of 1960. The perigee was 1.0 nautical mile lower than planned, and the apogee was 4.6 nautical miles below that planned. The offset yaw-steering technique placed the spacecraft into an orbital plane very close to the plane of the target-vehicle orbit and the slant range to the target vehicle was a nominal 572 nautical miles.

During the following period of 4 hours 7 minutes, seven maneuvers were performed by the crew to effect a rendezvous with the Augmented Target Docking Adapter. The first three maneuvers were conducted using ground-computed values and pointing data. The terminal phase initiate maneuver was conducted using information from the onboard computer, ground computer, and onboard charts. The final three maneuvers were conducted line of sight using the onboard computer and displays. Continuous radar lock-on was achieved at a range of 120 nautical miles, and no significant losses of lock occurred until the rendezvous was completed at 4 hours 15 minutes ground elapsed time.

The crew reported that the shroud of the Augmented Target Docking Adapter had not separated from the vehicle and that docking could not be accomplished. After approximately 45 minutes of station keeping and inspection of the target vehicle, the crew performed a radial separation maneuver in preparation for an equi-period rendezvous which was to be performed using onboard optical techniques. The ground-computed maneuvers and IGS solutions were not used for this rendezvous. Terminal phase initiate and the accompanying midcourse corrections and braking maneuvers were performed to complete the equi-period rendezvous at approximately 6 hours 36 minutes ground elapsed time.

After station keeping for approximately 40 minutes, a second separation maneuver was performed by the crew in preparation for a rendezvous from above which was to be completed after the scheduled sleep period. During this third rendezvous, seven maneuvers were performed, beginning at 18:23:19 ground elapsed time, resulting in a rendezvous at approximately 21 hours 42 minutes ground elapsed time. The maneuvers resulted in a rendezvous which simulated a Command Module rendezvous with a Lunar Module in a lower orbit around the moon. It was discovered that the crew will probably not be able to track the target optically on the dayside pass during this type of rendezvous due to the low light contrast and high relative velocity between the target and the sunlit background. The crew stated that radar was essential until the range had closed to approximately two nautical miles.

After completing the third rendezvous, and prior to starting the preparations for extravehicular activity, the command pilot requested that the activity be delayed due to crew fatigue and ground control concurred. The preparations were postponed and the extravehicular activity was rescheduled for the following day. At 22 hours 59 minutes ground elapsed time, the crew performed the final separation maneuver from the Augmented Target Docking Adapter. The crew rested and conducted experiments for the remainder of the second day.

After the second scheduled sleep period, the crew began preparations for extravehicular activity and the right-hand hatch was opened at 49 hours 23 minutes ground elapsed time. The extravehicular phase of the flight proceeded in a satisfactory manner until the pilot began preparing the Astronaut Maneuvering Unit for donning. He immediately encountered greater than anticipated difficulty in maintaining his position relative to the work area. This increased workload apparently caused the Extravehicular Life Support System to become overloaded with moisture and the pilot's visor began fogging at the sides. The fogging steadily increased to the point that he was unable to see clearly in any direction. Rest periods were taken by the pilot in an unsuccessful attempt to clear the visor, and, because it was anticipated that the workload to complete the Astronaut Maneuvering Unit evaluation would result in additional fogging, the extravehicular operations were terminated. The pilot ingressed the spacecraft and the hatch was closed at approximately 51 hours 28 minutes ground elapsed time. The spacecraft hatch was open for a total of 2 hours and 5 minutes for extravehicular activity. The remainder of the third day was spent conducting experiments.

After the third scheduled sleep period, the crew began preparations for retrofire and landing in the revolution-46 primary recovery area. The retrofire sequence was initiated over the Canton station in the Pacific Ocean at 71:46:44 ground elapsed time. The landing point achieved was approximately one-third mile from the planned landing point, and 3.5 nautical miles from the prime recovery ship (U.S.S. Wasp). After landing at 72:20:50 ground elapsed time, the crew elected to stay in the spacecraft and were hoisted aboard the prime recovery ship at approximately 73 hours 13 minutes ground elapsed time, about 52 minutes after landing.

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2.0 INTRODUCTION

A description of the Gemini IX-A mission and a discussion of the results are contained in this report. The report covers the time from the start of the simultaneous countdown of the Target Launch Vehicle/Augmented Target Docking Adapter and the Gemini Space Vehicle to the date of publication of this report. Detailed discussions are found in the major sections related to each principal area of effort. Some redundancy may be found between the various sections where it is required for a logical presentation of the subject matter. Included in the appendix (section 12.0) is a report on the Target Launch Vehicle which failed during the Gemini IX mission. The evaluation of the Gemini Agena Target Vehicle flown on the Gemini IX mission is discussed in a supplemental report to this report. No other formal report will be made on the Gemini IX mission.

Data were reduced from telemetry, onboard records, and ground-based radar tracking, but were reduced only in areas of importance. In evaluating the performance of the Target 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 these results 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 experiments flown on this mission are presented in section 8.0, and preliminary results and conclusions of the 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 IX-A mission were as follows:

(a) Perform rendezvous and docking

(b) Conduct extravehicular activities.

The secondary objectives of the Gemini IX-A mission were as follows:

(a) Perform rendezvous and docking during the third revolution

(b) Conduct systems evaluation

(c) Perform equi-period re-rendezvous

(d) Conduct experiments

(e) Conduct docking practice

(f) Perform re-rendezvous from above

(g) Demonstrate a controlled reentry.

At the time of publication of this report, more detailed analyses of data on the performance of both launch vehicles and the Radio Guidance System were continuing. Analysis of the spacecraft Inertial Guidance System was 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 9.

3.0 VEHICLE DESCRIPTION

The manned vehicle for the Gemini IX-A mission consisted of Spacecraft 9 and Gemini Launch Vehicle (GLV) 9. The second vehicle consisted of the Augmented Target Docking Adapter (ATDA) and the Target Launch Vehicle (TLV) 5304.

The general arrangement and major reference coordinates of the manned Gemini Space Vehicle are shown in figure 3.0-1. Section 3.1 of this report describes the spacecraft configuration including the Extravehicular Life Support System (ELSS), 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 TLV and ATDA are shown in figure 3.0-2. Section 3.5 describes the TLV configuration, and section 3.6 provides the weight and balance data of the ATDA and the TLV. Section 3.7 describes the ATDA configuration.

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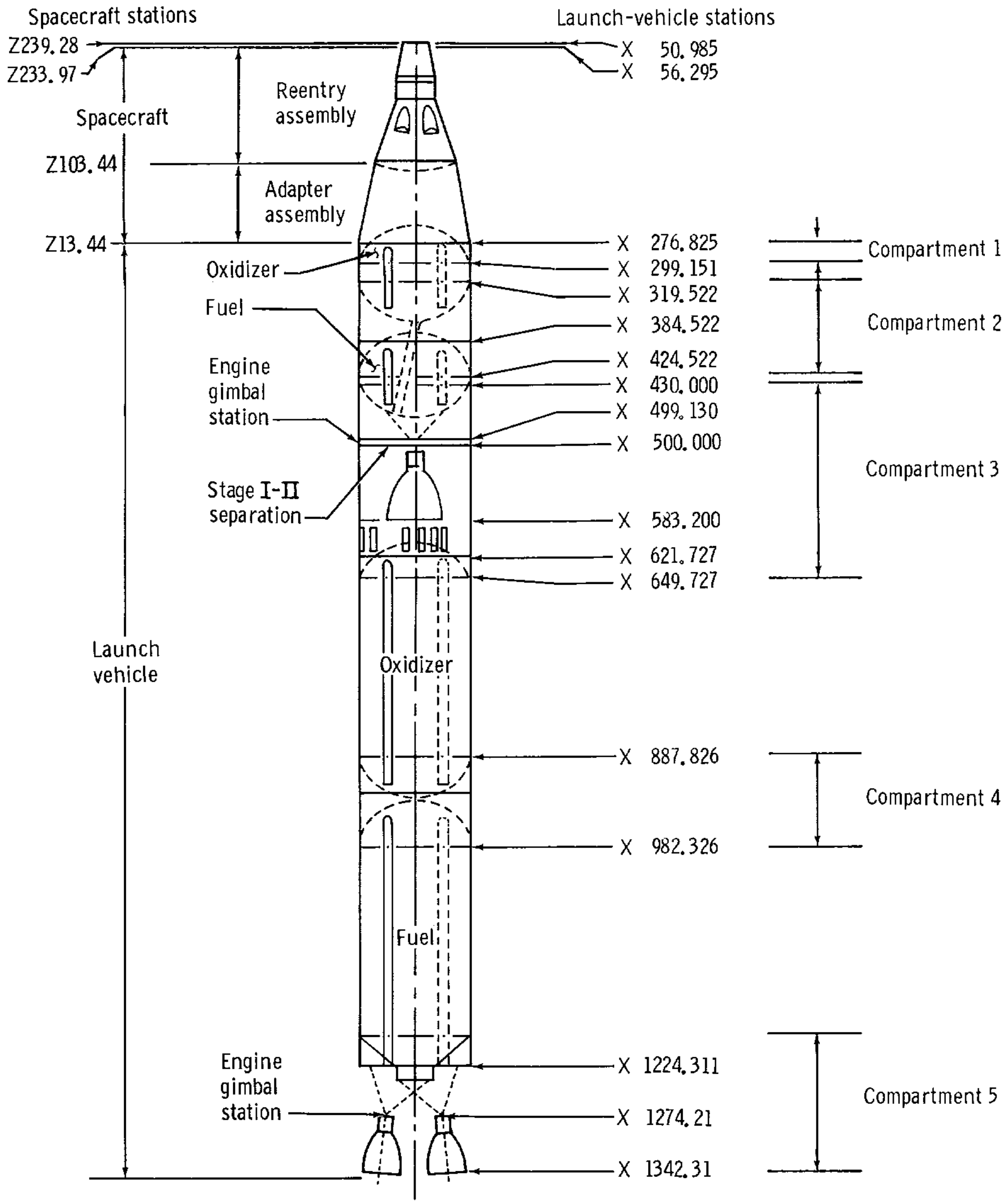
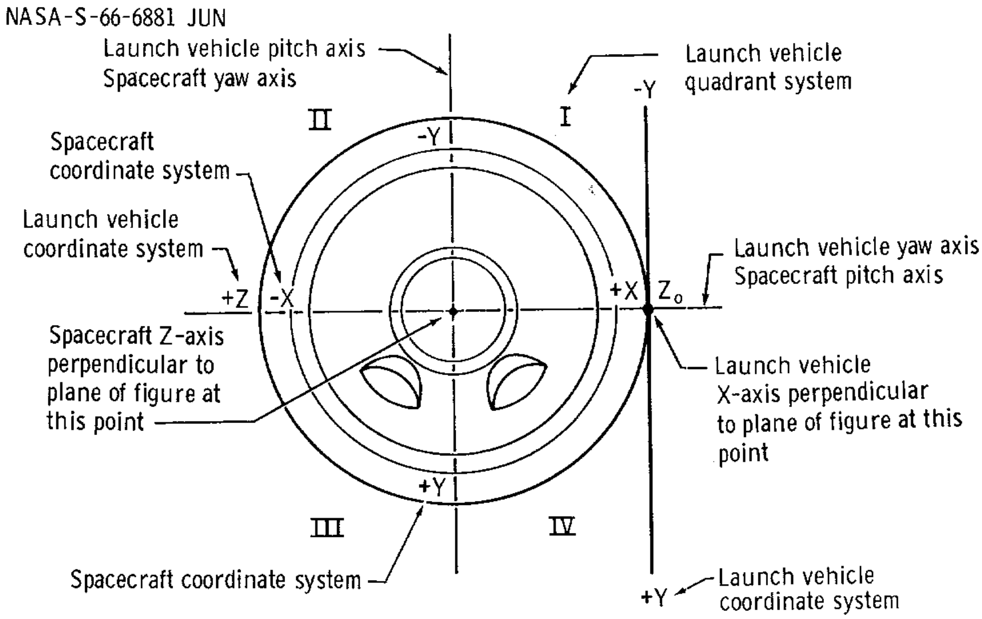
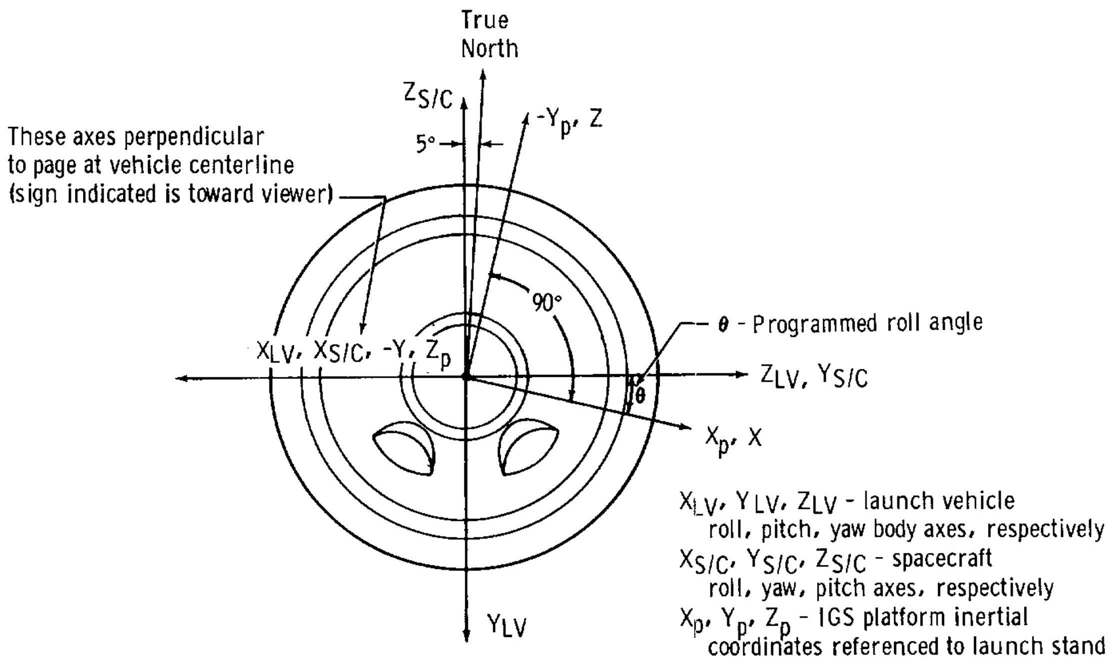


(a) Launch configuration.

Figure 3.0-1. - GLV - spacecraft relationships.



Dimensional axes



Guidance coordinates

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

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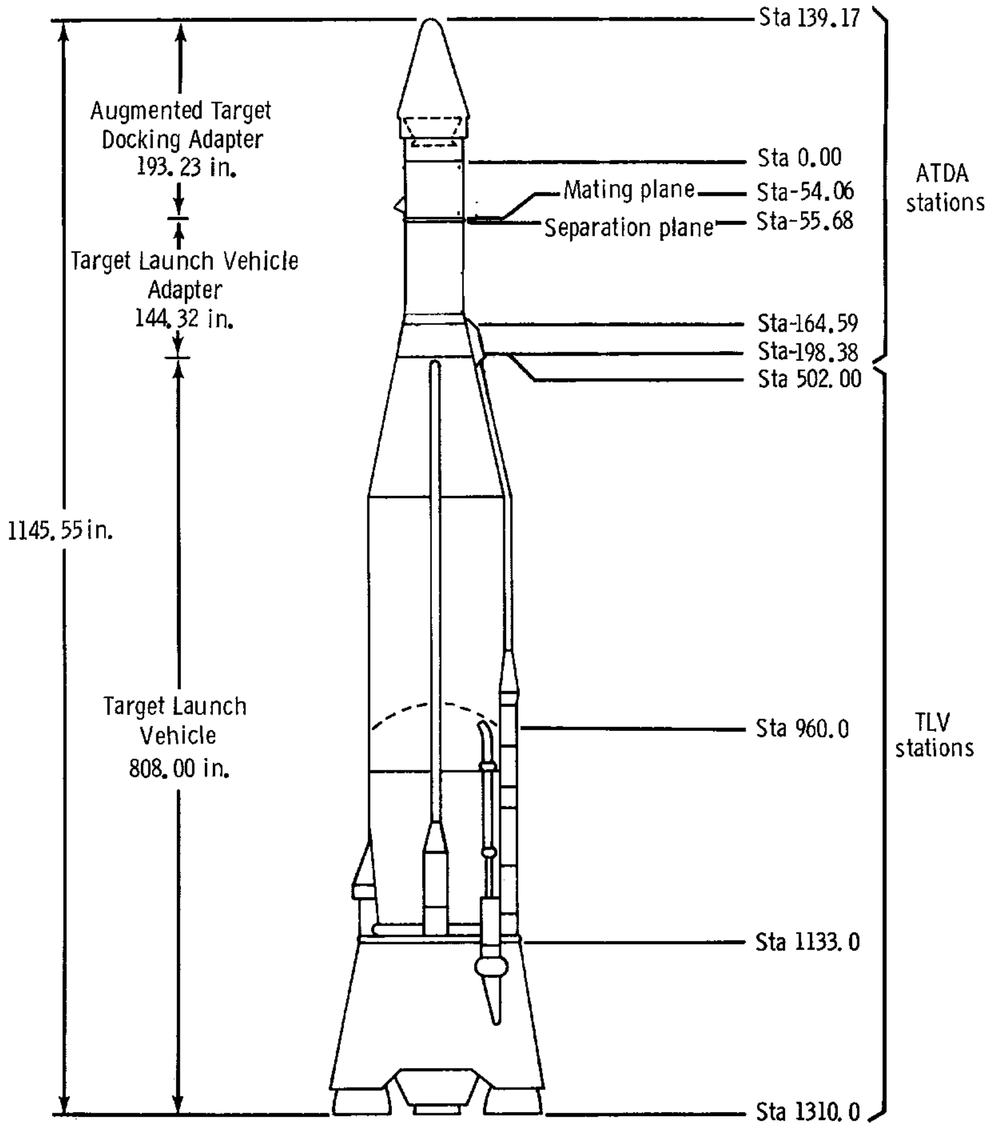

(a) Launch configuration.

Figure 3.0-2. - TLV/ ATDA relationship.

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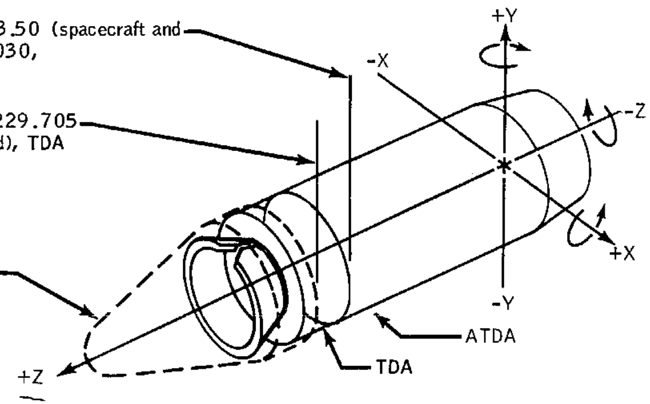
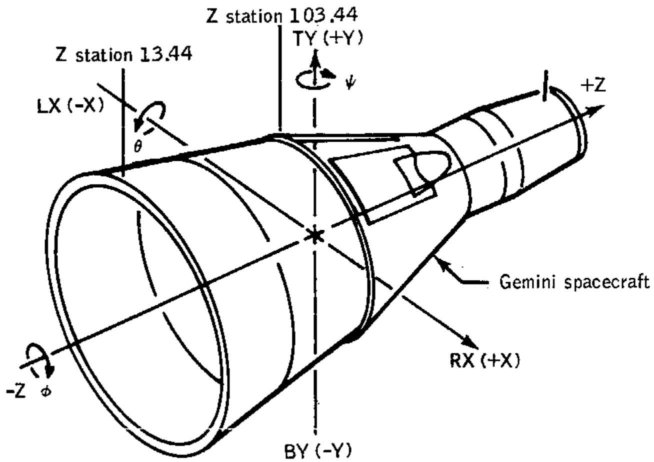
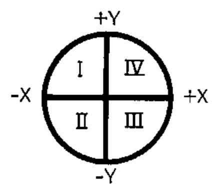
Note:

1. The coordinate axes for the TDA are the same as that shown for the Gemini spacecraft in the rigidized configuration.
2. Positive sense of axes and angles are indicated by arrows.





ATDA quadrant designations
(looking toward docking cone)



1. Spacecraft- ATDA contractor design and weights group coordinate system
   - +Y up in direction of crew's head (yaw axis)
   - +Z forward in direction crew is facing (roll axis) spacecraft
   - -Z forward in direction crew is facing (roll axis) ATDA
   - +X in direction of crew's right arm (pitch axis)
2. Spacecraft- ATDA contractor guidance and control mechanics and aerodynamics groups coordinate system
   - -Z up in direction of crew's head (yaw axis)
   - +X forward in direction crew is facing (roll axis)
   - +Y in direction of crew's right arm (pitch axis)

(b) Dimensional axes and guidance coordinates, ATDA.
Figure 3.0-2. - Continued.

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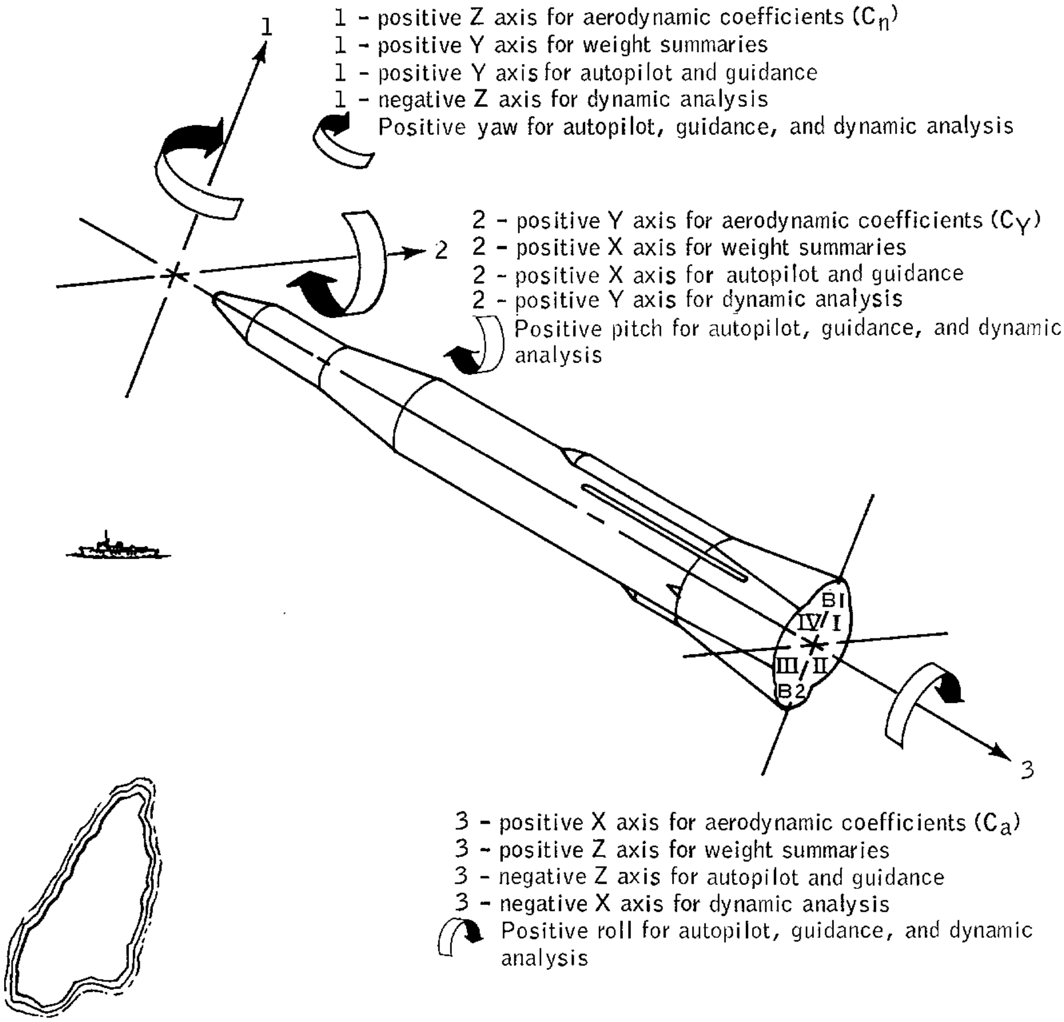

Vehicle shown in flight attitude

(c) Dimensional axes and guidances coordinates, TLV.

Figure 3.0-2. - Concluded.

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3.1 GEMINI SPACECRAFT

The structure and major systems of Spacecraft 9 (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 9 describe the modifications incorporated into the subsequent spacecraft. Except for the window covers and the extravehicular equipment, Spacecraft 9 closely resembled Spacecraft 8 (ref. 9), and only the significant differences (table 3.1-I) between those two spacecraft are included in this report. A detailed description of Spacecraft 9 is contained in reference 10.

3.1.1 Spacecraft Structure

The primary load-bearing structure of Spacecraft 9 was essentially the same as that of Spacecraft 8. However, the few major changes are described in the following paragraphs.

An externally mounted transparent cover for each hatch window was installed to protect the windows from the film encountered during the powered phase of flight. The covers were outer spacecraft windows removed from spacecraft flown on previous Gemini missions. In addition, each cover incorporated two hook-type hinges inserted in the hinge pins mounted on the hatch, two torsion springs for jettisoning, and a machined tab for attachment of the latch-release mechanism in the hatch. A molded gasket seal between the cover frame and outer window frame was attached to the cover frame to afford a positive parting plane at cover jettison. The covers were jettisoned manually by rotating a latch-release knob mounted inside the spacecraft on each hatch after the spacecraft was inserted into orbit.

The Astronaut Maneuvering Unit (AMU) was mounted in the adapter section in place of the Extravehicular Support Package (ESP) installed on Spacecraft 8. Also, a spring-clip device was added to the right-hand handhold assembly in the adapter AMU installation to restrain the 25-foot umbilical to the handhold for donning the AMU. (For additional changes required to support the extravehicular activities, see paragraph 3.1.2.12.)

3.1.2 Major Systems

3.1.2.1 <u>Communications System</u>.- The Communications System was basically the same as the one used on Spacecraft 8, including the voice control center modification to permit recording of the pilot's voice while communicating with the command pilot during EVA. This modification also permitted the recording of UHF communications from the ground.

3.1.2.2 Instrumentation and Recording System.- Wiring changes were incorporated to permit recording of AMU telemetry data while the AMU was in the stowed configuration. The changes provided the capability to extend the AMU telemetry antennas (by initiating the EVA BARS EXTEND switch) and to record the telemetry data on track B of the tape recorder. There was no playback capability; therefore, the data could be reduced only after recovery of the spacecraft.

3.1.2.3 Environmental Control System.- The Environmental Control System (ECS) was similar to the one used on Spacecraft 8.

3.1.2.4 Guidance and Control System.- The Guidance and Control System, including the Auxiliary Tape Memory Unit, was basically the same as the Spacecraft 8 system except for the following changes.

During previous missions, the horizon sensor system has been susceptible to certain combinations of atmospheric and earth-climate conditions. These conditions have caused reduced accuracy of the pitch and roll output signals. On Spacecraft 9, the secondary horizon sensor system was modified to include a narrower band-pass optical system for flight evaluation. This modification replaced the 8-micron filter and collimator with a narrow band (13 to 22 microns to roll off of germanium) optical coating. This spectral band-pass selection reduced the erroneous inputs by rejecting the ozone absorption band and the water-absorption continuum window.

The OAMS CNTL POWER switch was changed to a four-pole single-throw lever-lock switch to provide the crew with the capability to remove power from all Orbital Attitude and Maneuver System (OAMS) thrusters by actuating one switch.

To preclude the firing of an OAMS thruster by a shorted internal connection, all exposed connections in the orbital attitude and maneuver electronics (OAME) package were insulated with a light coat of varnish or epoxy.

3.1.2.5 Time Reference System.- The Time Reference System configuration was the same as the one used on Spacecraft 8.

3.1.2.6 Electrical System.- The Electrical System included a fuel-cell power system that was the same as the Spacecraft 8 system.

3.1.2.7 Propulsion System.- The Propulsion System (figs. 3.1-2 and 3.1-3) was the same as the system flown on Spacecraft 8.

3.1.2.8 Pyrotechnic System.- Except for deletion of the pyrotechnic devices associated with the Gemini VIII experiments, the Pyrotechnic System was similar to the one used on Spacecraft 8.

3.1.2.9 Crew-station furnishings and equipment.- The following changes were incorporated into 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 also included minor changes in the nomenclature of indicators and switch positions (see fig. 3.1.4).

(a) Both attitude indicators were changed to include the improved quantitative markings used on the command pilot's attitude indicator on Spacecraft 8.

(b) Each of the six ammeters on the fuel-cell monitor and control panel were used to monitor the fuel-cell stack currents. (On Spacecraft 8, the two center ammeters monitored the two main-bus currents.)

(c) The MMU DEPLOY-OFF-TM ON switch replaced the BACK-PACK DEPLOY-SAFE switch that was installed on Spacecraft 8.

(d) The elapsed-time digital clock display light was dimmed mechanically (using a hood assembly) instead of electrically (using a potentiometer).

(e) The MMU hydrogen-peroxide warning indicator was added to the annunciator panel on the center panel.

(f) Displays and controls were installed as required for the experiments (see section 8.0).

(g) Eight switches located on the center and lower instrument panels and on the right circuit-breaker panel were replaced by switches which had increased pushbutton travel and actuator plate spring preload. The change was the result of one of this type switch having been pressure sensitive on Spacecraft 6 and thereby not actuating properly.

3.1.2.9.2 Miscellaneous equipment changes: The following changes were made in the spacecraft cabin.

(a) A single quick-release pin capable of simultaneously stowing the ejection control handle and safetying the ejection control mechanism replaced the safety pin and quick-release stowage pin used on previous spacecraft.

(b) The spacecraft hatch-holding device was modified to permit rigging the hatches with a maximum closing force of 40 pounds. Also, a loop was added to the end of the fabric strap on the pilot's hatch-closing device to provide a means for the command pilot to assist in closing the right-hand hatch.

3.1.2.9.3 Stowage facilities: The stowage containers are shown in figure 3.1-5. Table 3.1-II lists the major items of equipment stowed in the containers at launch.

3.1.2.10 Landing System.- No significant changes were made to the Landing System.

3.1.2.11 Postlanding and Recovery System.- No significant changes were made to the Postlanding and Recovery System.

3.1.2.12 Extravehicular equipment.- The following modifications were incorporated in the spacecraft and space suit to support the extravehicular activities (EVA). In addition, the Extravehicular Life Support System (ELSS) and the AMU were provided to equip the pilot for the EVA.

3.1.2.12.1 Structural modifications: The structural modifications (handrail, Velcro patches) installed on Spacecraft 8 were included on Spacecraft 9. The adapter modifications to provide handholds, foot supports, and floodlighting were also on Spacecraft 9. In addition, two aluminum stirrups were installed on the footbar to provide foot restraint during AMU donning activities. The modifications which were made to the adapter-equipment-section thermal curtain to accommodate the ESP on Spacecraft 8 were the same as those required on Spacecraft 9 for the AMU (see fig. 3.1-6).

3.1.2.12.2 Space suits: The space-suit configuration for the command pilot was basically the same as that used on the Gemini VIII mission — a G4C suit with a light coverlayer. The helmet pressure visor for both crewmen was fabricated from a polycarbonate material which is more resiliant than plexiglass.

The space suit for the pilot was also a G4C suit but included an extravehicular coverlayer and a modified overvisor. The EVA coverlayer was modified to provide protection to the legs from the high-temperature, high-velocity plume impingement of the lower forward-firing and the downward-firing thrusters of the AMU.

The visor modifications included a polycarbonate pressure visor and a single-lens sun visor in lieu of the multi-lens unit previously provided. The use of the single-lens sun visor was possible because the polycarbonate pressure visor provided the impact protection required in the sun visor. A low-emittance coating was applied to the exterior surface of the pressure visor to prevent interior surface cooling during nighttime EVA.

3.1.2.12.3 Extravehicular Life Support System: The ELSS was a semi-open-loop system utilizing externally supplied oxygen for ventilation and for removal of carbon dioxide. This system, contained in a chestpack, was the same as the ELSS flown on the Gemini VIII mission, but with the following modifications:

(a) The umbilical configuration was modified such that, when the umbilical was attached to the chestpack oxygen fitting and the electrical connection, the umbilical assembly extended forward of the pilot instead of aft.

(b) Bypass flow was rerouted downstream of the ejector throat by a tube from the existing bypass valve.

(c) The chestpack wiring was altered to permit paralleling of the ELSS power switching relay contacts.

(d) A light was added to the chestpack panel to indicate the use of either spacecraft or ELSS power. On the Spacecraft 8 ELSS, the RCS AMU warning light illuminated when the power source changed from the spacecraft power supply to the ELSS battery. The Gemini IX-A configuration permitted the use of the RCS light for AMU status reporting and the additional light (S/C POWER) illuminated when spacecraft power was utilized and extinguished when a switchover to ELSS battery power occurred.

3.1.2.12.4 Astronaut Maneuvering Unit: The AMU was a highly compact unit consisting of a basic structure and six major systems. These systems were: propulsion, flight control, oxygen supply, power supply, malfunction detection, and communications.

The structure consisted of a backpack shell, two folding sidearm controllers, and folding nozzle extensions. The size of the backpack was determined primarily by the size of the hydrogen peroxide, oxygen, and nitrogen tanks. The thrusters were located in the corners of the structure to provide controlling forces and moments about the center of gravity of the entire AMU. The remainder of the components were located in the available space inside the backpack. The total volume and shape were further constrained by the stowage location in the equipment of the adapter section. A thermal curtain covered the stowage cavity to provide passive temperature control for the AMU. It was jettisoned prior to the start of AMU donning activities. As a part of the donning activities, the pilot unfolded the nozzle extensions and sidearm controller heads. This permitted the control handles to be in an accessible position.

The evaluation of the AMU was an integral part of an experiment; therefore, for a detailed description of the AMU, see section 8.1, EXPERIMENT D-12, ASTRONAUT MANEUVERING UNIT.

TABLE 3.1-I.- SPACECRAFT 9 MODIFICATIONS

System	Significant differences between Spacecraft 9 and Spacecraft 8 configurations
Structure	(a) A transparent cover for each hatch window was installed. (b) EVA provisions were modified.
Communications	No significant difference.
Instrumentation and Recording	(a) Additional telemetry of the OAMS was installed. (b) Wiring changes were incorporated to permit recording AMU telemetry data while the AMU was stowed. (c) Additional telemetry parameters were installed to provide data for evaluation of the ATMU.
Environmental Control	A modification was incorporated to eliminate air from the drinking water system.
Guidance and Control	The secondary horizon sensor system was modified to include a narrower band-pass optical system for flight evaluation.
Time Reference	No significant difference.
Electrical	No significant difference.
Propulsion	(a) A switch and associated circuitry were added to permit the crew to remove power from all OAMS thrusters by actuating one switch. (b) All exposed connections in the OAME package were insulated.
Pyrotechnics	No significant difference.

TABLE 3.1-I.- SPACECRAFT 9 MODIFICATIONS - Continued

System	Significant differences between Spacecraft 9 and Spacecraft 8 configurations
Crew-station furnishings and equipment	(a) Both attitude indicators had improved quantitative markings. (b) Each of the six ammeters on the fuel-cell control panel were used to monitor fuel-cell stack currents. (c) An MMU switch replaced the BACK-PACK switch. (d) An MMU H_2O_2 indicator was added to the center panel. (e) A mechanical dimmer was installed on the elapsed-time digital-clock-display light. (f) Displays and controls were added as required to support the experiments. (g) A single quick-release pin capable of simultaneously stowing the ejection control handle and safetying the mechanism was used. (h) The hatch-closing device and fabric strap were modified to allow the command pilot to assist in closing the hatch. (i) Both helmet pressure visors were made of a polycarbonate material.
Landing	No significant change.
Postlanding and Recovery	No significant change.
EVA equipment	(a) Foot restraints were mounted on the adapter foot support. (b) The pilot's sun visor was a single-lens type.

TABLE 3.1-I.- SPACECRAFT 9 MODIFICATIONS - Concluded

System	Significant differences between Spacecraft 9 and Spacecraft 8 configurations
EVA equipment - concluded	(c) The pilot's EVA coverlayer was modified to provide protection from AMU thruster firing. (d) The AMU replaced the ESP. (e) The ELSS bypass flow was rerouted downstream of the ejector throat. (f) The ELSS wiring was altered to permit paralleling of the power switching contacts. (g) The ELSS RCS light was used for AMU status reporting and a new S/C POWER light indicated which power source was in use.

TABLE 3.1-II.- CREW-STATION STOWAGE LIST

Stowage area (See fig. 3.1-4)	Item	Quantity
Centerline stowage container	70-mm camera	1
	16-mm camera	2
	70-mm camera, super-wide angle, with film magazine	1
	18-mm lens, 16-mm camera	1
	75-mm lens, 16-mm camera	1
	5-mm lens, 16-mm camera	1
	Lens, f/2.8	1
	16-mm film magazine	10
	70-mm film magazine	3
	Ring viewfinder	1
	Sighting device	1
	Mirror mounting bracket	1
Left sidewall containers	Personal hygiene towel	1
	Waste container	1
	Urine receiver	1
	Penlight	1
	Defecation device	1
	Voice tape cartridge	5
	Velcro pile, 2 by 6 in.	1
	Velcro hook, 2 by 6 in.	1
	Plastic zipper bag	6
	Urine hose assembly	1
	Bandolier	1
	Urine sample bag	6
	Adjustable wrench, 6 in.	1
	Pilot's preference kit	1

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

Stowage area (See fig. 3.1-4)	Item	Quantity
Left aft stowage container	Components for EVA consisting of:	
	EVA remote control cable, 16-mm camera	1
	ELSS umbilical assembly	1
	Jumper cable	1
	Dual connector	2
	ELSS restraint assembly	2
	Pressure thermal gloves	1 pr.
	EVA rearview mirror	1
	EVA wrist mirror and band	1
	Thermal cover, 16-mm camera	1
	EVA remote control cable	1
	EVA hand pad	2
	Movie camera adapter	1
	ELSS hose, short, with interconnector	1
	ELSS hose, long, with interconnector	1
	Extension cable assembly	1
	Hose assembly, red, 9 in.	1
	Hose assembly, blue, 9 in.	1
	Pouch, harness, electrical cable, "Y" connector	1
	Hose nozzle interconnector	1
Left pedestal	Lightweight headset with oral temperature probe	1
	Velcro, 8 by 1 in.	4

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

Stowage area (See fig. 3.1-4)	Item	Quantity
Left footwell	Personal hygiene towel	1
	Tissue dispenser	1
	Auxiliary window shade	1
	Reflective window shade	1
Right sidewall containers	Inflight medical kit	1
	Personal hygiene towel	1
	Waste container	1
	Penlight	1
	Defecation device	1
	Voice tape cartridges	8
	Velcro pile, 2 by 6 in.	1
	Velcro hook, 2 by 6 in.	1
	Plastic zipper bag	6
	Oral hygiene kit	1
	Surgical scissor	1
	Debris cutter	1
	Urine sample bag	6
	Pilot's preference kit	1
Right aft stowage container	Mirror mounting bracket	1
	16-mm camera bracket	1
	18-mm lens, 16-mm camera	1
	70-mm camera	1
	16-mm film magazine	10
	Spotmeter and exposure dial	1
	70-mm film magazine	4

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

Stowage area (See fig. 3.1-4)	Item	Quantity
Right aft stowage container - concluded	Postlanding kit assembly	1
	Manual inflator, blood pressure	1
	Waste container	2
	Defecation device	4
	Circuit breaker module	2
	Urine sample bag	12
	Zodiacal camera	1
	Mounting bracket, 70-mm camera	1
Right pedestal	Lightweight headset with oral temperature probe	1
	Velcro, 8 by 1 in.	4
Right footwell	Orbital-path display assembly	1
	Celestial display - Mercator	1
	Personal hygiene towel	1
	Tissue dispenser	1
	Auxiliary window shade	1
	Reflective window shade	1
Plotboard pouch	Flight data book	2
	Flight booklet	1
	Flight data cards	2
	Splash curtain clip	6
	Transparent reticle	1
Orbital utility pouch	Helmet stowage bag	2

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

Stowage area (See fig. 3.1-4)	Item	Quantity
Right and left circuit breaker fairings	Clamps for urine collection device	2
	Latex roll-on cuff (urine system)	4
	Tape, 3/4 in.	10 ft
	Urine receiver removable cuff	2
	Tape, 1/2 in. by 10 ft	2
	Glareshield	1
Center stowage rack	ELSS chestpack	1
Hatch torque box	Sextant	1
	Lens, 50-mm, f/0.95	1
	Objective filter, 50-mm lens	1
Water management console	Roll-on cuff receiver (urine system)	1

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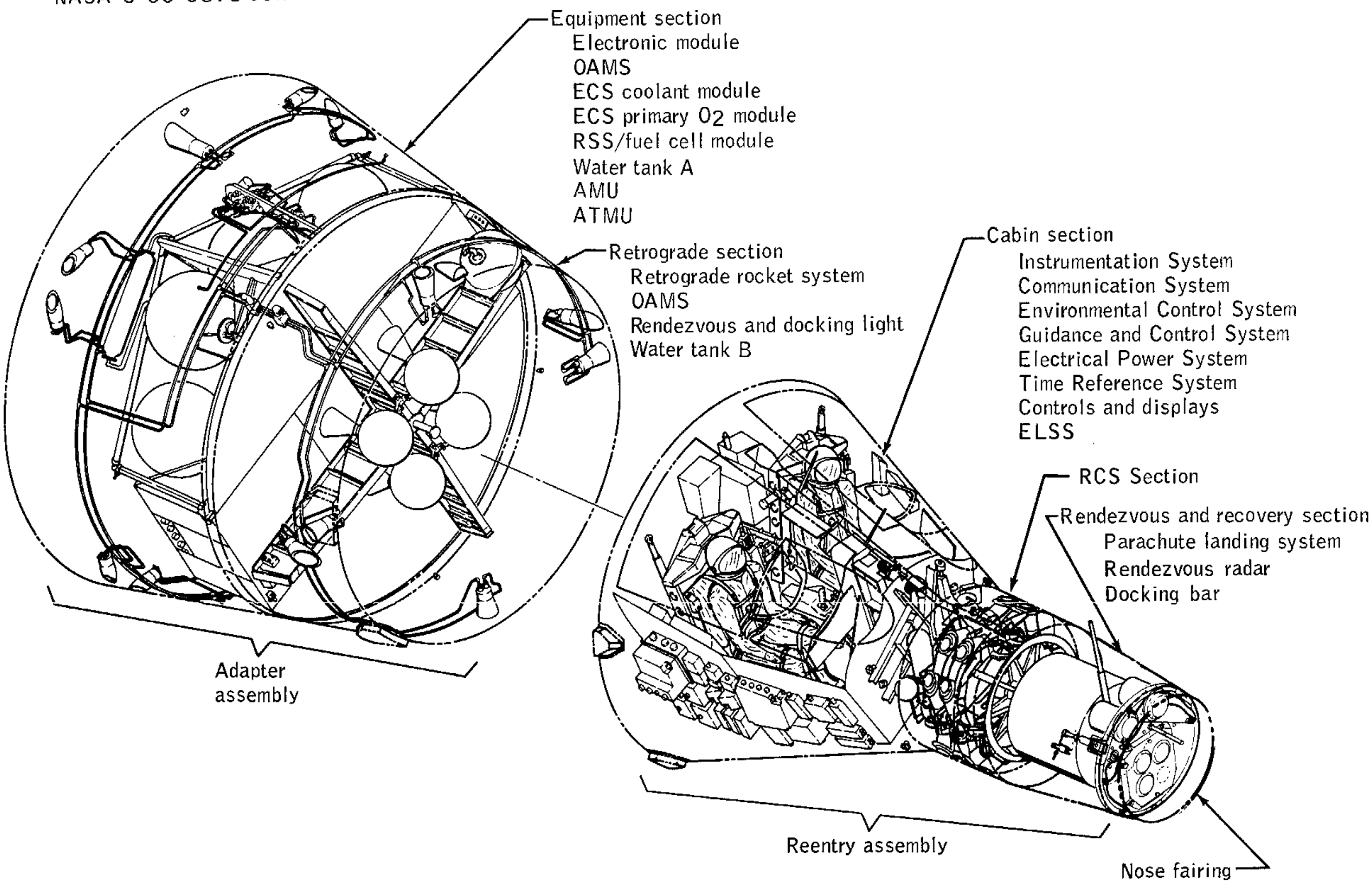

Figure 3.1-1. - Spacecraft arrangement and nomenclature.

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OAMS thrusters		
5	6	Pitch up
1	2	Pitch down
3	4	Yaw right
7	8	Yaw left
3	7	Roll clockwise
4	8	Roll counterclockwise
9	10	Translate forward
11	12	Translate aft
	13	Translate right
	14	Translate left
	15	Translate up
	16	Translate down

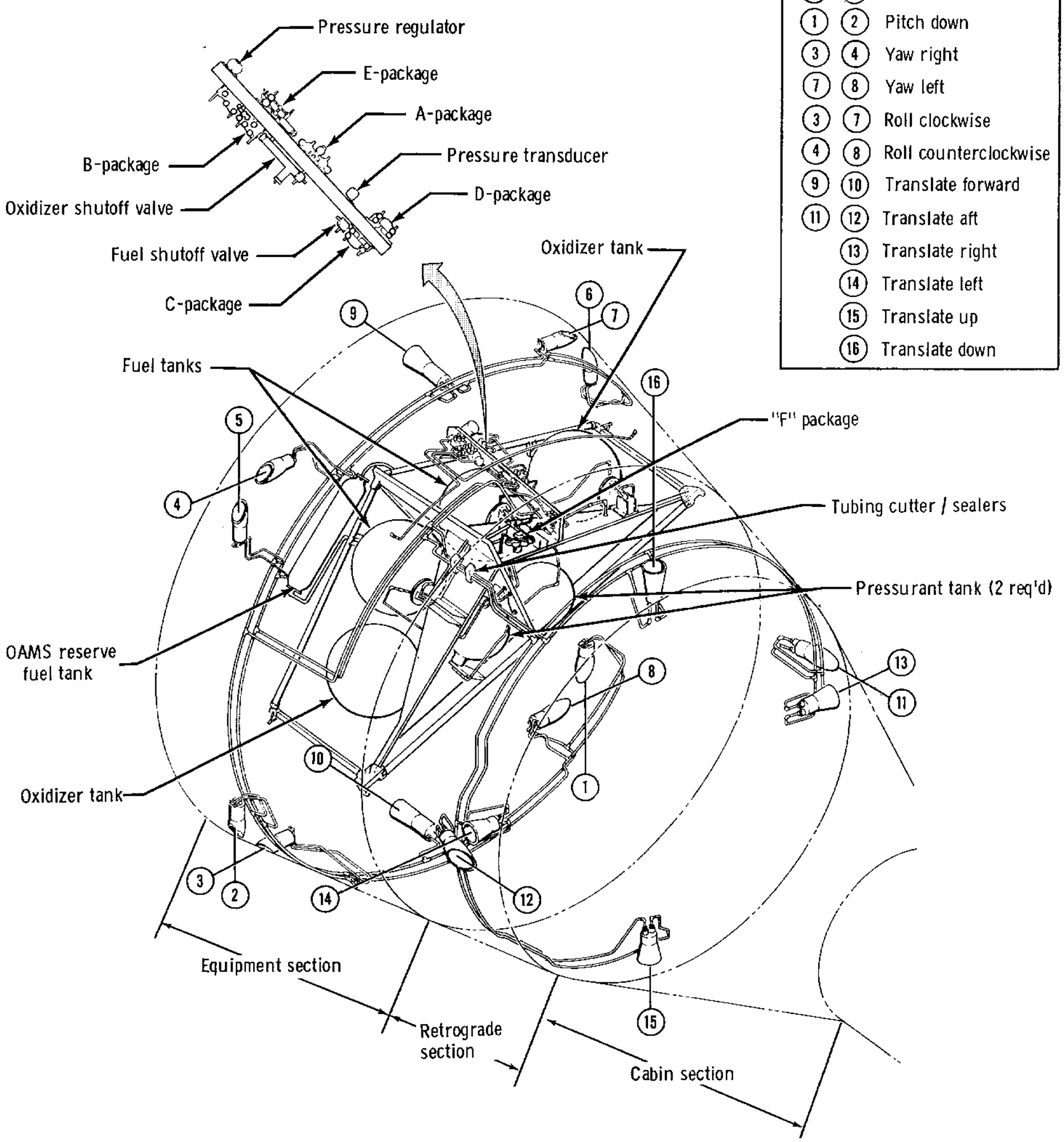

Figure 3. 1-2. - Orbital Attitude and Maneuver System.

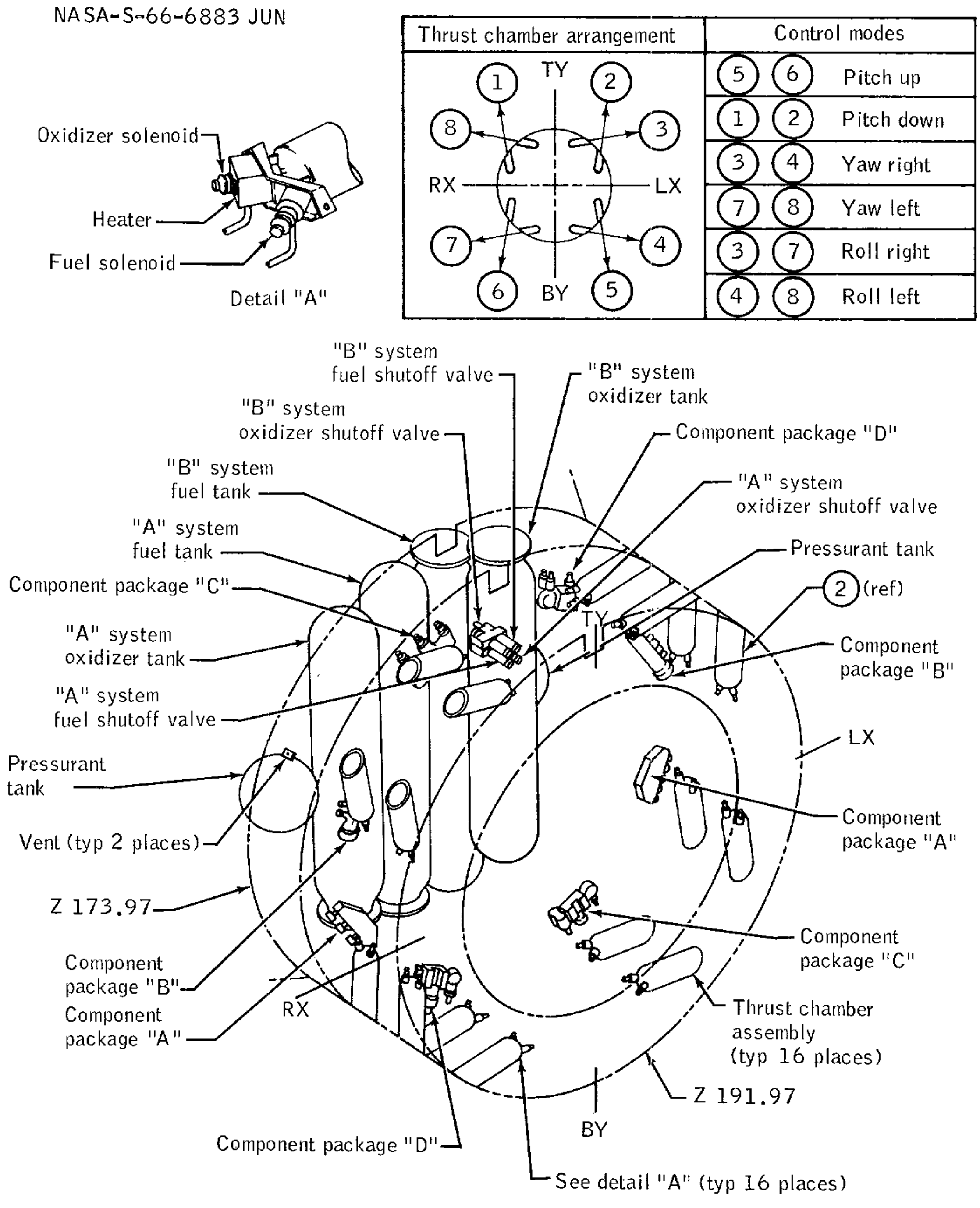

Figure 3.1-3. - Reentry Control System.

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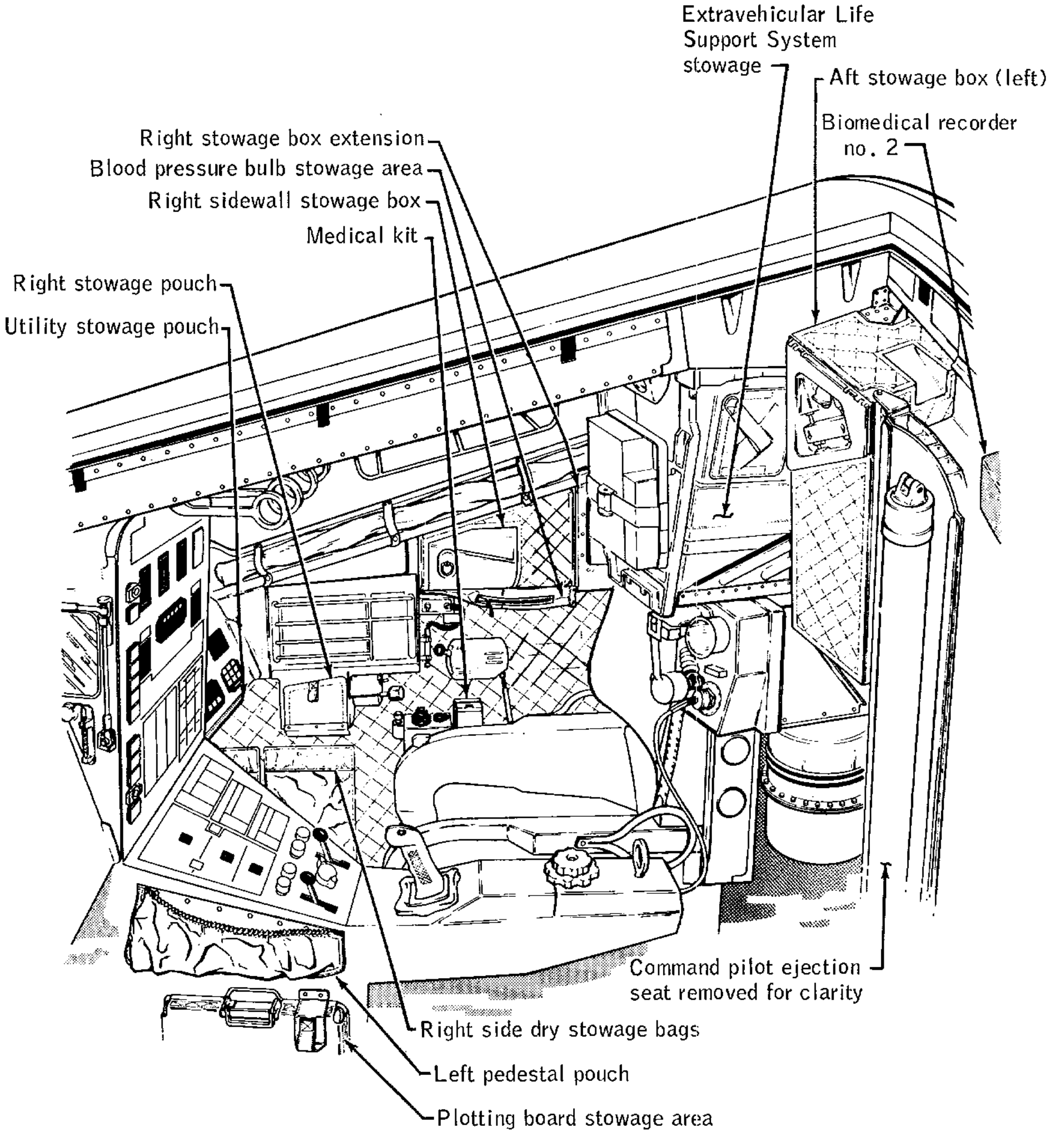

(b) View looking into pilot's side.

Figure 3.1-5.- Concluded.

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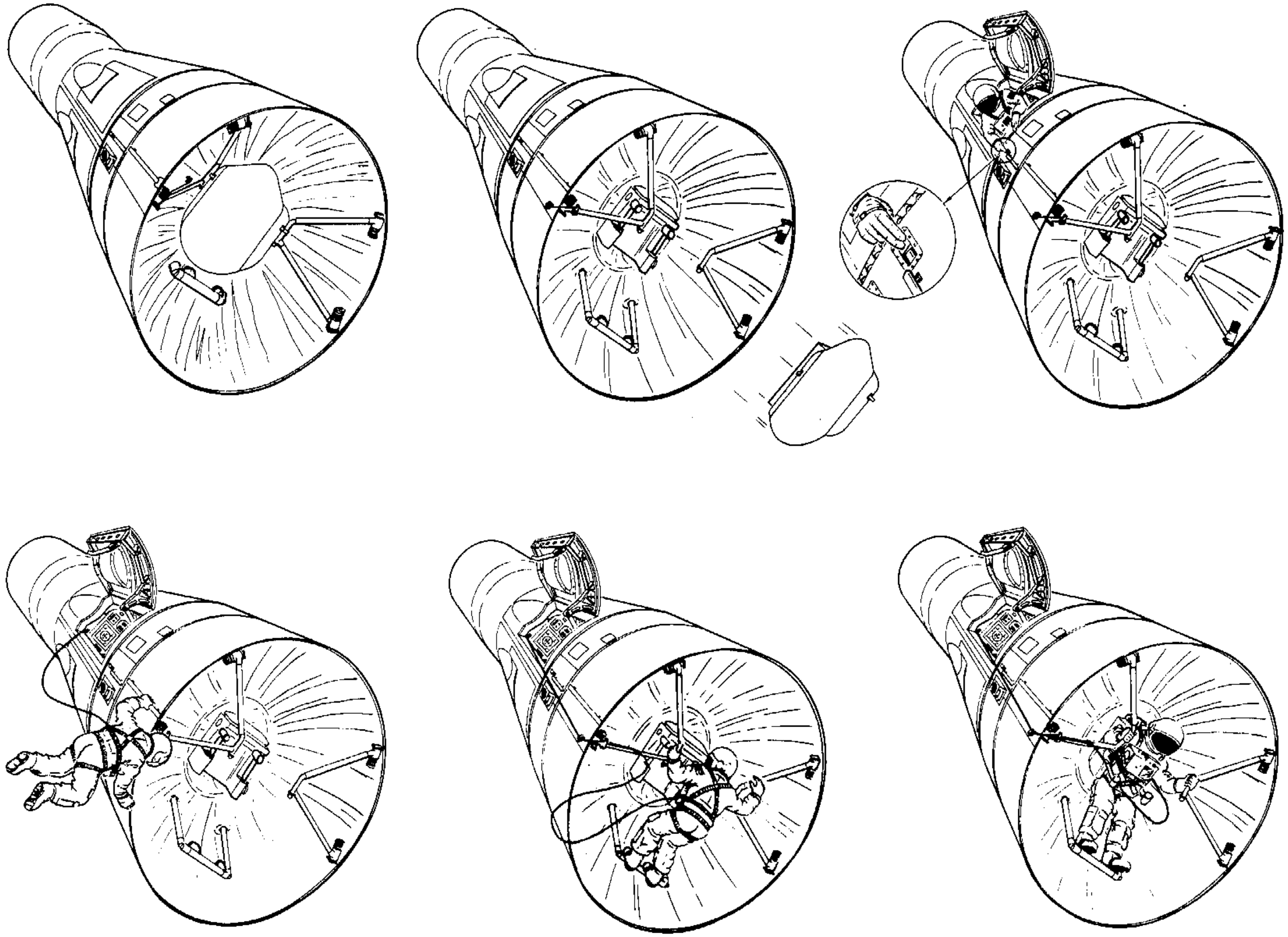

Figure 3.1-6. - Planned sequence for donning Astronaut Maneuvering Unit.

3.2 GEMINI LAUNCH VEHICLE

The Gemini Launch Vehicle (GLV-9) was of the same basic configuration as those used for previous Gemini missions. Table 3.2-I lists the significant differences between GLV-9 and GLV-8 (ref. 9).

TABLE 3.2-I.- GLV-9 MODIFICATIONS

System	Significant differences between the GLV-9 and GLV-8 configurations
Structure	(a) The skin cutout for compartment 3 air conditioning was deleted. (b) High-strength bolts replaced the AN-4 bolts used previously.
Propulsion	(a) The redundant propellant-level sensors at the outage position were deleted from both stages. (b) The TCPS function was removed from the Stage I engine. Launch bolt release and staging signals were received from MDTCPS.
Flight Control	No significant change.
Radio Guidance	No significant change.
Hydraulic	No significant change.
Electrical	The Stage I TCPS sensing circuit was changed to a MDTCPS sensing circuit.
Malfunction Detection	Separate IPS and APS busses were provided for MDTCPS on Stage I subassemblies 1 and 2.
Instrumentation	Two measurements from the MDTCPS APS/IPS busses were added.
Range Safety and Ordnance	No significant changes.

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3.3 WEIGHT AND BALANCE DATA

Weight and balance data for the Gemini IX-A Space Vehicle are as follows:

Condition	Weight (including spacecraft), lb (a)	Center-of-gravity location, in. (a), (b)		
		X	Y	Z
Ignition	344 225	773.7	-0.050	59.96
Lift-off	340 732	774.0	-0.050	59.96
Stage I burnout (BECO)	86 224	441.0	-0.180	59.48
Stage II start of steady-state combustion	74 509	343.5	-0.05	59.45
Stage II engine shutdown (SECO)	14 222	281.5	-0.230	59.13

[a]Weights and center-of-gravity data were obtained from 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 Y-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 9 weight and balance data are as follows:

Condition	Weight, lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Launch, gross weight	8268	-0.13	+1.84	104.84
Retrograde	5635	+0.03	-1.11	130.01
Reentry (0.05g)	4874	+0.13	-1.48	136.16
Main parachute deployment	4475	+0.10	-1.65	129.46
Landing (no parachute)	4364	+0.10	-1.65	127.43

[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

Section 3.4 is not applicable to the Gemini IX-A mission.

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3.5 TARGET LAUNCH VEHICLE

The Target Launch Vehicle (TLV-5304) was an Atlas Standard Launch Vehicle (SLV-3) and was of the same basic configuration as the TLV-5302 used for the Gemini VIII mission (ref. 9). Table 3.5-I lists the significant differences between TLV-5304 and TLV-5302. Also included are the modifications incorporated on TLV-5304 to support the Augmented Target Docking Adapter (ATDA) which replaced the Gemini Agena Target Vehicle (GATV) for this mission. These modifications are described further in the following paragraphs.

3.5.1 Structure

No significant primary structural changes were required; however, modifications to the TLV adapter were required to support the ATDA. The shaped-charge ring located near the forward end of the adapter section and used to separate the ATDA from the TLV was ignited by ground command through a command receiver mounted in the ATDA equipment section.

The weight of the TLV adapter was reduced by approximately 42 pounds (from the GATV configuration) by removing all equipment not required to support the ATDA mission. Equipment removed included the GATV jettison rails, pyrotechnic cover, retrorockets and associated fairings, electrical harness, battery relay box, destruct package, and miscellaneous clips and brackets. Vent holes in the forward access panel were enlarged, and the debris shield at the forward end of the adapter was replaced by a thermal shield.

3.5.2 Major Systems

3.5.2.1 Propulsion System.- A 0.15-inch spacer and gasket were added between the fuel duct and probe on the fuel staging valve to increase the nominal staging poppet valve opening to 1.72 inches. Test data and analysis of a failure on another Atlas flight revealed that the minimum poppet valve opening must be 1.6 inches.

The sustainer liquid oxygen and fuel weep holes were modified to eliminate the recurrence of a corrosion problem which could cause pinhole leaks.

3.5.2.2 Hydraulic System.- Four hydraulic pressure transducers were replaced, and four associated current regulators were added to eliminate recurring wiper lift off which had been caused by response to low-level, high-frequency pressure fluctuations.

ATDA-peculiar modifications: The sustainer lubrication-oil tank was replaced with a larger tank. Installation of this tank extended the sustainer operating time to 368 seconds, which was sufficient to support the longer powered flight necessary to place the ATDA into orbit.

3.5.2.3 Guidance System.- The Guidance System antenna configuration for the ATDA mission was modified to obtain the desired radiation pattern by installing an antenna identical to that used on Mercury-Atlas Launch Vehicles. Also, the antenna waveguides were modified to interface with the different antenna.

3.5.2.4 Flight Control System.- The vernier-engine servoamplifier was modified to change the vernier-engine bias angle from 50 to 45 degrees in yaw. This 5-degree margin eliminated the control deadband zone which could have caused steady-state oscillation during the vernier solo phase of the flight.

3.5.2.4.1 Autopilot: The null integrator circuit was no longer required; therefore, the diodes were removed and the switch was deactivated. Also, the programmer was modified to keep the integrator circuits active from lift-off until programmer reset.

3.5.2.4.2 ATDA-peculiar modifications: The ATDA direct-ascent trajectory required end-to-end gains, frequency-response characteristics, and a pitch program which were peculiar to the TLV control system for the ATDA mission.

New end-to-end gains were incorporated by modifying the displacement gyro package; the rate gyro package was not changed. The programmer package was revised to incorporate pitch-program requirements and revised switching requirements. One signal wire from the programmer to the displacement gyro package was added for gain change. Setting of the backup staging accelerometer was changed to the new nominal staging acceleration value.

3.5.2.5 Electrical System.- The distribution box was modified to monitor the three-phase, 400-cycle ac input. If fluctuation did occur, the programmer was reset. All electrical interface harnesses between the TLV and TLV adapter were removed for the ATDA mission.

3.5.2.6 Pneumatic System.- An internal helium pressure transducer and the associated telemetry wiring were added to confirm that the helium-shutoff valve had operated prior to lift-off. (This valve failed to shut off during a previous Atlas flight.)

ATDA-peculiar modifications: Liquid-oxygen tank-pressure oscillations at lift-off are larger for lighter TLV payloads. Because of the lighter payload of the ATDA mission, the differential pressure across the intermediate bulkhead could have been lower than the minimum design value. Therefore, pressure schedules for the TLV were changed as follows:

(a) The liquid-oxygen-tank pressure regulator was adjusted from a range of 28.5 to 31.0 psig to a range of 24.7 to 26.0 psig.

(b) The boiloff valve was adjusted from a range for cracking pressure of 4.7 to 5.8 psig to a range for cracking pressure of 3.0 to 4.0 psig.

(c) The relief valve was adjusted to reduce the cracking pressure from the range of 32.1 to 34.7 psig to the range of 27.1 to 29.8 psig.

3.5.2.7 Instrumentation System.- The telemetry system of the Instrumentation System was modified to support the changes described in paragraphs 3.5.2.4 and 3.5.2.5.

3.5.2.8 Range Safety Command System.- The Range Safety Command System was modified to delete the destruct package from the TLV adapter for the ATDA mission.

TABLE 3.5-I.- TLV-5304 MODIFICATIONS

System	Significant differences between TLV-5304 and TLV-5302 configurations
Structure	All equipment not required to support the ATDA mission was removed from the TLV adapter, and the debris shield at the forward end was replaced by a thermal shield. (ATDA only)
Propulsion	1. The fuel staging poppet valve was modified to increase the opening to 1.72 inches. 2. Four hydraulic pressure transducers were replaced, and four associated current regulators were added to eliminate wiper liftoff in the transducers. 3. A larger lubrication-oil tank was installed. (ATDA only)
Guidance	The antenna and associated waveguides were modified to obtain desired radiation pattern. (ATDA only)
Flight Control	1. The vernier-engine servoamplifier was modified to change the vernier-engine bias angle. 2. The displacement gyro package and programmer package were modified to obtain mission-peculiar flight control. (ATDA only)
Electrical	1. The distribution box was modified to monitor the three-phase, 400-cycle ac input. 2. All electrical interface harnesses between the TLV and TLV adapter were removed. (ATDA only)
Pneumatic	1. An internal helium pressure transducer and associated telemetry wiring were added to confirm that the helium-shutoff valve had closed prior to lift-off. 2. Pressure schedules were changed. (ATDA only)
Instrumentation	Modifications were incorporated to monitor the three-phase, 400-cycle ac input to the electrical distribution box and the closure of the helium shutoff valve.
Range Safety	The destruct package was removed from the TLV adapter. (ATDA only)

3.6 WEIGHT AND BALANCE DATA

Weight and balance data for the Target Launch Vehicle/Augmented Target Docking Adapter (TLV/ATDA) were as follows:

Condition	Weight (including ATDA), lb (a)	Center-of-gravity location, in. (a)		
		X	Y	Z
Ignition	266 371	--	--	--
Lift-off	264 320	0.4	-0.5	850.9
Booster engine cutoff (BECO)	76 860	-1.77	1.35	977.6
Sustainer engine cutoff (SECO)	9 571	-4.08	7.25	862.4

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

The ATDA weight and balance data were as follows:

Condition	Weight, lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Launch, gross weight	2 006	0.3	0.1	-26.3
Separation	(b)	(b)	(b)	(b)
Insertion weight (in-orbit)	(b)	(b)	(b)	(b)

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

[b]Not available because shroud did not jettison.

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3.7 AUGMENTED TARGET DOCKING ADAPTER

The Augmented Target Docking Adapter (ATDA) was utilized on the Gemini IX-A mission as a target vehicle for rendezvous and docking with Spacecraft 9. (See fig. 3.7-1.) The purpose of the ATDA was to:

(a) Provide a target vehicle for Spacecraft 9.

(b) Provide a docking vehicle for the spacecraft and provide a rigid connection between the two vehicles.

(c) Unrigidize, disconnect, and allow the spacecraft to disengage from the ATDA.

(d) Provide multiple docking capability.

3.7.1 Structure

The ATDA consisted of an ascent shroud, a Target Docking Adapter (TDA), a cylindrical equipment section, a reaction control section, and a battery module. The overall ATDA structure was 241.25 inches long and 60 inches in diameter. After separation of the ascent shroud, the orbited vehicle was 146.75 inches long. (See fig. 3.7-1.)

3.7.1.1 <u>Ascent shroud</u>.- The ATDA ascent shroud was very similar to the shroud used on the Gemini Agena Target Vehicle (GATV). It consisted of two phenolic fiberglass half-shells clamped together by an externally mounted restraining strap. The nose of the shroud was latched together by two latches which were held in place by a spring and cable system. Vents provided for the GATV configuration were closed for use with the ATDA. The shroud was clamped to the forward end of the TDA section and the shroud separation sequence was initiated two seconds before ATDA separation from the Target Launch Vehicle (TLV).

Separation was initiated by the receipt of one of the two redundant sequence-timer signals. (See fig. 3.7-2.) This signal caused the cartridges in each of four explosive bolts to fire, releasing the restraining strap which in turn released the shroud latches. Two spring-loaded separators were installed to provide the necessary force to insure positive separation. Pivot brackets restrained the base of the shroud to insure nose-first separation. (See fig. 3.7-3.)

3.7.1.2 <u>Target docking adapter</u>.- The TDA was bolted to the forward end of the equipment section and consisted of a thin-skin semimonocoque cylindrical section, docking cone, ATDA status-display panel, radar transponder, two L-band antenna systems, acquisition and approach lights, mooring drive system, latching mechanism, and spacecraft-ATDA hardline umbilical connections.

3.7.1.3 Equipment section.- The equipment section, which was approximately 54 inches long and 60 inches in diameter, housed all electronics equipment (fig. 3.7-4). This section consisted of a ring-stiffened cylindrical shell which was semimonocoque at the forward end and monocoque at the aft end. Two internal crossbeams provided support for the electrical equipment; four access doors were included to allow for installation, checkout, and removal of the equipment. The aft end of the section provided support for the reaction control section and was also attached to the separation assembly of the TLV adapter.

Six running lights – two red, two amber, and two green – were installed on the equipment section (fig. 3.7-1). Each light was covered with a dome-shaped quartz-glass protective cover.

The ATDA was separated from the TLV by cutting the structure with redundant strands of a flexible linear shaped charge (FLSC). Eight preloaded bungee cords provided a separation velocity of approximately 3 ft/sec. The design of the FLSC separation assembly was the same as that utilized for separating the equipment and retrograde sections of the Gemini spacecraft adapter assembly. The only change between the two assemblies was that the ATDA charge was approximately two-thirds as long as that used on the spacecraft and no tube cutter was required for the ATDA separation.

3.7.1.4 Reaction control system section.- The reaction control section was of titanium sheet metal construction. It consisted of a thin cylinder, with nine external webs supporting the longitudinal stiffeners, that was enclosed by machined titanium rings at each end. This section was attached to the equipment section by a method similar to the attachment of the Reentry Control System to the conical section of the Gemini spacecraft.

3.7.1.5 Battery module.- The battery module was of aluminum sheet metal, crossbeam construction with individual compartments for each battery and support for the stub antenna. The module was covered by a cylindrical can to provide protection for the batteries and to provide support for the separation system and separation guide rails. The battery module was attached to the reaction control section by utilizing the existing Gemini rendezvous and recovery section attach points and main parachute fitting attach points.

3.7.2 Major Systems

3.7.2.1 Communications system.- The communications system provided the necessary information, acquisition, and command links between the spacecraft and the ATDA and between the ground stations and the ATDA. The system consisted of a tracking subsystem, a telemetry transmission subsystem, and a command subsystem.

3.7.2.1.1 Tracking subsystem: The tracking subsystem was installed to enable the ground tracking stations to track the ATDA. Complete redundancy was accomplished by the use of two C-band radar transponders mounted in the equipment section. Each transponder was controlled by a separate Digital Command System (DCS) channel and was used in conjunction with its own C-band helical antenna.

Each C-band radar transponder consisted of a receiver, a decoder section, a transmitter, a power supply, a common RF input and output connector, and an RF circulator. Each C-band helical antenna set comprised three helical antennas and one power divider. The transponders were activated by ground command via the DCS.

3.7.2.1.2 Telemetry transmission subsystem: The telemetry (TM) transmission subsystem functioned as the link which carried real-time information, generated in the ATDA, to the ground stations. Two TM transmitters operated in conjunction with two diplexers, three coaxial switches, and three antennas. The TM subsystem was used during all phases of the mission when the ATDA was in contact with the ground stations and was controlled by DCS command. The three antennas that supplied the radiation coverage were the UHF whip antenna, the ascent antenna, and the UHF stub antenna.

The UHF whip antenna provided omnidirectional yaw-plane coverage and was utilized in conjunction with the UHF stub antenna for telemetry and digital command. It was also used in conjunction with the ascent antenna to receive ground commands until ATDA separation.

The UHF stub antenna provided omnidirectional roll coverage after ATDA separation. The ascent antenna, mounted under the shroud, was used to receive ground commands until ATDA separation.

3.7.2.1.3 Command subsystem: The DCS provided a real-time command link for ATDA utilization of ground commands. The receiver-decoder unit and three relay units, mounted in the equipment section, were operational from prelaunch throughout the mission. The DCS received and decoded command transmissions, and, after decoding by the decoder, the commands operated various relays which controlled the ATDA (see fig. 3.7-5).

Complete control over the ATDA included initiation of TLV/ATDA separation, equipment redundancy selection, lighting control, control rate selection, telemetry antenna selection, and dock and undock control. Selective control over the ATDA was provided by the air-to-air spacecraft/ATDA link, which included an undock, zero deg/sec rate selection, and an acquisition-lights-off command.

3.7.2.2 Instrumentation.- The instrumentation system monitored specific ATDA systems, conditions, or events by sensing, conditioning, and encoding the data for transmission to the ground via the telemetry transmission subsystem.

The system consisted of two Gemini dc-to-dc converters, a standard Gemini PCM programmer, a Gemini signal conditioner package, and instrumentation sensors.

3.7.2.3 Guidance and Control System.- The Guidance and Control System consisted of the Target Stabilization System and the rendezvous radar transponder.

3.7.2.3.1 Target Stabilization System: The ATDA Target Stabilization System (TSS), in conjunction with the reaction control system, provided three-axis rate stabilization of the vehicle. The TSS was designed to control the vehicle to fixed turning rates about two axes and provide rate damping about the third axis or to provide rate damping about all three axes.

The TSS consisted of an attitude control electronics (ACE) package, an orbital attitude and maneuver electronics (OAME) package, two rate gyro packages, and two power inverters, all standard Gemini components. (See fig. 3.7-6.) The ACE accepted input signals from the rate gyros and a system control panel. The signals from the system control panel simulated hand-controller rate-command signals, which, in conjunction with the rate gyro signals, established the target-vehicle two-axis turning rates. The following portions of the ACE were utilized:

(a) Mode logic – to set the appropriate deadband

(b) Power supply – to convert 26 V ac, 400 cps power into firing commands to the OAME.

The OAME, consisting of attitude solenoid valve drivers and spike-suppression circuits, converted the signals received from the ACE into firing commands for the reaction control system.

The rate gyros sensed body angular rates about the pitch, roll, and yaw axes of the spacecraft. Each rate gyro package contained three gyros mounted orthogonally. In addition to inputs to the ACE, input to the telemetry system was also provided by the rate gyros.

The biased rate damping mode was to establish fixed turning rates about the pitch and roll axes of the target vehicle and rate-damping (±0.25 deg/sec) in the yaw axis. The fixed turning rates would have (1) reduced the spacecraft radar error (target angle measurement) caused by ellipticity of the target-vehicle antenna pattern, and (2) provided more uniform exposure of the surface of the ATDA to the sun's radiation. This mode was selected automatically after TLV/ATDA separation; however, a DCS backup was provided in case the automatic selection malfunctioned.

The normal rate damping mode was to have rate-damped the ATDA about all three axes, and was to have been used during the docking phase of the mission. The rates would have been damped to 0.0 ±0.25 deg/sec in pitch and yaw, and to 0.0 ±0.5 deg/sec in roll. During the docking maneuver, the TSS could be left in this mode or turned off after the body rates had been damped. This mode could be selected by DCS or spacecraft L-band command.

Upon command from the DCS, a complete secondary (redundant) set of TSS equipment could be selected. The secondary TSS was comprised of a rate gyro package, a power inverter, the ACE power supply, ACE and OAME electronic sections, and an OAME jet-valve driver. The preamplifiers and logic circuits of the ACE were not redundant and, therefore, were shared by both primary and secondary systems. Both primary and secondary systems would provide the same control mode characteristics.

Either of the reaction control system rings could also be selected by DCS command; however, interlocks were provided to preclude simultaneous operation of both A-ring and B-ring, or primary and secondary TSS. After docking, the TSS could be shut off via the spacecraft/TDA umbilical connection in addition to the DCS control. However, the TSS could be turned on only by DCS command.

3.7.2.3.2 Rendezvous radar transponder system: The rendezvous radar transponder system received and amplified signals from the spacecraft and provided a return signal to the spacecraft. The transponder was also capable of receiving a spacecraft-generated command and supplying the ATDA with a control signal which would command the TDA cone to unrigidize, acquisition lights to turn off, and rate control to be in either mode. These discrete commands were to be initiated by the crew and could be transmitted via the radar RF link when separated or the hardline connection when docked. The rendezvous radar transponder system consisted of a transponder package, a boost regulator, and an antenna system.

The transponder package included receiver and transmitter groups and a sub-bit detector which was a component part of the command link. (The output signal of the transponder was connected to a logic package which increased the amplitude necessary for switching and decreased the switching time of the antennas.) The unit operated cooperatively with the spacecraft radar throughout a range of 50 feet to 180 nautical miles, provided the angle between the spacecraft Z-axis and the line-of-sight to the ATDA did not exceed 0 ±8.5 degrees.

The antenna system provided left-hand circular polarization over a sphere of coverage compatible with the spacecraft radar. The antenna system consisted of two spiral antennas, fed in parallel, and a multi-element dipole. The dipole boom was extended 8 seconds after TLV/ATDA separation.

3.7.2.4 Electrical system.- The function of the electrical system was to provide all of the ATDA dc power during the airborne phase of the mission. The ATDA electrical system provided unregulated dc main, control, and squib bus power (see fig. 3.7-7). The electrical power source was located in the battery section, and the power distribution busses were located in the equipment section. All ac power required by the various systems was generated within the using system by self-contained inverters.

Three silver-zinc batteries, with a rating of 400 ampere-hours each, were installed as the primary source to support a 5-day mission. Two additional silver-zinc batteries, with a rating of 15 ampere-hours each, were installed as the power source for firing the pyrotechnics, and for the common control bus. Redundancy in the electrical system was accomplished by providing two isolated squib busses for the ATDA pyrotechnics.

No circuit protection was provided in the ATDA except in the pyrotechnic and latch-actuator heater circuits. This concept is identical to that used in the GATV power system.

3.7.2.5 Reaction control system.- The reaction control system was identical to the Gemini spacecraft reentry control system. The function of this system was to provide, in conjunction with the TSS, rate control about the three axes during orbital flight. Two complete and independent systems were installed to provide 100 percent system redundancy. Each system had eight thrust chamber assemblies (TCA's), rated at 23.5 pounds nominal thrust. Firing of the TCA's provided roll, pitch, and yaw capability.

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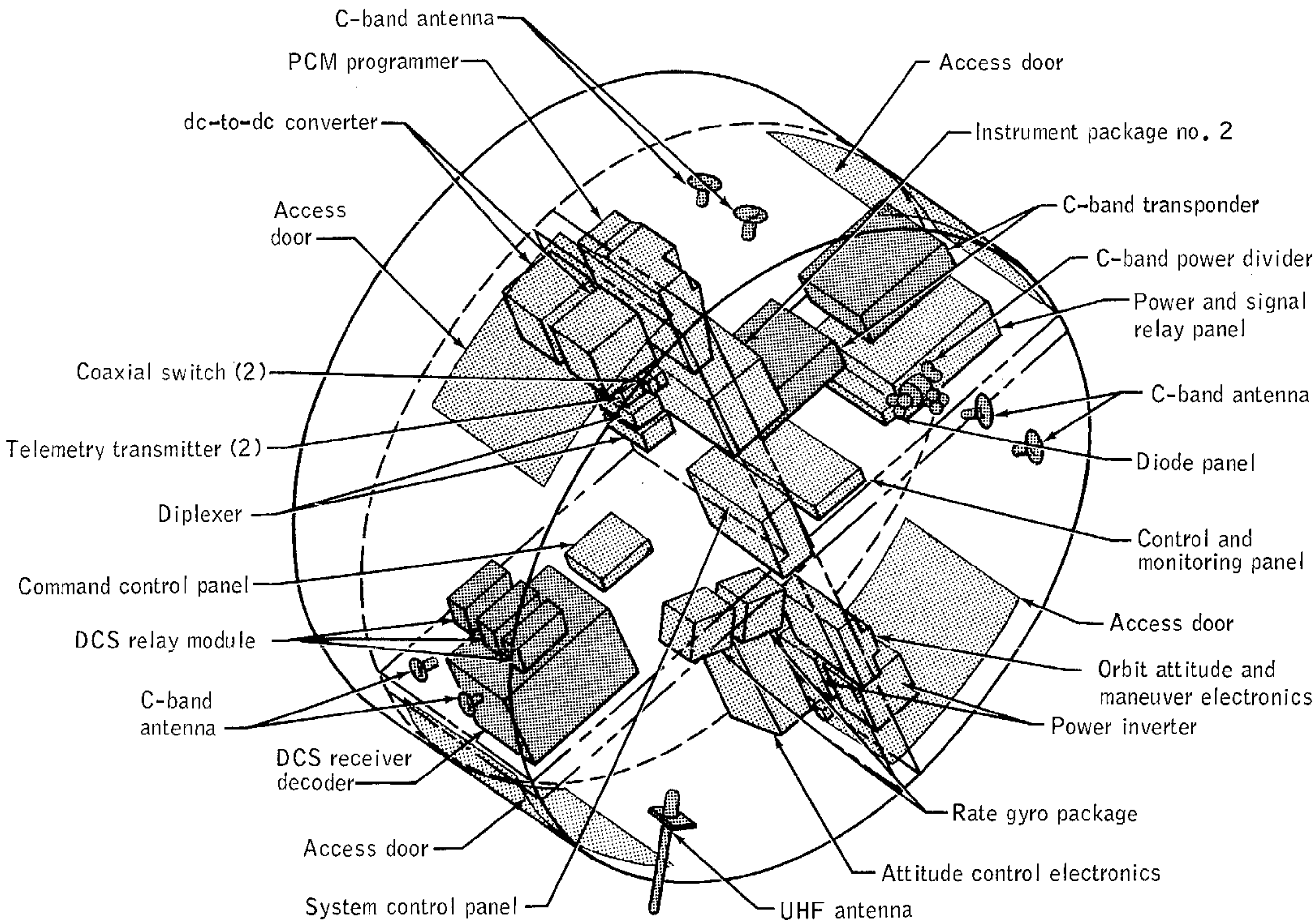

Figure 3.7-4. - ATDA equipment section.

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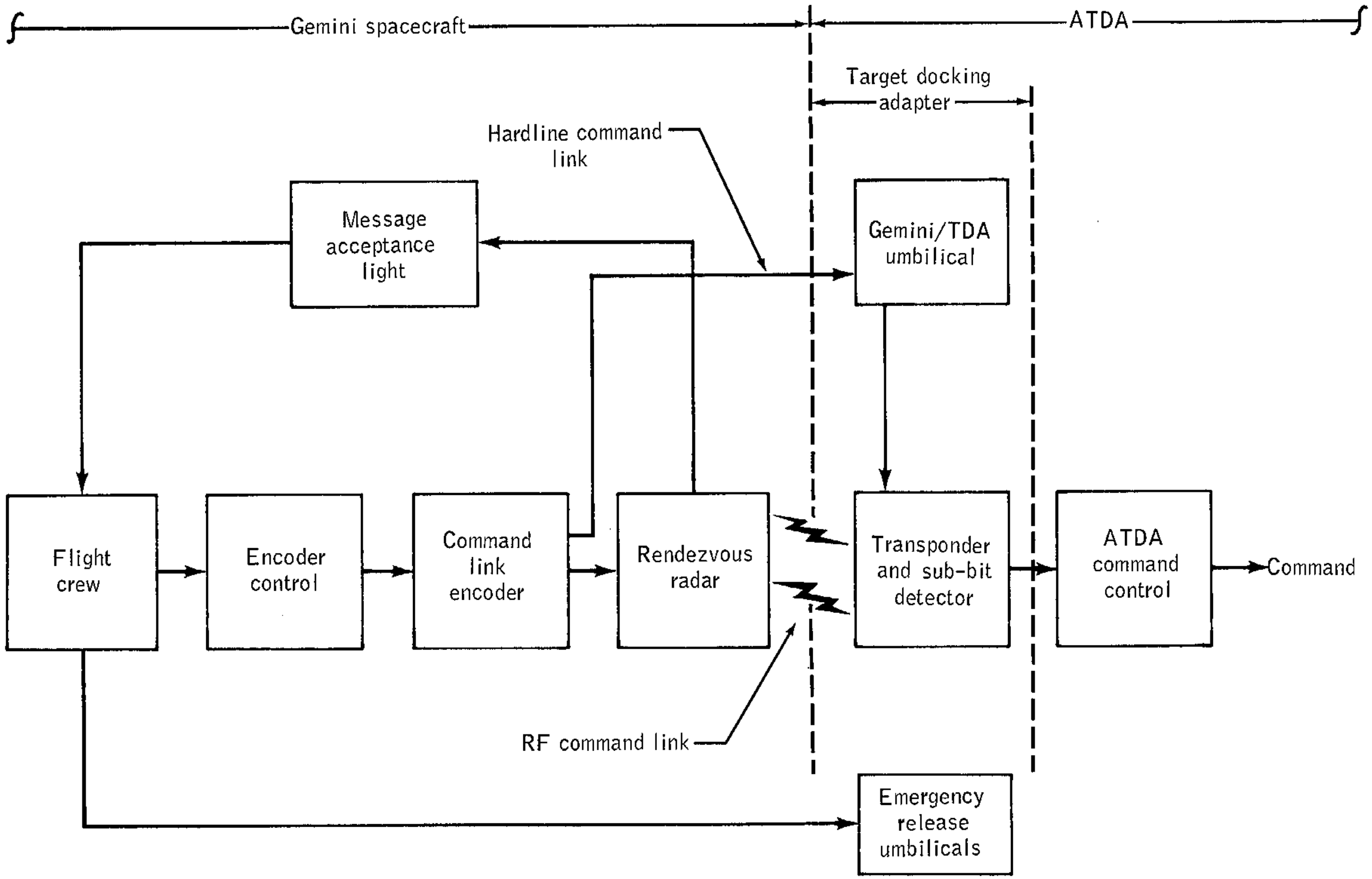

Figure 3.7-5. - Command link block diagram.

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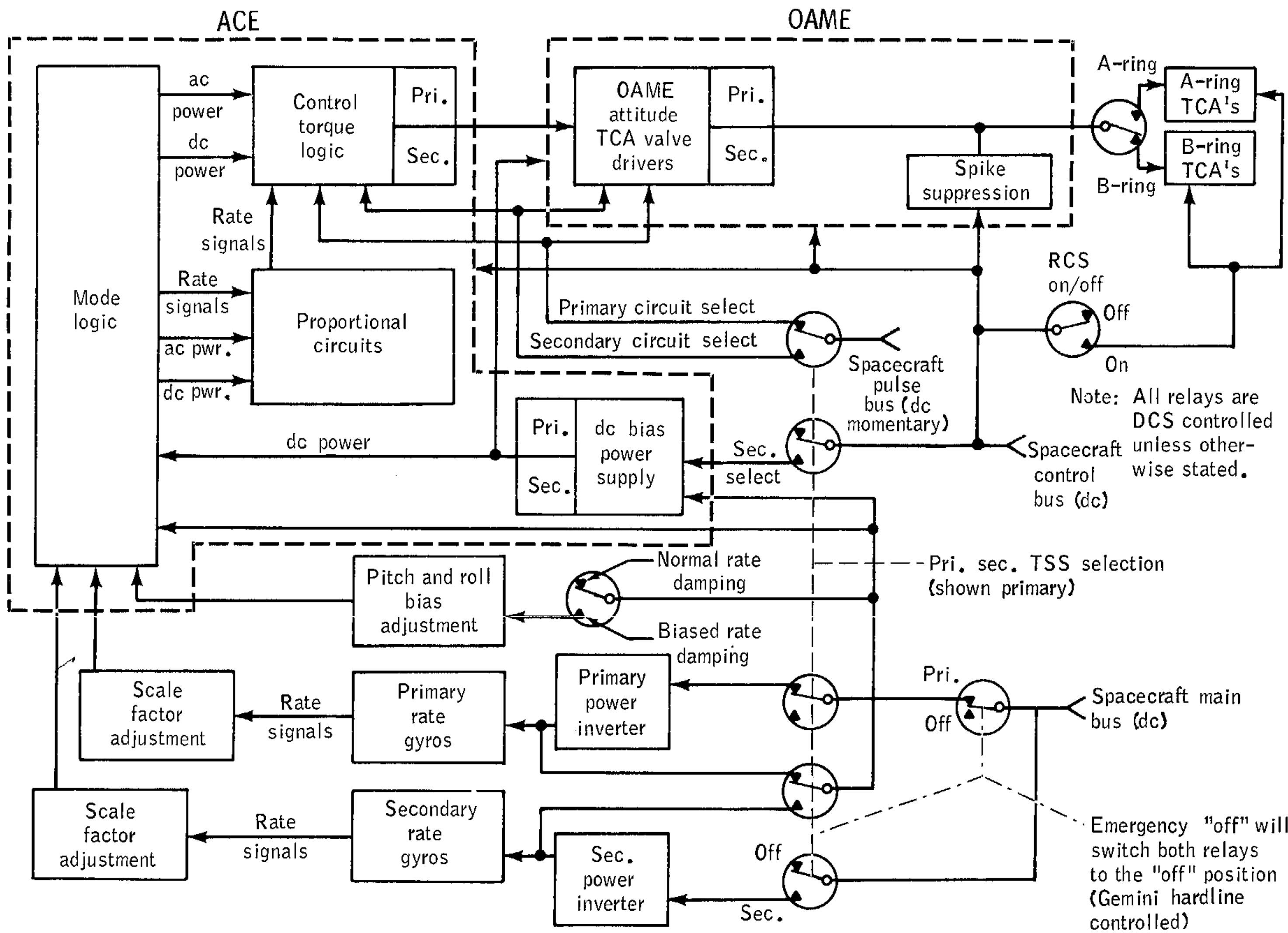

Figure 3.7-6. - TSS functional block diagram.

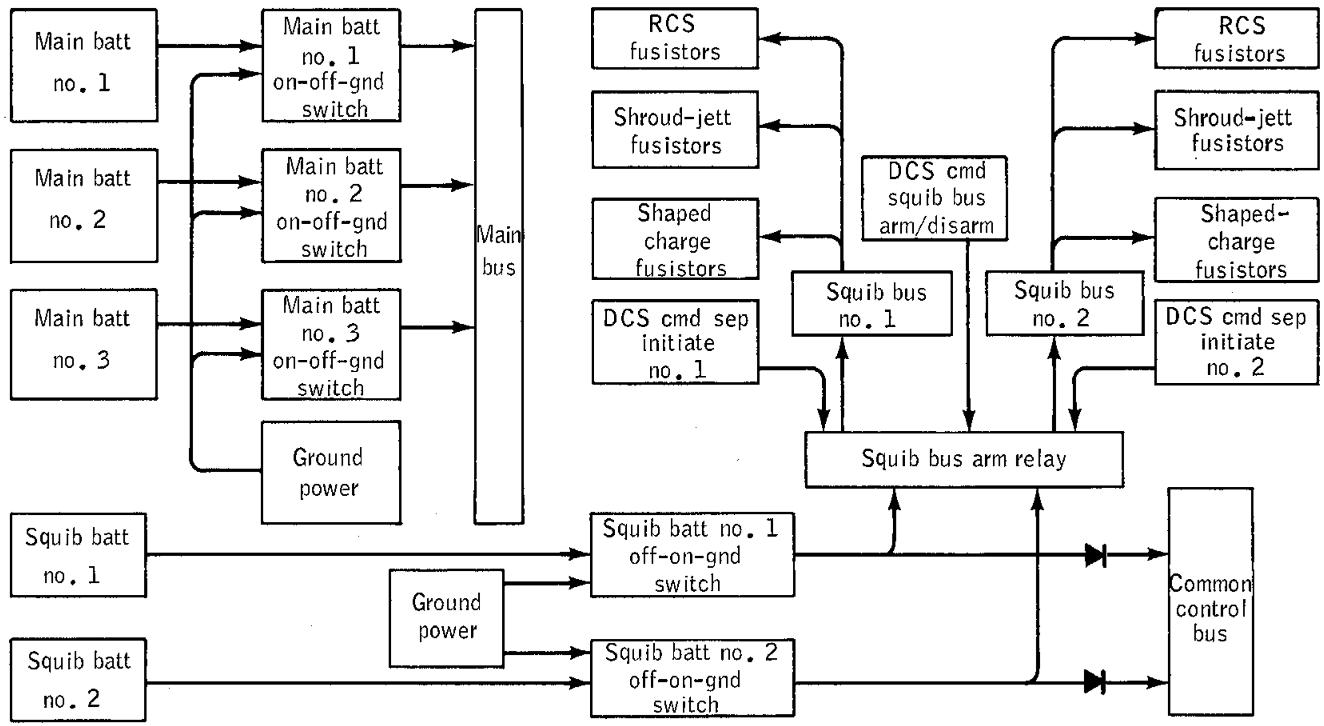

Figure 3.7-7. - ATDA power and power distribution.

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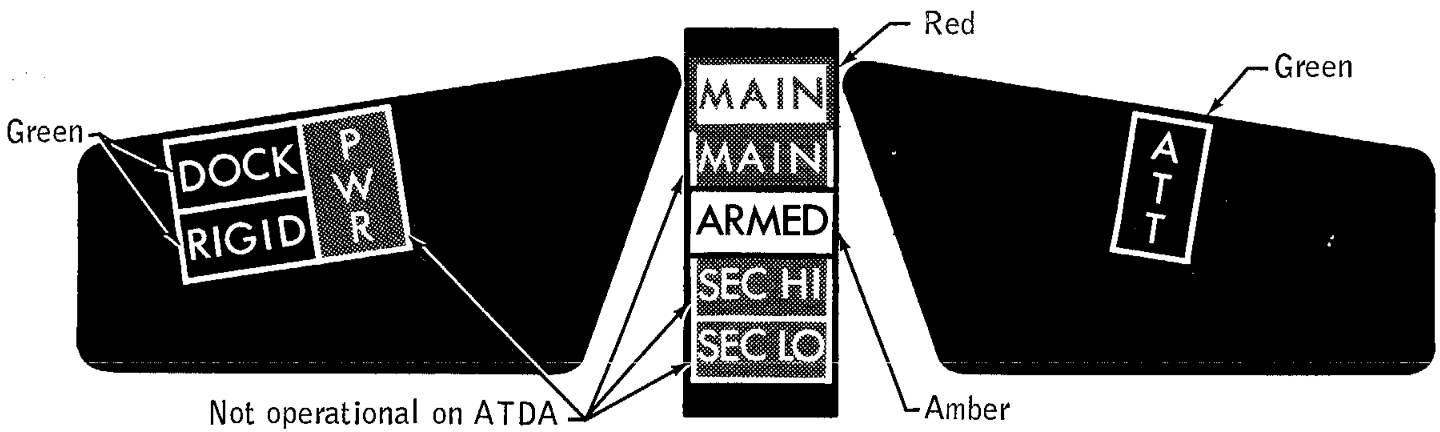

Figure 3.7-8. - Status display panel.

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4.0 MISSION DESCRIPTION

4.1 ACTUAL MISSION

The Gemini IX-A mission was initiated with the lift-off of the Target Launch Vehicle/Augmented Target Docking Adapter (TLV/ATDA) on June 1, 1966. The TLV sustainer engine operated for a longer duration than on any previous Atlas flight, and the ATDA was inserted into a near-circular orbit with an apogee of 161.5 nautical miles and a perigee of 158.5 nautical miles, referenced to the Fischer ellipsoid earth model of 1960. Telemetry data indicated that the protective shroud had not separated from the ATDA.

Problems in the T-3 minute launch-azimuth update to the spacecraft Inertial Guidance System prevented the launch of the Gemini space vehicle on June 1, 1966. The Gemini Space Vehicle was launched on June 3, 1966. Although the T-3 update was again not received by the spacecraft, a backup procedure, developed during the two-day launch postponement, prevented a hold in the countdown.

At 4 hours 15 minutes ground elapsed time (g.e.t.), the crew reported that the M=3 rendezvous was complete and that they were station keeping with the ATDA at a range of 60 feet. The crew also reported that the shroud was still attached to the ATDA, and they were able to maneuver close enough to the ATDA to describe the shroud in detail. After two attempts to release the shroud by ground commands were unsuccessful, the alternate flight plan was placed in effect. (See figure 4.1-1.) The alternate flight plan had been developed immediately after the first telemetry data indicated that the shroud might not have jettisoned and docking would not be possible. This plan was formalized prior to lift-off of the Gemini Space Vehicle. Because the shroud was still attached, docking could not be accomplished and the spacecraft moved away from the ATDA at 5 hours 1 minute g.e.t. to get into position for the equi-period rendezvous. The first equi-period rendezvous maneuver was initiated at 5:45:20 g.e.t., and the rendezvous was successfully completed at 6 hours 36 minutes when the crew reported station keeping with the ATDA at a range of 60 feet. At 7:14:58 g.e.t., the spacecraft separated from the ATDA to get into position for the third rendezvous (rendezvous from above). This sequence of events was necessary to complete the three rendezvous operations and avoid station keeping during a night period. The rendezvous from above and ahead of the ATDA was to simulate conditions which would result if the Apollo Command Module were required to rendezvous from above with a disabled Lunar Module.

After eating and sleeping (the micrometeorite collector door was opened during the sleep period), the crew initiated the third rendezvous at 18:23:19 g.e.t., and the rendezvous was completed at 21 hours 42 minutes g.e.t. Additional attempts to release the ATDA shroud by ground command were not successful. The crew inspected the ATDA from close range and determined that an attempt to remove the shroud by an extravehicular pilot would not be practical.

Also, the crew reported that they were fatigued and requested that extravehicular activities (EVA) be postponed until the next day. Ground control concurred with this request, and several experiments were scheduled to be conducted in place of the EVA on the second day of the flight. A decision was made to not attempt further operations with the ATDA, and at 22 hours 59 minutes g.e.t. the spacecraft moved away from the ATDA.

After a three-hour rest period, the crew conducted several experiment sequences during revolutions 17 through 21. The micrometeorite collector door was opened at the start of the next sleep period and closed at the end of the sleep period.

Preparations for EVA were started at 45 hours 30 minutes g.e.t. on revolution 29 and were completed approximately 30 minutes ahead of schedule. The cabin was depressurized incrementally to zero, and the pilot opened the hatch at 49 hours 23 minutes. The extravehicular activities were proceeding according to plan until the pilot reached the adapter and started preparations for donning the Astronaut Maneuvering Unit (AMU). The pilot experienced difficulty keeping his body properly positioned in the work area and, because of this, was working much harder than anticipated. After the pilot had been outside the spacecraft for one hour and five minutes, he reported that moisture had started to fog his pressure visor. When efforts to clear the fogging visor were not successful, the pilot was unable to continue with the AMU evaluation, and further extravehicular activities were cancelled. After two hours and five minutes of EVA, the pilot was back in the spacecraft.

After a sleep period, the crew powered up the spacecraft and prepared for retrofire. Retrofire was initiated at 71:46:44 g.e.t. during revolution 45, and the reentry was nominal. The spacecraft landed at 72:20:50 g.e.t. within one mile of the target landing point and in visual contact of the prime recovery ship, U.S.S. Wasp. Pararescue men attached and inflated the flotation collar four minutes after landing, and the crew and the spacecraft were hoisted onboard the aircraft carrier 53 minutes after landing.

The ATDA remained in orbit with an estimated lifetime of 53 days. The ATDA was monitored by remote sites until ATDA revolution 63 when all sites were released except the Rose Knot Victor tracking ship. The Rose Knot Victor monitored the ATDA from ATDA revolution 63 until the ATDA was powered down to a minumum power configuration on revolution 78.

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4.2 SEQUENCE OF EVENTS

The times at which major events were planned and executed are presented in tables 4.2-I and 4.2-II for the Gemini Space Vehicle spacecraft. Table 4.2-III shows the sequence of events for the Target Launch Vehicle/Augmented Target Docking Adapter.

TABLE 4.2-I.- SEQUENCE OF EVENTS FOR GEMINI SPACE VEHICLE LAUNCH PHASE

Event	Time from lift-off, sec		Difference, sec
	Planned	Actual	
Stage I engine ignition signal (87FS1)	-3.40	-3.14	+0.26
Stage I MDTCPS makes, subassembly 1	-2.30	-2.32	-0.02
Stage I MDTCPS makes, subassembly 2	-2.30	-2.30	0.00
Shutdown lockout (backup)	-0.10	-0.09	+0.01
Lift-off (pad disconnect separation)	13:39:33.335 G.m.t.		
Roll program start (launch azimuth = 87.4 deg)	18.48	18.42	-0.06
Roll program end	20.48	20.42	-0.06
Pitch program rate no. 1 start	23.04	22.97	-0.07
Pitch program rate no. 1 end, rate no. 2 start	88.32	88.06	-0.26
IGS update sent	105.00	Inhibited	
Control system gain change no. 1	104.96	104.64	-0.32
Pitch program rate no. 2 end, rate no. 3 start	119.04	118.66	-0.38
Stage I engine shutdown circuitry armed	114.64	144.20	-0.44
IGS update sent	145.00	Inhibited	
Stage I MDTCPS unmakes	151.57	152.44	+0.87
BECO (stage I engine shutdown) (87FS2)	151.65	152.49	+0.84
Staging switches actuate	151.65	152.48	+0.83
Signals from stage I rate gyro package to flight control system discontinued	151.65	152.48	+0.83
Hydraulic switchover lockout	151.65	152.48	+0.83
Telemetry ceases, stage I	151.65	152.48	+0.83
Staging nuts detonate	151.65	152.48	+0.83
Stage II engine ignition signal (91FS1)	151.65	152.48	+0.83
Control system gain change	151.65	152.48	+0.83
Stage separation begin	152.35	153.23	+0.88
Stage II engine MDFJPS make	152.55	153.20	+0.65
Pitch program rate no. 3 ends	162.56	161.44	-1.12
RGS guidance enable	162.56	161.43	-1.13
First guidance command signal received by TARS	169.00	168.19	-0.81
Stage II engine shutdown circuitry armed	317.44	315.79	-1.65
SECO (stage II engine shutdown) (91FS2)	338.99	339.79	+0.80
Redundant stage II shutdown	338.99	339.81	+0.82
Stage II MDFJPS break	339.29	339.94	+0.65
Spacecraft separation	369.29	366.72	-2.57
OAMS on	369.29	366.10	-3.19
OAMS off (final)	399.22	[a]436.40	37.18

[a]Over a 70-second time interval, 4 maneuvers were made which totaled 41.4 seconds thrust time.

TABLE 4.2-II.- SEQUENCE OF EVENTS FOR GEMINI SPACECRAFT

Event	Ground elapsed time, hr:min:sec		Difference, sec
	Planned[a]	Actual	
Orbital phase			
Rendezvous in third revolution			
Phase-adjust maneuver	00:49:03	00:49:03	0
Corrective-combination maneuver	01:55:17	01:55:17	0
Coelliptic maneuver	02:24:51	02:24:52	+1
Terminal-phase-initiate maneuver	03:35:35	03:36:02	27
First midcourse correction	NA	03:48:35	NA
Second midcourse correction	NA	04:00:27	NA
Braking maneuver	04:07:59	04:06:08	-111
Equi-period rendezvous			
Radial-separation maneuver	05:01:00	05:01:00	0
Horizontal-adjust maneuver	05:45:00	[b]05:45:20	+20
Terminal-phase-initiate maneuver	06:14:07	06:15:12	+65
First midcourse correction	NA	06:20:20	NA
Intermediate correction	NA	06:26:59	NA
Second midcourse correction	NA	06:28:34	NA
Braking maneuver	06:33:58	06:29:26	-272
Rendezvous from above			
Separation maneuver	07:14:58	[c]07:14:58	0
Phase-adjust maneuver	18:23:19	[c]18:23:19	0
Height-adjust maneuver	19:08:16	[c]19:08:16	0
Coelliptic maneuver	19:54:24	[c]19:54:24	0
Terminal-phase-initiate maneuver	[d]20:55:28	21:02:28	[d]+420
First midcourse correction	NA	[c]21:14:36	NA
Second midcourse correction	NA	[c]21:26:36	NA
Braking maneuver	21:27:57	n/a	n/a
Separation maneuver	22:59:00	22:59:00	0
True anomaly adjust maneuver	53:41:35	53:41:35	0

[a]The planned values given for the orbital and rendezvous maneuvers are the latest information forwarded to the crew prior to initiating the maneuver.

[b]Check point only — maneuver not required.

[c]Based on pilot report.

[d]The ground commanded time was determined to be in error by the crew.

NA — Not applicable.

n/a — Not available.

TABLE 4.2-II.- SEQUENCE OF EVENTS FOR GEMINI SPACECRAFT - Concluded

Event	Ground elapsed time, hr:min:sec		Difference, sec
	Planned[a]	Actual	
Reentry phase			
Adapter equipment section separation	71:45:44	71:45:50	6
Retrofire initiation	71:46:44	71:46:44	0
Begin blackout	72:08:56	72:08:32	-24
End blackout	72:13:44	72:13:54	+10
Drogue parachute deployment	72:15:33	72:15:48	+15
Pilot parachute deployment, main parachute initiation	72:17:09	72:17:11	+2
Two-point suspension	NA	72:18:23	NA
Landing	72:21:09	72:20:50	-19

[a]The planned values given for the orbital and rendezvous maneuvers are the latest information forwarded to the crew prior to initiating the maneuver.

NA — Not applicable.

TABLE 4.2-III.- SEQUENCE OF EVENTS FOR TLV/ATDA LAUNCH PHASE

Event	Planned time from lift-off, sec	Actual time from lift-off, sec	Difference, sec
Lift-off	15:00:02.363 G.m.t.		
Initiate roll program	2.00	2.05	+0.05
Terminate roll program	15.00	15.05	+0.05
Initiate pitch program	15.00	15.20	+0.20
Booster engine cutoff (BECO)	117.94	117.21	-0.73
Booster section jettison (BECO + 3.0 sec)	120.94	120.18	-0.76
Sustainer engine cutoff (SECO)	347.99	348.70	+0.71
Vernier engine cutoff (VECO)	366.08	367.54	+1.46
ATDA separation	368.08	383.41	[a]+15.33

[a]Separation signal was sent from the Mission Control Center - Houston through Digital Command System.

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4.3 FLIGHT TRAJECTORIES

The launch and orbital trajectories referred to as planned are either preflight-calculated nominal trajectories (refs. 11 through 14) 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 attitudes and sequences, as determined from airborne instrumentation. For all trajectories except the actual launch phase, 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 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. Ground tracks of the first five revolutions, the revolution during which the rendezvous from above was executed, and the final revolution from retrofire to landing are shown in figure 4.3-1.

4.3.1 Gemini Spacecraft

4.3.1.1 Launch.- The launch trajectory data shown in figure 4.3-2 are based on real-time output of the Range Safety Impact Prediction Computer (IP 3600), the Guided Missile Computer Facility (GMCF), and the Bermuda radar. 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; and Bermuda used data from an FPS-16 radar. Data from these tracking facilities were used during the following time periods:

Facility	Time from lift-off, sec
IP 3600 (FPS-16 and MISTRAM)	0 to 55
GMCF (GE MOD III)	55 to 386
Bermuda (FPS-16)	386 to 411

The actual launch trajectory, as compared with the planned launch trajectory in figure 4.3-2, was essentially nominal during stage I powered flight. At first-stage engine cutoff (BECO), however, the velocity was low by 22 ft/sec, and the altitude and flight path angle were high by 1635 feet and 0.07 degree, respectively. After BECO, the Radio Guidance System (RGS) had only very small errors to correct and guided Stage II to a near-nominal insertion of the spacecraft. At second-stage engine cutoff (SECO), the velocity and altitude were less than nominal by 15 ft/sec, and 980 feet, respectively. The flight-path angle was a negative 0.11 degree, rather than the nominal zero degree. At spacecraft separation the velocity, altitude, and flight-path angle were low by 16 ft/sec, 2299 feet, and 0.11 degree, respectively.

Table 4.3-I contains a comparison of planned and actual conditions at BECO, SECO, and spacecraft separation. The preliminary conditions were obtained from MISTRAM and Canary Island tracking data. The final conditions for BECO were obtained from the Inertial Guidance System (IGS); and final conditions for SECO and spacecraft separation were obtained by integrating the best estimated trajectory orbital fit back through the spacecraft separation maneuver and the tail-off impulse, both determined from telemetry records of IGS data. (NOTE: This best estimated trajectory contained complete first-revolution tracking data.)

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 conditions obtained from GE MOD III contained a velocity and a flight-path angle that were low by 13 ft/sec and high by 0.05 degree, respectively, when compared with the more accurate orbital ephemeris data obtained later. The conditions obtained by MISTRAM showed velocity and flight-path angle to be high by 5 ft/sec and low by 0.11 degree, respectively, when compared with the later ephemeris data.

4.3.1.2 Orbit.- Because the main objective of the Gemini IX-A mission was to rendezvous and dock with the Augmented Target Docking Adapter (ATDA), 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 spacecraft insertion to retrofire. The planned elements shown in these tables were obtained from orbital ephemerides generated using the sequences in references 11 and 12, and the actual elements were obtained by integrating the Gemini tracking network vectors after each midcourse and terminal-phase maneuver.

The planned trajectory and the actual trajectory for the first rendezvous (M=3 rendezvous) are presented in table 4.3-III and figure 4.3-4. The planned trajectory and planned maneuvers for the initial rendezvous in spacecraft revolution 3 were obtained from the real-time solution using the Eglin revolution 29 vector for the ATDA and the Bermuda revolution 1 vector for the spacecraft. The ground-commanded maneuvers were determined from various spacecraft and ATDA vectors as the planned maneuvers were updated after each maneuver. The actual trajectory during the initial rendezvous phase was reconstructed utilizing anchor vectors obtained from best estimated trajectory (see reference 15) and the actual maneuvers, as derived from the Inertial Guidance System (IGS) postflight analysis, applied as instantaneous changes in velocity until rendezvous. The spacecraft vector was determined prior to the first maneuver. The ATDA vector was from a best estimated trajectory listed in reference 15. 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 orbital insertion, indicated a nominal situation for obtaining a third-orbit rendezvous. Spacecraft lift-off was on time and the only anomalies indicated were an underspeed of about 5 ft/sec at insertion after the separation maneuver and a negative inertial flight-path angle of 0.08 degrees instead of the nominal 0.03 degree. The effective results of the anomalies were to cause the phase-adjust maneuver $\left(N_{C1}\right)$ to increase from the premission expectation of approximately 54 ft/sec to 75 ft/sec and to cause the corrective-combination maneuver $\left(N_{CC}\right)$ to increase slightly from a nominal zero ΔV to approximately 14 ft/sec.

4.3.1.2.1 First rendezvous: At spacecraft insertion, the range between the Spacecraft 9 and the ATDA was approximately 570 nautical miles and the out-of-plane velocity error resulting after the GLV ascent-yaw-steering was less than 10 ft/sec.

At 49 minutes 3 seconds ground elapsed time (g.e.t.), a phase-adjust maneuver $\left(N_{C1}\right)$ was initiated near first apogee. The horizontal, posigrade ΔV of 74.8 ft/sec was applied with the aft-firing thrusters. The resultant altitude at perigee was about 125 nautical miles. At 1:55:17 g.e.t., the corrective-combination maneuver was initiated and performed nominally. The actual ΔV of 14.8 ft/sec was applied with the aft-firing thrusters at a pitch-up attitude of 44.3 degrees and yaw-left of 67.5 degrees.

The coelliptic maneuver $\left(N_{SR}\right)$ was initiated at 2:24:51 g.e.t. and performed nominally with the aft-firing thrusters. The actual ΔV of 53.4 ft/sec was applied at a pitch-down attitude of 40.1 degrees and a yaw-left attitude of 2.9 degrees. The corrective-combination maneuver and the coelliptic maneuver varied slightly from the planned maneuvers because it was decided to change the planned time of the coelliptic maneuver to approximately 8 minutes earlier. At 2:24:51 g.e.t., favorable tracking conditions would exist over Carnarvon so that it would be possible to monitor and evaluate the coelliptic maneuver. It was decided that no constraints would be violated by changing the time of N_{SR} and that all previous conditions required for rendezvous would still be met. It was therefore decided to recompute N_{CC} and N_{SR} so that the N_{SR} maneuver could be performed over Carnarvon. The resultant spacecraft orbit was about 148 by 149 nautical miles and the differential altitude (Δh) between the spacecraft and the ATDA orbits was about 12 nautical miles. Prior to terminal phase initiate (TPI), the Δh varied from 12.0 to 12.2 nautical miles with a value of 12.1 nautical miles at TPI.

The TPI maneuver was initiated at 3:36:02 g.e.t. when the elevation angle to the ATDA was approximately 27.2 degrees and the range was about 25.2 nautical miles. A total ΔV of 28.4 ft/sec was applied with the aft-firing thrusters at an effective pitch-up angle of 31.9 degrees and a yaw-left angle of 2.7 degrees.

The intermediate corrections, at ωt = 81.8 degrees and 33.6 degrees (the angles of orbit travel to rendezvous), were applied 12 and 24 minutes later and required about 6 ft/sec ΔV and 5 ft/sec ΔV, respectively.

The terminal phase finalize (TPF) maneuver was initiated at 4:06:08 g.e.t. and braking thrusts were applied intermittently over the next 9 minutes. An effective resultant velocity of about 33 ft/sec was added to the spacecraft orbit; however, because of the semi-optical approach technique, a ΔV of approximately 46 ft/sec was used. By 4:15:00 g.e.t., the spacecraft was less than 100 feet from the ATDA and the crew was station keeping.

The total translation cost of the terminal phase maneuver was approximately 100 pounds of propellant, which was close to the preflight nominal. The total translation cost of the rendezvous maneuvers, including terminal phase, was 240 pounds, which was also close to the preflight nominal.

4.3.1.2.2 Second rendezvous (equi-period rendezvous): The planned trajectory and the actual trajectory for the second rendezvous (equi-period rendezvous) are presented in table 4.3-III and figure 4.3-4. The planned trajectory and planned maneuvers for the equi-period rendezvous were obtained from the real-time solution using the Ascension revolution 31 ATDA vector for both vehicles. (That is, the spacecraft and ATDA were assumed to be in the same orbit prior to the radial-separation maneuver.)

This equi-period rendezvous was performed completely by the crew using only information displayed from the platform, optical aids, and onboard charts, except for the time of the 20 ft/sec radial-separation maneuver which was ground updated. The 20 ft/sec radial-separation maneuver was initiated on time at 5:01:00 g.e.t. The actual trajectory was obtained by using the actual maneuvers, as derived from the IGS postflight analysis, applied as instantaneous changes in velocity until rendezvous.

Approximately 45 minutes after the radial-separation maneuver (5:45:20 g.e.t.), the crew computed the required horizontal corrective maneuver, which was zero. Thus, from all indications, the 20 ft/sec radial-separation maneuver was performed exactly as planned.

At approximately 6:15:12 g.e.t., the crew initiated the terminal phase transfer to the ATDA. The crew performed corrective maneuvers at 6:20:20, 6:26:59, and 6:28:34 g.e.t. and then initiated the braking maneuvers at 6:29:26 g.e.t. The crew reported that station keeping began at 6 hours 36 minutes g.e.t. Except for the application of one correction that was computed using a handheld sextant, the equi-period rendezvous was as close to nominal as could be expected.

The translation cost of propellants for the terminal phase of the second rendezvous, including line-of-sight corrections, totaled about the equivalent of 45 pounds of propellant. This propellant cost was within 10 pounds of the prelaunch nominal.

4.3.1.2.3 Third rendezvous (rendezvous from above): The planned and actual trajectories for the third rendezvous (rendezvous from above) are presented in table 4.3-III and figure 4.3-4. The third rendezvous exercise was initiated at 7:14:58 g.e.t. with a 3.7 ft/sec horizontal retrograde maneuver. The planned trajectory and planned maneuvers were obtained from a real-time solution using the Ascension revolution 38 ATDA vector for both vehicles.

The ground-commanded maneuvers were determined from various Spacecraft 9 and ATDA vectors as the planned maneuvers were updated after each spacecraft maneuver. The actual trajectory during the third rendezvous phase was reconstructed using anchor vectors from the best estimated trajectory (see reference 11) and maneuvers, as derived from pilot report and IGS postflight analysis, applied as instantaneous changes in the velocity until rendezvous. The spacecraft and ATDA were assumed to be in the same orbit prior to the 3.7 ft/sec separation.

A small quantity of real-time telemetry data were obtained, and the IGS radar data compared very closely with the actual reconstructed trajectory.

The only IGS postflight telemetry data received were for the TPI maneuver. The remaining portion of the mission was reconstructed using the pilot's report and information derived from engineering estimates of the postflight data.

Following a five-revolution coast period after the 3.7 ft/sec separation maneuver, ground computations indicated a nominal situation for effecting the rendezvous from above. The only anomaly indicated was a variation of the Δh after the coelliptic maneuver. The coellipticity variation after the coelliptic maneuver is accredited to the errors in tracking between the spacecraft maneuvers. Tracking data from the Carnarvon station indicated a TPI time of 20:55:28 g.e.t.; however, this time was updated by the crew to 21:02:28 g.e.t., and later tracking data confirmed the crew update. At 18:23:19 g.e.t., a phase-adjust maneuver was performed. The scheduled height-adjust maneuver was performed at 19:08:16 g.e.t.

The coelliptic maneuver was performed at 19:54:24 g.e.t. This posigrade maneuver was applied with the forward-firing thrusters and placed the spacecraft in a 166 by 167 nautical mile orbit. Following the N_{SR} maneuver, the differential altitude (Δh) varied from 7.5 nautical miles to 8.3 nautical miles and back down to 6.7 nautical miles at the time of terminal phase initiation. This anomaly is accredited to the fact that a single-station solution from Ascension revolution 38 tracking data was used for the ATDA for the final portion of the third rendezvous. Also, the Antigua revolution 13 spacecraft vector used for the N_{SR} update or the TPI solution could have been slightly in error. This, along with the fact that the desired coellipticity was not achieved, is the reason that TPI slipped from 20:55:28 to 21:02:28 g.e.t.

The TPI maneuver was begun at 21:02:28 g.e.t. with an elevation angle to the ATDA of approximately -27.5 degrees at a range of 14.5 nautical miles. A total ΔV of 18 ft/sec was applied with the aft-firing thrusters, using about 19 pounds of propellant, at an effective pitch down of 19.4 degrees and a yaw left of 179.7 degrees. The propellant usage for TPI was approximately 1 pound over nominal.

The intermediate corrections, at ωt = 81.8 degrees and 33.6 degrees, were applied 12 and 24 minutes later and required about 8 and 19 ft/sec ΔV, respectively. These maneuvers used approximately 22 pounds more propellant than the 2.8-pound nominal.

The time of the TPF maneuver is not exactly known because of loss of data. An effective velocity of about 20 ft/sec was added to the spacecraft orbit; however, because of the poor visibility of the ATDA against an earth background, the semi-optical approach technique required approximately 85 pounds of propellant. Because of the poor visibility for the line-of-sight control and braking, the 85 pounds of propellant used represented about 25 pounds more than the preflight nominal of 60 pounds. The total propellant used from initiation of separation maneuver to station keeping was about 174 pounds which was about 54 pounds over the nominal 120 pounds required for the planned rendezvous from above. By 21:42:00 g.e.t., the spacecraft was in position with the ATDA and the crew was station keeping.

4.3.1.3 Reentry.- The planned and actual reentry trajectories are shown in figure 4.3-5. The planned trajectory was determined by integrating the Canary Island vector in revolution 45 through planned retrofire sequences determined by the RTCC, and simulating a 55-degree contour-line bank-angle reentry according to Math Flow 7 described in reference 16. The actual trajectory was obtained by integrating the Canary Island vector to retrofire, then integrating the White Sands vector after retrofire to the Grand Bahama Island vector, through Math Flow 7 guidance. The Grand Bahama Island vector was then integrated to landing through the actual parachute-deployment sequences. Actual reentry angles during the communications blackout were not available because of a failure in the onboard tape recorder.

The reconstructed reentry trajectory agrees very well in relation to the actual reentry events. Communication blackout times agree within 10 seconds of actual blackout, maximum acceleration loads compare with telemetry within 0.1g at analogous time, and parachute-deployment altitudes at recorded sequence times are in accord with those reported in section 5.1.11. Table 4.3-II contains a comparison of reentry dynamic parameters and landing points. The final landing point was within one nautical mile of the planned landing point. (See section 5.1.5 for a more detailed description of the landing.)

4.3.2 Target Launch Vehicle/Augmented Target Docking Adapter

4.3.2.1 Launch.- The launch trajectory data for the Target Launch Vehicle/Augmented Target Docking Adapter (TLV/ATDA) as 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 (FPS-16, TPQ-18, FPQ-6)	0 to 125
IP 3600 (FPQ-6, TPQ-18)	125 to 353
IP 3600 (TPQ-18)	353 to 387
IP 3600; BDA (TPQ-18, FPS-16)	387 to 409
BDA (FPS-16)	409 to RTCC orbit phase

The actual launch trajectory, as compared with the planned trajectory in figure 4.3-6, was essentially nominal during the TLV/ATDA launch phase. At BECO, the velocity, altitude, and flight-path angle were high by 11 ft/sec, 746 feet, and 0.29 degree, respectively. After BECO, the Radio Guidance System had very little error to correct and guided the sustainer to a nominal insertion of the ATDA. At SECO the velocity was low by 9 ft/sec, and altitude and flight-path angle were high by 140 feet and 0.02 degree, respectively. At vernier engine cutoff (VECO), the velocity was measured as exactly nominal and the altitude and flight-path angle were high by 79 feet and low by 0.01 degree, respectively.

4.3.2.2 Orbit.- The ATDA was placed into the desired orbit for the planned Gemini Space Vehicle launch and rendezvous. Table 4.3-V contains a comparison of the planned and actual insertion conditions of the ATDA. The preliminary conditions were obtained by integrating the Carnarvon vector during the first revolution back to ATDA separation, and the final conditions were obtained by integrating the first revolution best estimated trajectory vector back to ATDA separation.

The difference in velocity between VECO and ATDA separation was 4 ft/sec. This can be attributed to the following:

(a) The bungee cord that imparts a separation ΔV to the ATDA

(b) The popgun effect of the shaped charge firing

(c) Uncertainty in the inertial velocity as measured at VECO and ATDA separation.

The bungee cord was designed to impart a separation ΔV of 2.7 ± 0.8 ft/sec to the ATDA. Because the shroud did not jettison from the ATDA, the additional weight reduced the expected separation ΔV to 2.3 ± 0.7 ft/sec.

The popgun effect on the ATDA is expected to impart a ΔV of approximately 1 to 2 ft/sec. The popgun phenomenon due to shaped charge firing has been observed repeatedly on the Gemini spacecraft at separation. The ΔV gained by the Gemini spacecraft has been approximately 1 ft/sec as measured by the onboard platform. The same type of shaped charge is used to separate both vehicles; therefore, from energy and momentum considerations, the ΔV expected on the ATDA is approximately 1 ft/sec, assuming that the popgun effect behavior is similar for both vehicles. This is not strictly true, however, as the ATDA is inserted a few feet into the adapter shroud on the TLV sustainer. This would allow the gas pressure from the shaped charge to act on the ATDA for a longer period of time than would be possible on the spacecraft, and additional velocity could result. The three-sigma uncertainty in the velocity measurement is approximately ± 1 ft/sec.

Table 4.3-VI contains a comparison of ATDA orbital parameters for every 12 revolutions until retrofire of the Gemini spacecraft. The ATDA was calculated to have a total orbital lifetime of 53 days.

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 127.8 and 85.6 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 19-hour orbital lifetime. The Goddard Space Flight Center predicted reentry in revolution 13 within a reentry corridor over the Indian Ocean.

4.3.4 Gemini Target Launch Vehicle

The sustainer of the Target Launch Vehicle was inserted into an orbit with apogee and perigee altitudes of 160.5 and 157.5 nautical miles, respectively. The Gemini network tracking radars and the NORAD network tracking sensors were able to skin-track the TLV during the mission, and NORAD estimated a 20-day lifetime. Estimated apogee and perigee altitudes at Gemini IX-A retrofire were 160.2 and 145.0 nautical miles, respectively.

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	151.65	152.48	152.48
Geodetic latitude, deg north	28.54	28.54	28.54
Longitude, deg west	79.65	79.64	79.64
Altitude, feet	208 328	209 950	209 963
Altitude, n. mi.	34.3	34.5	34.5
Range, n. mi.	47.7	47.9	47.9
Space-fixed velocity, ft/sec	9 765	9 765	9 766
Space-fixed flight-path angle, deg	19.92	19.94	19.97
Space-fixed heading angle, deg east of north	88.42	88.37	88.38
SECO			
Time from lift-off, sec	339.00	339.79	339.79
Geodetic latitude, deg north	28.90	28.89	28.89
Longitude, deg west	71.81	71.80	71.80
Altitude, feet	529 055	528 570	528 075
Altitude, n. mi.	87.1	87.0	86.9
Range, n. mi.	461.6	462.0	462.0
Space-fixed velocity, ft/sec	25 641	25 628	25 626
Space-fixed flight-path angle, deg	0.0	-0.14	-0.11
Space-fixed heading angle, deg east of north	86.84	86.85	86.84

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

Condition	Planned	Actual	
		Preliminary	Final
Spacecraft separation			
Time from lift-off, sec	369.00	366.80	366.80
Geodetic latitude, deg north	28.99	28.97	28.98
Longitude, deg west	69.59	69.80	69.80
Altitude, feet	529 105	526 821	526 806
Altitude, n. mi.	87.1	86.7	86.7
Range, n. mi.	578.7	567.2	567.3
Space-fixed velocity, ft/sec	25 723	25 706	25 707
Space-fixed flight-path angle, deg	0.00	-0.11	-0.11
Space-fixed heading angle, deg east of north	87.96	87.86	87.86
Maximum conditions			
Altitude, statute miles	194.0	193.4	193.4
Altitude, n. mi.	168.7	168.2	168.2
Space-fixed velocity, ft/sec	25 745	25 739	25 740
Earth-fixed velocity, ft/sec	24 378	24 371	24 372
Exit acceleration, g	7.3	7.2	7.2
Exit dynamic pressure, lb/ft^2	750	748	748
Reentry deceleration, g (tracking data)	5.0	5.5	5.5
Reentry deceleration, g (telemetry data)		5.4	5.4
Reentry dynamic pressure, lb/ft^2	337	372	372
Landing point			
Latitude, north	27°52'	27°51'	27°51'
Longitude, west	75°00'	74°56'	74°56'

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

Maneuver	Condition	Before maneuver			After maneuver		
		Planned (a)	Actual		Planned (a)	Actual	
			Preliminary (b)	Final (a)		Preliminary (b)	Final (a)
M=3 Phase adjust (N_{C1})	Apogee, n. mi.	148.6	144.0	144.0	148.6	146.9	146.0
	Perigee, n. mi.	86.7	85.7	85.7	117.9	125.0	123.6
	Inclination, deg	28.88	28.90	28.91	28.88	28.93	28.92
	Period, min	88.82	--	88.78	89.40	--	89.60
Corrective combination (N_{CC})	Apogee, n. mi.	148.6	146.9	146.0	148.5	149.7	148.7
	Perigee, n. mi.	117.9	125.0	123.6	118.0	124.9	124.9
	Inclination, deg	28.88	28.93	28.92	28.88	28.91	28.91
	Period, min	89.40	--	89.60	89.40	--	89.67
Coelliptic (N_{SR})	Apogee, n. mi.	148.5	149.7	148.7	148.8	149.2	148.0
	Perigee, n. mi.	118.0	124.9	124.9	146.4	148.5	145.2
	Inclination, deg	28.88	28.91	28.91	28.88	28.90	28.91
	Period, min	89.40	--	89.67	89.98	--	90.07
Terminal phase initiate (TPI)	Apogee, n. mi.	148.8	149.2	148.0	164.9	160.8	160.0
	Perigee, n. mi.	146.4	148.5	145.2	147.2	146.8	147.1
	Inclination, deg	28.88	28.90	28.91	28.88	28.91	28.91
	Period, min	89.98	--	90.07	90.23	--	90.32
Terminal phase finalize (TPF)	Apogee, n. mi.	164.9	160.8	160.0	161.4	161.0	161.0
	Perigee, n. mi.	147.2	146.8	147.1	158.8	160.0	157.6
	Inclination, deg	28.88	28.91	28.91	28.88	28.88	28.91
	Period, min	90.23	--	90.32	90.49	--	90.54

[a]The altitude is computed above the Fischer ellipsoid earth model of 1960; period and inclination are osculating elements.

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

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

Maneuver	Condition	Before maneuver			After maneuver		
		Planned (a)	Actual		Planned (a)	Actual	
			Preliminary (b)	Final (a)		Preliminary (b)	Final (a)
Radial separation	Apogee, n. mi.	161.2	161.0	161.0	161.9	163.1	163.3
	Perigee, n. mi.	158.5	160.0	157.6	157.2	157.1	156.6
	Inclination, deg	28.88	28.88	28.91	28.88	28.91	28.89
	Period, min	90.47	--	90.54	90.47	--	90.49
Equi-period rendezvous	Apogee, n. mi.	161.9	163.1	163.3	161.2	161.0	161.0
	Perigee, n. mi.	157.2	157.1	156.6	158.5	160.0	157.6
	Inclination, deg	28.88	28.91	28.89	28.88	28.88	28.91
	Period, min	90.47	--	90.49	90.47	--	90.54
Retrograde separation	Apogee, n. mi.	161.2	161.0	161.0	160.5	160.4	159.8
	Perigee, n. mi.	158.5	160.0	157.6	157.8	158.4	156.2
	Inclination, deg	28.88	28.88	28.91	28.88	28.89	28.89
	Period, min	90.47	--	90.54	90.43	--	90.50
Rendezvous from above Phase adjust (N_{C1})	Apogee, n. mi.	160.2	160.0	159.1	160.2	161.0	159.5
	Perigee, n. mi.	157.6	158.1	155.7	157.6	158.1	156.5
	Inclination, deg	28.88	28.88	28.91	28.88	28.88	28.90
	Period, min	90.42	--	90.48	90.42	--	90.49
Height adjust (N_H)	Apogee, n. mi.	160.2	161.0	159.5	167.9	167.2	166.7
	Perigee, n. mi.	157.6	158.1	156.5	158.3	159.8	158.2
	Inclination, deg	28.88	28.88	28.90	28.88	28.88	28.90
	Period, min	90.42	--	90.49	90.57	--	90.67

[a]The altitude is computed above the Fischer ellipsoid earth model of 1960; period and inclination are osculating elements.

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

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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 (a)	Actual		Planned (a)	Actual	
			Preliminary (b)	Final (a)		Preliminary (b)	Final (a)
Rendezvous from above (continued) Coelliptic (N_{SR})	Apogee, n. mi.	167.9	167.2	166.7	168.7	168.2	168.2
	Perigee, n. mi.	158.3	159.8	158.2	167.0	168.0	166.4
	Inclination, deg	28.88	28.88	28.90	28.88	28.88	28.91
	Period, min	90.57	--	90.67	90.73	--	90.81
Terminal phase initiate (TPI)	Apogee, n. mi.	168.7	168.2	168.2	168.0	166.9	166.4
	Perigee, n. mi.	167.0	168.0	166.4	158.5	160.1	158.6
	Inclination, deg	28.88	28.88	28.91	28.88	28.88	28.91
	Period, min	90.73	--	90.81	90.57	--	90.64
Terminal phase finalize (TPF)	Apogee, n. mi.	168.0	166.9	166.4	161.0	161.0	160.3
	Perigee, n. mi.	158.5	160.1	158.6	158.3	159.0	156.8
	Inclination, deg	28.88	28.88	28.91	28.88	28.88	28.91
	Period, min	90.57	--	90.64	90.45	--	90.51
Separation	Apogee, n. mi.	Not computed	161.0	160.3	Not computed	161.0	160.0
	Perigee, n. mi.		159.0	156.8		157.9	155.4
	Inclination, deg		28.88	28.91		28.89	28.91
	Period, min		--	90.51		--	90.58
True anomaly	Apogee, n. mi.	Not computed	161.2	159.9	Not computed	157.5	156.5
	Perigee, n. mi.		157.3	154.3		145.8	144.0
	Inclination, deg		28.91	28.91		28.90	28.90
	Period, min		--	90.47		--	90.19

[a]The altitude is computed above the Fischer ellipsoid earth model of 1960; period and inclination are osculating elements.

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

TABLE 4.3-III.- SPACECRAFT RENDEZVOUS MANEUVERS

Maneuver	Planned[a]	Ground commanded	Actual
M=3 rendezvous			
Phase adjust (N_{C1})			
G.e.t., hr:min:sec	00:49:07	00:49:03	00:49:03
ΔV, ft/sec	74.2	75.0	74.8
Pitch, deg	0.0	0.0	-0.2
Yaw, deg	0.0	0.0	-0.1
Δt_B, sec	99.0	100.0	94.6
Corrective combination (N_{CC})			
G.e.t., hr:min:sec	01:55:20	01:55:17	01:55:17
ΔV, ft/sec	10.9	14.6	14.8
Pitch, deg	2.6	44.1	44.4
Yaw, deg	-60.0	-66.9	-67.5
Δt_B, sec	15.0	19.0	20.3
Coelliptic (N_{SR})			
G.e.t., hr:min:sec	02:32:59	02:24:51	02:24:51
ΔV, ft/sec	42.5	54.0	53.4
Pitch, deg	1.5	-40.7	-40.1
Yaw, deg	-0.7	-2.8	-2.9
Δt_B, sec	57.0	71.0	70.0
Terminal phase initiate (TPI)			
G.e.t., hr:min:sec	03:37:11	03:35:35	03:36:02
ΔV, ft/sec	25.2	[b]26.8	28.4
Pitch, deg	30.8	30.1	27.6
Yaw, deg	2.6	4.8	0.7
Δt_B, sec	33.0	35.0	37.0

[a]The planned values given for the orbital and rendezvous maneuvers are the latest information forwarded to the crew prior to initiating the maneuver.

[b]Maneuvers sent referenced to line-of-sight were 26.7 forward, 2.2 right and 1.3 up.

TABLE 4.3-III.- SPACECRAFT RENDEZVOUS MANEUVERS - Continued

Maneuver	Planned[a]	Ground commanded	Actual
M=3 rendezvous - Concluded			
82-degree correction (COR_1)			
G.e.t., hr:min:sec	Not computed	Not sent	03:48:35
ΔV, ft/sec	Not computed	Not sent	[c]6.1
Pitch, deg			[d]31.9
Yaw, deg			[d]-1.0
Δt_B, sec			8.0
33-degree correction (COR_2)			
G.e.t., hr:min:sec	Not computed	Not sent	04:00:27
ΔV, ft/sec	Not computed	Not sent	[c]5.0
Pitch, deg			[d]89.6
Yaw, deg			[d]-2.0
Δt_B, sec			7.0
Terminal phase finalize (TPF)			
G.e.t., hr:min:sec	04:09:39	04:07:59	04:06:08
ΔV, ft/sec	32.8	Not sent	[e]33.0
Pitch, deg	52.4		[f]111.3
Yaw, deg	-174.2		[f]-4.0
Δt_B, sec	56.0		[g]550.0

[a]The planned values given for the orbital and rendezvous maneuvers are the latest information forwarded to the crew prior to initiating the maneuver.

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

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

[e]This is the resultant ΔV applied during the braking; however, the total ΔV expended during semi-optical approach was about 46 ft/sec.

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

[g]Braking lasted intermittently for about 9 minutes.

TABLE 4.3-III.- SPACECRAFT RENDEZVOUS MANEUVERS - Continued

Maneuver	Planned[a]	Ground commanded	Actual
Equi-period rendezvous			
Radial separation			
G.e.t., hr:min:sec	05:00:59	05:01:00	05:01:00
ΔV, ft/sec	20.0	20.0	20.0
Pitch, deg	-90.0	-90.0	-85.0
Yaw, deg	0.0	0.0	0.9
Δt_B, sec	36.0	35.0	35.0
Horizontal crossing correction			
G.e.t., hr:min:sec	05:45:00	Not sent	05:45:20
ΔV, ft/sec	0.0	Not sent	0.0
Pitch, deg	0.0		0.0
Yaw, deg	0.0		0.0
Δt_B, sec	0.0		0.0
Terminal phase initiate (optical)			
G.e.t., hr:min:sec	06:15:43	06:14:07	06:15:12
ΔV, ft/sec	3.5	Not sent	[c]2.1
Pitch, deg	55.8		[c]40.7
Yaw, deg	0.0		[e]-1.2
Δt_B, sec	3.0		4.0

[a]The planned values given for the orbital and rendezvous maneuvers are the latest information forwarded to the crew prior to initiating the maneuver.

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

[e]This is the resultant ΔV applied during the braking; however, the total ΔV expended during semi-optical approach was about 46 ft/sec.

TABLE 4.3-III.- SPACECRAFT RENDEZVOUS MANEUVERS - Continued

Maneuver	Planned[a]	Ground commanded	Actual
Equi-period rendezvous - Concluded			
Optical correction (COR_1)			
G.e.t., hr:min:sec	Not computed	Not sent	06:20:20
ΔV, ft/sec	Not computed	Not sent	[c]1.1
Pitch, deg			[d]51.8
Yaw, deg			[d]-1.2
Δt_B, sec			3.0
Optical intermediate correction (C_{INT})			
G.e.t., hr:min:sec	Not computed	Not sent	06:26:59
ΔV, ft/sec	Not computed	Not sent	[c]1.0
Pitch, deg			[d]67.6
Yaw, deg			[d]-1.5
Δt_B, sec			2.0
Optical correction (COR_2)			
G.e.t., hr:min:sec	Not computed	Not sent	06:28:34
ΔV, ft/sec	Not computed	Not sent	[c]3.0
Pitch, deg			[d]67.2
Yaw, deg			[d]-1.0
Δt_B, sec			4.0
Terminal phase finalize (TPF)			
G.e.t., hr:min:sec	Not computed	Not sent	06:29:26
ΔV, ft/sec	Not computed	Not sent	[c]35.0
Pitch, deg			[d]60.0
Yaw, deg			[d]0.2
Δt_B, sec			[h]420.0

[a]The planned values given for the orbital and rendezvous maneuvers are the latest information forwarded to the crew prior to initiating the maneuver.

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

[d]Approximate line-of-sight angles to target during correction.

[h]Braking lasted intermittently for about 7 minutes.

TABLE 4.3-III.- SPACECRAFT RENDEZVOUS MANEUVERS - Continued

Maneuver	Planned[a]	Ground commanded	Actual
Rendezvous from above			
Separation			
G.e.t., hr:min:sec	07:14:58	07:14:58	[i]07:14:58
ΔV, ft/sec	3.7	3.7	3.7
Pitch, deg	0.0	0.0	0.0
Yaw, deg	0.0	0.0	0.0
Δt_B, sec	7.0	7.0	7.0
Phase adjust $\left(N_{C1}\right)$			
G.e.t., hr:min:sec	18:23:23	18:23:19	[i]18:23:19
ΔV, ft/sec	1.8	2.0	2.0
Pitch, deg	0.0	0.0	0.0
Yaw, deg	0.0	0.0	0.0
Δt_B, sec	2.4	3.0	3.0
Height adjust $\left(N_{H1}\right)$			
G.e.t., hr:min:sec	19:08:21	19:08:16	[i]19:08:16
ΔV, ft/sec	16.4	17.0	17.0
Pitch, deg	0.0	0.0	0.0
Yaw, deg	0.0	0.0	0.0
Δt_B, sec		22.0	22.0
Coelliptical $\left(N_{SR}\right)$			
G.e.t., hr:min:sec	19:53:38	19:54:24	[i]19:54:24
ΔV, ft/sec	15.8	14.4	14.4
Pitch, deg	-14.3	-38.0	-38.0
Yaw, deg	180.0	180.0	180.0
Δt_B, sec		25.0	25.0

[a]The planned values given for the orbital and rendezvous maneuvers are the latest information forwarded to the crew prior to initiating the maneuver.

[i]Based on pilot log and other best estimate data.

TABLE 4.3-III.- SPACECRAFT RENDEZVOUS MANEUVERS - Concluded

Maneuver	Planned[a]	Ground commanded	Actual
Rendezvous from above - Concluded			
Terminal phase initiate (TPI)			
G.e.t., hr:min:sec	20:57:44	20:55:28	21:02:28
ΔV, ft/sec	15.9	[j]16.7	18.0
Pitch, deg	-25.7	-27.4	-19.5
Yaw, deg	-180.0	-178.3	-180.0
Δt_B, sec	20.9	21.0	22.0
82-degree correction (COR_1)			
G.e.t., hr:min:sec	Not computed	Not sent	21:14:36
ΔV, ft/sec	Not computed	Not sent	No data available
Pitch, deg			
Yaw, deg			
Δt_B, sec			
33-degree correction (COR_2)			
G.e.t., hr:min:sec	Not computed	Not sent	21:26:36
ΔV, ft/sec	Not computed	Not sent	No data available
Pitch, deg			No data available
Yaw, deg			No data available
Δt_B, sec			No data available
Terminal phase finalize (TPF)			
G.e.t., hr:min:sec	21:30:26	21:27:57	No data available
ΔV, ft/sec	21.7	Not sent	No data available
Pitch, deg	-56.5		No data available
Yaw, deg	0.1		No data available
Δt_B, sec	21.7		No data available

[a]The planned values given for the orbital and rendezvous maneuvers are the latest information forwarded to the crew prior to initiating the maneuver.

[j]Maneuvers sent referenced to line-of-sight were 16.5 forward, 2.5 right, and 0.3 up.

TABLE 4.3-IV.- COMPARISON OF SPACECRAFT ORBITAL ELEMENTS

Revolution	Condition	Planned (a)	Actual	
			Preliminary (b)	Final (a)
1 (Insertion)	Apogee, n. mi.	148.6	144.0	144.0
	Perigee, n. mi.	86.7	85.7	85.7
	Inclination, deg	28.88	28.90	28.91
	Period, min	88.82	--	88.78
3 Before M=3 rendezvous	Apogee, n. mi.	148.8	149.2	148.0
	Perigee, n. mi.	146.4	148.5	145.2
	Inclination, deg	28.88	28.90	28.91
	Period, min	89.98	--	90.07
4 Before equi-period rendezvous	Apogee, n. mi.	161.9	163.1	163.3
	Perigee, n. mi.	157.2	157.1	156.6
	Inclination, deg	28.88	28.91	28.89
	Period, min	90.47	--	90.49
12 Before rendezvous from above	Apogee, n. mi.	168.7	168.2	168.2
	Perigee, n. mi.	167.0	168.0	166.4
	Inclination, deg	28.88	28.88	28.91
	Period, min	90.73	--	90.81
15 After rendezvous from above	Apogee, n. mi.	161.0	161.0	160.3
	Perigee, n. mi.	158.3	159.0	156.8
	Inclination, deg	28.88	28.88	28.91
	Period, min	90.45	--	90.51
45 Retrofire	Apogee, n. mi.	Not computed	157.0	155.5
	Perigee, n. mi.		145.9	143.8
	Inclination, deg		28.91	28.91
	Period, min		--	90.19

[a]The altitude is computed above the Fischer ellipsoid earth model of 1960; period and inclination are osculating elements.

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

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

Condition	Planned	Actual	
		Preliminary	Final
BECO			
Time from lift-off, sec	117.94	Not available	117.21
Geodetic latitude, deg north	28.54		28.54
Longitude, deg west	80.04		80.04
Altitude, feet	213 896		214 642
Altitude, n. mi.	35.2		35.3
Range, n. mi.	26.9		27.8
Space-fixed velocity, ft/sec	8 418		8 429
Space-fixed flight-path angle, deg	36.61		36.90
Space-fixed heading angle, deg east of north	85.08		85.13
SECO			
Time from lift-off, sec	347.99	Not available	348.70
Geodetic latitude, deg north	29.00		29.00
Longitude, deg west	72.06		72.00
Altitude, feet	979 325		979 465
Altitude, n. mi.	161.1		161.2
Range, n. mi.	448.6		452.2
Space-fixed velocity, ft/sec	25 298		25 290
Space-fixed flight-path angle, deg	0.04		0.06
Space-fixed heading angle, deg east of north	88.51		88.55
VECO			
Time from lift-off, sec	366.08	Not available	367.54
Geodetic latitude, deg north	29.03		29.02
Longitude, deg west	70.77		70.68
Altitude, feet	979 523		979 602
Altitude, n. mi.	161.1		161.2
Range, n. mi.	516.6		520.6
Space-fixed velocity, ft/sec	25 365		25 365
Space-fixed flight-path angle, deg	0.0		-0.01
Space-fixed heading angle, deg east of north	89.17		89.23

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

Condition	Planned	Actual	
		Preliminary	Final
ATDA insertion			
Time from lift-off, sec	376.08	383.41	383.41
Geodetic latitude, deg north	29.03	29.03	29.03
Longitude, deg west	70.05	69.55	69.55
Altitude, feet	979 543	978 807	979 526
Altitude, n. mi.	161.2	161.1	161.2
Range, n. mi.	554.3	580.4	580.5
Space-fixed velocity, ft/sec	25 368	25 370	25 369
Space-fixed flight-path angle, deg	0.0	-0.01	-0.01
Space-fixed heading angle, deg east of north	89.54	89.80	89.81
Maximum conditions			
Altitude, statute miles	185.6	185.4	185.7
Altitude, n. mi.	161.4	161.2	161.5
Space-fixed velocity, ft/sec	25 374	25 370	25 370
Earth-fixed velocity, ft/sec	23 989	23 972	23 972
Exit acceleration, g	8.3	8.2	8.2
Exit dynamic pressure, lb/ft^2	955	904	904

TABLE 4.3-VI.- COMPARISON OF PLANNED AND ACTUAL OSCULATING ELEMENTS AT ATDA INSERTION

Condition	Planned	Actual	Difference
Semimajor axis, n. mi.	3604.8	3605.2	+0.4
Eccentricity	0.0007	0.0008	+0.0001
Inclination, deg	28.88	28.87	-0.01
Inertial ascent node, deg	68.09	68.07	-0.02
Apogee altitude, n. mi.	166.3	167.1	+0.8
Perigee altitude, n. mi.	161.2	161.0	-0.2
Period, min	90.49	90.50	+0.01
True anomaly, deg	1.85	-14.2	-16.05
Argument of perigee, deg	87.31	103.89	+16.58
Latitude of perigee, deg N	29.00	28.11	-0.89
Longitude of perigee, deg W	94.73	54.29	-40.44
Latitude of apogee, deg S	29.00	28.11	-0.89
Longitude of apogee, deg E	73.93	114.37	+40.44

TABLE 4.3-VII.- COMPARISON OF ATDA ORBITAL ELEMENTS

Revolution	Condition	Planned (a)	Actual	
			Preliminary (b)	Final (a)
Insertion (1)	Apogee, n. mi.	161.4	162.1	161.5
	Perigee, n. mi.	158.8	160.8	158.5
	Inclination, deg	28.88	28.89	28.91
	Period, min	90.49	--	90.58
12	Apogee, n. mi.	161.4	162.1	161.5
	Perigee, n. mi.	158.6	160.4	158.2
	Inclination, deg	28.88	28.88	28.91
	Period, min	90.48	--	90.56
24	Apogee, n. mi.	161.1	161.8	161.1
	Perigee, n. mi.	158.4	160.4	158.2
	Inclination, deg	28.88	28.88	28.91
	Period, min	90.47	--	90.55
36	Apogee, n. mi.	160.9	161.2	160.5
	Perigee, n. mi.	158.2	160.0	158.1
	Inclination, deg	28.88	28.88	28.91
	Period, min	90.46		90.54
48	Apogee, n. mi.	160.4	161.0	160.4
	Perigee, n. mi.	158.0	159.1	157.0
	Inclination, deg	28.88	28.91	28.91
	Period, min	90.45	--	90.53
60	Apogee, n. mi.	Not calculated	160.8	159.0
	Perigee, n. mi.		158.6	156.5
	Inclination, deg		28.90	28.91
	Period, min		--	90.51
Gemini IX-A retrofire 71	Apogee, n. mi.	Not calculated	160.5	158.9
	Perigee, n. mi.		157.7	155.2
	Inclination, deg		28.89	28.91
	Period, min		--	90.48

[a]The altitude is computed above the Fischer ellipsoid earth model of 1960; period and inclination are osculating elements.

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

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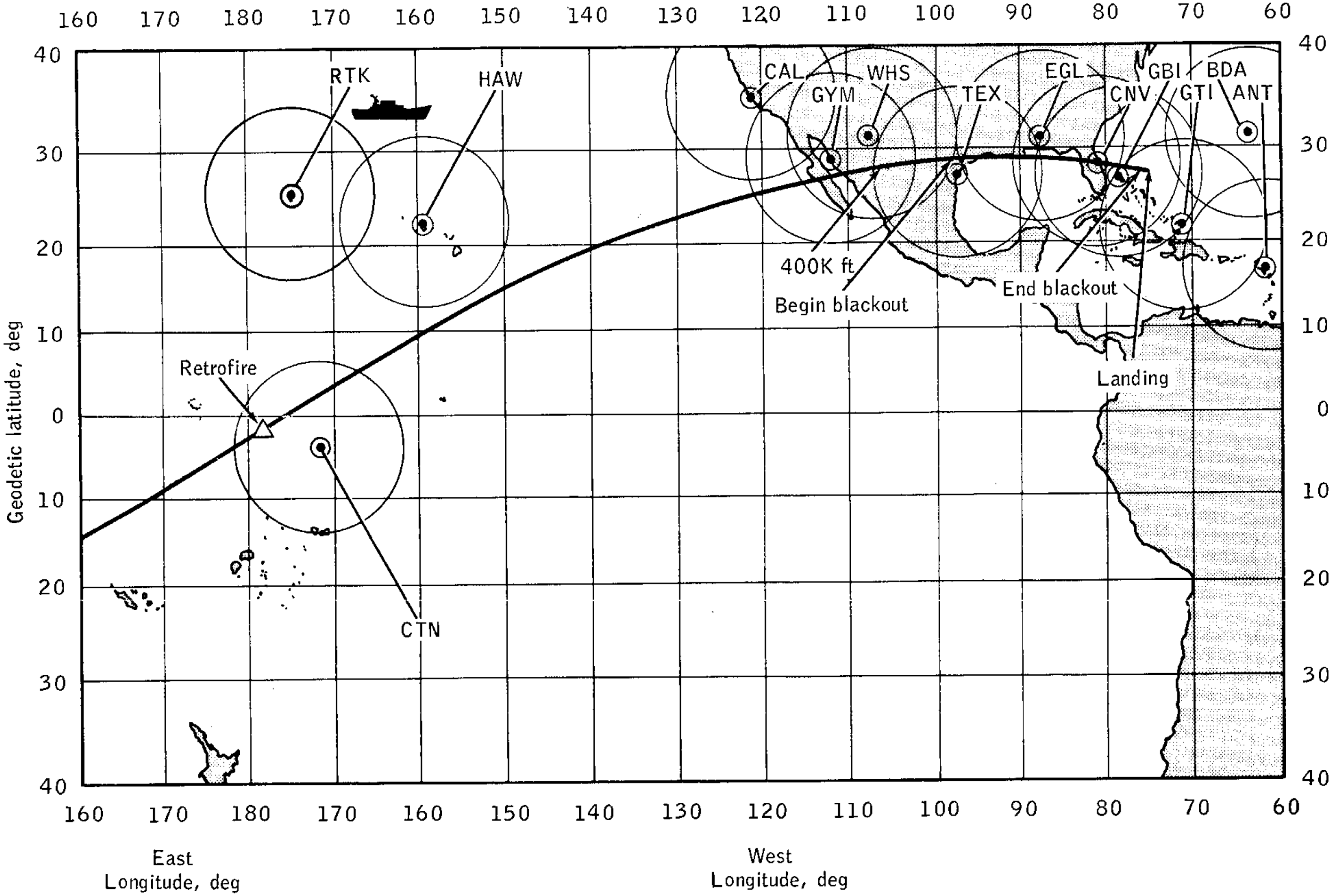

(c) Reentry.

Figure 4.3-1.- Concluded.

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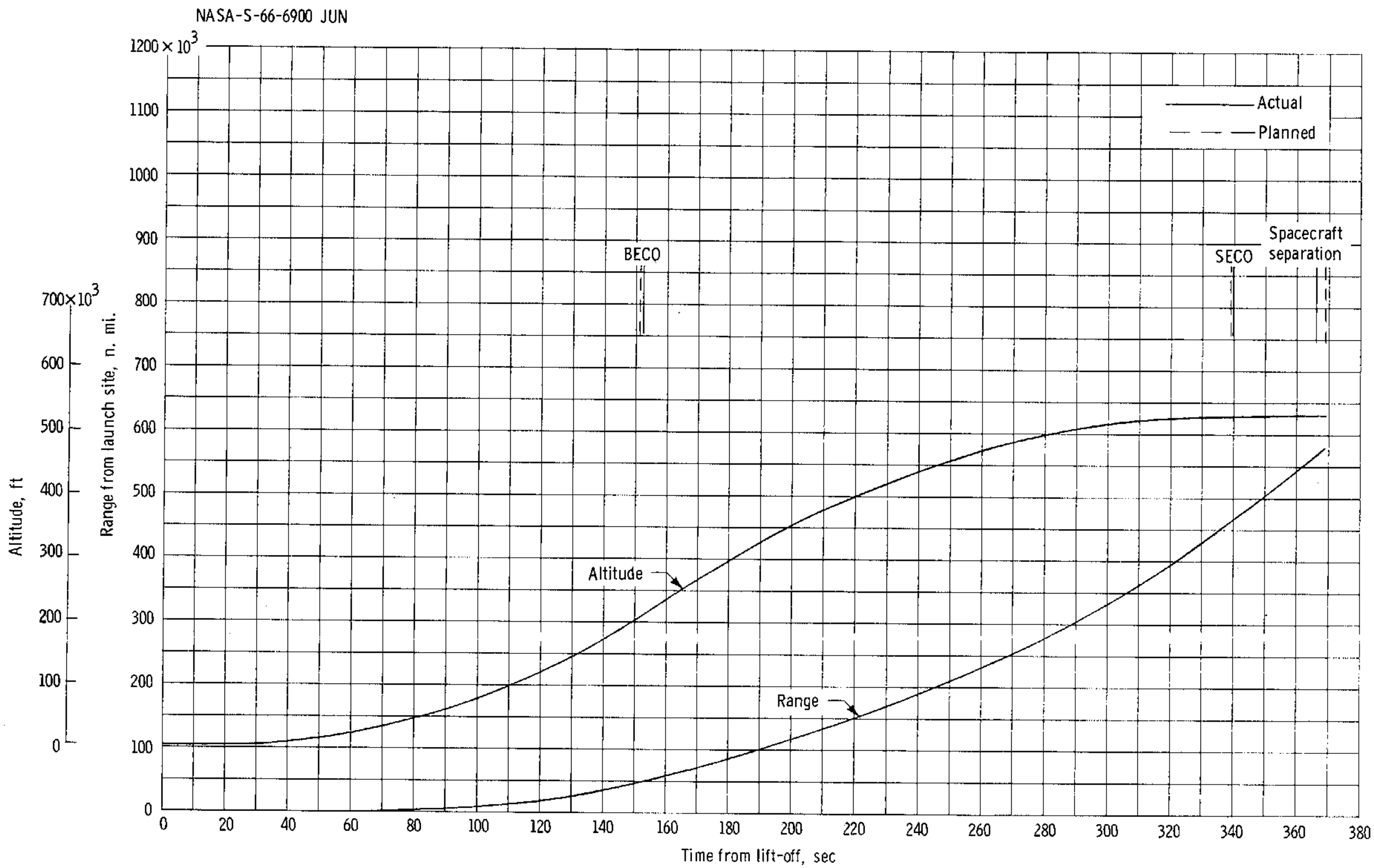

(a) Altitude and range.

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

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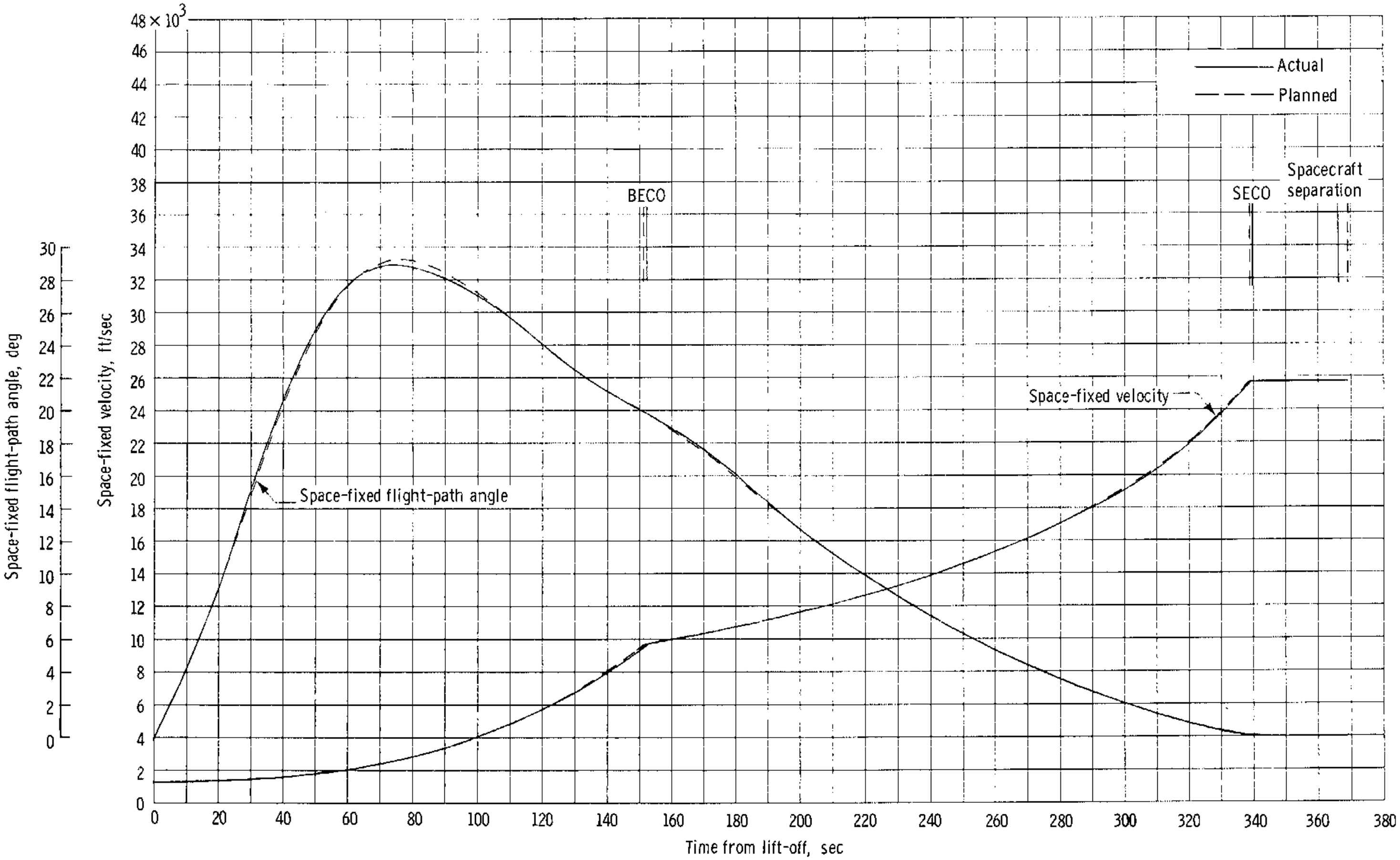
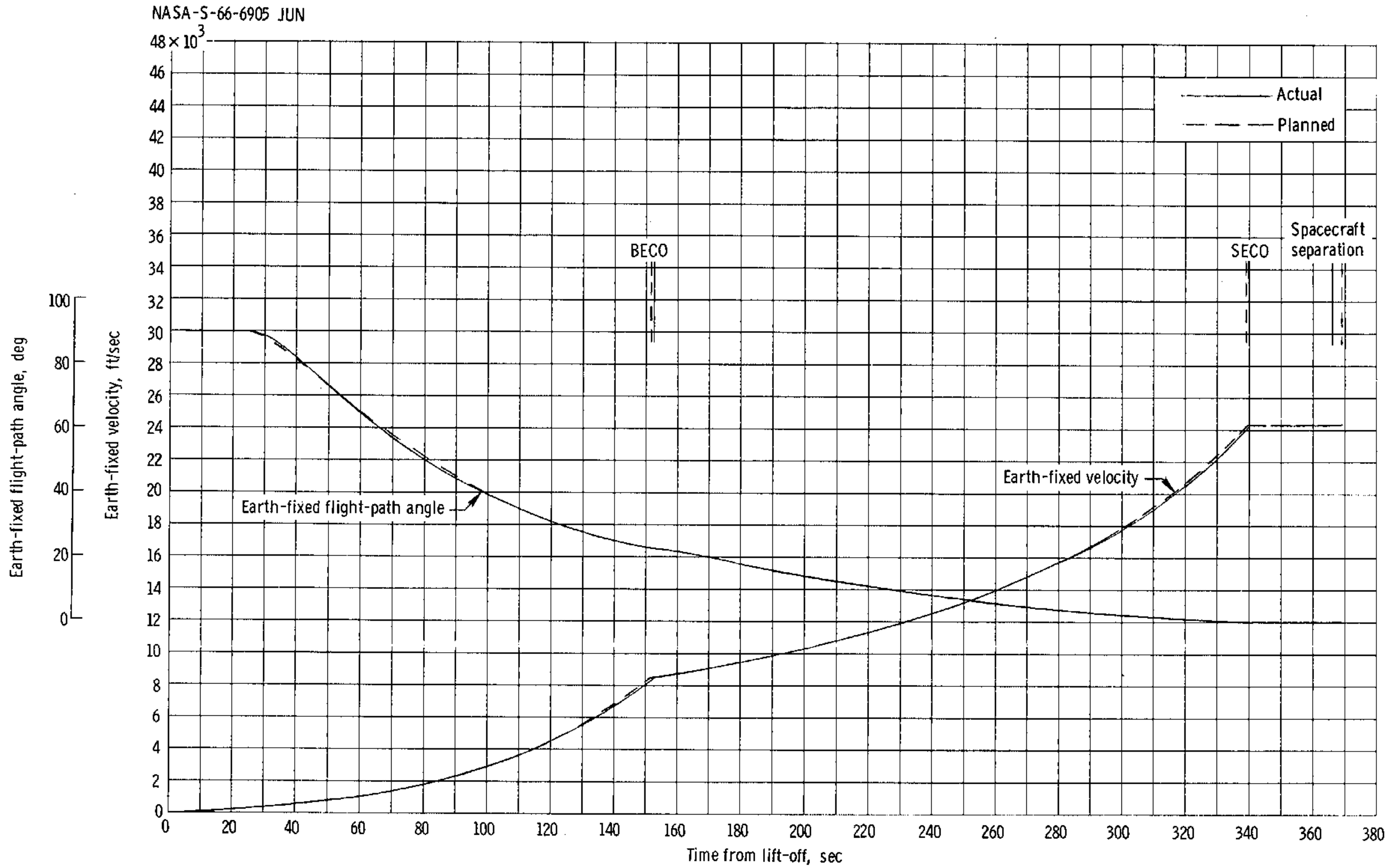


(b) Space-fixed velocity and flight-path angle.

Figure 4.3-2. - Continued.



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

Figure 4.3-2. - Continued.

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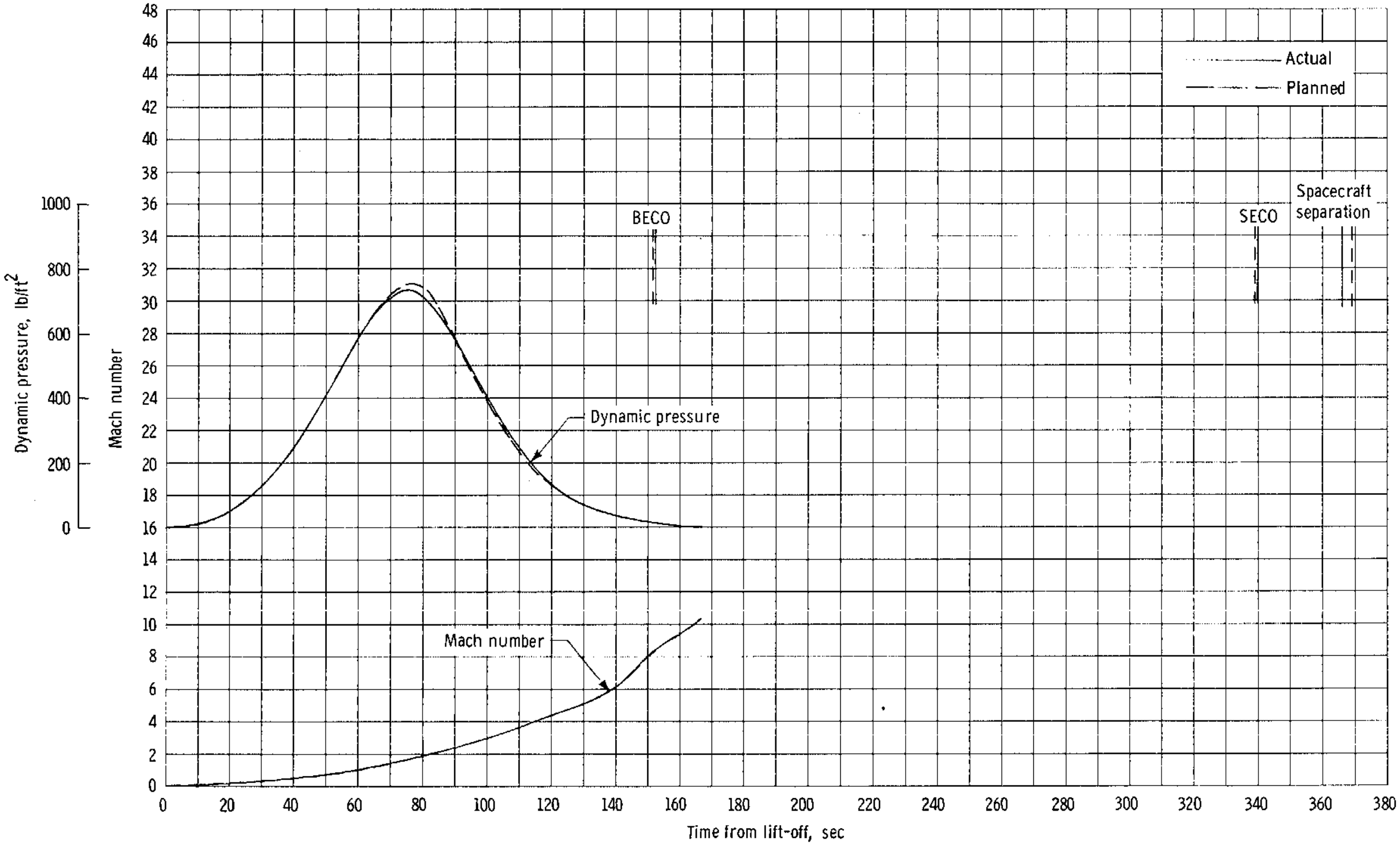

(d) Dynamic pressure and Mach number.

Figure 4.3-2. - Continued.

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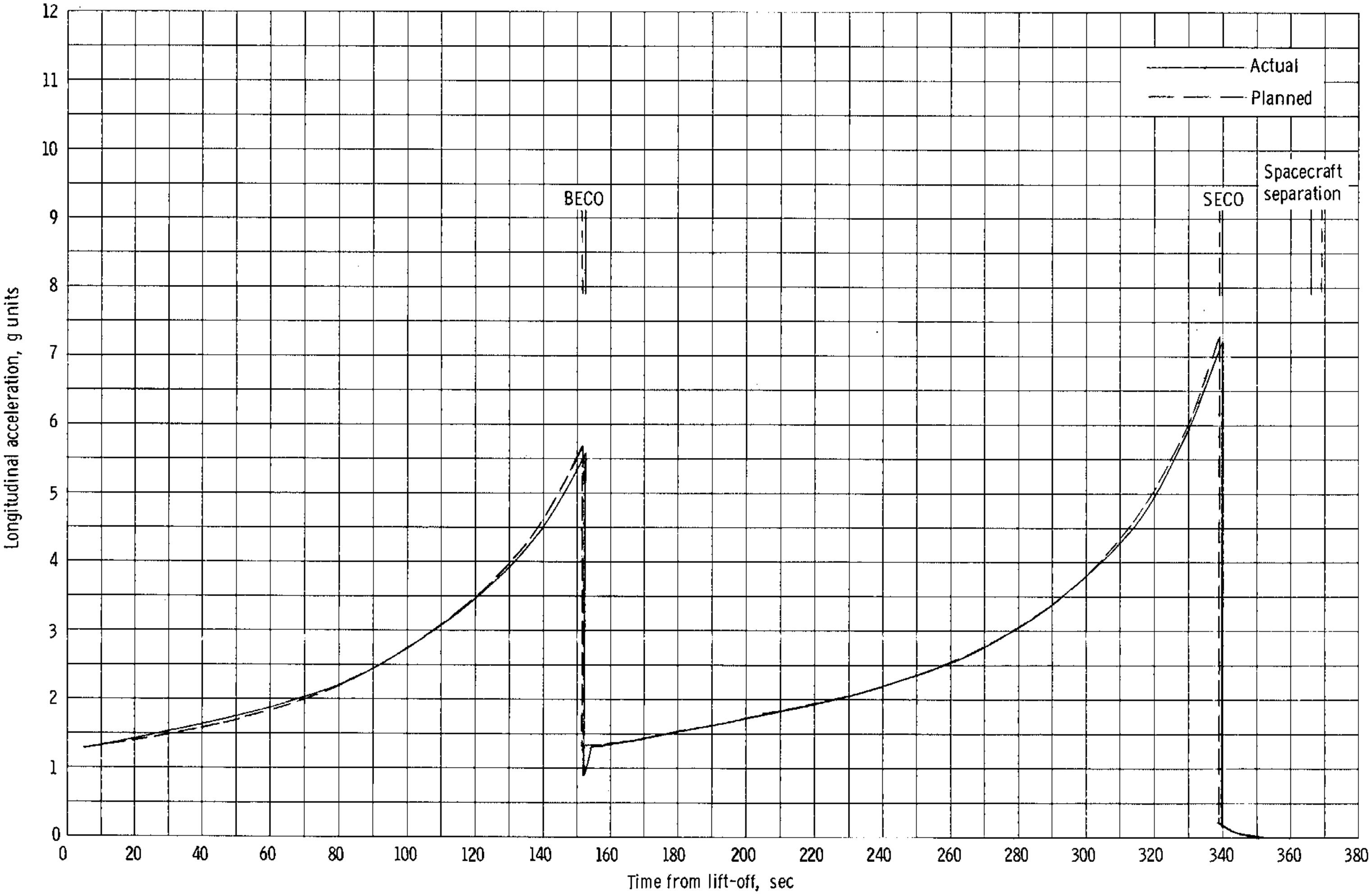

(e) Longitudinal acceleration.

Figure 4.3-2.- Concluded.

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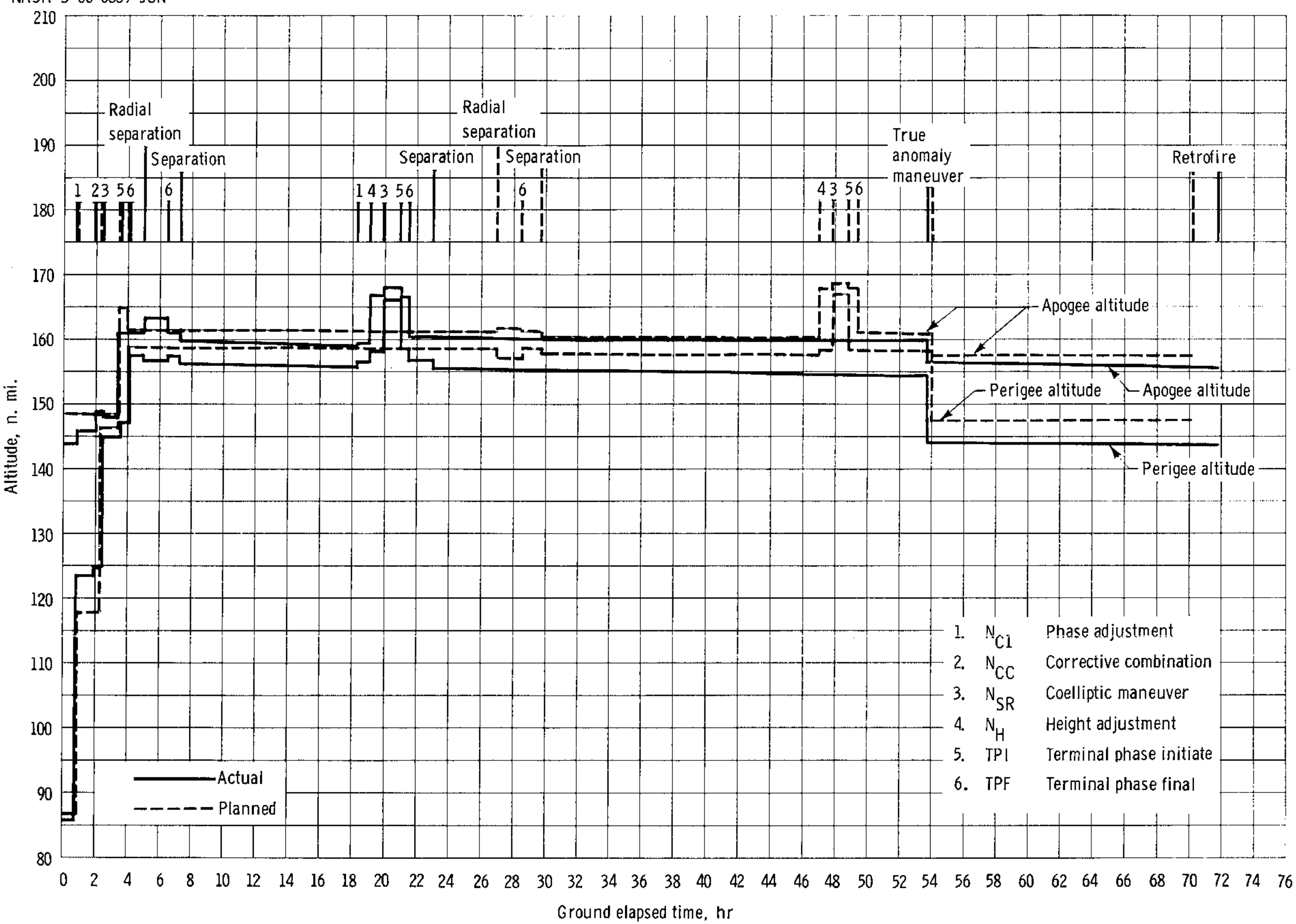

Figure 4.3-3. - Apogee and perigee altitudes for the Gemini IX-A mission.

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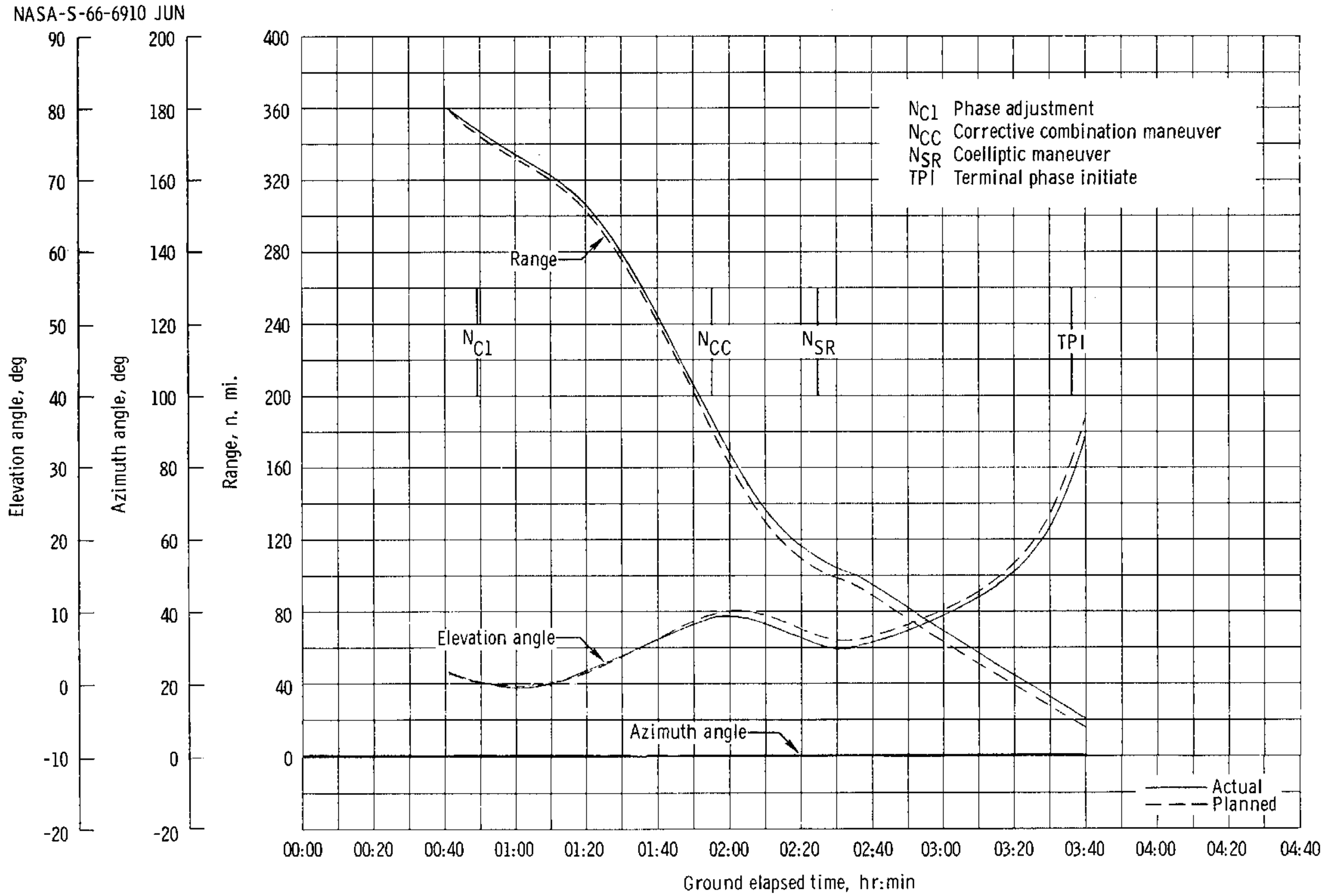

(a) Relative range, azimuth, and elevation from Spacecraft 9 to ATDA during midcourse maneuvers for M=3 rendezvous.

Figure 4.3-4. - Rendezvous during the Gemini IX-A mission.

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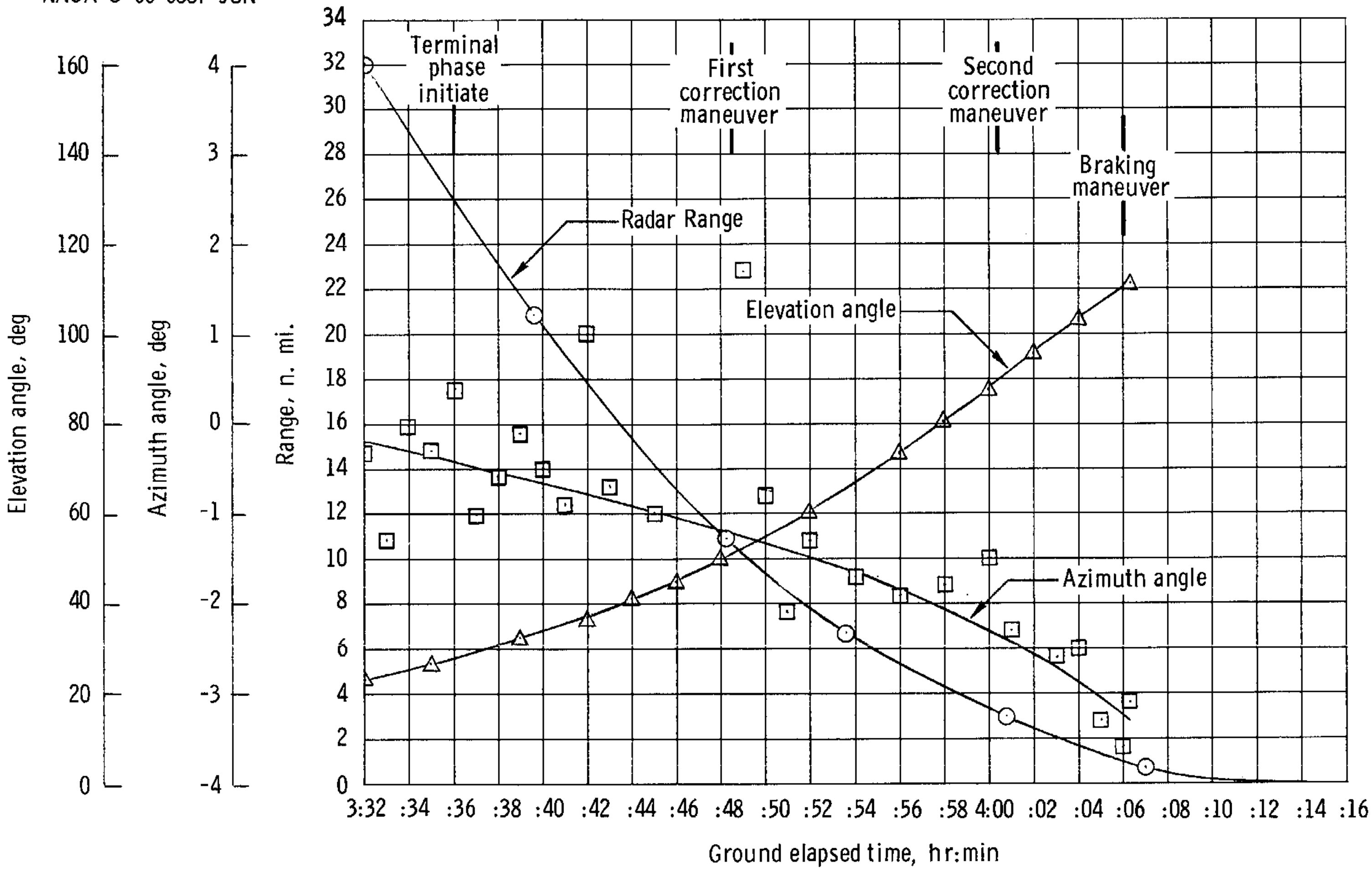
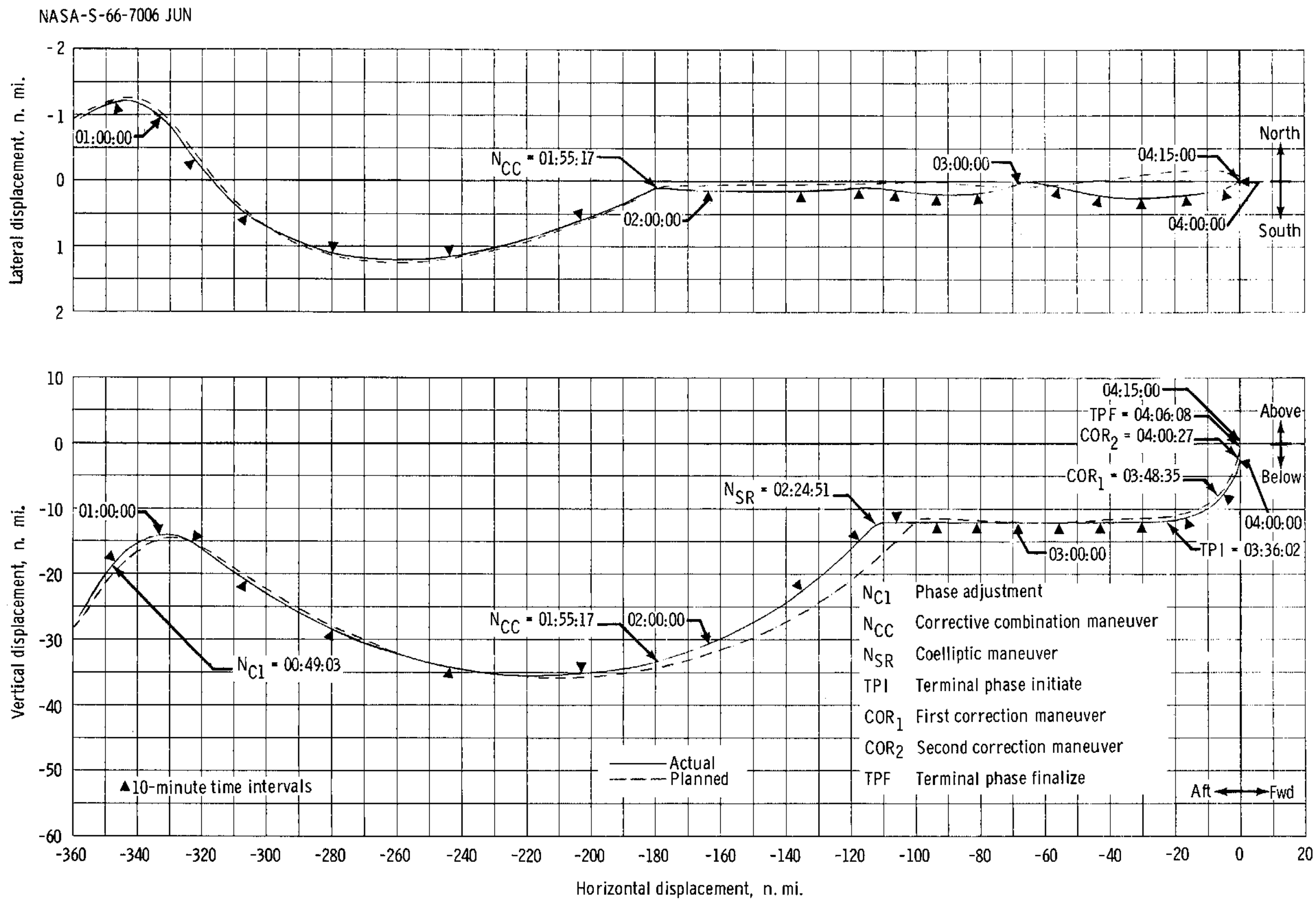


(b) Relative range, azimuth, and elevation from Spacecraft 9 to ATDA during terminal phase maneuvers of M = 3 rendezvous.

Figure 4.3.-4 - Continued.



(c) Relative trajectory profile for M = 3 rendezvous, measured from ATDA to Spacecraft 9 in curvilinear coordinate system.

Figure 4.3-4. - Continued.

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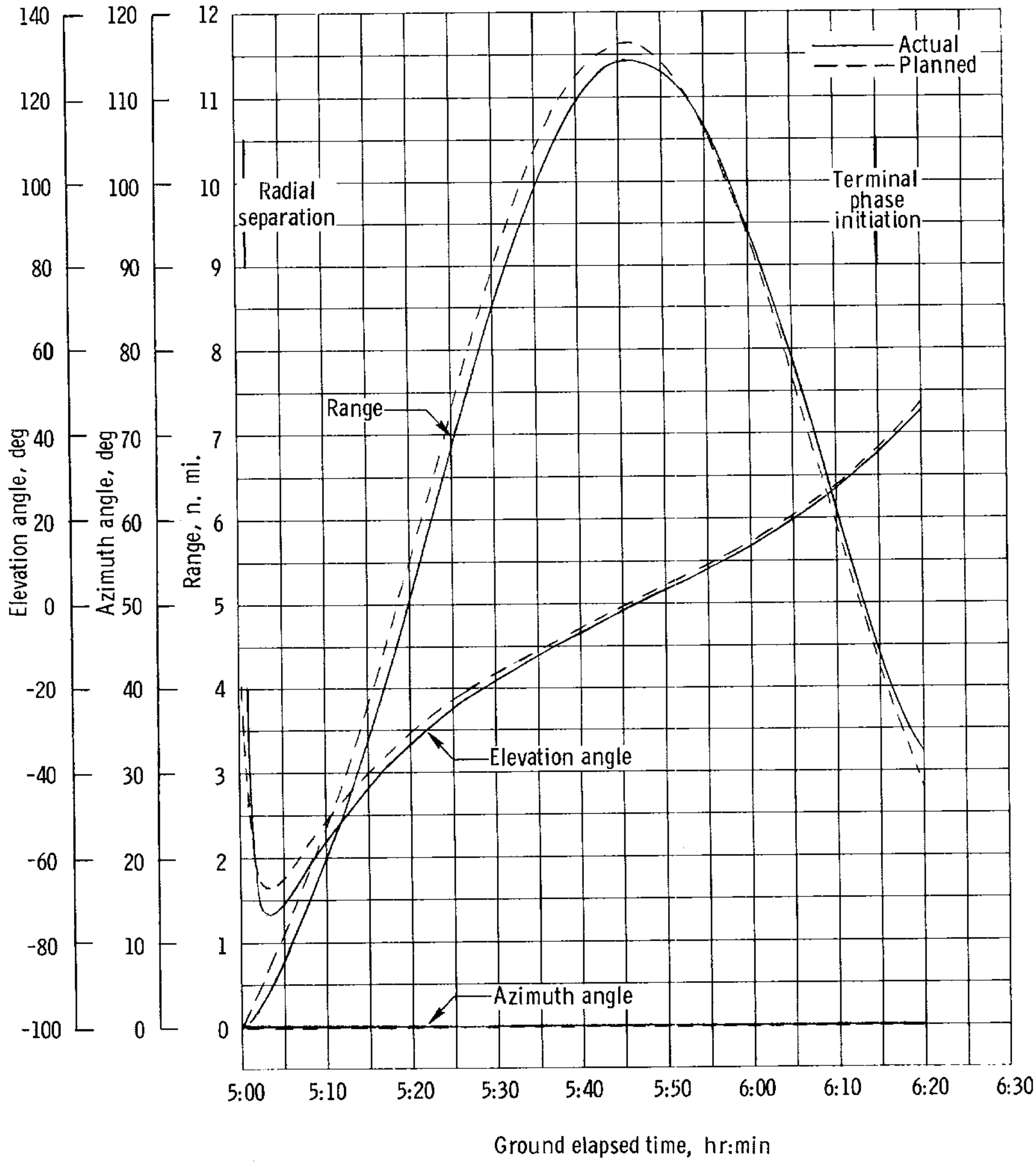

(d) Relative range, azimuth, and elevation from Spacecraft 9 to ATDA during midcourse maneuvers of the equi-period rendezvous.

Figure 4.3-4 - Continued.

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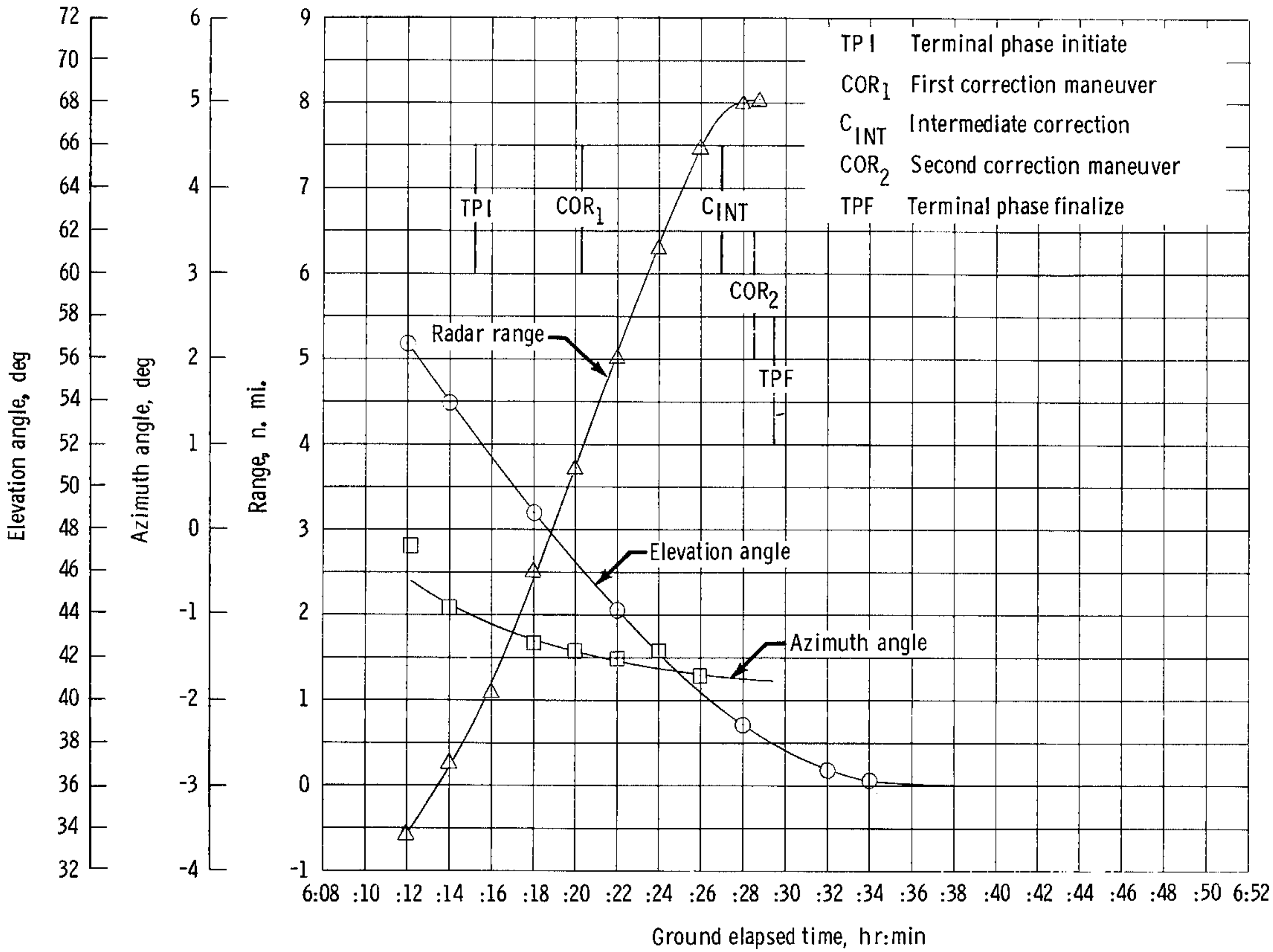

(e) Relative range, azimuth, and elevation from Spacecraft 9 to ATDA during terminal phase of the equi-period rendezvous.

Figure 4.3-4. - Continued.

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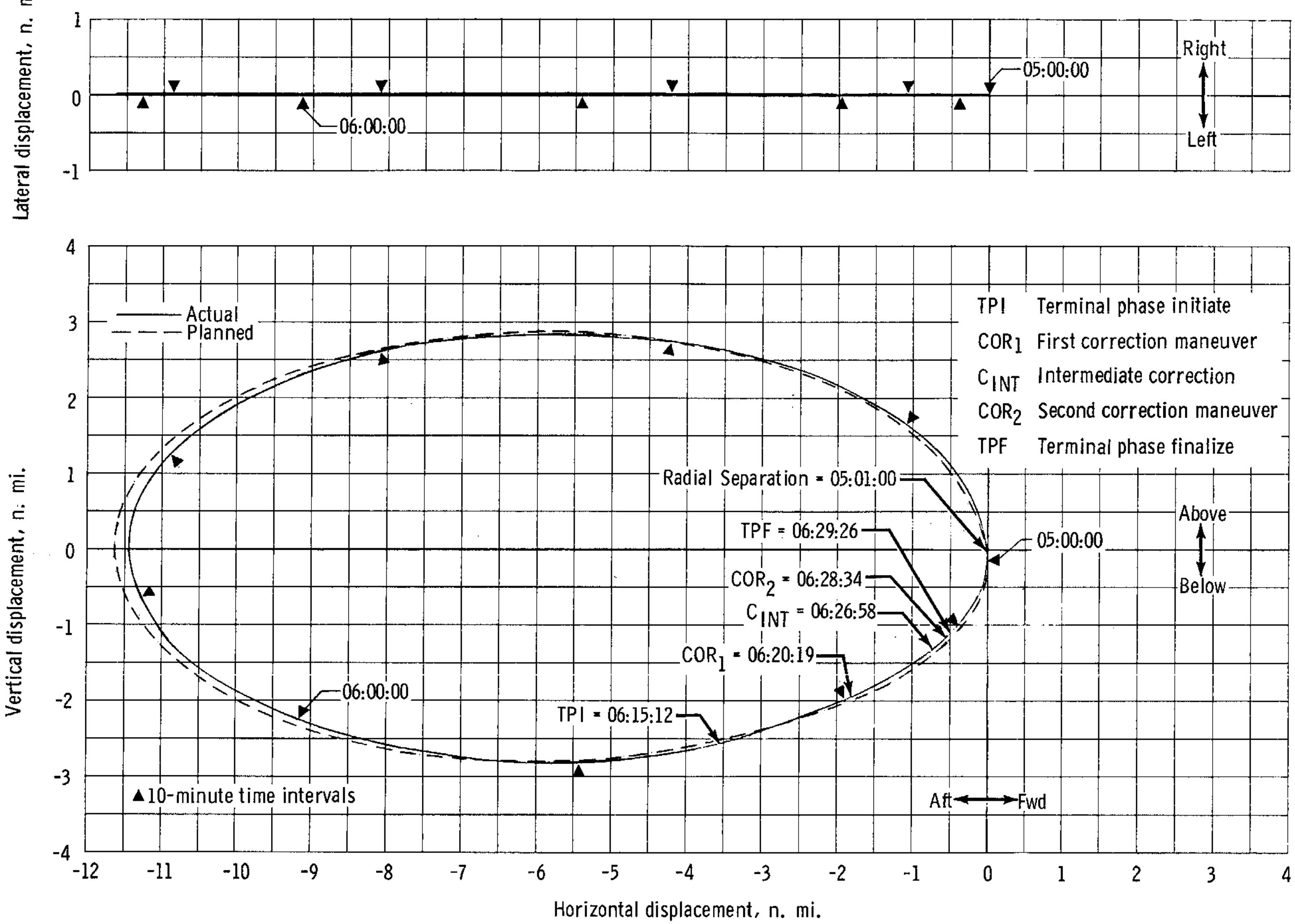

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(f) Relative trajectory profile of the equi-period rendezvous measured from ATDA to Spacecraft 9 in curvilinear coordinate system.

Figure 4.3-4. - Continued

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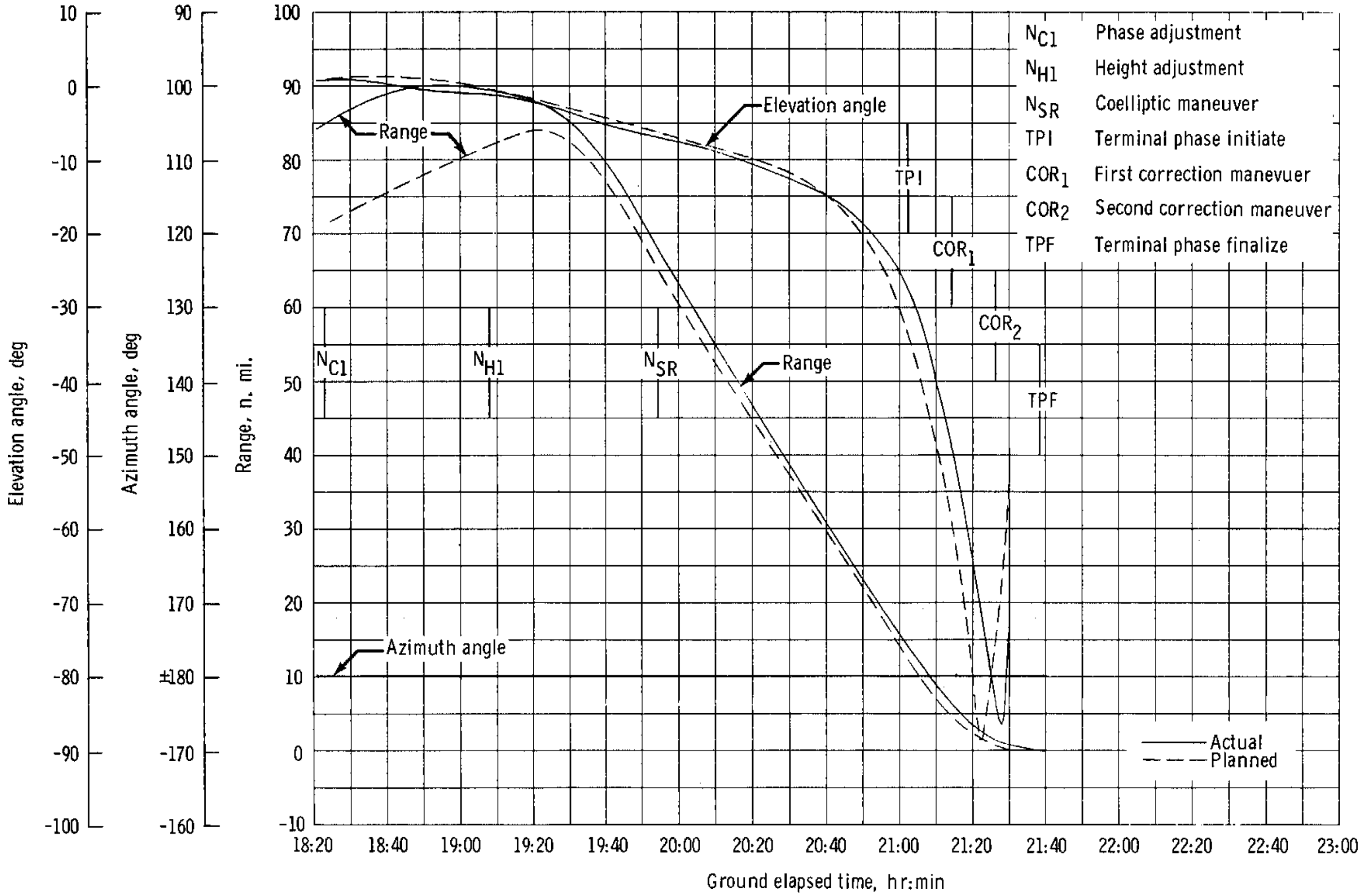

(g) Relative range, azimuth, and evaluation from Spacecraft 9 to ATDA during midcourse and terminal phase maneuvers of the rendezvous from above.

Figure 4.3-4. - Continued.

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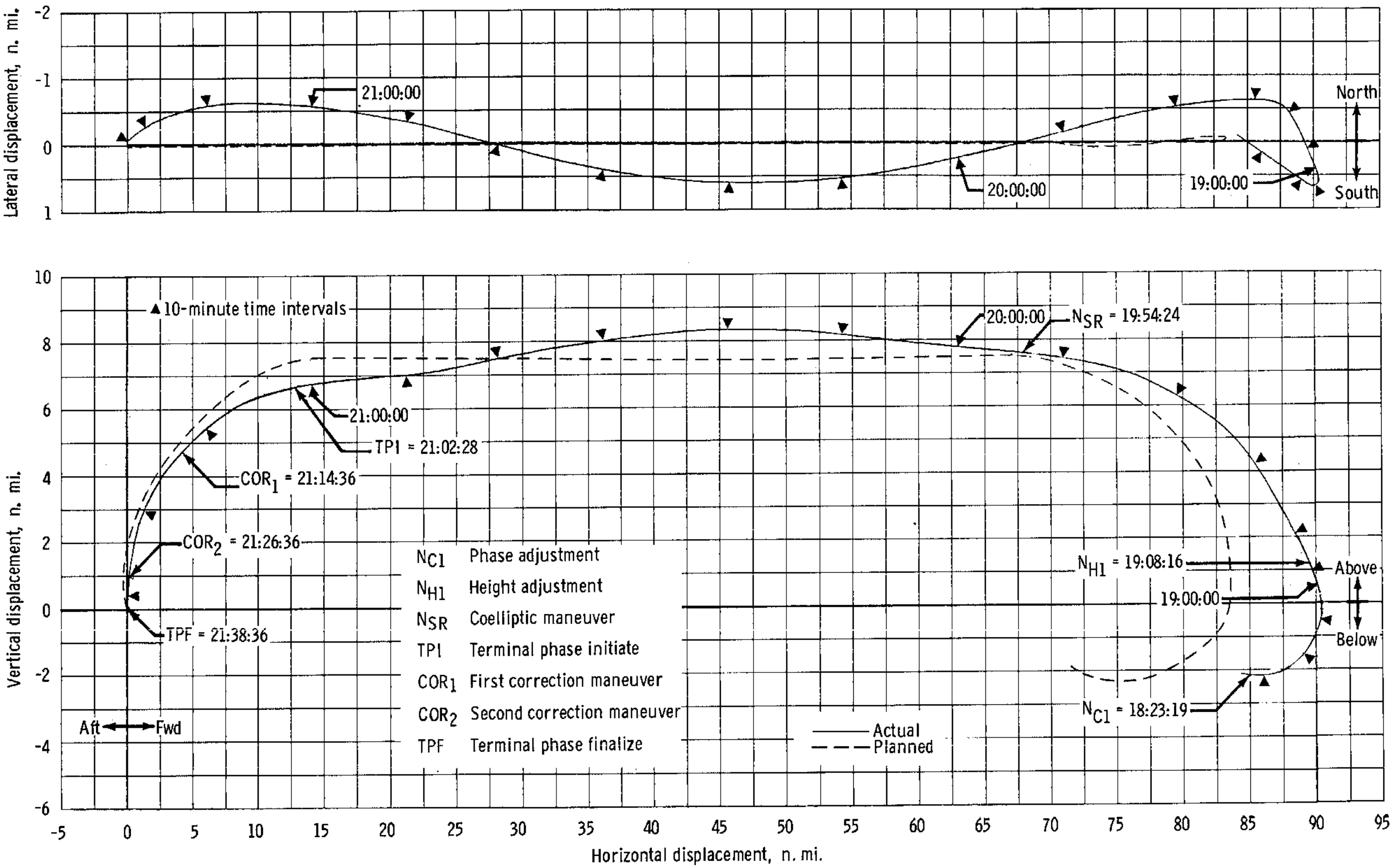

(h) Relative trajectory profile of the rendezvous from above, as measured from ATDA to Spacecraft 9 in curvilinear coordinate system.

Figure 4.3-4. - Concluded.

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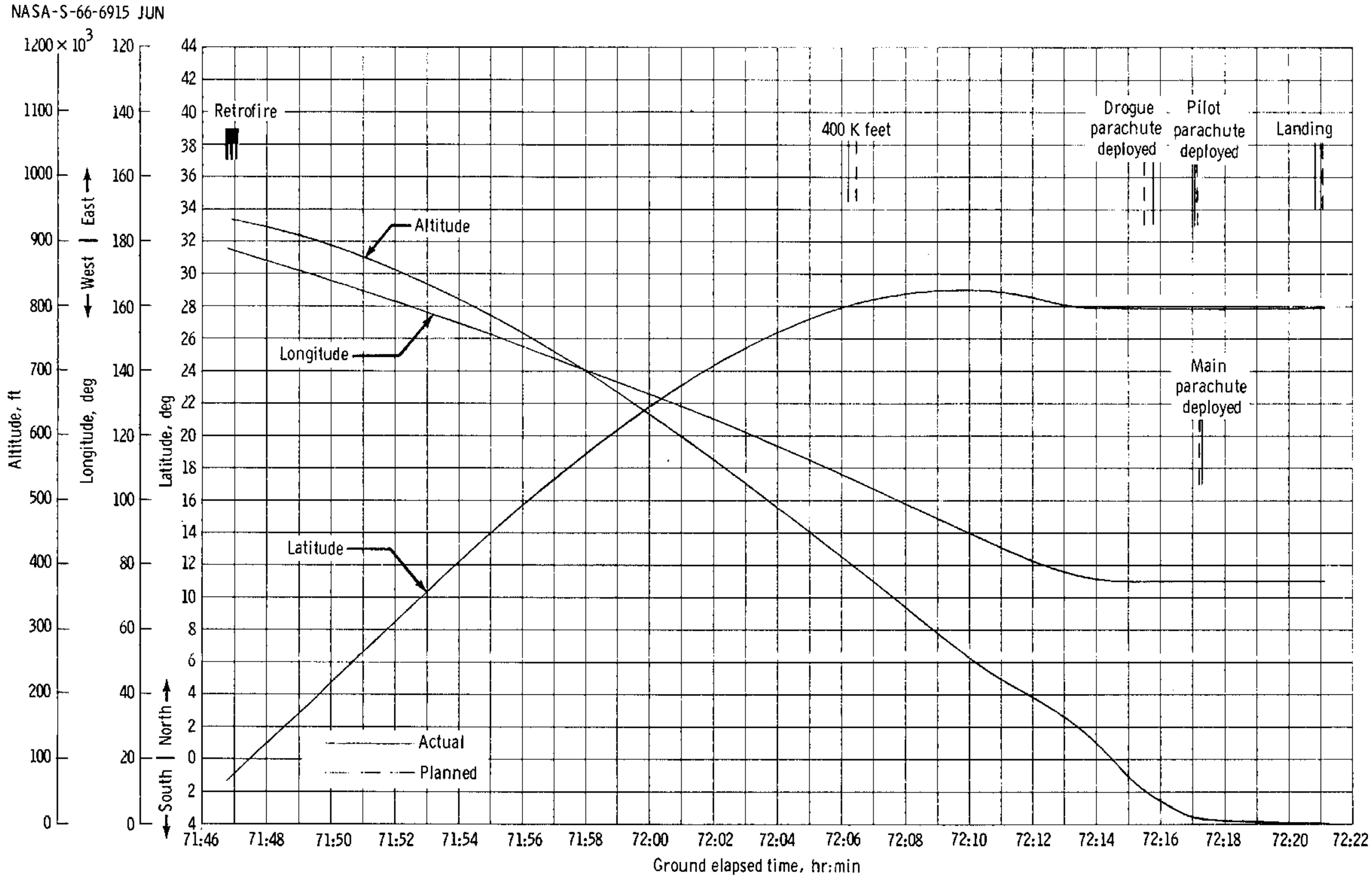

(a) Latitude, longitude, and altitude.

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

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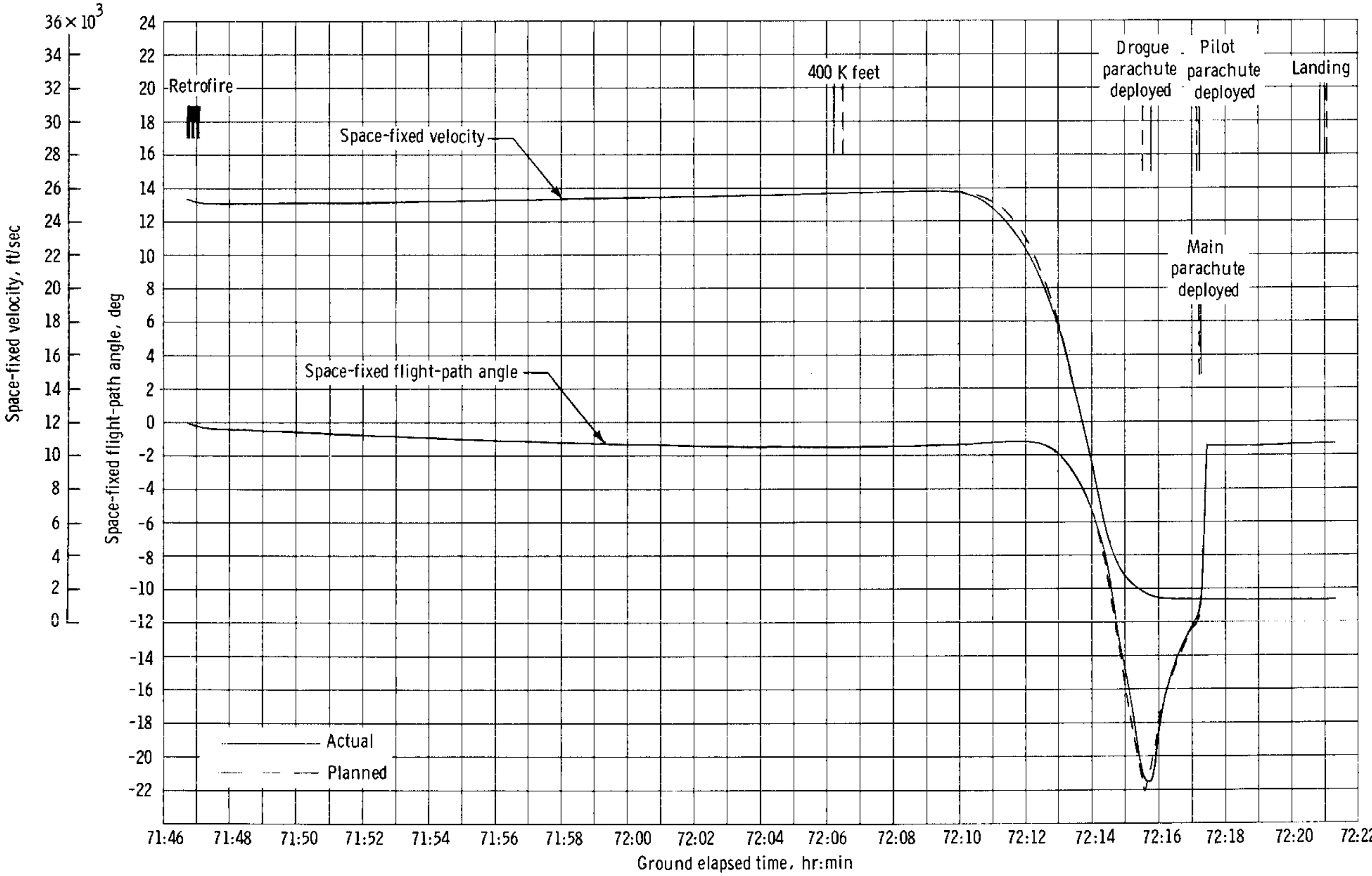

(b) Space-fixed velocity and flight-path angle.

Figure 4.3-5. - Continued.

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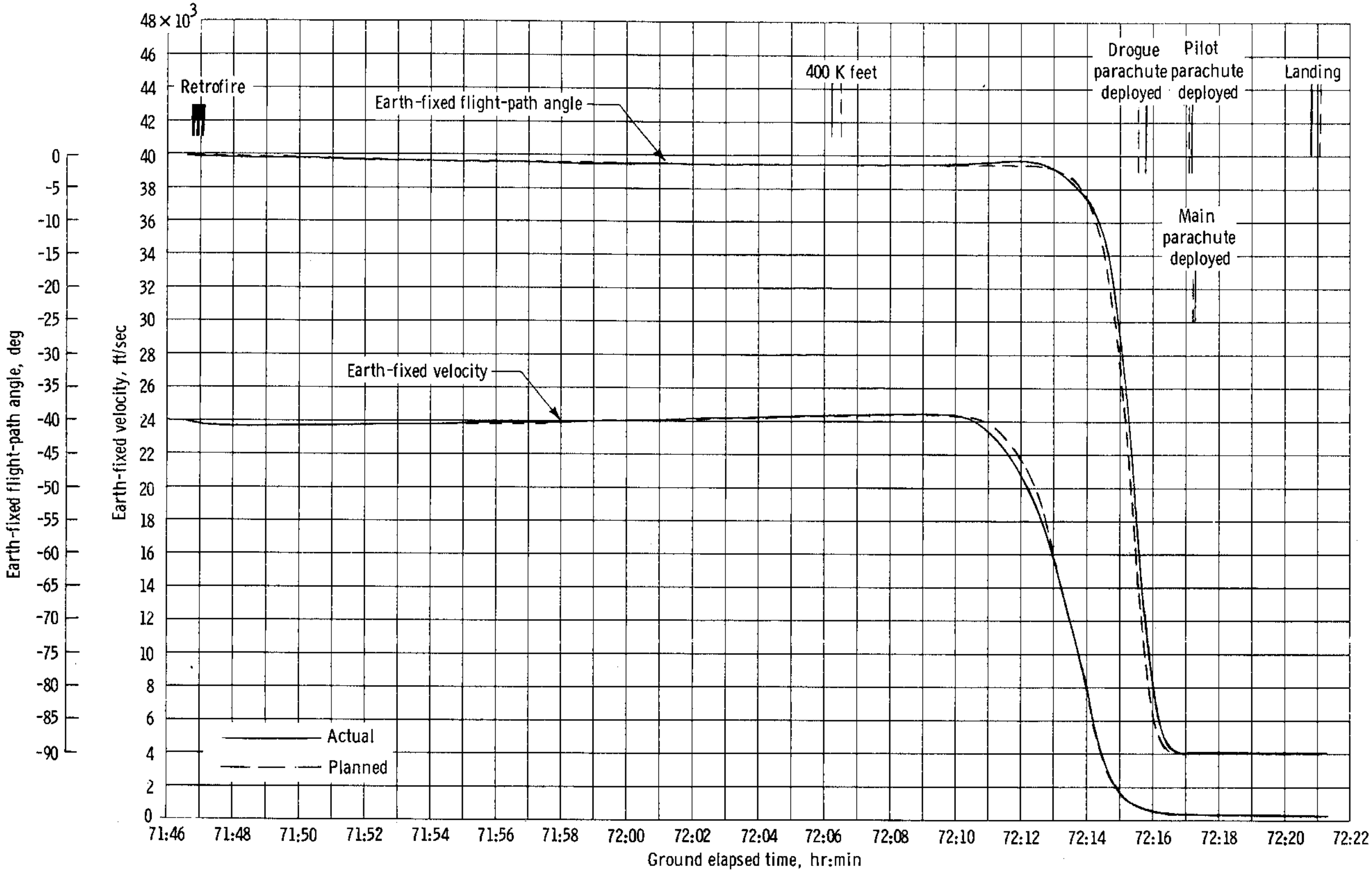

(c) Earth-fixed velocity and flight-path angle.

Figure 4.3-5. - Continued.

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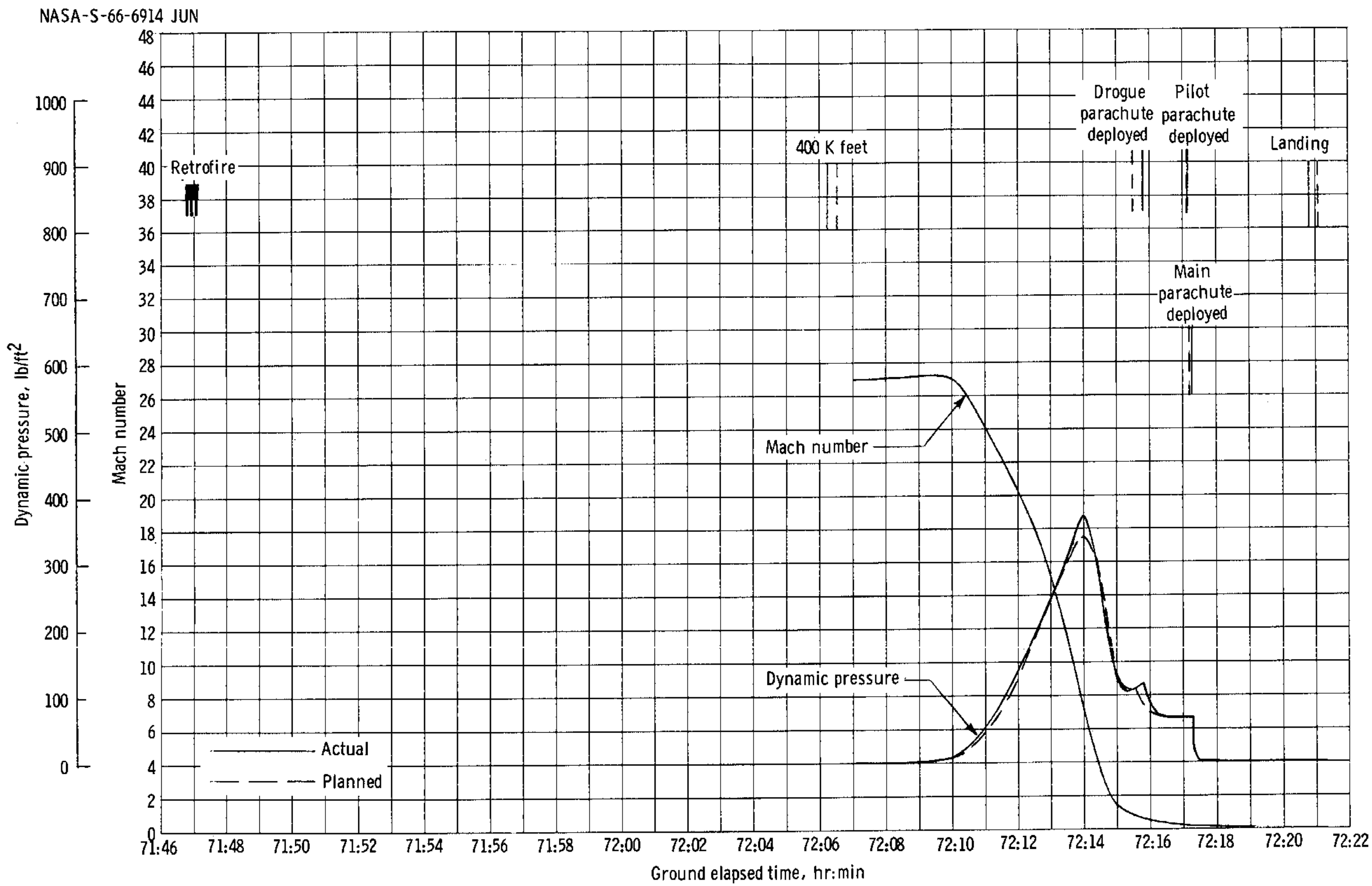
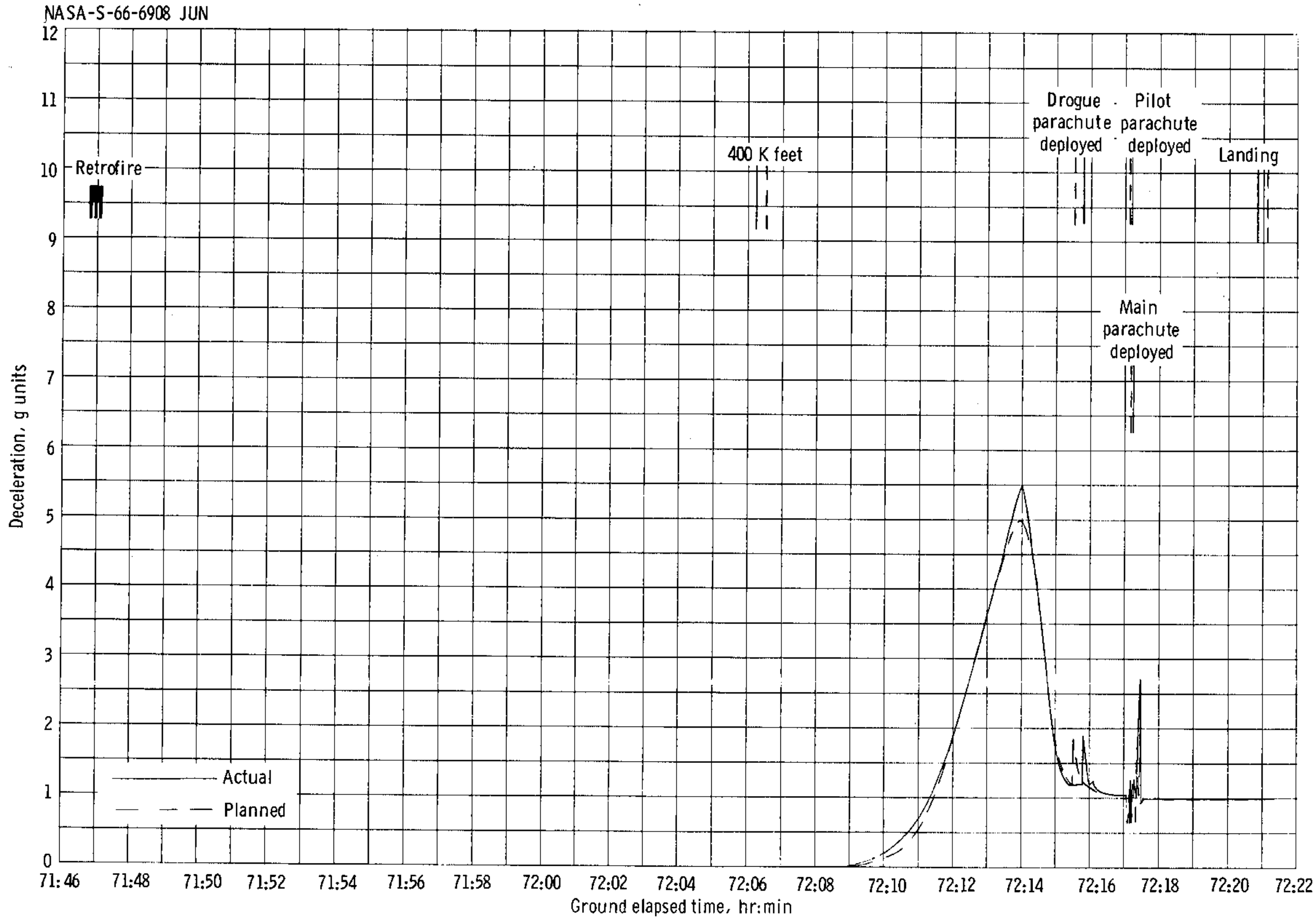


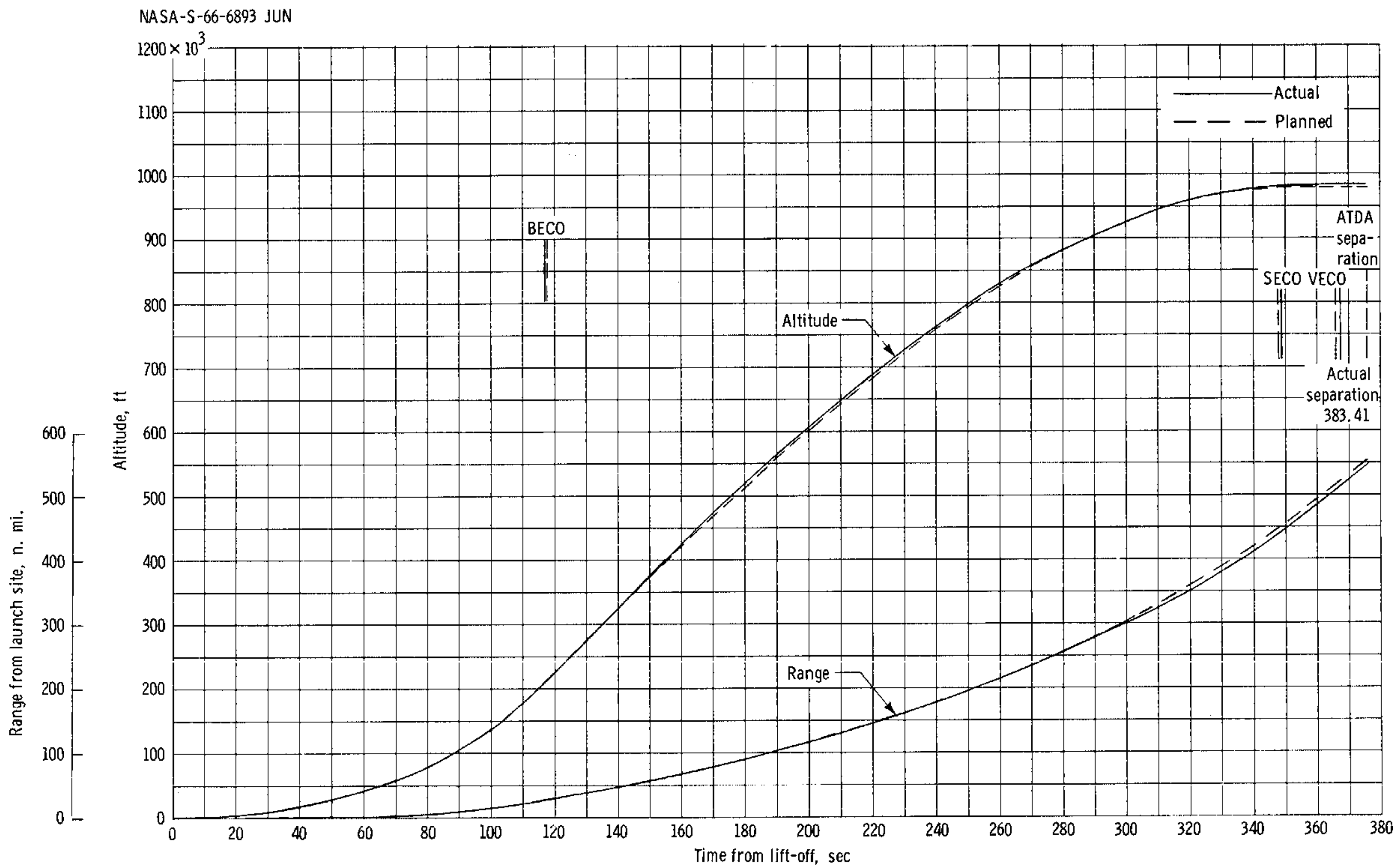
(d) Dynamic pressure and Mach number.

Figure 4.3-5. - Continued.



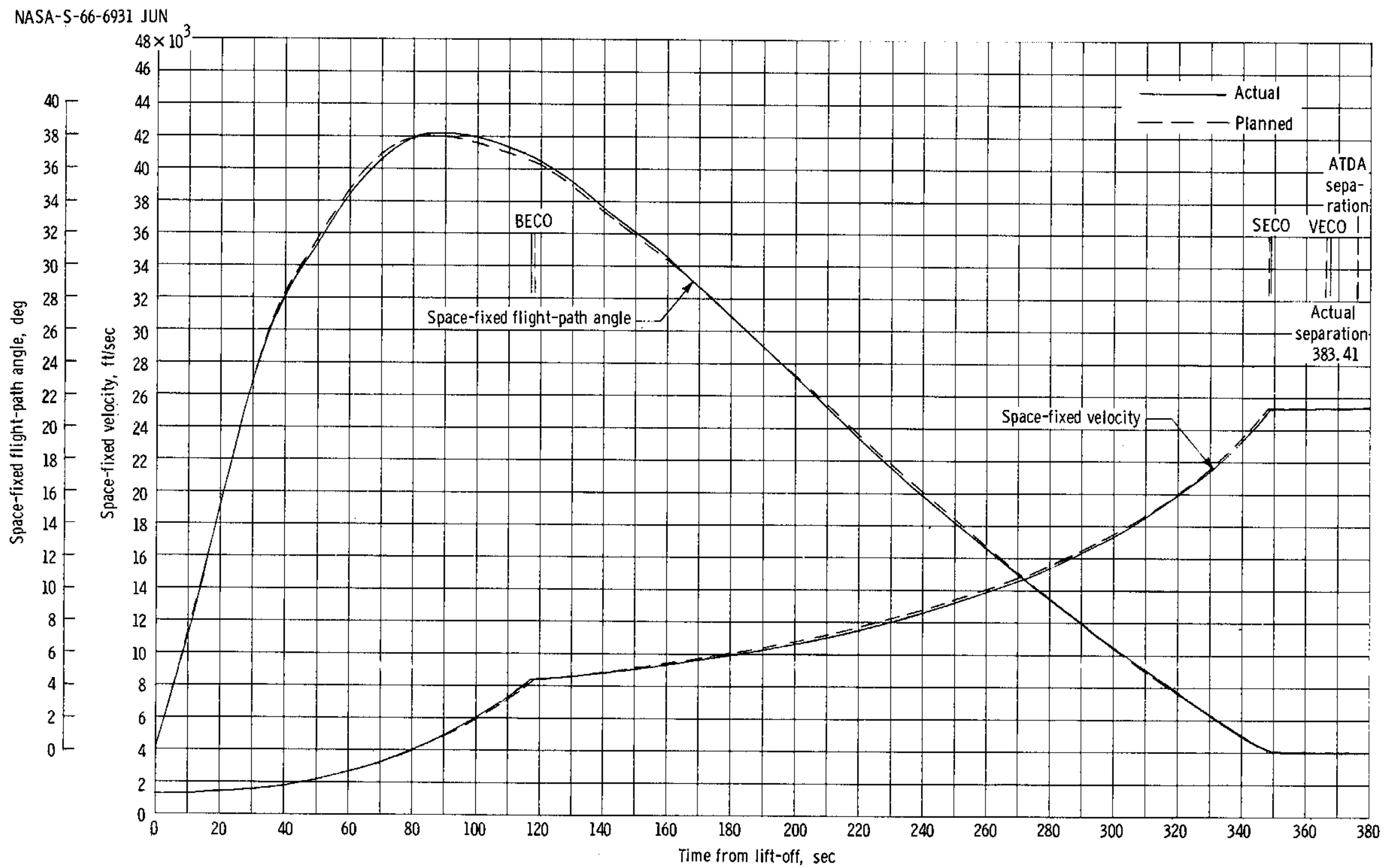
(e) Longitudinal deceleration.

Figure 4.3-5. - Concluded.



(a) Altitude and range.

Figure 4.3-6. - Trajectory parameters for TLV/ATDA launch phase.



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

Figure 4.3-6. - Continued.

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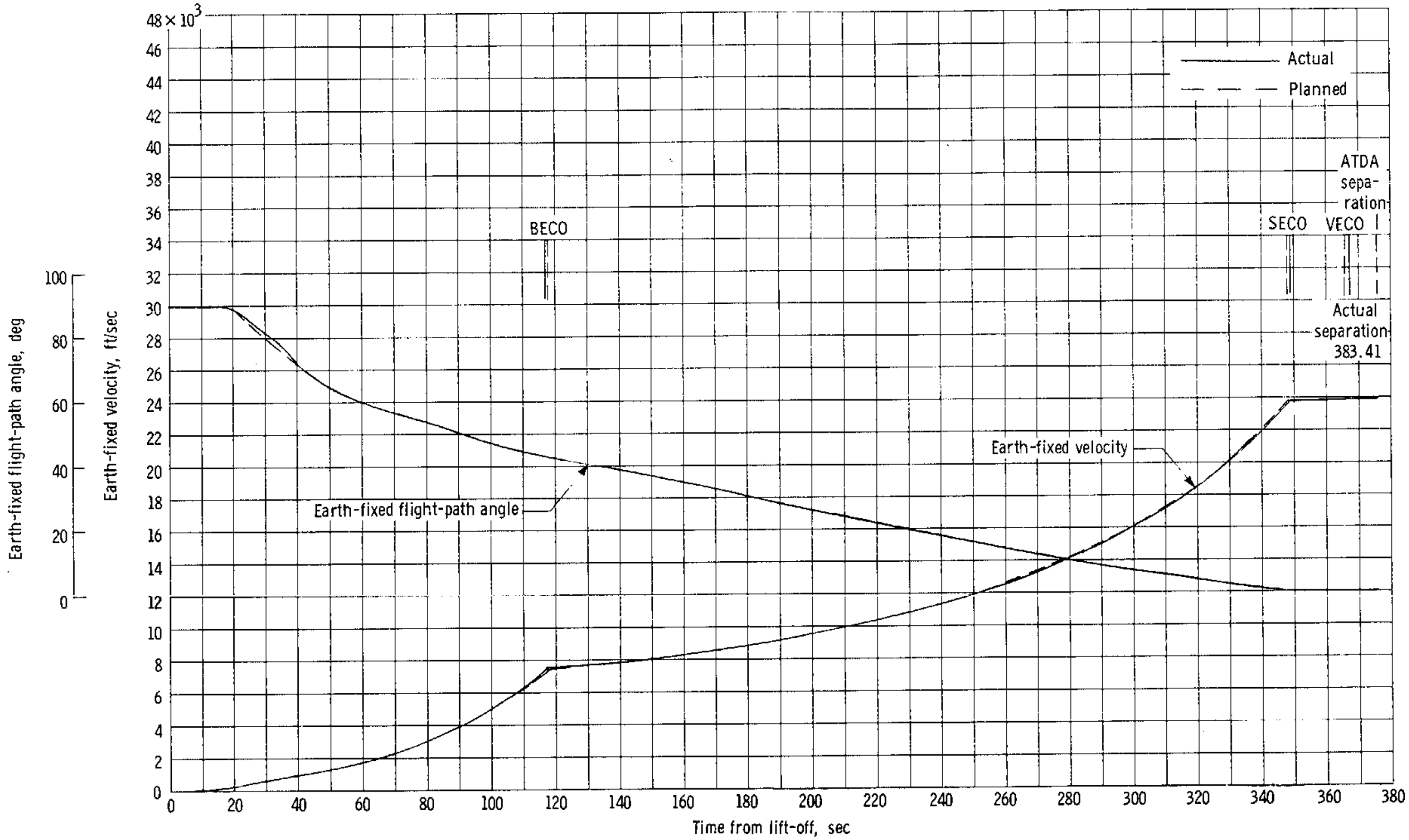

(c) Earth-fixed velocity and flight-path angle.

Figure 4.3-6. - Continued.

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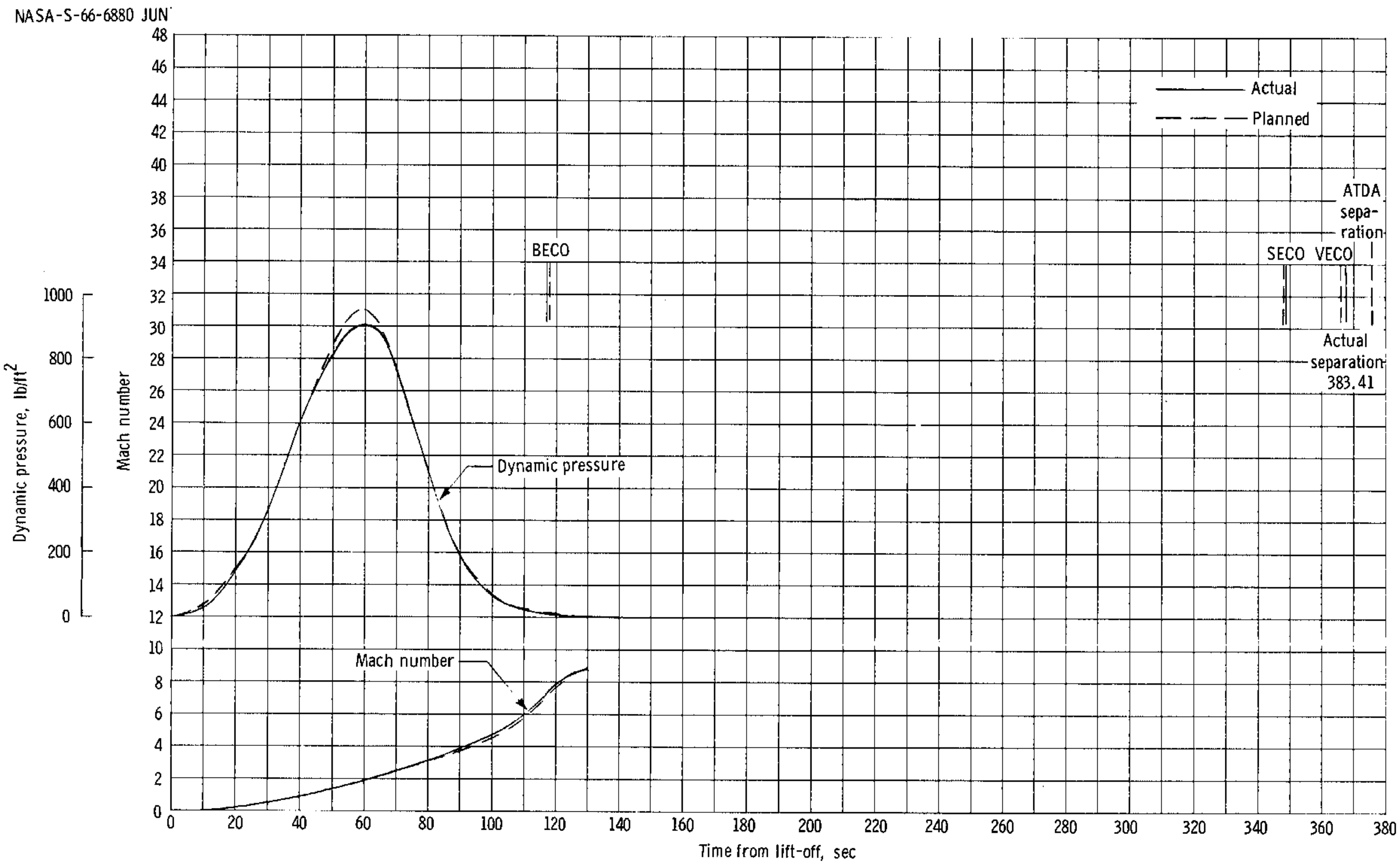

(d) Dynamic pressure and Mach number.

Figure 4.3-6. - Continued.

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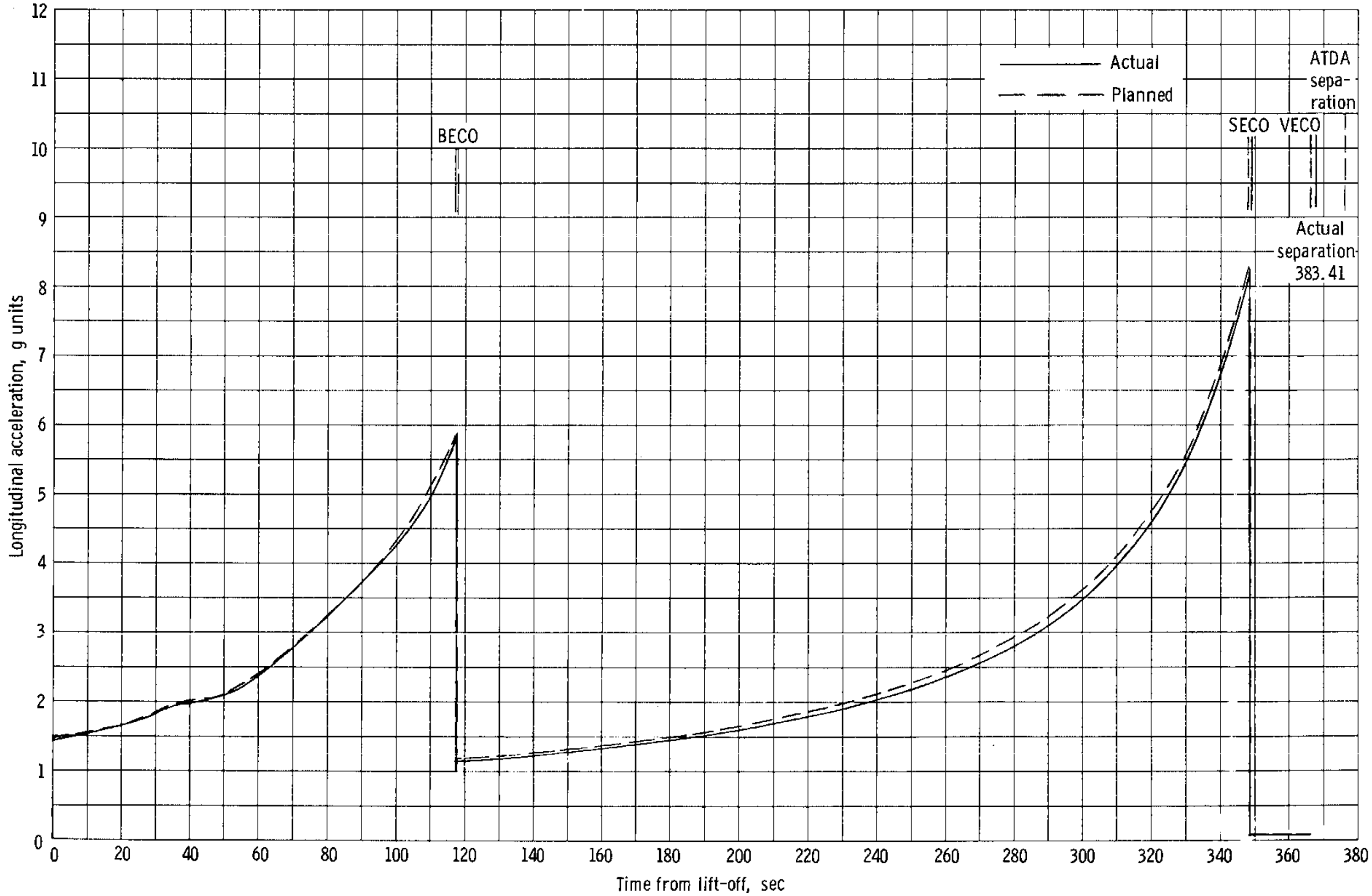

(e) Longitudinal acceleration.

Figure 4.3-6. - Concluded.

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5.0 VEHICLE PERFORMANCE

5.1 SPACECRAFT PERFORMANCE

5.1.1 Spacecraft Structure

The spacecraft structure sustained the loading and environment of the mission satisfactorily. The crew reported a hard landing and this probably caused the bent shingle noted by the swimmers while attaching the flotation collar. Ground tests have shown that damage of this type can be incurred when the 30-ft/sec descent rate is combined with the design limits of wind drift, parachute swing velocities, and landing attitude. The ground tests demonstrated the ability of the pressure vessel to retain its integrity under these circumstances. A comparison of the Spacecraft 9 shingle damage with damage incurred during the tests indicated that the Gemini IX-A landing loads were probably less than the design limits.

The launch window covers, used for the first time on Gemini IX-A, withstood the launch environment and jettisoned satisfactorily, although they were only partially successful in preventing the deposit of residue. Tests are underway to identify the composition of this residue so that corrective action can be taken.

The crew reported minor difficulties with some of the adapter-mounted mechanisms installed to support the extravehicular activities associated with donning of the Astronaut Maneuvering Unit (AMU). The umbilical guide, one of the handholds, and the AMU thermal protective cover did not deploy properly. After hearing the pilot's description of what he found, this was attributed to a failure of the protective cover to maintain alignment during the deployment sequence. Ground tests will be conducted so that proper changes can be made to future spacecraft. The pilot also reported that one general illumination light in the adapter failed to illuminate. The possibility that this may have been caused by pyrotechnic-induced shock is being investigated.

Difficulty was experienced with mounting the extravehicular activity (EVA) camera in its bracket. It was found that the camera mating part was not chamfered sufficiently to cam the detent easily. This will be corrected on later spacecraft.

The pilot reported some degradation of the holding characteristics of the Velcro on the rendezvous and recovery section. From his description, it appears that launch heating may have fused some of the hook ends of the material, reducing its holding power. The Velcro mounted on the conical section was not evaluated. That in the forward portion of the adapter was, and it appeared to be undamaged. It was subjected to less launch heating than the forward portion of the spacecraft.

Postflight inspection revealed blistered paint on the ejection seat and the egress kit, indicating temperatures of higher than 200 degrees F in this area during the hatch-open period of EVA. A review had been conducted to determine whether heating would damage components inside the spacecraft. However, because the temperatures were higher than expected, another review will be conducted to ensure that temperature-critical components such as pyrotechnics and experiments are properly protected.

The crew reported difficulty in opening and closing the hatch during EVA. Postflight tests have revealed an increase in hatch-actuator loads during extension and retraction near the midpoint of travel. The measured load was approximately 100 pounds, which translates into 53 pounds when applied perpendicularly to the hatch at the cloth handle. Lubrication was applied to the piston of another actuator with similar characteristics, and the load was lowered to one-half the former value. A critical disassembly and inspection of the flight actuator was made. The appearance of all internal parts of the device was normal. Lubricating the actuator would have afforded a slight advantage and will be accomplished during launch preparations. However, with the present hatch-closing device, forces would have still been higher than desired and an improved device is being designed for Spacecraft 10, 11, and 12.

The apparent heat-shield stagnation point measured 19.6 inches below the center. This indicates an angle of attack of 10 degrees, which is considered nominal.

5.1.2 Communications Systems

All spacecraft communications equipment performed in a satisfactory manner and without evidence of malfunction. During the postmission debriefings and data analyses, a few minor areas of concern were noted and investigated.

Seven tapes of acceptable quality were recorded on the spacecraft voice tape recorder during the mission. Portions of both transmitted and received voice communications were recorded.

Communications blackout occurred during reentry from 72:08:32 to 72:13:54 ground elapsed time (g.e.t.). These times were determined from the real-time telemetry signal-strength charts recorded at the Texas station and the Mission Control Center at Cape Kennedy (MCC-C), respectively.

During this mission, as in previous missions, there were several instances of poor intelligibility during air-to-ground voice communications. The usual causes are microphone positioning, low audio level, and interference caused by high breath noise. Background noise, probably due to air turbulence in the space suit, was intense at times and seemed to vary with crew movement or with suit or neck dam adjustment. Because of automatic-gain-control action in the microphone amplifiers, a lower than normal or a momentary decrease in the voice level of a crewmember was transmitted to the ground with a substantial increase in background noise level. Interference with mission operations caused by these minor abnormalities was insignificant; however, when they occurred simultaneously with a noisy circuit between the Mission Control Center at Houston (MCC-H) and the remoted network station, the result was poor intelligibility at MCC-H.

5.1.2.1 Ultrahigh frequency voice communications.- Ultrahigh frequency (UHF) voice communications were satisfactory for mission support except during the extravehicular activities (EVA). During this period, UHF voice communications between the crew and the ground were weak, broken, distorted, and barely adequate to support the mission. Postflight investigation of the MCC-H air-to-ground voice tapes and the onboard voice tapes indicated that a lower-than-normal audio level from the command pilot caused the voice operated transmitter (VOX) to release too soon, resulting in broken words and syllables. The voice quality of the command pilot was degraded in comparison with the pilot during those mission phases which required the crewmembers to wear flight helmets, that is, launch, EVA, and reentry. This may have been because the microphones in the command pilot's flight helmet were different in style from those in the pilot's flight helmet. It is not planned to use VOX communications during future Gemini flights.

UHF voice communications were excellent between the spacecraft and the recovery forces from the end of reentry blackout until after landing.

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 HF equipment was not used during the orbital mission phase. Because of the rapid deployment of pararescue personnel, no attempt was made to use the HF equipment for either direction-finding or voice communications during the postlanding mission phase.

5.1.2.3 Radar transponder.- The radar transponder configuration 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. Beacon-sharing operations by ground radars were satisfactory. There were no problems with the spacecraft equipment. C-band tracking during reentry was satisfactory. During the communications blackout, the spacecraft was skin-tracked by Merritt Island Launch Area and Patrick Air Force Base radars. The recovery ship reported skin-track radar contact at the end of communications blackout at a range of 304 nautical miles.

5.1.2.4 Digital Command System.- The performance of the Digital Command System (DCS) was satisfactory throughout the mission. The T-3 minute update could not be transferred to the spacecraft because of a failure in ground equipment, and the two ascent guidance updates were inhibited from being transmitted to the spacecraft. This problem is discussed in section 6.2 of this report. Flight control personnel reported that all commands sent to the spacecraft were validated.

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 antennas which were deployed operated properly during the mission, as evidenced by the adequate performance of the communications system. The HF whip antenna installed on the adapter assembly was not extended in orbit. The HF whip antenna installed on the reentry assembly was not deployed for the postlanding phase of the mission. After recovery, the UHF descent antenna was examined and found to be bent at the base, which caused it to deploy at an angle of 30 degrees from the stowed position. Recovery photographs indicated that this condition existed prior to attachment of the flotation collar. Postflight inspection revealed that the mounting fitting was reversed at the base and therefore the whip pointed toward the heat shield rather than forward in the stowed position. In this position, friction between the element screw ends and the deploying parachute bridle would tend to jam the antenna element into the base. It is believed that this occurred at two-point suspension and caused a permanent bend in the antenna at the base. Corrective action to prevent a recurrence of this situation will be taken.

5.1.2.7 Recovery aids.- All communications recovery aids operated normally with the exception of intercommunications equipment employed between the swimmers and the spacecraft. 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 35 nautical miles. UHF voice communications between the spacecraft and the recovery forces were satisfactory. The flashing light extended normally, but was not necessary and was not turned on by the crew. During the recovery phase of the mission, prior to opening the hatches, communications between swimmers and the crew were intermittent. Postflight investigation indicated that the swimmers' intercommunications equipment was defective. The 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 except for the following three anomalies:

(a) The pulse code modulation (PCM) tape recorder malfunctioned during the fifth revolution.

(b) The Reactant Supply System (RSS) hydrogen cryogenic-mass-quantity measurement failed.

(c) The left-hand suit-inlet air-temperature measurement (CC03) failed.

5.1.3.1 PCM tape recorder failure.- During the fifth revolution, the PCM tape recorder failed in the record mode. A review of the real-time data from the tracking ship Rose Knot Victor (RKV) confirmed that the recorder slowed down to approximately 30 percent of the normal record speed at 6:35:52.6 g.e.t., resumed normal speed 1.8 seconds later, and then stopped completely approximately 2 minutes later at 6:37:49.3 g.e.t. Data received at the tracking ship Coastal Sentry Quebec (CSQ) showed no tape motion during a pass later in the fifth revolution. When the tape dump was initiated over Hawaii, a tape-motion indication was obtained for only 8.1 seconds in the playback direction. Near the end of the Hawaii pass, when the playback command was removed, tape motion was obtained for 18.3 seconds in the record direction until loss of signal (LOS) occurred. The Hawaii station was unable to decommutate the PCM data signal obtained from the ground-recorded tape made during the spacecraft pass. A later playback of the spacecraft recorder, commanded during the sixth revolution over Hawaii, gave similar results. In an attempt to conserve tape availability for the extravehicular activity (EVA) and reentry periods, a playback command was sent from the CSQ at approximately 10:36:00 g.e.t. in an attempt to reach the start-of-tape position. At 20:06:06 g.e.t., the playback command was removed by the Carnarvon (CRO) station, and the tape-recorder-control circuit breaker was turned off by the crew. At approximately 48:59:00 g.e.t., this circuit breaker was turned on by the crew over Carnarvon, and tape motion in the record direction was indicated in the real-time data. At 49:40:22 g.e.t. during revolution 31 over Texas (TEX), tape motion stopped. A tape playback attempt over Carnarvon on the revolution 32 pass resulted in no useful dump data being received. The recorder was left in the record mode until approximately 58 hours g.e.t., at which time the tape recorder circuit breaker was turned off by the RKV.

After recovery of the spacecraft, data were obtained for the following times from the PCM tape recorder:

Start, g.e.t.	Stop, g.e.t.	Interval, hr:min:sec
6:08:17	6:37:32	0:29:15
7:44:57	9:13:11	1:28:14
9:18:49	10:35:44	1:16:55
48:59:03	49:33:04	0:34:01
	Total	3:48:25

Postflight examination of the recorder showed that the magnetic tape had come off one of the guide rollers, altering its path in such a manner that it would move in the record direction, but would not move in the playback direction.

Further examination has revealed the probable cause of the failure and two lesser conditions which may have contributed to it. A metallic chip was found clinging to a gear tooth in the recorder reel-drive assembly. Bright spots were also found on the gear and indicate that the chip at some time could have been lodged in a position to have prevented motion between the mating gears. The function of the gear system, in conjunction with four negator-type springs, is to provide relative torque motion between the supply and the take-up reels as tape diameters change and to maintain tension in the magnetic tape. It was found during the examination of the recorder that the negator spring was prewound to 80 turns instead of the nominal 88 turns. This creates a marginal condition at the point where half the tape is on the supply reel and half on the take-up reel. Furthermore, the recorder was found to have a loose pinch belt. This belt must hold the magnetic tape against the drive capstan in order to impart the necessary constant linear speed to the tape.

Although the sequence of events cannot be established with certainty by postflight examination, they may logically have occurred as follows. The aforementioned RKV real-time data confirm the slowdown of tape motion at 6:35:52.6 g.e.t. This may have resulted from the chip being caught in the gear teeth and forced out 1.8 seconds later. At that time, the tape resumed normal record speed until 6:37:49.3 g.e.t.

when possibly the chip became caught firmly between the teeth and stopped all record motion. When playback was commanded during the pass over Hawaii, in the fifth revolution, the tape motion was reversed, and this could have dislodged the metallic chip from between the gear teeth. With the tape tension less than normal and the pinch belt too loose, the tape probably came off the vertical roller on the take-up reel side and stopped tape motion in the playback direction. The recorder could now move only in the record direction, with tape movement in the playback direction being prevented by the drag caused by the altered tape path. Plans for corrective action to prevent a recurrence of this problem are being formulated by the spacecraft contractor and the tape-recorder vendor. Additional evidence that less than normal tape tension may have contributed to the problem is found in a similar failure in the recorder for Spacecraft 10. With the negator spring prewound to less than the required 88 turns and commanding playback near the mid-tape position where tension may be lost, the tape would come off the guide roller as it did on the Spacecraft 9 recorder.

5.1.3.2 System performance.- During this mission, 258 parameters were monitored. At 26:58:00 g.e.t. the RSS hydrogen cryogenic-mass-quantity measurement, parameter CA09, failed. An inflight calibration of this parameter was obtained, verifying proper operation of the measurement circuitry from the adapter control box to the cabin indicator and into the PCM system. This means that the failure must have occurred in the transducer or its wiring into the control box and could not be verified by postflight examination because the transducer and control box were located in the adapter section.

The left-hand suit-inlet air-temperature measurement, parameter CC03, failed at approximately 69:13:00 g.e.t. Further discussion regarding this failure will be found in section 5.1.4.

5.1.3.3 Delayed-time data quality.- The delayed-time data received at the Mission Control Center at Cape Kennedy (MCC-C), Texas, and Hawaii ground stations are summarized in table 5.1.3-I. This table shows the results of the four computer-processed data dumps prior to the failure of the PCM tape recorder. For all ground stations listed, the usable data exceed 99.66 percent.

5.1.3.4 Real-time data quality.- The real-time data received by the Texas, Hawaii, MCC-C, Canary Island, Carnarvon, Bermuda, Grand Bahama Island, Coastal Sentry Quebec, and Antigua ground stations and the real-time data received by aircraft 490, 493, and 628 during reentry are summarized in table 5.1.3-II. For all the ground stations and aircraft, the usable data recovered exceeded 97.94 percent. All percentages were derived from computer-processed data edits.

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	Subframes	Percent	
Cape Kennedy	Launch/1, 2	2:59:05	107 448	295	0.274	99.726
Texas	3	1:19:56	47 963	373	0.777	99.223
Hawaii	3, 4	1:16:07	45 667	15	0.032	99.968
Summation		5:35:08	201 078	683	0.339	99.661

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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	
Texas	4, 15, 17, 29, 30, 31, 32, 43, 44	77:18	185 505	3 768	2.031	97.969
Hawaii	4, 5, 6, 17, 32, 33, 34, 35	72:34	174 160	3 788	2.175	97.825
Cape Kennedy	2/3, 42/43, 43/44, 44/45	34:00	81 611	776	0.950	99.050
Grand Canary Island	12, 13, 14, 15, 30	45:34	109 360	3 828	3.500	96.500
Carnarvon	12, 13, 29, 30, 31, 32, 33	53:15	127 810	2 024	1.583	98.417
Bermuda	14, 15, 44	26:24	63 356	416	0.656	99.344
Grand Bahama Island	13/14	8:34	20 577	24	0.116	99.884
Coastal Sentry Quebec	34	8:46	21 289	1 184	5.561	94.439
Antiqua	32/33	9:01	21 652	56	0.258	99.742
Aircraft 490	Reentry	5:29	13 151	264	2.007	97.993
Aircraft 493	Reentry	6:31	15 651	284	1.814	98.186
Aircraft 628	Reentry	7:14	17 376	1 072	6.169	93.831
Summation		354:40	851 498	17 484	2.053	97.947

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5.1.4 Environmental Control System

Performance of the Environmental Control System was satisfactory throughout the mission. All measured parameters varied within the expected ranges of values throughout the flight, except as discussed herein.

5.1.4.1 Cabin pressure decay.- The cabin pressure decayed from the normal control value of 5.1 psia at 66 hours 30 minutes g.e.t. to 4.68 psia at 67 hours 36 minutes g.e.t. The crew returned the cabin pressure to normal by using the cabin repressurization valve. The decay continued, and the repressurization valve was used a second time to raise the pressure back to normal. The water seal was closed at this time, thus deactivating the pressure relief valve, and was left closed for about 30 minutes and then opened. The cabin pressure remained normal for the remainder of the mission.

Postflight tests on the fully redundant cabin pressure regulator displayed normal performance of both sides of the regulator. Tests are now being conducted on the cabin pressure relief valve.

5.1.4.2 Left-hand suit-inlet temperature.- The left-hand suit-inlet temperature instrumentation failed at 69 hours 13 minutes g.e.t. This failure was similar to the failure which occurred during the Gemini VII mission when free moisture caused a short circuit in the temperature sensor. The failure occurred shortly after a sleep period when the crew had the system adjusted for minimum cooling, thus allowing a buildup of moisture in the suit circuit. A failure analysis is being performed on the temperature sensor.

5.1.4.3 Increased carbon dioxide indication.- The indicated carbon dioxide (CO_2) level increased approximately one mm Hg from 1.57 to 2.54 mm Hg as recorded by the Canary Island tracking station on the last orbit. The crew stated that the increase was not indicated by the cabin meter. The rate of increase in CO_2 partial pressure recorded is too rapid to be the result of any failure in the CO_2 absorber, but is attributed to the closing of the tape-recorder circuit breaker. Previous test history has shown that electromagnetic interference resulting from activation of the tape-dump cycle has caused the indicated level to increase momentarily.

5.1.4.4 Water inflow at landing.- The crew reported water entering the cabin at landing. Investigation has shown that this water entered through the cabin pressure relief valve during the initial pitch down of the spacecraft upon landing. The valve releases inward at a pressure of 15 inches of water. Prior to the mission, procedures on this flight had been changed to close the water seal during the post-landing checklist instead of at an altitude of 2000 feet as on previous flights. This change was incorporated to prevent the entrance of undesirable fumes into the cabin during landing and postlanding phases of flight. After the flight, the cabin was subjected to a pressure test, and the leakage rate was normal. The procedures will be investigated and changed if required.

5.1.4.5 Depletion of drinking water.- The crew reported that during the last orbit they were unable to withdraw drinking water from the water metering device. This indicates either failure of the water metering device, blockage in a water line, or depletion of the adapter drinking water supply. Inspection of the spacecraft upon return to St. Louis revealed that the cabin drinking-water storage tank was empty except for a small residual. Postflight data analysis and spacecraft component inspection provided the following information:

(a) The water metering device functioned normally.

(b) Drinking water available for consumption from the adapter storage tank was 33.75 pounds versus an indicated 26.16 pounds withdrawn through the water metering device.

(c) The cabin drinking-water storage tank was found to have a 32 and 72 cc/hr water leak at 15 psid and 20 psid, respectively. This test was made after the tank had been removed from the spacecraft.

(d) The cabin drinking-water storage-tank check valve was found to leak nitrogen at a rate of 500 scc/sec at 5 psid. Further investigation showed that the check valve was held open by aluminum oxide particles.

(e) Corroded aluminum flare-saver washers were found in the water system plumbing.

This investigation led to the conclusion that the failure to dispense drinking water during the final orbit of the flight was caused by the depletion of the adapter drinking water supply by leakage out of the cabin drinking-water storage tank and the associated loss of pressure to force water out of the cabin tank. The measured leak rate would result in a loss of approximately 5 pounds of water during the mission. The leak rate could have easily been greater with the tank installed in the spacecraft, accounting for the additional 2.5 pounds that were apparently lost.

The loss of water from the cabin drinking-water storage tank was a result of leakage through the check valve after adapter separation.

5.1.4.6 Extravehicular Life Support System.- The Extravehicular Life Support System (ELSS) performed according to specifications during the extravehicular portion of the mission. Adequate cooling was provided to the extravehicular crewman at all times except at ingress, when the pilot experienced heavy sweating. The ELSS was set for medium flow throughout the majority of the first day-side pass during EVA and the pilot stated that he was comfortable. High flow was first selected when the pilot experienced local heating on his back. The valve was left in this position during the remainder of EVA, primarily in an attempt to clear the visor. The pressure gage on the suit indicated 3.9 psia throughout the EVA period. Telemetry data showed 3.72 psia which is normal.

Two anomalies occurred during EVA. The visor began fogging just after sunset of the first day-side pass, and the heat exchanger dried out at some point during the extravehicular portion of the mission.

5.1.4.6.1 Visor fogging: Higher workloads than expected were evident throughout the EVA. The evaporator/condenser was designed for a nominal metabolic rate of 1400 Btu/hr and a maximum of 2000 Btu/hr for periods of short duration. Medical data indicate that these rates may have been exceeded, which, in effect, would overpower the capabilities of the evaporator/condenser. Preflight and postflight correlation of pulse-rate energy expenditure indicates that rates of 2000 Btu/hr and higher may have been prevalent throughout the majority of the EVA and that rates were in excess of 3200 Btu/hr during ingress. Cooling capability was adequate, even on medium flow, but the evaporator/condenser could not keep up with the thermal load and prevent fogging. Furthermore, fogging was probably induced by the high respiration rates of 30 to 40 breaths per minute observed during the EVA. This breath rate would humidify 55 to 75 percent of the total oxygen flow to the helmet to near saturation, sufficiently raising the dew point around the visor to allow fogging at normal visor operating temperatures and also inhibit clearing of the visor.

5.1.4.6.2 Evaporator/condenser dryout: The ELSS evaporator/condenser contained 0.596 pound of water at lift-off. The amount of water remaining after flight was determined to be 0.246 pound. The pilot stated that during ingress he became uncomfortably warm. From this fact, and from the amount of water remaining, it is evident that the evaporator/condenser performance was degraded due to dryout sometime during the mission, presumably at or near the time of ingress. Dryout was a result of a higher-than-anticipated metabolic load which, in turn, caused the water capacity to be less than required.

5.1.4.6.3 Postflight investigation: An altitude chamber test with Gemini IX-A flight hardware will be performed to investigate the cause of visor fogging. Work rates consistent with those experienced during the mission will be simulated, and the evaporator/condenser will have the same amount of water.

5.1.5 Guidance and Control System

5.1.5.1 Summary.- The Guidance and Control System performed satisfactorily throughout the mission. Inertial Guidance System (IGS) ascent guidance was as expected with no inflight launch-azimuth updates. Rendezvous guidance was satisfactory, although a start-compute-discrete anomaly (see section 5.1.5.4) required a procedural corrective measure in order to compute the terminal-phase-initiate parameter for the first rendezvous. Reentry guidance and navigation were excellent, as indicated by the small miss distance. Radar performance was normal throughout the mission, considering the signal-strength variations caused by the tumbling target. The Auxiliary Tape Memory Unit (ATMU) was utilized for the second time and performed properly. Table 5.1.5-1 contains a summary of significant guidance and control events.

5.1.5.2 Inertial Guidance System performance evaluation.-

5.1.5.2.1 Ascent phase: The IGS roll, pitch, and yaw steering commands are presented in figure 5.1.5-1. Superimposed on the IGS steering quantities are the steering signals from the primary Radio Guidance System (RGS) along with the upper and lower IGS attitude-error limit lines for a nominal Gemini IX-A mission. The IGS values were within the preflight predicted zero-wind limits with differences between the two systems, other than those noted below, attributable to known programmer and timing differences, initial engine misalignments, and drift in the primary guidance Three-Axis Reference System (TARS). Analog time histories of predicted pitch and yaw attitude errors for winds at T-3 hours are shown for the first 90 seconds of flight. These caused pitch excursions beyond the lower zero-wind limit early in flight.

The IGS pitch attitude error represented a normal response to the primary pitch steering commands until about lift-off (LO) + 335 seconds, when the pitch limit was exceeded. This condition was similar to that experienced during the Gemini VIII launch and was caused by primary system errors.

The T-3 minute update for launch azimuth was not successful; therefore, the ground-computed velocity updates at LO + 100 and LO + 140 seconds, although correct, were inhibited. These inflight updates are used by the Gemini Inertial Guidance System (IGS) to correct for the minor launch vehicle and spacecraft installation misalignments of the inertial platform. Because of the missing updates, the out-of-plane velocity component (nominally zero at SECO) was erroneously indicated by the IGS to be -159 ft/sec at SECO. During Stage II guidance, the IGS yaw-steering signals exceeded the predicted limit after

LO + 225 seconds and increased to the limiting command of 6 degrees at LO + 334 seconds.

If a guidance switchover had occurred early in second stage flight, the vehicle would have achieved an insertion vector with a flight-path angle within ±0.01 degree of nominal, an in-plane velocity error of approximately -10 ft/sec, and an out-of-plane velocity of approximately +160 ft/sec.

The resulting apogee and perigee would have been closer to nominal than that achieved by the primary system; however, considerable spacecraft propellant would have been required to correct the out-of-plane velocity in order to achieve a rendezvous with the target vehicle.

SECO and spacecraft separation were normal, as indicated by the rates and accelerations shown in figure 5.1.5-2. An increase in acceleration prior to separation, noted on the Gemini VIII mission, was not present on this flight.

The inertial velocity (V) and velocity to be gained at perigee to correct apogee $\left(V_{ga}\right)$ as calculated in the Insertion Velocity Adjust Routine (IVAR) by the computer were not affected by the large out-of-plane component and were, therefore, used to correct toward the desired orbit instead of the Incremental Velocity Indicator (IVI) display.

These quantities were read via the Manual Data Readout Unit (MDRU), and thrust was applied until nominal values were obtained. The crew also noted from the negative radial velocity readout of the MDRU prior to separation that the flight-path angle achieved by the launch vehicle was low. Therefore, a zero pitch angle as indicated on the Flight Director Indicator (FDI) was held during the correction. Because the Inertial Measurement Unit (IMU) was referenced to launch-pad inertial coordinates (at lift-off), the actual pitch angle above the local horizontal varied from 12 to 18 degrees during the three post-separation thrust periods and resulted in a vertical component being applied. Figure 5.1.5-3 shows a time history of attitude and velocity during this period. Although the IVI and FDI IVAR commands were not followed explicitly, sufficient information was available to the crew from the IGS to allow them to obtain a near-nominal orbit. The corrections that were applied corrected the post-SECO preseparation orbit of 85.6 by 127.8 nautical miles to one of 85.7 by 144.0 nautical miles. If the IVAR commands had been followed explicitly, the apogee achieved would have been 142.7 nautical miles. The velocity to be gained at apogee to correct perigee would have been 2 ft/sec and, if applied, would have raised perigee by 1.1 nautical miles to the planned 86.7 nautical miles.

The IVI display, as actually computed by the onboard IVAR, was reconstructed using the IGS navigational and gimbal-angle data with the results presented in figure 5.1.5-4. The crew-reported readings of 60 ft/sec forward occurred about three seconds after SECO. The next crew-reported reading of 17 ft/sec forward, 26 ft/sec left, and 153 ft/sec down appears to have been read at about LO + 372 seconds, or about three seconds following the initiation of the roll maneuver. The components of the large out-of-plane velocity were displayed on all three windows as the yaw-gimbal angle changed; therefore, because of this large relative magnitude with respect to the in-plane velocity, the IVI displays changed more as a function of attitude changes than a function of thruster activity. 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. The crew readings and the comparisons shown in the table verify that the orbit insertion equations and the computer/IVI interface were operating properly.

The velocity residuals obtained with GE MOD III final data were used to determine a set of IMU component errors which would induce velocity error propagations as shown in figure 5.1.5-5.

The ramp-like trend in the X-axis velocity residuals from lift-off and the step change observed at first stage engine cutoff indicate accelerometer-scale-factor and timing errors. A timing error between IGS and tracker, a scale factor error in the IGS timing, and a shift in the accelerometer scale factor were used to fit the velocity error trend along the X-axis. A history of the preflight IMU component stability (fig. 5.1.1-6) indicated X-accelerometer shifts of approximately the same magnitude as those estimated from flight data. The Z-velocity residuals represent the resulting crossrange velocity error assuming the actual launch-azimuth updates generated by GE/Burroughs had been transmitted, received, and properly used by the spacecraft computer. A set of errors which can propagate like the Z-velocity and Y-velocity residuals is shown in table 5.1.5-III. The inertial component and tracking errors obtained from a preliminary error coefficient recovery program (ECRP) are also presented in the table.

The IGS position and velocity errors at SECO + 20 seconds are presented in table 5.1.5-IV. The quantities were obtained from calculations using the best estimated trajectory. An estimate of injection parameters at SECO + 20 seconds, determined from the IGS and other sources, is given in table 5.1.5-V. The performance was well within the expected tolerances and design limits of the Inertial Guidance System.

5.1.5.2.2 Orbital phase: A summary of the translation maneuvers performed is shown in table 5.1.5-VI. Although telemetry data are not available to analyze all the maneuvers, there are indications that the capability for precise performance was available. The platform-accelerometer bias was checked periodically and updated twice during the mission. The errors in bias noted before any updates, however, were not sufficient to cause significant or even noticeable errors.

The IMU was aligned frequently during the mission with no reported difficulty. A representative sample of alignment accuracies, prior to major maneuvers for which data are available, is included as table 5.1.5-VII.

The rendezvous radar was turned on to standby at 2:10:58 g.e.t. and switched to the search mode at 2:13:01 g.e.t. Normal radar lock-on occurred approximately three seconds later at a range of 130 nautical miles (792 110 feet). Figures 5.1.5-7 and 5.1.5-8 contain a time history of radar parameters during the first and third rendezvous.

Maximum lock-on range occurred during revolution 34 (54 hours 10 minutes g.e.t.) when the radar tracked the Augmented Target Docking Adapter (ATDA) to approximately 187 nautical miles. This maximum range is considered normal because of the tumbling target. The radar signal strength variations noted throughout both rendezvous operations were caused by the tumbling ATDA and normal antenna switching. These variations were expected because of the continuously changing antenna look angle of the tumbling ATDA. Because of the fluctuating signal strength, the normal single crossover from the narrow-band to the wide-band receiver amplifiers did not occur. The signal strength fluctuations were large enough in some cases to break radar-lock, especially during initial radar tracking. The radar lock-on dropouts were, for the most part, two seconds or less in duration.

The signal strength fluctuations also caused numerous range-rate transients. Because range rate is obtained by differentiating analog range, any decrease in the voltage which represents analog range, caused by missing reply pulses, appeared as impulses in the range-rate measurements. Approximately 30 successive missing pulses are required, however, before the radar-lock indicator shows loss of radar lock. This condition was most prevalent before analog-range acquisition (approximately 300 000 feet) but persisted throughout radar tracking.

The tumbling ATDA target also caused errors in the radar angle measurements because of the ellipticity of the reply signal and angle noise associated with low signal strength. Figure 5.1.5-9 contains a representative comparison of changes in gimbal angles with changes in radar angles at various ranges. The spacecraft-to-ATDA L-band command link functioned normally when used to turn off the acquisition lights on the ATDA. Radar environmental parameters were nominal throughout the mission.

The first maneuver for the M=3 rendezvous was initiated from a position below and behind the ATDA. Under normal circumstances the computer would have been placed in the rendezvous mode shortly after the coelliptic maneuver N_{SR} and left in that mode until the start of the braking sequence. However, due to the start-compute-discrete anomaly which was present during this period, the computer would compute only one value of total-velocity-to-rendezvous ΔV_T before apparently sensing the start-compute-discrete in the ON state and entering the calculation and command sequence for terminal phase initiate (TPI).

To overcome this problem, the crew reinitialized the rendezvous mode each time a value of ΔV_T was computed. The final reinitialization was timed to produce a ΔV_T at the nominal time for TPI. As a result, only four values of ΔV_T were computed prior to the M=3 rendezvous

Postflight simulations using both the Real Time Computer Complex (RTCC) and best-estimated-trajectory state vectors for the period between N_{SR} and TPI were used to produce ΔV_T time histories for the period between 2 hours 40 minutes g.e.t. and 3 hours 42 minutes g.e.t. These ΔV_T curves, along with the radar-angle and gimbal-angle time histories, are shown in figure 5.1.5-10. As shown in this figure, good agreement exists between the two ΔV_T curves and the values of ΔV_T computed in real time onboard the spacecraft.

Table 5.1.5-VIII shows the calculated solutions for the maneuvers and the maneuvers actually applied during the first rendezvous sequence. Although the crew followed the onboard-computer solutions for the two midcourse corrections, they executed a TPI maneuver that was based more on their backup calculations and the the ground-computed values for the vertical component. The crew was concerned about the 8-ft/sec up component in the computer solution.

In order to evaluate the effect of these differences in velocity components, three simulations of the first rendezvous were run using the state vectors from the best estimated trajectory. The following programs of thrusts were used in these runs:

(a) Run 1: Used the TPI maneuver and midcourse corrections actually applied in flight, as recorded from the spacecraft inertial platform.

(b) Run 2: Used the onboard-computer solution for the TPI maneuver, and the simulator solutions for the two midcourse corrections.

(c) Run 3: Used the TPI maneuver and midcourse corrections computed by the simulator.

The thrust histories of these three simulations are shown in table 5.1.5-IX, and the resulting relative trajectories are shown in figure 5.1.5-11.

Although the state vectors used contained errors which show up in the miss distance, the data in table 5.1.5-IX show that, while radar errors caused a larger vertical correction than necessary (8 ft/sec), the total vertical correction using the onboard computer solution was 15 [8+11 - (3+1)] ft/sec (composed of the TPI and first midcourse correction adjusted for state-vector errors). This was 8 ft/sec more than the 7(3+2+2) ft/sec actually applied in the up-down direction. If equivalent-accuracy midcourse corrections are assumed, 4 ft/sec should be added to the onboard-computer solution, giving 19 ft/sec total, or a difference of 12 ft/sec. Variations in the braking maneuver for different braking schedules would tend to reduce the fuel penalty caused by this difference.

An explanation of the values produced by the computer at TPI can be found in the time histories of the pitch gimbal angle and the radar elevation angle for the period immediately preceding TPI. These data are shown in table 5.1.5-X. The radar samples in this table were taken at 100-second intervals beginning at 3:20:26 g.e.t. Examination of the last sample taken shows that, although the spacecraft pitch angle increased 1.5 degrees, the elevation angle decreased 1.39 degrees. This means that the look angle increased 2.89 degrees over the last 100 seconds. Because it is unlikely that this could happen in flight, it is reasonable to assume that the elevation angle read in the final radar sample prior to TPI was in error.

In the computer logic of the rendezvous mode, the smoothed relative velocity of the spacecraft to the target vehicle is determined by sequentially pairing the last radar sample with earlier samples to derive values of relative velocity and then averaging the values so obtained. For TPI of the first (M=3) rendezvous, the eighth or last radar sample was paired with each of the first four samples. The final radar sample had an elevation angle that was too small. This caused the value of vertical closing rate calculated by the computer to be too low. To overcome this apparent low vertical closing rate and place the spacecraft on an intercept trajectory, the computer indicated an up component.

Table 5.1.5-XI shows the vector components for the TPI and terminal phase finalize (TPF) for two consecutive points on the best-estimated-trajectory ΔV_T curve, all taken from the region around the flight TPI. The data show, at least in this particular case, that changes in TPI vector components (especially in $\Delta\dot{Y}$, the vertical component) may be compensated by changes in the corresponding TPF components with almost no change in the total velocity increment.

Figure 5.1.5-12 shows a time history of radar range over the final few minutes of the M=3 rendezvous trajectory. An extrapolation of the data recorded up to the braking sequence shows that the TPI maneuver and midcourse corrections applied by the crew put the spacecraft on a trajectory which would have brought it to a point less than 1100 feet from the ATDA.

During the second or equi-period rendezvous, neither the radar nor the computer was used. This rendezvous was accomplished using the attitude indicator, a miniature handheld sextant, and onboard charts. The onboard charts were prepared prior to the mission to utilize data from the attitude indicator and sextant and to provide solutions for the rendezvous maneuvers. No telemetry data were available from the platform attitude indicators and accelerometers due to the PCM tape recorder failure and no computer rendezvous solutions were used; therefore, an analysis of the equi-period rendezvous maneuvers could not be conducted.

The third rendezvous (rendezvous from above) was a simulation of an Apollo Lunar Module abort and was initiated from a position above and ahead of the target. Because the onboard telemetry recorder was not functioning during this rendezvous from above and because TPI was the only maneuver performed over a ground station, only limited analysis was possible.

The calculated solutions for the maneuvers and the maneuvers actually applied during the rendezvous from above are shown in table 5.1.5-XII. As in the first rendezvous, the crew chose to use their backup solution for TPI, but followed the computer for the midcourse corrections.

Figure 5.1.5-13 shows time histories of available radar, gimbal angle, and ΔV_T data as well as a ΔV_T curve generated by a postflight simulation using RTCC state vectors. The difference between the values of ΔV_T obtained in flight and those obtained from the simulation indicated that the RTCC state vectors contained relatively large errors.

Simulations of the third rendezvous were made from the RTCC state vectors in the same manner as for the first rendezvous. The resulting relative trajectories are shown in figure 5.1.5-14, and the thrust histories are shown in table 5.1.5-XIII. Because no telemetry data were available for either of the two midcourse corrections, both the magnitude and the direction of the thrust vectors applied had to be estimated from data recorded in the crew's flight log.

The inaccuracies in these estimates, as well as in the state vectors, are shown by the large miss distance in the run which used the flight values for all three maneuvers (run 1). The large first midcourse correction in run 2, which used TPI from the onboard-computer solution, shows that the solution was not correct for the problem defined by the state vectors used. Using table 5.1.5-XIII and figure 5.1.5-14, it can be concluded that the left-right onboard-computer solutions for the midcourse corrections (table 5.1.5-XII) probably could have been avoided by applying the onboard-computer TPI right-left solution of 2 ft/sec to the left. Then the velocity requirement for these maneuvers would have been reduced about 6 ft/sec, assuming 4 ft/sec midcourse-correction errors. The trajectory plots in figure 5.1.5-14 show that lateral-displacement errors can be corrected on the first or second crossing of the zero-displacement error line; however, it can also be seen that the second crossing is more sensitive to errors in the midcourse corrections. At TPI, the computer solution called for a down component instead of the up component actually applied and a larger forward component than was used. These maneuvers would have produced a steeper and faster transfer and would have reduced the final midcourse correction from 10 ft/sec to probably less than 2 ft/sec in the downward direction. This 8 ft/sec reduction of the vertical velocity requirement added to the 4 ft/sec reduction of right-left requirement would have resulted in a total saving of 12 ft/sec. This analysis shows that velocity differences of these magnitudes at TPI cause only minor changes in propellant consumption.

The performance of the Auxiliary Tape Memory (ATMU) was nominal in every respect. The automatic mode, Reprogram/Verify, was used to transfer and verify the reentry program (Module IV) stored on the A section of the tape. The reprograming was initiated over Hawaii during revolution 35. The crew also verified the computer's previously stored reentry program against the redundantly stored reentry Module IVB.

In view of the start-compute discrete problem, a special procedure was developed and transmitted to the crew which would have allowed loading and use of the reentry program if the abnormal condition recurred. This procedure involved zeroing certain locations in the memory used and normally zeroed by the reentry program. Had the problem occurred during computer power-up, the crew was instructed to use the ATMU manual mode to reprogram Module IA prior to reprograming or using the reentry program. Had the problem recurred prior to retrofire, the crew was instructed to switch from the reentry mode to the spare position on the computer-mode switch and to switch back into reentry one second prior to retrofire to initiate reentry navigation.

Prior to retrofire, the crew verified the stored reentry program once again against ATMU Module IVA.

For the first time during a Gemini mission countdown, the ATMU was used to verify the internal memory of the computer during both precount and final count. This eliminated the requirement for the lengthy (relative to using the ATMU) memory verify with ground equipment, which had been standard practice for all countdowns prior to Gemini IX-A.

5.1.5.2.3 Reentry phase: The IGS operated properly throughout the retrofire and reentry phases of flight, and the spacecraft was sighted on the main parachute by personnel onboard the recovery ship. The performance of the retrorockets was 1.06 percent higher than nominal, causing the footprint to shift 46.2 nautical miles up range.

From retrofire to an altitude of 400K feet, a 10-degree bank angle was flown as planned. At 72:06:27.840 g.e.t., the computer indicated the 400K-foot level by commanding the 0-degree bank angle. This time compared within one second of the value computed on the ground when using tracking data acquired after retrofire. No IGS data are available for the period between 72:08:24.984 g.e.t. and 72:13:53.915 g.e.t. because of the communications blackout. At 72:14:53.540 g.e.t., the computer properly terminated guidance at a density altitude factor of 4.66083, which indicates proper functioning of the reentry mode at this time. Table 5.1.5-XIV shows the actual telemetry data of various IGS parameters at 400K feet and at guidance termination.

Figure 5.1.5-15 shows a comparison after blackout between position from tracking and position from IGS. The radar data in comparison with the available IGS data seem to indicate the possibility of a small up-date initial-condition error of approximately 0.96 nautical miles propagating throughout the trajectory.

The difference between IGS navigation and tracking position was approximately 2.2 nautical miles at guidance termination. This navigation error was well within the variation expected because of initial condition uncertainty and normal IMU misalignment and component errors. The guidance error at this time was 3.03 nautical miles. Between guidance termination at approximately 80K feet and drogue deploy at 40K feet, the spacecraft translated about 5 nautical miles downrange toward the planned landing point. The position of the spacecraft at drogue parachute deployment was 0.38 nautical mile west of the planned landing point measured by radar-corrected IGS data. The landing point as reported by the recovery forces was 2.3 nautical miles west of the planned landing point. Figure 5.1.5-16 shows these relative positions of the spacecraft with respect to the planned target.

A manual closed-loop reentry was flown by the crew based on the roll commands generated by the onboard computer. For the Spacecraft 9 reentry, a four-degree-of-freedom reentry program reconstruction with the actual IVI quantities and the Woomera retrofire conditions revealed that the automatic reentry would have had a miss distance of 3.80 nautical miles at guidance termination which would have resulted in a 1.04 nautical mile overshoot at 50K feet.

5.1.5.3 <u>Control system performance evaluation</u>.-

5.1.5.3.1 Attitude Control and Maneuver Electronics (ACME): Performance of the control system was nominal throughout the mission. During EVA preparations, the scanner heater circuit breaker was inadvertently turned off. This circuit breaker applies power to a relay which controls the phase of pitch and roll feedback into the ACME when in platform control mode. As a result, spurious thruster firing commands were generated, causing the spacecraft to move off the null position. The crew immediately switched to DIRECT and regained control of the spacecraft. The trouble was quickly diagnosed by ground personnel, and the analysis was transmitted to the crew. After resetting the circuit breaker, the system returned to normal.

The extravehicular activities (EVA) operation was examined in detail to determine the response of the control system to the disturbances introduced by the pilot moving about the spacecraft. The platform control mode was the primary mode used throughout this period. Pulse mode was utilized periodically to return the spacecraft to the null position. The use of pulse mode resulted from the system having been turned off to allow the pilot to maneuver in the vicinity of the thrusters. The maximum attitude excursions noted in the data available while in the platform control mode were approximately 2.80 degrees. Maximum angular accelerations noted while control power was off were 1.2 deg/sec^2 in pitch and yaw and 3.6 deg/sec^2 in roll. These represent approximately 25 percent of the available control authority in pitch and yaw and 57 percent in roll. Figure 5.1.5-17 shows platform mode activity, EVA disturbance effects, and pulse mode operation during EVA.

Reentry data became available at 71:51:13.4 g.e.t., 4 minutes 29 seconds after retrofire. The control system was in pulse mode using Reentry Control System (RCS) A-ring from that time until communications blackout. Coming out of blackout, the control system was in reentry rate-command mode and remained there through drogue parachute deployment, at 72:15:47.8 g.e.t. At 72:15:23.0 g.e.t. the RCS B-ring was turned on and used in parallel with the A-ring through drogue parachute deployment. The maximum rates observed prior to drogue parachute deployment (in the data available) were 6 deg/sec in both pitch and yaw. Figure 5.1.5-18 contains a time history of control parameters during reentry.

5.1.5.3.2 Horizon sensors: The horizon sensors performed nominally throughout the mission. The new model secondary sensor unit with narrower bandwidth was proved on this mission, by crew reports, to be less sensitive to sun interference at sunrise and sunset, although occasional losses of track were experienced. The crew also reported normal automatic alignment performance and no difference between the primary and secondary units, although the secondary unit appeared to provide tighter track. The narrow-band scanner was used to align for reentry, and preliminary data indicate that the navigation errors at guidance termination, taking into account update errors, were smaller than on previous flights.

5.1.5.4 Start-compute-discrete anomaly.- During the period from 2:21:13 g.e.t. to 4:00:44 g.e.t., an anomaly appeared in the operation of the start-compute discrete. This is the discrete which normally indicates to the onboard computer that the START button has been depressed. In the time period noted, telemetry data from the onboard computer shows 14 instances in which the computer sensed the start compute discrete in the on state in which the crew reported that the START button had not been depressed. A time history of significant events during this period is presented in table 5.1.5-XV.

The state of the start-compute discrete in the catch-up mode can be determined from computer telemetry words by the examination of the attitude commands, the computed IVI displays, and the logic time (Tx). From the time the computer is placed in the catch-up mode until the computer senses that the start-compute discrete is on, attitude commands are computed and Tx = 0. When the computer senses that the start-compute discrete is in the ON state, the computation of attitude commands is set to 100 seconds. If the start-compute discrete is on when the computer tests it for the first time after switching into the catch-up mode, the attitude commands will not be computed and will remain at the initialized values of zero.

In the rendezvous mode, the start-compute discrete is not tested for approximately the first 715 seconds of operation. Therefore, the state of the discrete remains unknown during this period. After this time, however, the computer begins testing the start-compute discrete and will calculate and display TPI maneuver commands as soon as the discrete is sensed in the ON state. The start-compute discrete is tested in the prelaunch mode only in the computer power-up sequence. Therefore, the state of the discrete is unknown when the computer is in this mode.

The FDI problem reported by the crew was a result of the start-compute discrete anomaly rather than a separate problem. When the crew put the computer in the catch-up mode at 2:21:13 g.e.t., the start-compute discrete was on the first time it was tested. This meant that the attitude commands were not set to their proper values but were left at the zero values to which they had been set as a part of the catch-up mode initialization logic.

The crew knew the approximate attitude (pitch down 41 degrees, yaw left 2 degrees) to which the spacecraft should be commanded for the N_{SR} maneuver and began to orient the spacecraft to this attitude. However, because of the start-compute discrete anomaly, the computer was commanding the Flight Director Indicator to display an error for other than zero degrees in pitch and yaw. Therefore, as the spacecraft pitched down, the pitch FDI moved upward, hitting the limit as the pitch angle passed 340 degrees. Continued pitching did nothing to improve the problem. The computer was then cycled to the rendezvous mode and back to the catch-up mode. This time, however, as the data in table 5.1.5-XV show, the start-compute discrete was off the first time it was tested, and the apparent pitch FDI problem did not appear.

In the onboard computer, the start-compute discrete is the output of a latch. This latch is set by the output of a relay triggered by the START button. The latch is reset by a discrete output from the computer. Once the latch has been set, it should remain set until it is reset by the computer.

From the onboard-computer telemetry words and a knowledge of the operation of the catch-up and rendezvous modes, the following conclusions about the operation of the start computer latch can be drawn:

(a) The latch was not on continuously during the period of the anomaly. At 2:22:57 g.e.t. and again at 2:24:06 g.e.t. the latch was OFF when the computer tested it.

(b) Once the latch was set, it remained set until the computer reset it. If the latch was on the first time it was tested in the catch-up mode, the attitude commands were never computed, showing that the latch did not go off.

(c) The computer was able to reset the latch. If the latch had not been reset, it would not have been off at 2:22:59 g.e.t. and 2:24:06 g.e.t.

These data and conclusions tend to support the supposition that the start-compute discrete anomaly was caused by an intermittent input to the start-compute latch. Investigation of this anomaly is continuing, and testing of the components involved is currently underway.

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	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	13:39:33.335 G.m.t.
18.48	18.42	Start roll program	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	
20.48	20.42	Stop roll program	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	
23.04	22.97	Start pitch program 1	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	
88.32	88.06	Stop pitch program 1 Start pitch program 2	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	
104.96	104.64	No. 1 gain change	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	
119.04	118.66	Stop pitch program 2 Start pitch program 3	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	
151.65	152.49	BECO	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	
162.56	161.44	Stop pitch program 3	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	

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

Time from lift-off, sec		Event	Component status					Remarks
Planned	Actual RGS		ACME	Computer	IMU	Horizon sensor	Radar	
169.00	168.19	First guidance command	Rate command (OAMS control power - off)	Ascent	Free	Primary	Off	
338.99	339.79	SECO	Rate command, then direct	IGS backup	Ascent	Free	Off	
369.29	366.72	Spacecraft separation	Direct, then rate command	IGS backup	Ascent	Free	Off	ΔV = 29.6 ft/sec
Ground elapsed time hr:min:sec								
Planned	Actual							
0:06:09	0:06:06	OAMS initiate for spacecraft separation	Direct, then rate command	Ascent	Free	Primary	Off	ΔV = 29.6 ft/sec
0:06:39	0:07:16	OAMS off	Rate command	Ascent	Free	Primary	Off	
	0:10:25	Horizon sensor check	Platform	Prelaunch	SEF	Secondary	Off	
0:49:03	0:49:03	Phase-adjust maneuver	Platform	Catch-up	Orbit rate	Secondary	Off	ΔV = 74.8 ft/sec
	0:51:21	Accelerometer bias update	Platform	Catch-up	Orbit rate	Secondary	Off	
1:55:17	1:55:17	Corrective-combination maneuver	Rate command	Catch-up	Orbit rate	Secondary	Off	ΔV = 14.8 ft/sec
	2:13:03	Radar lock-on	Pulse	Rendezvous	Orbit rate	Secondary	On	
2:24:51	2:24:52	Coelliptic maneuver	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 53.4 ft/sec

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

Ground elapsed time, hr:min:sec		Event	Component status					Remarks
Planned	Actual		ACME	Computer	IMU	Horizon sensor	Radar	
	[a]2:50:31	Computer anomaly	Platform[a]	Catch-up[a]	Orbit rate[a]	Secondary	On	
	[a]3:14:47	Rendezvous mode	Pulse	Rendezvous	Orbit rate	Secondary	On	
03:35:35	3:36:02	Terminal phase maneuver	Rate command	Rendezvous	Orbit rate	Secondary	On	ΔV = 28.4 ft/sec
	3:48:35	First midcourse correction	Rate command	Rendezvous	Orbit rate	Secondary	On	ΔV = 4.1 ft/sec
	4:00:27	Second midcourse correction	Rate command	Rendezvous	Orbit rate	Secondary	On	ΔV = 4.0 ft/sec
	4:06:08	Terminal phase final	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 24.1 ft/sec
	4:15:00 (approx)	Station keeping	Pulse, Horscan, rate command	Catch-up	Orbit rate	Secondary	On	
05:01:00	5:01:00	Radial separation maneuver	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 19.9 ft/sec
	6:15:12	Terminal phase initiate	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 2.1 ft/sec
	6:20:20	First midcourse correction	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 1.1 ft/sec
	6:26:59	Intermediate correction	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 1.0 ft/sec
	6:28:34	Second midcourse correction	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 3.0 ft/sec
	6:29:26	Terminal phase final	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 16.3 ft/sec

[a]Data were obtained from crew reports or air-to-ground communications.

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

Ground elapsed time hr:min:sec		Event	Component status					Remarks
Planned	Actual		ACME	Computer	IMU	Horizon sensor	Radar	
7:14:58	[a]7:14:58	Second separation maneuver	Platform	Catch-up	BEF	(b)	On	Used engines 9 and 10 BEF, Retrograde
	[a]7:38:05	Power down	(b)	(b)	(b)	(b)	(b)	
	[a]17:45:00	Power up	(b)	(b)	(b)	(b)	(b)	
18:23:19	[a]18:23:19	Phase-adjust maneuver	(b)	(b)	(b)	(b)	(b)	
19:08:16	[a]19:08:16	Height-adjust maneuver	(b)	(b)	(b)	(b)	(b)	
	[a]19:32:13	Radar lock-on	(b)	(b)	(b)	(b)	On	
19:54:24	[a]19:54:24	Coelliptic maneuver	(b)	(b)	(b)	(b)	On	
20:55:28	21:02:28	Terminal phase initiate	Rate command	Rendezvous	Orbit rate	Primary	On	ΔV = 18.0 ft/sec
	(b)	First midcourse correction	(b)	(b)	(b)	(b)	On	
	(b)	Second midcourse correction	(b)	(b)	(b)	(b)	On	
	(b)	Termianal phase final	(b)	(b)	(b)	(b)	On	
	21:42:00 approx.	Station keeping	Pulse, direct, rate command, Horscan	(b)	(b)	Primary	Off at 21:44:56 g.e.t.	
22:59:00	[a]22:59:00	Third separation maneuver	(b)	(b)	(b)	(b)	Off	

[a]Data were obtained from crew reports or air-to-ground communications.

[b]Not available.

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

Ground elapsed time, hr:min:sec		Event	Component status					Remarks
Planned	Actual		ACME	Computer	IMU	Horizon sensor	Radar	
	[a]23:56:15	Accelerometer bias check	Platform	Catch-up	Orbit rate	(b)	Off	Update sent at 27:12:43
	[a]31:30:00 (approx)	Power down	(b)	(b)	(b)	(b)	(b)	
	[a]45:00:00 (approx)	Power up	(b)	(b)	(b)	(b)	(b)	
45:30:00	--	Extravehicular activities preparation	(b)	Off[a]	(b)	(b)	(b)	
	[a]47:10:40	Scanner heater circuit breaker problem	Platform[a]	Off[a]	SEF[a]	(b)	(b)	
	[a]49:21:55	Open hatch	Platform	Off	Orbit rate	Primary	(b)	
53:41:35	[a]53:41:35	True anomaly adjust maneuver	(b)	(b)	(b)	(b)	(b)	
55:46:37	55:50:41	Auxiliary tape memory loaded	Horscan	Prelaunch	(b)	Secondary	(b)	
	[a]60:20:00 (approx)	Power down	(b)	(b)	(b)	(b)	(b)	
	[a]66:47:00	Power up	(b)	(b)	(b)	(b)	(b)	
	[a]70:29:15	Reentry update	(b)	(b)	(b)	(b)	(b)	
	[a]70:31:51	Arm reentry control system	(b)	(b)	(b)	(b)	(b)	

[a]Data were obtained from crew reports or air-to-ground communications.

[b]Not available.

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

Ground elapsed time, hr:min:sec		Event	Component status					Remarks
Planned	Actual		ACME	Computer	IMU	Horizon sensor	Radar	
71:46:44	71:46:43.7	Retrofire	Rate command	Reentry	Free	Secondary	Off	ΔV = 321.3 ft/sec 2-ring reentry control system
72:06:21	72:06:27	400K feet	Pulse	Reentry	Free	Off	Off	
72:08:45	72:08:28	Begin blackout	Pulse	Reentry	Free	Off	Off	
72:13:51	72:13:54	End blackout	Reentry rate command	Reentry	Free	Off	Off	
72:15:27	72:15:47.8	Drogue parachute deployment	Reentry rate command	Reentry	Free	Off	Off	

[a]Data were obtained from crew reports of air-to-ground communications.

[b]Not available.

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

	Actual	Reconstructed
Velocity to be applied at apogee, V_{gp}, ft/sec	1.730	1.731
Velocity to be applied at perigee, V_{ga}, ft/sec	-0.895	-0.869
Radial velocity, V_p, ft/sec	-0.855	-1.125
Inertial velocity, V, ft/sec	25 749	25 749
IVI fore-aft, window, $\Delta V_{X_{S/C}}$, ft/sec	8.046	8.764
IVI right-left, window, $\Delta V_{Y_{S/C}}$, ft/sec	-158.427	-158.414
IVI up-down, window, $\Delta V_{Z_{S/C}}$, ft/sec	2.543	2.836
Time to apogee, T_{AP}, sec	3 203.6	3 203.8

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		-0.01	0	N	+0.1				
Y_p-gyro		-0.01	N	-0.3	0				
Z_p-gyro		-0.08	N	0	-1.1				
g-sensitive drift	0.5 deg/hr/g	deg/hr/g				deg/hr/g			
X_p-gyro spin-axis unbalance		0.014	0	0	-0.1		0	0	0
Y_p-gyro spin-axis unbalance		-0.38	0	-0.6	0		0	0	0
Z_p-gyro spin-axis unbalance		-0.09	0	0	+0.7	-0.015 ± 0.35	0	0	1.1
X_p-gyro input-axis unbalance		-0.37	0	N	+4.5		0	0	0
Y_p-gyro input-axis unbalance		0.4	+1.4	+10.9	0	0.29 ± 0.31	0.8	8.0	0
Z_p-gyro input-axis unbalance		-0.01	N	0	-0.3	0.076 ± 0.35	0	0	1.4
Accelerometer bias	300 ppm	ppm				ppm			
X_p		50	+0.2	-1.2	0				
Y_p		52	0	0	-0.8				
Z_p		-33	0	-0.4	0				
Accelerometer scale factor	360 ppm	ppm				ppm			
X_p		300	+7.4	0	0	313 ± 31	+7.7	0	0
Y_p		-41	0	0	N		0	0	0
Z_p		-100	0	+0.7	0	-208 ± 174	0	+1.4	0

N = negligible

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

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
Misalignments									
Azimuth misalignment	60 arc sec	48 arc sec	N	0	+5.7	48 ± 5	0	0	+5.7
Pitch misalignment	100 arc sec	-80 arc sec	0	-9.6	0	-92 ± 44	0	-11.0	0
Time bias		20 sec	+4.5	+1.0	N	17 ± 6	+3.8	+0.9	0
IGS time scale factor	50 ppm	-80 ppm	-6.1	-1.4	N	-63 ± 20	-4.8	-1.1	0
Total velocity error			+7.4	-0.9	+8.6		+7.5	-1.8	+8.2

External tracker errors

System	Range bias, ft	P-bias, ft	Q-bias, ft	Azimuth, radians	Elevations, radians	Refraction, n units
GE MOD III (final)	25 ± 34	N/A	N/A	$-1 \times 10^{-5} \pm 0.56 \times 10^{-4}$	$-0.47 \times 10^{-4} \pm 0.21 \times 10^{-3}$	0.25 ± 13
MISTRAM 100K	31 ± 30	-0.6 ± 1.0	2.5 ± 1.0	N/A	N/A	Not available

N = negligible

N/A = not applicable

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

Error	Position, ft			Velocity, ft/sec		
	X	Y	Z	X	Y	Z
IMU	+1000 ± 200	-160 ± 50	+960 ± 100	+9.0 ± 1.0	-0.4 ± 3.0	+8.7 ± 2.0
Navigation	+920	+20	+2900	+0.6	+0.3	+8.5
Total guidance	+1920 ± 200	-140 ± 50	+3800 ± 100	+9.6 ± 1.0	-0.1 ± 3.0	+17.2 ± 2.0

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TABLE 5.1.5-V.- 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	25 723	0.00	-	-	-
Inertial guidance system	25 715	-0.07	25 295	4630	-28
Preliminary best estimate trajectory	25 706	-0.10	25 285	4630	133
MISTRAM 10K	25 700	-0.05	25 283	4612	163
MISTRAM 100K	25 709	-0.15	25 283	4665	132
GE MOD III (final)	25 706	-0.10	25 285	4630	133
GE MOD III (real time)	25 694	-0.06			
MISTRAM Impact Predictor	25 712	-0.22			

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TABLE 5.1.5-VI.- TRANSLATION MANEUVERS

Event	Ground elapsed time, hr:min:sec	Actual				Planned $\Delta\bar{V}$, ft/sec
		Components, ft/sec			Total $\Delta\bar{V}$, ft/sec	
		ΔV_X	ΔV_Z	ΔV_Z		
Tail-off	0:05:39.8	+90.4	+20.1	+3.0	92.7	--
Separation and IVAR	0:06:06.1	+29.5	-1.0	+1.5	29.6	20.0
M=3 rendezvous						
Phase-adjust	0:49:03.3	+74.8	-0.2	+0.2	74.8	73.6
Corrective-combination	1:55:17.4	+4.1	-10.3	+9.8	14.8	14.7
Coelliptic	2:24:51.7	+40.6	+34.7	+2.1	53.4	54.0
Terminal phase initiate	3:36:02.2	+24.1	-15.0	-1.1	28.4	
First midcourse correction	3:48:35.4	-3.6	+0.7	-1.8	4.1	
Second midcourse correction	4:00:26.8	+2.8	-2.8	0.0	4.0	
Terminal phase finalize	4:06:08.2	+21.0	+11.7	-2.0	24	
Radial separation	5:00:59.8	-0.3	-19.9	-0.1	19.9	20.0

TABLE 5.1.5-VI.- TRANSLATION MANEUVERS - Continued

Event	Ground elapsed time, hr:min:sec	Components, ft/sec			Total $\Delta\bar{V}$, ft/sec	Planned $\Delta\bar{V}$, ft/sec
		ΔV_X	ΔV_Y	ΔV_Z		
Equi-period rendezvous						
Terminal phase initiate[a]	6:15:11.9	+0.5	-2.0	0.0	2.1	
First midcourse correction	6:20:19.7	-0.3	-1.1	-0.1	1.0	
Intermediate corrections	6:26:58.8	-0.6	-0.7	-0.1	1.0	
Second midcourse correction	6:28:33.8	-2.2	-2.0	+0.1	3.0	
Terminal phase finalize	6:29:26.4	+4.7	+15.6	-0.8	16.3	
Separation[a]	7:14:58	-3.7	0	0	(b)	3.7
Rendezvous from above						
Phase-adjust[a]	18:23:19	2	0	0	(b)	2.0
Height-adjust[a]	19:08:16	17	0	0	(b)	17.0
Coelliptic[a]	19:54:24	11.4	-8.9	0	(b)	14.4
Terminal phase initiate	21:02:27.5	-17.0	±6.0	+0.1	18.0	

[a]Data were obtained from crew reports or air-to-ground communications.

[b]No data available.

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TABLE 5.1.5-VI.- TRANSLATION MANEUVERS - Concluded

Event	Ground elapsed time, hr:min:sec	Components, ft/sec			Total $\Delta\bar{V}$, ft/sec	Planned $\Delta\bar{V}$, ft/sec
		ΔV_X	ΔV_Y	ΔV_Z		
First midcourse correction[c]	(b)	4 aft (along line of sight)	1 up	3 lt	(b)	
Second midcourse correction[c]	(b)	2 fwd (along line of sight)	10 dwn	7 rt	(b)	
Terminal phase finalize	(b)	(b)	(b)	(b)	(b)	
Separation	(b)	(b)	(b)	(b)	(b)	(b)
True-anomaly adjust[a]	53:41:35	(b)	(b)	(b)	(b)	25.0
Retrofire	71:46:44	+295.9	+125.2	-3.8	321.3	318.4

[a]Data were obtained from crew reports or air-to-ground communications.

[b]No data available.

[c]Incremental velocity indicator values in spacecraft coordinates obtained from pilot's log.

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TABLE 5.1.5-VII.- PLATFORM ALIGNMENT ACCURACY PRIOR TO MAJOR MANEUVERS

Maneuver	Ground elapsed time, hr:min:sec	Alignment accuracy (horizon sensor output minus gimbal angle)	
		Pitch, deg	Roll, deg
Phase adjust	0:49:03	+0.2	+0.3
Corrective combination	1:55:17	+0.3	+0.4
Coelliptic	2:24:52	+0.3	+0.4
First terminal phase initiate	3:36:02	+0.2	-0.2
Radial separation	5:01:00	+0.3	+0.4
Second terminal phase initiate	6:15:12	-0.3	+0.4
Coelliptic	19:54:24	+0.2	+0.3
Retrofire	71:46:44	+0.1	+0.1

TABLE 5.1.5-VIII.- COMPARISON OF COMPUTED SOLUTIONS WITH VELOCITY CHANGES ACCOMPLISHED ON M=3 RENDEZVOUS MANEUVERS

Maneuver	Solution			Applied maneuvers
	Onboard computer	Onboard backup	Ground backup	
Terminal phase initiate, ft/sec	26 forward	24 forward	26.7 forward	27 forward
	4 right	--	2.2 right	2 right
	8 up	0 up	1.5 up	3 up
First midcourse correction, ft/sec	2 aft	3 aft	--	2 aft
	3 right	--	--	1 right
	2 up	4 up		2 up
Second midcourse correction, ft/sec	3 forward	0 forward	--	3 forward
	0 right	--	--	0 right
	2 down	3 down	--	2 down

TABLE 5.1.5-IX.- THRUST HISTORIES FOR THREE SIMULATIONS

OF M=3 RENDEZVOUS

Maneuver	Flight value (run 1)	Computer and simulator (run 2)	Simulator only (run 3)
Terminal phase initiate, ft/sec	28 forward 2 right 3 up	26 forward 4 right 8 up	27 forward 3 right 1 down
First midcourse correction, ft/sec	2 aft 1 right 2 up	0 forward 1 left 11 down	1 aft 0 right 0 down
Second midcourse correction, ft/sec	3 forward 0 right 2 down	1 forward 0 right 0 down	0 forward 0 right 0 down

TABLE 5.1.5-X.- CHANGE IN PITCH GIMBAL ANGLE AND RADAR ELEVATION PRECEDING TPI ON M=3 RENDEZVOUS

Radar sample	Pitch, deg	Elevation, deg	Look angle, deg	Change in pitch, deg	Change in elevation, deg	Change in look angle, deg
1	15.7	0.24	15.46	--	--	--
2	16.1	0.00	16.10	0.4	-0.24	0.64
3	17.0	0.00	17.00	0.9	0.00	0.90
4	17.7	-0.05	17.75	0.7	-0.05	0.75
5	19.3	0.24	19.06	1.6	0.29	1.31
6	20.6	0.67	19.93	1.3	0.43	0.87
7	21.9	1.10	20.80	1.3	0.43	0.87
8	23.4	-0.29	23.69	1.5	-1.39	2.89

TABLE 5.1.5-XI.- COMPARISON OF CALCULATED VELOCITY COMPONENTS FOR M=3 RENDEZVOUS

	Terminal phase initiate				Terminal phase finalize				$\Delta\overline{V}_T$, ft/sec
	$\Delta\dot{X}$	$\Delta\dot{Y}$	$\Delta\dot{Z}$	$\Delta\overline{V}$	$\Delta\dot{X}$	$\Delta\dot{Y}$	$\Delta\dot{Z}$	$\Delta\overline{V}$	
	+23.2	-14.8	-1.7	+27.6	+20.1	+25.3	+1.6	+32.3	+59.9
RTCC	+21.7	-21.0	-1.7	+30.3	+21.5	+19.1	+1.4	+28.8	+59.1
Difference	+1.5	+6.2	0	-2.7	-1.4	+6.2	+0.2	+3.5	+0.8
	+22.4	-17.5	-2.8	+28.6	+20.7	+22.7	+2.1	+30.8	+59.4
Best estimated trajectory	+21.0	-23.5	-2.9	+31.6	+22.1	+16.6	+2.0	+27.7	+59.3
Difference	+1.4	+6.0	+0.1	-3.0	-1.4	+6.1	+0.1	+3.1	+0.1

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TABLE 5.1.5-XII.- COMPARISON OF COMPUTED SOLUTIONS WITH VELOCITY CHANGES ACCOMPLISHED ON RENDEZVOUS FROM ABOVE

Maneuver	Onboard computer	Onboard backup	Ground backup	Applied maneuvers
Terminal phase initiate, ft/sec	19 forward 2 left 4 down	16.5 forward 0 left 3 up	16.5 forward 2.5 right 0.3 up	16 forward 0 left 1 up
First midcourse correction, ft/sec	4 aft 5 left 1 up	None computed		[a]4 aft [a]3 left [a]1 up
Second midcourse correction, ft/sec	2 forward 7 right 10 down	1 aft 0 left 5 down		[a]2 forward [a]7 right [a]10 down

[a]As reported by crew.

TABLE 5.1.5-XIII.- THRUST HISTORIES FOR THREE SIMULATIONS

OF RENDEZVOUS FROM ABOVE

Maneuver	Flight value (run 1)	Computer and simulator (run 2)	Simulator only (run 3)
Terminal phase initiate, ft/sec	16 forward 0 left 1 up	19 forward 2 left 4 down	29 forward 4 right 15 down
First midcourse correction, ft/sec	4 aft 3 left 1 up	34 forward 5 left 1 down	0 forward 0 left 0 down
Second midcourse correction, ft/sec	2 forward 7 right 10 down	2 forward 0 right 0 down	0 forward 0 right 0 down

TABLE 5.1.5-XIV.- COMPUTER TELEMETRY REENTRY PARAMETERS

Parameters	Time in mode = 2 731.7 sec Altitude = 400K ft			Time in mode = 3 240.4 sec Guidance termination		
	Telemetry	MAC	IBM	Telemetry	MAC	IBM
Radius vector, ft	21 304 156.0			20 995 235.0		
Velocity, ft/sec	24 373.6			1 751.8		
Flight-path angle, deg	-1.522			-29.124		
Downrange error, n. mi. . . .	NA			.143		
Crossrange error, n. mi. . . .	2.528			-1.348		
Bank-angle command, deg . . .	0.0			-2.499		
Latitude, deg	27.997			27.715		
Longitude, deg	253.804			284.928		
Density altitude factor . . .	0.0			4.661		
Zero lift range predicted, n. mi.	NA			5.174		
Range to target, n. mi. . . .	1 649.681			3.854		
Spacecraft heading, deg . . .	82.266			98.839		

NA = not available.

TABLE 5.1.5-XV.- START-COMPUTE-DISCRETE ANOMALY EVENTS

Ground elapsed time, hr:min:sec	Computer mode	START COMP discrete	Comments
1:57:16	Catch-up	Off	Maneuver commands for coelliptic maneuver (N_{SR}) inserted. Crew reports normal operation during this period.
2:12:04		On	Crew reports START button depressed here. Operation still normal.
2:12:09	Rendezvous	Unknown	
2:21:13	Catch-up	On	First apparent anomaly. IVI displays N_{SR} commands. Crew reported FDI anomaly during this period.
2:22:54	Rendezvous	Unknown	
2:22:57	Catch-up	Off	FDI commands displayed.
2:23:01		On	IVI displays N_{SR} commands.
2:23:47	Rendezvous	Unknown	
2:23:49	Catch-up	On	IVI displays N_{SR} commands. No FDI commands.
2:24:04	Rendezvous	Unknown	
2:24:06	Catch-up	Off	FDI commands displayed.
2:24:09		On	IVI displays N_{SR} commands.
2:24:54		On	N_{SR} thrusting begins.
2:29:06	Rendezvous	Unknown	
2:29:09	Catch-up	On	IVI displays N_{SR} commands. No FDI commands.

TABLE 5.1.5-XV.- START-COMPUTE-DISCRETE ANOMALY EVENTS - Continued

Ground elapsed time, hr:min:sec	Computer mode	START COMP discrete	Comments
2:29:21	Rendezvous	Unknown	
2:29:25	Catch-up	On	IVI displays N_{SR} commands. No FDI commands.
2:29:45	Rendezvous	Unknown	
2:29:49	Catch-up	On	IVI displays N_{SR} commands. No FDI commands.
			Manual Data Insertion Unit addresses 25, 26, 27 inserted as zero by crew.
2:30:42	Rendezvous	Unknown	
2:30:45	Catch-up	On	IVI displays zeroes.
2:31:52	Rendezvous	Unknown	
2:43:42			ΔV_T displayed
2:43:47		On	IVI displays TPI commands.
2:46:30	Catch-up	On	IVI displays TPI commands.
2:46:42	Prelaunch	Unknown	
2:47:06	Catch-up	On	IVI displays TPI commands.
2:47:21	Rendezvous	Unknown	
2:59:13			ΔV_T displayed.
2:59:18		On	IVI displays TPI commands.
3:01:28	Catch-up	Unknown	Computer not in mode long enough to indicate state of start-compute discrete.
3:01:32	Prelaunch	Unknown	

TABLE 5.1.5-XV.- START-COMPUTE-DISCRETE ANOMALY EVENTS - Concluded

Ground elapsed time, hr:min:sec	Computer mode	START COMP discrete	Comments
3:02:40	Rendezvous	Unknown	
3:14:30			ΔV_T displayed.
3:14:35		On	IVI displays TPI commands.
3:18:59	Catch-up	On	IVI displays TPI commands.
3:20:21	Rendezvous	Unknown	
3:32:13			ΔV_T displayed.
3:32:18		On	IVI displays TPI commands.
3:36:03			TPI thrusting begins.
3:48:20			First midcourse thrusting begins.
4:00:20			Second midcourse thrusting begins.
4:00:44	Catch-up	Off	After this time, the operation of the start-compute discrete appears to be normal.

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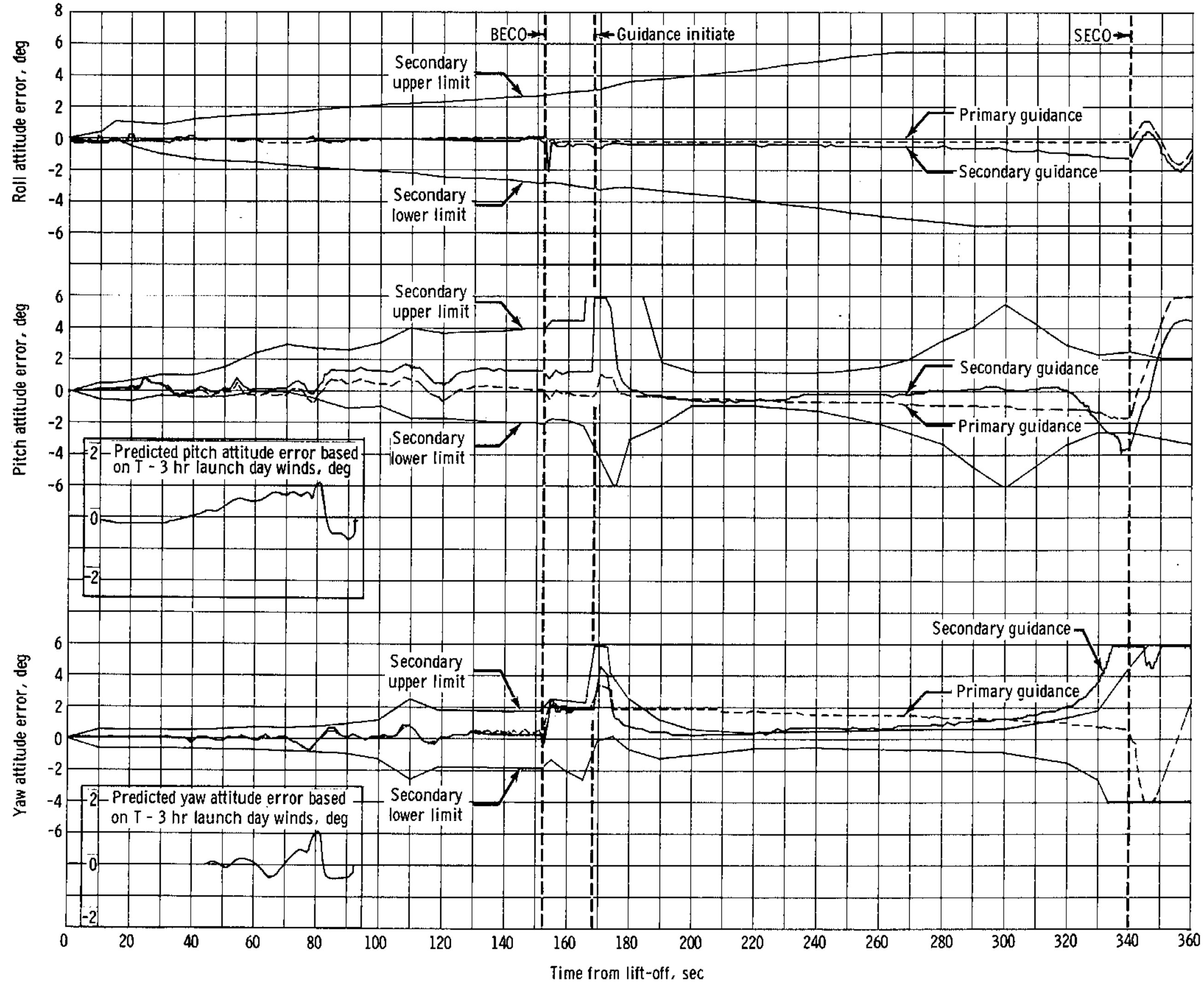

Figure 5.1.5-1. - Comparisons of launch vehicle and spacecraft steering errors.

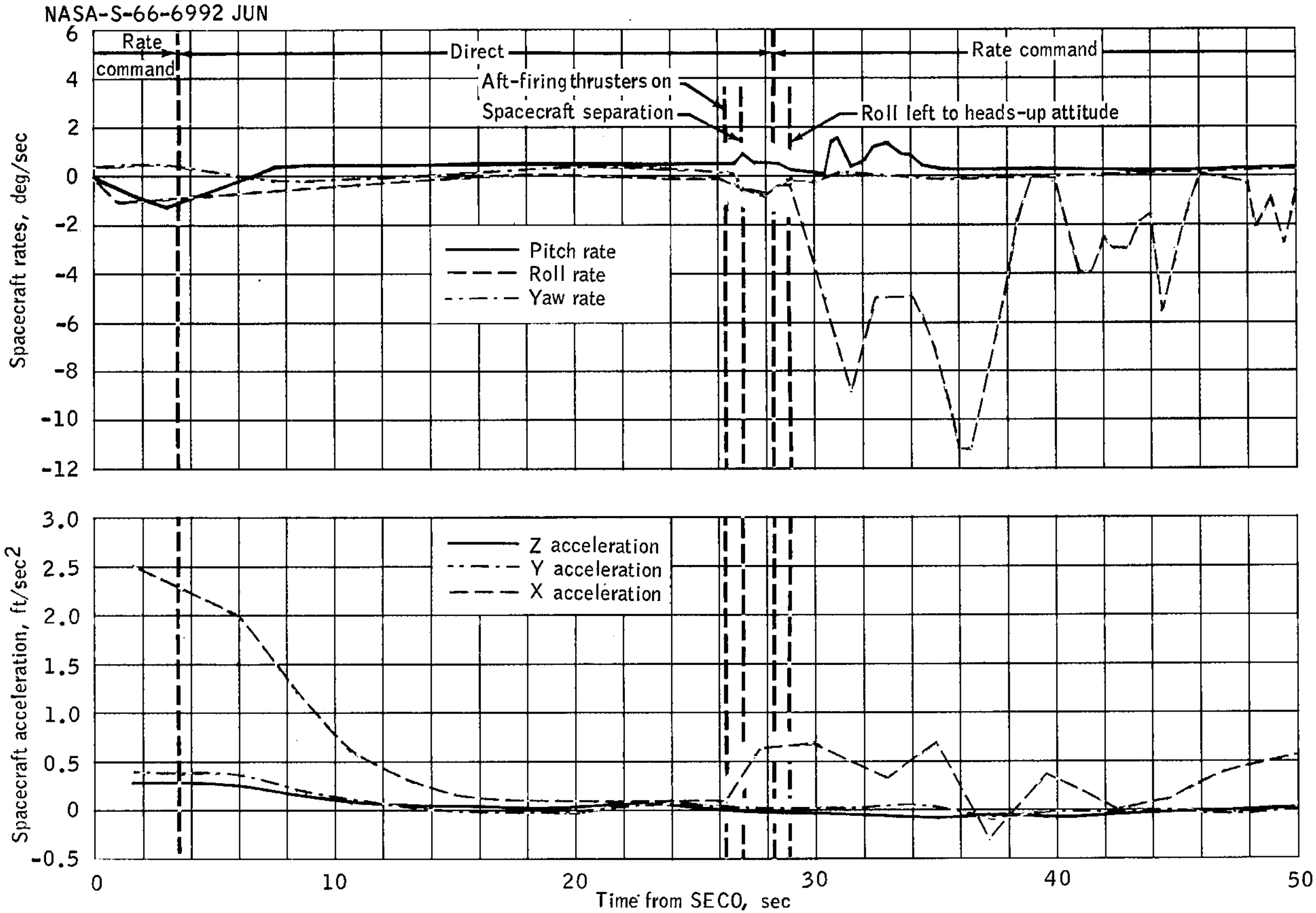

Figure 5.1.5-2. - Post-SECO acceleration and rate profile.

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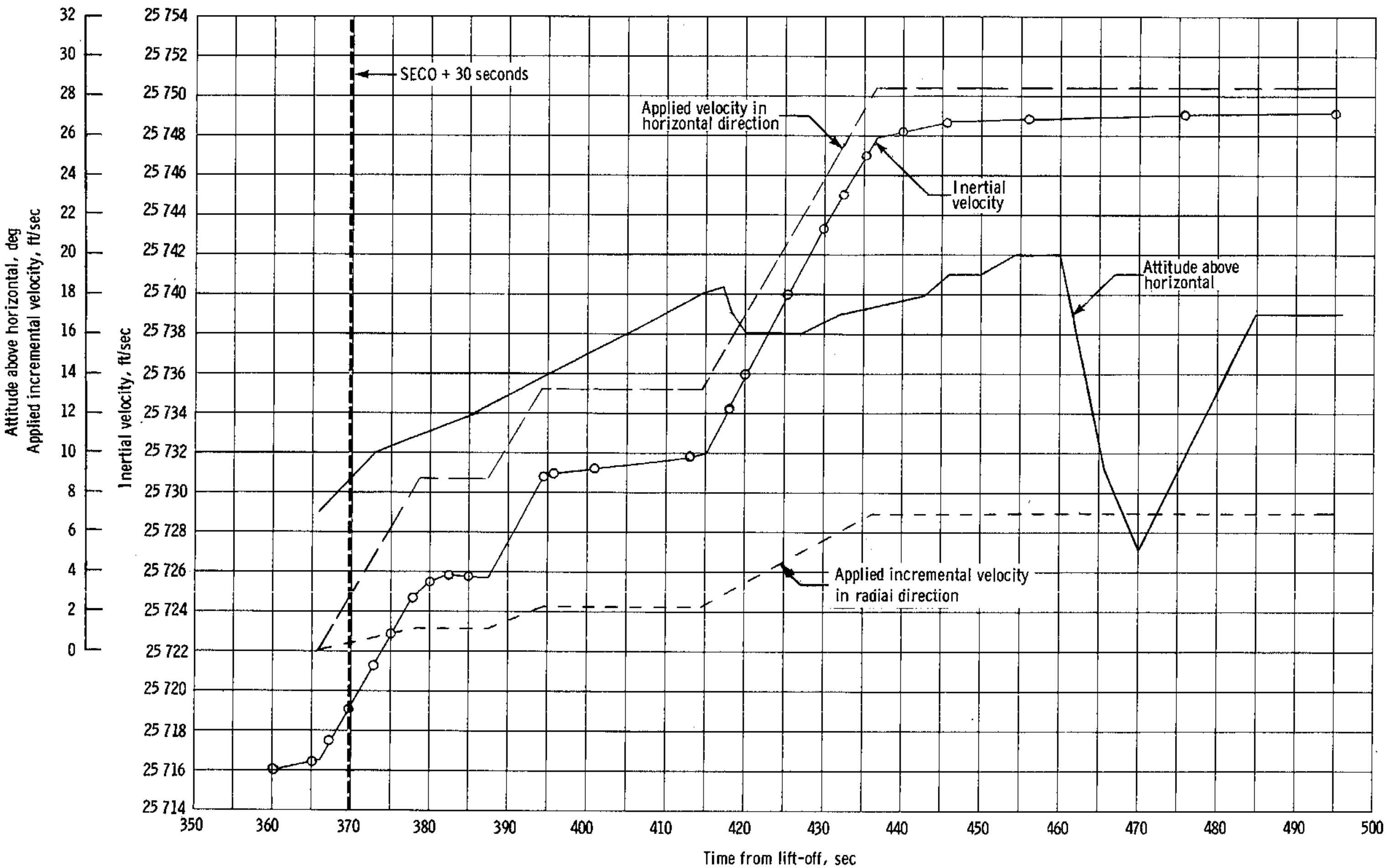

Figure 5.1.5-3.- Insertion maneuvers.

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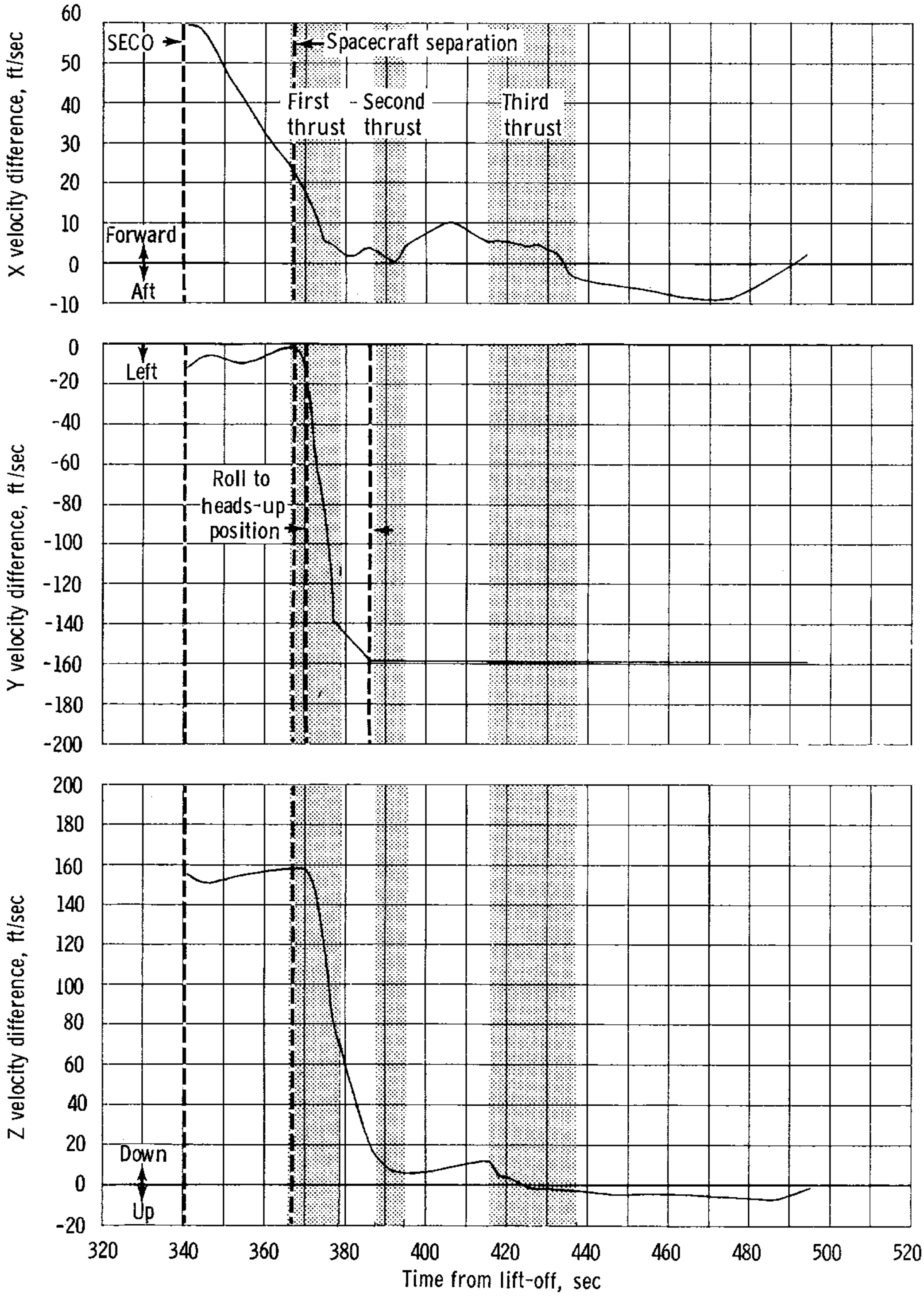

Figure 5.1.5-4.- Reconstructed IVI readings during insertion maneuvers.

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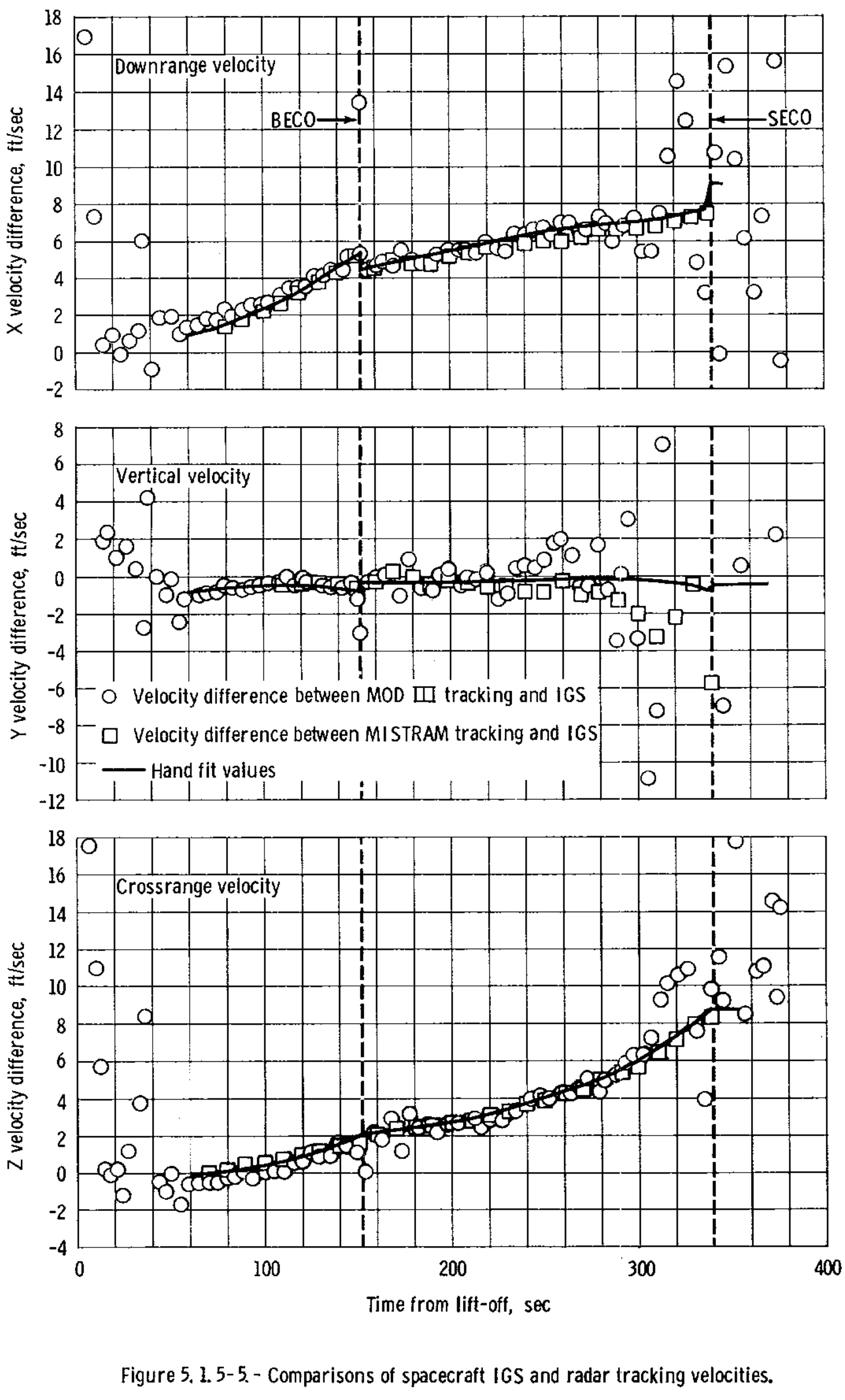

Figure 5.1.5-5.- Comparisons of spacecraft IGS and radar tracking velocities.

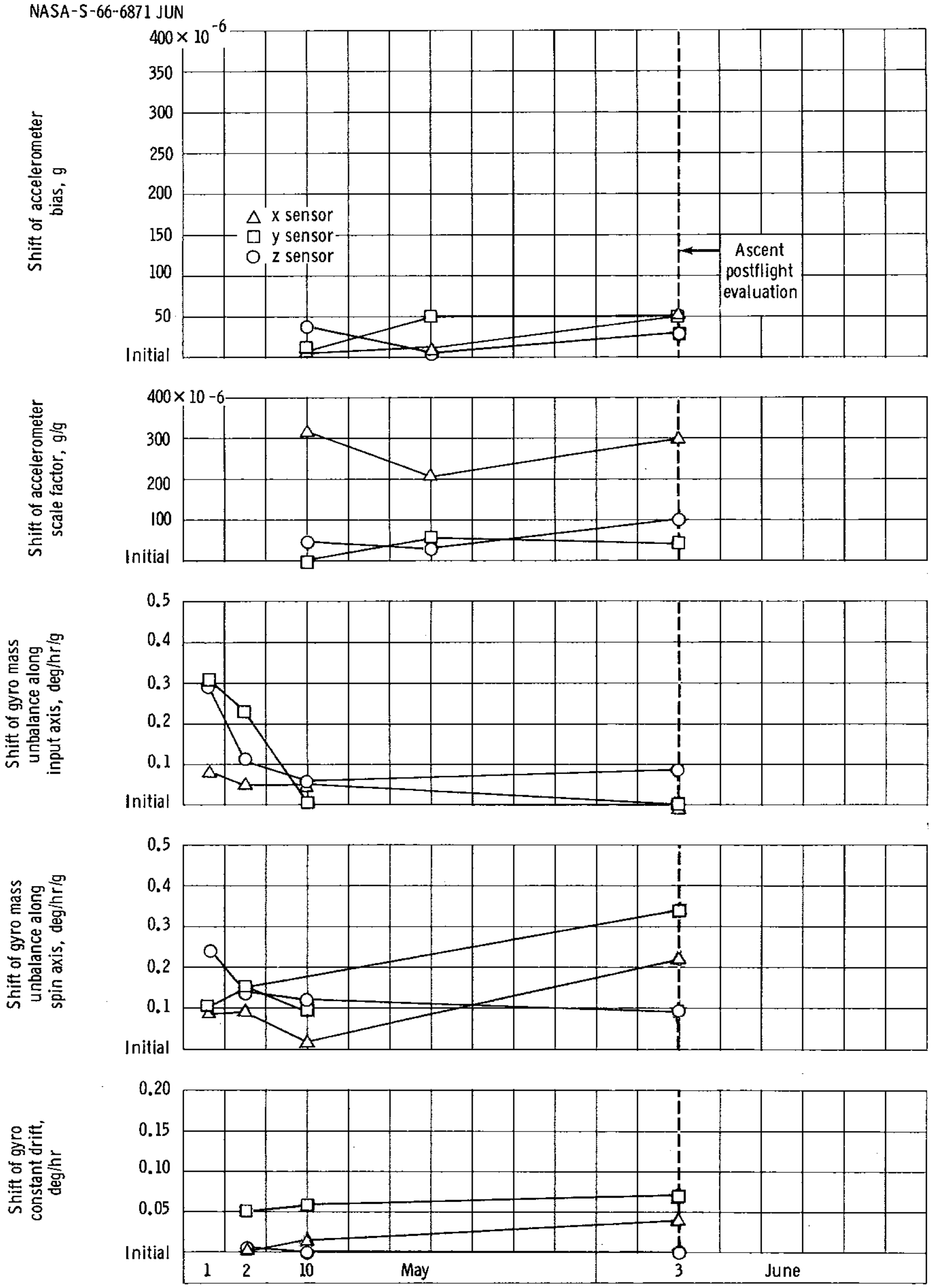

Figure 5.1.5-6. - IMU error coefficient history.

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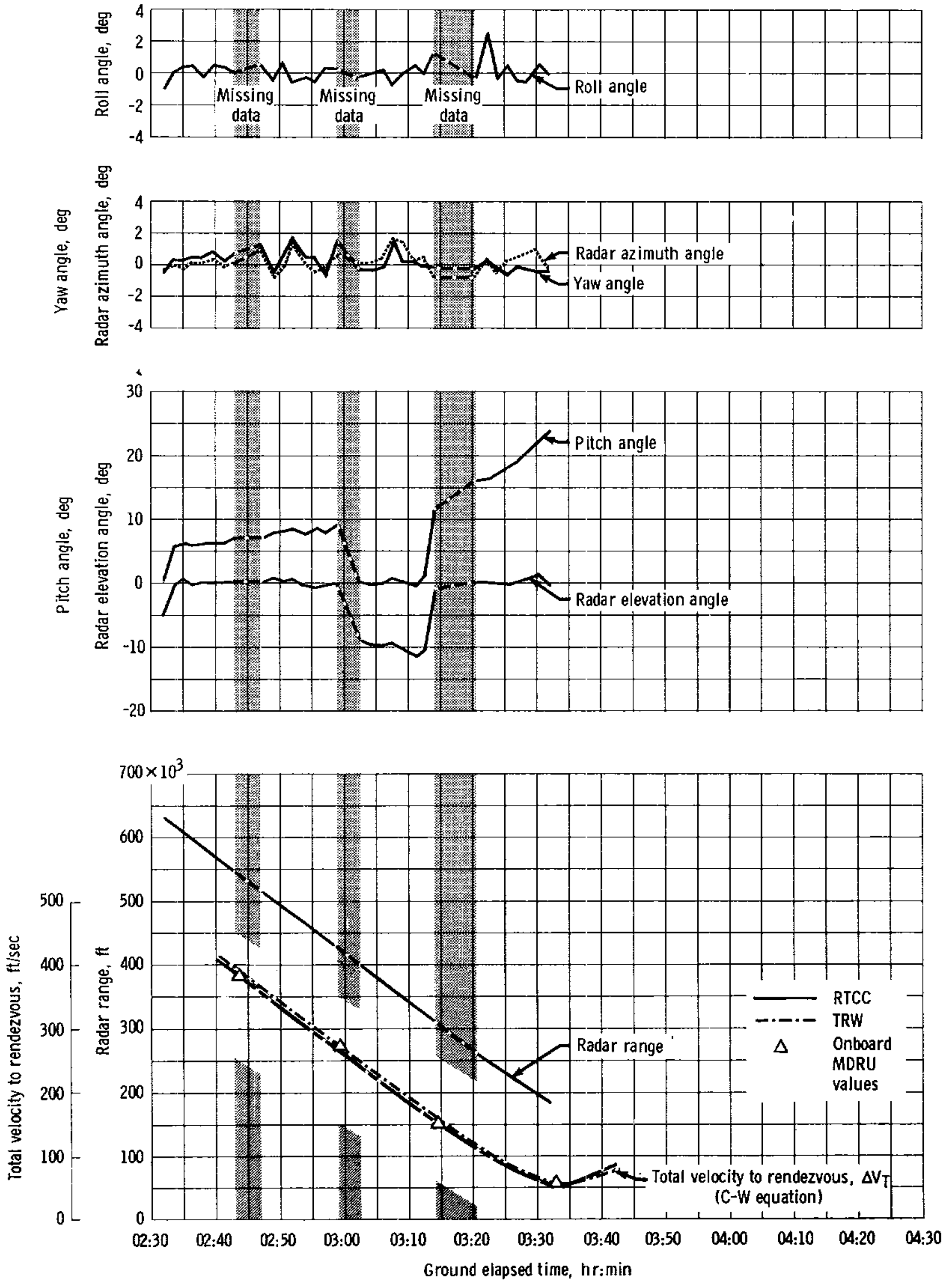

Figure 5.1.5-10. - Total-velocity-to-rendezvous comparison for the M-3 rendezvous.

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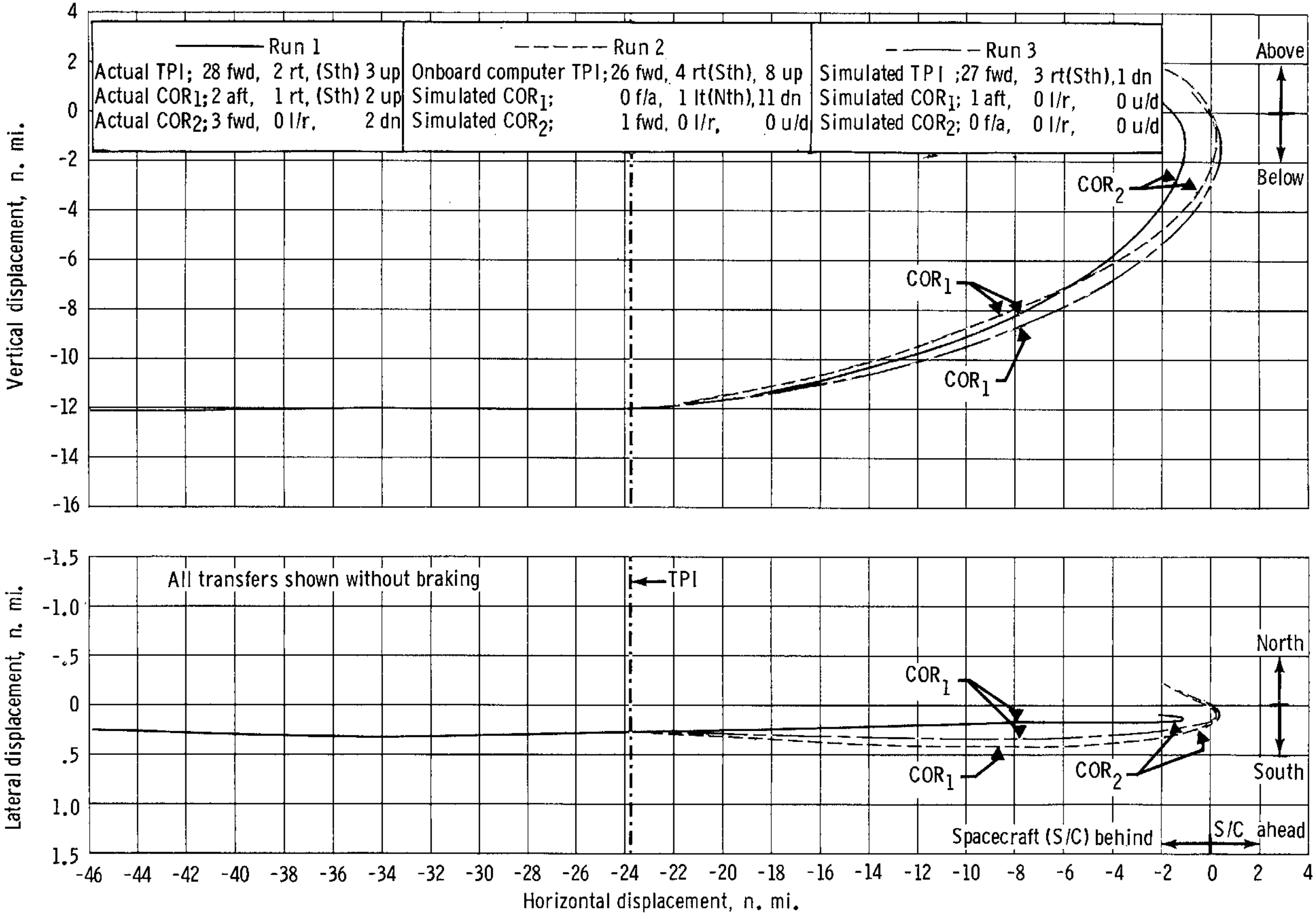

Figure 5.1.5-11. - Simulated relative trajectory profile, measured from ATDA to Spacecraft 9, for M=3 rendezvous.

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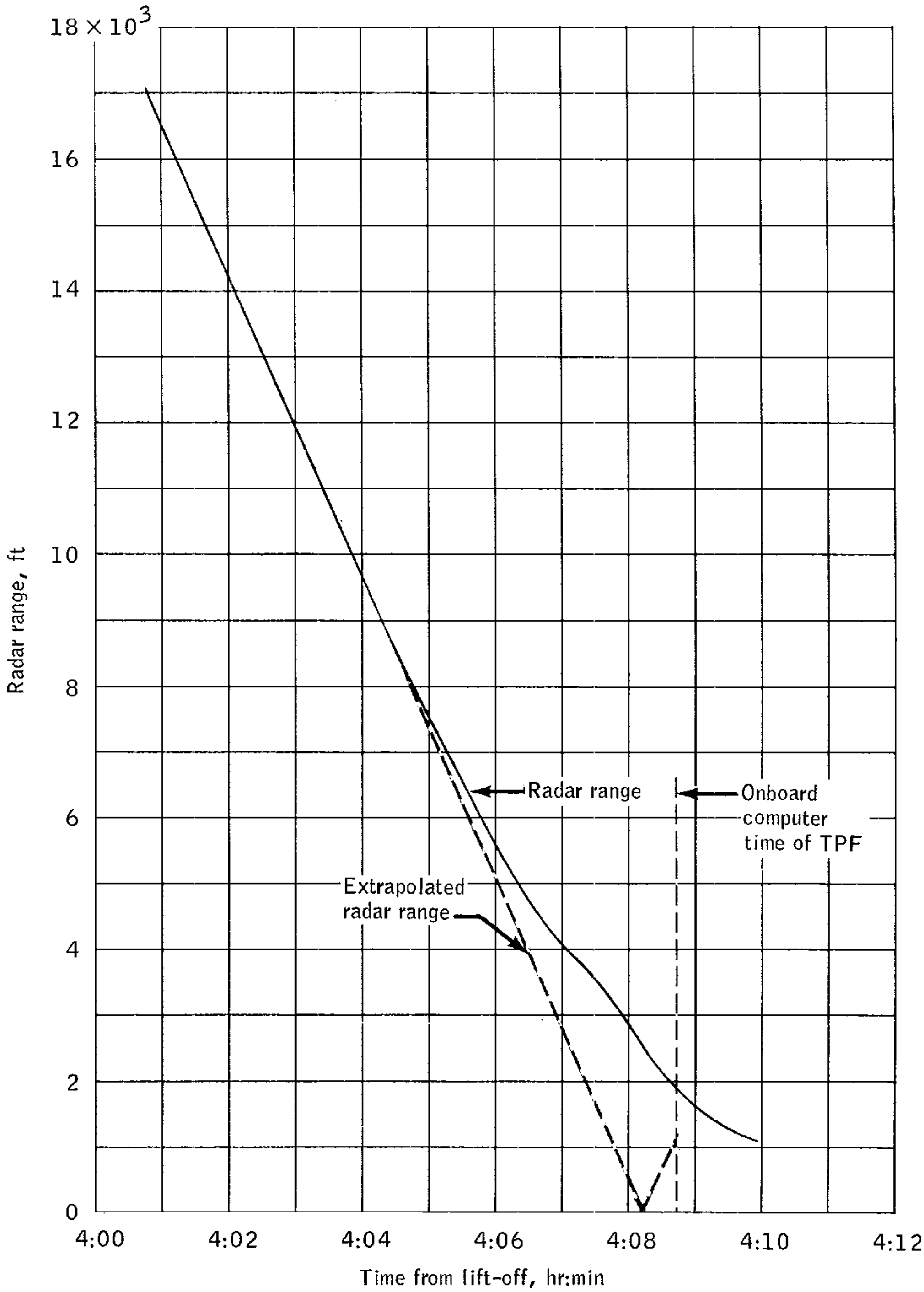

Figure 5.1.5-12.- Closing trajectory for M = 3 rendezvous.

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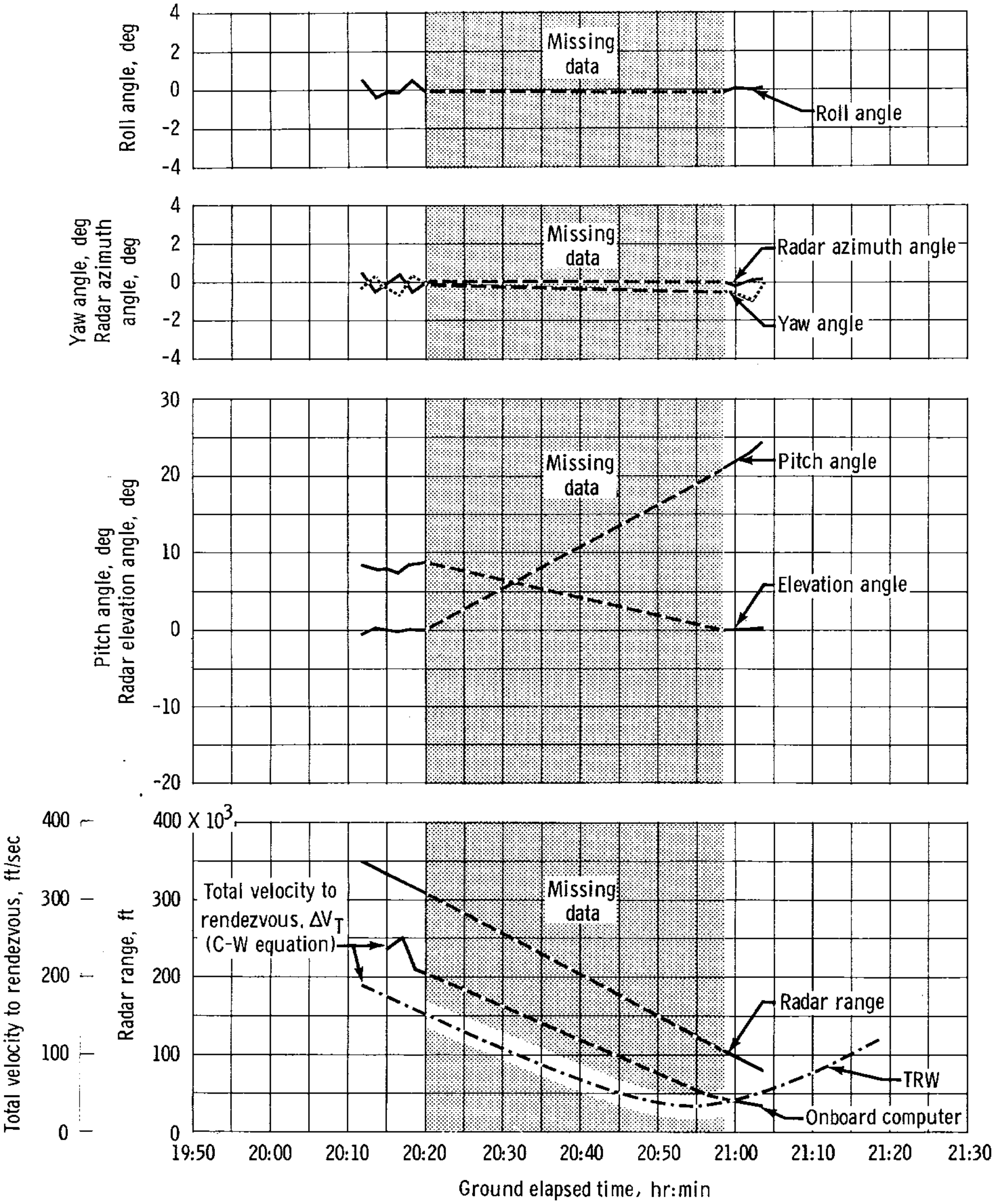

Figure 5.1.5-13. - Total-velocity-to-rendezvous comparison for the rendezvous from above.

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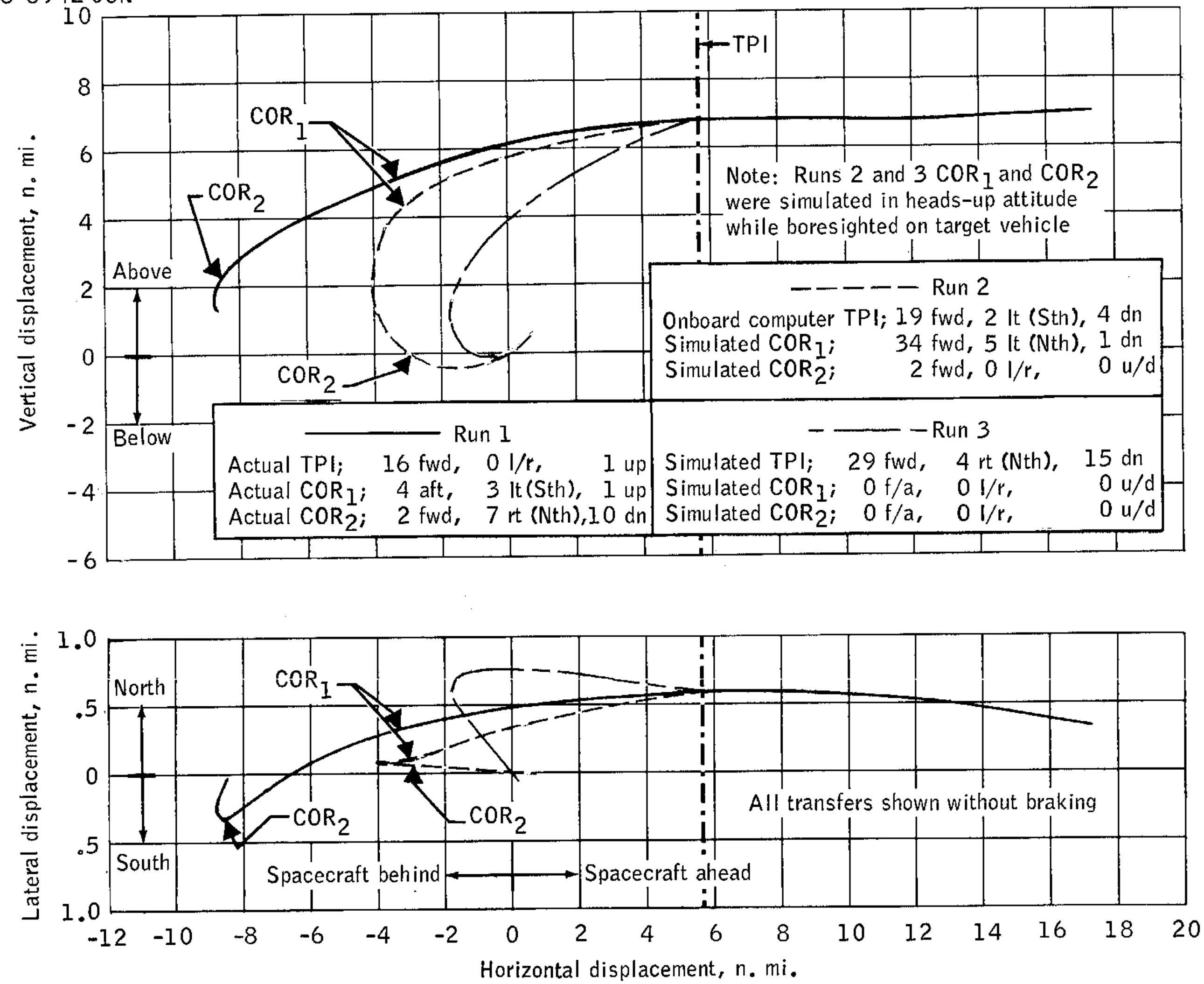

Figure 5.1.5-14. - Simulated relative trajectory profile, measured from ATDA to Spacecraft 9, for rendezvous from above.

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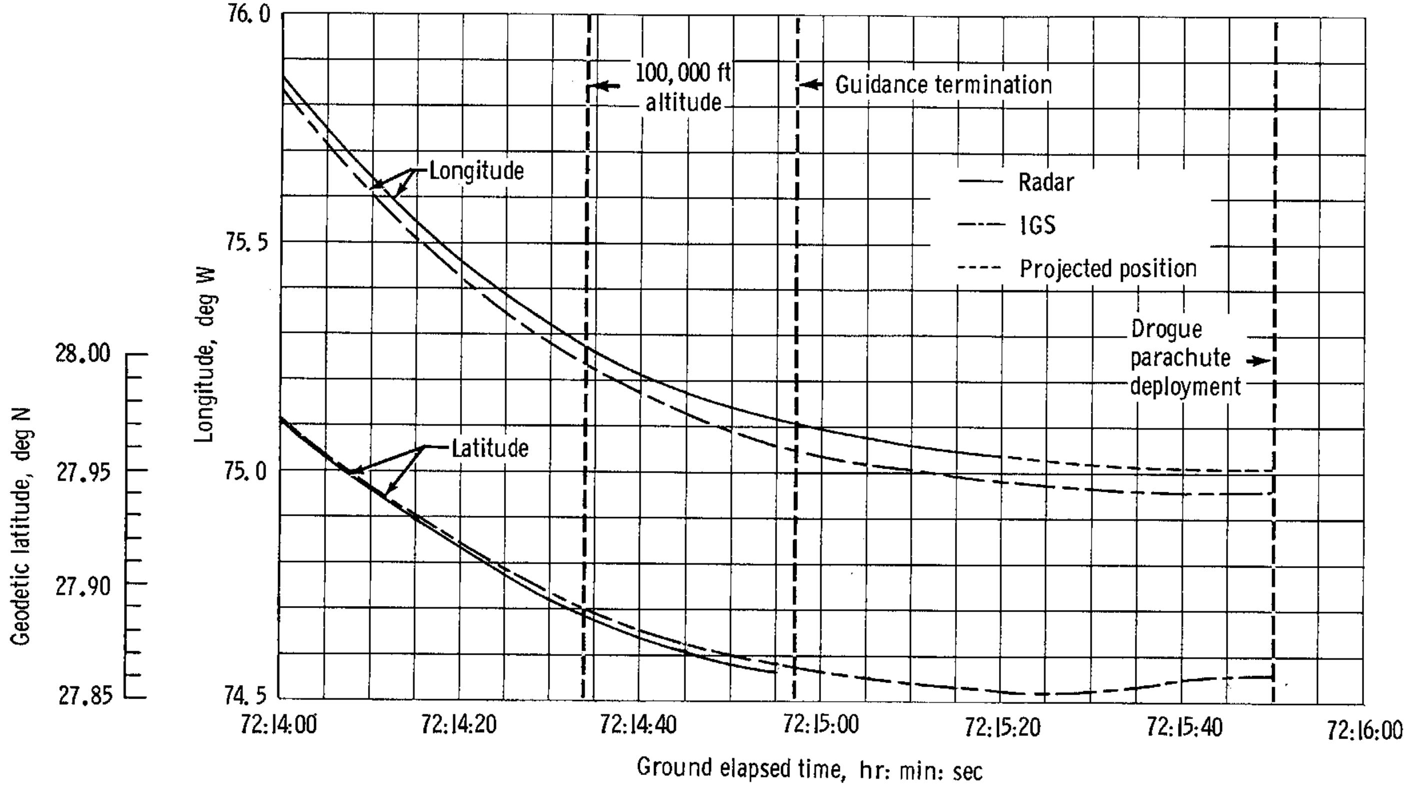

Figure 5.1.5-15. - Comparison of longitude and latitude from IGS and tracking data.

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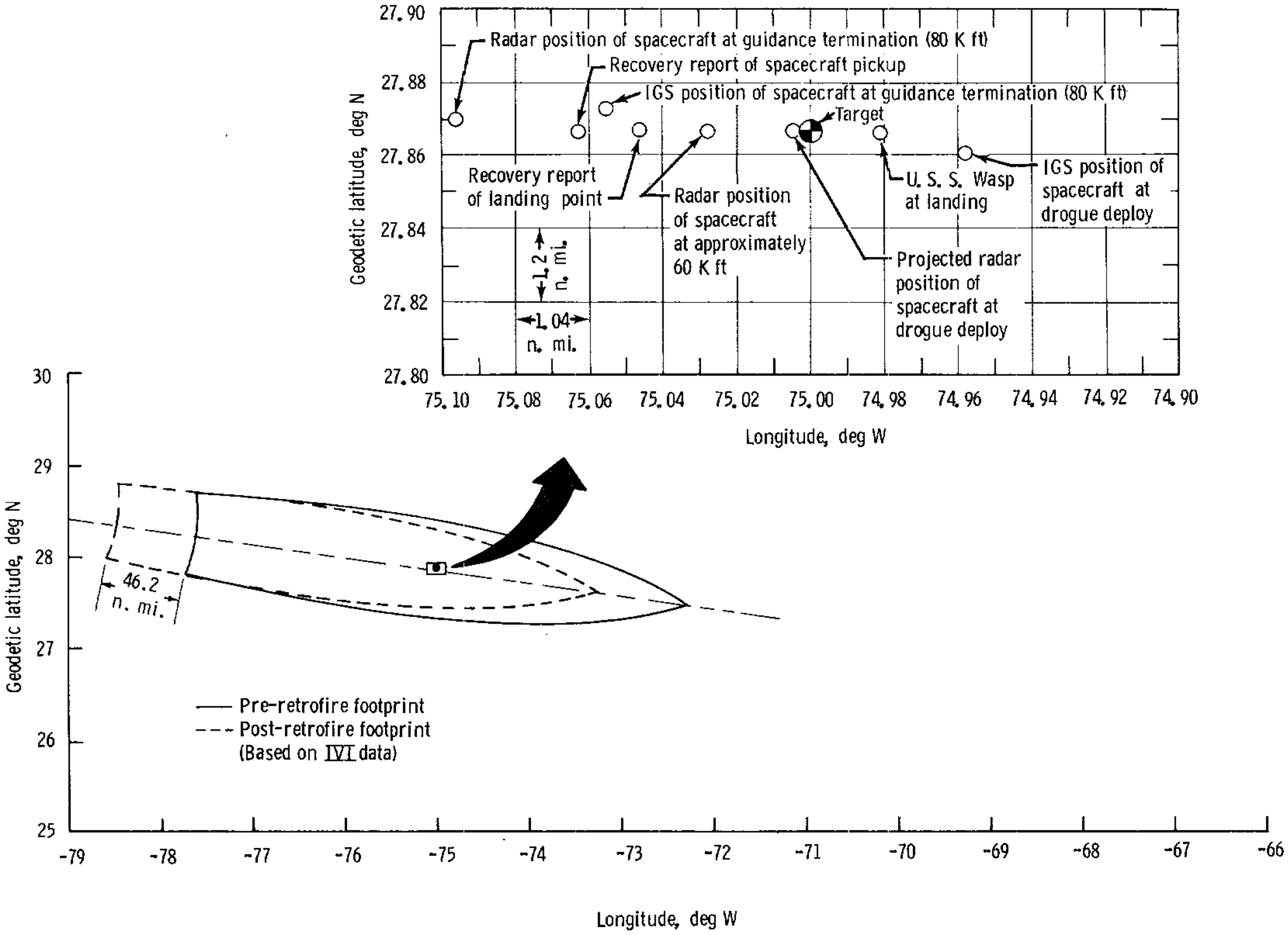

Figure 5.1.5-16. - Touchdown comparisons.

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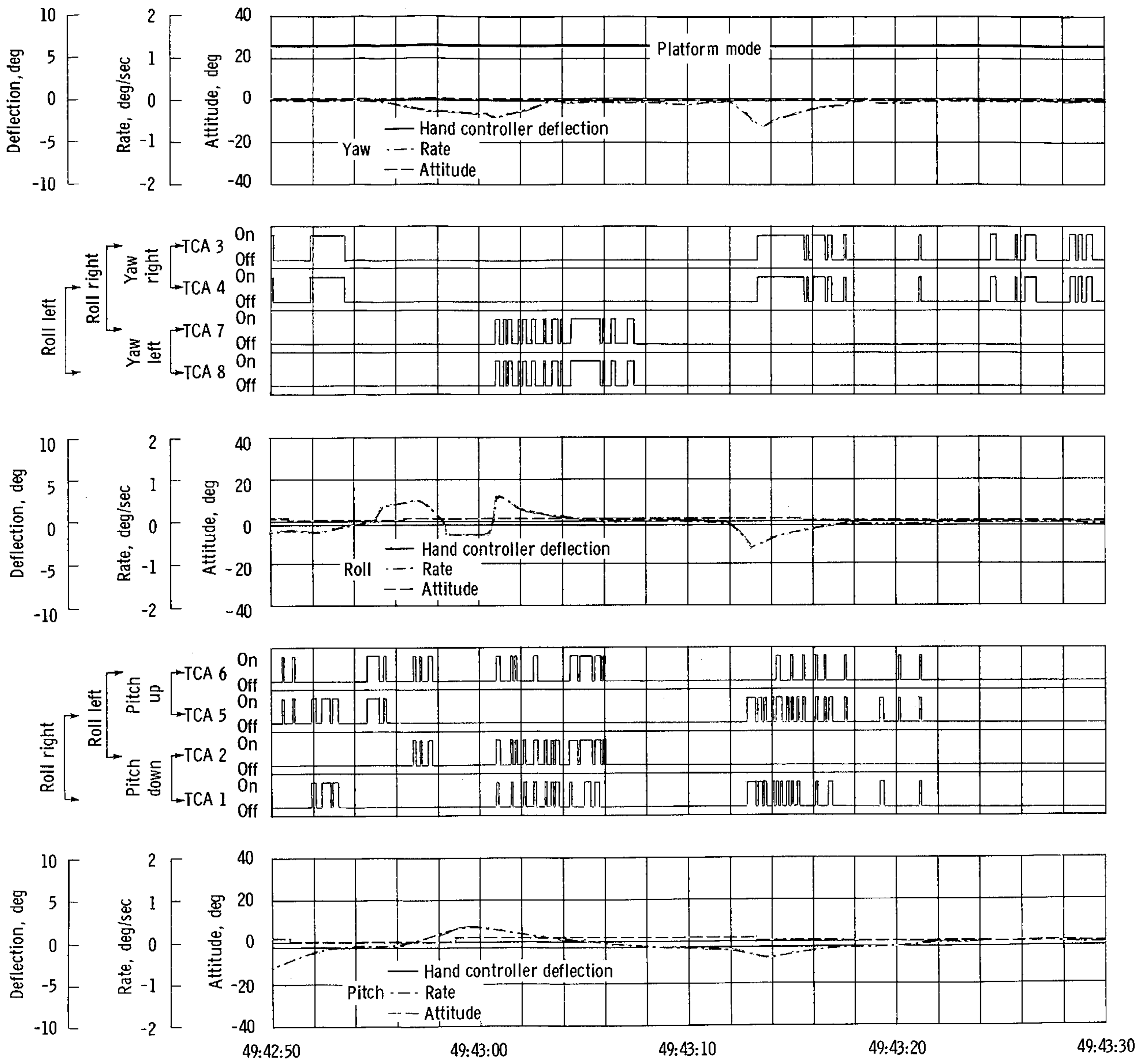

(a) Platform mode.

Figure 5.1.5-17. - Control system performance during the extravehicular activities.

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5.1.6 Time Reference System

Analysis of available data indicates that all components of the Time Reference System (TRS) performed according to specifications. The spacecraft electronic timer was inadvertently turned off for 2.1 seconds during the forty-third revolution sometime between the Kano station and the Guaymas station. The electronic timer began counting elapsed time approximately 6 milliseconds after lift-off. Maximum error during the first 248 409.366 seconds (69:00:09.366 ground elapsed time (g.e.t.)) of flight was approximately 871 milliseconds, or 3.48 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 71:46:44 g.e.t. The electronic timer read 72:22:08 when it was turned off by the crew approximately a minute and 20 seconds after landing.

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 battery-operated G.m.t. clock and the mechanical G.m.t. clock, but made no special accuracy checks. The clocks were not compared with an accurate clock during the recovery sequence. Satisfactory timing on tapes from the biomedical tape recorder and the onboard 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 throughout the mission. The performance of the fuel cells was excellent even though the average spacecraft electrical loads were higher than for any previous missions. The only flight anomaly was the failure in the hydrogen quantity-sensor circuitry. This failure did not in any way compromise the mission.

5.1.7.1 Silver-zinc batteries.- The main-bus and squib-bus batteries performed satisfactorily during the mission; however, relatively low voltages were exhibited by two of the four main batteries in flight under a 10-ampere test load. These two flight batteries also exhibited poor main-bus load-sharing characteristics when paralleled with the fuel cells during the launch and preretrofire period. However, when the fuel cells were shut down prior to equipment adapter separation, the four main batteries shared the spacecraft load reasonably well, and the load sharing started to improve. Immediately after adapter separation, the main-bus voltage was 23.3 volts with a spacecraft load of 36 amperes, which is representative of a normal system response. Based on these results, the present technique of inflight battery testing is questionable. It is probable that insufficient time was allowed during some of the tests to permit the battery to stabilize sufficiently to show its true condition.

5.1.7.2 Fuel-cell power system.- The fuel-cell power system performed as required in delivering electrical power to the spacecraft systems. The fuel cells supplied a total of 2393 ampere hours of electrical power during the mission. The electrical load ranged from 13 amperes when the spacecraft was powered-down, to a maximum of 52 amperes at full load. The aborted launch, EVA, and three rendezvous operations, in addition to the normal launch and preretrofire loads, resulted in the greatest total of high current levels of any mission (fig. 5.1.7-1). The first and second activations of the sections were performed on May 21 and May 31, 1966, respectively. The fact that the second-activation polarization curves (figs. 5.1.7-2 and 5.1.7-3) were approximately 0.3 volts to 0.5 volts higher than previously experienced can probably be attributed to the short storage period after initial activation. As on previous missions, the performance of the sections decayed during the prelaunch low-load standby periods. During the mission, the net performance decays of 0.003 volt per hour and 0.005 volt per hour at 10 and 24 amperes per section, respectively, were consistent with performance decays on previous spacecraft. The section load-sharing (fig. 5.1.7-4) varied between 48 percent and 52 percent, and the load-sharing of the three stacks within each section varied from 30 percent to 50 percent. (See fig. 5.1.7-5.)

Activation of the hydrogen-to-oxygen differential-pressure lights during all hydrogen purges verified the improvements in hydrogen vent-port design made initially on Spacecraft 8. No ice build-up on the port was noticed during EVA.

5.1.7.3 Reactant supply system.- The reactant supply system performed as expected throughout the mission. A single anomaly occurred in the hydrogen subsystem when, at 26 hours 58 minutes g.e.t., it was noted that the hydrogen-quantity indication was zero. A calibration was performed and normal operation of the telemetry channel was indicated; therefore, the failure was concluded to be in the sensor circuitry in the adapter.

5.1.7.4 Fuel-cell water-storage system.- The fuel-cell water-storage system functioned normally throughout the flight. However, drinking water was dumped to ensure that adequate storage space for the fuel-cell product water was available. Two factors caused this requirement: (1) spacecraft power consumption was approximately 200 ampere-hours greater than predicted, thus generating about 5.7 pounds more fuel-cell product water than expected, and (2) crewmen water-consumption at the end of the second day was about one-half that anticipated.

5.1.7.5 Power distribution system.- At approximately 1 hour 30 minutes g.e.t., an intermittent open circuit was noted in the command pilot's dual utility cord which was being used to distribute power to the optical sight and a camera. The branch which supplies power to the optical sight had malfunctioned, and after troubleshooting, the command pilot switched cords with the pilot. The faulty cord is undergoing failure analysis at the spacecraft contractor's facility.

Postflight inspection of Spacecraft 9 revealed several blown fusistors in the pyrotechnic firing circuitry, a condition which is considered normal.

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

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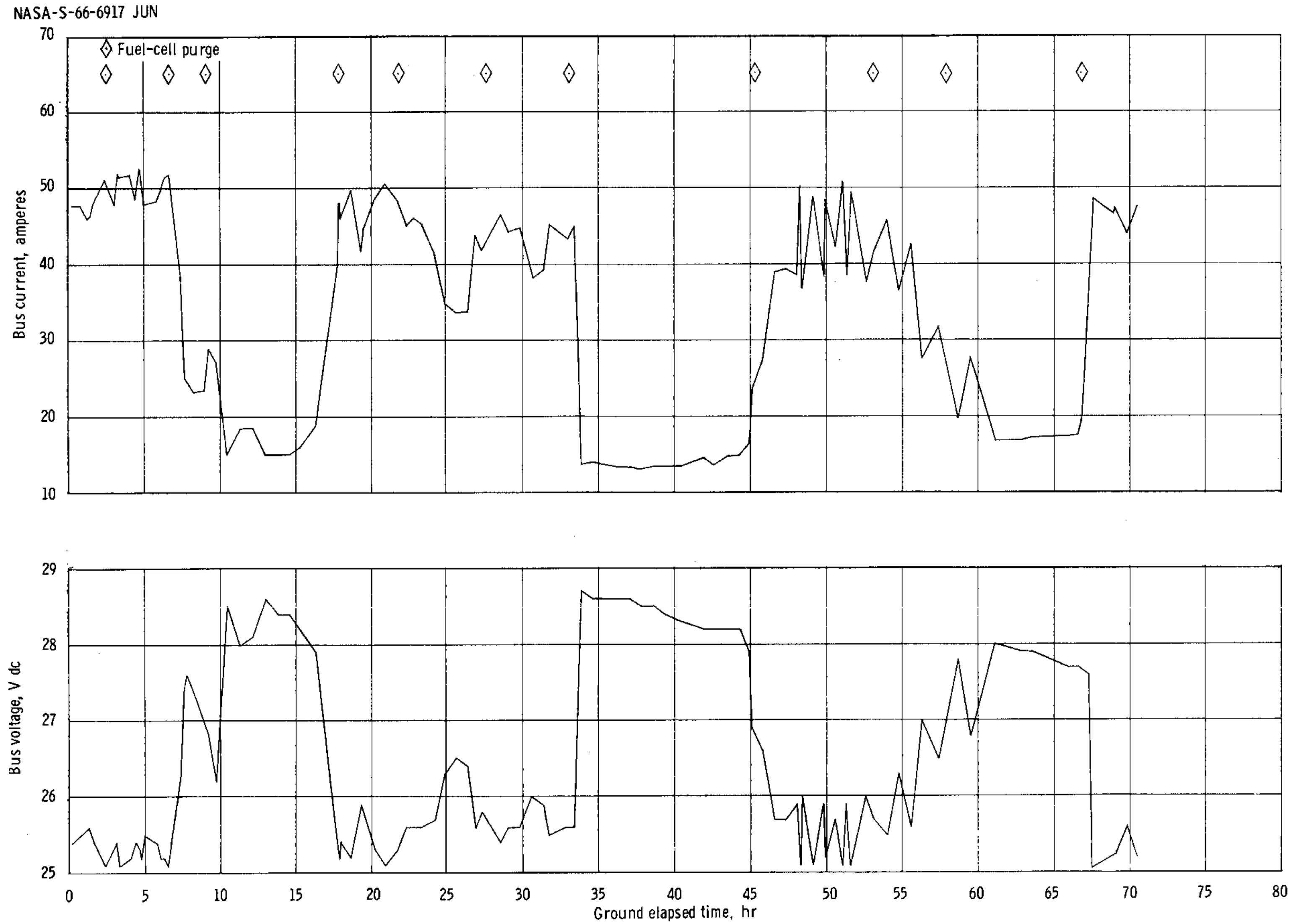

Figure 5.1.7-1. - Spacecraft 9 fuel-cell performance.

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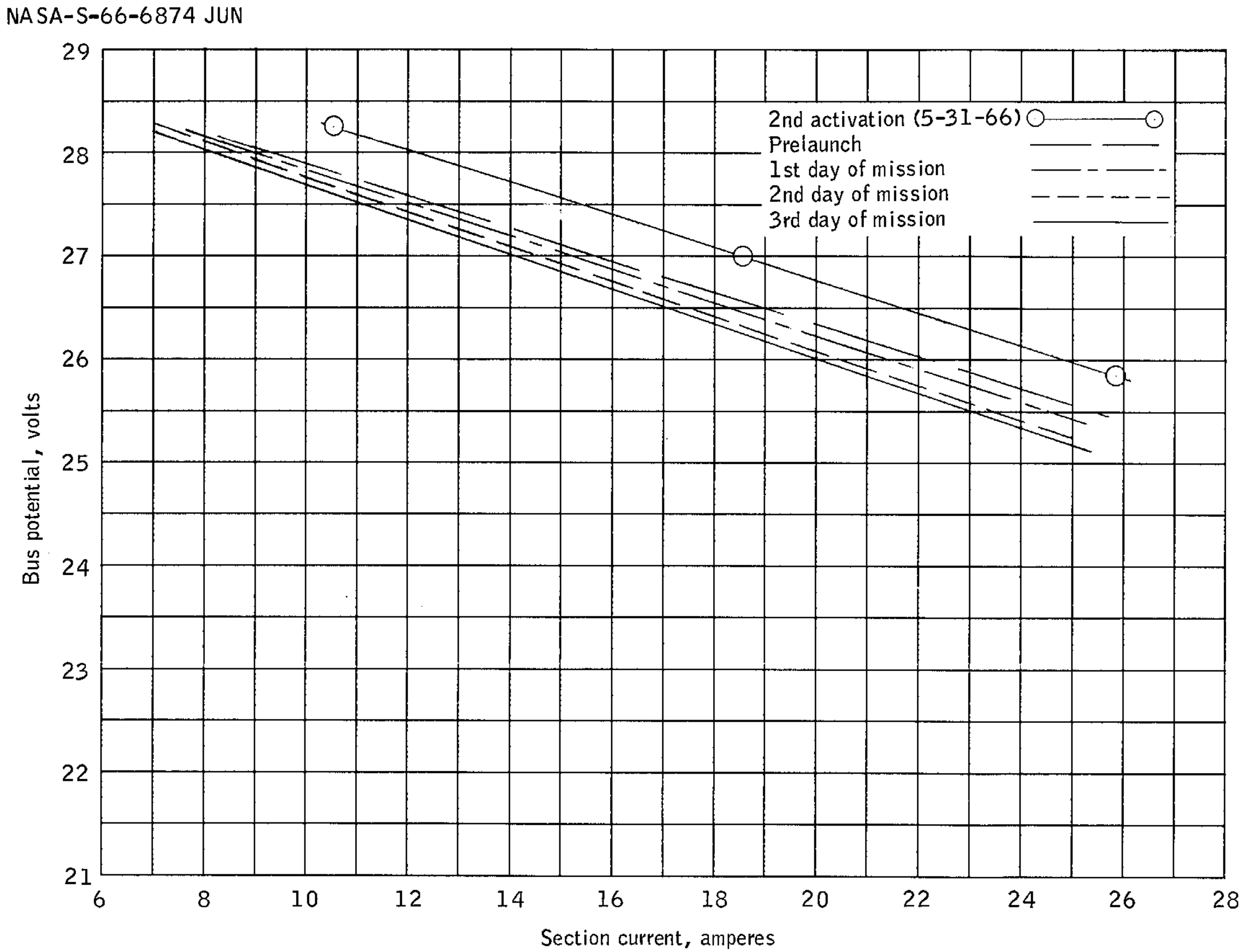

Figure 5.1.7-2.- Fuel-cell section 1 performance (uncorrected for temperature and pressure).

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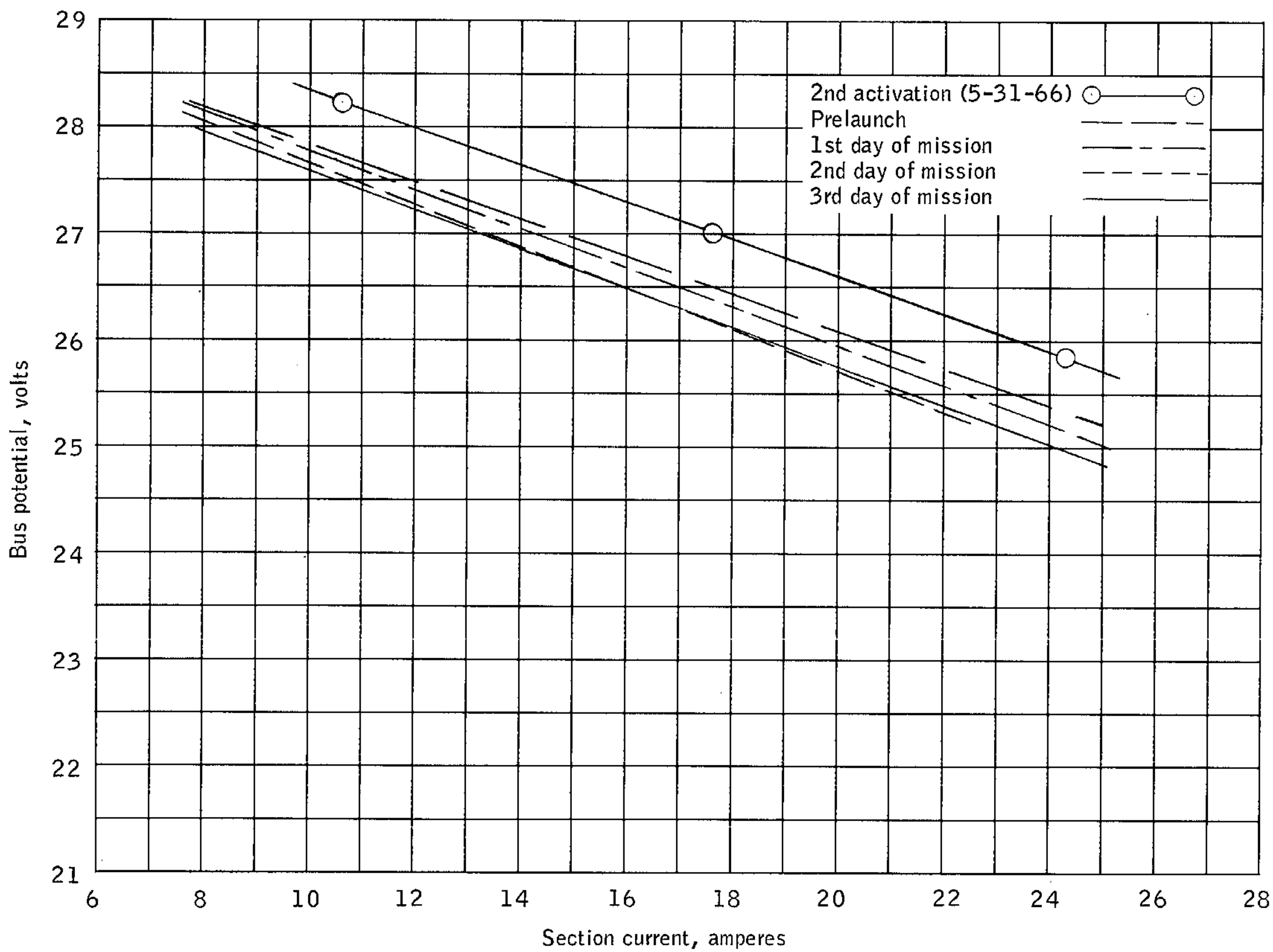

Figure 5.1.7-3.- Fuel-cell section 2 performance (uncorrected for temperature and pressure).

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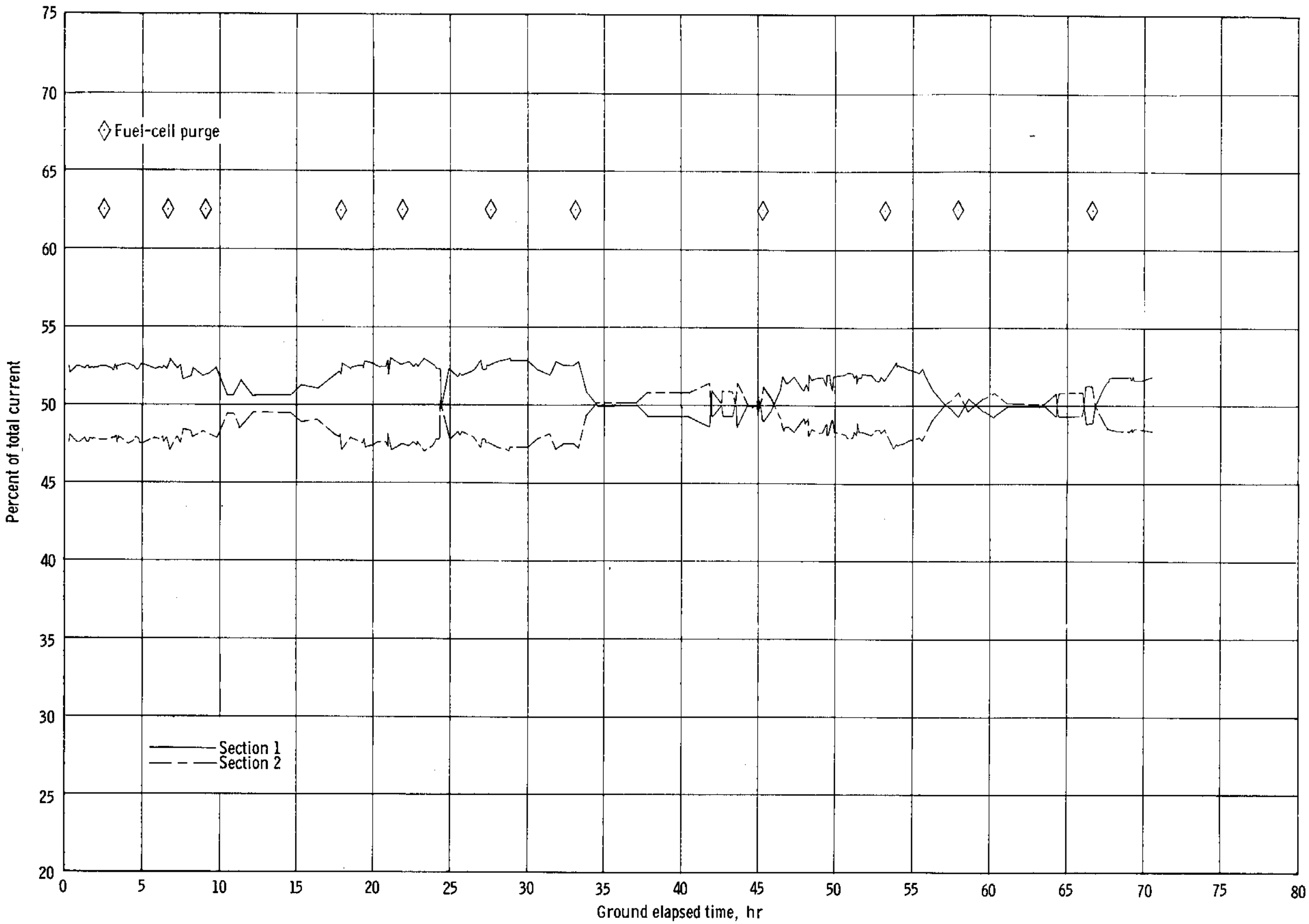

Figure 5.1.7-4. - Load sharing between fuel-cell sections.

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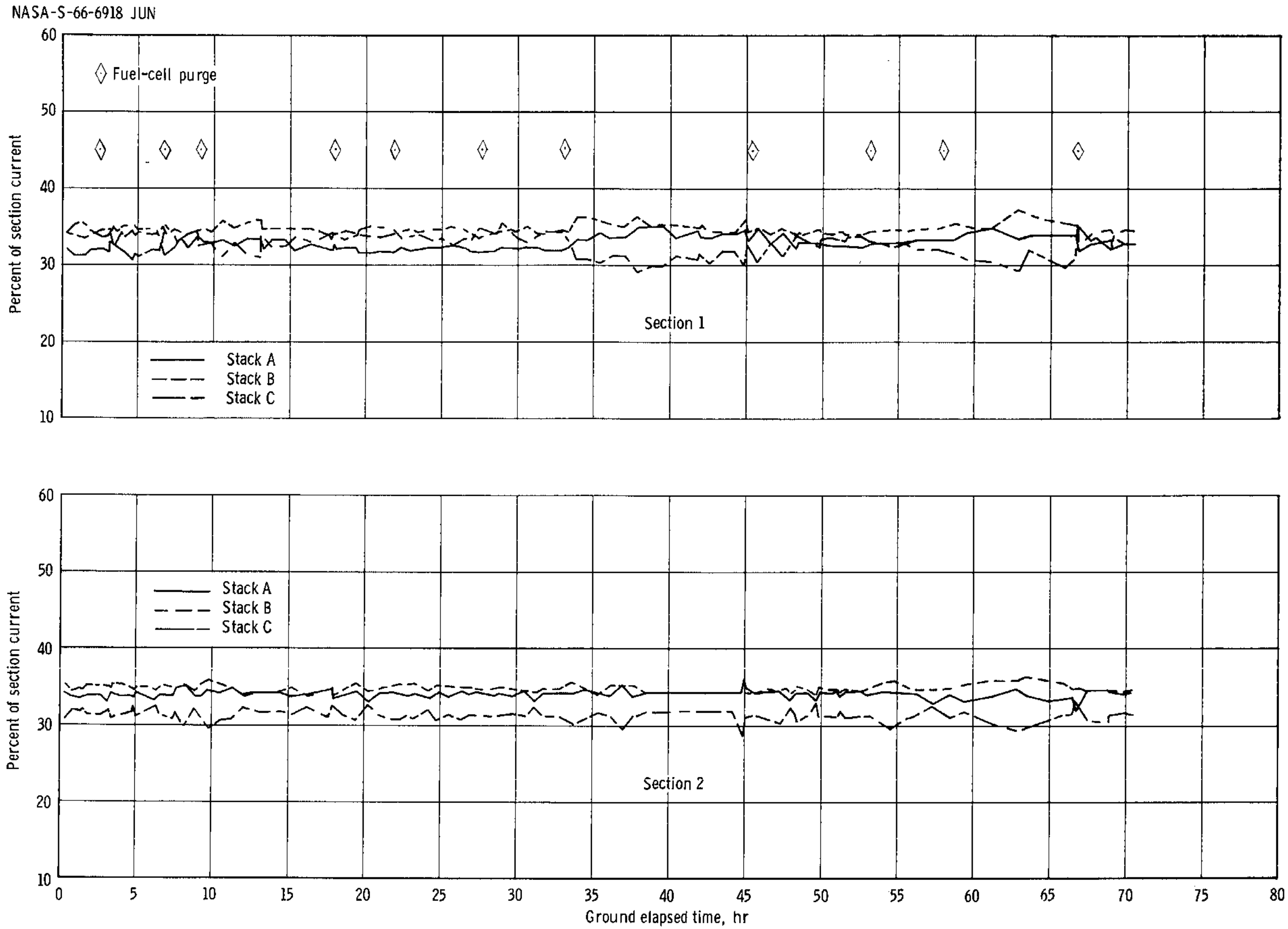

Figure 5.1.7-5. - Load sharing between fuel-cell stacks.

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5.1.8 Spacecraft Propulsion System

Flight performance of the Orbital Attitude and Maneuver System (OAMS) was, in general, satisfactory. The usual period of low thrust during the initial firing of these engines was noted, and short periods of low thrust from some attitude engines was revealed by postflight analysis. There were no detected anomalies associated with the Reentry Control System (RCS), other than the flames observed in several engines after closure of the motor valves, and the apparent high mixture-ratio of the B-ring as revealed by postflight deservicing. The performance of the Retrograde Rocket System was satisfactory.

5.1.8.1 Orbital Attitude and Maneuver System.- Activation of the system occurred approximately 30 minutes before the attempted launch on June 1, 1966. A normal static-fire test was performed before the attempted launch and again prior to the launch on June 3, 1966.

5.1.8.1.1 Maneuver engine performance: The crew reported no problems with the maneuver engines. Based on analysis of the limited data available, their performance was satisfactory. Low thrust transients, similar to those encountered on previous missions, were noted during the first firing of aft-firing engines 9 and 10 and forward-firing engines 11 and 12. It is quite certain that this resulted from trapped gas in the feed lines.

The PCM tape recorder failed in flight with resultant loss of data; therefore, precise firing duration times are unavailable. It is estimated that the total firing time of the maneuver-engines was 940 seconds, most of which was accumulated on the aft engines.

5.1.8.1.2 Attitude engine performance: Examination of control-system commands and rates produced show that the OAMS attitude engines experienced several periods of reduced performance. At various times during the mission, degraded thrust from a single engine of a pair, as evidenced by yaw/roll or pitch/roll coupling, was noted. Apparent reduced thrust was encountered on engines 1, 3, and 4. The problem with engine 4 may have been the result of contamination since the thrust reduction was noted early in the mission (revolution 1) but subsequently cleared. Engine 3 operated at thrust levels between 100 and 0 percent. The data available were insufficient to determine the cause of the noted short periods of degraded thrust. Similar conditions have been noted in earlier missions. It appears that the most serious consequence from such occurrences on a short mission, such as Gemini IX-A, would be slightly higher than nominal propellant usage to correct for engine-coupling effects.

The total firing time of the attitude engines was calculated to be 1870 seconds. Details of each engine firing time are not available because of the failure of the PCM tape recorder.

5.1.8.1.3 Propellant utilization: The total quantity of usable oxidizer and fuel was 336 and 343 pounds, respectively, when referenced to the preflight estimated mixture ratio of 0.98. By the end of the mission, calculations show that the actual mixture ratio was 1.06 and that a total of 700 pounds of propellant were consumed. Fourteen pounds of propellant (at a mixture ratio of 0.7) remained in the system at the time of adapter equipment section separation. In addition to the usable propellant, there were about 7 pounds of entrapped fuel and 50 pounds of entrapped plus unusable oxidizer. The propellant consumed over the duration of the mission is compared with the preflight planned usage rate in figure 5.1.8-1. Included are the mixture ratio variations that were used to establish the flight propellant usage quantities.

5.1.8.2 Reentry Control System.-

5.1.8.2.1 Flight: Reentry Control System (RCS) activation occurred at approximately 71:31:56 g.e.t., and checkout was performed during revolution 44.

The only RCS problem reported by the crew occurred after drogue parachute deployment and closure of the motor valves. A large yellowish flame, about 9 inches long, was observed coming from engine 8 on the A-ring, and engines 3 and 8 on the B-ring. Flames from the number 8 engine continued until landing. As a consequence of the unusual burning, the crew delayed going to the two-point suspension to prevent possible damage to the main parachute bridle. The exact cause of this burning is not known at the present time, but an investigation is in progress.

5.1.8.2.2 Postflight: Information derived from the results of the postflight deservicing is shown in the following table.

	A-ring	B-ring
Source pressure remaining, psia . . .	1245	1925
Usable oxidizer deserviced, lb . . .	1.66	9.03
Usable fuel deserviced, lb	0.78	8.88
Total propellant consumed, lb	31.74	16.73

Based on the propellant remaining in the A-ring, the overall mixture ratio was an acceptable 1.22 as compared with the nominal 1.30. However, based on the propellant remaining in the B-ring, the mixture ratio was 1.64 which is considerably higher than nominal. This apparent high mixture ratio could have been the result of a number of causes which are being investigated.

5.1.8.3 Retrograde rocket system.- All four retrorockets fired nominally in the automatic sequence, following initiation of retrofire at 71:46:43.7.

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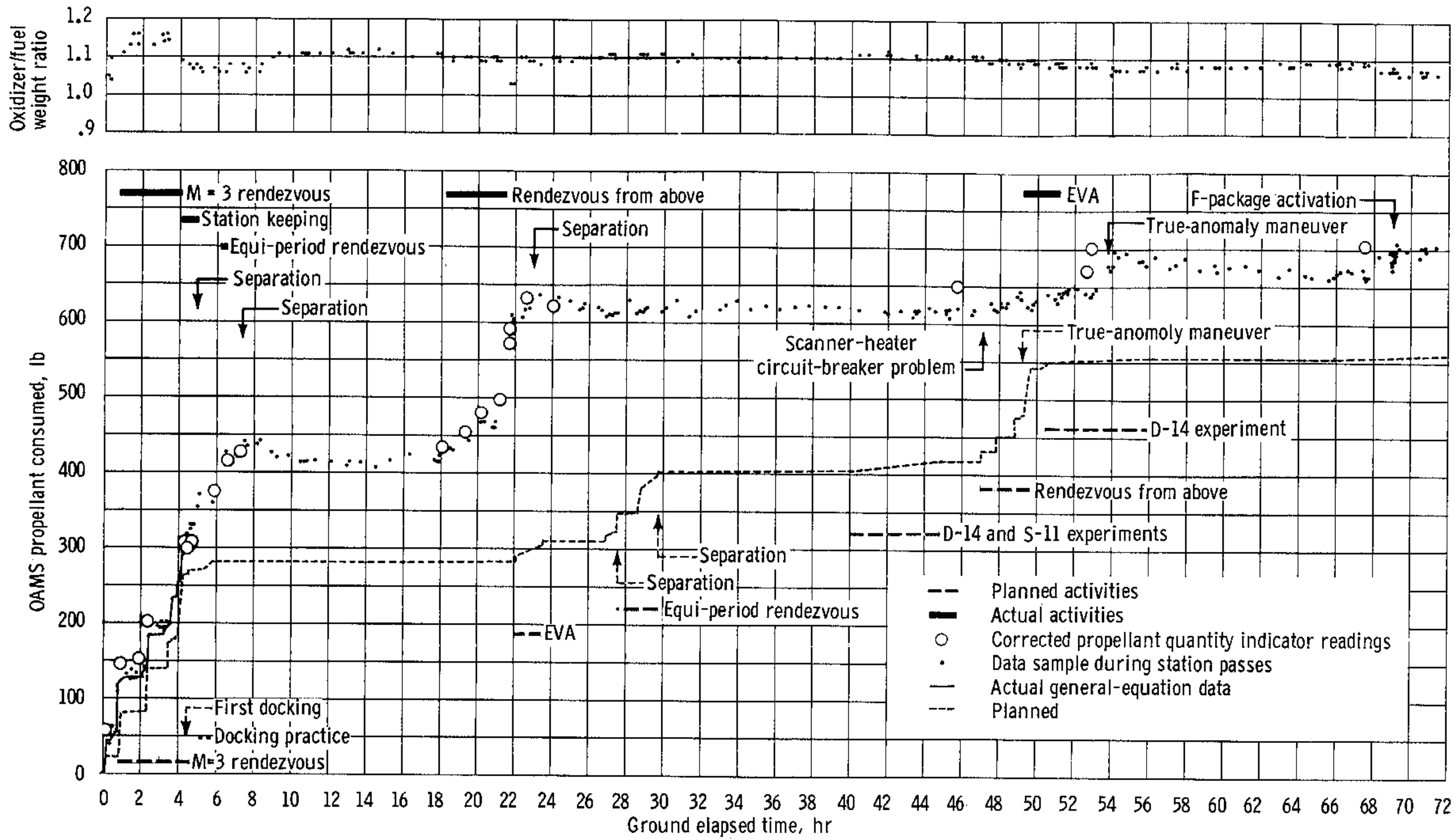

Figure 5.1.8-1. - OAMS propellant consumption.

5.1.9 Pyrotechnics

All pyrotechnics functions were satisfactorily performed. One of the redundant cartridges on the hoist-loop door-release guillotine did not fire; however, the door was properly released by the other cartridge. The cause of this anomaly is under investigation. Examination of the launch vehicle—spacecraft separation plane during EVA showed that the Station Z13 shaped-charge cut was clean and the back-up strips were not retained on the adapter.

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5.1.10 Crew Station

5.1.10.1 Crew-station design and layout.- The crew-station design and layout were satisfactory for the Gemini IX-A mission. Minor problems or noteworthy conditions are described below.

5.1.10.1.1 Displays and controls: The displays and controls functioned normally for this mission. During the period from 1 to 3 minutes after lift-off, the command pilot was unable to see the displays on his instrument panel because of the direct view of the sun in his window. The polaroid window filters were not being used at this time because the attachments for these filters were designed for zero-gravity mounting. A suitable filter is being considered for use during launch.

The fuel-cell power system monitor display of the six stack currents was satisfactory for this mission. There was no requirement for main-bus current display during the mission.

The G.m.t. clock was used only for the stopwatch function, and no reference to G.m.t. was required during the mission. All mission timing was based on ground elapsed time displayed on the digital elapsed-time clock on the center instrument panel.

5.1.10.1.2 Equipment stowage: Equipment stowage provisions were satisfactory for the mission except as discussed below. The door of the centerline stowage area opened and closed freely in orbit. The removal of the rubber bumper below the lower shelf of the centerline stowage area apparently eliminated the unsatisfactory deflection of the shelf which occurred during the Gemini VIII mission.

The stowage provisions for the extravehicular umbilical were satisfactory for launch. After the EVA the crew was unable to pack the umbilical in its stowage bag and, therefore, was unable to restow the umbilical in the left-aft stowage container. The umbilical was held in the right footwell under the pilot's feet for reentry and landing.

5.1.10.1.3 Lighting: The interior cabin lighting was satisfactory for the mission. The docking light on the adapter section was satisfactory for illuminating the target vehicle for station keeping on the dark side. This light did not provide sufficient illumination for seeing the target and judging distance and closing rate during the rendezvous braking maneuvers; however, this is a function for which it was not originally intended.

5.1.10.1.4 Crew furnishings: The ejection seats were not used except for restraint and support of the crew. The ejection-control-mechanism safety pin on the pilot's seat was very difficult to remove and install. The command pilot's seat pin functioned satisfactorily. Postflight examination showed that the safety pin in the pilot's seat was scratched. It is now evident that sharp edges on the hole which receives the pin caused the scratches and also the difficulty in removal and insertion.

The upper surface of the headrest and lower seat contour on the pilot's seat were found to be blistered after the flight. This blistering of the paint was apparently caused by the exposure of the pilot's seat to the sun while the right hatch was open for EVA.

5.1.10.2 Pilots' operational equipment.-

5.1.10.2.1 Optical sight: The optical sight functioned satisfactorily during all three rendezvous operations under daylight and dark-side conditions. The command pilot reported that the optical-sight alignment agreed with the analog display of the radar line-of-sight within 1/4 degree. The intensity and intensity adjustment of the sight image were satisfactory for all the rendezvous lighting conditions except when a very bright earth background was encountered. During the rendezvous from above, the command pilot was unable to see the reticle pattern against a background of sunlit desert. However, the target was also not visible at this time. The optical-sight image was visible against a cloud background soon thereafter. The crew reported that the optical sight was effective in estimating range and range rate during the visual phase of the rendezvous approach. The optical sight was also reported to be useful during the braking maneuvers.

5.1.10.2.2 Miniature sextant: The miniature sextant was used for target-to-horizon measurements during the second rendezvous. The crew reported that the maximum usable angle on this sextant was 70 to 75 degrees. The crew also reported that the use of the sextant was time consuming because of the time to acquire the target and the need to hold the sextant under the light to read the numbers on the dial. No attempt was made to evaluate the high magnification eyepiece nor to use the sextant optics to estimate range rate.

5.1.10.2.3 Still cameras: A new configuration 70-mm general-purpose camera was used for the first time in orbit on this mission. The camera operated satisfactorily, and numerous high-quality, high-resolution photographs were obtained. During EVA, the command pilot used this camera in a pressurized space suit without difficulty. The camera's low profile facilitated its use between the command pilot's space-suit visor and the window, a distance of less than eight inches. The camera was also used successfully for experiment photography.

The 70-mm super-wide-angle camera was also used during EVA. The back of the camera was mounted on the front of the ELSS using Velcro. Several excellent photographs were obtained from outside the spacecraft; however, some of the exposures had light streaks.

5.1.10.2.4 Sequence cameras: Two 16-mm sequence cameras were used on this mission. One of these cameras was mounted in the left window for rendezvous, station keeping, and general orbital photography. The other camera was mounted outside the spacecraft during EVA for photographing the pilot. The cameras were also mounted in the two windows to obtain reentry photographs down to approximately 100 000 feet. Seventeen 16-mm magazines were exposed during the flight; however, the magazine from the external EVA camera was inadvertently lost in orbit during ingress. The magazine taken of EVA from inside the cabin was underexposed and therefore poor in quality. All other 16-mm photography was satisfactory.

The 16-mm sequence camera used for external EVA photography was stuck in the six-frames-per-second mode after EVA. This discrepancy was corrected later when the pilot tapped the camera with his hand.

5.1.10.2.5 Utility cords: One of the two dual electrical utility cords failed open during the flight. A similar failure occurred on the Gemini V mission. A modified utility cord incorporating strain relief in the electrical leads was incorporated after the failure in Gemini V, but it is now apparent that this corrective action was not entirely adequate. The crew shifted electrical cords in the cabin and completed all required electrical functions involving use of these cords.

5.1.10.2.6 Water-metering device: During the third day of the mission, the crew reported that the trigger on the water-metering device could not be moved to the full-up position, and the counter was not actuating properly. Subsequently, the water-metering device operated normally until the last few orbits of the mission. Postflight testing of the water-metering device showed it to perform in a completely normal manner. It was found, however, that the condition described by the crew could be duplicated by placing the trigger lock in a partially locked position. This was probably the cause of the problem. Just prior to reentry, the crew discovered that the flow of water was reduced to less than one third of normal. The postflight investigation indicated that this latter problem was due to water depletion (see section 5.1.4).

5.1.10.3 Pilots' personal equipment.-

5.1.10.3.1 Food: The crew consumed approximately 1/2 of the food carried in the spacecraft. It was reported to be satisfactory for the mission. The food was eaten when activities and time permitted rather than on a predetermined schedule. No problems were reported concerning preparation of the food.

During EVA, the food was stowed in the left footwell and was exposed to a vacuum for more than two hours with no detrimental effects.

5.1.10.3.2 Waste equipment: The chemical urine-volume measuring system was the only urine collection system used during the mission. It operated normally throughout the mission, although the crew reported objectionable back pressure in the urine receiver. There was no significant urine leakage or spillage during the mission. A second urine system, which did not incorporate a volume measuring capability, was carried but not used on the mission.

The launch-day urine bags were dumped early in the mission and jettisoned during EVA. The crew reported that the clamps used to seal the launch-day urine bags were not strong enough to accomplish the intended task. Postflight testing will be conducted on like items, because the flight items were jettisoned in orbit.

The defecation equipment was not required during this mission.

5.1.10.4 Space suits.-

5.1.10.4.1 Command pilot's space suit: The space suit worn by the command pilot functioned normally during the entire mission. The only anomalies were a small opening in the thumb seam of the right glove and corrosion in one of the latching dogs in the right wrist disconnect fitting. The seam opening had no significant effect on the structural integrity of the glove; however, the glove has been returned to the space suit contractor for failure analysis. The corrosion in the wrist disconnect was probably caused by postflight handling procedures which may not have directed proper suit venting and drying before shipment. Neither of these discrepancies had any effect on the mission.

5.1.10.4.2 Pilot's space suit: The space suit worn by the pilot operated satisfactorily for the intravehicular phases of the mission (see section 5.1.10.5.2). The leak rate of the suit after the mission was 11 scc/min as compared with the allowable value of 1000 scc/min. During the rendezvous phase of the mission, the pilot worked almost continuously reading books and charts and recording data. He reported that, with his helmet removed, the space suit neck ring restricted the downward movement of his head when attempting to see the books and charts. This condition is considered to be principally a problem of zero gravity rather than space suit design. Similar comments have been made by previous crews.

5.1.10.5 Extravehicular equipment.-

5.1.10.5.1 Extravehicular Life Support System: The Extravehicular Life Support System (ELSS) was used for the first time on this mission. It was onboard the spacecraft for Gemini VIII mission but was not used because of the early termination of the mission.

The ELSS performed normally during the EVA preparation period from 48 hours 10 minutes ground elapsed time (g.e.t.) to cabin depressurization at 49 hours 19 minutes g.e.t. The ELSS continued to perform normally in the medium flow rate from the time of hatch opening at 49 hours 23 minutes g.e.t. until just before the end of the first daylight period during EVA. The pressure in the space suit remained steady at 3.7 psia and the pilot reported being comfortable. At approximately 50 hours 18 minutes g.e.t., the pilot shifted to the high flow rate on the ELSS because of hot spots on his back (see paragraph 5.1.10.5.2). At approximately 50 hours 28 minutes the pilot's visor began to fog. This was about 8 minutes after local sunset and followed a period of particularly high workload resulting from the pilot's attempts to connect the AMU tether hooks and lower the AMU controller arms. Throughout the remainder of the night period, the ELSS was operated on high flow in an attempt to clear the visor. Because of the visor fogging, the crew terminated the AMU evaluation.

The pilot reported that he was neither cool nor hot and that his only problem was visor fogging. After resting, the visor fogging began to clear gradually during the second daylight EVA period. At 51 hours 16 minutes g.e.t., the pilot's visor was 60 percent clear. At this time, he retrieved the docking-bar mirror, and the added workload caused the visor fogging to increase. Ingress to the cabin occurred at 51 hours 21 minutes g.e.t. and produced heavy fogging. When the hatch was closed at 51 hours 28 minutes g.e.t. the pilot's visor was completely fogged over again. After locking the hatch and repressurizing the cabin at 51 hours 42 minutes g.e.t., the pilot was perspiring very profusely and was noticeably overheated. The interior of his space suit was soaking wet, and portions of the ELSS suit loop had become saturated with water. The ELSS problems are discussed in section 5.1.4.

5.1.10.5.2 Extravehicular space suit: The G4C-32 space suit worn by the pilot performed normally in all respects during the extravehicular activities except for the visor fogging, overheating in the back area, and fogging of the pressure gage.

(a) Visor fogging: The inner visor of the G4C-32 space suit was made of a polycarbonate material with a low-emittance coating on the outer surface. When the visor was examined after the flight, substantial areas of the low-emittance coating had been rubbed off. The pilot indicated that he wiped the visor to clean it prior to reentry, but that he had inspected it and verified that the coating was not disturbed prior to EVA. Postflight testing of the area where the low-emittance coating was not rubbed off, showed that the coating was well within specifications. Calculations based on the known properties of the visor indicate that the inner visor temperature was probably between 75° F and 80° F at the time fogging occurred.

With the ELSS set for high-flow rate, approximately 186 liters of oxygen per minute are delivered to the visor area. With a respiration rate of 40 breaths per minute, at which the pilot was breathing just prior to the time of fogging, the inspired volume would be approximately 140 liters per minute. This respiration rate would consume a large portion of the oxygen being supplied to the visor area, and would result in an estimated dew point between 70° and 80° F. The workload and the respiration rate of the pilot apparently exceeded the combined capabilities of the ELSS and the space-suit ventilation system.

(b) Localized overheating: At 50 hours 18 minutes, the pilot reported extreme heating in a localized area in the small of his back. Postflight inspection of the thermal coverlayer of the G4C-32 space suit showed that the superinsulation had parted from the seam adjacent and parallel to the zipper in the small of the back. This separation resulted from a repair made at the suit contractor's factory in the final month before flight. The separation was noted at the time of repair, and the edges of the superinsulation were repaired with aluminized Mylar tape but were not restitched. Without being stitched in place, the superinsulation moved away from the zipper seam and created a thermal short in the back of the suit.

The design of the superinsulation and its supporting structure is being reviewed to ensure that present fabrication techniques do not introduce excessive structural loading in the insulation layers.

(c) Pressure gage fogging: The pilot reported fogging of the pressure gage on the left arm of the space suit. Postflight inspection indicated the presence of water in the gage face and cover area. The face of the gage is not sealed from the space suit interior, and fogging has occurred during ground testing. However, the fogging never became so severe that the gage was unreadable. Such was also the case in flight. Although the pilot could not read the markings on the gage, he could see the needle sufficiently to know that the pressure indication had not changed throughout the EVA. A review of the design is being made to determine the feasibility of isolating the face of the gage from the environment within the suit.

5.1.10.5.3 Spacecraft provisions: The spacecraft provisions for EVA included the adapter handrails, the adapter footbar and handbars, the EVA lights, the hatch closing aids, the nose tether-attach point, the camera mount, and the adapter umbilical guide and clip.

The adapter handrails provided a satisfactory handhold for the pilot when he was in transit from the reentry assembly to the aft edge of the adapter. While moving along the handrails, the pilot moved sideways rather than hand-over-hand. Because there were no restraints for the feet, the pilot's legs tended to point away from the spacecraft. This body attitude did not interfere with movement along the handrails. The pilot reported that lack of handrails on the reentry assembly made transit along the surface to the nose of the spacecraft more difficult than on the adapter.

The adapter interior handbars and footrail were unsatisfactory for maintaining a stable body position while the pilot was attempting to prepare the AMU for donning. The stirrups that were added to the footbar did not restrain the pilot's feet adequately to permit him to use both hands as he had in training. The most demanding two-handed tasks were connecting the AMU tether hooks to the pilot's tether jumper, and lowering the AMU attitude controller arm. These devices were evaluated during a number of zero-g aircraft tests and were satisfactory. The problems encountered during the mission were probably caused by the differences between long-term zero-g conditions and short-term (20 seconds) operations in the zero-g aircraft. The footrail and handbars were satisfactory for AMU donning after the EVA pilot backed into the AMU. The configuration of the footrail and handbars gave adequate support for two-handed operation under this condition.

The external adapter surface light and the adapter internal lighting were adequate for darkside operations, except that the light on the upper handbar failed to illuminate. The lack of this light was not serious because one of the portable penlights was used in place of the failed light. On the basis of this mission, the pilot indicated that the external adapter surface lights were adequate for egress from the cabin during the darkside operation.

The hatch holding device performed as planned. The hatch closing lanyard was satisfactory during the final portion of hatch closure but did not provide assistance to the crew during the intermediate portions of hatch travel. The command pilot attempted to assist the pilot in closing the hatch. However, the hatch was still difficult to move through the midtravel range. See section 5.1.1 for an analysis of the hatch opening and closing problems.

The nose tether-attach point was not used in the Gemini IX-A mission. The umbilical clip in the adapter section did not restrain the umbilical longitudinally. The use of a more positive holding feature on this clip is being studied for future missions.

The pilot reported that the extravehicular camera mount was difficult to install. The pilot had to hit the camera mount to get it to lock in place on the spacecraft adapter fitting. This was duplicated postflight on a similar mount by overtightening the detent adjust screw.

5.1.10.5.4 Miscellaneous EVA equipment: The 25-foot umbilical performed satisfactorily during the entire EVA operations. It imparted little or no force on the pilot while he was free of the spacecraft. It was usable out to a distance of 25 feet for the function of returning to the spacecraft. The umbilical was totally ineffective for maneuvering in any direction other than toward the spacecraft.

The pilot reported that the Velcro hand pads were unsatisfactory because the pads were not stiff enough and the straps were too susceptible to coming off his hands. The pilot was able to complete sufficient evaluation to establish that the inability to introduce body torques using the Velcro pads was a substantial handicap.

During the EVA operations, the pilot reported that the ELSS was riding up on his chest and causing helmet interference. This condition indicates the ELSS restraint straps were not holding the ELSS in the proper position — centered on the pilot's chest.

Attaching and removing the 16-mm camera thermal cover was time-consuming. The flight crew indicated the need for an improved thermal cover for any future mission where such a cover is used.

One of the penlights stowed in the AMU tether bag for use in the adapter was found to be inoperative. The defective penlight was discarded in orbit, and because it was not recovered, the exact cause of failure is unknown. The qualification test results for this light are being reviewed to determine whether design weakness could have caused this failure.

5.1.10.6 <u>Bioinstrumentation</u>.- The bioinstrumentation equipment performed satisfactorily during this mission, and satisfactory biomedical data were obtained on both pilots.

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5.1.11 Landing System

The parachute landing system operated satisfactorily, with all system events occurring when commanded by the flight crew and within established tolerances. Figure 5.1.11-1 illustrates the occurrence of the major events with respect to ground elapsed time and pressure altitude.

The crew reported that they experienced much higher forces than they expected when the spacecraft landed. The data indicate that the parachute system functioned within design limits; therefore, it is concluded that these forces resulted from the combination of wind drift, normal swing on the parachute, and contact of the spacecraft on the bad side of a wave. (For further information, refer to section 5.1.1.) Photographs and motion pictures of the spacecraft shortly before landing also indicate that the parachute system was functioning properly. The canopy exhibited a good shape, and the spacecraft was in the correct attitude for landing.

The main parachute was recovered; however, the rendezvous and recovery section, with the attached drogue and pilot parachutes, sank and could not be recovered. Postflight evaluation of the main parachute revealed it to be in excellent condition. Examination of all other landing-system components confirmed satisfactory operation.

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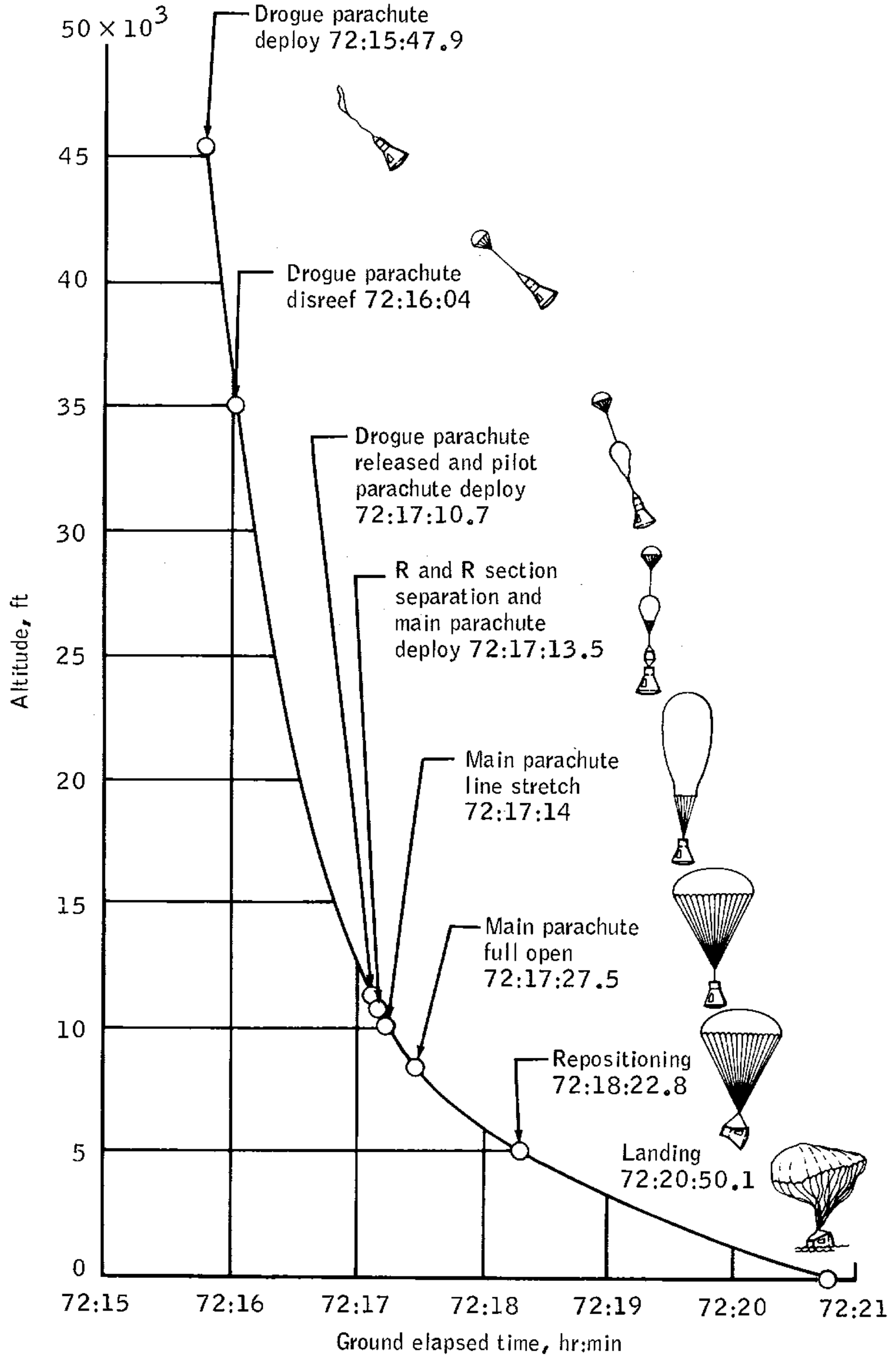

Figure 5.1.11-1. - Landing system performance.

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 crew. The crew did not attempt to extend the HF antenna. 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).

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5.2 GEMINI LAUNCH VEHICLE PERFORMANCE

The Gemini Launch Vehicle (GLV) was launched on time after a countdown that involved no unplanned holds. All systems performed satisfactorily, and, although the flight-path angle and velocity at insertion were slightly below those desired, they were within the predicted limits and a satisfactory orbital insertion of the spacecraft was achieved.

Real-time calculations, performed during the countdown, indicated that the nominal payload capability would exceed the spacecraft weight by 380 pounds, and computations of minimum capability (minus 3 sigma) predicted a payload margin of minus 235 pounds. Postflight-reconstructed burning-time margin was +1.71 seconds, indicating that the achieved vehicle performance was equivalent to 8881 pounds, 613 pounds over the spacecraft weight, or 233 pounds more than the predicted nominal capability.

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5.2.1 Airframe

Flight loads on the launch vehicle were well within its structural capability and comparable to loads experienced on previous missions. The vibration and acceleration environment was comparable to previous flights with the exception of longitudinal oscillation (POGO) which had the lowest amplitude experienced in the Gemini Program.

5.2.1.1 Structural loads.- Ground winds of approximately 20 miles per hour induced prelaunch lateral oscillations which caused a maximum bending moment on the Gemini Launch Vehicle equal to 32 percent of the design-limit wind-induced bending moment.

Estimated loads on the launch vehicle during the Gemini IX-A mission are shown in the table below. The loads are related to design loads of spacecraft in the weight range from 8000 pounds to 8500 pounds. These data indicate that the highest percentage of design loading occurred at station 1188 during the maximum-$q\alpha$ region of flight, as shown in the following table.

Station	Maximum $q\alpha$			Pre-BECO		
	Load, lb	Percent of design		Load, lb	Percent of design	
		Limit	Ultimate		Limit	Ultimate
276	35 880	42.7	34.2	48 290	57.5	46.0
320	154 100	50.2	40.2	269 580	88.1	70.5
935	460 370	76.8	61.4	440 690	73.5	58.8
1188	516 370	96.2	77.0	455 950	85.0	68.0

5.2.1.2 Longitudinal oscillation (POGO).- Data indicated the same intermittent characteristic of the suppressed longitudinal oscillation that had been experienced on previous flights. Maximum response at the spacecraft-launch vehicle interface occurred at lift-off (LO) + 121.8 seconds, with an amplitude of ±0.11g and a corresponding frequency of 10.9 cycles per second on filtered data. This is the lowest level of POGO experienced to date.

5.2.1.3 Post-SECO disturbance.- There were two minor indications of disturbances on the low-range accelerometer data after Stage II engine cutoff (SECO). The first disturbance occurred at SECO + 10.8 seconds with an amplitude of 0.04g peak-to-peak. The second disturbance occurred at SECO + 16.7 seconds with an amplitude of 0.05g peak-to-peak. These amplitudes are comparable to those experienced on previous flights.

5.2.2 Propulsion

5.2.2.1 Engines.-

5.2.2.1.1 Stage I: Performance of the Stage I engine was essentially normal throughout the start, steady state, and shutdown phases of flight. Two deviations from nominal are briefly discussed below.

(1) The Stage I mixture ratio at standard inlet conditions was outside of the 3-sigma run-to-run repeatability of ±1.38 percent, as may be seen in table 5.2.2-I. This off-nominal condition led to a fuel-depletion shutdown.

(2) The difference between the subassembly 1 and subassembly 2 start transients, shown in figure 5.2.2-1, was due to chamber-pressure transducer-response differences. The difference in transducers attenuates the indications of the real spike pressures which were between the 520 psia and the 710 psia recorded. The time from the Stage I ignition signal to bootstrapping was somewhat shorter than had been experienced in Gemini; however, this did not reflect as a hard start.

Steady-state operation (thrust and specific impulse) of the engine was close to predicted, as can be seen from a comparison of the planned and the actual flight-average data shown in figure 5.2.2-2 and in table 5.2.2-I.

5.2.2.1.2 Stage II: The Stage II engine-start transient appeared to have an abnormally slow chamber-pressure $\left(P_c\right)$ rise. A similar indication was present in the data from GLV-5 and GLV-8 and is believed to be caused by moisture freezing in the P_c sensing line or transducer cavity during Stage I flight. After staging telemetry blackout, normal P_c operation was indicated by these data.

Stage II engine steady-state performance was satisfactory throughout flight. Table 5.2.2-II and figure 5.2.2-3 show the planned and actual engine performance at standard-inlet and flight-average conditions. Good agreement with planned values can be noted.

The Stage II engine-shutdown transient was a normal command-type shutdown with a P_c profile (fig. 5.2.2-4) and shutdown impulse very close to that of GLV-8. The table below presents predicted and actual shutdown impulses for GLV-8 and GLV-9.

Flight	Predicted, lb-sec	Actual, lb-sec
Gemini VIII	36 100 ± 7000	35 535
Gemini IX-A	36 100 ± 7000	35 422

5.2.2.2 Propellants.-

5.2.2.2.1 Loading: The results of the three GLV-9 propellant loadings are presented in table 5.2.2-III. All loadings were made within the required accuracy of ±0.35 percent. The change in the Stage II requested values between launch attempts was due to a change made in the predicted Stage II inflight suction conditions. This required a reevaluation of the Stage II engine performance and a small revision to the Stage II propellant loading.

The launch-attempt actual values were derived from flowmeter readings corrected for actual flowmeter temperatures during loading. The actual flight values are the result of a propellant-load reconstruction using GLV-9 engine performance and level-sensor data.

5.2.2.2.2 Utilization: Outages, both predicted and actual, are shown in the following table. Stage I oxidizer outage is the amount of usable oxidizer remaining after the fuel-depletion shutdown. Stage II oxidizer outage is the amount of usable oxidizer which would have remained if a command shutdown had depleted all of the usable fuel.

Engine	Predicted mean, lb	Predicted maximum, lb	Actual, lb
Stage I	563	1645	664 oxidizer
Stage II	209	621	108 oxidizer

The amount of propellants remaining at Stage II engine shutdown could have sustained Stage II flight an additional 1.71 seconds. This is 0.42 seconds greater than the predicted nominal burning-time margin of 1.29 seconds at Stage I engine ignition, indicating a higher than nominal overall propulsion-system performance.

5.2.2.3 Pressurization.- The predicted and actual GLV-9 tank pressures for various flight times are given in tables 5.2.2-IV and 5.2.2-V. The close agreement to the actual pressures indicates nominal performance of the pressurization system.

TABLE 5.2.2-I.- STAGE I ENGINE PERFORMANCE

Parameter	Preflight prediction	Postflight reconstruction	Percent difference
Standard inlet condition performance			
Thrust, lb	435 450	435 631	+0.03
Specific impulse, lb-sec/lb	259.54	259.97	+0.17
Engine mixture ratio, oxidizer to fuel	1.9563	1.9275	-1.47
Oxidizer flow rate, lb/sec	1109.89	1102.96	-0.62
Fuel flow rate, lb/sec	567.87	572.75	+0.86
Flight average performance			
Thrust, lb	461 353	461 137	-0.05
Specific impulse, lb-sec/lb	276.21	276.95	+0.27
Engine mixture ratio, oxidizer to fuel	1.9389	1.9147	-1.25
Oxidizer flow rate, lb/sec	1101.61	1093.44	-0.74
Fuel flow rate, lb/sec	568.68	571.61	+0.52
Burning time, sec	155.74	155.63	-0.07

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TABLE 5.2.2-II.- STAGE II ENGINE PERFORMANCE

Parameter	Preflight prediction	Postflight reconstruction	Percent difference
Standard inlet condition performance[a]			
Thrust, lb	101 920	102 425	+0.50
Specific impulse, lb-sec/lb	311.11	311.82	+0.23
Engine mixture ratio, oxidizer to fuel	1.8365	1.8011	-1.93
Oxidizer flow rate, lb/sec	210.53	209.63	-0.43
Fuel flow rate, lb/sec	114.38	116.14	+1.54
Flight average performance[b]			
Thrust, lb	100 525	101 136	+0.61
Specific impulse, lb-sec/lb	311.97	312.39	+0.13
Engine mixture ratio, oxidizer to fuel	1.7939	1.7752	-1.04
Oxidizer flow rate, lb/sec	207.06	207.25	+0.09
Fuel flow rate, lb/sec	115.17	116.50	+1.15
Burning time, sec	187.58	187.31	-0.14

[a]Includes roll control thrust.

[b]Does not include roll control thrust.

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TABLE 5.2.2-III.- GLV-9 PROPELLANT LOADING SUMMARY

Date, 1966	Tank	Requested, lb	Actual, lb	Percent change
May 16	Stage I oxidizer	171 077	171 094	+0.01
	Stage I fuel	88 896	88 937	+0.05
	Stage II oxidizer	39 165	39 162	-0.01
	Stage II fuel	22 056	22 057	0
May 31	Stage I oxidizer	171 077	171 080	0
	Stage I fuel	88 896	88 937	+0.05
	Stage II oxidizer	39 222	39 236	+0.04
	Stage II fuel	22 004	22 026	+0.10
June 2	Stage I oxidizer	171 077	170 908	-0.04
	Stage I fuel	88 896	88 901	+0.01
	Stage II oxidizer	39 222	39 280	+0.15
	Stage II fuel	22 004	22 081	+0.35

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TABLE 5.2.2-IV.- STAGE I ULLAGE GAS PRESSURE

Tank	Lift-off - 3.14 sec		Lift-off + 47 sec		Lift-off + 97 sec		Lift-off + 147 sec	
	Predicted, psia	Actual, psia	Predicted, psia	Actual, psia	Predicted, psia	Actual, psia	Predicted, psia	Actual, psia
Oxidizer	32.5	32.8	19.5	20.2	18.3	19.2	19.8	20.0
Fuel	29.5	29.9	23.0	23.3	22.6	23.0	23.2	23.9

TABLE 5.2.2-V.- STAGE II TANK ULLAGE GAS PRESSURE

Tank	Lift-off + 152.48 sec (staging)		Lift-off + 192 sec		Lift-off + 242 sec		Lift-off + 292 sec	
	Predicted, psia	Actual, psia	Predicted, psia	Actual, psia	Predicted, psia	Actual, psia	Predicted, psia	Actual, psia
Oxidizer	55.5	55.6	20.6	20.9	12.7	13.1	9.3	9.9
Fuel	50.5	51.2	48.6	48.3	48.5	48.2	48.2	48.4

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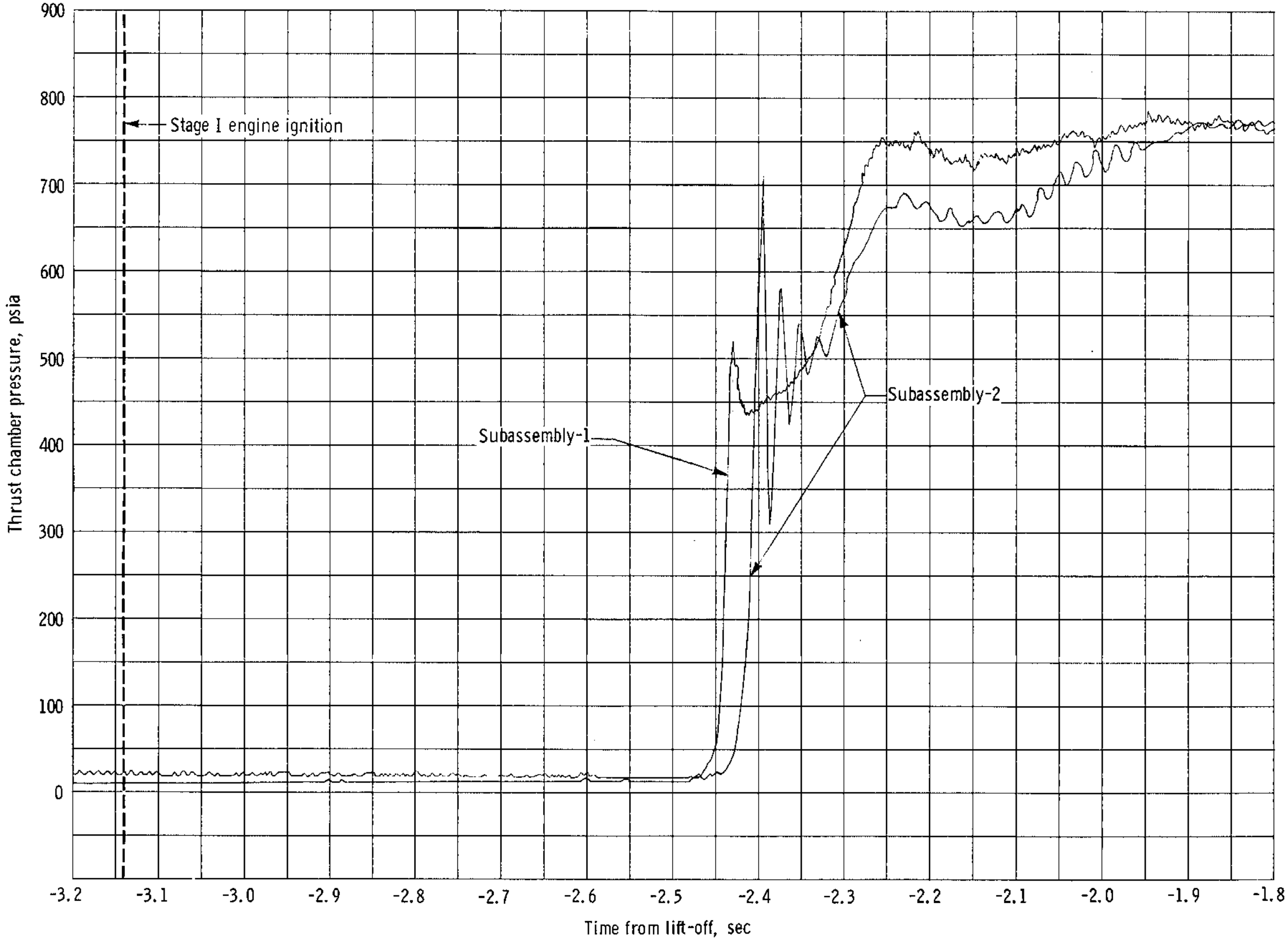

Figure 5.2.2-1. - Stage I engine start transient.

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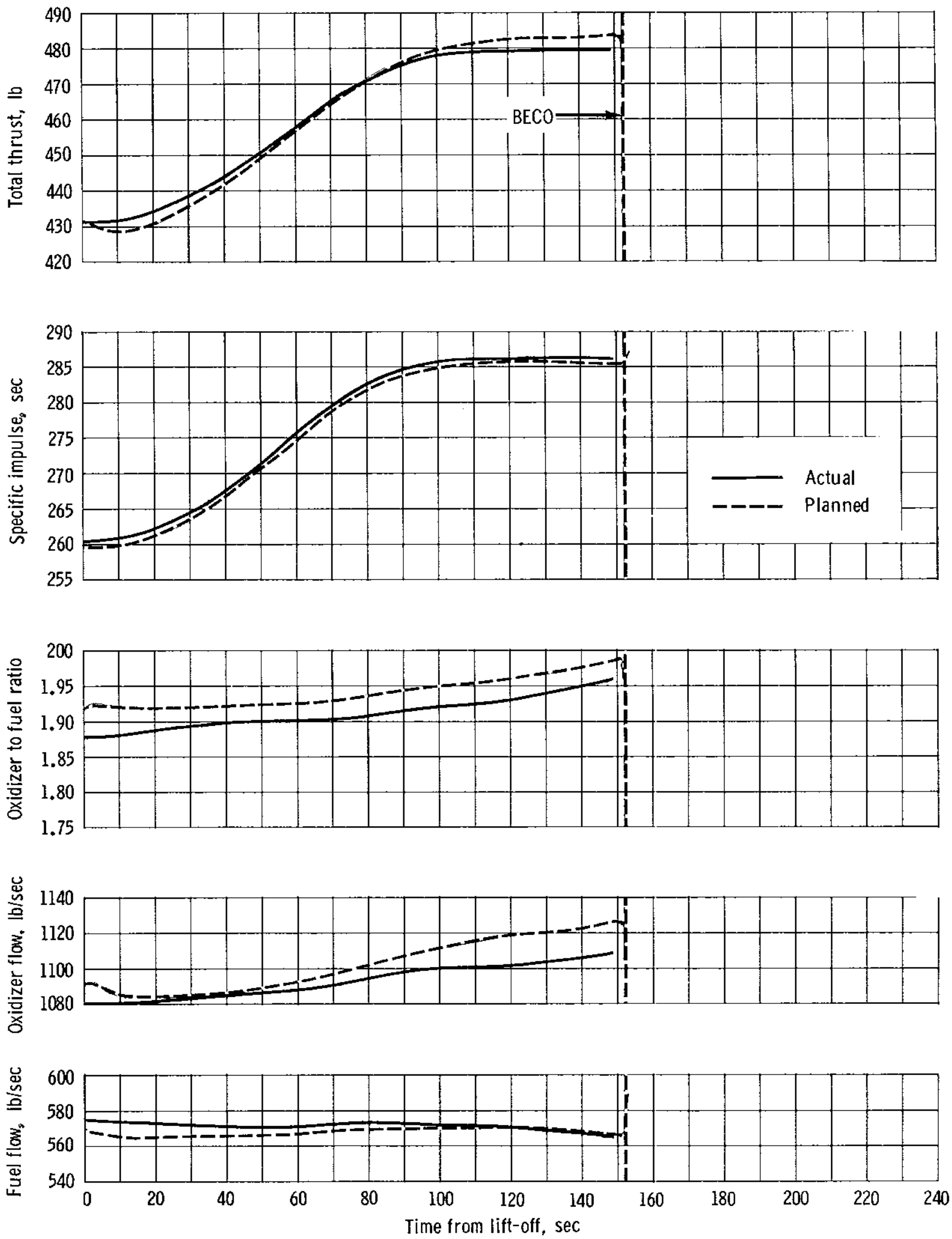

Figure 5.2.2-2. - Stage I engine performance.

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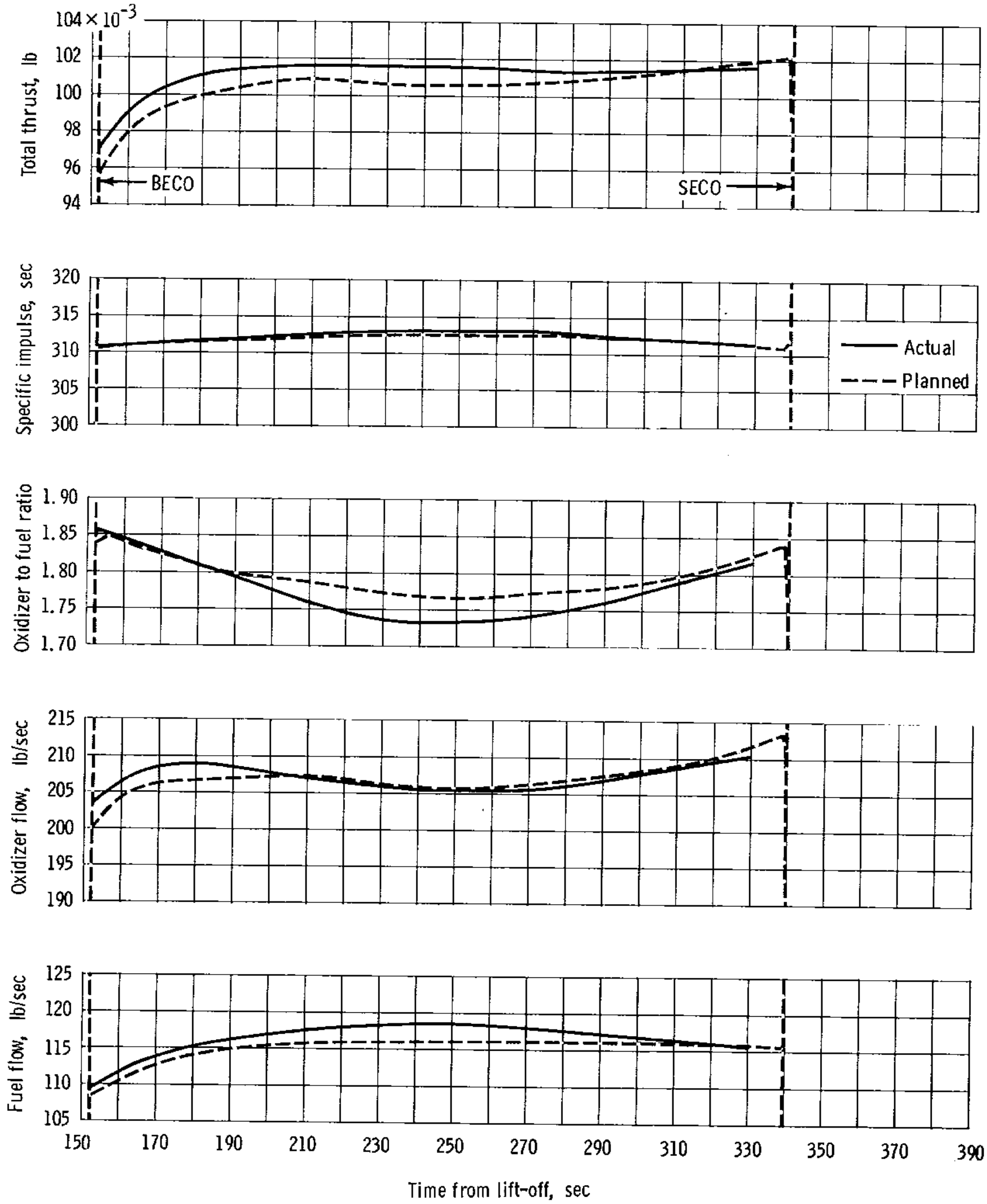

Figure 5.2.2-3. - Stage II engine performance.

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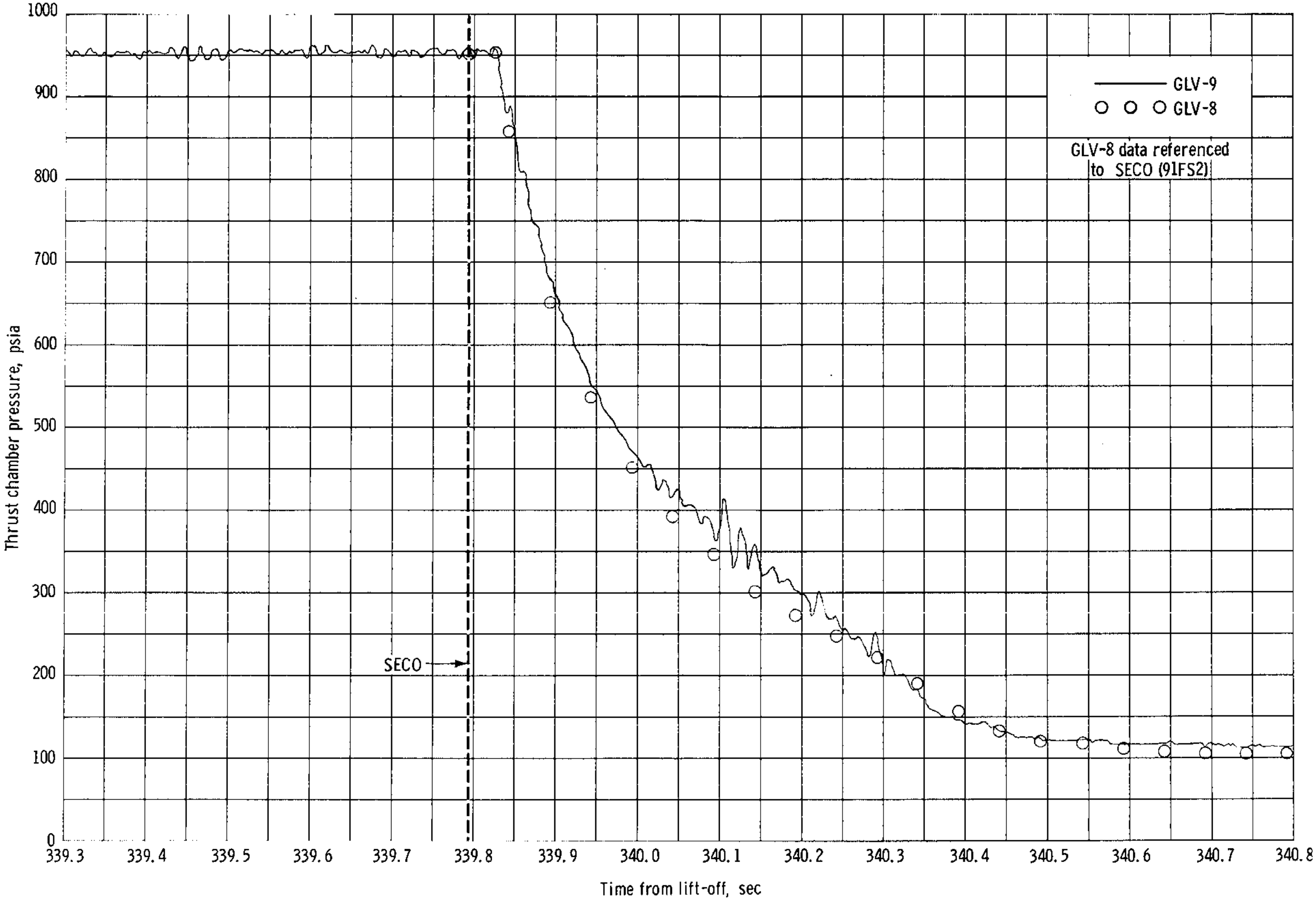

Figure 5. 2. 2-4. - Stage II engine shutdown transient.

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5.2.3 Flight Control System

Performance of the flight control system was satisfactory from lift-off to spacecraft separation. No flight control anomalies were encountered throughout powered flight. The primary flight control system in conjunction with the GE MOD III guidance system inserted the spacecraft into an acceptable orbit for the mission. The secondary guidance system performed satisfactorily; therefore, switchover could have been successfully accomplished at any time during the powered phase of flight.

5.2.3.1 Stage I flight.- The response to ignition transients was normal. Values of peak actuator travel recorded during the ignition and holddown period are listed in the following table.

Actuator	Maximum travel		
	Maximum during ignition		Maximum during null check, in.
	Maximum travel, in.	Time from lift-off, sec	
Pitch, 1_1	-0.06	-2.40	-0.02
Yaw-roll, 2_1	+0.07	-2.39	-0.01
Yaw-roll, 3_1	+0.12	-2.41	+0.02
Pitch, 4_1	-0.12	-2.42	+0.01

The combination of thrust and engine misalignment resulted in two small roll transients. At lift-off minus 0.150 second, a transient of +0.2 deg/sec occurred; at lift-off, a second transient of -1.1 deg/sec occurred. An actuator bias of 0.36 degree had been set into the yaw-roll actuators and this apparently reduced the roll transient at lift-off to one half of that indicated on GLV-8.

The first stage programmed roll and pitch programs were performed as planned and were normal in rates and duration, as listed in table 5.2.3-I. The discretes initiated by the Three Axis Reference System (TARS) were executed at the pre-set times.

Primary (TARS) and secondary (Inertial Guidance System (IGS)) attitude-error signals shown in figure 5.1.5-1 correlated well throughout Stage I flight. These attitude errors indicate the response of the control system to the first-stage guidance programs and the vehicle disturbances caused by the prevailing winds aloft. The maximum vehicle rates and attitude errors which occurred during Stage I flight are presented in table 5.2.3-II. The dispersions between the primary and secondary attitude-error signals were the result of a combination of drift in the TARS and in the IGS inertial measurement unit (IMU), errors in TARS roll and pitch guidance programs, and cross-coupling of the reference axes within each of the systems.

5.2.3.2 Staging sequence.- Maximum attitude errors and rates were higher than normal during the staging sequence, probably as a result of the Stage I fuel-depletion shutdown transient. The maximum attitude errors and rates recorded during staging are given in table 5.2.3-III.

5.2.3.3 Stage II flight.- The primary and secondary attitude error signals are shown in figure 5.1.5-1. The Stage II attitude biases resulted from the Stage II thrust-vector misalignment, the center-of-gravity offset from the longitudinal axis, and the offset of the roll thrust from the longitudinal axis. Telemetry data indicate normal primary system response to the radio-guidance pitch commands. No hardware discrepancies related to the off-nominal SECO + 20 second dispersions were noted in the data.

The response of the primary flight-control system to the Radio Guidance System yaw-left command was satisfactory. The differences noted in the yaw-attitude errors between the primary and secondary guidance systems were due to intentional inhibiting of the two secondary-system updates at LO + 105 seconds and LO + 145 seconds. In the event of a switchover to secondary guidance, inhibiting these corrections to the secondary guidance system would have resulted in a large out-of-plane velocity error at insertion.

5.2.3.4 Post-SECO and separation phase.- Vehicle attitude rates between SECO and spacecraft separation were normal. The maximum rates experienced during this period are listed in table 5.2.3-IV. Maximum pitch and yaw attitude excursions of 6 degrees were experienced at separation, as shown in figure 5.2.3-1. There was no detrimental effect on spacecraft separation as a result of these errors. The vehicle roll attitude between SECO and separation was normal, as shown in figure 5.2.3-1.

TABLE 5.2.3-I.- TARS ROLL AND PITCH PROGRAMS

Program	Actual time, LO + sec	Planned time, LO + sec	Rate gyro, deg/sec	Torquer monitor, deg/sec	Nominal rate, deg/sec
Roll					
Start	18.42	18.48	-1.25	-1.22	-1.25
Stop	20.42	20.48			
Pitch, step 1					
Start	22.97	23.04	-0.70	-0.72	-0.709
Pitch, step 2					
Start	88.06	88.32	-0.50	-0.54	-0.516
Pitch, step 3					
Start	118.66	119.04	-0.23	-0.23	-0.235
Stop	161.44	162.56			

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

Axis	Maximum rate, deg/sec	Time from lift-off, sec	Maximum attitude error, deg	Time from lift-off, sec
Pitch	+0.2 -1.17	0.8 55.6 and 75.5	+0.99 -0.63	107.5 77.9
Yaw	+0.68 -0.38	82.0 87.4 and 111.6	+0.89 -0.58	108.4 77.6
Roll	+0.68 -1.44	152.4 19.1	+0.21 -0.37	18.8 and 152.4 0.5

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

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Axis	Maximum rate gyro, deg/sec	Time from BECO[a], sec	Maximum rigid body rates, deg/sec	Time from BECO[a], sec	Maximum vehicle attitude change, deg	Time from BECO[a], sec
Pitch	+2.44 -5.08	0.04 0.75	-1.0	0.8	-0.73	1.6
Yaw	+2.60 -2.37	0.75 0.76	+1.65	1.6	+1.99	2.7
Roll	+3.50 -5.65	1.35 0.41	+2.50	1.0	-1.84	1.1

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[a]BECO occurred 152.48 seconds after lift-off.

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

Condition	Rate, deg/sec
Pitch	
Maximum positive rate at SECO + 1.0 sec	+0.88
Maximum negative rate at SECO + 0.1 sec	-0.09
Rate at SECO + 20 sec	0.0
Rate at spacecraft separation (SECO + 26.9 sec)	+1.17
Yaw	
Maximum positive rate at SECO + 15.5 sec	+0.59
Maximum negative rate at SECO + 2.6 sec	-1.27
Rate at SECO + 20 sec	+0.59
Rate at spacecraft separation (SECO + 26.9 sec)	+0.45
Roll	
Maximum positive rate at SECO + 0.3 sec	+0.30
Maximum negative rate at SECO + 8.6 sec	-0.48
Rate at SECO + 20 sec	+0.20
Rate at spacecraft separation (SECO + 26.9 sec)	+0.08

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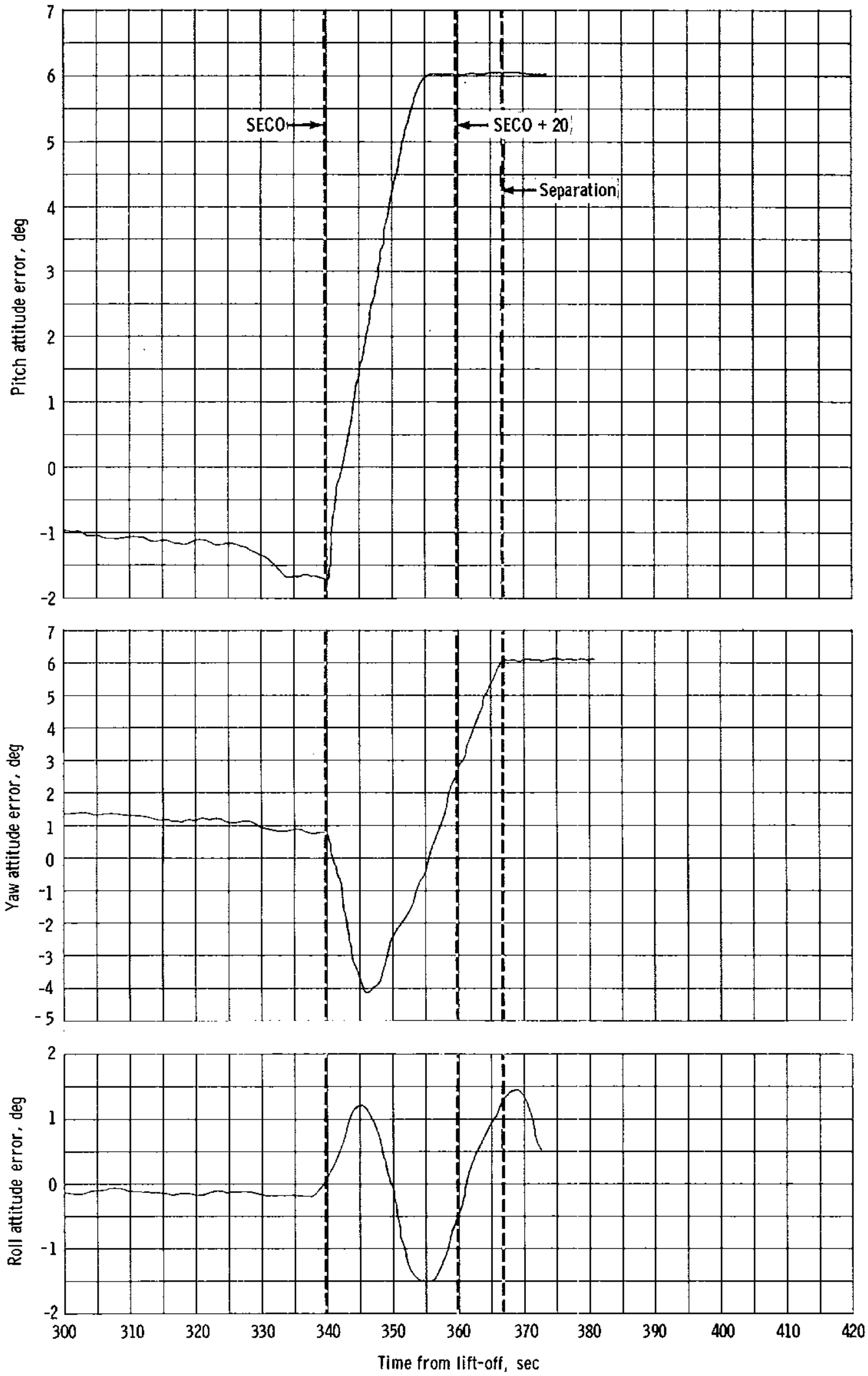

Figure 5.2.3-1. - Attitude errors after SECO.

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5.2.4 Hydraulic System

The vehicle hydraulic systems performed satisfactorily during Stage I and Stage II flight. No anomalous pressures were noted during steady-state flight, indicating low flow demands and a smooth flight. Prior to the simulated flight test, the engine-driven pumps were replaced with newly cleaned units, and the action of the pressure compensators in these units was verified by a Gaussmeter check. Table 5.2.4-I shows selected hydraulic system pressures.

The hydraulic test selector valve, which had malfunctioned on GLV-8, was replaced twice on GLV-9 because of anomalous prelaunch performance. Improper adjustment of the air gap in the solenoid of the control valve was determined to be the problem. By the addition of shims, this condition was corrected in the final valve installed on GLV-9. Subsequent performance in tests and prelaunch checks was satisfactory.

TABLE 5.2.4-I.- HYDRAULIC PRESSURES

Event	Stage I		Stage II system, psia
	Primary system, psia	Secondary system, psia	
Starting transient (minimum)	2720	--	--
Starting transient (maximum)	3260	3600	3860
Steady state	3100	3150	3000
BECO	2800	2840	--
SECO	--	--	2860

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-I, was within acceptable limits. The trajectory was nominal, and the errors at BECO, compared with the no-wind prelaunch nominal trajectory, were 22 ft/sec low in velocity, 1635.0 feet high in altitude, and 0.05 degree high 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 60.0 seconds after SECO. There was a 2.2-second period of intermittent lock until final loss-of-signal at 62.2 seconds after SECO. Track was maintained to an elevation angle of 1.55 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 + 38.3 seconds to LO + 381.0 seconds (41.2 seconds after SECO). Rate lock was maintained to an elevation angle of 2.0 degrees above the horizon.

Pitch steering commands were initiated as planned by the airborne decoder at LO + 168.19 seconds. At this time, 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 5.0 seconds. Pitch steering at guidance initiate was indicative of a nominal first-stage trajectory. The steering gradually returned, during the following 17.0 seconds, to relatively small pitch-down commands slowly varying from 0.6 to 0.11 deg/sec. At LO + 280 seconds, because of noisy tracking data, the rates became oscillatory. This particular phenomena is a normal characteristic of tracking data when the system is being influenced by atmospheric effects. Past experience had shown the noise to increase as the tracking elevation angle decreases. As a result, the commands varied between 0.1 and 0.18 deg/sec (pitch-down) from LO + 280 seconds until approximately 26 seconds before SECO. The pitch commands then gradually experienced a peak-to-peak (-0.2 to +0.3 deg/sec) single-cycle phenomena until termination of guidance (SECO - 2.5 seconds). During this time, a phase difference was noted in the steering commands between the RGS and the IGS. That is, the RGS sent pitch-down and pitch-up commands, while the IGS commanded a steady pitch-up. The phenomena that appeared in the RGS pitch-steering commands near SECO is currently attributed to low-frequency tropospheric effects. These effects are not predictable and, therefore, are not accounted for in the guidance system. Analysis has shown that on the Gemini VIII mission and again on the Gemini IX-A mission the major contributor to the errors at SECO + 20 seconds were these low-frequency tropospheric effects.

Yaw steering was initiated, as planned at LO + 168.19 seconds. The commands were indicative of the large dog-legged trajectory (-0.51 degree wedge angle) executed during the second-stage flight. The philosophy behind the dog-legged trajectory, executed on this flight through means of the Radio Guidance System, was to remove the out-of-plane wedge angle (position error) that existed between the in-orbit target vehicle and the GLV at lift-off. This was accomplished empirically by means of a prelaunch targeting procedure and through use of the target-vehicle real-time ephemeris data to compute the proper 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) was dependent on the prelaunch (T - 120 minutes) GLV performance prediction. As a result, yaw-left commands of 100 percent (2.0 deg/sec) were sent for a duration of 4.0 seconds. The steering gradually returned, 19 seconds later, to yaw-right commands of less than 0.04 deg/sec until termination of guidance (SECO - 2.5 seconds). At SECO + 20 seconds, the yaw velocity was 0.0 ft/sec and the yaw position was -7129 feet, as compared with the planned values of 0.5 ft/sec and -7462 feet (prelaunch guidance residuals due to insertion targeting accuracies).

SECO occurred at LO + 339.784 seconds at an elevation angle of 6.70 degrees above the horizon. The SECO + 20 second conditions were 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 may be attributed primarily to noise in the guidance data. Evaluation of the near-nominal shutdown thrust transient indicates that it contributed only 2.0 ft/sec of the estimated 16.0 ft/sec total underspeed at SECO + 20 seconds.

Likewise, the yaw-position errors (smallest to date) and yaw-velocity errors at SECO + 20 seconds resulted in the spacecraft having to make a 14.8 ft/sec out-of-plane maneuver in its second orbit (see section 4.0). Vehicle attitude rates were 0.0 deg/sec pitch down, 0.59 deg/sec yaw right, and 0.20 deg/sec roll clockwise at SECO + 20 seconds.

The ground-based A-1 guidance computer, in conjunction with the GE MOD III tracking and missileborne guidance system, performed satisfactorily during prelaunch and flight. No anomalies were encountered with the airborne pulse, rate, and decoder hardware. All guidance-system generated discretes were executed nominally and on time.

The prelaunch transmission and verification of the target ephemeris data at approximately T-25 minutes, between the Real Time Computer Complex at the Mission Control Center in Houston and the Guided Missile Computer Facility at Cape Kennedy were satisfactory.

The T-3 minute spacecraft targeting updates were generated and transmitted satisfactorily from the A-1 guidance computer. These updates were not transmitted to the spacecraft Digital Command System because of a failure in the ground equipment at T-3 minutes. Due to this ground equipment problem, a decision was made to manually inhibit the two spacecraft IGS updates transmitted at LO + 105 seconds and LO + 145 seconds. The updates were generated by the A-1 computer as planned and verified as correct by a recording. Further discussion of this ground transmission problem is in sections 5.1.5 and 6.2.1.

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5.2.6 Electrical

The Instrumentation Power Supply (IPS) provided power at a nominal 30 volts throughout the countdown and flight. The respective values of current and voltage indicate that the staging sequence (fire staging nuts) resulted in a momentary short, through the staging-nut squib wires to the structure; the short was cleared by the physical separation of the two stages. The Auxiliary Power Supply (APS) also performed nominally. This system reflected the usual temporary short caused by the staging sequence. The spacecraft-separation event is distinctly evident on both the APS and IPS by transients on the respective current traces.

The remaining electrical systems on the vehicle (5-volt instrumentation power supply; 115-volt, 400-cycle supply; 40-volt supply; and the 25-volt-dc supply) also reflected nominal operation throughout the flight.

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5.2.7 Instrumentation

5.2.7.1 Ground.- For the final countdown and launch of Gemini IX-A, there were 104 measurements programmed for use on the landline system. Data acquisition was 100 percent with no problems. The umbilical-connectors separation sequence was as planned and complete in 0.807 seconds. The actual separation time and the rise of the launch vehicle at connector separation was the closest yet to the planned events.

5.2.7.2 Airborne.- There were 188 measurements scheduled for use on this launch. Data review disclosed no transducer measurement anomalies. Review of real-time telemetry data disclosed a momentary loss of launch-vehicle telemetry immediately after lift-off (LO + 0.6 sec). Review of signal-strength records and a subsequent data playback revealed this problem was associated only with the Tel II data. Investigation is continuing to determine the reason for this condition. The expected telemetry data loss at staging lasted 320 milliseconds. Loss-of-signal for the launch-vehicle telemetry occurred at approximately LO + 490 seconds.

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5.2.8 Malfunction Detection System

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

5.2.8.1 Engine MDS.- The malfunction-detection thrust-chamber pressure switch (MDTCPS) and malfunction-detection fuel-injector pressure-switch (MDFJPS) actuations were as follows:

Switch	Condition	Actuation time from lift-off, sec	Pressure, psia
Subassembly 1 MDTCPS	Make	-2.327	570
	Break	+152.443	550
Subassembly 2 MDTCPS	Make	-2.295	600
	Break	+152.431	530
Subassembly 3 MDFJPS	Make	+153.201	(a)
	Break	+339.937	(a)

[a]MDFJPS operational pressures are not known because there is no equivalent analog sensor.

5.2.8.2 Airframe MDS.- The MDS rate-switch package operated properly throughout the flight.

No vehicle overrates occurred during powered flight, and no spurious outputs were generated. After spacecraft separation, there were three operations of the low-rate switch contacts. Comparison of these actuation times with flight-control rate-gyro data indicate that the rate-switch operations were in agreement with preflight calibrations.

5.2.8.3 Tank Pressure Indications.- All MDS tank-pressure transducers operated properly throughout the flight. Maximum difference between paired transducers was 1.2 percent of full scale.

TABLE 5.2.8-I.- GEMINI IX-A MALFUNCTION DETECTION SYSTEM SWITCHOVER PARAMETERS

Parameter	Switchover setting	Maximum or positive	Time from lift-off, sec	Minimum or negative	Time from lift-off, sec
Stage I primary hydraulics	Shuttle spring (1500 psia equivalent)	3300 psi	-2.08	2720 psi	-2.34
Stage I secondary hydraulics	None	3570 psi	-2.53	2840 psi	BECO
Stage I tandem actuators					
No. 1 subassembly 2 pitch	±4.0 deg	+0.70	78.6	-0.38	24.2 and 93.8
No. 2 subassembly 2 yaw-roll	±4.0 deg	+0.50	83.0	-0.44	78.0
No. 3 subassembly 1 yaw-roll	±4.0 deg	+0.43	78.0	-0.43	83.0
No. 4 subassembly 1 pitch	±4.0 deg	+0.40	24.2 and 93.8	-0.70	78.6
Stage I pitch rate[a]	+2.5 deg/sec -3.0 deg/sec	+0.10	1.0	-1.10	56.0
Stage I yaw rate[a]	±2.5 deg/sec	+0.50	82.5	-0.33	112.5
Stage I roll rate[a]	±20 deg/sec	+1.01	152.4	-2.73	152.8
Stage II pitch rate[a]	±10 deg/sec	+0.50	154.8	-2.00	172.0
Stage II yaw rate[a]	±10 deg/sec	+1.68	154.1	-2.30	172.0
Stage II roll rate[a]	±20 deg/sec	+2.50	+153.9	-0.40	155.3

[a]Positive indicates pitch up, yaw right, or roll clockwise. Negative indicates pitch down, yaw left, or roll counterclockwise.

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5.2.9 Range Safety and Ordnance

The performance of all Range Safety and Ordnance items was satisfactory.

5.2.9.1 Flight termination system.- Both GLV command receivers displayed adequate received signal throughout powered flight and beyond spacecraft separation.

The following command facilities were used:

Time from lift-off, sec	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 260	Grand Bahama Island (GBI) 10kW transmitter and steerable antenna
260 to 432	Grand Turk Island (GTI) 10kW transmitter and steerable antenna

Auxiliary Stage II Engine Cutoff (ASCO) was transmitted over the GTI transmitter at LO + 339.81 for 4.43 seconds.

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

5.2.10.1 Launch attempt.- Propellant loading was initiated at 04:03 G.m.t. on May 31, 1966, and was accomplished in 3 hours 32 minutes. Only one Aerospace Ground Equipment (AGE) problem occurred during countdown of the launch vehicle. At T-40 minutes, during the running of the Program Sequence Test, the Airborne-Beacon Test-Set (ABETS) frequency drifted out of tolerance. The ABETS unit was retuned, and a retest was satisfactorily performed. The Gemini Launch Vehicle (GLV) count continued and the planned hold at T-3 minutes was reached without incident. When the countdown was resumed, the ground equipment could not transfer the refined targeting information to the spacecraft computer because of ground equipment problems. The mission was recycled for launch on June 3, 1966.

5.2.10.2 Recycle.- The recycle activities consisted of offloading of GLV propellants, removing the start cartridges and destruct initiators, draining and purging the fuel side of the Stage I engine, and performing special engine inspections to ensure that oxidizer was not leaking through Stage I thrust-chamber valves. In addition, the oxidizer standpipes for longitudinal oscillation suppression were drained and purged, including the remote-charge AGE system. Electrical power was left on the vehicle during this period.

5.2.10.3 Launch.- The final countdown was initiated through the range sequencer for a launch (T-0) at the nominal 13:39:30 G.m.t. on June 3, 1966. The second propellant loading had been completed in 3 hours 15 minutes. Because the prevalves had remained open during the recycle, a weight correction was applied to the loading schedule to compensate for the difference in configuration. The automatic oxidizer-standpipe charging procedure was accomplished at T-176 minutes, and the airborne disconnects were manually removed.

No delays were encountered and launch was successfully accomplished at 13:39:33 G.m.t.

Motion-picture film of the launch shows that the drop-weight systems for the upper spacecraft umbilical connector functioned properly. This indicates that the revised lanyard lengths, to permit the upper lanyard to exert the initial force on the connector, were satisfactory.

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5.3 SPACECRAFT—GEMINI LAUNCH VEHICLE INTERFACE PERFORMANCE

The various aspects of the Spacecraft—Gemini Launch Vehicle interface, as defined in reference 17, 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 observations.

The electrical circuitry performed as anticipated. Review of electrical data disclosed the presence of shorting during the spacecraft—launch vehicle separation event. However, no problems were experienced on either the spacecraft or the launch vehicle. The Malfunction Detection System performed without incident. The spacecraft Inertial Guidance System yaw steering commands to the launch vehicle deviated from those of the launch-vehicle Radio Guidance System toward the end of Stage II flight, as a result of manually inhibited launch-azimuth updates to the spacecraft.

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5.4 GEMINI AGENA TARGET VEHICLE PERFORMANCE

Section 5.4 of this report is not applicable to the Gemini IX-A Mission. A detailed evaluation of the Gemini Agena Target Vehicle flown on the Gemini IX Mission on May 17, 1966, is contained in a supplemental report to this report. See section 12.4 for the title and number of this supplemental report.

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5.5 TARGET LAUNCH VEHICLE PERFORMANCE

The performance of the Target Launch Vehicle (TLV) was satisfactory. The TLV sustainer engine operated for a longer duration than on any previous Atlas flight, and boosted the Augmented Target Docking Adapter (ATDA) to the required velocity and position for direct insertion into the specified orbit. Insertion parameter values indicated an orbit very close to the planned 161.2-nautical-mile circular orbit. A Digital Command System (DCS) ground transmitter link was utilized to initiate the TLV/ATDA separation sequence of events.

The TLV/ATDA was launched from Complex 14, Air Force Eastern Test Range (ETR), at 15:00:02.363 G.m.t. on June 1, 1966. No holds or difficulties were encountered during the TLV/ATDA launch countdown.

All times in this section, unless otherwise noted, are referenced to 2-inch motion of the TLV as zero time. This time is quoted in the preceding paragraph.

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 1.12g peak-to-peak at approximately lift-off (LO) + 7 seconds and was damped by LO + 25 seconds. This oscillation is excited during release of the launcher hold-down arms.

Axial-accelerometer data indicated peak accelerations at booster engine cutoff (BECO) and sustainer engine cutoff (SECO) of 5.85g and 8.23g, respectively. The expected accelerations were 5.91g and 8.25g.

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 recorded boost-phase temperature was 115° F and occurred in the area of the sustainer fuel pump at LO + 90 seconds. The minimum temperature recorded during the boost phase was 49° F and occurred on the sustainer instrument panel at lift-off.

Booster-section jettison at LO + 120.179 seconds and ATDA separation at LO + 383.409 seconds were normal. Gyro and acceleration 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, utilizing MA-5 booster, sustainer, and vernier components, was satisfactory in performance and operational characteristics. Because of the extended sustainer engine firing time dictated by the mission requirements, the sustainer engine was equipped with a larger capacity lubrication-oil tank (additional 1.5 gallons), which proved adequate for the mission. No degradation of sustainer engine performance resulted from the extended firing time.

A comparison of actual computed thrust obtained during flight with the predicted thrust levels is shown in the following table.

TLV Engine Performance

Engine		Thrust, lb			
		Lift-off	BECO	SECO	VECO
Booster	Predicted	330 477	380 794	NA	NA
	Actual	327 852	379 761	NA	NA
Sustainer	Predicted	58 057	81 916	80 685	NA
	Actual	56 758	81 178	79 138	NA
Vernier	Predicted	1 153	1 411	1 152	1 155
	Actual	1 089	1 439	1 077	900

NA - Not applicable

The engines started at LO - 2.73 seconds, and ignition, thrust rise, and thrust levels were normal prior to lift-off. The booster engines were cut off by a guidance system command at LO + 117.207 seconds. The sustainer engine operation was terminated upon command at LO + 348.700 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 the VECO command occurring at LO + 367.537 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, time from lift-off, sec	Start of thrust decay, time from lift-off, sec
BECO	117.207	117.317
SECO	348.700	348.743
VECO	367.537	367.659

The environmental temperature measurements reflected normal radiation heating during the sustainer phase of flight and indicated no evidence of cyrogenic leaks.

5.5.2.2 Propellant utilization.- The propellant utilization system operated properly throughout the flight. Propellant residuals at SECO were calculated by utilization of the uncovering times of the instrumented head-pressure ports in the liquid-oxygen and fuel tanks in conjunction with the flow rates determined between sensor stations 5 and 6 (corrected for the propellant-utilization valve-angle change after sensor station 6 uncovered). Usable propellant residuals based on this method of calculation are presented in the following table.

Condition	Liquid oxygen, lb	Fuel, lb	Time from SECO to theoretical liquid-oxygen depletion, sec	Excess fuel at theoretical liquid-oxygen depletion, lb
Predicted	561	362	2.90	69
Actual	330	369	1.82	225

5.5.2.3 Propellant loading.- The normal propellant loading procedure was used for this vehicle. Fuel was tanked to a level 12 gallons above the 100-percent probe on May 31, 1966. Liquid oxygen was tanked during the countdown to near the 100-percent probe and maintained at this level until the liquid-oxygen fill system was closed. Total fuel and liquid-oxygen weights prior to launch were 76 919 pounds and 175 000 pounds, respectively.

5.5.3 Flight Control System

The performance of the Flight Control System was satisfactory. Attitude control and vehicle stability were maintained throughout the flight, and the proper sequence of events was performed by the autopilot programmer. Moderate transients at lift-off were rapidly damped following autopilot activation at 42-inch motion, as indicated by initial engine movements at LO + 0.74 seconds. The lift-off roll transient reached 0.7 degree in the counterclockwise direction at a peak rate of 2.6 deg/sec.

The usual longitudinal mode that is apparent at lift-off excited the second lateral bending mode of the TLV. Maximum oscillations in pitch at a frequency of 8.4 cps reached 3.8 deg/sec peak-to-peak and became completely damped by LO + 8 seconds.

Gyro data provide 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 1.4 degrees. The programmer enabled guidance steering at LO + 95 seconds; however, no steering commands were required, and none were evidenced in the data.

Low-amplitude TLV propellant sloshing which began at a frequency of 1.2 cps was observed between LO + 60 seconds and LO + 112 seconds, with maximum peak-to-peak rates of 0.9 and 0.8 deg/sec in pitch and yaw, respectively. The usual coupling into the roll plane was evident at a maximum peak-to-peak rate of 1.8 deg/sec.

Vehicle lateral bending at low amplitudes was evident throughout the period from lift-off to staging; however, the maximum oscillations at a frequency of 5.2 to 5.6 cps did not exceed 0.4 deg/sec peak-to-peak.

The guidance-initiated staging discrete signal was indicated at the programmer input at LO + 117.06 seconds, and the resultant switching sequence was successfully executed. Vehicle transients associated with BECO and booster-section jettison were not excessive and were quickly damped by the autopilot. Rigid-body oscillations at a frequency of 0.26 cps in pitch and yaw were excited by booster jettison but did not exceed 0.5 deg/sec peak-to-peak. The oscillations were also evident at a reduced amplitude until SECO.

The vehicle first bending mode which usually occurs after BECO was evident in the pitch and yaw planes. The maximum zero-to-peak amplitude sensed by the TLV rate gyros was 1.1 deg/sec in pitch at a frequency of 6.0 cps. The oscillation had decayed to zero prior to booster-section jettison.

Proper system response was exhibited to all guidance steering commands. The sustainer cutoff signal was received by the programmer at LO + 348.70 seconds. No vernier steering commands were planned for this flight, and the vernier control phase was normal, with the VECO signal being received by the programmer at LO + 367.53 seconds.

Gyro and accelerometer data indicate the initiation of shroud jettison at LO + 381.48 seconds and ATDA/TLV separation at LO + 383.41 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.

Ullage pressures in the liquid-oxygen and fuel tanks were stable at 24.6 psig and 64.5 psig, respectively, at lift-off, and at 25.3 psig and 66.1 psig at BECO. The differential pressure across the propellant-tank intermediate bulkhead was normal at 16.0 psid (fuel-tank pressure minus liquid-oxygen head pressure plus ullage pressure) at lift-off, 17.7 psid at BECO, and 16.8 psid at SECO and VECO. The minimum bulkhead differential pressure experienced during flight was 12.0 psid at LO + 0.45 seconds.

During the boost phase, 68.8 pounds of the 152.4 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 790 psig at BECO.

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 3130 psig in the booster system and 3010 in the sustainer/vernier system. These pressures were satisfactorily maintained until the respective engine cutoffs.

After SECO and the cessation of sustainer-pump output, the sustainer/vernier system reverted to vernier-solo accumulator operation. The vernier system pressure was 1480 psig at VECO. All return system pressures were normal.

5.5.5 Guidance System

The TLV was guided by the autopilot and by the MOD III-G Radio Guidance System (RGS), both of which performed satisfactorily throughout the countdown and powered flight. The three planned discrete commands and closed-loop steering commands were properly generated and transmitted by the ground equipment, and all commands were received and decoded by the TLV equipment. The ATDA, unlike the Gemini Agena Target Vehicle, did not utilize discrete commands from the TLV.

5.5.5.1 Programmed guidance.- The initial open-loop steering of the TLV, as indicated by rate and displacement gyro outputs from the autopilot, was nominal. The pre-set roll and pitch programs of the TLV Flight Control System successfully guided the vehicle into the planned trajectory (refer to section 5.5.3).

5.5.5.2 Radio Guidance System.-

5.5.5.2.1 Booster steering: The radio-guidance ground station acquired the TLV in the cube-acquisition mode, as planned, with vehicle-borne rate and track lock-on established at LO + 55.5 and LO + 57.0 seconds, respectively. Normal lock-on was maintained until approximately LO + 435 seconds, when tracking was intentionally terminated.

Booster steering, implemented to correct open-loop dispersions, was enabled by the TLV Flight Control System at LO + 95 seconds, as planned. However, no corrections were required; therefore, no steering commands were generated, and none were evidenced on telemetered decoder output data. The staging signal, indicated at the autopilot programmer input, occurred at LO + 117.06 seconds at an elevation angle of 49.9 degrees. The errors at BECO were 11 ft/sec high in velocity, 746 feet high in altitude, and 0.29 degrees high in flight-path angle (refer to table 4.3-V).

5.5.5.2.2. Sustainer steering: Sustainer steering was initiated at LO + 132.5 seconds. The peak pitch command was an initial 100 percent pitch-up command for 2.5 seconds which decreased to less than 15 percent by LO +138 seconds. Yaw steering was active, with both positive and negative commands below 15 percent until approximately LO + 145 seconds. Subsequently, pitch and yaw commands were less than

15 and 5 percent, respectively, for the sustainer steering phase. The sustainer cutoff signal was measured at the programmer input at LO + 348.70 seconds.

5.5.5.2.3 Vernier steering: No vernier steering maneuvers were planned for this flight. Consequently, no steering commands were generated during vernier-solo operation. The vernier cutoff signal, as indicated at the programmer input, occurred at LO + 367.53 seconds and at an elevation angle of 12.3 degrees. The VECO conditions were very close to the nominal insertion parameters. The insertion velocity was 0.9 ft/sec low, the vertical velocity was 5.0 ft/sec low, the lateral velocity was 2.8 ft/sec right, and the radius was 37 feet low. The following table compares the actual insertion conditions with the planned conditions.

VECO conditions	Planned	Actual
Time from lift-off, sec	366.077	367.537
Space-fixed velocity, ft/sec	25 364.5	25 363.6
Vertical velocity, ft/sec	-2.8	-7.8
Yaw velocity, ft/sec	0.0	+2.8
Radius, ft	21 888 821	21 888 784

5.5.6 Electrical System

Operation of the electrical system was satisfactory during countdown operations and throughout the flight. All electrical parameters were at normal levels and remained within tolerance. There was no evidence of any unusual transients; however, a ripple voltage was evident on the measurement S209V (programmer 28 V dc test) data during the latter portion of the vernier-solo phase.

These data indicate that a ripple voltage of 0.61 volts peak-to-peak, at a mean frequency of 18.1 cps, existed on the main vehicle dc bus. The ripple started gradually at LO + 326.0 seconds, then slowly decreased, and finally disappeared at LO + 427.4 seconds. A ripple on the main vehicle dc bus was noted during ground checks of this vehicle, as well as on several previous vehicles. No detrimental effects have ever been apparent as a result of these fluctuations, and the magnitude and frequency of the fluctuations, in all cases, have been well within electrical system and user system specifications.

5.5.7 Instrumentation System

5.5.7.1 Telemetry.- The TLV telemetry system operated satisfactorily throughout the flight. One lightweight telemetry package was utilized to monitor a total of 110 parameters on 9 continuous and 5 commutated channels. The usable data provided a system recovery of 100 percent.

Measurement A743T (ambient temperature at sustainer instrument panel) indicated an open circuit after TLV booster-section jettison, but provided satisfactory data during the period of predominant interest.

Measurement S54R (yaw rate gyro) had a superimposed 17 to 19 cps frequency of up to 10 percent indicated-bandwidth amplitude. Because this frequency was not evidenced on the data played back at ETR and because the amplitude was appreciably less on the secondary recording track than on the primary track, it appears that these data were not a valid indication of airborne-system operation.

5.5.7.2 Landline.- The landline instrumentation system carried a total of 48 analog and 54 discrete vehicle measurements. All 102 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 command signals were received or generated. Range-safety plots and telemetry readouts in Central Control were normal throughout the flight.

The RF signal strength received at command receiver 1 indicated that sufficient signal margins were available for proper operation of the RF command link at all times during the flight.

5.6 ATDA/TLV INTERFACE PERFORMANCE

The Target Launch Vehicle/Augmented Target Docking Adapter (TLV/ATDA) interface performed as expected. Accelerometer data indicate that separation was nominal. The bungee cords functioned properly, providing the necessary separation velocity of the ATDA from the TLV.

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5.7 SPACECRAFT/AUGMENTED TARGET DOCKING ADAPTER INTERFACE

Performance of the Spacecraft/Augmented Target Docking Adapter (ATDA) interface was satisfactory throughout the flight. The mooring drive system operated satisfactorily in the undocked mode at all times, even though the docking cone was contacting the partially separated shroud. Because docking could not be achieved, the performance of the status display panel, the hardline electrical circuits, and the mooring drive system in the docked mode was not evaluated.

The acquisition lights were acquired on the first rendezvous at an approximate range of 20 nautical miles. Estimated brightness at that range was equivalent to that of a second or third magnitude star. Because of the position of the shroud, the upper light was fully uncovered, but the lower one was substantially obscured.

The ATDA running lights, which were of greater intensity than those on the Gemini Agena Target Vehicle, operated normally. The red lights were discernible at ranges up to 8 nautical miles, and the amber and green lights were visible at lesser ranges. Because the ATDA was tumbling, the lights were of little value for attitude determination.

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5.8 AUGMENTED TARGET DOCKING ADAPTER PERFORMANCE

The performance of the Augmented Target Docking Adapter (ATDA) was as expected with the exception of two anomalies which occurred near insertion. The shroud failed to separate completely, and the primary target stabilization system (TSS 1) exhausted the total fuel supply in the reaction control system B-ring in 129 seconds. The shroud and stabilization anomalies are discussed under Structure (section 5.8.1) and Guidance and Control (section 5.8.3), respectively.

5.8.1 Structure

With the exception of the shroud anomaly, the ATDA structure performed as expected. The launch loads, vibration, and heating were sustained satisfactorily. The pyrotechnic mechanism and bungee cords, which separate the ATDA from the Target Launch Vehicle (TLV), performed properly. The external paint pattern on the ATDA provided adequate passive thermal control for the equipment during orbital flight, and adequate brightness for visual acquisition during rendezvous.

5.8.1.1 Ascent Shroud.- The ATDA shroud performed its function of protecting the payload through the launch phase, but failed to completely release upon command at two seconds prior to separation from the TLV. Investigation revealed the failure to be the result of an improper installation. Four electrical-connector quick-disconnect lanyards had not been attached to the forward band clamp as required. Although pyrotechnic separation performed satisfactorily, electrical wiring held the two shroud halves onto the ATDA. The shroud prevented docking by not allowing access to the docking cone.

A description of the shroud is given in section 3.7. Figure 3.7-3, in that section, shows the details of the shroud separation sequence. The normal sequence of events for separation is as follows:

(a) Voltage is applied to the four pyrotechnic bolts which clamp the shroud in place on the forward end of the Target Docking Adapter (TDA). Two of these bolts are on the top and bottom of the shroud at the TDA/shroud interface. The other two preload a band clamp on the right and left side of the shroud at a station 22 inches forward of the TDA/shroud interface.

(b) Pyrotechnic charges are ignited to fail the four bolts.

(c) Spring cartridges at the TDA/shroud interface force the aft ends of the shroud halves into pivot fittings attached to the TDA on the left and right sides, separating the aft ends of the shroud halves approximately 1.6 inches.

(d) The 1.6-inch aft-end motion coupled with release of a 3/8-inch warp in each half shell of the shroud provides clearance for the release of internal lanyards attached to spring-loaded latches in the nose of the shroud.

(e) The nose latches release, permitting two 150-pound preloaded primary springs, located approximately 40 inches ahead of the TDA/ shroud interface, to start forcing the shroud halves apart.

(f) When the shroud halves have pivoted apart about 15 degrees, the band-clamp electrical-connector quick-disconnect lanyards become taut, releasing the disconnects and allowing the band clamp to fall free.

(g) The primary springs continue to force the shroud halves apart to an angle of about 26 degrees between halves, where the primary spring cartridges become completely unloaded.

(h) Rotational momentum of the shroud halves separates them further to about 30 to 35 degrees where the lanyards on the interface electrical quick disconnects become taut, releasing the connectors and allowing the shroud halves to fall free. Separation of both interface connectors opens a 24-volt circuit monitored by telemetry and provides the telemetry signal indicating shroud separation.

The normal sequence was interrupted at step (f) when the electrical connectors were not released. Wiring to the connectors prevented the band-clamp halves from falling free, absorbed the primary spring force and the opening momentum of the shroud halves, and stopped the separation sequence. The shroud halves had pivoted to about 30 degrees of separation, as shown by figure 5.8.1-1, when the wiring became taut on the ends of the band-clamp halves, as shown by figure 5.8.1-2.

The disconnects are located under the small fairing shown on figure 5.8.1-2. If the lanyards had been connected to their respective band-clamp ends, they would have become taut in similar fashion to the wires shown in the figure. Due to slack in the wiring, the lanyards would have become taut first and released the disconnects with about 2 pounds of tension, whereas the wiring requires several hundred pounds to break.

A number of factors contributed to the disconnect lanyards not being attached to the band-clamp ends. These factors primarily relate to certain differences in operating procedures. Each factor was somewhat minor in itself, but in series they produced an error which resulted in a flight problem. Good basic operating procedures involve the following:

(a) Preparation and a thorough review of a detailed installation procedure

(b) Checkout of the procedure by performing a complete trial run or rehearsal of the installation in the presence of experienced personnel

(c) Documentation of deviations to the procedure, if such are taken during final flight installation

(d) The use of engineering specialists to witness and advise during the installation.

The written procedure for the installation of the shroud was the same as that which had been successfully used on the Agena. A post-flight review, however, established that the procedure was not sufficiently detailed to preclude installation errors such as the failure to hook up the lanyards.

Although the procedure was inadequate for installation of the lanyards, the installation error would have been recognized and corrected during a complete trial mate in the presence of someone who knew the procedure in detail. A trial mate was performed, but it did not proceed to the point of lanyard installation because, for safety reasons, pyrotechnic bolts (with disconnects and lanyards) were not used, and no dummy pyrotechnic bolts were available. Attention was primarily centered around the mechanical tensioning differences between the ATDA shroud installation and that of the GATV/TDA configuration.

When the actual installation was made, the assigned personnel taped the disconnect lanyards to the respective wire bundles instead of attaching them to the band-clamp ends. The use of tape was not specified, either in the procedure or on the drawings. This then, in itself, was a deviation from procedure and, if documented as such, would probably have revealed the basic error. The assumption made was that the lanyards were provided for ground handling only and, thus, were of no consequence.

As insurance in obtaining proper flight configuration, engineering specialists are frequently used to verify the step-by-step assembly and installation of flight hardware, especially a configuration that has not flown before. An engineering specialist witnessed and approved a major portion of the shroud installation. At one point he interceded and corrected the nose-latch lanyard installation. However, when it appeared that he was no longer needed, he was released shortly before the electrical disconnect lanyards were encountered.

This anomaly strongly points up the value of a detailed procedure which has been reviewed by design and operations engineers, technicians, et cetera, and has been checked out by a complete demonstration. It also emphasizes the importance of documenting every procedure deviation, though it may seem negligible at the time.

5.8.1.2 Thermal control.- The ATDA equipment temperatures were controlled passively by the selection of external paints and finishes with the desired thermal properties. With the exception of the black external docking cone surface being painted aluminum, for better visual properties, the TDA on the ATDA was the same as for the Gemini Agena Target Vehicle configuration. The majority of the TDA was bare Alclad aluminum, with an absorbtivity to emissivity ratio (α/ϵ) of 6.0, which represents a heat absorber. The transponder cover, however, was painted white ($\alpha/\epsilon = 0.24$), with the transponder thermally shunted to it, for rejection of the high thermal output of the transponder. Most of the remaining ATDA surface was painted with an aluminized acrylic enamel having a $\alpha/\epsilon = 1.33$.

Thermocouples were located for inflight monitoring of the transponder, equipment bay, and battery module temperatures. Figure 5.8.1-3 shows the flight history of these measurements. During the standby periods of the flight, the temperatures dropped toward lower design limits, these being -20° F for the transponder, 15° F for the equipment bay, and 50° F for the battery module. The rendezvous periods are apparent in the figure by the temperature recovery as the result of all electronic equipment being activated.

Although the temperatures of the equipment bay and battery module dropped somewhat lower than expected for the tumbling rates measured by telemetry and witnessed by the crew, the temperatures were sufficiently above the lower design limits not to cause concern. The transponder temperature was expected to fall rapidly in the inoperative state because of its thermal shunt to the white cover. As shown by the figure, however, this temperature increased rapidly to optimum operating values when activated.

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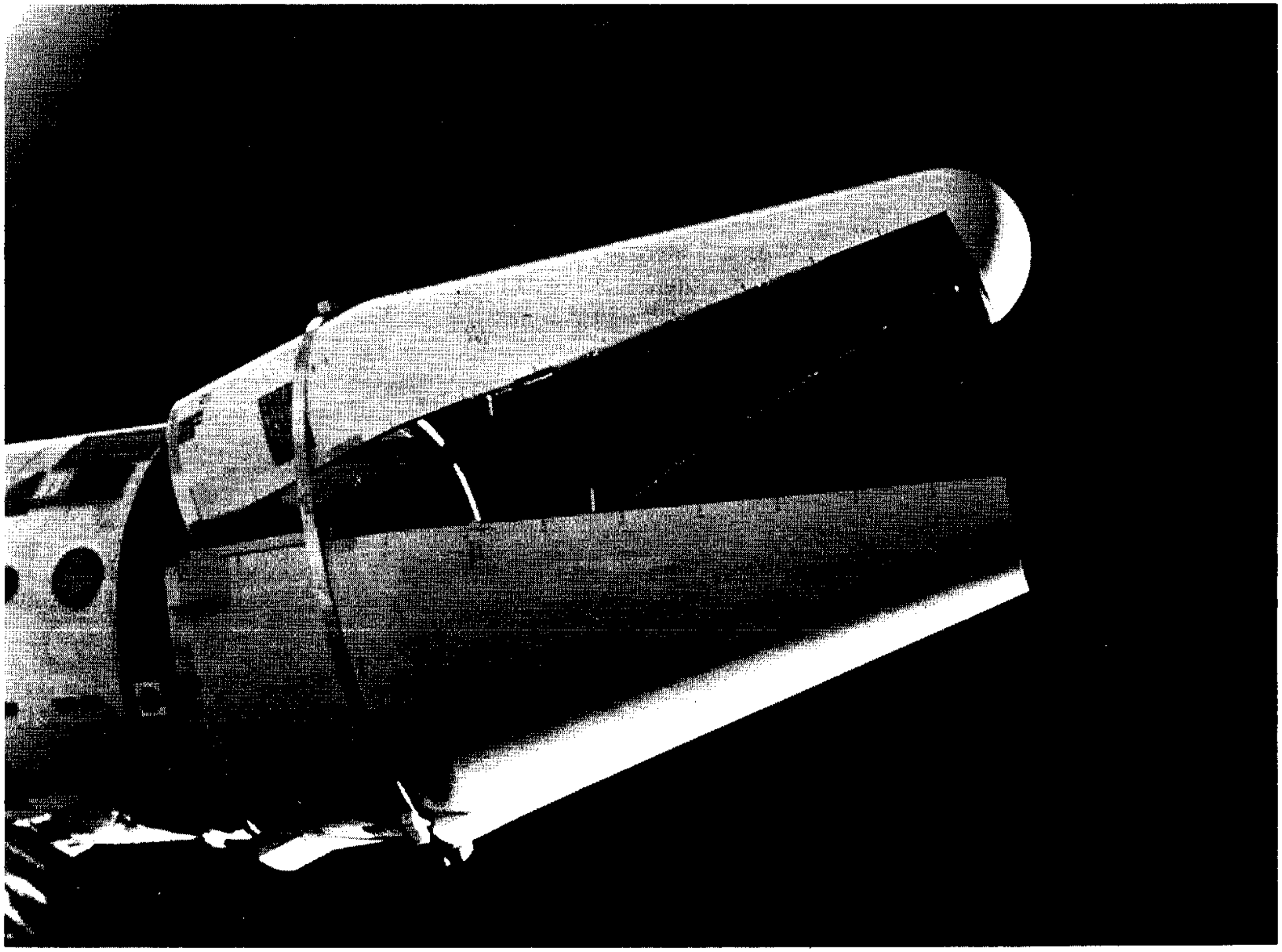

Figure 5.8.1-1. - Partially detached ATDA shroud.

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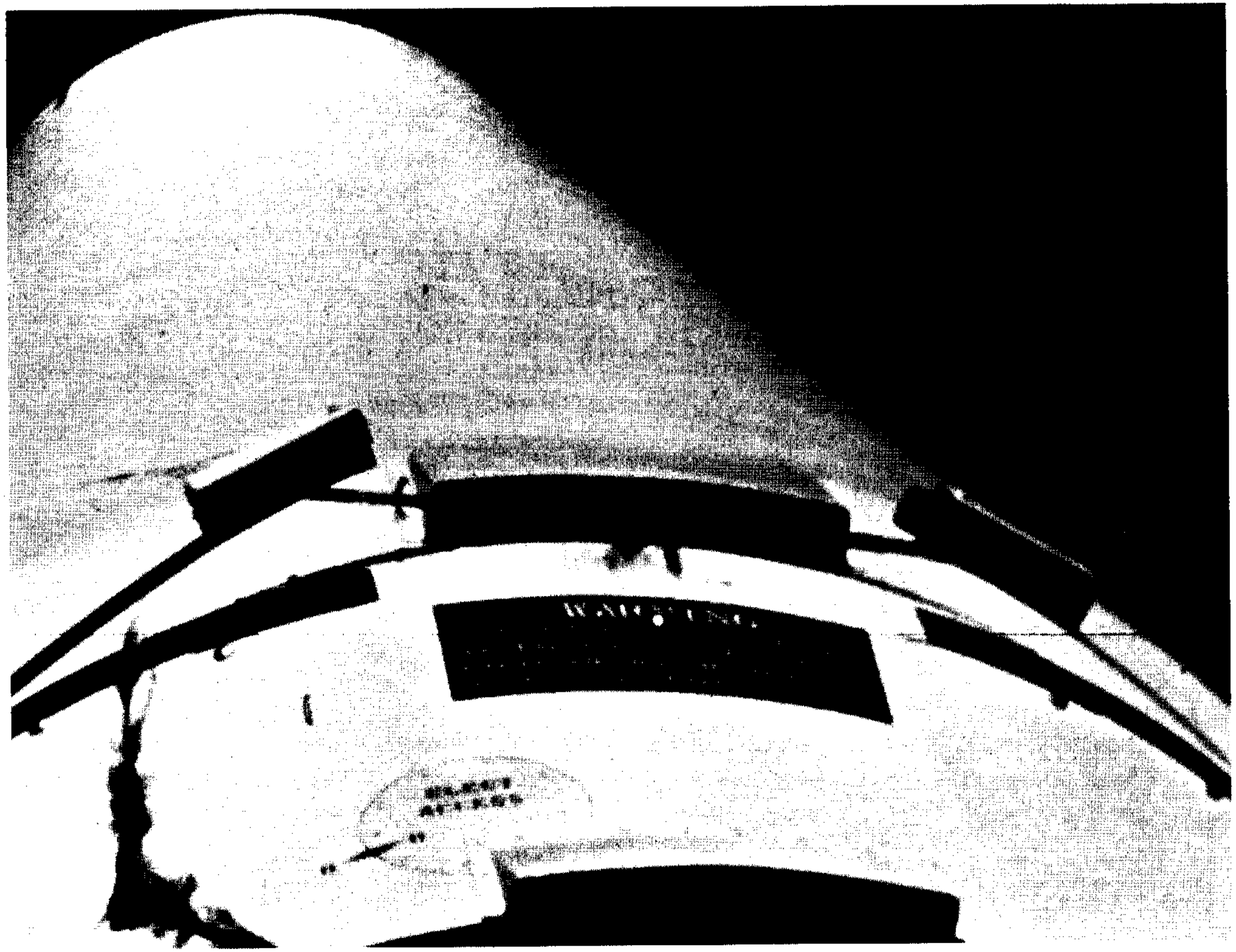

Figure 5.8.1-2. - Band-clamp halves held together by wiring to the connectors.

NASA-S-66-6894 JUN

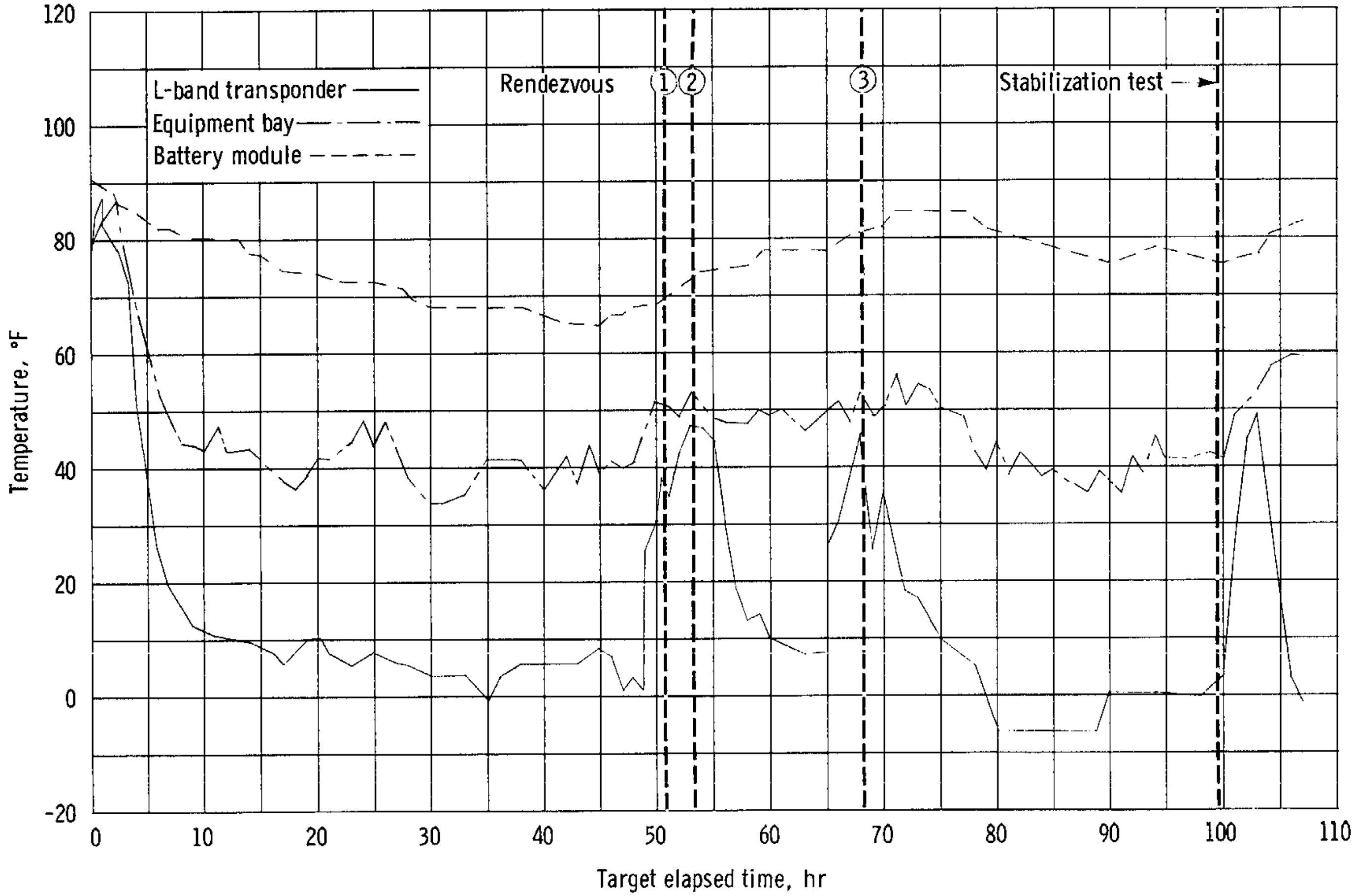

Figure 5.8.1-3. - ATDA orbital equipment temperatures.

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5.8.2 Communications System

The ATDA communications system consisted of a tracking subsystem, a telemetry subsystem, and a digital command system. All ATDA communications equipment performed in a satisfactory manner and without evidence of malfunction. The failure of the shroud to separate from the ATDA did not have any effect on the operation of the communication equipment.

5.8.2.1 Tracking subsystem.- The operation of the C-band transponder was satisfactory, as evidenced by the excellent tracking information supplied by the network stations. The primary C-band transponder was used throughout the mission.

5.8.2.2 Telemetry subsystem.- The operation of the telemetry transmitters was normal as indicated by the quantity and quality of the telemetry 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. The telemetry signal-strength chart from Mission Control Center-Cape Kennedy (MCC-C) did indicate a loss of RF track during the launch phase of the ATDA; however, the signal-strength chart from the Grand Bahama Island telemetry station indicated a good RF track and nominal signal levels during the same period.

5.8.2.3 Digital Command System.- The performance of the Digital Command System was satisfactory throughout the mission. Flight control personnel reported that all transmitted ground commands were validated.

5.8.2.4 Antenna system.- All antenna systems deployed and operated properly during the mission.

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5.8.3 Instrumentation System

The instrumentation system performed satisfactorily during the mission with no known anomalies.

A total of 43 parameters were monitored on the mission. Of this total, 8 parameters were low level (0 to 20 mV dc), 16 were high level (0 to 5 V dc) and 19 were bi-level (0 or 28 V dc).

An evaluation of the real-time data from typical stations is summarized in table 5.8.3-I. From these stations the usable data averaged better than 97 percent of the received data. All percentages were derived from computer-processed data edits.

TABLE 5.8.3-I.- ATDA REAL-TIME DATA RECEIVED FROM SELECTED STATIONS

Station	Revolution	Total data received		Total losses		Usable data, percent
		Duration, sec	Total master frames	Master frames	Percent	
MCC-C	Launch	541.94	21 677	263	1.2	98.8
MCC-C	1/2	418.25	16 730	296	1.8	98.2
ANT	Launch	267.7	10 708	556	5.2	94.8
ANT	16/17	555.5	22 220	1060	4.8	95.2
TEX	43	516.34	20 653	1000	4.8	95.2
BDA	44	547.03	21 881	304	1.4	98.6
CYI	44	521.14	20 845	300	1.4	98.6
TEX	32	507.84	20 314	276	1.4	98.6

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5.8.4 ATDA Guidance and Control

The ATDA target stabilization sytem (TSS) maintained rates within design limits but commanded abnormal thruster activity resulting in rapid propellant depletion. This may have been caused by the failure to jettison the launch shroud completely.

Following insertion, the primary control system was activated in the high-rate mode, using reaction control system (RCS) B-ring. For approximately 1.6 seconds, the pitch, yaw, and roll rates were normal, with normal thruster activity. Following this, rates remained under control but with abnormally high thruster activity and unusual rate indications. The rates were held in the vicinity of their correct values and were adequate to conduct rendezvous. Figure 5.8.4-1(a) shows the thruster firings, angular rates, main bus voltage, and B-ring source pressure following control system activation. The figure also shows that, after about 2 minutes, the rate indications and thruster activity again became more nearly nominal; at about the same time, the B-ring source pressure indicated propellant depletion. Crew reports and motion pictures confirm that the ATDA rates were low during braking and station keeping.

The control system was re-activated following the third rendezvous, using the secondary electronics and rate gyros and the A-ring. The performance in the high-rate mode was similar to that previously obtained using the primary electronics and rate gyros and the B-ring, as may be seen by comparing figure 5.8.4-1(b) with figure 5.8.4-1(a). Several commands were sent to the ATDA to determine the effects of switching rate modes and rigidizing and unrigidizing the docking cone in various sequences. The control system performance for each of these tests was similar to that obtained following the initial activation at insertion. The amount of time required for stabilization varied, but not in a fashion which correlated with the command sequence.

A partially detached, but rigid, shroud would have a significant effect on the vehicle dynamics. The slopes of the rate data were used to compute angular accelerations over the first rate commands at the initial control system activation, based on the indicated sharing of the roll-yaw thrusters. These were compared with the theoretical values with and without the shroud as indicated in the following table.

Direction	Acceleration, deg/sec^2		
	Measured	Theoretical, no shroud	Theoretical, with shroud
Roll	9.8	10.40	8.35
Pitch	8.7	9.09	6.95
Yaw	3.8	4.55	3.48

From these data, it may be concluded that the shroud was having little or no effect on the moments of inertia of the vehicle over the small amount of angular travel required to reach the commanded angular rates, but was subsequently oscillating to produce accelerations leading to control system instability. It has not been positively established that the shroud was the sole cause of the abnormal control system performance. If the shroud had been rigidly attached to the Augmented Target Docking Adapter and normal target stabilization system operation obtained, then, due to the increased inertia, the accelerations should have been lower and the thruster activity therefore should have been less than with the shroud off; however, the commanded rates would have been achieved. A detailed examination of the rate data shows that periods occurred after the beginning of the abnormal behavior period when the rates in all axes did not respond to the thruster firing. Conversely, the data show periods when rates changed and no thruster firing could be correlated with the change.

Figure 5.8.4-2 compares the thruster firing with the deadband operation for selected times during the abnormal behavior period. The figure indicates that disturbance torques were apparently being generated by some unexplained mechanism. Any attitude motions within the deadband of an axis would be less than one degree at lower than 0.4 deg/sec which would be difficult to detect by observation; therefore, the possibility of an intermittent loose connection of the shroud cannot be eliminated.

An analysis was conducted to postulate hardware failures that could produce the flight results. Based on the results of this analysis, it was determined that no single failure or even dual failure could be identified that would result in the flight conditions. It was also noted that the bus voltage, which is sampled once every second, shows a change at the initiation and termination of the increased fuel-consumption period.

To further evaluate this anomaly, the spacecraft contractor will perform analog-computer simulations of the ATDA control system including the effects of the shroud.

The shroud anomaly had little or no effect on the rendezvous radar transponder antenna characteristics. The ATDA was observed visually by the crew at a range of about 50 nautical miles, and the acquisition lights were reported at a range of 25 miles. Review of photographs shows that the acquisition lights were partially obscured by the shroud.

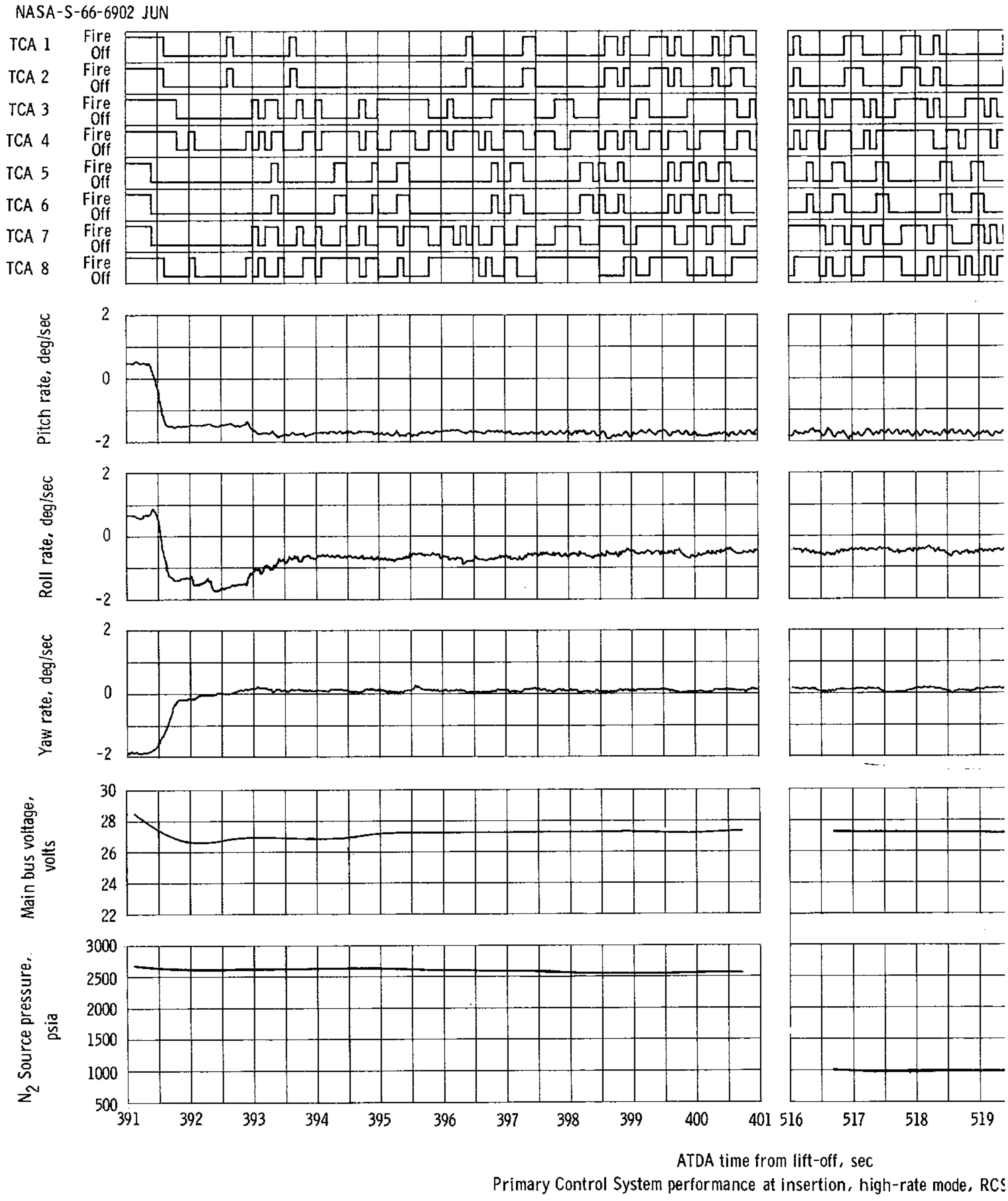

Figure 5.8.4-1. - Comparison of ATDA primary and secondary control system performance.

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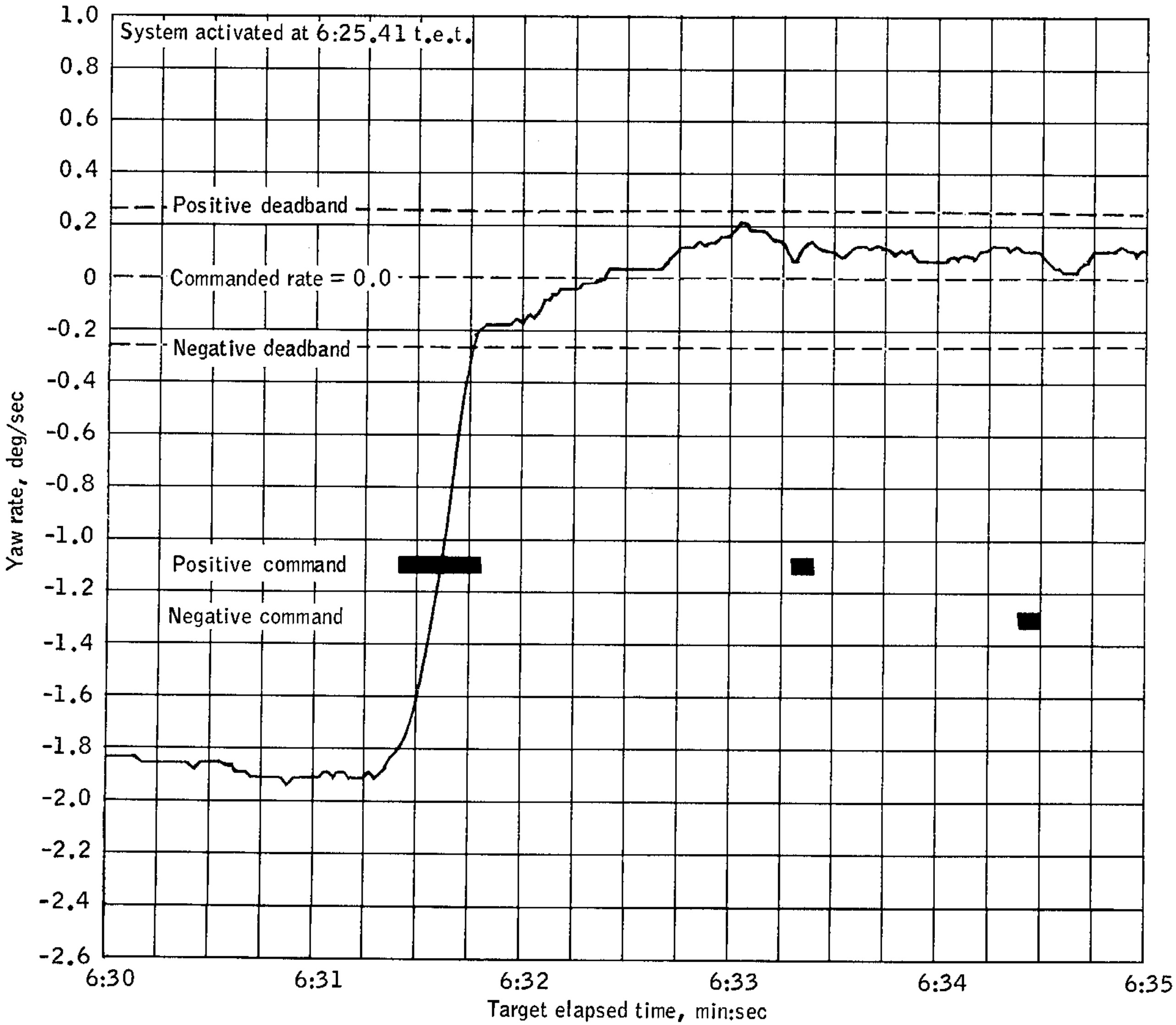

(a) Yaw rate.

Figure 5.8.4-2.- Initial ATDA rate anomaly.

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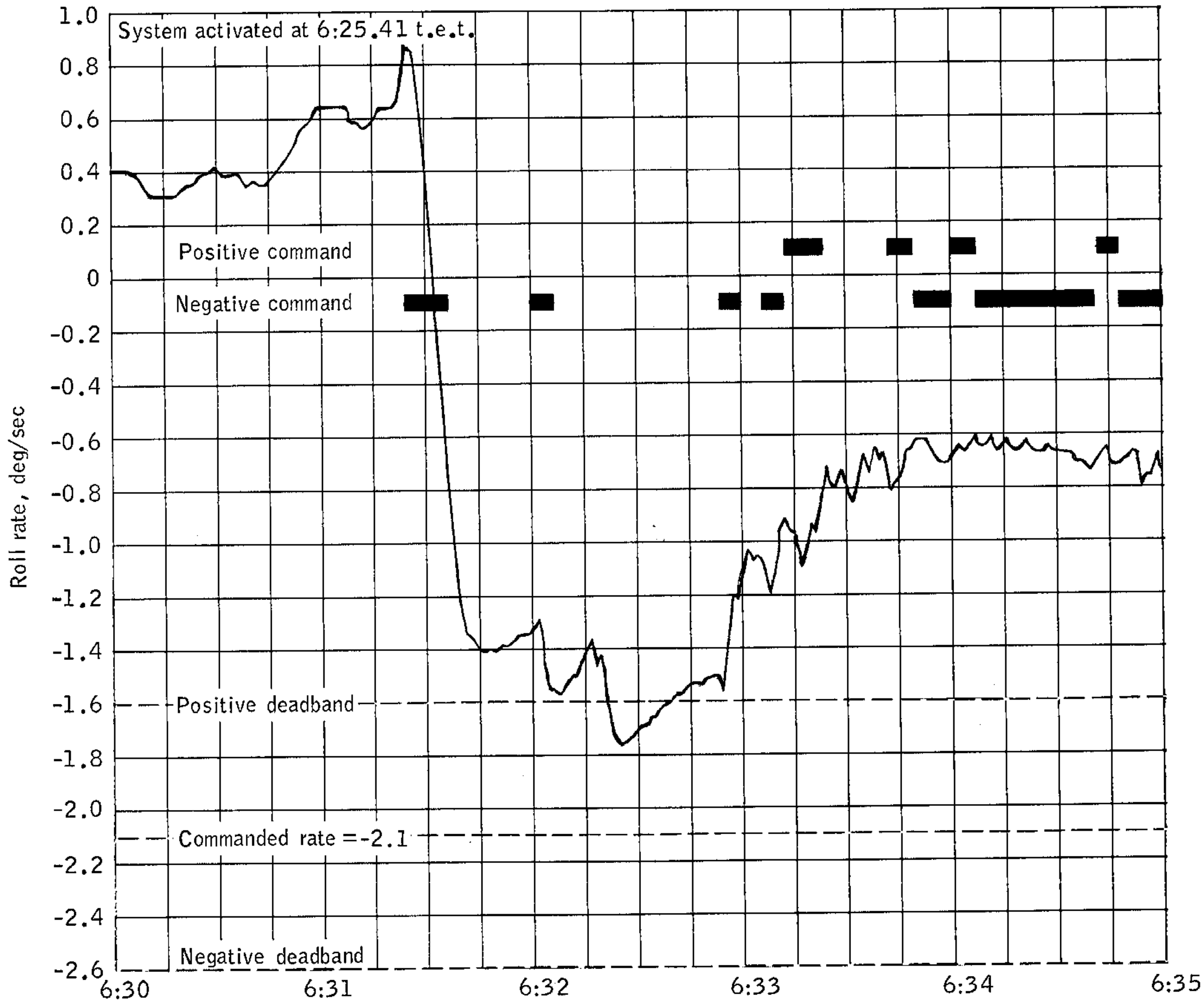

(b) Roll rate.

Figure 5.8.4-2.- Continued.

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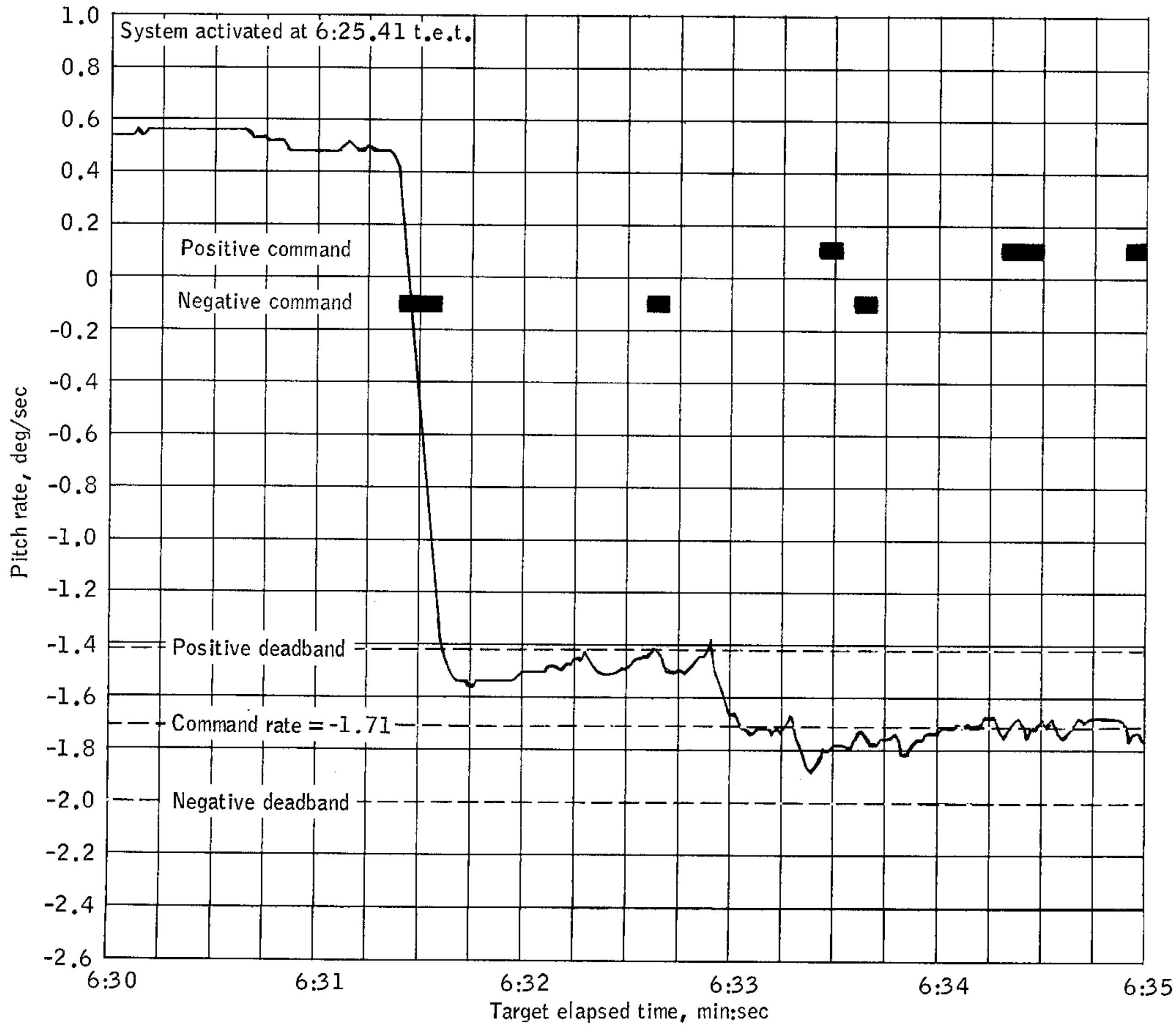

(c) Pitch rate.
Figure 5.8.4-2.- Concluded.

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5.8.5 Electrical System

The performance of the electrical system was nominal throughout the ATDA mission. All sequential events occurred successfully.

After the completion of the Gemini IX-A mission, the Rose Knot Victor (tracking ship) conducted tests at 107:47:53 ATDA elapsed time to determine whether the ATDA batteries were still operable. During this test, the main bus supported a maximum load of 21.5 amperes at 23.6 volts when the TDA rigidizing-unrigidizing sequence was performed for the last time. The common-control and squib-bus batteries appeared normal with voltage levels of 24.5 volts on the common control bus, 25.4 volts on squib bus 1, and 25.2 volts on squib bus 2.

A second postmission test was conducted over Cape Kennedy at 306:34:30 ATDA elapsed time to determine whether the batteries were still operable. This test showed the two 15-ampere-hour common-control-bus and squib-bus batteries apparently to be depleted, as the ATDA systems which were dependent on these batteries did not respond to any commands. However, the C-band transponder operated satisfactorily, which confirmed that the main batteries were not depleted.

After 308:18:30 hours of flight, at the completion of the second check of the batteries, the three 400-ampere-hour main-bus batteries had supplied 808 ampere-hours of current.

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5.8.6 ATDA Propulsion System

There are no known anomalies associated with the ATDA propulsion system. The only unplanned system event was the high rate of propellant consumption. This rate was an order of magnitude greater than the predicted maximum propellant usage rate and was a result of response to the ATDA target stabilization system (TSS) commands rather than an abnormal propulsion system performance. (See section 5.8.3.)

The ATDA was launched with the B-ring of the reaction control system selected for operation after separation. Electrical power to the thruster solenoids was off and the reaction control system heaters were on at this time. At 6 minutes 23 seconds target elapsed time (t.e.t.) the A-package squib valve was fired, thus activating the system. The regulated pressure rose to a maximum value of 298 psia within 3 seconds and subsequently stabilized at 296 psia. The source pressure decreased normally from 3100 to 2690 psia. Power was applied to the solenoids at 6 minutes 32 seconds t.e.t. Single firings of engines 1 and 2 drove the pitch rate toward its high-rate deadband. A continuous firing of engines 3, 4, and 8 drove the yaw and roll rates toward their deadbands. After approximately one second, abnormal activity of all engines was commanded by the TSS, rapidly depleting the propellant in the B-ring. This activity is estimated to be about a 40 percent duty cycle. Usage appeared to be about the same on all engines. However, the exact duty cycle is impossible to ascertain from engine-firing data because of the telemetry limitations. The propellant was expended in 129 seconds, compared with 50 seconds if all eight engines had been on continuously. Near the end of this period, the engine activity diminished rapidly and became normal in appearance.

The A-ring was activated at 22:29:42 spacecraft ground elapsed time (g.e.t.) (69:09:13 t.e.t.). The regulated pressure stabilized normally at a value of 300 psia. The source pressure dropped from 2812 to 2360 psia at activation and stabilized. The decrease in source pressure from 3080 psia at launch to 2812 psia at activation is attributed to a lower temperature at activation. The pressure at activation is equivalent to a temperature of 42° F, which is not unreasonable based on the measured equipment-bay temperatures.

During ATDA tests immediately following activation of the A-ring, three brief periods of heavy thruster activity occurred. The rate of reaction control system source pressure decrease was approximately the same as experienced during the depletion of propellant in the B-ring after ATDA insertion. The reaction control system was next used during a series of tests during ATDA revolution 57 over the United States to evaluate the effects of a rigidized-versus-unrigidized docking cone (and attached shroud) on the control-system stability. There were three periods of heavy engine activity during which the pressurant gas decreased at a rate similar to earlier heavy activity; therefore, approximately a 40 percent duty cycle was commanded. An estimated 12 to 13 pounds of propellant remained in the A-ring after these tests were completed.

The command pilot reported seeing yellowish liquid droplets coming from an ATDA reaction control system nozzle. No definitive explanation of these droplets is yet available. A brownish-yellow color adjacent to some of the nozzles is also visible in the 70-mm photographs made of the ATDA following reaction control system firings. Similar coloration was also discernible in photographs made during the Gemini VII/VI-A mission.

6.0 MISSION SUPPORT PERFORMANCE

6.1 FLIGHT CONTROL

The Gemini IX-A mission was controlled from the Mission Control Center (MCC-H) at the Manned Spacecraft Center, Houston, Texas. This portion of the report is based on real-time observations and may not agree with the detailed postflight analysis and evaluation in other sections of this report.

6.1.1 Premission Operations

6.1.1.1 Premission activities.- The flight control team at MCC-H conducted simulations and provided support to Launch Complexes 14 and 19 during the premission phase. Support was provided for the Final Systems Test; for the Simultaneous Launch Demonstration on May 10, 1966; for the Final Simulated Flight on May 11, 1966; for the launch countdown on May 14, 16, and 17, 1966; and for the launch attempt on May 17, 1966, when the Gemini Agena Target Vehicle failed to achieve orbit. The Gemini IX-A mission was scheduled for May 31, 1966, utilizing the Augmented Target Docking Adapter (ATDA). Support was provided for the ATDA on May 22, 25, and 28, 1966, and for the spacecraft for an additional Final Simulated Flight on May 26, 1966.

6.1.1.2 Documentation.- Documentation for the mission was adequate in all areas. Because of the change in target vehicles, a large amount of documentation had to be revised in a short time period. All mission documentation was updated in a timely manner.

6.1.1.3 MCC/network flight-control operations.- The flight control personnel began deployment to the remote sites on May 2, 1966, and the Manned Space Flight Network went on mission status initially on May 3, 1966. Mission status was terminated on May 17, 1966, after the Gemini IX mission was cancelled and mission status was resumed on May 24, 1966. Flight control personnel return from the Hawaii, Canary Islands, and Guaymas tracking stations to assist in the mission preparation. The Carnarvon, Australia (CRO), team remained on station, and the Rose Knot Victor (RKV) and the Coastal Sentry Quebec (CSQ) tracking ships returned to port for logistics support. The command and telemetry data flow tests between MCC-H and the remote sites were conducted successfully, and the sites were ready to support the Gemini IX-A mission which had been rescheduled for June 1, 1966.

6.1.1.4 Target Launch Vehicle/Augmented Target Docking Adapter countdown.- The Target Launch Vehicle/Augmented Target Docking Adapter (TLV/ATDA) countdown proceeded smoothly and slightly ahead of schedule during most of the tests. At T-275 minutes, the 24 ATDA command relays were confirmed by blockhouse personnel to be in the proper launch configuration, and the primary-execute set and reset commands were transmitted from the MCC-H. At T-260 minutes, the real-time command tape, with the six separation commands, was loaded into the MCC-H command system. The command system was then disarmed until after TLV sustainer engine cutoff (SECO), when it was armed to transmit the separation sequence commands. The alternate impact predictor (IP) high-speed data line dropped out at T-215 minutes. A power supply was replaced at Cape Kennedy, and the IP high-speed data line was back in service at T-208 minutes. At T-116 minutes, the Patrick radar was reported inoperative, but was returned to operational status at T-104 minutes.

6.1.2 Powered Flight

6.1.2.1 TLV/ATDA powered flight.- The predicted TLV lift-off time was 15:00:00 G.m.t. During powered flight, the TLV trajectory was very close to nominal, and all events occurred at the nominal elapsed times. The desired and recorded TLV/ATDA insertion cutoff conditions are shown in the following table.

Condition	Desired	IP (raw)	Bermuda
Mission recommendation . . .	Go	Go	Go
Velocity ratio, V/V_R	1.000	1.000	1.000
Velocity, ft/sec	25 365	25 362	25 364
Flight-path angle, deg . . .	0.0	-0.02	-0.01
Altitude, n. mi.	161.0	161.0	161.0

The resultant orbit based on the transferred Bermuda (BDA) insertion vector was 161.5 by 158.6 nautical miles. Subsequent low-speed tracking data from the Eastern Test Range (ETR) showed the orbit to be 162.1 by 160.8 nautical miles. The TLV/ATDA ascent sequence was nominal up to the point of ATDA separation from the TLV. ATDA separation was initiated by the time-critical sequence of seven real-time commands, with the sixth command being transmitted at vernier engine cutoff (VECO) + 10 seconds and the seventh command being transmitted within the next 2 seconds. The following events should have occurred within 12 seconds of the sixth command (primary execute):

(a) ATDA shroud separation . . . . 0 seconds

(b) ATDA-TLV separation . . . . +2 seconds

(c) Reaction control system B-ring activated . . . . +2 seconds

(d) L-band boom antenna extended . . . . +10 seconds

(e) Status display panel lights, running lights, and acquisition lights on . . . . +10 seconds

(f) Docking cone unrigidized . . . . +12 seconds

(g) Self-reset of the separation sequence relay logic . . . . +12 seconds

All but one of the above sequences functioned normally. This anomaly was the failure of the protective aerodynamic shroud to separate. Telemetry indicated that one or both of the ATDA shroud telemetry disconnect plugs had failed to disconnect properly. The MCC-H along with Cape telemetry personnel confirmed that a disturbance torque was noted at this time on a specific launch-vehicle telemetry parameter. This indicated that at least one of the four explosive bolts that retain the shroud had been detonated at the proper time.

Another anomaly noted early in the ATDA flight was the abnormally high activity of the reaction control system B-ring thrusters. Upon activation of the B-ring at ATDA separation, the target stabilization system (TSS) was to provide rate stabilization of the ATDA in the high-rate mode. The TSS did not respond normally, and the high thruster activity depleted the B-ring propellants in approximately 2 minutes. At propellant depletion, telemetry became noisy and unusable. Just prior to ATDA loss of signal, 20 seconds of telemetry data appeared to be valid. At this time, the telemetry signal received at MCC-H indicated that the TSS high rates of approximately 2 deg/sec in roll and pitch had reduced to the TSS low rates of zero deg/sec, plus and minus the deadband limits of 0.25 deg/sec in pitch and yaw and 0.5 deg/sec in roll.

6.1.2.2 Prelaunch 2 (time period between TLV/ATDA lift-off and Gemini Space Vehicle lift-off).- The Canary Islands (CYI) tracking station experienced an auto-tracking problem with their radar on the first ATDA pass. This meant that the recommended lift-off times and launch window information, as well as the final targeting for the spacecraft launch, had to be based on the ETR (launch phase) radar data and the Carnarvon (CRO) and Woomera, Australia (WOM), radar data.

Telemetry readouts indicated the protective shroud on the ATDA had not jettisoned, and an alternate mission plan was developed for the contingency of the spacecraft being unable to dock with the ATDA.

Final targeting was based on CRO and WOM tracking data, and the recommended lift-off time was 16:38:23 G.m.t. with a biased launch aximuth of 97.7 degrees.

The launch windows associated with these data were as follows:

Latest lift-off time, G.m.t.	Spacecraft rendezvous apogee no.
16:38:58	3
16:40:41	4
16:42:55	5
16:44:09	6

The Agena Ephemeris Data (AED) transfer was performed successfully at T-22 minutes, and validated at T-18 minutes. At T-3 minutes, the final targeting updates to the onboard backup guidance system were not received by the spacecraft from the GE/Burroughs guidance facility. The launch window for that day was violated, and the spacecraft launch was postponed until June 3, 1966, when two launch windows were available. Radar tracking was continued on the ATDA through revolution 3, and recommended lift-off times calculated for the second day launch windows were as follows:

Window	Time, G.m.t.
First pane	13:39:39
Second pane	15:15:13

The TLV was tracked through revolution 3 of the ATDA, and separation studies between the ATDA and its launch vehicle were conducted in the Auxiliary Computer Room (ACR).

To preclude the T-3 minute update anomaly from prohibiting launch on June 3, 1966, several procedures were developed. Sufficient radar tracking allowed flight control personnel to utilize the ACR to predict the required T-3 minute targeting quantities for an on-time launch. These values were then converted to digital quantities that could be accepted by the spacecraft computer in the event the normal update at T-3 minutes could not be transferred. Tests were conducted successfully on June 2, 1966, using both the AED transfer method and the backup procedure described above. The digital quantities were transmitted and verified on the day before launch but were transmitted again with minor corrections at T-15 minutes in the terminal countdown.

6.1.2.3 Final Gemini Space Vehicle countdown.- The terminal phase of the launch countdown was picked up by the MCC-H at T-680 minutes and proceeded ahead of schedule. A problem was discovered at approximately T-120 minutes when telemetry data indicated that the primary horizon scanner on the spacecraft was acquiring intermittent track. The horizon scanner fairing was removed, and the horizon simulator was used to verify the operation of the scanner. All checks were normal and the primary scanner was verified to be operating properly. Another problem noted in the terminal countdown occurred at T-20 minutes when the Booster Tanks Monitor could not communicate on UHF with the spacecraft crew. This was a patching error, and proper procedures should preclude a recurrence.

ATDA tracking data for revolution 29 from the Grand Canary Island defined a requirement for a spacecraft lift-off time of 13:39:33 G.m.t. and a biased launch azimuth of 87.4 degrees. Final targeting quantities based on the CRO and WOM tracking data were as follows:

Recommended lift-off time, G.m.t.	13:39:33
Biased launch azimuth, deg	87.4
Desired velocity, ft/sec	25 723.5

Other spacecraft launch window information associated with the ATDA orbit was as follows:

Latest lift-off time, G.m.t.	Spacecraft rendezvous apogee no.
13:40:10	3
13:41:53	4
13:43:36	5
13:45:19	6

At T-3 minutes, the GE/Burroughs computed launch azimuth transmitted to MCC-H was in agreement with the RTCC calculations; however, the update again did not transfer from the GE/Burroughs guidance facility to the spacecraft computer.

With the ATDA orbit well established after two days of tracking, the ACR, using an ATDA vector from the RTCC and the predicted T-0 time, ran a targeting run at T-3 hours to predict any minor corrections to the four IGS targeting quantities which GE/Burroughs would transmit to the spacecraft at T-3 minutes. The quantities required to enable these corrections were then converted to a digital command format and loaded into the Master Digital Command System (MDCS) in the MCC-H. This command load was then transmitted to the spacecraft computer and verified at T-15 minutes in the terminal countdown. The spacecraft then had a valid load which was good for the predicted T-0 time when GE/Burroughs attempted to load the spacecraft backup guidance system with the final targeting quantities at T-3 minutes.

After the GE/Burroughs transmission failed to reach the command transmitter at the T-3 minute attempt, the input to the digital command system from the GE/Burroughs buffer was disabled. This also inhibited GE/Burroughs transmissions of the plus-time updates to correct the Spacecraft Inertial Guidance System (IGS) azimuth alignment. These updates are necessary to ensure that the IGS out-of-plane errors at insertion are small. Inhibiting them resulted in the Spacecraft Inertial Guidance System indicating a large out-of-plane velocity at SECO + 20 seconds, when in fact the out-of-plane velocity was very small.

This procedure was acceptable for two reasons:

(a) The large out-of-plane velocity that might result if a switchover occurred was acceptable.

(b) A well-established ATDA orbit enabled the ACR to run the targeting program far enough in advance of the predicted T-0 time so that the four IGS quantities could be checked, converted to digital parameters, loaded into the MDCS, transmitted to the spacecraft computer at T-15 minutes, and validated.

6.1.2.4 Gemini Space Vehicle powered flight.- The Gemini Space Vehicle lift-off occurred on time at 13:39:33 G.m.t. Impact predictor (IP) smooth data were good at lift-off and remained solid throughout powered flight. GE/Burroughs achieved solid lock early and was selected as the primary data source at 55 seconds into Stage I flight. The initial flight-path angle during Stage I operation was approximately 0.02 degree lower than the preflight calculated nominal at its maximum, and at staging it was approximately 0.2 degree high. Data quality during Stage II operation was very good, and all sources agreed. The last 10 seconds of powered flight showed a low trend (approximately 0.18 degree low in flight-path angle) on plotboard I (V/VR versus flight-path angle) until a velocity ratio of about 0.975 was reached. At this time, the trajectory began to converge to the nominal. During powered flight, the maximum deviation from nominal altitude was indicated to be less than one nautical mile. The cutoff appeared slightly low in flight-path angle on the V/VR versus flight-path-angle plotboard, but well within the acceptable region for using the onboard Insertion Velocity Adjustment Routine (IVAR) solution. From the sources shown in the table, the RTCC computed the following trajectory parameters at cutoff.

Source	Velocity, ft/sec	Flight-path angle, deg	Altitude, n. mi.	Wedge angle, deg
GE/Burroughs	25 694	-0.06	86.8	0.02
IP Smooth	25 712	-0.22	86.8	0.03
IP Raw	25 790	-0.55	86.6	0.04
Bermuda	25 702	-0.10	86.8	0.05

Prior to performing the IVAR maneuver, the crew read out two spacecraft computer values as follows:

Velocity to be gained at perigee, ft/sec (Core 95) . . . +0021

Total inertial velocity, ft/sec (Core 72) 25 714

Following the IVAR maneuver, the crew read out computer address 95 as -0002 and computer address 72 as 25 749. This was in excellent agreement with Bermuda tracking data. High-speed averaged data from Bermuda after thrusting indicated the velocity was 25 749 ft/sec, the altitude was 86.0 nautical miles, and the flight-path and wedge angles were 0.0 degree.

6.1.3 Spacecraft Orbital Flight

The Bermuda high-speed insertion vector was transferred to the orbit phase and predicted an initial orbit of 85.9 by 149.7 nautical miles. Low-speed tracking from the Grand Canary Island station gave an orbit of 85.9 by 146.6 nautical miles and indicated a required plane change maneuver of approximately 10 ft/sec.

The Grand Canary Island station had no indication of PCM tape motion on their first pass. A check with the PCM ground stations showed the tape motion parameter reading 77 percent of full scale. Further checks at the MCC-H indicated that the remote sites were set-up to give an ON indication at 95 percent of full scale while the MCC-H indication was driven at 50 percent of full scale. Remote site personnel were instructed to trigger their indicators at 50 percent of full scale, and the problem was solved.

The associated phasing maneuver is given below, and, for comparison, the prelaunch nominal values are included:

	Prelaunch nominal	Bermuda tracking	Canary Islands update to crew
Ground elapsed time of maneuvers, min:sec	49:25	49:05	49:03
ΔV, ft/sec	55.2	73.6	75.0
Thrust time, sec	74	99	100

Because of the negative flight-path angle at insertion, the maneuver was performed at approximately 15 degrees from the spacecraft line of apsides. The effect of traveling through perigee (which implies a high catch-up rate over an increased time span), coupled with maneuvering off the line of apsides, caused the phasing maneuver to increase over the prelaunch value by 19.8 ft/sec. The crew report of the phase-adjust maneuver over Carnarvon on revolution 1 indicated that the maneuver was executed on time and that the residuals were nulled to the proper values. The accelerometer bias check was also performed over Carnarvon, and a decision was made to update the accelerometer biases over the United States on revolution 1. All three biases were uplinked to the spacecraft over the United States on revolution 1 and were verified to be correct. After the update, the Y-axis and Z-axis looked very good, but the X-axis was still slightly off. The decision was made to proceed with the rendezvous with the current X-axis bias, because the error was very small.

A preliminary two-impulse maneuver plan was generated based on Canary Islands tracking, and the plan indicated an optimum solution for the specified inputs at an altitude differential (Δh) of 11.0 nautical miles.

The time for the coelliptic maneuver occurred after the Carnarvon pass and the post-maneuver tracking from the Australian site could not be included in the terminal phase update. The total ΔV for this plan was 107 ft/sec (excluding the phase-adjust maneuver (N_{C1})), with the time for the corrective combination maneuver (N_{CC}) of 2:11:18 ground elapsed time (g.e.t.), and 2:36:02 g.e.t. for the coelliptic maneuver (N_{SR}).

Additional two-impulse calculations were made to move the maneuver earlier and take advantage of the Carnarvon post-maneuver tracking following the N_{SR} maneuver. The Carnarvon tracking was used to base another set of two-impulse calculations, and the final N_{CC} and N_{SR} updates were based on California and White Sands tracking.

The maneuvers for this plan were as follows:

	N_{CC}	N_{SR}
Maneuver-initiate time, g.e.t, hr:min:sec	1:55:17	2:24:51
$\Delta\overline{V}$, ft/sec	14.6	54.0
Pitch, deg	+44.1	-40.7
Yaw, deg	-66.9	-2.8
Thruster	Aft	Aft

	N_{CC}	N_{SR}
ΔV_X, ft/sec	+4.1	+40.9
ΔV_Y, ft/sec	-10.2	+35.2
ΔV_Z, ft/sec	+9.7	+2.0

These maneuvers were passed to the crew over Texas on revolution 1, along with the Δh (12 nautical miles) for the coelliptic orbit. The total ΔV for this plan was 129 ft/sec, including terminal phase but excluding the phase-adjust maneuver $\left(N_{C1}\right)$.

Over Hawaii on revolution 2, the crew reported a problem with the start-compute discrete in the onboard computer. It appeared that the start-compute discrete was latched up continuously. When the crew switched to the catch-up mode, the computer running light would come on. If the computer was placed in the rendezvous mode, the computer would collect only eight radar data points, compute the required ΔV for rendezvous, and go into terminal phase calculations. These functions do not normally occur until the START button is depressed. The crew also reported that the radar was working normally and that the optical and radar boresight agreed quite closely. A time for entering the rendezvous mode was calculated on the ground and transmitted to the crew. This time would circumvent the start-compute-discrete anomaly by allowing the computer to accept the desired eight radar data points and function normally for terminal phase. The crew switched to the rendezvous mode at the pre-established time, and the computer performed normally during terminal phase. Also, the start-compute-discrete anomaly cleared itself during the terminal phase. After rendezvous, the crew reported that the computer was working normally in all modes. The exact cause for the start-compute-discrete anomaly could not be determined, and the anomaly did not reoccur during the remainder of the mission.

Terminal phase initiate (TPI) nominally was to occur 3.5 minutes before spacecraft darkness, at 3:37:10 g.e.t. Tracking data after the N_{CC} and N_{SR} maneuvers through Woomera on revolution 2 indicated that TPI should occur 5 minutes 3 seconds earlier than nominal. The composite Hawaii, California, and Eglin solution was used to give the crew their final TPI update. These data showed that the TPI should occur at 3:35:35 g.e.t., which was only 1 minute 35 seconds earlier than desired.

The update parameters for TPI were passed to the crew over the United States at the beginning of revolution 3 and are shown in the following table:

Maneuver-initiate time, g.e.t., hr:min:sec	3:35:35
$\Delta\bar{V}$, ft/sec	26.8
Pitch, deg	+30.1
Yaw, deg	+4.8
Thruster	Aft
ΔV_X, ft/sec	+23.1
ΔV_Y, ft/sec	-13.4
ΔV_Z, ft/sec	-2.0

Over Bermuda on revolution 3, radar transponder signals were received simultaneously from both the ATDA and the spacecraft. This did not create any problem; however, the ATDA C-band transponder was commanded off over Carnarvon during spacecraft revolution 3.

The spacecraft and target trajectories were computed in the ACR, based on White Sands tracking after N_{SR}, and the printout indicated that the two orbits were coelliptic within ±0.1 nautical mile. Numerous readings were given by the flight crew after radar lock-on, which occurred at approximately 2:14:00 g.e.t. These readings agreed very closely with the ground-computer printout.

Over Hawaii on revolution 3, the crew reported they were station keeping with the ATDA at a distance of approximately 20 feet and that the shroud was still attached to the ATDA. After making a close inspection, they further reported that the explosive bolts had fired. However the wires to the pyrotechnics were not disconnected and the bandclamp was still in place. Over the United States on revolution 3, the ATDA acquisition lights were commanded off, and the docking cone was rigidized and unrigidized by ground commands in an unsuccessful attempt to free the shroud. High and low rates were also commanded on and off in an attempt to free the shroud, but without success. This test was visually observed by the crew while they were station keeping with the ATDA. Based on the crew reports and ground evaluation, it was decided that docking could not be accomplished because the shroud was still attached.

A radial separation maneuver for the equi-period rendezvous was computed to be required at 10 minutes prior to the next spacecraft darkness. The update quantities for the radial separation maneuver were passed to the crew over the United States at the beginning of revolution 4. These quantities were as follows:

Maneuver-initiate time, g.e.t., hr:min:sec	5:01:00
Thrust time, sec	35
ΔV, ft/sec	20.0
Pitch, deg	-90
Yaw, deg	0
Thrusters	Forward

This maneuver was performed on time and accurately by the crew, and, as a result, the planned horizontal-adjust maneuver was not required at the time the target and the spacecraft reached the same altitude, one-half orbit later.

The crew reported that the equi-period rendezvous with the ATDA had been completed over the Rose Knot Victor (RKV) tracking ship at 6 hours 36 minutes g.e.t. It was also noted by ground personnel during this pass that the tape motion indicator indicated the tape recorder was not functioning properly.

The Orbit Attitude and Maneuver System (OAMS) propellant required for the remainder of the flight plan was being monitored very closely. An indication of 40 percent remaining on the onboard gage was necessary at the end of the first day to complete the remaining flight-plan items. Over the RKV on revolution 5, the crew was advised of this requirement, and also of the separation maneuver to set up the phasing for the rendezvous from above on the following day. The separation maneuver required a ΔV of 3.7 ft/sec at 7:14:58 g.e.t. Subsequent to the maneuver, it was decided to position the TPI and TPF maneuvers to occur in daylight.

Over the Hawaii station on revolution 5 (7:38:20 g.e.t.), a tape playback was received which indicated that the playback speed was much lower than normal. Various playback speeds and ground-station configurations were used in attempts to reduce the data but were all unsuccessful. Over the Coastal Sentry Quebec (CSQ) on revolution 7 (10 hours 30 minutes g.e.t.), a continuous playback command was transmitted in an attempt to reposition the tape. This was done so that if the recorder were functioning normally in the record mode, the EVA data could be recorded. Continuous dump modulation (unusable) was received until Carnarvon on revolution 13, when the playback command was removed and the tape recorder turned off. With the apparent loss of the onboard tape recorder, all OAMS firing time analysis was estimated rather than measured.

After the decision was made to relocate the terminal phase of the rendezvous from above to occur in daylight, the ACR began generating a maneuver plan to achieve that objective. Due to limited tracking coverage prior to and during the midcourse maneuvers, it was decided to reposition the maneuver points. Moving each maneuver ahead in time by approximately 23 minutes allowed more optimum ground coverage and also decreased the magnitude of the second phasing maneuver $\left(N_{C1}\right)$. Both the phase-adjust maneuver $\left(N_{C1}\right)$ and the height-adjust maneuver $\left(N_{H1}\right)$ were updated to the crew over Antigua on revolution 12. The updates were as follows:

	N_{C1}	N_{H1}
Maneuver-initiate time, g.e.t., hr:min:sec	18:23:19	19:08:16
$\Delta\bar{V}$, ft/sec	2.0	17.0
Pitch, deg	0	0
Yaw, deg	0	0
Thruster	Aft	Aft
ΔV_X, ft/sec	2.0	17.0
ΔV_Y, ft/sec	0	0
ΔV_Z, ft/sec	0	0

Over Carnarvon on revolution 12, the crew reported that N_{C1} had been executed on time. The elevation angle to the spacecraft during the

Carnarvon pass was too low to obtain good tracking data. However, tracking data from the Woomera station confirmed the nominal N_{C1} maneuver. Over Antigua during revolution 13, the crew reported that N_{H1} had been executed nominally. In order to update N_{SR} with tracking data after completion of the N_{C1} and N_{H1}, the Antigua track was interrupted, and an updated coelliptic maneuver was calculated in the ACR based on this vector. These data were passed to the crew over the Canary Islands during revolution 13 and were as follows:

Maneuver-initiate time, g.e.t., hr:min:sec	19:54:24
$\Delta\overline{V}$, ft/sec	14.4
Pitch, deg	-38.1
Yaw, deg	180
Thruster	Forward
ΔV_X, ft/sec	11.3
ΔV_Y, ft/sec	8.9
ΔV_Z, ft/sec	0

Following the N_{SR} maneuver, the differential altitude (ΔH) varied from 8.3 to 6.7 nautical miles. This anomaly was due to radar tracking errors in the ATDA Ascension revolution 38 vector and the spacecraft Antigua revolution 13 vector. Radius from center of the earth to the target was updated as 7 496 600 yards and the elevation angle to align the platofrm was -9 degrees in pitch. The Canary Islands tracking data contained several bad points and were not used to update the spacecraft ephemeris. The TPI update was based on Antigua data from revolution 13 and was transmitted to the crew over Carnarvon on the same revolution. The update was:

Maneuver-initiate time, g.e.t., hr:min:sec	20:55:28
$\Delta\overline{V}$, ft/sec	16.7
Pitch, deg	-26.7

Yaw, deg	-172.1
Thruster	Aft
ΔV_X, ft/sec	-14.7
ΔV_Y, ft/sec	+7.5
ΔV_Z, ft/sec	2.0

This update indicated that TPI would occur very near to spacecraft sunrise. However, at the time of the update onboard range and elevation readings disagreed with the ground-computed trajectory, and computations by the crew indicated that TPI would occur close to the nominal time – 10 minutes into daylight. Over Antigua on revolution 14, the crew initated TPI, 10 minutes 45 seconds after spacecraft sunrise.

All of the maneuvers for the third rendezvous had to be computed by the ACR due to the inability of the Real Time Computer Complex (RTCC) to generate a rendezvous plan in which the spacecraft is ahead and above the target at TPI.

After the third rendezvous was completed at approximately 21 hours 20 minutes g.e.t., the command pilot recommended that extravehicular activity (EVA) be postponed until the third day as the crew were extremely tired. The ground controllers concurred with this recommendation and advised the crew to perform a 3-ft/sec retrograde maneuver at thier convenience to provide positive separation from the ATDA. This maneuver was performed at 22 hours 59 seconds g.e.t.

Over the United States during revolution 15, an accelerometer bias check was performed. Based on this and previously collected data, the biases of the X and Y axes were updated for the final time on the subsequent pass over the United States. These biases were again rechecked by telemetry on subsequent days of the mission and found to be very stable.

A 3-hour rest period followed for the crew, and the remainder of the second day was spent conducting experiments. Six sequences of the UHF-VHF Polarization (D-14 experiment) were completed satisfactorily, and three complete and one partial sequence of the Airglow Horizon Photography (S-11) experiment were conducted successfully.

The oxygen crossfeed was opened, and the pyrotechnic valve on the hydrogen tank activated over the United States on revolution 15 at 23 hours 58 minutes g.e.t. The pressures in the two oxygen tanks equalized as expected, and the crew reported they heard the hydrogen squib ignite.

Over the Hawaii station during revolution 17, the crew reported that the Reactant Supply System (RSS) hydrogen quantity indicator was indicating zero. A calibration check on the gaging system was performed during the following stateside pass with a calibration-1 indication of 22.8 percent and a calibration-2 indication of 76.4 percent. This verified the integrity of the quantity control unit and isolated the failure to either the tank sensor or the wiring between the tank and the quantity control unit. This failure posed no problem due to the backup method of calculating hydrogen quantity in the RTCC using pressure and temperature of the stored hydrogen.

During the second night, some concern arose as to the amount of storage room remaining for the fuel-cell product water. Calculations and estimates at 35 hours g.e.t. indicated that the higher-than-predicted power profile and the lower-than-predicted water consumption by the crew had combined to produce an 18-pound increase of water in the tanks over what was expected. Over the United States on revolution 28, the crew was advised to transfer four pounds of water into the evaporator while the MCC-H monitored the drop in water pressure. Using these two values, additional calculations indicated that enough storage space remained to conduct EVA without any concern about completely filling the tank during that period.

During EVA preparation, the crew reported over Tananarive during revolution 30 (47 hours 7 minutes g.e.t.) that, while in the platform mode, the spacecraft had started to roll with rates as high as 30 deg/sec. The crew indicated that the problem was with the OAMS thrust chamber assembly (TCA) 3. A few minutes later, over the Carnarvon station, the crew also reported that, in order to get the FDI nulled, it was necessary to fly toward the yaw indicator and away from the pitch and roll indicators. The flight controller advised Carnarvon to ask the crew to check the scanner-heater curcuit breaker. When the crew closed that circuit breaker, the problem was cleared. This circuit breaker controls power to the pitch and roll FDI phase-reversal circuit. The scanner-heater circuit breaker is located on the right-hand switch panel and was inadvertently hit and opened by the pilot during the EVA preparation.

Just prior to EVA, the tape recorder was turned on over Carnarvon on revolution 31. The tape motion indicator illuminated, and the tape recorder appeared to be working normally. The crew was also given a go for cabin depressurization. The spacecraft hatch was opened over the Canton Island station at 49 hours 23 minutes g.e.t.

The first dayside period of EVA was conducted normally, and all spacecraft systems functioned well. Over Carnarvon on the next revolution, the command pilot reported that the extravehicular pilot was having difficulty seeing due to visor fogging. During the Astronaut Maneuvering Unit (AMU) checkout and donning phase, the extravehicular pilot experienced some difficulty unstowing one of the attitude-control arms. Communications between the command pilot and the extravehicular pilot were somewhat garbled when the pilot was disconnected from the spacecraft electrical umbilical. A procedure had been developed for the pilot to remake the spacecraft electrical umbilical and move away from the adapter area in the AMU and conduct a voice check with the command pilot. Fogging on the pilot's visor increased to approximately 75 percent, and oxygen high flow, along with several rest periods, was utilized unsuccessfully to provide clearing. The command pilot recommended that the AMU portion of the EVA be eliminated due to the fogging on the visor. The flight controller agreed with this recommendation, and the pilot began ingress. Hatch closure occurred over Antigua at 51 hours 28 minutes g.e.t. During the stowage of EVA equipment, the RCS A-1 pitch circuit breaker was inadvertently opened by one of the crew members. The flight controller advised the crew, and the circuit breaker was closed.

The AMU systems remained stable, and the AMU was not jettisoned from the equipment adapter.

Over the Coastal Sentry Quebec tracking ship on revolution 35, the crew transferred an additional four pounds of water, some of which they dumped and some of which they drank, which ensured that no fuel-cell product-water storage problems would occur during the remainder of the mission.

It was then decided to conduct the orbit-shaping maneuver with the remaining OAMS fuel to position retrofire for 46-1 so that it would occur at the 270-degree true anomaly. At that time it was calculated that 11 pounds of OAMS fuel was available for the maneuver. This would leave a full reserve fuel tank plus 10 pounds for gaging uncertainty. The orbit-shaping maneuver parameters were:

Maneuver-initiate, g.e.t., hr:min:sec	53:41:35
ΔV, ft/sec	25
Pitch, deg	0
Yaw, deg	180
Thrusters	Aft

The crew reported that at the completion of this maneuver, the OAMS reserve tank pressure had dropped from 300 psi to 285 psi. This indicated that the main fuel tank was empty; however, subsequent to this report the reserve tank pressure increased back to the original value of 300 psi. During revolution 43 over United States the OAMS reserve tank pressure dropped to 250 psi, indicating an empty main fuel tank, and the reserve tank was activated. Ground calculations at that time correlated very well with the actual amount of known fuel in the reserve tank.

A procedure for loading Module IV in the spacecraft computer was transmitted to the crew over the CSQ on revolution 35. This procedure would assist the crew in identifying a start-compute problem should it recur with Module IV loaded. A contingency procedure for starting the computer at retrofire was also transmitted to the crew in the event the start-compute problem arose. This problem did not recur, however, so the contingency procedures were not implemented.

Module IV was loaded by the crew during revolution 36. The 46-1 preretrofire command load and time-to-retrofire (Tr) were transmitted and verified over the CSQ on revolution 36.

Between 62 hours 54 minutes g.e.t. and 63 hours 27 minutes g.e.t., one of the crew apparently turned the suit temperature control valve to full off. The immediate effect of this was a rise in the suit heat-exchanger inlet temperature, which increased 35 to 40° F over a 3-hour period. The suit-inlet temperatures followed this trend and reached 93.3° F (command pilot) and 93.8° F (pilot) by the time the crew was awakened at Carnarvon at 66 hours 39 minutes g.e.t. An hour later at 67 hours 24 minutes g.e.t. the suit heat-exchanger was once again in the loop with the suit heat-exchanger temperature being 44° F, and suit-inlet temperature being 58° F for the command pilot and 56° F for the pilot. A decrease in cabin pressure from 5.0 psid to 4.68 psid between Carnarvon and the United States was also noticed at this time. It was believed that because the suit heat-exchanger was turned off during the night, a large moisture buildup occurred in the cabin and suit loops and that, when the moisture was exposed to the operating heat exchanger, enough moisture was condensed to cause the drop in cabin pressure.

The accelerometer biases were continually checked by telemetry after the spacecraft was powered up on the day of reentry. All axes were correct and stable.

Tracking data from Carnarvon (revolution 42) through Bermuda (revolution 44) indicated the ground elapsed time of retrofire computed (GETRC) was 71:46:43. Canary Islands radar data changed the retrofire time by one second (GETRC = 71:46:44), and all subsequent tracking agreed perfectly.

Over the United States during revolution 43, the crew reported that they had bumped the electronic-timer circuit breaker to the OFF position and the T_r was lagging by 2.25 seconds. However, it was decided not to update it at that time, because the final retrofire update and T_r were scheduled to be transmitted on the next pass over the United States. The Reentry Control System (RCS) was armed over the United States on revolution 44, and the 46-1 preretrofire load and T_r were transmitted and verified to be correct.

6.1.4 Reentry

Retrofire occurred at 13:26:17 G.m.t. (71:46:44 g.e.t.) on June 6, 1966. The crew reported that all four retrorockets had fired on time and resulted in Incremental Velocity Indicator (IVI) readings of 296 aft, 4 right, and 125 down. Subsequent telemetry from Hawaii showed that the velocity changes during retrofire were 297.5 aft, 3.9 right, and 126 down. These values were utilized in computing the backup guidance quantities.

After blackout, the crew reported that the Flight Director Indicator (FDI) needles were nulled and that all systems were operating normally. After loss of telemetry signal, the crew reported that the onboard-computer solution was indicating that the landing point would be approximately 3 nautical miles downrange from the target point. Final telemetry indications received at the MCC-H showed that the crossrange error was 1.91 nautical miles and the downrange error was 1.38 nautical miles.

6.1.5 Augmented Target Docking Adapter Orbital Flight

Over the Canary Islands on the first pass, a planned command sequence was initiated to properly configure all ATDA command relays following the events occurring at ATDA separation from the Target Launch Vehicle. During the remainder of revolution 1, ATDA operations were primarily confined to further analysis of the no-shroud-separation indication and the unexpected depletion of the propellant in the B-ring of the reaction control system.

The Gemini IX-A launch was postponed on June 1, 1966, when the launch window was violated. At this time, ATDA activities centered in the following areas:

(a) Analysis of the unexpected fuel depletion in the reaction-control-system B-ring and the apparent low rates in the TSS.

(b) Proper configuration of the ATDA acquisition and running lights for the M=3 rendezvous and the equi-period rendezvous.

(c) Maintaining a minimum electrical load on the critically limited common control bus that was powered by the two squib batteries.

(d) Analysis of the ATDA shroud installation to determine what action, if any, would permit its removal.

The ATDA was powered down for the next two revolutions with the exception of telemetry and the C-band transponder, and these were commanded off over Texas on ATDA revolution 3.

During ATDA revolution 4 over the Hawaii station, the telemetry was commanded on, and the separation command sequence reinitiated in an unsuccessful attempt to obtain shroud separation. The pulse bus was inadvertently left on at loss of signal (LOS), placing an undesired load on the control bus. This was corrected by the RKV a few minutes later when the ATDA pulse bus was disarmed by ground commands.

Over Hawaii on ATDA revolution 5, the primary TSS, and then the secondary TSS, were commanded on without the reaction control system power on. The objective of this test was to determine the status of the secondary TSS operation and determine whether both systems would indicate the same attitude rates. Both systems indicated the same rates. The C-band transponder was commanded on over Carnarvon on ATDA revolution 15 to obtain tracking data for an ephemeris update. Over the United States on revolution 16/17, another test on the TSS was conducted to evaluate the secondary TSS capability prior to activating the reaction-control-system A-ring. Primary TSS and secondary TSS were commanded on with the different rates selected. Reaction-control-system power was then commanded on and the empty B-ring was selected. Thruster activation signals were analyzed for the proper operation. Both systems appeared to be responding normally when each rate was outside its deadband, except for the thruster 7 solenoid driver in the primary TSS. This circuit had been working properly during the separation sequence, but was now open. The rates measured from the primary and secondary TSS both appeared to be valid, and deadbands for high rates and low rates appeared normal. Over Hawaii on ATDA revolution 18, the C-band transponder was commanded off and remained in that configuration until revolution 26 over the Canary Islands when it was turned back on. Over the United States on ATDA revolution 30, the secondary TSS was commanded on, with the reaction-control-system power off, again with no change in the ATDA rates. The separation sequence was repeated over the Hawaii station on ATDA revolution 31, and again the no-shroud-separation indication was present. On ATDA revolution 32, the Gemini IX-A crew advised the Hawaii station that the shroud was being held together by the restraining band.

They also advised that the ATDA roll rate would be acceptable for docking if the shroud were completely separated. Over the United States during the same revolution, the acquisition lights were turned off and the ATDA docking cone rigidized and unrigidized in an unsuccessful attempt to free the shroud. Disturbance torques of the rigidize/unrigidize motor caused pitch and yaw rates to begin oscillations between ±2.4 deg/sec. Roll rates remained at -2.2 deg/sec. During ATDA revolution 33 over the United States, the crew transmitted the L-band command to select TSS low rates and turn off the acquisition lights.

A test was conducted on ATDA revolution 43 over the United States in another unsuccessful attempt to separate the shroud. First, the reaction-control-system A-ring was armed, the docking cone was unrigidized, and the secondary TSS selected. Then the reaction-control-system power was turned on for 9 seconds. During this 9-second period, activity was observed on all thrusters with short pulse durations along with excessive propellant usage. Reaction-control-systems power was turned off, and the thruster activity and propellant usage stopped. Control power was turned back on, and the TSS functioned normally. The TSS high rates were then selected, and thruster activity and propellant usage were observed for a 5-second period, at which time it stopped without any corrective action. An additional TSS test was conducted over the United States on ATDA revolution 57/58 to determine whether a rigidized or an unrigidized docking cone had any effect. It was concluded that the rigidizing and unrigidizing had no direct effect. The exact cause of the TSS problem could not be determined during the mission.

The ATDA was monitored by remote sites until ATDA revolution 63 when all sites except the RKV were released. During its return to port, the RKV monitored the ATDA and powered down the ATDA to a minimum-power configuration on revolution 78.

The last command activity over the United States was on revolutions 94 and 96 when the C-band transponder was turned on by the Cape for two revolutions to provide a beacon for tracking-range calibration.

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6.2 NETWORK PERFORMANCE

The network was placed on mission status for Gemini IX on May 3, 1966, and supported the Gemini IX and Gemini IX-A missions satisfactorily.

6.2.1 MCC and Remote Facilities

The network configuration and general support required for each station are indicated in table 6.2-I. After the unsuccessful launch attempt of May 17, 1966, Guaymas support was released with the exception of air-to-ground and acquisition aid. Figure 4.3-1 shows the worldwide network stations. In addition, 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 track of the Gemini Launch Vehicle (GLV), Target Launch Vehicle (TLV), Augmented Target Docking Adapter (ATDA), and spacecraft.

6.2.2 Network Facilities

Performance of the network is reported on a negative basis by system and site. All performance not discussed in this report was satisfactory.

6.2.2.1 Remote sites.-

6.2.2.1.1 Telemetry: The telemetry ground stations supporting Gemini IX-A had no significant equipment problems during the mission, with one exception. During spacecraft revolution 1, Carnarvon had a data dropout and was requested by flight control to switch to PMC2. Because the station was being configured for Apollo, the switch-over could not be made, and approximately one minute of data was lost. The short time available between May 17, 1966, and June 1, 1966, was not sufficient to allow all the telemetry data-handling program to be changed to the ATDA format. Consequently, the telemetry data-handling capabilities were somewhat less than normal. The ATDA 2-kbps data from the Bermuda and Texas stations could be sent to MCC-H. The Eastern Test Range (ETR) downrange stations could handle either Gemini or ATDA 40.8 kbps data, but not both simultaneously. There were a few minor procedure and documentation problems, but these were quickly solved and had no serious impact on the mission.

6.2.2.1.2 Radar: Several minor radar equipment problems resulted in the loss or rejection of tracking data during the mission. Canary Island radar was not used for the first ATDA pass because of a problem in the ranging system, and the data from ATDA revolution 28 were rejected by the MCC-H computers because of an alignment problem. Both problems were corrected prior to the next pass. Canary Island data for spacecraft revolution 11 were partially rejected because of an intermittent problem in the radar-data formatting equipment. This problem was also corrected prior to the following revolution. The Eglin radar did not track the ATDA on ATDA revolution 43 because of range gear drive motor and digital-data power supply problems, which were corrected prior to revolution 44. Canary Island data on spacecraft revolution 28 were rejected due to the failure of an encoder which was replaced before revolution 29. Woomera data on spacecraft revolution 44 were rejected. No cause has yet been found. Other radar problems encountered were the result of spacecraft orientation, pointing data errors, and the non-availability of the computer to provide pointing information. As a rule, these problems resulted in only a partial loss of data on a single pass.

6.2.2.1.3 Command: The spacecraft onboard guidance equipment is normally updated at T-3 minutes. This update capability provides the required launch azimuth to the Inertial Guidance System and is mandatory for launch. The data link delivering the updated data from the GE/Burroughs guidance equipment to the command transmitter failed at the critical T-3 instant and caused the mission to be postponed for 2 days. Two digital logic modules in the buffer which interfaces the GE/Burroughs equipment with the command system were diagnosed to be faulty and replaced. Procedures were worked out to backup the guidance update capability in case this link should fail again, as it did on the second launch attempt. The backup procedure permitted the countdown and mission to continue. The actual cause of these failures has not been identified. The situation is presently being investigated by Goddard Space Flight Center (GSFC) and will be reported at a later date.

No significant command problems occurred during the flight.

6.2.2.2 Remote Site Data Processor.- Several minor Remote Site Data Processor (RSDP) program problems existed during the Gemini IX-A mission. However, these were solved by using procedures which were developed and issued to the sites. There were several cases of the RSDP faulting for no apparent reason, but these cases were all under different conditions and none of the faults could be duplicated. These anomalies are under investigation.

6.2.2.2.3 Goddard Space Flight Center: There were no significant problems with the GSFC computer during the mission.

6.2.2.2.4 Real Time Computer Facility: There were no significant problems with the AFETR Real Time Computer Facility (RTCF) computers during the mission.

6.2.2.3 Communications.-

6.2.2.3.1 Ground communications: The Bermuda undersea cable suffered a break on April 28, 1966, and was repaired on May 4, 1966. The alternate command line to Bermuda had numerous noise problems during prelaunch activities. The line was turned over to the contractor each time and was eventually restored to a usable condition.

During the F-7 day simultaneous launch demonstration (SLD), two major communication problems occurred. The first was the loss of 9-out-of-11 ground operational-support-system voice circuits caused by severe microwave fade. An emergency backup routing of voice lines from MCC-H to GSFC has since been established through the ETR. The second major problem was the loss of all 12 receiving teletype circuits between MCC-H and GSFC (6 out totally, 6 marginal). All circuits were operational prior to simulated lift-off.

The alternate command line to Cape Kennedy experienced noise during F-1 day activity. The lines were turned over to the contractor numerous times and were restored to a satisfactory condition.

The GSFC B-Comm processor became inoperative during the terminal count on June 3, 1966, due to a bad card in the multiplexer. The A-Comm processor was operational at all times, and the B-Comm processor was repaired prior to launch.

6.2.2.3.3 Frequency interference: California, Hawaii, and Carnarvon reported some interference in the HF air-to-ground frequency.

6.2.2.4 Additional comments.- A power failure, caused by a procedural error, occurred at MCC-C at approximately T-9 on May 17, 1966. Procedures were established to prevent recurrence of this problem.

TABLE 6.2-I.- GEMINI IX-A NETWORK CONFIGURATION

Stations[a] \ Systems	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	Biomedical remoting	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	X
A/C		X																	X			
ANT	X	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		X	O	X	X
CAL	X	X	X													X					X	X
CNV[b]			X			X				X				X					X	X		
CRO	X		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	X
CTN	X	X																	X		X	X
CYI	X		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	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	X
KAW	X		X	X			X	X					X	X	X	X	X		X	X	X	X
KNO	X	X																	X		X	X
MIA			X																			
PAT			X																			
PRE			X																			
RKV	X			X			X	X					[X]	X	X		X		X	X	X	X
RTK	X	X	X																X		X	X
TAN	X	X																	X		X	X
TEX	X	X		X		X					X	X	X	X		X	X		X	(*)	X	X
WHS	X		X																		X	X
WLP[c]																X					X	X
WOM[d]	X		X																		X	X

[a]Location of stations is shown in figure 4.3-1.

[b]Wind profile measurements in support of recovery operations.

[c]If available.

[d]Non-interference basis.

Legend:

(X) Master Digital Command System.

O Remoting.

(*) Real-time and remoting.

[X] Post-pass biomed 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 IX-A recovery support. (The same forces were committed and on station May 17, 1966.)

The planned landing areas, in which support forces are pre-positioned 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 IX-A 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 provided landing areas periodically throughout the flight and prior to the nominal end-of-mission. Typical recovery support in these areas (fig. 6.3-3) consists of a destroyer equipped with a spacecraft retrieval crane and, at nearby air bases, search/rescue aircraft on alert (fig. 6.3-4).

The fourth type of planned landing area is the primary landing area where the spacecraft would land following a nominal mission. For Gemini IX-A, this area was located in the West Atlantic, Zone 1; because of its higher probability of use, the recovery support deployed consisted of the aircraft carrier U.S.S. Wasp (CVS 18), helicopters, tracking aircraft, and search/rescue aircraft. Support provided for this area is shown in figure 6.3-5.

The contingency recovery forces consisted of aircraft deployed to staging bases around the world (fig. 6.3-4) 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. 6.3-4) to take advantage of the nearby location of recovery forces.

6.3.2 Location and Retrieval

Gemini IX-A was programmed for a West Atlantic landing at the beginning of revolution 46. The coordinates of 46-1 (Revolution-46, Zone-1) were latitude 27°52' N., longitude 75°00' W. Retrofire was normal, and the primary recovery ship (U.S.S. Wasp) had visual contact with the spacecraft descending on its parachute at a range of approximately 3.5 nautical miles.

The following is a sequence of recovery events on June 6, 1966:

June 6, 1966 G.m.t., hr:min	Ground elapsed time, hr:min	Event
13:19	71:39	Kindley Rescue 2 on station, 195 nautical miles downrange from aiming point.
13:20	71:40	Kindley Rescue 1 on station, 195 nautical miles uprange from aiming point.
13:52	72:12	U.S.S. Wasp on station for 46-1 aiming point (27°52' N., 75°00' W.)

June 6, 1966 G.m.t., hr:min	Ground elapsed time, hr:min	Event
13:52	72:12	U.S.S. Wasp acquired radar contact with spacecraft.
13:54	72:14	U.S.S. Wasp in communications with spacecraft on UHF.
13:58	72:18	Personnel aboard U.S.S. Wasp sight spacecraft on main parachute at a range of 3.5 nautical miles. Rendezvous and recovery section also sighted on drogue parachute.
14:00	72:21	Spacecraft landing (fig. 6.3-6).
14:01	72:21	Swimmers and flotation collar deployed from helicopter.
14:04	72:24	Flotation collar attached and inflated.
14:53	73:13	Spacecraft and flight crew onboard U.S.S. Wasp (crew remained in spacecraft during retrieval). Pickup point was reported to be 27°52.0' N., 75°3.8' W. (fig. 6.3-7).

Communications between recovery elements and the spacecraft were good. Assignments were carried out efficiently by all recovery personnel. Of particular concern to the swimmers preparing to jump from the helicopter to the spacecraft was the report that the spacecraft was leaking, and the fact that the crew had requested deployment of swimmers to the spacecraft as soon as possible. After jumping to the spacecraft, the swim-team leader connected the interphone to the spacecraft and established contact with the crew. They reported that all was well, and the swim-team leader then assisted the two other swimmers in attaching the flotation collar to the spacecraft. After inflation of the collar, the swimmers attempted to reestablish communication with the crew through the interphone but were unsuccessful (see section 6.3.3.7). Unable to detect any motion or activity in the spacecraft and thinking the crew was in difficulty, the swim-team leader immediately released the hatch tool and started to open the hatch. While doing so, the hatch suddenly flew open and knocked the swimmer into the raft alongside the spacecraft. The swimmer received a slight head injury but was able to remain at the scene and await pickup by the ship. It was reported later by the command pilot that he noticed the swimmer preparing to open the hatch and, realizing the spacecraft cabin might still be pressurized, he attempted to release the hatch slowly from inside. The forces, however, were too great for him to restrain the hatch.

The rendezvous and recovery section sank before swimmers were able to secure a line to it. The main parachute was recovered and returned to the Cape.

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 type	Mode
Search 1 SH-3A	14:01 14:01	8000 8000	Approx. 15	SPP SPP	CW Pulse
Air Boss 1 S-2E	13:56	5000	14	ARA-25	ADF
Search 2 SH-3A	14:00	8000	17	SPP	Pulse
Search 3 SH-3A	13:59 13:59	5500 5500	5 5	SPP SPP	CW Pulse

The aircraft and helicopters from the aircraft carrier were initially in position for the landing in an array as shown in figure 6.3-4. Under these circumstances and considering that the spacecraft landed so near the carrier, the ranges in the above table may not be too significant. Normal recovery beacon operation, however, was verified.

6.3.3.2 HF transmitter (15.016 mc).- The HF antenna was not erected and there was no HF activity.

6.3.3.3 UHF voice transmitter (296.8).- The U.S.S. Wasp reported excellent UHF communications at 13:54 G.m.t., six minutes prior to landing. Other aircraft and helicopters from the carrier also reported UHF reception from the spacecraft during descent and after landing.

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 crew.

6.3.3.6 Fluorescein sea marker.- The sea dye marker diffusion was normal and was sighted at a range of 2 nautical miles by the recovery ship. Carrier aircraft reported sighting the sea marker at ranges from 1 to 3 nautical miles. The spacecraft was still releasing dye at spacecraft pickup time, approximately 50 minutes after landing.

6.3.3.7 Swimmer interphone.- Two interphones were carried to the spacecraft by the swimmers. Initial communications were good; however, after installation of the collar, the swimmers were unable to contact the crew on the interphone. Not knowing whether the phone was inoperative or whether the crewmen were in difficulty, the swimmers chose to open the hatch rather than try the second interphone.

Prior to the swimmers boarding the helicopters, the phones were checked and found operative. Upon returning to the carrier, the swimmers, with NASA recovery personnel, checked the phones again and did find intermittent operation with the phone used at the recovery scene. A failure analysis will be performed on this phone.

6.3.4 Postretrieval Procedures

The spacecraft was powered down by the crew prior to retrieval and the crew remained in the spacecraft during the retrieval and pickup operations. The spacecraft was retrieved with the carrier "Boat and Aircraft" crane and placed in the spacecraft dolly.

Operations concerning condition of the spacecraft were as follows:

(a) The HF antenna was not extended.

(b) The recovery and UHF antennas were erected. The recovery antenna was bent (swimmers made the same observation prior to their deployment).

(c) The flashing light and recovery loop were erected. The light had not been activated.

(d) The right-hand spacecraft window was 90 percent fogged, and the left-hand window was 40 percent fogged.

(e) The heating effects on the spacecraft appeared very similar to other Gemini spacecraft.

(f) The main parachute jettisoned normally and was recovered.

(g) The gear in the spacecraft was stowed neatly, and it was reported that there was some water on the spacecraft floor. The right-hand goose-neck mirror was cracked.

(h) There was one bent shingle on the forward end of the left-hand landing-gear well door. (Swimmers noticed the bent shingle prior to installation of the flotation collar.)

(i) Swimmers noticed that the RCS thrusters were "burping" for approximately 2 minutes after landing.

Approximately one hour after recovery, the flight was sent from the U.S.S. Wasp to Patrick Air Force Base with film of the recovery. All "urgent-return" flight items, including spacecraft onboard PCM and biomedical data, film, pilots suit, et cetera, were returned to Cape Kennedy and Houston on an aircraft leaving the U.S.S. Wasp three hours after recovery.

The flight crew departed the carrier for Cape Kennedy approximately 2:00 p.m. e.s.t., June 6, 1966.

The spacecraft was off-loaded from the carrier at Boston, Massachusetts at approximately 2:00 p.m. e.d.t., June 8, 1966. Deactivation procedures were conducted at Boston.

6.3.5 Spacecraft 9 RCS Deactivation

The Spacecraft 9 RCS deactivation was accomplished on June 8 and 9, 1966, at the Boston Naval Shipyard, Boston, Massachusetts. Work was started at 2:30 p.m. e.d.t., on June 8 and completed at 8:30 a.m. e.d.t., June 9, 1966. The deactivation was normal in all respects, and no emergency situations were encountered.

Fuel and oxidizer remaining in the spacecraft RCS were as follows:

A-ring fuel	0.78 pound
A-ring oxidizer	1.66 pounds
B-ring fuel	8.88 pounds
B-ring oxidizer	9.03 pounds

The preflight loads were 15.80 pounds of fuel in each ring and 20.20 pounds of oxidizer in each ring.

Following deactivation at the Boston Naval Shipyard, the spacecraft was transported to South Weymouth Naval Air Station for transport to St. Louis, Missouri. Spacecraft 9 was delivered to St. Louis, Missouri, at 6:30 p.m. c.d.t. on June 9, 1966.

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TABLE 6.3-I.- RECOVERY SUPPORT

Landing area	Access time, hr:min		Support
	Aircraft	Ship	
Launch site area:			
Pad	0:05		4 LARC (amphibious vehicle) 1 LCU (large landing craft) with spacecraft retrieval capabilities
Land	0:10		2 LVTR (amphibious vehicle) with spacecraft retrieval capabilities
Water (if flight crew ejects)	0:02		3 M-113 (tracked land vehicles)
Water (if flight crew is in spacecraft)	0:15		4 CH-3 (helicopters) (3 with rescue teams) 1 boat (50-foot) with water salvage team)
Launch abort area:			
A_1	4:00	11:00	1 CVS (aircraft carrier) with on-board helicopter capabilities, 3 DD (destroyers), 1 AO (oiler)[a], and 6 aircraft on station (4 HC-97 and 2 HC-130).
A_2	4:00	38:00	
B	4:00	5:00	
C	4:00	12:00	
C	4:00	36:00	
Primary landing area:			
West Atlantic (Zone 1)	1:00	4:00	1 CVS (aircraft carrier) from area A, station 3. 2 S-2E - Air Boss 1 and 2 3 EA-1F (2 relay + 1 backup) 1 EA-1E (radar search) 6 SH-3A (3 search, 1 photo, and 2 swimmer)

[a]An oiler (AO) was deployed to this area primarily for logistic purposes; however, it also provided recovery support in the East Atlantic Zone.

TABLE 6.3-I.- RECOVERY SUPPORT - Concluded

Landing area	Access time, hr:min		Support
	Aircraft	Ship	
Secondary landing areas:			
East Atlantic (Zone 2)	0:30 strip alert	6:00	1 DD (destroyer)[a]
West Pacific (Zone 3)		6:00	2 DD (destroyers) (rotating on station)
Mid-Pacific (Zone 4)		6:00	1 DD (destroyer)
End-of-mission landing area:			
46-1	1:00	4:00	Same as for "West Atlantic" with the following additions:
			2 HC-97 (rescue aircraft)
Contingency			29 aircraft on strip alert at staging bases throughout the world and aboard carrier
Total			8 ships, 6 helicopters, 35 aircraft

[a]An oiler (AO) was deployed to this area primarily for logistic purposes; however, it also provided recovery support in the East Atlantic Zone.

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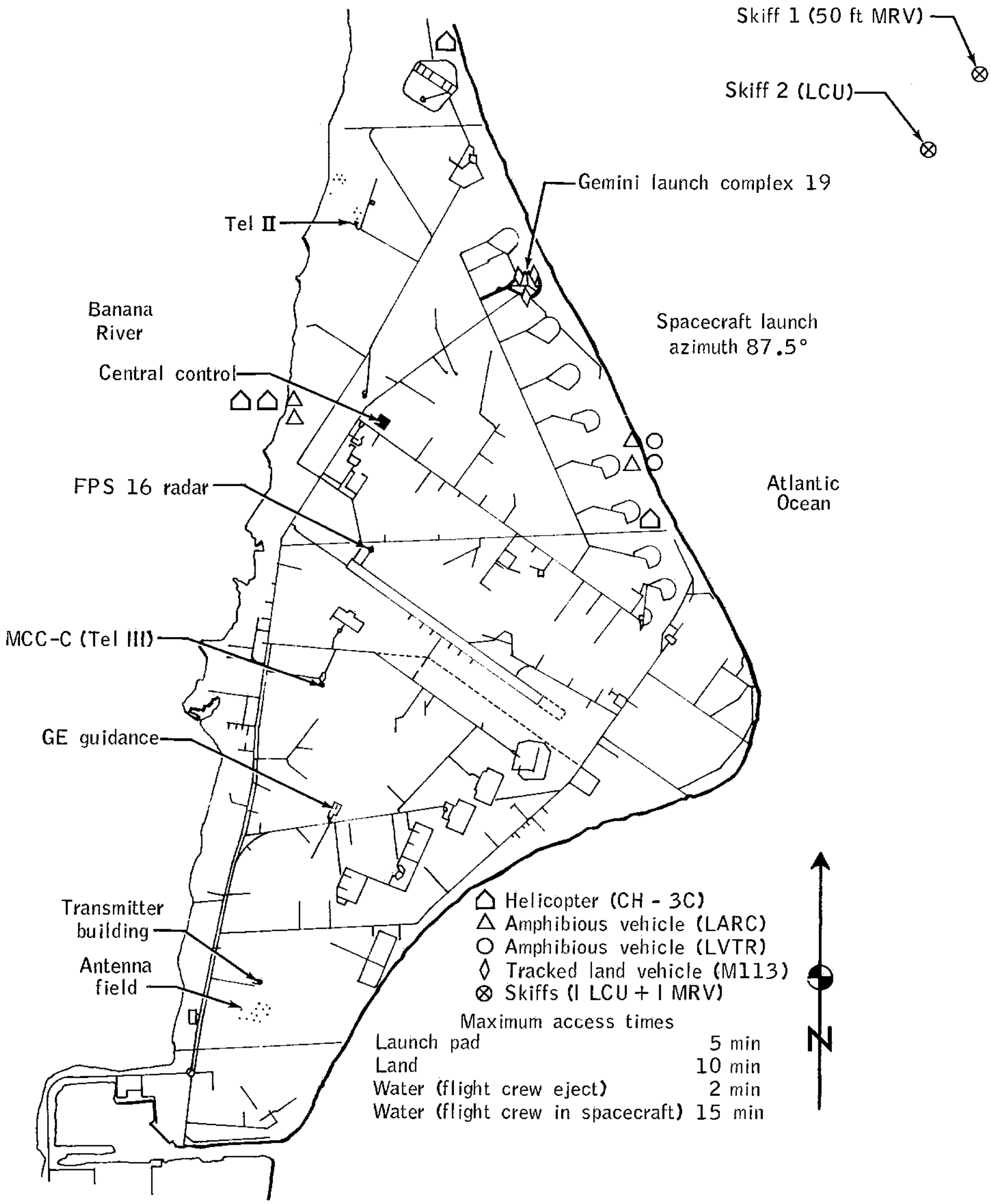

Figure 6.3-1. - Launch site landing area recovery force deployment.

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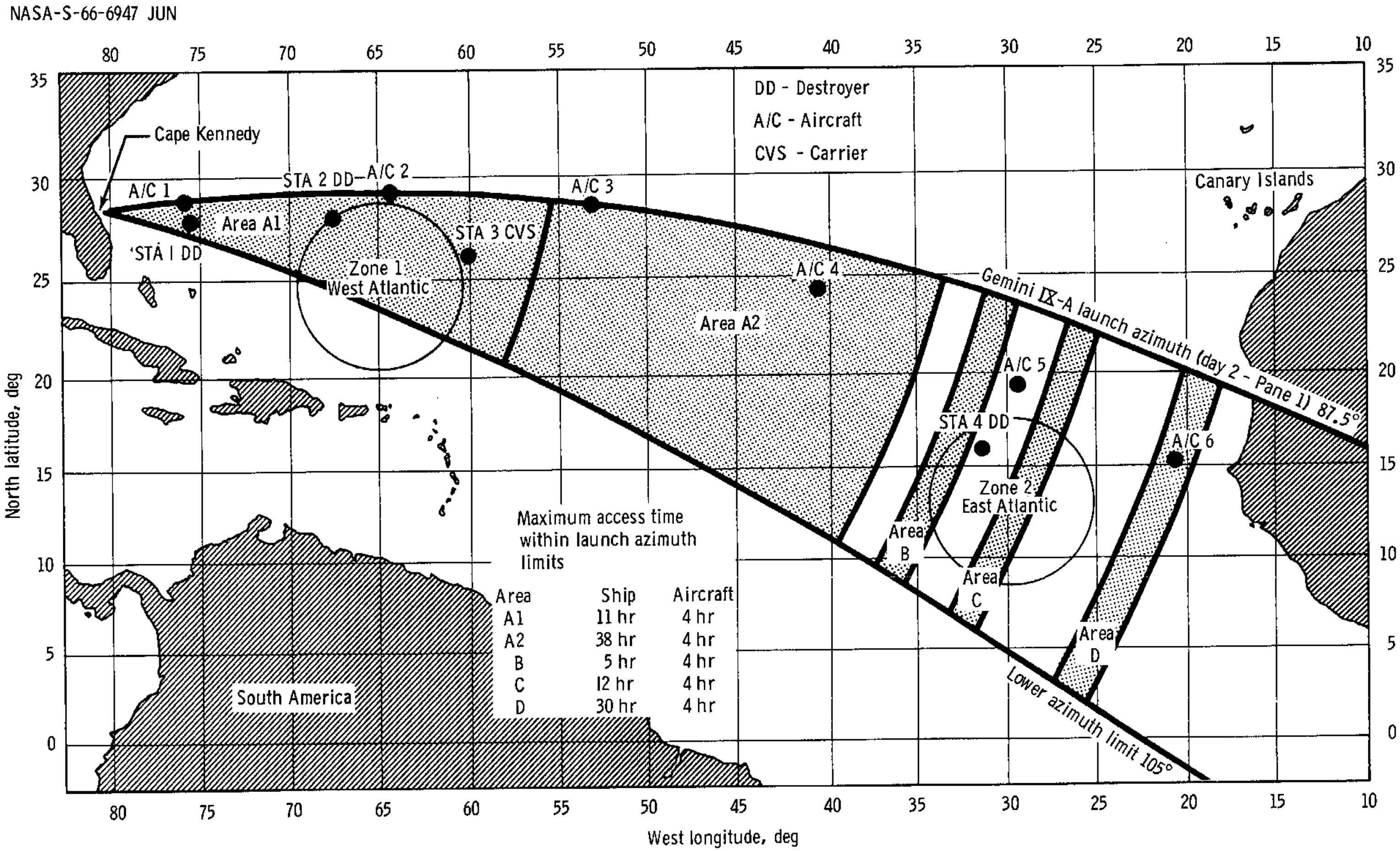

Figure 6.3-2. - Gemini IX-A launch abort areas and recovery ship and aircraft deployment.

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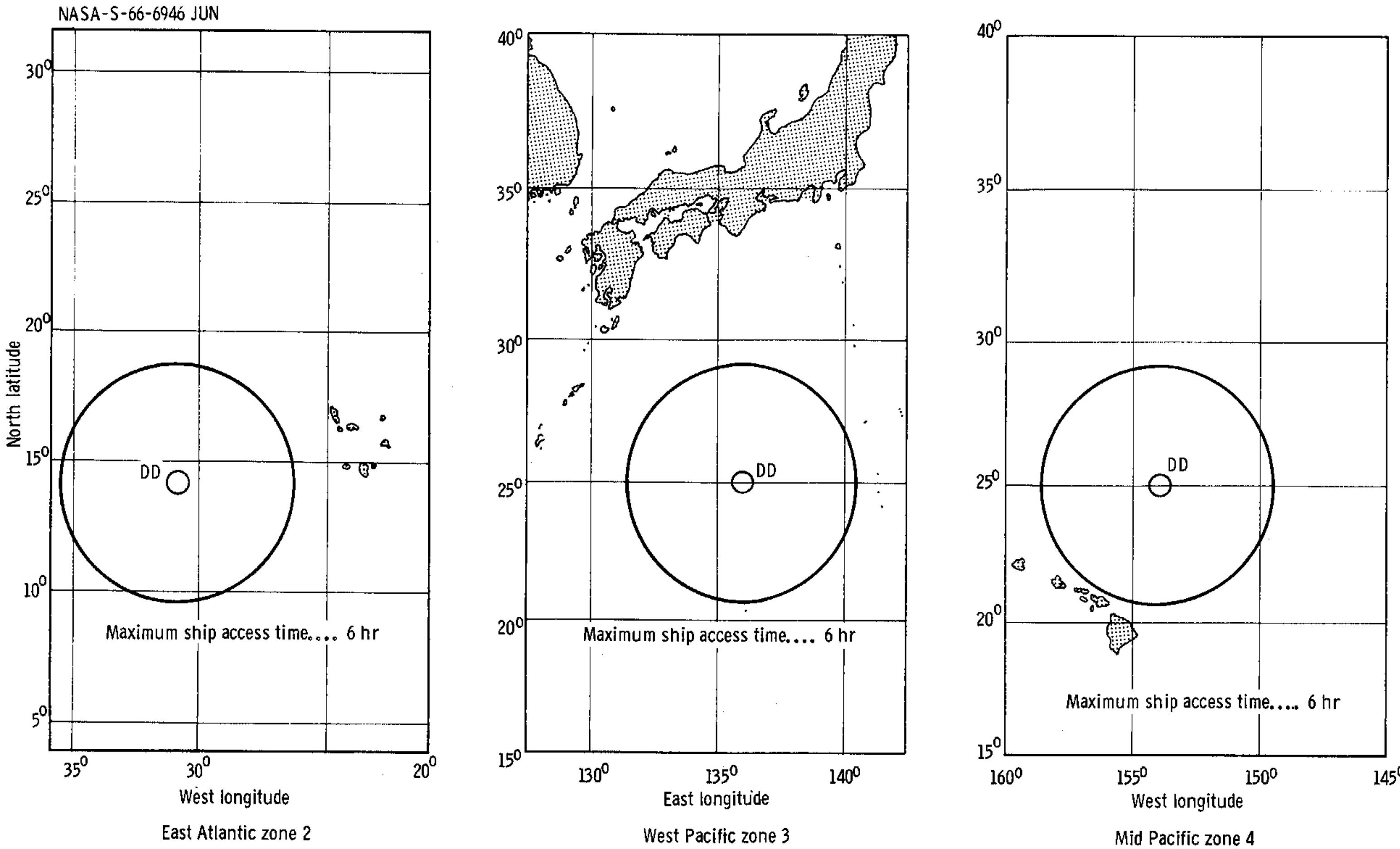

DD - Destroyer

Figure 6.3-3. - Gemini IX-A landing zone location and force deployment.

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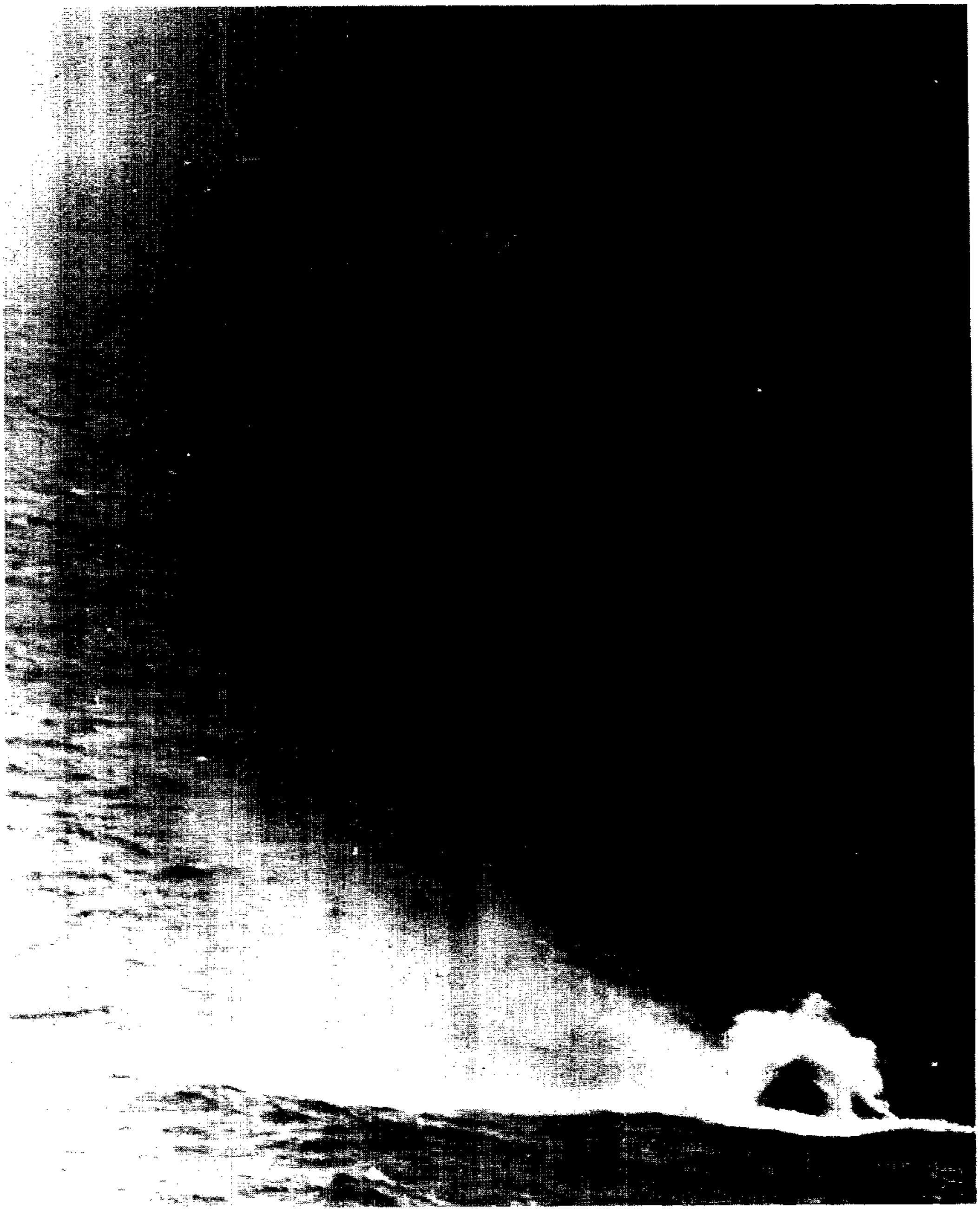

Figure 6.3-6. - Spacecraft landing.

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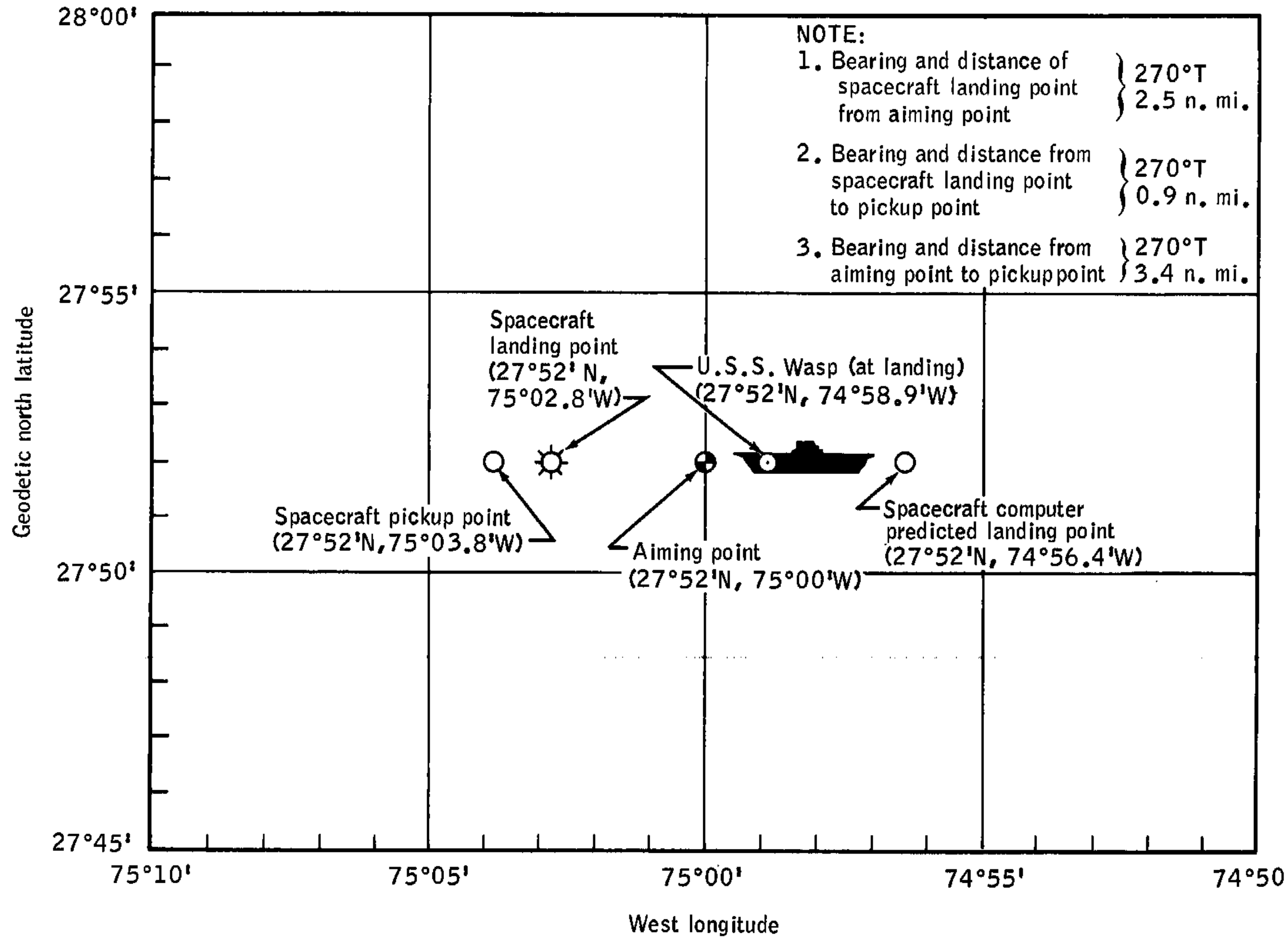

Figure 6.3-7. - Spacecraft landing information, as determined on the prime recovery ship.

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7.0 FLIGHT CREW

7.1 FLIGHT CREW PERFORMANCE

7.1.1 Crew Activities

The flight crew followed the flight plan through the first rendezvous and station keeping. Because the Augmented Target Docking Adapter (ATDA) shroud had failed to jettison, docking could not be accomplished. At the time the crew reported that the shroud was still attached, the alternate flight plan for this contingency was put into effect. After a period of station keeping and ATDA shroud inspection, a radial separation maneuver was performed in preparation for the equi-period (second) rendezvous. The second rendezvous was accomplished successfully; however, because docking was still not possible, the crew performed a second separation maneuver prior to the first sleep period, with a third rendezvous scheduled to occur after the sleep period. The third rendezvous was also accomplished successfully; however, due to the Orbital Attitude and Maneuver System (OAMS) fuel status and because the crew were tired, the plan for conducting extravehicular activities (EVA) while station keeping with the ATDA was abandoned. The EVA was then rescheduled for the third day of flight, and the crew performed a final separation from the ATDA.

A short rest period was scheduled prior to the start of activities for the second day. During the second day, activities were performed on schedule and many experiment objectives were accomplished. After the second sleep period, the crew started EVA preparation and accomplished all planned evaluations during the first dayside period of EVA. During the first nightside period, while checking out and donning the Astronaut Maneuvering Unit (AMU), the pilot experienced visor fogging which resulted in termination of the AMU experiment (D-12). The remainder of the mission proceeded as scheduled. Figure 7.1.1-1 presents the timeline of crew activities as performed during the Gemini IX-A mission.

7.1.1.1 Prelaunch through insertion.- The crew entered the spacecraft at the proper time in the countdown and completed all prelaunch checks and operations on time. Lift-off (LO) was evident to the crew as a decrease in noise and vibration and as a feeling of pulsing motion. The entire powered-flight phase was nominal; however, in the period between LO + 60 seconds and LO + 170 seconds, the command pilot was blinded by the sun's rays and was not able to monitor certain critical launch displays. The flight controllers were immediately notified of this problem. The command pilot was able to see the launch vehicle fuel and oxidizer pressure gages for Stage I by removing his right hand from the attitude controller and shading his eyes, but he was still unable to see other critical displays. The flame front associated with staging was visible out of the windows despite the sun, but was apparently not so pronounced as on some of the previous Gemini flights. No low-level longitudinal oscillations (POGO) were apparent to the crew.

Special window covers on this spacecraft did not completely prevent the windows from smudging during launch. Insertion was within the 3-sigma dispersions, and the Insertion Velocity Adjust Routine (IVAR) readings appeared at second stage engine cutoff (SECO) plus three seconds showing expected vectors. There were no noticeable angular rates from SECO until SECO plus 27 seconds when the crew separated the spacecraft from the second stage of the launch vehicle. The window covers and the fairings on the spacecraft nose and horizon scanners were then jettisoned, and the insertion checklist was completed according to the flight plan.

7.1.1.2 M=3 (first) rendezvous.- The M=3 (first) rendezvous activities consisted of the planned pretransfer maneuvers and terminal rendezvous maneuvers.

7.1.1.2.1 Pretransfer maneuvers: The crew reported that the pretransfer maneuvers were made on time and that no problem was encountered in nulling the residuals. Fuel remaining after the corrective combination maneuver (N_{CC}) was 81 percent, according to the onboard gage.

Intermittent radar lock-on occurred when the radar was activated at a range of approximately 130 nautical miles. Lock-on was fairly steady at 120 nautical miles and, at ranges of 25 nautical miles or less, lock-on was broken only occasionally. (See sections 5.1.5 and 7.1.2 for a more detailed description of the radar performance.)

The first indication that a computer anomaly had occurred was noted before the coelliptic maneuver (N_{SR}). (See sections 5.1.5 and 7.1.2 for a detailed description of this computer problem.) The crew was able to perform the N_{SR} maneuver as planned, despite the malfunction.

The ATDA was first seen at a range of 50 nautical miles approximately 24 minutes before sunset, at a pitch-up angle of 14 degrees. The elevation of the sun above the line-of-sight was approximately 86 degrees. The crew reported that the brightness of the ATDA at this range was approximately equivalent to a fifth-magnitude star, and that the reticle pattern obscured the target even though the dimming control was set at the lowest level. However, at 30 nautical miles, the target was no longer obscured by the reticle pattern.

7.1.1.2.2 Terminal phase maneuvers: The crew conducted a terminal phase initiate maneuver (TPI) that agreed with the ground and backup solutions; however, the onboard-computer solution differed from this maneuver by 7 ft/sec in the up direction. The midcourse solutions obtained from the backup calculations and the onboard computer were similar, and therefore, the onboard computer solution was applied. The discrepancy in the onboard-computer solution for TPI is discussed in section 5.1.5 of this report.

The ATDA, which had been visible in reflected sunlight from 50 nautical miles, disappeared at sunset (about 5 minutes after TPI); however, the ATDA acquisition light appeared almost immediately thereafter. The acquisition light was first seen at a range of about 20 nautical miles. As the ATDA tumbled with the shroud attached, the acquisition lights would appear and disappear. At 4 to 5 nautical miles range, the crew were able to see the red running lights, and the green and amber running lights became visible as the range decreased.

The crew reduced range rate to 19 ft/sec at a range of one nautical mile; at 3000 feet, the range rate was further reduced to 15 ft/sec; and at 1000 feet, the range rate was reduced to 10 ft/sec. Station keeping began at 4 hours 15 minutes ground elapsed time (g.e.t.) at a distance of 60 feet with 58 percent of the OAMS propellant remaining.

7.1.1.3 Separation maneuver and sextant practice.- The second (equi-period) rendezvous was initiated at about 5 hours 1 minute g.e.t. with a radial separation maneuver of 20 ft/sec. The crew reported that this maneuver was made as planned and the residuals were close to zero.

The crew was scheduled to use the sextant and make measurements of the angle between the target and the horizon during the period between 5 and 13.5 minutes after separation. Immediately after separation, the target was difficult to track against the sunlit earth. As the terminator was approached, tracking with the reticle was easier, but it was still very difficult, if not impossible, to acquire the ATDA in the 8-degree field-of-view of the sextant against the earth background. Therefore, the sextant measurements were not accomplished as planned.

7.1.1.4 Equi-period rendezvous.- A sextant reading of the target-to-horizon angle was successfully made in darkness at about 30 minutes after separation. The results of this measurement could have been used to calculate an adjustment maneuver; however, the crew chose to solve for this maneuver by using onboard charts and entering the time at which the ATDA crossed the local horizontal. The solution from the charts was 0.1 ft/sec forward and 0.2 ft/sec up. Because of the small magnitude indicated for this maneuver, it was not applied.

The target was first seen after sunrise when the crew rolled the spacecraft to a heads-down attitude. Although the sun was only approximately 30 degrees from the target line-of-sight, the nose of the spacecraft, in the heads-down attitude, blocked the sun from shining directly into the spacecraft windows. At a range of about nine nautical miles, the target was estimated to have a brightness approximately equivalent to a minus 2 magnitude star, or brighter.

For the equi-period rendezvous, the calculation of TPI was to start with a measurement of the time when the ATDA reached 43 degrees above the horizon, as measured with the sextant, and noting the pitch angle to the target. It was not noticed that the sextant had been set to 47 degrees until about the time the 43-degree position had passed. Because two measurements of angle and time are required to get a rate of angle change for calculating TPI, and the second angle was measured late, the terminal phase initiation was also late. The late TPI resulted in high pitch angles for the first midcourse correction and sextant measurements were not used to obtain a solution. This correction was calculated from platform-angle measurements; however, sextant measurements were successfully made for the second midcourse correction. The pilot had a great deal of difficulty in measuring the last angle with a sextant setting of 80 degrees. Because the measurement of an angle this large had to be made in the lower left-hand corner of the spacecraft window, only a small arc of the horizon was visible and consequently the sextant may have been rolled with respect to the horizon. It was determined from the flight log that, although the recorded time increment measured between angle measurements was 3 minutes and 5 seconds, a time increment of 3 minutes was used in the chart to calculate the maneuver. The pilot calculated a thrust-down time of 2 seconds for this correction using 3 minutes for the time increment. If he had used 3 minutes and 5 seconds, the calculated thrust time would have been about 1.5 seconds. The crew reported that after applying this correction, line-of-sight control in the opposite direction was required.

Braking was accomplished by using the rate of change of target size in the reticle. Starting at a range of about one nautical mile, the shape of the ATDA was discernible. Braking based on range measurements that used the sextant optics and reticle was not attempted because the tumbling ATDA did not present a good target in the small field of view afforded at the braking ranges.

7.1.1.5 Rendezvous from above.- The rendezvous from above (third rendezvous) activities were similar to those of the first rendezvous and consisted of pretransfer and terminal phase rendezvous maneuvers.

7.1.1.5.1 Pretransfer maneuvers: The crew reported that all pretransfer maneuvers were made on time and that no difficulty was encountered in reducing the residuals to the required values. After the N_{SR} maneuver, the OAMS propellant remaining was reported as 32 percent according to the onboard propellant gage.

Shortly after the moon appeared, the target was visible in reflected moonlight. The target brightness increased until sunrise, which occurred when the spacecraft and the ATDA were approximately 20 nautical miles apart. With the target in the sunlight, the crew estimated the brilliance to be four to five times that of Venus; however, as the terminator was crossed, the apparent brightness decreased rapidly and the target seemed to disappear against the sunlit earth background.

Computer solutions, which were being calculated after the platform alignment between N_{SR} and TPI, were somewhat erratic, with every fourth or fifth solution being higher than expected. During the 15-minute platform alignment initiated at a pitch-down angle of 9 degrees, the sine of the radar elevation angle, as read from the computer, oscillated erratically.

7.1.1.5.2 Terminal phase rendezvous maneuvers: The target disappeared prior to TPI and was not visible until the range was less than three nautical miles, and midcourse corrections had to be made while tracking the target with instruments only. The onboard-computer solution indicated 4 ft/sec down as compared with 3 ft/sec up from the backup solution for TPI. The relative-trajectory polar plot and the history of range variation indicated that an up correction was needed, and the ground solution also indicated a small up correction of 0.3 ft/sec. Therefore, the backup solution was applied. TPI was initiated when the COMP light came on at 21:02:28 g.e.t.

A 5-minute platform alignment was initiated shortly after TPI; therefore, a second backup solution was not obtained. Radar lock-on was not broken during the platform alignment. Because the range in the computer did not change during the alignment, the range and range rate could not be monitored.

The last onboard-computer midcourse calculation indicated 10 ft/sec down and 7 ft/sec right and the backup chart indicated 5 ft/sec down. The crew applied the 10 ft/sec down, but later noted that this correction appeared to have been too large. The first onboard-computer midcourse calculation was a correction of 4 ft/sec left, which was in the opposite direction of the last correction.

When the target was visible at a range of approximately three nautical miles, it was a small black dot which would disappear against the terrain features. When the white markings finally became visible, the target could be tracked against the blue background of the water below.

The crew reduced the range rate to about 14 ft/sec at approximately one nautical mile and to approximately 7 ft/sec at 2000 feet. The crew estimated that overbraking during the final approach cost about 10 to 15 pounds of propellant. Station keeping was initiated with 18 percent of the propellant remaining; therefore, approximately 14 percent of the total OAMS propellant was used from shortly after the last N_{SR} maneuver to the completion of the rendezvous.

7.1.1.6 Experiments.-

7.1.1.6.1 Experiment M-5, Bio-Assays of Body Fluids: The experiment equipment was used by both crewmen for all urinations, and also for overboard dumping of drinking water to ensure ample storage space for the fuel-cell product water. One sample bag was returned unlabeled, but was later identified. The pilot encountered difficulty in keeping the sample bags on the collector device and had to hold the bag in place; however, urine release to the cabin interior was minimal. Accidental extra actuations which released additional tritiated water were properly logged.

7.1.1.6.2 Experiment S-1, Zodiacal Light Photography: This experiment was performed during the night period starting at 54 hours 57 minutes g.e.t. but was conducted from inside the spacecraft, rather than during EVA as planned. Ground-supplied information and crew-generated procedures were used, and resulted in 17 1/2 excellent airglow and Milky Way photographs out of a possible 18. These results were better than anticipated and much better than the crew believe that they could have done during EVA. The crew stated that long exposure times with handheld cameras were not feasible during EVA because of the difficulty in maintaining a steady position without the use of hands or firm body restraints.

7.1.1.6.3 Experiment S-10, Agena Micrometeorite Collection: The experiment package and mounting accessories on the ATDA were both viewed at close range and photographed during station keeping. The package assembly was in excellent condition, and there appeared to be no launch or orbit contamination that would hinder operation during a future mission. The package was not recovered because EVA was not performed while in the vicinity of the ATDA.

7.1.1.6.4 Experiment S-11, Airglow Horizon Photography: This experiment was performed on three nightside passes as follows:

Sequence 01	29:16:27 g.e.t.
Sequence 03	29:49:02 g.e.t.
Sequence 01	30:46:45 g.e.t.
Sequence 03	31:19:20 g.e.t.
Sequence 01	32:17:04 g.e.t.
Sequence 03	32:49:39 g.e.t.

The sequence 02 run, scheduled for 56:49:17 g.e.t., was not performed because the camera had been restowed in the bottom of one of the aft food boxes and would have been very difficult to unstow. However, during the earlier runs, the crew took two extra photographs, one of 5 seconds duration and one of 10 seconds duration, with the filter installed. A total of 45 very good pictures were obtained.

The pilot had difficulty in keeping his body in position to aim the camera because of the particular camera-mounting configuration. The flight crew stated that the task would have been an order-of-magnitude easier had the camera been boresighted along the spacecraft X-axis.

7.1.1.6.4 Experiment S-12, Micrometeorite Collection: The experiment doors were opened and closed during the flight as follows:

Opened	9 hours 00 minutes g.e.t.
Closed	17 hours 00 minutes g.e.t.
Opened	34 hours 30 minutes g.e.t.
Closed	44 hours 30 minutes g.e.t.

The crew could not hear the opening or closing operation from inside the spacecraft. The door was closed and locked during the EVA preparation and the operation of the pyrotechnic lock was heard by the crew. The pilot encountered no difficulty recovering the package during EVA and no tether was used during this recovery.

7.1.1.6.5 Experiment D-12, Astronaut Maneuvering Unit: The pilot completed 80 percent of the AMU donning procedure before the decision was made to terminate the AMU experiment. Communications were clear from the command pilot to the pilot from the spacecraft transmitter but were garbled from the pilot to the command pilot over the AMU RF link. The problem of maintaining position when preparing the AMU for donning resulted in a much higher-than-anticipated total workload which finally resulted in fogging of the pressure visor, and EVA was terminated prior to evaluating the AMU.

7.1.1.6.6 Experiment D-14, UHF-VHF Polarization: The following experiment passes were completed by the crew:

Hawaii	26:52:34 g.e.t.
Antigua	27:16:33 g.e.t.
Hawaii	28:28:25 g.e.t.
Hawaii	30:04:25 g.e.t.
Hawaii	31:39:57 g.e.t.
Hawaii	33:15:34 g.e.t.

All the runs were performed using the spacecraft platform and attitude displays, and no operational difficulties were encountered; however, the antenna was broken off by the pilot during EVA. The signal strength received at the ground stations was about 20 db lower than expected. Also, part of the data from the first pass was lost because of a problem encountered with positioning the ground antenna. The remaining data were usable and are being analyzed.

7.1.1.7 <u>Extravehicular crew performance</u>.- Preparations for EVA started at 45 hours 30 minutes g.e.t. and took about 3 hours of the 3 hours and 45 minutes allocated. A spacecraft control problem was encountered during this period because of the inadvertent opening of the scanner-heater circuit breaker. This problem took about 25 minutes to analyze and correct. The hatch was opened about 10 minutes prior to sunrise, and the pilot's activities in the vicinity of the hatch went well, despite the tendency to float out of the spacecraft. At this time, the pilot reported the Extravehicular Life Support System (ELSS) chestpack was riding much higher on his body than was expected. It was so high the guard on the ELSS TEST-DIM-BRIGHT switch interfered with the neck-ring latch when the pilot turned his head. The pilot also encountered difficulty in pushing off toward the nose of the spacecraft to attach the docking-bar mirror.

The pilot determined that the only value of the tether as a maneuvering aid was that it enabled him to return to the location of the tether attachment to the spacecraft. It was also determined that Velcro pads could be helpful in maintaining a station but were of doubtful value in maintaining attitude. The Velcro-pad evaluation was shortened because one pad was lost when it slipped off the pilot's hand during his first attempt to use the pads. During the entire EVA, the pilot reported a tendency for his body to move away from the spacecraft or work area, and to rotate around his point of attachment to the spacecraft.

The AMU thermal-cover pyrotechnic device fired satisfactorily, but the cover, one adapter handbar, and the umbilical guide did not deploy completely. The pilot, being thoroughly familiar with the deployment mechanism, proceeded to manually deploy these items with very little effort.

The major problem with the EVA became apparent when the pilot was making the tether connections. There were three tether connections to make, and all were two-handed operations. In one-g and zero-g aircraft training, each connection required between 15 and 20 seconds to make. In flight, however, the foot-restraint system (stirrups) proved inadequate, and the pilot was not able to use both hands for working more than about five seconds at a time. Within five seconds, his feet would float out of the stirrups and he would have to use one hand to reposition himself. Despite this difficulty, the pilot was able to make two of the three tether connections, eliminating the 25-foot tether-hook connection. During the period the pilot was working on the tether connections, he also commented that, while his overall comfort level was acceptable, a portion of his lower back was very hot. He waited until sunset before doing any more work, assuming the direct solar radiation on his back was the source of the problem. (Postflight analysis has identified the problem as a tear in the suit superinsulation. This tear allowed localized heating.) The pilot had considerable difficulty in unstowing the AMU attitude-controller arm. He attributed this primarily to the inadequacy of the stirrups.

At about this time, the pilot's visor was noticeably fogged, but he continued through the AMU preparations as called to him by the command pilot. The pilot had considerable difficulty getting the AMU oxygen valve started open. The pilot turned around and backed into the AMU, and the changeover to the AMU electrical system was accomplished. Communications on the AMU were marginally acceptable, but they were expected to improve when the pilot moved away from the adapter. The pilot next prepared for making the AMU restraint-harness connection. At this time, the magnitude of the fogging problem became fully known to the pilot. He had trained to use the two mirrors in the adapter to verify all his connections, and to observe the mating of the restraint-harness buckle. He was not able to see any of the connections through the mirrors at this time. The pressure gage on his suit was also partially fogged, but it was readable. The crew decided to terminate the EVA portion of the mission because the pilot was not able to visually confirm all connections. The pilot's visor became partially defogged while he was standing in the seat prior to ingress, but became fogged over again during ingress. This fogging was probably caused by a heavy workload encountered because the hatch was difficult to close.

7.1.1.8 Retrofire and reentry.- Retrofire and reentry were nominal in all respects. The crew completed the preretrofire checklist, and the Reentry Control System (RCS) was armed over the United States where ground telemetry stations were available. The Digital Command System (DCS) load was checked and the platform was then aligned until approximately three minutes prior to retrofire. All switches and systems operated properly, and an automatic retrofire was accomplished.

Downrange and crossrange errors were nulled on the error indicators and the 400K feet indication occurred within one second of the ground predicted time. All indications and checks gave the crew a high degree of confidence in the accuracy of the Inertial Guidance System.

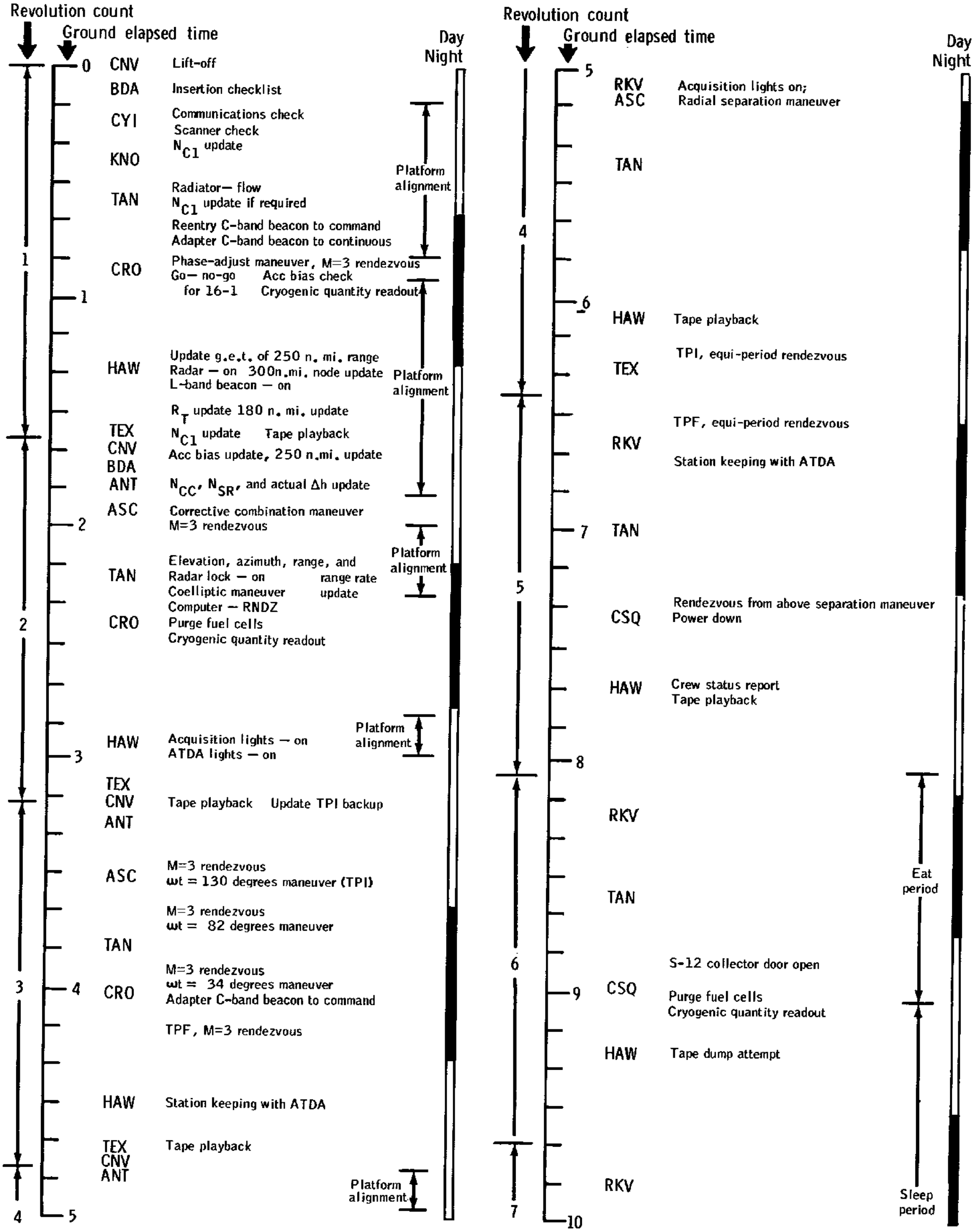
7.1.1.9 Landing and recovery.- The drogue parachute was deployed at 50 000 feet and oscillations were less than expected or previously experienced. The main parachute was deployed at 10 600 feet, followed by a normal disreefing. Two-point suspension was activated later than planned due to the RCS thruster burning as reported in paragraph 7.1.2.5. Landing shock was more severe than anticipated, and recovery operations progressed nominally (paragraph 7.1.2.5).

7.1.1.10 Mission training and training evaluation.- Flight crew training was accomplished generally as outlined in the Mission Training Plan. Table 7.1.1-I shows the training time for the Gemini IX-A crew.

The command pilot had trained previously for the Gemini III and Gemini VI/VI-A missions and was able to react in a timely manner after the loss of the prime crew about 2 1/2 months before the scheduled launch date. The crew's efficient utilization of their time, combined with the availability and performance of the simulators, mockups, and other training devices, resulted in all training requirements being met or exceeded well before the scheduled launch date.

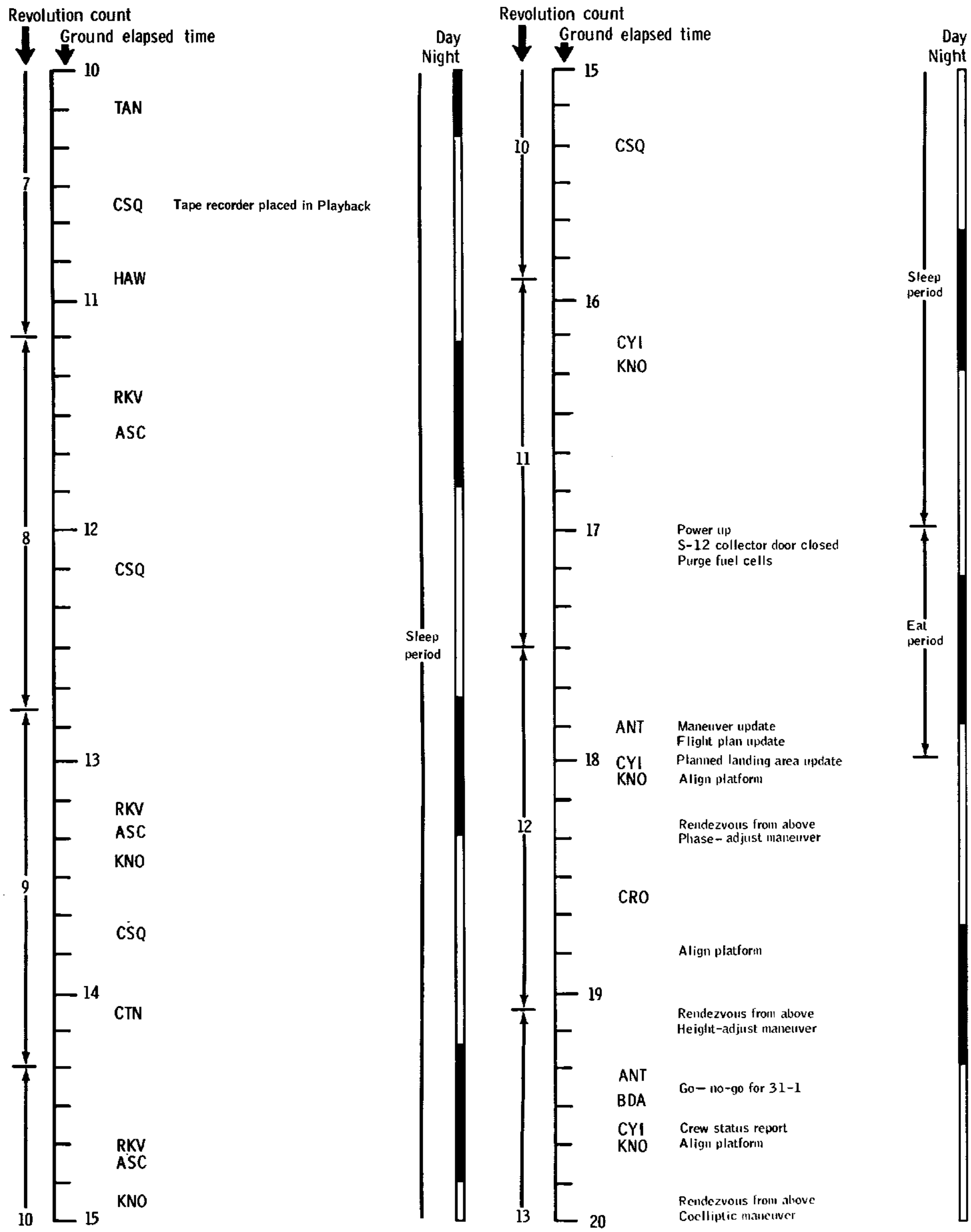
TABLE 7.1.1-I.- CREW TRAINING SUMMARY

Activity	Training time, hr	
	Command pilot	Pilot
Gemini system briefings	52	121
Operational briefings	36	51
Gemini Mission Simulator	161	147
Dynamic Crew Procedures Trainer	10	3
Translation and Docking Trainer	4	2
Rendezvous simulation	61	61
Extravehicular activity training	84	115
Egress training	0	10
Planetarium	0	16
Spacecraft Systems Tests (SST)	71	70

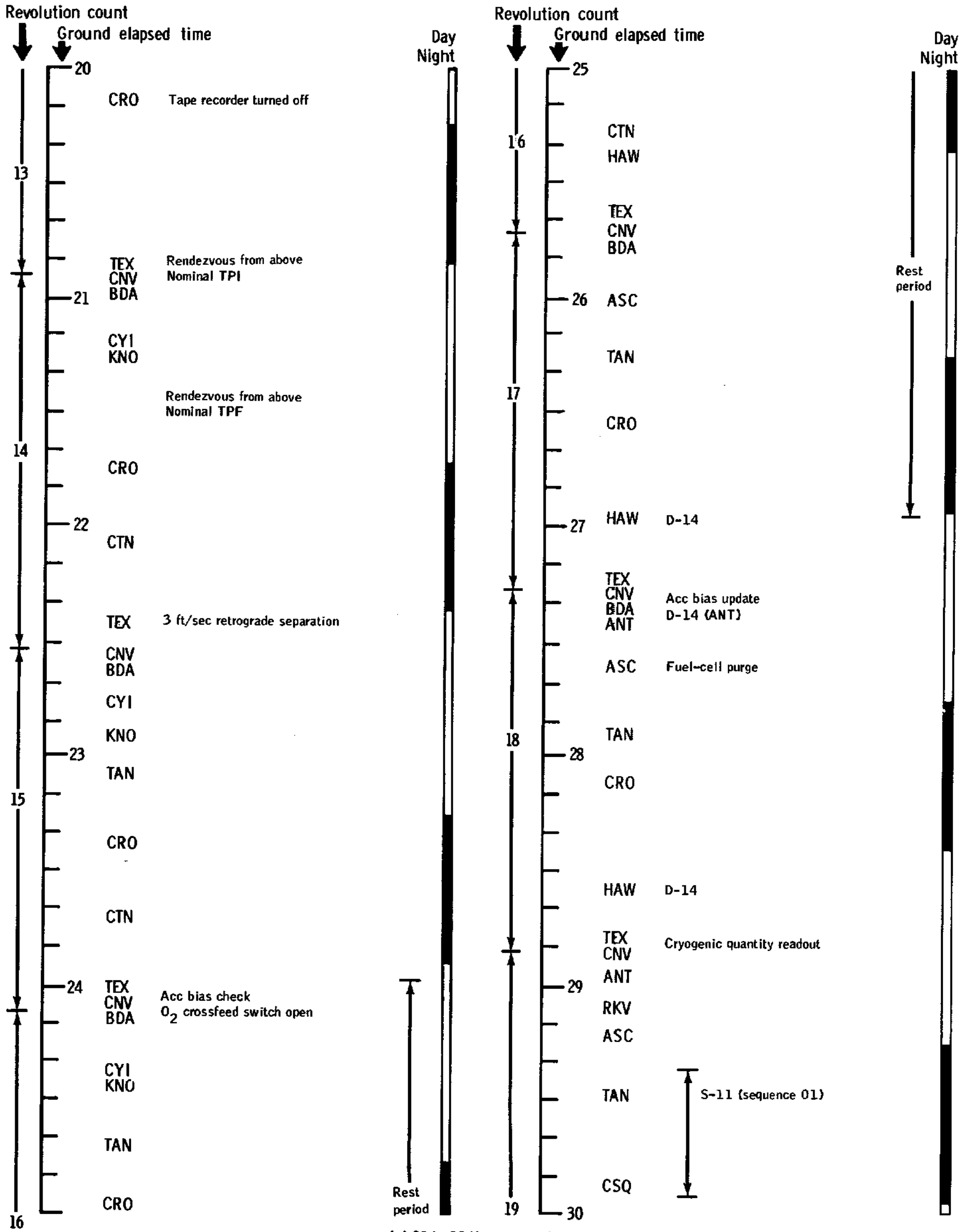


(a) 0 to 10 Hours g.e.t.

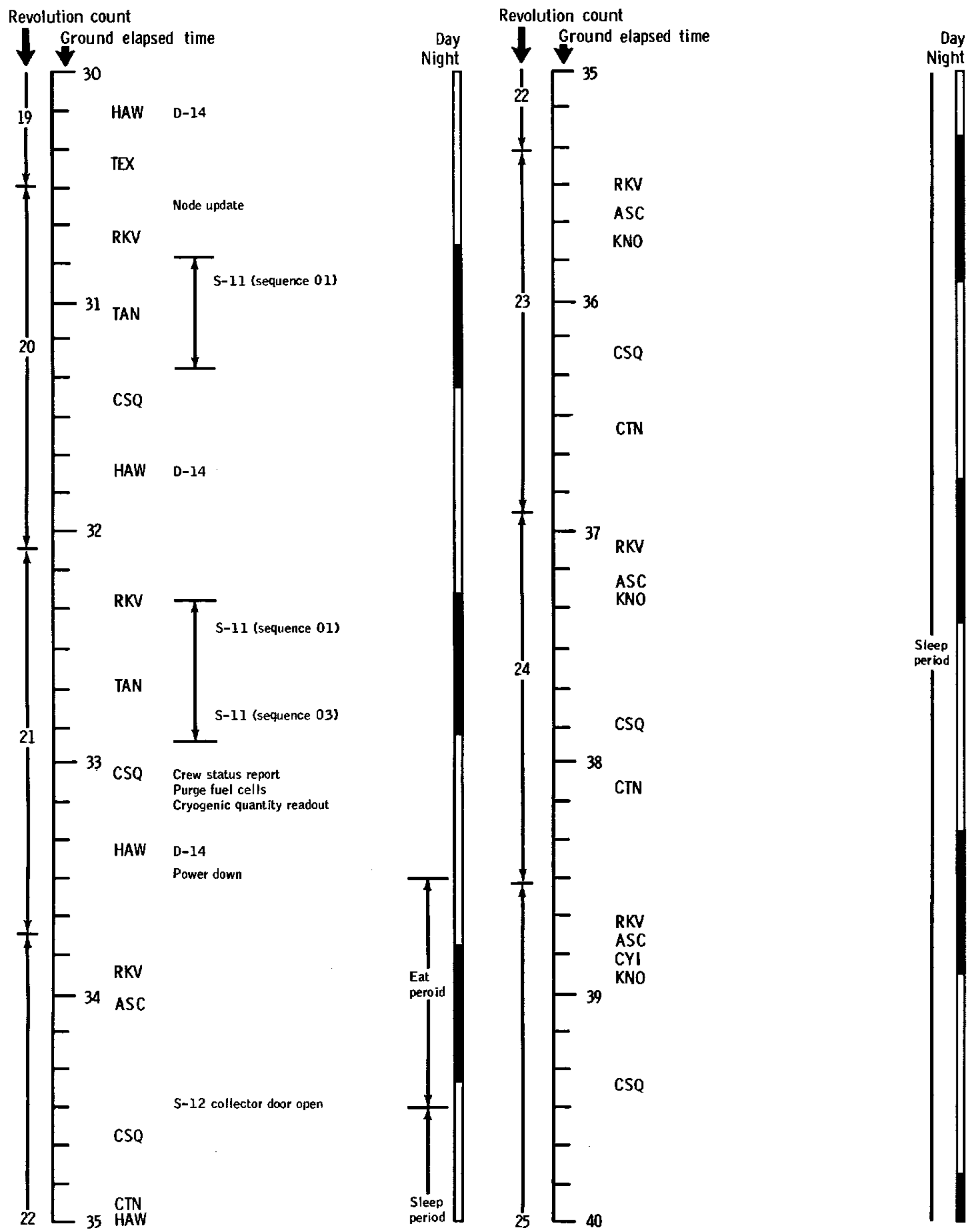
Figure 7.1.1-1. - Summary flight plan.



(b) 10 to 20 Hours g.e.t.
Figure 7.1.1-1. - Continued.

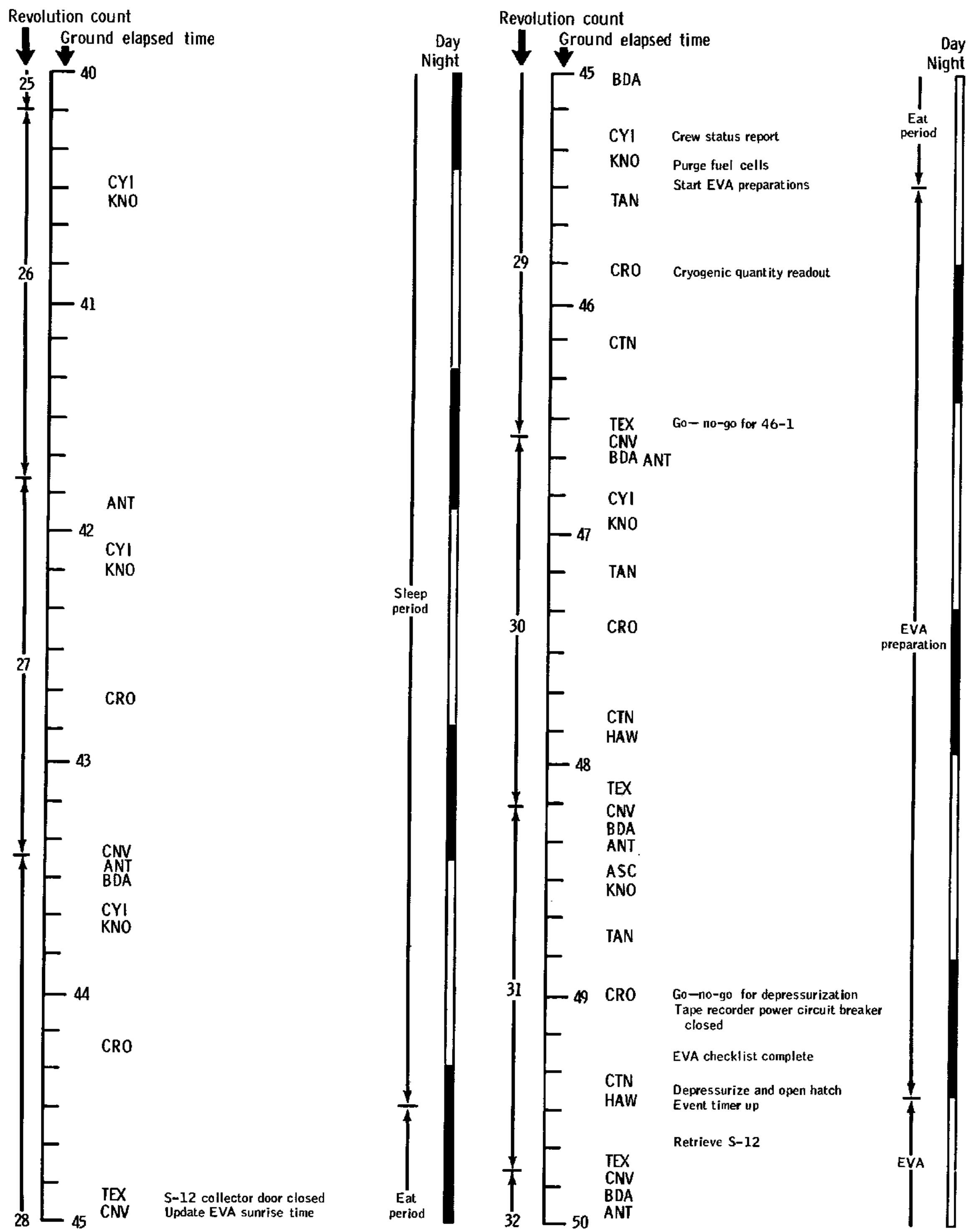


(c) 20 to 30 Hours g.e.t.
Figure 7.1.1-1. - Continued.

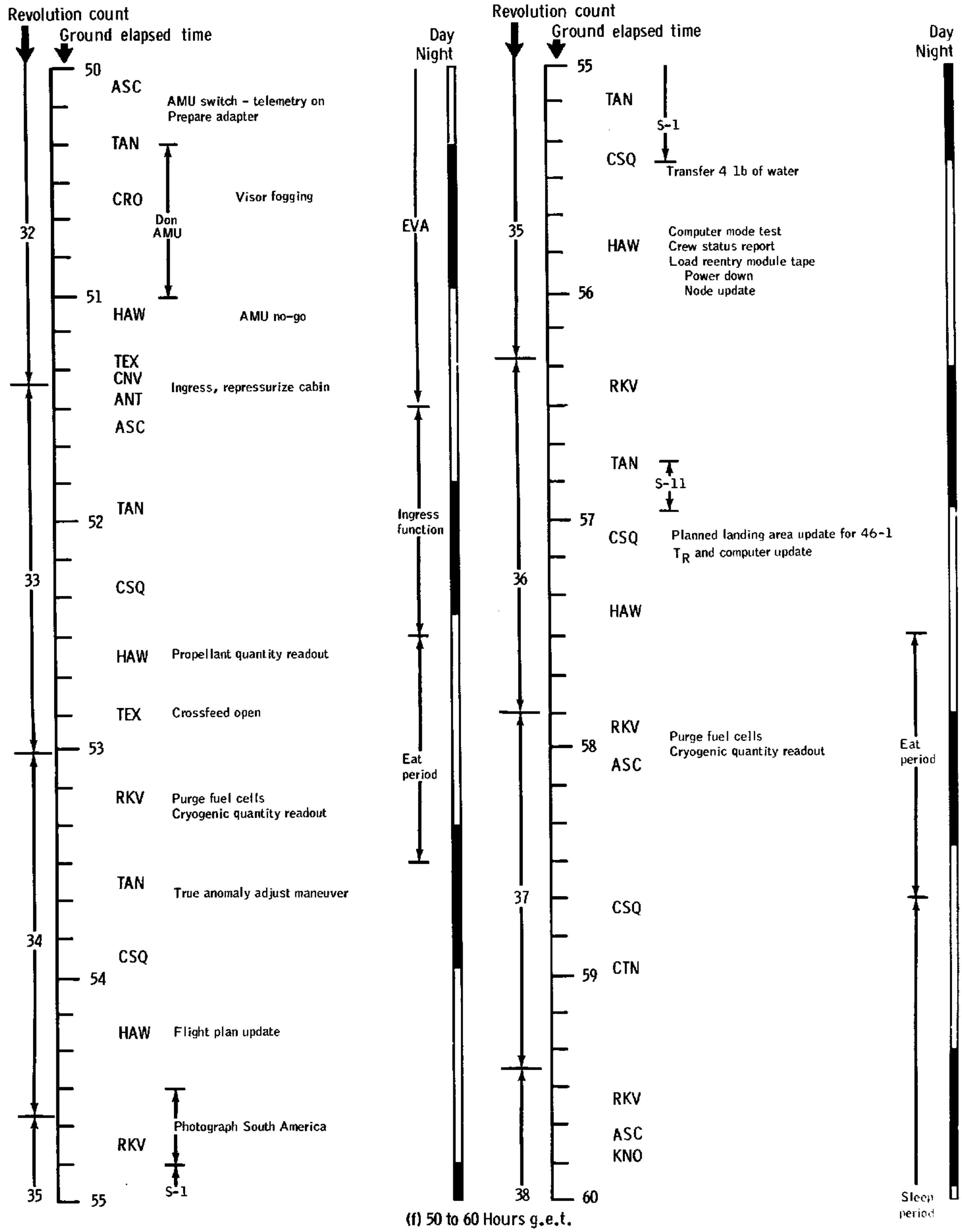


(d) 30 to 40 Hours g.e.t.

Figure 7.1.1-1. - Continued.

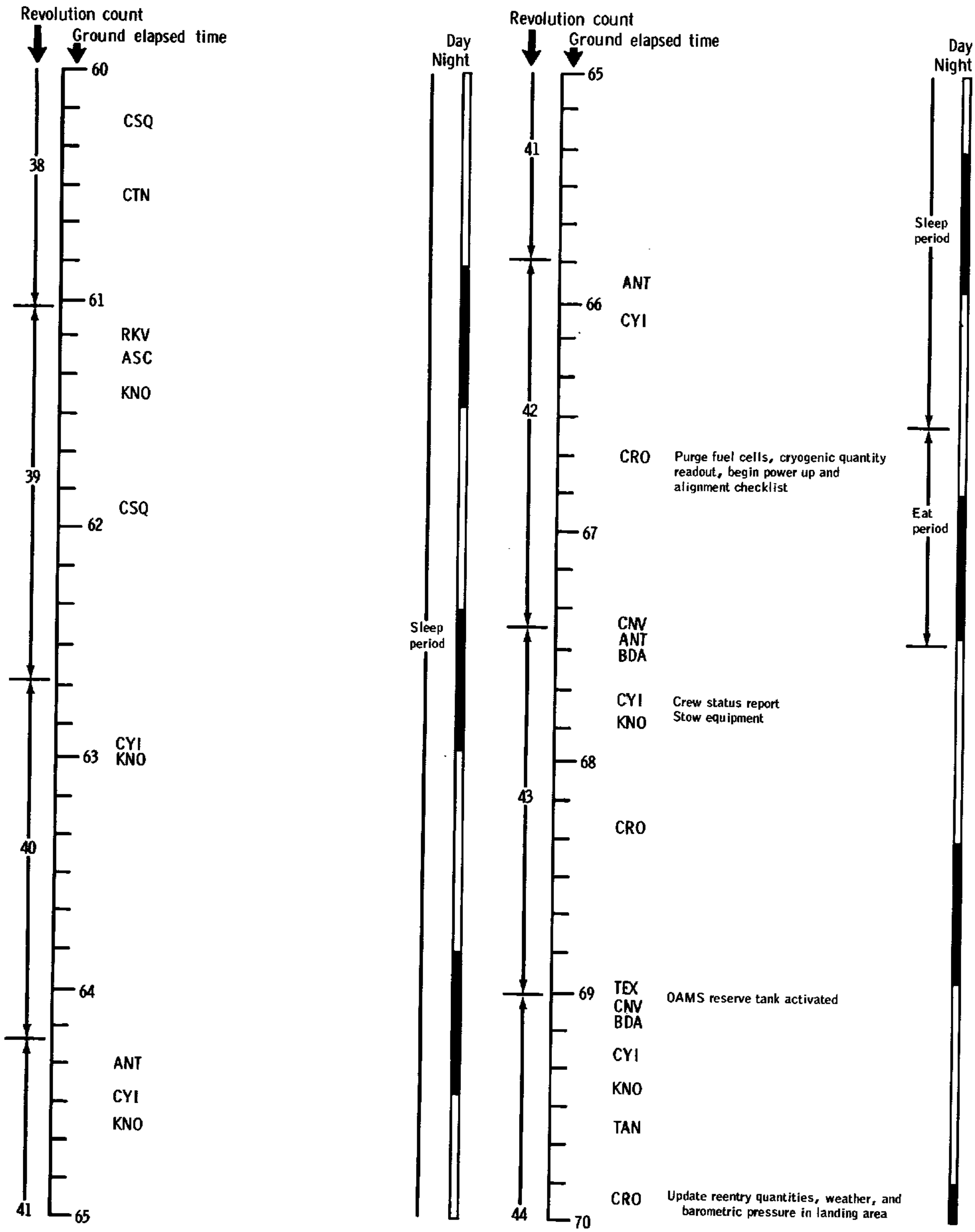


(e) 40 to 50 Hours g.e.t.
Figure 7.1.1-1. - Continued.



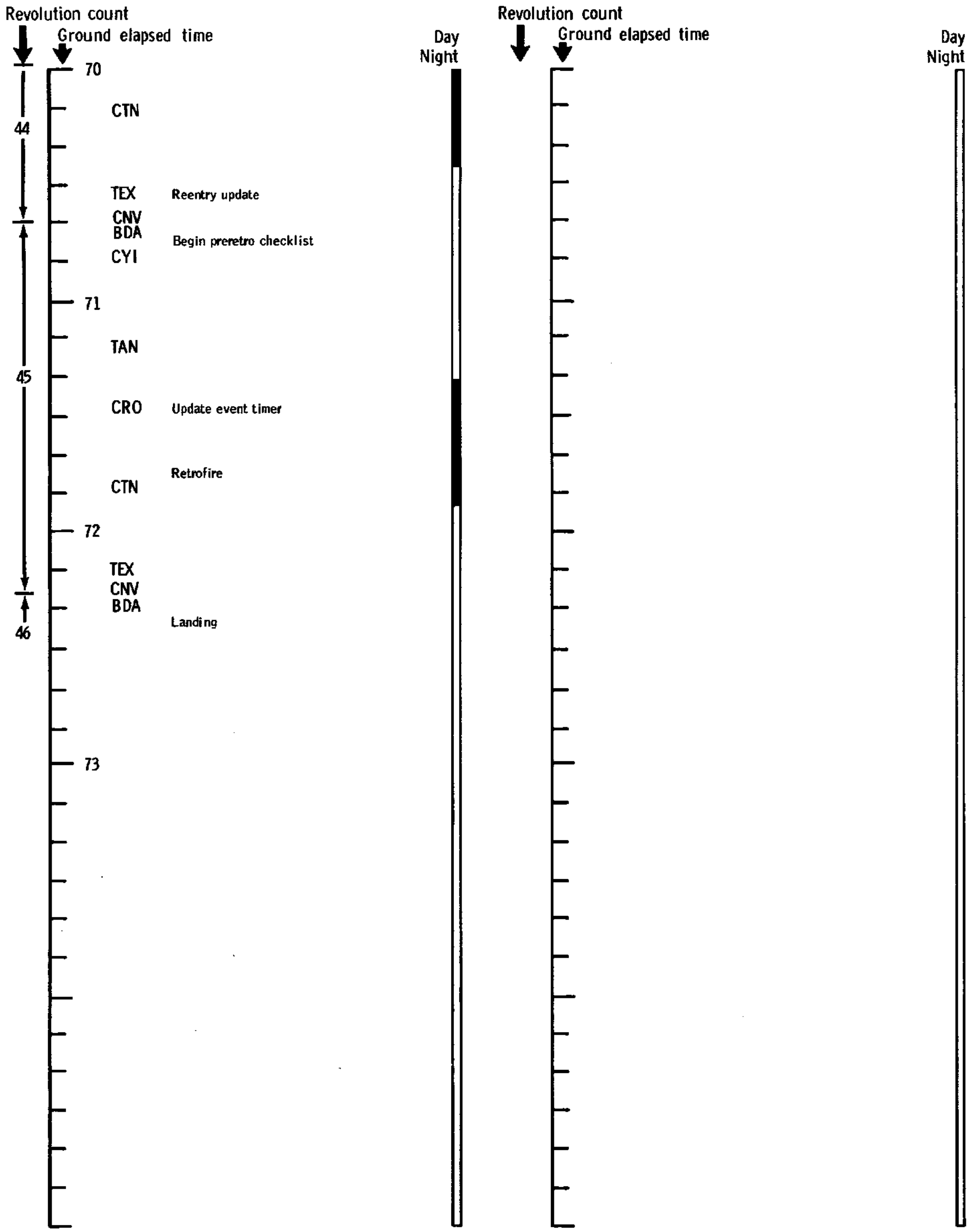
(f) 50 to 60 Hours g.e.t.

Figure 7.1.1-1. - Continued.



(g) 60 to 70 Hours g.e.t.

Figure 7.1.1-1. - Continued.



(h) 70 to 73 Hours g.e.t.
Figure 7.1.1-1. - Concluded.

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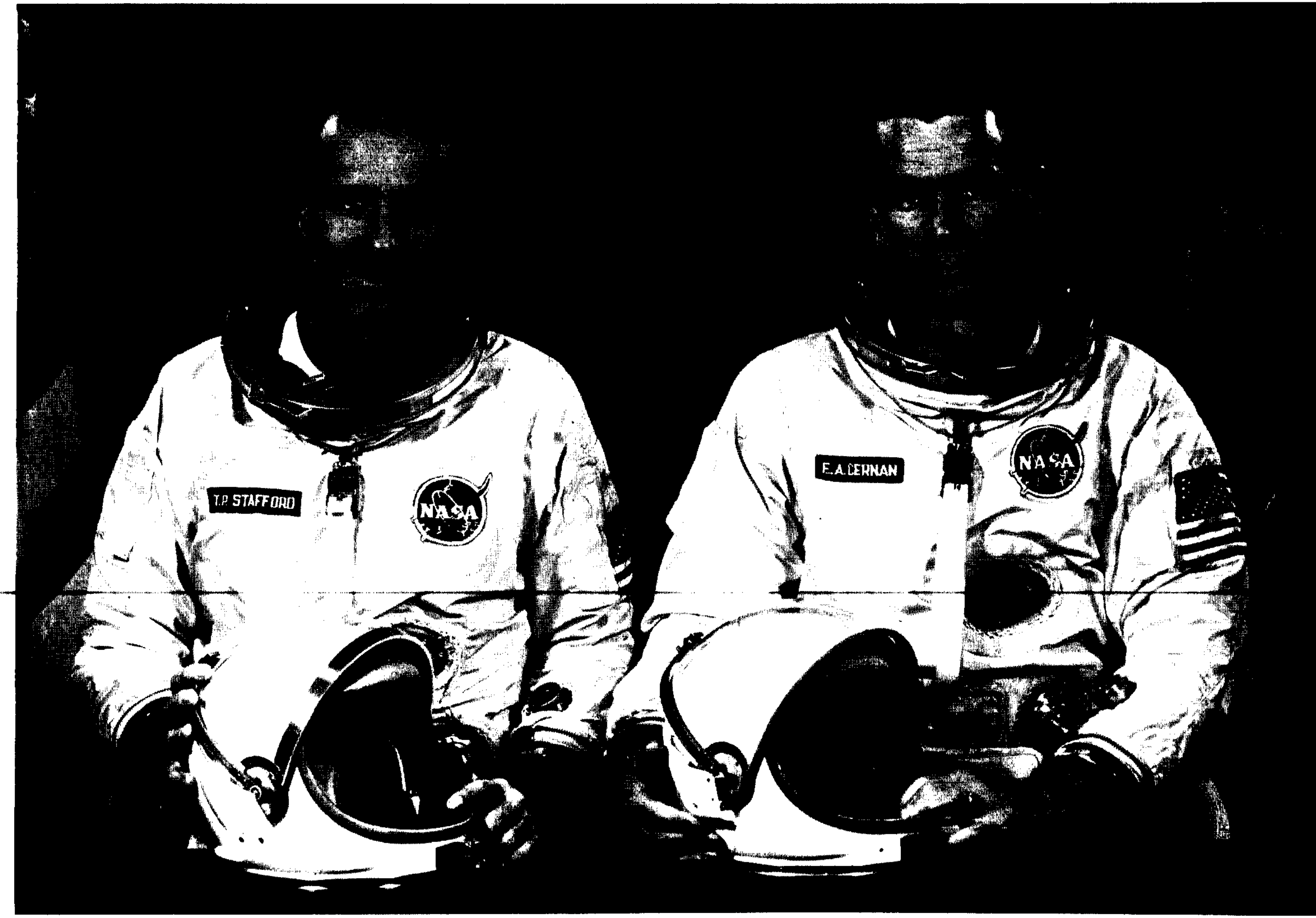

Astronaut Thomas P. Stafford, Command Pilot, and
Astronaut Eugene A. Cernan, Pilot.

7.1.2 Gemini IX-A Pilots' Report

Immediately after the loss of the Gemini Agena Target Vehicle on May 17, 1966, an all-out effort was initiated to ensure accomplishment of the three rendezvous operations of the original flight plan while using the Augmented Target Docking Adapter (ATDA). In one case, the rendezvous was altered to reduce the differential altitude for the rendezvous from above to approximately 7.5 nautical miles, using the Gemini spacecraft to perform the positioning maneuvers. The final flight plan with the ATDA basically included all the items of the original flight plan with the exception of using the Gemini Agena Target Vehicle propulsion system.

7.1.2.1 Prelaunch.- The simultaneous launch attempt with the TLV/ATDA on June 1, 1966, was satisfactory until the T-3 minute Digital Command System (DCS) update was not received. Two additional attempts to update the DCS were made in the next five minutes but were unsuccessful. The launch window stayed open until the opportunity for an M=6 rendezvous had passed, after which the launch for that day was cancelled.

Telemetry indicated that the nose shroud of the ATDA had not jettisoned in the separation sequence. It was arranged for the flight crew to spend several hours performing a thorough examination of each component contained within the shroud and noting the functions that these components perform during the separation sequence.

The launch of the Gemini Space Vehicle was rescheduled for June 3, 1966. On that date the crew ingressed at T-115 minutes, and the subsequent tasks and checkout progressed smoothly to the scheduled hold at T-3 minutes. When the countdown was resumed at T-3 minutes, the Inertial Guidance System (IGS) launch-azimuth update was not received, and the crew was informed by the Mission Control Center-Houston (MCC-H) that the backup plan to launch without these updates would be used. Further information indicated that a large out-of-plane Insertion Velocity Adjust Routine (IVAR) reading on the Incremental Velocity Indicators (IVI's) could be expected at insertion because, due to the uncertainty of the ground system, the two inflight updates to the IGS would not be transmitted.

7.1.2.2 Powered flight.- The blockhouse Capsule Communicator's countdown from 10 seconds to engine ignition progressed very smoothly. At engine ignition, the noise level built up in pulsating steps and then stabilized. Three seconds after engine ignition, a definite indication of lift-off could be felt. This was accompanied by a definite decrease in both noise and vibration levels.

The roll program was initiated at approximately 18 seconds after lift-off and was completed on time. The pitch program also was initiated on time. At approximately 50 seconds after lift-off, the vehicle had pitched and rolled to an attitude to where the sun was shining directly into the command pilot's eyes and was of such intensity that he could not see the forward instrument panel. He attempted to shield his eyes from the sun by placing his right hand in front of his face while keeping the left hand on the abort handle. During this period, the pilot had to make all of the required transmissions to the Capsule Communicator. This situation did not clear up until approximately 3 minutes after lift-off, and, for this period of time, only the stage II tank pressure gages could be seen. Physiological indications of POGO (low-frequency longitudinal oscillations) were not experienced during Stage I flight. The staging sequence occurred at the predicted time and was accompanied by a flame front that was somewhat less than what the command pilot had noted on Gemini VI-A.

Radio guidance initiate occurred at 2 minutes 48 seconds after lift-off, with very small deviations in attitudes and rate. Vehicle steering during this period was extremely smooth.

The IGS Flight Director Indicators were zeroed from guidance initiate until approximately 4 minutes into the flight, at which time the yaw-error indicator slowly began to drive toward the negative side; however, the pitch-error indicator remained at zero. By the time of Stage II engine cutoff (SECO) the yaw errors had built up to a maximum. All spacecraft systems operated satisfactorily throughout the first and second stages of powered flight, and normal pressures and temperatures were observed. The temperature and pressure of the hydrogen peroxide in the Astronaut Maneuvering Unit (AMU) held very stable at the pre-launch values. The fuel-cell performance was nominal throughout launch and shared the load with the main batteries. Fuel-cell differential-pressure warning lights were not observed to illuminate during powered flight.

The second stage burned slightly longer than expected, and SECO was observed to occur at approximately 5 minutes 39 seconds ground elapsed time (g.e.t.). After SECO, the combined GLV-spacecraft vehicle exhibited no residual rates. At SECO + 3.5 seconds, the fore-aft window of the IVI indicated 60 ft/sec forward with a large velocity in the down window (GLV yaw), indicating that the out-of-plane velocity on the IGS was as predicted. This large velocity produced transients in the fore-aft window each time a slight degree of yaw was experienced.

7.1.2.3 Pretransfer maneuvers.- Separation occurred on time. After a small separation thrust in a 90-degree right bank, the IVI's read 70 ft/sec forward, 26 ft/sec left, and 153 ft/sec down. The spacecraft was maneuvered to a zero roll and yaw attitude and then pitched up to null the pitch-error indicator to zero. The pilot coordinated the computer values, primarily for total inertial velocity and velocity to be gained at perigee, to ascertain the correct values of the incremental velocity required. At the completion of the separation maneuver, the total inertial velocity read 25 749 ft/sec; the velocity to be gained at perigee to correct apogee read -2 ft/sec, and the velocity to be gained at apogee for perigee read zero ft/sec. The insertion checklist was completed within approximately 10 minutes. The platform alignment was performed in both the pulse and platform control modes. The yaw-left tendency due to the water-boiler thrust was noted at this time. The primary and secondary scanners were checked and found to be in extremely close alignment with each other and with the visual reference to the horizon.

7.1.2.4 Rendezvous.-

7.1.2.4.1 First rendezvous (M=3): The midcourse rendezvous maneuvers performed prior to the terminal phase initiation are shown in table 7.1.2-I(a).

All of these pretransfer maneuvers were performed on the basis of ground-computed updates. The maneuvers were initiated on time with the computer reference being used for the pointing commands. The control mode flown was platform or rate command, depending upon whether or not the maneuver was made at 0, 0, 0 attitude. No difficulty was encountered in nulling out the residual desired velocities in the spacecraft body axes displayed by the computer in the Manual Data Readout Unit (MDRU). The in-plane residuals were reduced to within ±0.1 ft/sec, while residuals normal to the line of sight were easily held to within ±0.2 ft/sec. One computer anomaly was observed during the coelliptic maneuver $\left(N_{SR}\right)$; however, this anomaly did not affect the accuracy of the maneuver. When the catch-up mode of the computer was selected, the start-compute cycle was observed to initiate without depressing the START button. This automatic initiation of the computation cycle was subsequently observed in the rendezvous mode of operation.

The radar was turned on prior to the N_{SR} maneuver and intermittent lock-on was obtained at a range of 130 nautical miles. The lock-on cycle increased in time and became solid at 120 nautical miles. The radar attitude indicators were fairly smooth and exhibited only small oscillations after the initial lock-on. As the distance to the target decreased, the bias of the indicators and the sinusoidial oscillations superimposed on this bias both increased in amplitude. The initial bias and oscillations were observed to be less than ±1 degree. The attitude indicator motion gave valid indications when the radar switched from the dipole to the spiral antenna and vice versa. The bias would build up to a maximum of approximately one degree and then immediately jump back towards the zero position or even to the other side where the previous bias had been observed.

At a range of approximately 50 nautical miles, the ATDA appeared as a faint sixth-magnitude star in reflected sunlight. It was noted that the boresight between the optical sight and the radar was approximately 0.25 degree up from the zero radar boresight position.

After N_{SR}, onboard calculations of $\Delta\Delta R$ (difference between increments of actual range to the target as compared with the nominal increments) indicated that a differential attitude of 12.3 miles below the target had been obtained with zero ellipticity between the two orbits. Table 7.1.2-II shows the three solutions that were available to the crew prior to the initiation of the transfer maneuver. This table is presented in the form of ground-computed, IGS-computed, and onboard-backup solutions, and, derived from these solutions, the velocity changes that were ultimately used for the transfer maneuver. The IGS-computed and the onboard-backup midcourse solutions are shown, together with the midcourse corrections that were actually applied.

The apparent intensity of the ATDA increased in the reflected sunlight with decreasing range until it appeared slightly brighter than Venus, just before the ATDA passed the sunline. The ATDA disappeared in darkness approximately 5 minutes after the terminal phase initiate (TPI) maneuver and then reappeared as a flashing blue light. The appearance of the flashing light indicated to the crew that the nose shroud had probably been jettisoned. The flashing light would appear for periods of 10 to 12 seconds, disappear, and then reappear, indicating that the ATDA was rotating in various axes. At a range of approximately 4 miles, the red running lights were observed in addition to the flashing light. At a range to the ATDA of approximately 1 mile, a slight separation was observed between the red running lights. The amber and green lights had not been observed up to this time.

As range to the target decreased, the radar indicators of the Flight Director Indicator displayed increased bias and larger oscillations. These excursions did not present a problem to the crew, however, because the target was tracked continually with the optical sight. After the final midcourse correction, the line-of-sight errors were monitored and observed to be zero. At a range of one nautical mile, a braking maneuver was initiated that reduced the range rate from 32 ft/sec to 19 ft/sec with the aid of the computer in the catch-up mode. Prior to braking, the addresses for $\Delta\dot{X}$, $\Delta\dot{Y}$, and $\Delta\dot{Z}$ had been zeroed so that all subsequent velocity changes could be totaled and displayed at any desired time.

A further reduction in range rate was initiated at a range of 1800 feet. This second braking maneuver resulted in lowering the range rate to 7 ft/sec at a range of 1300 feet to the ATDA. The spacecraft continued to close at 7 ft/sec until the range was indicating 900 feet. At this time, range rate was further reduced to 5 ft/sec and maintained until final braking was accomplished and, at 4:15:00 g.e.t., station keeping was started at a separation distance of 60 feet from the ATDA. Figure 7.1.2-1 is a replica of the onboard plot of the spacecraft position relative to the target during the terminal phase of rendezvous.

Braking occurred completely during the night phase, but visual acquisition was greatly enhanced by the nearly full moon that existed. The braking phase was completed with the moon approximately 4.5 degrees to the right of the ATDA. The target vehicle was observed to be in tumbling flight through the entire braking and subsequent station-keeping operation. Roll rate was approximately 3 deg/sec; pitch and yaw rates were relatively small but were coupled with roll in all axes.

At approximately 1000 feet separation, it became apparent that the nose shroud was still attached to the ATDA but in a partially deployed position. The station-keeping distance was closed to within 20 feet during the night period. When sunrise occurred, the separation distance was reduced to within 10 feet. It was determined at that time that all explosive bolts had fired but that the electrical initiating wires were holding the shroud in place. The two electrical umbilical disconnects near the rear of the shroud were also determined to be in place.

7.1.2.4.2 Second rendezvous (equi-period): The impossibility of docking at this time led to the adoption of the alternate flight plan. The equi-period separation maneuver was planned and executed at 5:01:00 g.e.t. (see table 7.1.2-I(b)). During separation and prior to darkness, the planned operations for sextant practice were attempted. However, it became evident that acquisition of a moving target over a lighted earth was a very difficult task. The use of the sextant was deemed not feasible during daylight operations with the target below the spacecraft. The crew, however, was successful in using the sextant to determine a pre-horizontal adjust correction by utilizing the ATDA flashing light and a night horizon. The ultimate correction for the horizontal adjust maneuver was determined as planned by maintaining the spacecraft at a 0, 0, 0 attitude and noting the elapsed time from the radial separation maneuver that the target passed through the zero position of the reticle. The time of the horizontal sighting was plotted on an onboard chart and showed that a horizontal adjust maneuver was not required.

After darkness occurred, the flashing light was intermittently seen as the ATDA tumbled, and the red running light was discernible out to a distance of approximately nine nautical miles. After sunrise, no visual acquisition of the target could immediately be made; however, at the programmed time, the spacecraft was rolled to an inverted attitude and, within 10 seconds, both crewmembers observed the ATDA in reflected sunlight at a magnitude estimated to be that of Venus or slightly greater. This magnitude gradually increased throughout the remainder of the rendezvous until the total outline of the ATDA could be seen in reflected sunlight at a range of approximately 1.5 nautical miles.

Terminal phase initiation was computed on the basis of an elevation-angle change over a prescribed period of time. The first and second midcourse corrections were based on a similar change in pitch angle. The second midcourse correction used data from the sextant as well as the timed rate of angle change indicated on the attitude indicator. (Table 7.1.2-III lists the onboard data for the second rendezvous maneuvers.) It was noted that sextant angles were very difficult to obtain with any assurance of accuracy when the angles exceeded approximately 70 degrees. Braking was initiated with the aid of the catch-up mode of the computer in a manner similar to that used on the prime rendezvous. No difficulties were encountered during the braking phase. The line-of-sight control was maintained while using the attitude indicator in orbit rate, which resulted in the reference rotating at 4 deg/min.

Optical estimations of range at less than one mile correlated very closely to that obtained from radar data. The braking phase and estimates of range and range rate were obtained using optical techniques. Figure 7.1.2-2 is the onboard plot of spacecraft position relative to the target vehicle during the equi-period rendezvous.

After station keeping for a period of time, a separation maneuver of 3.7 ft/sec retrograde occurred at 7:14:58 g.e.t. All residual velocities were nulled. The first two rendezvous operations resulted in a work task that precluded the crew having a specific meal period. Approximately one meal was shared by the crew during this period. The crew then consumed one meal each during the programmed meal period but obtained only light periods of sleep during the programmed sleep period.

7.1.2.4.3 Third rendezvous (rendezvous from above): Following the sleep period, a series of maneuvers was performed to place the spacecraft in a coelliptic orbit above and ahead of the target vehicle to set up the third rendezvous, the rendezvous from above. These maneuvers are depicted with the associated changes in velocity in table 7.1.2-I(c). After N_{SR}, the inflight calculations of range indicated that the spacecraft had an ellipticity relative to the target vehicle orbit. This ellipticity resulted in a differential altitude that varied from 7.5 nautical miles at N_{SR} up to 8.2 nautical miles and back down to 7.1 nautical miles just prior to terminal phase initiation. The values of the ground-computed, IGS-computed, and onboard-backup solutions of the transfer maneuver, in addition to those values actually used, are shown in table 7.1.2-IV. Included in this table are the backup and the closed loop midcourse correction.

After N_{SR}, as the range was closing, the radar indicators performed in a manner similar to that observed during the first rendezvous. As range decreased, the bias buildup of the indicated angles and the sinusoidal motion superimposed on this bias would increase just prior to antenna switching. As a result of light from the full moon reflected off of the target, at a range of 20 nautical miles, the target appeared approximately 5 to 10 times the diameter of Venus. This reflective target appeared as a whitish-blue disc that moved rapidly across the black earth below. At sunrise the color changed into a whitish-orange hue, and the apparent diameter immediately decreased in size. The target disappeared completely from view approximately 2 minutes after sunrise. The remainder of the rendezvous was conducted solely on radar. The target was again acquired through the use of radar at less than 3 nautical miles range when it appeared visually as a small black dot moving rapidly across the Sahara desert. The target was not continually visible until approximately 2 nautical miles range when it crossed over into the Indian Ocean. The reticle pattern of the optical site was lost from view due to the bright reflected sunlight from the Sahara desert when the spacecraft was pitched down in excess of 60 degrees. The reticle pattern did not reappear until the final phase of braking over the Indian Ocean.

Because the spacecraft altitude was approximately 25 nautical miles above the 146-nautical-mile circular reference orbit, the platform was aligned during the programmed period between 5 and 10 minutes after TPI. Braking was accomplished with the aid of the computer in the same manner as in the prime rendezvous; however, it was scaled down on the basis of the differential altitude. Figure 7.1.2-3 is a plot of the spacecraft position relative to the target during the final phase of rendezous.

After a station-keeping position was achieved, the spacecraft was stabilized on the tumbling ATDA, and the distance was reduced to approximately 3 inches. This position was maintained in order to obtain pictures of the area where the explosive bolts of the strap had fired but the electrical initiator wires were intact. After obtaining this photographic information, a retrograde maneuver was made to separate the spacecraft from the ATDA. Subsequent to the retrograde maneuver, radar lock-on was obtained in excess of 179 nautical miles.

7.1.2.5 Extravehicular activity.- Preparation for the extravehicular activity (EVA) commenced at the programmed time and progressed ahead of schedule until an apparent control system malfunction was noted. Analyzing the malfunction (determined to be an open scanner-heater circuit breaker) and applying the corrective action entailed approximately 25 minutes. In spite of this delay, the preparation for EVA was completed approximately 30 minutes ahead of the programmed time to depressurize the cabin. During this 30-minute period, the crew rested and reviewed the flight plan and emergency procedures. The final suit-integrity checks were made, and the cabin pressure was decreased in increments until it reached zero at 10 minutes prior to sunrise.

The hatch was then opened in darkness. No sudden opening of the hatch, as a result of residual pressure, was observed. The hatch was difficult to open, however, after it had moved approximately 10 inches from the full-closed position. At sunrise, the programmed checklist was continued by placing cameras, handrails, et cetera, in the proper position. The EVA docking-bar mirror was then taken to the nose of the spacecraft and placed on the docking bar by the pilot. Throughout this period, the pilot encountered difficulty in trying to stay in the seat because he tended to float up and out of the spacecraft. The same difficulty was experienced in traversing the distance to the docking bar. Subsequently, a Velcro-pad evaluation took place, both at the retrograde adapter and at the rendezvous and recovery section of the spacecraft. The pilot returned to the hatch area 15 minutes prior to sunset and stood in the seat. The EVA 16-mm camera was handed back to the command pilot, who changed the lens and the film pack without any difficulty. The camera was then reinstalled on the retrograde adapter. The hatch was closed with the same difficulty previously experienced in the midranges of travel. However, the travel over the last 10 inches prior to closing the hatch was quite easy. It is estimated that a force slightly in excess of 100 pounds was required to open or close the hatch in the mid-travel range.

The pilot then held on to the handrails on the adapter assembly while the command pilot extended the EVA handholds in the adapter equipment section. Upon indication from the command pilot that the EVA handholds had been extended, it was noted immediately by the pilot that they had either not extended fully or possibly not extended at all because the umbilical guide had not swung into view. However, the electrical signal had been received because the AMU telemetry antenna on top of the adapter assembly of the spacecraft was observed to extend. Upon reaching the adapter end of the spacecraft, the pilot observed that the thermal cover, the umbilical guide, and one handhold had not fully deployed. The footrail and the second handhold, however, were fully extended. With comparative ease, the pilot was able to free the cover by imparting a small force on the partially deployed handhold. The adapter was then properly configured and the pilot proceeded with the AMU donning.

During the period between the time the pilot started back to the adapter assembly of the spacecraft until he gave the go to turn the attitude control power switch on, he produced torques that yawed the spacecraft 150 degrees, rolled it past the vertical position, and pitched it down over 40 degrees. Throughout the first phases of EVA, the command pilot noted that the pilot perturbated the spacecraft attitude beyond what had been anticipated. The platform control mode was used whenever possible throughout the entire EVA. When the pilot's operations required him to be in the vicinity of the attitude thrusters, the command pilot turned off power to the control system and allowed the spacecraft to drift. In one 30-second period, the pilot pitched the spacecraft up 30 degrees before he was clear of the thrusters, after which the command pilot could restore the spacecraft to the original attitude. Pulse mode and occasionally direct mode, along with the rate-command mode, were used to reposition the spacecraft before the platform mode was re-activated.

The pilot remained comfortable, although not cool, during the entire period following egress, and maintained medium flow on the Extravehicular Life Support System (ELSS). A visual inspection of the adapter revealed that the separation plane had jagged edges as expected, but there was no loose debris like that seen on Spacecraft 6 and 7. As the AMU donning procedures began just prior to local sunset, the pilot became extremely warm in a localized area of the back. In an attempt to alleviate this local hot-spot, all work was ceased, and the ELSS was placed in high flow until darkness occurred. At sunset the warm area disappeared, and the pilot momentarily went back to medium flow on the ELSS.

As work continued in the donning of the AMU, the pilot observed fog starting to form on the lower part of his visor. At this point he returned to high flow on the ELSS chestpack. Additional work above what had been anticipated was required for the pilot to maintain his position in the adapter during the donning of the AMU. This work comprised approximately 50 percent of the total work output. The fogging increased until, during the final stages of donning the AMU, it was considered objectionable and possibly even a factor of safety in continuing. The handrail light looked like a headlight on an approaching automobile when driving in a fog. At this point the AMU had been fully donned with the exception of the oxygen hose and the restraint strap. The pilot then rested in the adapter until sunrise, hoping that this rest would eliminate or at least reduce the fogging. No significant improvement occurred after sunrise. The situation was evaluated by both the command pilot and the pilot. Both believed that further operations with the AMU would be limited and could produce a safety of flight hazard; therefore, continuation of the donning would not be feasible. All restraints and ties with the AMU and the adapter were then freed and the pilot commenced to leave the adapter without the AMU.

Returning to the hatch area was accomplished with little difficulty, although the fogging was 100 percent and did persist. The pilot noted that the hatch seal was still soft and pliable. It appeared similar in physical nature to what it had been following the initial opening. With the sunvisor up and the face effectively pointing toward the sun, the pilot's visor commenced to defog slowly but without enough effectiveness to warrant continued EVA. As a result, conducting the zodiacal light experiment (S-1) during EVA on the following night was not considered feasible. After returning the cameras to the cockpit and discarding the docking-bar mirror, the pilot began his ingress.

The only anomaly which occurred during ingress was that the hatch was again difficult to operate in the mid-range. As a result, the pilot had to rise out of the spacecraft slightly to move the hatch in impulsive increments in order to bring it down to a position where the command pilot could fully utilize the hatch-closing device to aid with the ingress. In contrast to the stiffness in the midrange, the freedom of movement of the hatch in the last 10 inches prior to being fully closed was exhibited by the command pilot being able to exert enough force actually to aid the pilot during ingress in getting fully within the envelope of hatch closure. Locking and subsequent cabin repressurization were as planned without further incident. When the hatch was locked and pressure started to rise, the command pilot noticed that the pilot's visor was completely fogged and consisted of a complete mass of condensation and water on the faceplate. He could not see the facial characteristics of the pilot through the fogged visor.

During the period of visor fogging, the pilot's suit pressure gage was also observed to be fogged. Subsequent to the EVA, the gage contained a great deal of moisture in the form of droplets.

7.1.2.6 Experiments.- The Airglow Horizon Photography experiment (S-11) was carried out as prescribed in the preflight documents. The one item noted by both pilots was that the bracket for the experiment was not aligned with the optical reticle and, thus, the spacecraft longitudinal axis. This non-alignment contributed to an increased workload required of both crewmembers in getting an accurate alignment of the camera during the night airglow. The experiment was carried out satisfactorily, and results appear to be very successful based upon the data that have been reviewed.

The Zodiacal Light Photography experiment (S-1), another low-level light photographic experiment, was conducted intravehicular rather than extravehicular because of problems previously noted during EVA. The camera was held against the window without a bracket for several 30-second-exposure time periods and the results were satisfactory. This method of stabilizing the camera against the spacecraft window appears to have worked well with exposure times as long as one-half minute. However, as covered in the S-11 experiment summary, the fact that the window and/or the camera was not boresighted with the spacecraft axis made pointing of the camera in a specific direction difficult. This feature required that the spacecraft be in a powered-up condition for both pointing and nulling of the three-axes rates within the desired limits.

The UHF/VHF Polarization experiment (D-14) was carried out in the prescribed manner. It should be emphasized that considerable fuel was used in flying to the desired attitude following a platform alignment because of the position of the antenna in the retrograde adapter. Careful consideration should be given to the alignment of all experiments to complement the crew workload and the fuel usage. The preliminary analysis of the results indicate that all data were satisfactory.

The Bio-Assays of Body Fluids experiment (M-5), was carried out completely. It was immediately noted by the flight crew that the obtaining of urine samples, labeling the bags, and stowing the bags throughout the entire mission created a workload that was estimated to be equal to 1.5 rendezvous operations. Every minor cockpit restowage required that the urine bags be stowed in several different places. This imposed a considerable workload upon the crew. When water was dumped through the urine dump system, care had to be taken in positioning the urine chemical sampling device.

The Astronaut Maneuvering Unit experiment (D-12), was not evaluated because of the fogged-visor complication experienced during EVA. However, as far as the donning procedures went and as far as the AMU systems were monitored, the AMU was fully operational and, in all respects, was ready to fly. It should be noted, however, that it was very difficult to open the oxygen valve, that the right-hand arm controller did not fully extend at first, and that the UHF communications between the pilot and the command pilot while the pilot was in the adapter were significantly degraded. The voice-operated transmitter (VOX) also appeared to be more sensitive than anticipated.

Because of the availability of film, electrical power, and OAMS fuel, a series of color photographs was made of the west coast of South America, including a majority of the Andes mountains. Strip maps were made of this terrain feature. On the morning prior to retrofire, maping photographs were taken across the Sahara desert in Africa and extending into the Indian Ocean to record both terrain and weather features.

7.1.2.7 Reentry.- The preparation leading up to retrofire involved the aligning of the platform and the checkout of the Auxiliary Tape Memory unit in which module 4A was automatically loaded and verified. The cockpit was restowed to the configuration outlined in the flight plan, with the exception that the umbilical and the extravehicular sun-visor were in the pilot's footwell. Neither of these two items caused any discomfort or any problems during the entire retrofire, reentry, and landing sequence.

The time-to-go-to-retrofire and the preretrofire command load were received by the Digital Command System and were satisfactory upon the first transmission. The MDRU readouts agreed with those received from the ground. Bank angles, times, and recovery call signals were all received satisfactorily prior to retrofire. The automatic retrofire sequence initiated the retrofire maneuver. All four retrorockets fired in the proper sequence; however, it appeared that there was a hesitation between the third and fourth retrorocket firing. The applied retrofire velocity increment, as indicated by the IVI, was 296 ft/sec aft, 4 ft/sec right, and 125 ft/sec down. Both Reentry Control System (RCS) A-ring and B-ring were used in the rate-command mode for retrofire. After adapter jettison, A-ring and pulse mode were used until guidance initiate. At that time, A-ring was continuously used with the reentry rate-command mode. After the drogue parachute deployed, the B-ring was also engaged. Both rings were turned off after 25 000 feet by the motor valves and then, when the propellant was depleted from the manifolds, electric power was turned off to both rings. Although both rings were completely shut down and not indicating any firing activity, a yellowish flame was noted from thruster 8 on both rings and from thruster 3 on the B-ring.

An altitude of 400 000 feet was indicated on the initial guidance system within one second of that predicted from ground data. The spacecraft was rolled left to a bank angle of 50 degrees. This was later changed to 28 degrees left as the ground update for the bank angle was received. The initial indication of downrange error was approximately 120 nautical miles, and this compared satisfactorily with the ground estimate of 94 nautical miles. The 28-degree bank angle was maintained for approximately 40 seconds while downrange and crossrange errors were monitored, and appeared to be approaching the desired null. The downrange and crossrange oscillations were far less than those experienced in the Gemini Mission Simulator. After approximately 40 seconds on the 28-degree-left bank angle, the roll indicator was followed for bank-angle commands. When the crossrange and downrange indicators achieved a zero-error position, the computer commanded full-left roll.

After the full-left roll indications were received and followed, the computer commanded a full-lift position for approximately 20 seconds, followed by a full-right roll. Maximum acceleration experienced during the reentry was approximately 6g. After the acceleration decreased to 3g, the pilot was able to read out the instantaneous latitude and longitude of the spacecraft. By projecting the various data points that he could read out over a time period, it was apparent some slight overshoot might be experienced even though the attitude indicator pointers were completely nulled and the computer was commanding full roll. At this time, the pilot instructed that the command pilot fly full-negative lift, which was accomplished, and this position (heads up) was maintained throughout the remainder of the lifting period. The drogue parachute was deployed slightly below 50 000 feet and was accompanied by oscillations similar to those that have been described on previous missions. At this time, the secondary RCS ring was placed on. At 27 000 feet, oxygen high-rate was actuated. No fumes were detected in the cabin throughout the entire reentry sequence. The main parachute was actuated at 10 000 feet, and it deployed in a nominal manner, was inspected for tears, and found to be in perfect condition. At this time, the previously described thruster fire was noticed, and the crew elected to remain in the single-point-suspension attitude for approximately 3000 feet. As soon as the thruster 3 fire went out, the spacecraft two-point-suspension attitude was initiated. This was accompanied by an oscillation that was estimated to be approximately three-fourths of that experienced on Gemini VI-A. The fire in thrusters 8 in both rings continued to burn until the spacecraft landed. UHF communications were good throughout all phases on the drogue parachute and after obtaining two-point suspension. The crew saw a recovery helicopter directly ahead of the spacecraft after the two-point suspension oscillations ceased.

Reentry photographic information was obtained by leaving both 16-mm Maurer movie cameras attached to the window brackets. These movie cameras were actuated in series during the ionization part of the reentry. As had been practiced in the simulator, both cameras were removed from the windows after the spacecraft passed 100 000 feet, and they were placed in the footwells, outside of the ejection envelope.

7.1.2.8 Landing.- The landing impact was greatly in excess of what the crew had expected. Water was noted in the command pilot's footwell immediately after landing. The cabin repressurization valve was immediately placed to the full-open position and the water seal closed as outlined in the checklist. It was later determined that this water probably entered the spacecraft through the cabin pressure relief valve. Approximately 10 seconds after obtaining a stable floating position, two pararescue swimmers were seen to land in the water within 10 feet of the spacecraft. Excellent UHF contact was made with the surrounding helicopters and other members of the recovery forces; therefore, the high frequency (HF) antenna was not extended.

The flotation collar was rapidly attached and inflated. The crew then opened the hatches to take advantage of the cool ocean air. The hatches were closed while the spacecraft was being hoisted aboard the carrier.

7.1.2.9 Systems operation.-

7.1.2.9.1 Platform: The platform was aligned using both primary and secondary horizon scanners. It was noted quantitatively that the secondary scanner may have produced a slightly more accurate platform alignment than did the primary. The secondary scanner did provide a better attitude control mode during the sunrise and sunset periods.

7.1.2.9.2 Computer: The computer performed nominally except during the period after the corrective combination maneuver and prior to the coelliptic maneuver through the first rendezvous. The crew noticed that, when switching into the catch-up mode, the START COMP light would come on and the IVI would count up to the value that was in the register for all three axes. In the rendezvous mode after eight data points were received, the total-velocity-to-rendezvous would be displayed, followed by the velocity to apply from that given point. After the first rendezvous was completed, no further anomalies were noted on the computer.

7.1.2.9.3 Environmental Control System: The Environmental Control System (ECS) performed satisfactorily; however, when the spacecraft was fully powered-up, both suit fans were required to keep the crew at the desired comfort level. For the first sleep period, approximately 2 to 3 hours were required for the crew to become comfortable using only one suit fan. During the second and third sleep periods, both coolant pumps were left in the B position, and only one suit fan was utilized. After 2 to 3 hours, the suit flow was reduced to the medium range.

The drinking-water gun had one mechanical malfunction. During the second day, the actuating lever on the back of the gun would not stroke to the full-top position. The crew examined the gun and were able to force the lever to the full-top position, and after a number of cycles, the anomaly disappeared. Just prior to retrofire, the quantity of water received from the drinking gun per cycle decreased noticeably, and very little water could be obtained from the gun. In fact, only one or two drops per cycle were obtained. Since this anomaly occurred just prior to retrofire, no further investigation was conducted in flight.

An ECS system anomaly was observed on the morning prior to retrofire when the crew noticed that the cabin pressure had dropped from 5.0 to 4.7 psid. The cabin repressurization valve was opened, and the pressure increased to 5.0. After approximately 30 minutes, the pressure again decayed to 4.7. The cabin pressure was again increased to 5.0, and the water seal was actuated. The cabin pressure then held steady for the next hour. After this period the water seal was opened, and the cabin pressure remained at 5.0 psi and held this value throughout the remainder of the flight.

7.1.2.9.4 Electrical System: The fuel-cell operation was excellent. The six ammeters used for monitoring the individual fuel-cell outputs indicated that each stack shared the load within ±0.5 ampere. One exception was noted on the third day when stack 2C dropped one ampere below the values that the other five stacks were carrying. The fuel-cell purges were nominal during the hydrogen sequence on each section. A differential-pressure warning light was observed when the oxygen was purged in either section. The differential pressure gage on the pilot's panel indicated only minor fluctuations, when either the hydrogen or water position was selected during the purges. The main battery voltages were seen to maintain their relative difference with battery 2 indicating about 2 volts higher than battery 1 which had the lowest voltage. Each battery maintained its relative position throughout the flight, including the retrofire and reentry sequence. There were no problems with circuit breakers opening except inadvertent actuation by the crew during EVA preparation, egress, and ingress into the spacecraft.

TABLE 7.1.2-I.- PRETRANSFER MANEUVERS

(a) Prime rendezvous

Maneuver	Ground elapsed time, hr:min:sec	ΔV, ft/sec	Spacecraft attitude, Yaw, deg	Spacecraft attitude, Pitch, deg	Control mode	Residual velocities, ft/sec ΔV_{xb}	Residual velocities, ft/sec ΔV_{yb}	Residual velocities, ft/sec ΔV_{zb}
Phase adjust	00:49:03	75.0	0	0	Platform	+0.1	+0.2	0.0
Corrective combination	01:55:17	14.6	67 left	44 up	Rate command	0.0	0.0	0.0
Coelliptic	02:24:51	54.0	3 left	41 down	Rate command	0.0	+0.1	+0.1

(b) Equi-period rendezvous

Maneuver	Ground elapsed time, hr:min:sec	ΔV, ft/sec	Yaw, deg	Pitch, deg	Control mode	ΔV_{xb}	ΔV_{yb}	ΔV_{zb}
Separation	05:01:00	20.0	0	90 down	Rate command	0.0	0.0	0.0

(c) Rendezvous from above

Maneuver	Ground elapsed time, hr:min:sec	ΔV, ft/sec	Yaw, deg	Pitch, deg	Control mode	ΔV_{xb}	ΔV_{yb}	ΔV_{zb}
Separation	07:14:58	3.7	180	0	Platform	0.0	0.0	0.0
Phase adjust	18:23:19	2.0	0	0	Platform	0.0	-0.1	0.0
Height adjust	19:08:16	17.0	0	0	Platform	0.0	+0.1	-0.2
Coelliptic	19:54:24	14.4	180	38	Rate command	0.0	+0.1	-0.2

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TABLE 7.1.2-II.- COMPARISON OF SOLUTIONS FOR THE TRANSFER MANEUVERS FOR THE M=3 RENDEZVOUS

Maneuver	Solution, ft/sec			
	Ground	Computer	Backup	Used
Transfer	26.7FWD 1.3UP 2.2R	26FWD 8UP 4R	24FWD 0 -	27FWD 1UP 2R
First correction			2AFT 3UP	0 0
Second correction (ωt = 82 deg)		2AFT 2UP 3R	3AFT 4UP -	2AFT 2UP 3R
Third correction			1AFT 1UP	0 0
Fourth correction (ωt = 33 deg)		3FWD 2DN 0	0 3DN -	3FWD 2DN 0

TABLE 7.1.2-III.- MANEUVERS FOR THE EQUI-PERIOD RENDEZVOUS

Maneuver	ΔV, ft/sec	Time, sec	Used, sec
Horizontal adjust	0.1 FWD 0.2UP	0 0.5 0.5	0 0
Transfer	-- --	2AFT 5DN	2AFT 5DN
First correction	-- --	0 2DN	0 2DN
Second correction	--	0 2DN	0 2DN

TABLE 7.1.2-IV.- COMPARISON OF SOLUTIONS FOR THE TRANSFER MANEUVERS FOR THE RENDEZVOUS FROM ABOVE

Maneuver	Solution, ft/sec			
	Ground	Computer	Backup	Used
Transfer	16.5FWD 0.3UP 2.5R	19FWD 1DN 2L	16.5FWD 3UP -	17FWD 3UP 0
First correction			0 3DN	0 0
Second correction (ωt = 82 deg)		4AFT 1UP 5L	Platform alignment, no solution	4AFT 1UP 3L
Third correction			2AFT 2DN	0 0
Fourth correction (ωt = 33 deg)		2FWD 10DN 7R	1AFT 5DN -	2FWD 10DN 7R

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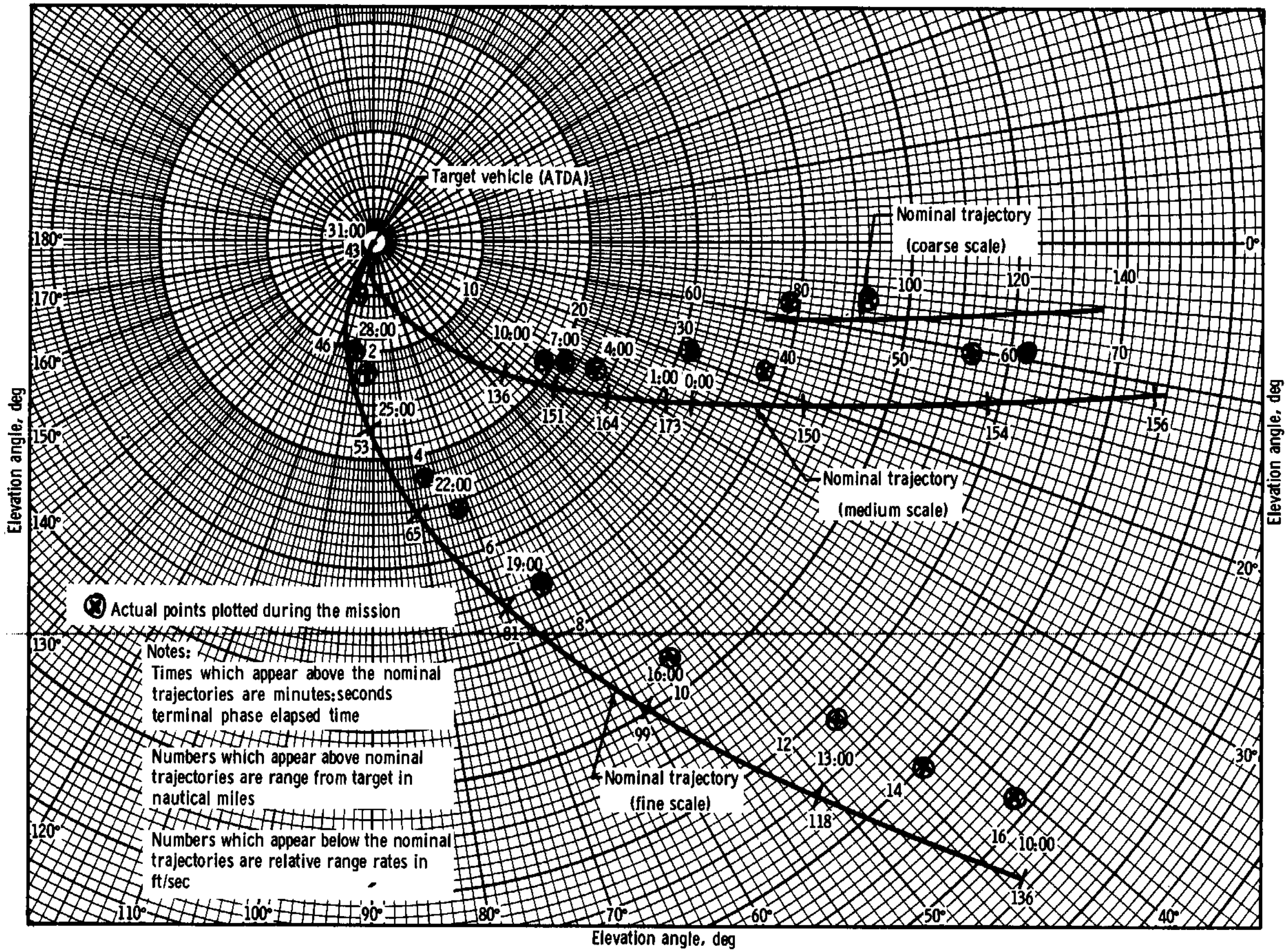

(a) M=3 rendezvous.

Figure 7.1.2-1. - Onboard target-centered coordinate plot of rendezvous.

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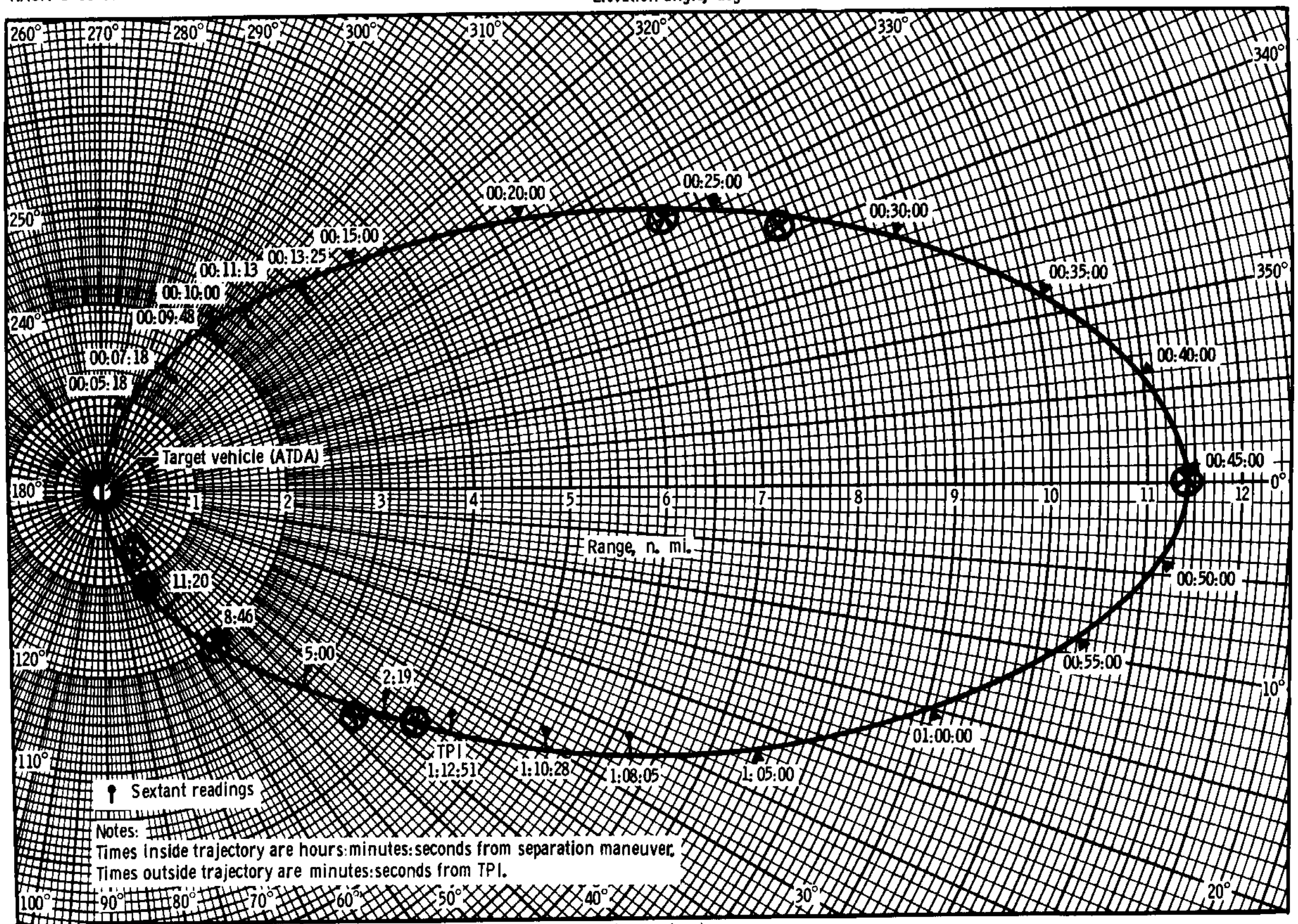
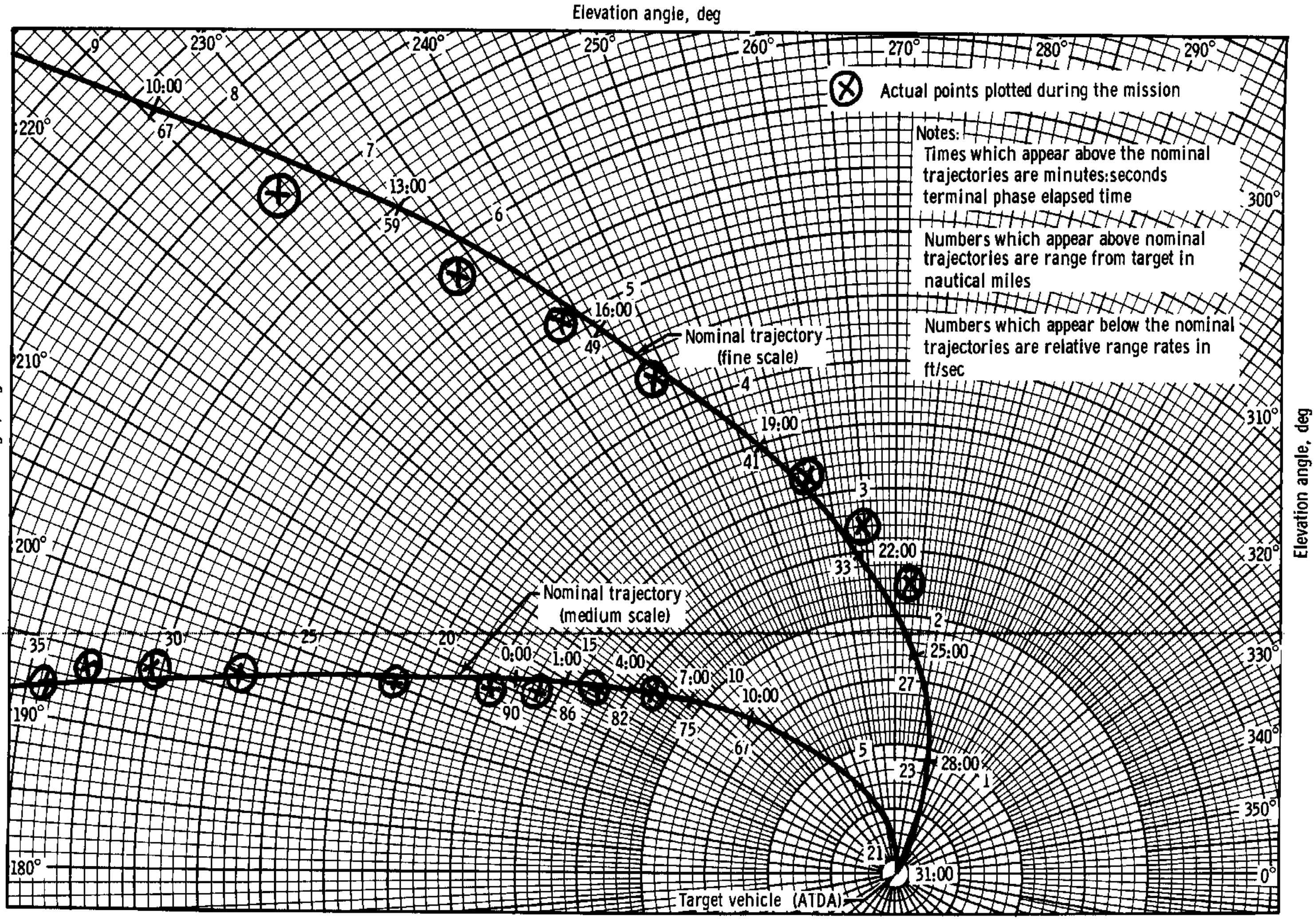
NASA-S-66-6981 JUN



(b) Equi-period rendezvous.

Figure 7.1.2-1. - Continued.

NASA-S-66-6979 JUN



(c) Rendezvous from above.

Figure 7.1.2-1. - Concluded.

7.2 AEROMEDICAL

Gemini IX-A was a three-day mission which included multiple rendezvous operations with the Augmented Target Docking Adapter (ATDA) and one extended period of extravehicular activity (EVA). Although this mission was planned as an operational mission to prove rendezvous techniques and to perform useful work during the extravehicular activities, physiological constraints caused some degree of real-time alteration in the mission. This portion of the report will present a medical analysis of these physiological constraints from the limited data which are available. The genesis of these physiological constraints will be explained and their impact on future space missions will be assessed in the following sections.

7.2.1 Preflight

7.2.1.1 Medical records review.- A complete review of the medical records of the crewmen was accomplished in February 1966. Following an aircraft accident which took the lives of the prime crew in February 1966, another review of the medical records was accomplished. The crewmen were tested for sensitivity to all medications in the inflight medical kit after they moved into crew quarters at Cape Kennedy in early May 1966. Because of the intensive training schedule established for the prime crew, all baseline medical-data collection procedures which were not considered essential for this flight were deleted from the scheduled preflight medical activities.

7.2.1.2 Health, fitness, and diet.- Neither of the crewmembers developed signs or symptoms of significant illness during the last 30 days prior to launch. During this period, they were cautious about casual exposure to large groups. The prime crew conducted vigorous exercises in the gymnasium daily in order to achieve a high level of physical fitness. A low-residue diet was prescribed in conjunction with a preflight laxative to reduce the probability that either crewmen would need to defecate during the three-day mission. The crew began the low-residue diet on May 14, 1966, in preparation for the attempted Gemini IX launch on May 17, 1966. After the Gemini IX mission was cancelled, they returned to a general diet and resumed the low-residue diet on May 28, 1966, in preparation for the scheduled Gemini IX-A launch of June 1, 1966. Following this attempt to launch, the crew remained on the low-residue diet until the launch of Gemini IX-A on June 3, 1966. Drug testing revealed that both prime crewmen experienced an undesirable reaction following ingestion of the laxative bisacodyl. An alternate preparation containing a combination of 50 mg danthron and 60 mg bis-2-ethylhexyl calcium sulfosuccinate (dioctyl calcium sulfosuccinate) was selected and tested. This preparation was found to be satisfactory.

7.2.1.3 Medical examinations.- The crewmembers were examined by an internist and the crew flight surgeons on May 6, 1966. The remainder of the medical specialty team, consisting of a neuropsychiatrist, an ophthalmologist, and an otorhinolaryngologist, conducted their examination of the crew on May 14, 1966. The crew flight surgeons conducted a preflight examination of the crew on May 17, 1966. Following the cancellation of the Gemini IX mission, re-examination of the crew was conducted by the crew flight surgeons on May 28, 1966. Prelaunch medical examinations were again conducted on the mornings of June 1 and June 3, 1966. Both crewmembers were found in excellent health by their examiners on each of these occasions.

7.2.1.4 Special baseline measurements.- Although this was basically an operational flight, one medical experiment (M-5, Bio-Assays of Body Fluids) was included in the mission plan. Moreover, because of available data on cardiovascular reflex changes noted in earlier Gemini flights, and the paucity of data concerning the physiological impact of extravehicular activities, the following measurements were performed on the Gemini IX-A crew.

7.2.1.4.1 Tilt studies: In keeping with the desire to limit time-consuming medical procedures to the absolute minimum necessary for mission support, only two preflight tilt tests were conducted on the pilot rather than the customary three. These tilt tests are depicted in figure 7.2-1. Because the data on the command pilot obtained prior to his participation in the Gemini VI-A mission were thought to be valid as baseline tilt measurements, only one tilt test was performed on the command pilot.

7.2.1.4.2 Bicycle ergometry: The flight plan included an extended period of extravehicular activities with a definite series of tasks programed for the extravehicular pilot. It was considered appropriate to measure his exercise capacity, both preflight and postflight, to gain data on the effect of the weightless mission and extravehicular activities on the pilot's measured capacity to take oxygen from the inspired air while performing work. The pilot accordingly accomplished a preflight ergometry test on May 6, 1966. Figure 7.2-2 shows the results of this test.

7.2.1.4.3 Laboratory studies: The protocol for the M-5 medical experiment required two 48-hour preflight periods of complete urine collection and the collection of two preflight blood samples for biochemical analysis. These were accomplished in conjunction with the scheduled preflight medical examinations. Following the Gemini IX cancellation on May 17, 1966, the M-5 experimenter requested an additional

24-hour urine collection period from each crewmember in order to resolve small discrepancies noted in comparing the results of the first two preflight urine samples. The crew supported this request, and the third urine specimen was obtained and shipped to the experimenter. Results of the examination of blood and urine are shown in tables 7.2-I through 7.2-III.

7.2.1.5 Prelaunch medical support.- Sensoring and bioinstrumentation checkout were accomplished in accordance with the crew countdown on each of the three scheduled launch dates. Launch morning activities are listed in table 7.2-IV. Following each of the two launch postponements, a brief conference was held between the flight crew and the crew flight surgeons regarding crew activities such as diet, training, and the rescheduling of medical examinations. The crew was considered to be in a high degree of physical and psychological readiness for the flight.

7.2.2 Inflight

The inflight portion of the aeromedical report includes events from lift-off to spacecraft landing, an elapsed time of 72 hours 21 minutes.

7.2.2.1 Physiological monitoring.- Physiological data and certain environmental parameters were monitored by physicians at the Mission Control Center in Houston (MCC-H) and at the remote network tracking sites. The electrocardiograms and pneumograms were relayed to MCC-H over voice data lines, either during the pass over the station or immediately after the pass. The quality of these data, when recorded at MCC-H, was satisfactory for clinical analysis. During the extravehicular activities, electrocardiograms from one set of sensors and the pneumogram tracings were recorded on the onboard biomedical tape recorder and also transmitted to the network tracking site in real time. These data were interrupted, however, for the period between 50 hours 44 minutes ground elapsed time (g.e.t.) and 51 hours 15 minutes g.e.t. when the pilot was disconnected from the spacecraft electrical umbilical. It was planned to transmit these data from the Astronaut Maneuvering Unit (AMU) to a receiver in the spacecraft which would, in turn, record the physiological data on the spacecraft tape recorder for postmission evaluation. Due to a failure in the spacecraft PCM recorder, physiological data for this period of time are not available.

7.2.2.1.1 Electrocardiograms: The rates and patterns of the electrocardiogram of each crewman remained within normal limits. During the flight, detailed analyses of the electrocardiograms for rates, patterns, and intervals were made during each pass by the remote-site physician and/or the physicians at MCC-H. The rates were transformed into graphs by the Aeromedical Staff Support Room personnel at MCC-H and further analyzed for trends or significant findings. Figure 7.2-3 shows the rates received at each station during the pass. The average, high, and low rates during the various station passes are shown in the figure. Figure 7.2-4 shows data for EVA obtained from real-time records and the biomedical tape recorder. The suit-inlet-temperature measurement was deleted through the EVA umbilical to provide wires for ELSS power; therefore, it was not available for real-time flight monitoring or postmission evaluation. Fortunately, the pilot participated in ergometry tests before and after this flight. Figure 7.2-5 is a plot of the pilot's heart rate against Btu output during the preflight and postflight ergometry studies, as well as a postflight calibrated exercise test using a single nine-inch step. It has been found through extensive studies in respiratory physiology that the heart rate recorded during ergometry and other exercise tests may be roughly correlated with the heat load produced. Obvious inaccuracies of this system are evident; however, a surprising degree of correlation does exist for this particular type of test. Accepting the inaccuracies and taking the available information, it can be estimated that the average heat load produced by the pilot was approximately 2200 Btu/hr, with a peak heat production of approximately 3500 Btu/hr during the hatch-closing procedure.

7.2.2.1.2 Respiration: Respiratory rates, as measured by the impedance pneumogram, were within the normal and expected range except during EVA. During hatch closure, the rates peaked to 42 breaths per minute. These rates are shown in figures 7.2-3 and 7.2-4. It is assumed that the increased respiratory rate during extravehicular activities is a normal physiological response to the increased workload; however, such an increase in respiratory rates may also be seen with simple hyperventilation or with an increase in carbon dioxide partial pressure. The increased respiratory rates may have contributed to the pilot's fluid loss, may have contributed a significant amount of water vapor to the EVA suit circuit, and possibly interferred somewhat with the exchange of gases in the faceplate area.

7.2.2.1.3 Oral temperature: An oral-temperature probe was attached to the ear piece of the lightweight headset. Oral temperatures were measured on each crewman twice during each flight day. These readings are included on figure 7.2-3.

7.2.2.2 Medical observations.-

7.2.2.2.1 Lift-off and powered flight: During powered flight, the crew experienced difficulty reading their instruments due to the position of the sun. The command pilot found that he had to shade his eyes with his right hand in order to read the instruments during the period from lift-off + 60 seconds to lift-off + 170 seconds. There were no unusual sensations described during powered flight or upon transition into the weightless state.

7.2.2.2.2 Environment: An evaluation of the Environmental Control System is found in section 5.1.4. It is interesting to note that the suit-inlet temperature measurement was approximately 10 degrees higher than the temperature taken in the suit heat exchanger. Unfortunately, suit-inlet temperature was not available during extravehicular activities. During the final night of this flight, suit and cabin temperatures were observed to rise to the high 80's. It was assumed that the crew had effectively closed the suit flow at this time and that these temperature readings, while technically correct, did not indicate any degree of crew discomfort. This was confirmed during the postflight debriefing. On the morning before retrofire, cabin pressure was reported at 4.8 psid. During the briefing, the crew reported that the pressure had reached 4.7 psid, and corrective action had been taken by the crew. They activated the repressurization valve and then observed the pressure decay again to 4.7 within one-half hour. Again they activated the repressurization valve and closed the water-seal valve. Thirty minutes later, the crew opened the water-seal valve. They later reported that the cabin pressure was holding at 5.0 psid.

The Extravehicular Life Support System (ELSS) is discussed in section 5.1.4. In the absence of telemetered environmental parameters, evaluation of the extravehicular suit environment was not possible.

7.2.2.2.3 Food, water, and sleep: Three meals of Gemini flight food per crewman per day were stored aboard the spacecraft. The crew found that they were unable to eat the programed food in sequence due to the extremely busy flight plan and were unable to report or log the food which was eaten by each crewman. It was therefore not possible for the aeromedical flight controllers to follow the caloric intake of either crewman during this flight, nor was it possible to reconstruct a reasonable assessment of the nutritional state of either crewman during this flight. The pilot was fatigued following EVA, but whether this could have been partly the result of insufficient caloric intake cannot be determined.

Prior to the flight, it was planned that each crewman would drink the same amount of water, log the amount, and report it to the ground when required. Total water-gun counts were reported as well as an assessment of the percentage consumed by the pilot or command pilot. Using this method, it was estimated that the pilot drank 4.4 pounds of water per day, and the command pilot drank 3.5 pounds per day. This was considered to be a reasonable real-time assessment, and, while normal methods of encouraging the crew to drink more water were advocated, the aeromedical flight controllers were not too concerned with the crew's water intake during the mission. After the mission, it was found that there had been an unreported failure of the crew's water supply system. This system was briefly intermittent on the second day and failed completely just prior to retrofire, when less than the programed amount of water was dispensed by the gun each time it was activated and finally no water was available. An engineering assessment of this failure is found in section 5.1.10. This information causes doubt as to the validity of the figures on water intake during the flight.

The additional postflight information that the pilot had lost over 13 pounds of body weight during this flight made it advisable to completely disregard the figures on water intake. This also tended to make the real-time evaluation of the pilot's physical condition before and after extravehicular activities somewhat less than valid.

After the stimulation of launch and the two successful rendezvous operations, the crew had difficulty settling down for the first night's sleep. They estimated six and one-half hours in naps. The second day, after a difficult rendezvous from above, the crew considered it inadvisable to proceed with the planned extravehicular activities because of crew fatigue. The ground controllers concurred with this decision, and the crew took a two-hour nap. This decision required mature judgment on the part of the command pilot and emphasized the fact that the ground controllers must rely heavily upon the crew's real-time evaluation of their physical state regarding fatigue. Further sleep periods are seen in figure 7.2-4.

7.2.2.2.4 Medications: On the second and third night, the command pilot took two APC tablets in an attempt to induce sleep, and, 30 minutes prior to retrofire, he took one Actifed tablet in an attempt to prevent ear blocks during reentry. During the postflight debriefing, the command pilot stated that during the first day he felt "billious". He attributed this to the change in diet and reported that he had decided to stay on liquids for most of the flight. The pilot felt some uneasy bowel symptoms during the flight and took two lomotil tablets in a further effort to prevent defecation.

7.2.2.2.5 Waste: Neither crewman found it necessary to defecate during this flight. The urine collection system was used repeatedly and performed normally. However, the sampling system which was required for the M-5 experiment was found to be time consuming and to cause rather distressing storage problems. The crew also reported some noticeable reverse pressure during urination. This was not uncomfortable, but may have interferred with complete bladder emptying.

7.2.2.2.5 Vision: During this flight, vision was again somewhat degraded by a thin greyish-black film on the outside of both windows as well as a silverish-grey film which developed on the inside of the outer pane and progressively increased throughout the flight. Even with this degree of degradation, several salient points of functional vision were clearly demonstrated. The Augmented Target Docking Adapter (ATDA) was observed in reflected sunlight at approximately 50 miles during the first rendezvous. Its brightness was between a fifth and sixth magnitude star. The flashing acquisition lights were visible at approximately 20 miles in the darkness on the first rendezvous. At eight miles, the crew could ascertain color, with the red running lights most easily visible, then the amber, and finally the green. At about one mile, they could actually discern that the shroud had not separated from the ATDA. During the rendezvous from above, when the target was seen at approximately 20 miles as an object approximately four to five times as bright as Venus, the brightness diminished with the approaching spacecraft day and disappeared as the ocean and the African coast came into the background. It was then impossible to see the target as there was no reflected light and very little difference in background contrast until the spacecraft approached within approximately 3 nautical miles of the target.

During extravehicular activities, the pilot reported no unusual visual phenomenon. During the first dayside pass, the pilot removed his extravehicular sunshade visor momentarily. The pilot reported no visual discomfort during the brief period during which the EVA visor was removed. The EVA visor was also removed during the night pass and, after fogging had occurred, the visor was left in the up (removed) position until approximately 20 minutes after sunrise on the second dayside pass.

7.2.2.2.6 Orientation: There were no orientation problems during the flight. The pilot reported that during extravehicular activities, although he had considerable difficulty positioning himself in order to do simple tasks and, although his visual references were somewhat compromised by faceplate fogging, there was no question as to his orientation at any time. After moving into the adapter section, the pilot's activities caused considerable motion of the spacecraft; however, the pilot retained orientation with respect to the spacecraft and was able to ignore the absolute motion.

7.2.2.2.7 Retrofire and reentry: Retrofire and reentry were normal. The sensations during deceleration were essentially the same as those reported by previous flight crews. The landing was considered to be harder than normal but well within the physiological limits of the individuals.

7.2.3 Postflight

This portion of the report includes aeromedical observations from the time of spacecraft landing at approximately 9:00 a.m. e.s.t. until after the final medical examinations were performed at Cape Kennedy. These data were obtained from clinical examinations, medical debriefings, and numerous laboratory determinations. Variations from normal include the following:

(a) Weight loss

(b) Hemoconcentration

(c) Mild transit reduction in pulse pressure and elevation in heart rate during the only postflight tilt study as compared with the preflight tilt

(d) Vesiciculation and subcutaneous hematoma formation under the sensor sites on the pilot

(e) Abrasion of the skin overlying the right and left proximal thumb joint.

7.2.3.1 Planned recovery procedures.- At the time of recovery, the time of crew departure from the carrier remained undetermined. Actual departure time was seven hours after spacecraft landing. The postflight medical evaluation was scheduled to be less detailed than those which had followed the previous Gemini flights. Routine tilt studies were scheduled the same as for previous missions, twice on recovery day and daily thereafter until the crewmembers' responses returned to preflight values. Laboratory procedures were planned to be limited to routine chest roentgenograms, complete blood counts, and urinalysis. Blood and urine specimens were to be collected for the M-5 experiment. Postflight medical examinations were also to be less comprehensive than those performed following previous flights, with special emphasis on the cardiovascular system. Therefore, only the internist-cardiologist member of the medical evaluation team was deployed to the prime recovery ship. Examinations of additional systems were performed as indicated by the NASA physician and/or the Department of Defense (DOD) members of the recovery medical team.

7.2.3.2 Recovery activities.- After spacecraft landing, the crew remained in the spacecraft until it was hoisted to the deck of the carrier. The crew egressed from the spacecraft without difficulty and walked with a normal gait. They were obviously in excellent spirits and gave no indications of ill effects from their space flight. The crew reported no effects of orthostatic hypotension either on the water or after egress from the spacecraft.

7.2.3.2.1 Examinations: A postflight medical examination, as programed, was completed within two hours after spacecraft landing. During the desuiting process, it was noted that the undergarments of both crewmen were completely saturated with perspiration. It was also noted that the pressure gage on the sleeve of the pilot's suit had visible fogging and water droplets on the glass covering. At the sites of the pilot's axillary electrocardiogram sensors, there was vesiciculation of the skin within the stomaseal ring under the sensor paste. In conjunction with this finding was an area of resolving hematoma, again under the area of the sensor paste. The vesiciculation had disappeared 24 hours later, but there was still a small area of hematoma at 24 hours after spacecraft landing. The skin of both astronauts, other than that described above, was in excellent condition and showed no signs of masceration, desclamation, or erythema. The internist report revealed no other changes with the exception of mild dehydration as manifested by weight loss. During this flight, the command pilot lost 5.2 pounds and the pilot lost 13.6 pounds. These weights were determined by subtracting the weights measured shortly after boarding the recovery ship from the weights measured during the preflight physical examinations. Allowances were made for the amount of water or fluids taken after recovery and prior to the postflight determination. Some inaccuracies in these weights may be expected due to the difference in scales used during the preflight and postflight determinations; however, the degree of accuracy is sufficient to determine that there was considerable weight loss by the pilot.

7.2.3.2.2 Tilt-table studies: One postflight tilt-table study was performed on each crewmember. These showed only minimal effects compared to those which have been noted on some previous short-duration flights. Both men tolerated the procedure well. Figures 7.2-1 and 7.2-2 present the data obtained during these studies.

7.2.3.3 Bicycle ergometer studies.- A bicycle ergometer test was performed by the pilot on the morning following spacecraft landing. The results of these studies are shown in figure 7.2-3. The pilot was found to have some degree of degradation in his ventilation, his oxygen uptake, and his endurance. The test was terminated due to pilot fatigue.

TABLE 7.2-I.- URINALYSIS

(a) Command Pilot

Determination	Preflight	Postflight	
	May 28, 1966	June 6, 1966	
Time (local)	08:00	12:15	19:15
Volume, cc	325	Unable to void	465
Color, appearance	Yellow, clear		Amber, clear
Reaction	pH 5		pH 6
Specific gravity	1.024		1.027
Albumin	Negative		Negative
Sugar	Negative		Negative
Microscopic	10-15 WBC, few bacteria and epithelial cells		Rare WBC, no RBC, no bacteria

(b) Pilot

Determination	Preflight	Postflight	
	May 28, 1966	June 6, 1966	
Time (local)	08:00	11:15	19:30
Volume, cc	375	Unable to void	395
Color, appearance	Yellow, clear		Light amber, heavy concentration
Reaction	pH 7		pH 5
Specific gravity	1.032		1.025
Albumin	Negative		Negative
Sugar	Negative		Negative
Microscopic	0-1 WBC		No RBC, WBC, bacteria. Heavy amorphous urate crystals.

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TABLE 7.2-II.- HEMATOLOGY

(a) Command Pilot

Determinations Date, 1966 Time, e.s.t.	May 11 07:30	May 14	May 28 08:00	June 6 12:15	June 6	June 9
WBC	-	-	7800	9700	7125	-
Neutrophiles, percent	-	-	60	60	59	-
Lymphocytes, percent	-	-	37	36	35	-
Monocytes, percent	-	-	3	-	4	-
Hematocrit, percent	-	-	43	-	44	-
Hemoglobin, gm	-	-	14.2	-	-	-
Eosinophiles, percent	-	-	-	2	1	-
Bands	-	-	-	2	1	-
RBC	-	-	-	-	-	-
Corrected sedimentation rate, mm/hr	-	-	-	-	-	-
Morphology	-	-	Normal	-	Normal	-
Sodium, mEq/l	139	141	-	143	148	144
Potassium, mEq/l	4.2	4.0	-	4.0	3.7	4.6
Chloride, mEq/l	103	102	-	100	103	100
Calcium, mg percent	9.5	9.9	-	10.0	9.6	9.3
Phosphate, mg percent	3.8	4.2	-	4.8	4.6	4.3
Creatinine, mg percent	1.5	1.5	-	1.8	1.2	1.3

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TABLE 7.2-II.- HEMATOLOGY - Concluded

(b) Pilot

Determinations Date, 1966 Time, e.s.t.	May 11 07:30	May 14	May 28 08:00	June 6 11:15	June 6 19:30	June 9
WBC	-	-	8600	11 150	9560	-
Neutrophiles, percent	-	-	54	75	54	-
Lymphocytes, percent	-	-	40	21	36	-
Monocytes, percent	-	-	6	-	7	-
Hematocrit, percent	-	-	44	51	46	-
Hemoglobin, gm	-	-	14.8	16.5	-	-
Eosinophiles, percent	-	-	-	-	3	-
Bands	-	-	-	4	-	-
RBC	-	-	-	5 050 000	-	-
Corrected sedimentation rate, mm/hr.	-	-	-	3	-	-
Morphology	-	-	Normal	RBC normal	Normal	-
Sodium, mEq/l	131	118	-	153	147	155
Potassium, mEq/l	4.3	3.8	-	3.6	3.6	4.5
Chloride, mEq/l	95	87	-	98	102	108
Calcium, mg percent	10.2	9.8	-	10.9	9.0	7.4
Phosphate, mg percent	4.7	4.6	-	3.4	4.3	3.2
Creatinine, mg percent	1.5	1.1	-	2.0	1.4	0.9

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TABLE 7.2-III. - URINE CHEMISTRIES

(a) Command Pilot

Determinations Date, 1966 Time, e.s.t.	May 9 10:15	May 9 13:15	May 9 15:40	May 9 17:45	May 9 21:30	May 10 07:15	May 10 21:30	May 11 07:00	May 12 11:30	May 12 14:10
Total volume, ml	240	513	160	195	490	225	200	290	275	230
Osmolality, mOs/kg . . .	152	177	108	96	166	183	194	303	142	116
Sodium, mEq/vol	22.6	29.7	16.6	17.9	39.2	15.7	16.8	27.8	17.6	16.1
Potassium, mEq/vol . . .	15.4	28.7	11.2	5.1	9.8	7.6	20.4	14.5	12.6	19.8
Chloride, mEq/vol	25.6	37.7	11.7	12.8	28.7	9.7	9.5	17.4	21.4	18.2
Calcium, mg/vol	26.2	51.8	20.0	25.5	42.6	27.7	23.8	34.5	31.4	23.7
Phosphate, g/vol	90	47	116	83	147	264	385	210	158	155
Creatinine, g/vol	368	238	240	185	258	581	461	687	325	244
17 hydroxycorticosteroids, mg/vol. . .	1.19	1.15	0.85	0.51	0.66	0.60	0.54	0.70	0.92	0.81

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TABLE 7.2-III.- URINE CHEMISTRIES - Continued

(a) Command Pilot

Determinations Date, 1966 Time, e.s.t.	May 12 17:45	May 12 19:30	May 12 23:10	May 13 04:00	May 13 08:15	May 13 12:50	May 13 16:00	May 13 22:00	May 14 07:00	May 27 14:30
Total volume, ml	490	60	240	435	340	475	345	480	205	470
Osmolality, mOs/kg . . .	191	QNS	177	150	164	136	143	126	148	222
Sodium, mEq/vol	51.0	QNS	31.7	25.2	25.8	25.6	37.3	16.3	17.6	56.4
Potassium, mEq/vol . . .	25.5	QNS	12.5	10.4	6.8	19.0	17.20	0.77	6.60	29.1
Chloride, mEq/vol	48.4	QNS	23.0	16.1	19.1	32.1	29.2	8.0	10.5	63.6
Calcium, mg/vol	43.6	QNS	28.6	32.6	14.6	46.6	41.1	41.8	21.3	47.5
Phosphate, g/vol	208	QNS	246	218	136	74	173	228	205	106
Creatinine, g/vol	276	QNS	378	291	394	229	223	301	429	282
17 hydroxycorti- costeroids, mg/vol. . .	0.88	QNS	0.65	0.30	0.72	1.12	0.65	0.52	0.38	0.70

QNS - quantity not sufficient.

TABLE 7.2-III.- URINE CHEMISTRIES - Continued

(a) Command Pilot

Determinations Date, 1966 Time, e.s.t.	May 27 18:45	May 27 23:00	May 28 08:00	June 7 09:10	June 7 15:30	June 7 23:00	June 8 07:30	June 8 08:30
Total volume, ml	350	250	280	465	-	170	395	255
Osmolality, mOs/kg	226	166	221	342	629	165	256	107
Sodium, mEq/vol	54.6	30.0	25.8	35.3	46	22.4	27.6	12.7
Potassium, mEq/vol	15.4	8.5	12.3	22.3	44	10.5	11.8	7.1
Chloride, mEq/vol	47.2	24.5	25.0	15.1	50.8	6.7	13.9	12.1
Calcium, mg/vol	52.5	35.8	32.2	67.0	19.0	54.9	51.0	20.0
Phosphate, g/vol	219	231	182	616	72.5	221	286	68
Creatinine, g/vol	408	353	611	944	250	393	601	225
17 hydroxycorti-costeroids, mg/vol.	1.09	0.58	0.99	1.86	56.5	0.62	1.23	0.64

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TABLE 7.2-III.- URINE CHEMISTRIES - Continued

(b) Pilot

Determinations Date, 1966 Time, e.s.t.	May 9 06:45- 10:25	May 9 14:30	May 9 17:15	May 9 19:00	May 9 21:30	May 10 07:10	May 10 14:00	May 10 18:00
Total volume, ml	410	375	310	295	195	450	90	305
Osmolality, mOs/kg	210	197	142	95	107	311	59	146
Sodium, mEq/vol	58.2	48.7	35.3	23.6	23.8	53.1	8.1	17.1
Potassium, mEq/vol	23.8	21.0	16.1	7.7	7.4	18.0	3.4	15.2
Chloride, mEq/vol	63.6	49.8	22.4	16.1	18.4	43.0	6.4	15.5
Calcium, mg/vol	25.8	27.4	14.9	13.9	19.1	42.3	9.7	16.5
Phosphate, g/vol	133	188	194	111	141	439	59	595
Creatinine, g/vol	257	317	296	161	222	695	134	308
17 hydroxy- corticosteroids, mg/vol . .	1.19	1.42	0.81	0.35	0.36	1.15	0.38	1.01

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TABLE 7.2-III.- URINE CHEMISTRIES - Continued

(b) Pilot

Determinations Date, 1966 Time, e.s.t.	May 10 21:00	May 11 08:00	May 12 08:00- 12:30	May 12 17:00	May 12 23:30	May 13 08:30	May 13 13:00	May 13 15:30
Total volume, ml	135	505	115	300	225	300	355	480
Osmolality, mOs/kg	114	301	66	182	184	266	251	115
Sodium, mEq/vol	16.2	38.4	9.9	33.6	31.5	40.8	54.0	28.8
Potassium, mEq/vol	10.3	24.2	10.6	31.8	19.3	25.8	34.8	11.5
Chloride, mEq/vol	10.5	30.4	13.8	36.6	25.8	36.6	53.6	19.1
Calcium, mg/vol	14.3	45.5	10.9	16.8	20.9	26.7	31.2	56.6
Phosphate, g/vol	182	391	52	390	203	405	222	276
Creatinine, g/vol	324	570	167	413	548	694	472	160
17 hydroxy- corticosteroids, mg/vol . .	0.49	1.66	0.57	1.30	1.22	1.36	2.02	0.54

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TABLE 7.2-III.- URINE CHEMISTRIES - Concluded

(b) Pilot

Determinations Date, 1966 Time, e.s.t.	May 13 22:00	May 14 07:00	May 27 11:00	May 27 14:00	May 27 18:30	May 27 22:30	May 28 08:00	June 6 21:30	June 7 07:30
Total volume, ml	335	475	160	460	305	210	375	325	395
Osmolality, mOs/kg	251	249	112	197	224	176	343	309	349
Sodium, mEq/vol	41.5	35.1	28.2	39.6	52.5	37.0	47.2	13.0	8.7
Potassium, mEq/vol	17.4	13.3	17.0	30.4	22.6	18.5	45.7	23.4	22.1
Chloride, mEq/vol	22.4	27.3	31.4	46.5	47.4	34.3	39.6	5.1	2.4
Calcium, mg/vol	35.2	37.5	11.7	18.9	27.8	18.7	27.8	34.1	29.2
Phosphate, g/vol	343	297	88	184	252	231	497	463	573
Creatinine, g/vol	539	551	146	274	440	326	761	823	864
17 hydroxy- corticosteroids, mg/vol . . .	1.03	1.37	0.32	1.15	1.24	.097	0.29	2.28	1.99

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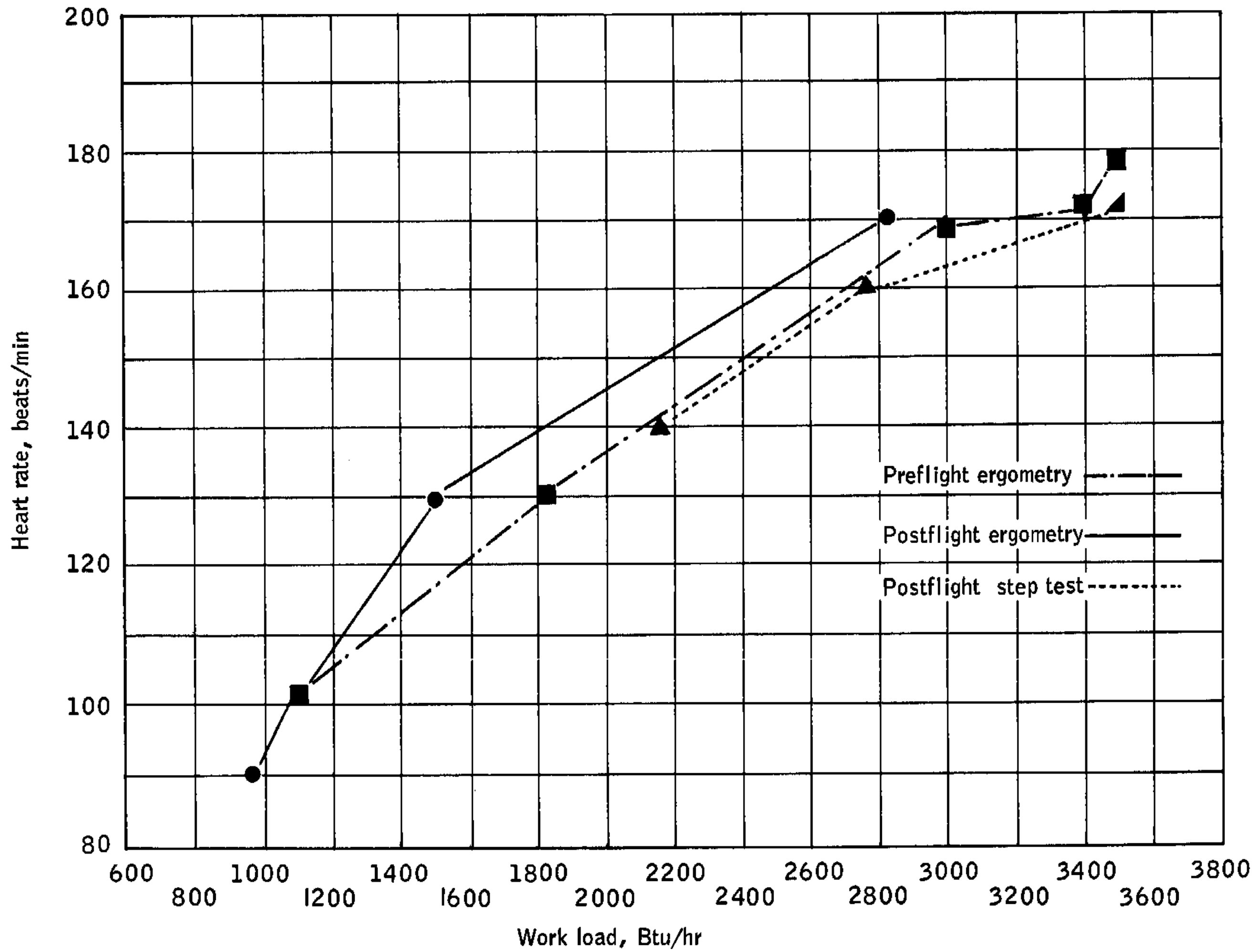

Figure 7.2-5. - Exercise studies on the Gemini IX-A pilot.

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8.0 EXPERIMENTS

Seven scientific, medical, or technological experiments were planned for the Gemini IX-A mission. The experiment number, title, principal investigator, sponsoring agency, and qualitative success are listed in table 8.0-I. The schedule of events during the mission is shown in table 8.0-II. This plan has been corrected to exact times as recorded on the onboard voice tapes and/or in the log recorded by the crew during the flight.

The Agena Micrometeorite Collection (S-10) and the Astronaut Maneuvering Unit (D-12) experiments were not completed because extravehicular activities were not conducted in the vicinity of the target vehicle. They are presently scheduled for Gemini missions X or XII. During the mission, several changes were made to the flight plan to schedule as much time as possible for experiments. Preliminary analysis of data indicates that four of the seven basic experiment objectives were obtained with the revised flight plan.

Each experiment scheduled for the Gemini IX-A mission is described in the sections that follow, and success or failure is so indicated. In these reports the principal investigators obtaining useful data have indicated only the quality of information received. Detailed analysis of photography and other prime data will require several months to determine useful results and reach definitive conclusions. Specific scientific, medical, or technological reports will be published later and will contain conclusions from the experiments.

TABLE 8.0-I.- EXPERIMENTS

Experiment number	Experiment title	Principal experimenter	Sponsor	Data results	Percent completion
D-12	Astronaut Maneuvering Unit	Deputy for Technology Headquarters, Air Force Space Systems Division, Los Angeles, California	Department of Defense	Cancelled after donning	10
D-14	UHF/VHF Polarization	U.S. Naval Research Laboratory, Washington, D.C.	Department of Defense	6 periods of useful data	100
M-5	Bio-Assays of Body Fluids	Space Medicine Branch, Crew Systems Division, NASA-MSC, Houston, Texas	NASA Office of Manned Space Flight	Poor	100
S-1	Zodiacal Light Photography	School of Physics, Institute of Technology, University of Minnesota, Minneapolis, Minnesota	Office of Space Sciences	Excellent (17 photographs)	100
S-10	Agena Micrometeorite Collection	Dudley University, Albany, New York	Office of Space Sciences	None	0
S-11	Airglow Horizon Photography	U.S. Naval Research Laboratory, Washington, D.C.	Office of Space Sciences	Excellent (44 photographs)	100
S-12	Micrometeorite Collection	Dudley University, Albany, New York	Office of Space Sciences	Useful data were obtained	100

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TABLE 8.0-II.- FINAL EXPERIMENT FLIGHT PLAN FOR GEMINI IX-A

Date, 1966	Experiment	Condition	Gemini ground elapsed time, hr:min	Revolution	Remarks
June 3	S-12	Start	9:29	6	Open collector door: pilot
		Finish	17:10	11	Close collector door: pilot
June 4	S-12	Start	35:48	24	Open collector door: pilot
		Finish	44:54	30	Close collector door: pilot
June 4	D-14	Start Finish	26:52 27:05	17	HAW test
June 4	S-11	Start Finish	29:12 30:00	19	CRO to CNV area
June 4	D-14	Start Finish	27:16 27:25	18	ANT test
June 4	D-14	Start Finish	28:28 28:41	18	HAW test
June 4	D-14	Start Finish	30:02 30:17	19	HAW test
June 4	S-11	Start Finish	30:21 31:25	20	CRO and CTN areas
June 4	D-14	Start Finish	31:42 31:57	20	HAW test
June 4	S-11	Start Finish	32:22 33:00	21	CRO and CTN areas

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TABLE 8.0-II.- FINAL EXPERIMENT FLIGHT PLAN FOR GEMINI IX-A - Continued

Date, 1966	Experiment	Condition	Gemini ground elapsed time, hr:min	Revolution	Remarks
June 4	D-14	Start Finish	33:15 33:28	21	HAW test
June 5	S-12	EVA	49:27	32	Retrieve S-12: pilot
June 5	S-10	Cancelled		14	Activate S-10: pilot
June 5	D-12	Cancelled	51:04	32	TM ON. Don AMU
June 5	D-12	Cancelled		32	AMU OFF
June 5	D-12	No data[a]		32	First tape dump after EVA. AMU off over TEX
June 5	S-11, D-14			32	Flight-plan update to crew. (over CYI)
June 5	S-10	Cancelled		15	Retrieve S-10 and attach to ELSS
June 5	S-1	Via space-craft window	54:37	35	Take photos: pilot (Pilot report to command pilot each exposure time)

[a]Tape recorder failed.

TABLE 8.0-II.- FINAL EXPERIMENT FLIGHT PLAN FOR GEMINI IX-A - Concluded

Date, 1966	Experiment	Condition	Gemini ground elapsed time, hr:min	Revolution	Remarks
June 5	S-1		55:15	35	S-1 completed
June 5	D-14	Cancelled[b]		35	ANT test
June 5	S-11	Cancelled[c]		36	TAN, CSQ area
June 6	Spacecraft	Landing	72:20:55	43	Recovery in West Atlantic

[b]Pilot broke antenna.

[c]Cancelled by crew at 56:50 g.e.t.

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8.1 EXPERIMENT D-12, ASTRONAUT MANEUVERING UNIT

8.1.1 Experiment Objectives

Maintenance, repair, resupply, crew transfer, rescue, satellite inspection, and assembly of structures in space are all operations of potential space systems. Many of the concepts for accomplishing these operations involve extravehicular activity and require the capability for man to maneuver in free space and translate short distances. The Astronaut Maneuvering Unit (AMU) experiment is one approach toward determining the basic hardware and operational criteria required to integrate these extravehicular activities into manned space flight. As an experiment, the AMU was not designed against any particular operational requirement or mission; rather, it was designed to provide as much experience in extravehicular maneuvering operations as the Gemini environment would accommodate. The experimental results were intended to provide basic information to establish the extent to which extravehicular operations may or should be employed in future space systems operations.

8.1.2 Equipment Concept

The Gemini spacecraft configuration and environment were major factors in establishing the AMU packaging configuration. Because stowage volume in the Gemini cabin is not adequate for the entire system and the life support system is needed for egress operations, the modular concept was virtually mandatory. The division of systems into the two basic modules shown in figure 8.1-1 was selected to make maximum utilization of the Extravehicular Life Support System (ELSS) chestpack developed by NASA for Gemini extravehicular operations. The chestpack contains the life support systems, emergency oxygen supply, and all of the AMU systems status and malfunction displays. The externally stored module, referred to as the backpack, contains the propulsion, flight control, oxygen supply, malfunction detection, and communication systems. With the Gemini pressure suit, these two modules comprise the AMU, a system that is essentially a miniature manned spacecraft.

8.1.3 AMU Backpack Subsystems

The backpack is a highly compact unit consisting of a basic structure and six major systems. These are the propulsion, flight control, oxygen supply, power supply, malfunction detection, and communications systems. The external features of the pack are shown in figure 8.1-2 and the internal equipment arrangement is shown in figure 8.1-3.

The structure consists of a backpack shell, two folding sidearm controllers, and folding nozzle extensions. The shell is a box-like structure consisting of three main beams and supporting shelves on which the components are mounted. The thrusters are located in the corners of the structure to provide controlling forces and moments about the center of gravity of the entire AMU. The remainder of the components are located in the available spaces inside the pack. The total volume and shape were determined somewhat by the stowage location in the Gemini spacecraft adapter equipment section. It was this constraint which required the folding features of the nozzle extensions and sidearm flight controllers. The sidearm controllers contain the controller heads by which the pilot commands translation and attitude. This allows the control handles to be in a readily accessible position for use with the pressure suit. The nozzle extensions position the exhaust plume from the hydrogen peroxide thrusters away from the helmet and shoulders of the suit. A weight breakdown by system is shown in table 8.1-I.

8.1.3.1 Propulsion system.- The propulsion system is a conventional monopropellant system which uses 90 percent hydrogen peroxide as the propellant and provides a total impulse of 3000 to 3500 lb-sec. The pressurant gas is nitrogen. The major components of this system are shown in figure 8.1-4. Figure 8.1-5 is a functional schematic. The nitrogen tank provides high-pressure gaseous nitrogen to a regulator, which in turn supplies regulated nitrogen at a nominal pressure of 455 psi to a bladder in the hydrogen peroxide tank. The selection of materials for this bladder represents one of the design problems encountered in the AMU program. Because of the long storage requirement – up to 10 days – a material which was practically inert in the presence of hydrogen peroxide was required. In addition, the bladder had to be capable of several fill and expulsion cycles. The material selected was Fluorel 2141 (a form of Viton), and it has proved to be highly satisfactory.

The flow of propellant to the thrust chamber assemblies is controlled through the manual valves which also control the electrical signals to the individual thruster valves. Two of these manual valves are provided, one for the primary control system and one for the alternate system. There are 12 thrust chambers and 16 control valves. As shown in figure 8.1-6, the primary control system utilizes 8 thrusters – 2 forward, 2 aft, 2 up, and 2 down. The forward-firing and aft-firing thrusters are used in various combinations for translation fore and aft, and for pitch and yaw control. The up-firing and down-firing thrusters are used for vertical translation and roll control. The alternate system uses entirely separate forward-firing and aft-firing thrusters but uses the same thrust chambers for vertical translation and roll control. However, separate control valves are used in the alternate system for the up, down, and roll commands. Relief valves are incorporated in both the nitrogen and hydrogen-peroxide lines as a safety feature. These vent into thrust-neutralizing overboard vents.

8.1.3.2 Flight control system.- The flight control system provides automatic attitude control and stabilization in three axes, and manual translation in two axes. Translation to the side can be accomplished by a 90 degree roll or yaw followed by translation in one of the modes available. The major components which make up the flight control system are shown in figure 8.1-7, and a functional block diagram is shown in figure 8.1-8. It should be noted that completely redundant systems are available. Control commands are made manually through the controller heads located on the sidearm controllers.

The left hand controls translation commands in the vertical (up and down) and horizontal (fore and aft) directions. The controller knob movement is direction oriented. That is, to translate forward, the knob is rotated forward; and, to translate up, the knob is rotated up. Also located on the left-hand controller assembly are the mode selection switch, a voice communication volume control, and a communications selector switch. The mode selection switch is used to select either automatic or manual attitude control and stabilization. The three-position communication selector switch permits the astronaut to select the most desirable mode of operation with respect to the voice-operated switches in the AMU transceiver. The right-hand controller provides control in pitch, yaw, and roll. These commands are also direction oriented; that is, to pitch down, the control knob is rotated in the down direction; and, to yaw right, the knob is rotated clockwise. When the knobs are released they return to the OFF position.

While under automatic stabilization, a fixed rate command (18 deg/sec in pitch and yaw, and 26 deg/sec in roll) is entered into the control system when the knob is rotated, and then the system goes into a hold position at the attitude at which the knob is released. The automatic mode will permit the pilot to "park" in space in a stabilized position, with the thrusters firing as required to maintain this position within a deadband of ±2.4 degrees. In the absence of the external torques, the period of limit-cycle operation within the deadband is in excess of 20 seconds about all three axes. In the manual mode, the gyros are out of the loop and a direct "fly-by-wire" system results. While in the manual mode, the thrusters will fire only on a manual command.

8.1.3.3 Oxygen supply system.- The purpose of the oxygen supply system (OSS) is to supply expendable oxygen to the chestpack ELSS at closely regulated values of temperature and pressure. The components of the OSS are shown in figure 8.1-9, and a functional schematic is shown in figure 8.1-10. A total of 7.3 pounds of gaseous oxygen is stored in the supply tank at a pressure of 7500 psi. A minimum of 5.1 pounds of oxygen can be delivered to the chestpack at a pressure of 97 ± 10 psi, and a temperature of 65° ± 10° F. Peak design flow is 8.4 lb/hr, with a normal flow of 5.0 ± 0.2 lb/hr. The delivered gas pressure can be maintained at the desired value until the tank pressure drops below 200 psi. A low-level switch illuminates a warning light on the chest pack when the tank pressure reaches 800 ± 160 psi.

8.1.3.4 Power supply system.- The power supply system supplies electrical power to the AMU systems for the planned mission duration with a 100 percent reserve capacity. The electrical power is provided by two batteries made up of silver-zinc cells enclosed in a sealed cylindrical can. This battery can is shown in figure 8.1-11. Two of the cans are mounted on the backpack to provide the required redundancy. One battery in the can provides power for the reaction control system. A block diagram of this arrangement is shown in figure 8.1-12. One set of taps on this battery provides plus and minus 16.5 volts for control-logic circuitry and a separate set of taps provides plus and minus 15 volts for the rate gyros and valve amplifiers. An entirely separate battery in the can provides 28 V dc power for the other systems. These separate batteries feed to a common bus, but are electrically isolated by diodes to prevent a short circuit in one battery from draining the other. This system is shown in a block diagram in figure 8.1-13. The batteries are installed as one of the last operations prior to mating the spacecraft adapter section to the launch vehicle because access to the backpack is not available subsequent to that time without demating. The batteries are isolated from the AMU systems by the main power switch, which the astronaut closes as part of the predonning procedure.

The power distribution unit contains mechanical fuses in all power circuits to protect the lightweight wires and cables used in the AMU electrical systems. Dual fuses are used to provide mechanical failure redundancy. That is, two fuses are used in parallel so that if one of the fuses fails open for some reason other than an overload, the other fuse will maintain the circuit integrity. The power distribution unit also includes the diodes which separate the two 28-volt systems electrically.

8.1.3.5 Malfunction detection system.- A malfunction detection system provides both crewmen with a warning when certain critical out-of-tolerance conditions exist. The critical parameters monitored are low fuel pressure, low oxygen supply, low fuel quantity, and those indicative of certain control system anomalies. A functional schematic of the system is shown in figure 8.1-14. The warning is given both as an intermittent tone in the headset and as a warning light on the chestpack display panel. Individual warning lights that identify the out-of-tolerance system are located on the upper surface of the chestpack. A manual switch is provided to permit the pilot to silence the audio tone if he so desires, but the warning light will remain on as long as the out-of-tolerance condition exists. However, if a new alarm condition occurs, the tone will come on again, and the appropriate warning light will appear on the chestpack.

8.1.3.6 Communications system.- The communications system consists of a telemetry system and a voice system. The major components of these systems, as well as the power supply and malfunction detection systems, are shown in figure 8.1-15. The telemetry system monitors certain backpack parameters and biomedical parameters and transmits the information over an RF link to the Gemini spacecraft, where it is received and stored on a tape recorder which is part of the basic spacecraft data acquisition system. The data is available for postflight analysis only. A functional schematic is shown in figure 8.1-16. Table 8.1-II is a list of the AMU telemetry parameters. The voice communications transceiver is a UHF transmitter/receiver which is controlled by redundant voice-operated switches. It is designed to be compatible with the basic spacecraft onboard communications system, and utilizes the microphone and earphones in the suit. A three-position switch mounted physically on the translation controller provides (1) continuity of the microphone leads for normal VOX mode of operation, (2) opening of the microphone leads or "listen" mode, and (3) momentary closing of the leads for transmitting. A functional schematic of this system is shown in figure 8.1-17. The signals from the telemetry transmitter and the transceiver are diplexed and radiated from a common antenna mounted on top of the backpack. While the AMU is stored in the spacecraft, certain parameters are fed to the spacecraft telemetry system and transmitted to the ground, and the pressure and temperature of the hydrogen peroxide are displayed on a panel in the spacecraft cockpit.

8.1.4 AMU Interfaces

8.1.4.1 Installation.- The AMU backpack is installed in the Gemini equipment adapter prior to mating the spacecraft to the launch vehicle. Locations of the AMU and the associated spacecraft hardware are shown in figure 8.1-18. Mechanical mating to the spacecraft is accomplished by mounting a four-legged structure or claw assembly to the backpack, and then pulling the claw down firmly against a sheet metal structure (torque-box assembly) with a tension bolt. The torque bolt is then hard-mounted to the blast-shield door. The bolt is severed by an electrically detonated, pyrotechnically operated guillotine actuated from the cockpit after the AMU has been donned. A pull-away electrical connector provides instrumentation and power leads for cabin monitoring and ground servicing and testing.

8.1.4.2 Servicing provisions.- To permit servicing of the AMU with hydrogen peroxide (H_2O_2) after mating the spacecraft to the launch vehicle, a service line is provided from the external surface of the adapter to the AMU H_2O_2 fill port. A second parallel line to the AMU regulated nitrogen port allows reservice of the system in the event an unstable condition in the H_2O_2 is detected or if the launch is delayed indefinitely. If, for some reason, the peroxide should become unstable and the pressure of the system should rise above 575 psia, the AMU relief valve would open and the H_2O_2 would be vented through a third line from the H_2O_2 vent to the adapter skin. The fill and reservice lines are severed by the same guillotine cutter which releases the AMU. The vent line is routed through a spring-loaded pull-off housing that separates when the AMU is released.

8.1.4.3 Thermal interface.- Because of temperature limitations of 40° to 100° F for various AMU components, a cover assembly is placed over the AMU to provide a means of passive thermal control. This cover rests against the ground-equipment connectors on the front of the AMU and is maintained in this position by a line which passes through the center of the AMU to a hard mounting point on the torque-box assembly. Jettisoning of the cover is accomplished by actuation of the cockpit EVA BARS EXT switch, which severs the attachment line by guillotine action.

8.1.4.4 Donning hardware.- Equipment is provided in the adapter to assist the astronaut in donning the AMU. This hardware, shown in figure 8.1-19, consists of a footrail, two handbars, an umbilical guide, and two floodlights for nightside operation. This equipment is deployed and properly positioned for AMU donning simultaneously with release of the thermal cover.

8.1.4.5 Instrumentation and communications.- To obtain AMU performance data during its mission, a telemetry receiver capable of accepting the diphase PCM format transmitted from the AMU is installed on the electronics module in the spacecraft adapter. The receiver demodulates the 433 mc received signal and provides a 5120 bits/second diphase signal to the spacecraft PCM recorder. These data are recorded on track B and stored for postflight analysis.

Two whip antennas are mounted on the adapter surface to receive the AMU telemetry transmissions, with only one antenna being in use at any time. Selection of the proper antenna is accomplished by coaxial switching from a signal provided by the telemetry receiver. The receiver provides automatic control of the coaxial switch to change antennas when the RF signal on the antenna in use drops below the preset level.

Although propellant status is monitored in the cockpit, the same pressure and temperature can be monitored through normal spacecraft telemetry to the ground. A 0 to 715 psia transducer and a thermistor in the AMU propellant tank are powered by the spacecraft for channels RA01 (H_2O_2 pressure) and RA02 (H_2O_2 temperature). Hydrogen peroxide pressure is available until the AMU telemetry switch is placed in the "Backpack" position during the donning phase of the extravehicular mission, while H_2O_2 temperature is available until AMU separation from the spacecraft.

During AMU activities, the spacecraft UHF transceiver is used to maintain communications between the extravehicular pilot and the command pilot. The AMU voice-operated transceiver operates on 296.8 mc, which, with its other operating characteristics, is compatible with the spacecraft communications system.

8.1.4.6 Crew-station displays.- Spacecraft crew-station displays and controls shown in figure 8.1-20 comprise the following:

(a) EXP PROP indicator - The EXP PROP indicates pressure and temperature of the hydrogen peroxide propellant stowed in the AMU.

(b) The MMU H_2O_2 warning light - A red light on the annunciator panel is illuminated when rising H_2O_2 pressure reaches 575 ± 20 psia.

(c) The BUS ARM/DOCK-SAFE-EXP switch - The BUS ARM switch, located on the Agena control panel, must be in the EXP position to energize experiment squib circuits before AMU cover release, footrail extension, telemetry antenna deployment, and AMU release.

(d) The MMU/DEPLOY-OFF-TM ON switch - Several functions are provided by this switch. In the spring loaded DEPLOY position, the AMU is released by a guillotine cutting the hollow retention bolt and servicing lines. In the TM ON position, the telemetry receiver and its associated antenna coaxial switch are powered, and the tape recorder B-track is activated.

(e) INDEX EXTEND-OFF-EVA BARS EXT switch - As previously described, the EVA BARS EXT position releases the AMU thermal cover and deploys the footbar and handrails to the operational position. In addition, the AMU telemetry antennas are deployed to permit reception.

8.1.4.7 <u>Extravehicular Life Support System/AMU Interfaces</u>.- The ELSS becomes an integral part of the AMU system during donning of the AMU. It provides electrical, mechanical, and life support connections between the extravehicular pilot and the AMU.

Through an umbilical from the AMU oxygen supply system, oxygen is delivered to the ELSS environmental control system at 97 ± 10 psia and 65 ± 10° F. A quick disconnect on this umbilical is attached to a mating connector on the ELSS to allow oxygen flow. The OSS interface is discussed in detail in section 8.1.1.3.3 of this report. Other AMU/ELSS interfaces are presented in the following paragraphs.

8.1.4.7.1 AMU restraint harness interface: A restraint harness is provided as part of the backpack. The harness restrains the backpack to the space suit by bearing on the forward surface of the ELSS.

8.1.4.7.2 Malfunction detection system interface: Four alarm lights, visible to the astronaut, are provided on the ELSS to indicate the existence of out-of-tolerance conditions in the backpack propulsion (fuel quantity and pressurization), oxygen supply, and reaction control systems. The lights are activated by electrical signals from the backpack through the AMU electrical umbilical. The signals are continuous as long as an alarm exists. Light arrangement and function are as follows:

(a) Oxygen warning light - Illuminates at 800 ± 160 psia tank pressure or when oxygen temperature drops below 5 ± 5° F.

(b) Fuel quantity warning light - Illuminates at 30 percent total fuel remaining.

(c) Fuel low pressure warning light - Illuminates at a nitrogen pressure of 650 ± 100 psia or H_2O_2 pressure of 350 ± 50 psia.

(d) Certain critical functions of the reaction control system are monitored, and the failure of any one of these functions will result in the illumination of a warning light.

A switch is installed on the ELSS to test the operation of the alarm lights and portions of the backpack abort alarm subsystem. The test signal to the backpack is provided through the electrical umbilical.

The ELSS supplies a 1700 cps audio tone signal to the backpack radio receiver/transmitter upon receipt of a signal from the AMU alarm subsystem through the electrical umbilical. The signal to the backpack is continuous until the reset (disable) switch is actuated. Actuation of the switch, which is located on the top of the ELSS, generates a signal to the backpack, via the electrical umbilical, to reset the alarm trigger in the backpack.

8.1.4.7.3 Telemetry Interface: The backpack telemeters the following parameters received from the space suit and ELSS through the ELSS/AMU electrical umbilical:

(a) Electrocardiogram

(b) Respiration rate

(c) Suit pressure.

8.1.4.7.4 H_2O_2 quantity indication interface: A meter, visible to the astronaut, is provided on the ELSS to indicate the quantity of H_2O_2 remaining in the backpack. Signals are supplied to the meter from the backpack through the electrical umbilical.

8.1.4.8 Space suit/AMU interface.- An exhaust plume heating analysis, conducted early in the program, indicated that the Gemini thermal coverall materials, HT-1 Nylon and Mylar, would be heated beyond acceptable limits during operation of the AMU. As a result of detailed analysis of the problem, a decision was made to add extensions to the upper forward thrusters and to modify the leg portion of the basic Gemini coverall. The coverall modification started with an investigation to select materials and establish the insulation requirements. A materials screening and testing program resulted in the selection of a coverlayer of a woven fabric (Chromel-R wire), a superinsulation spacer material of fiberglass, and a reflective material of aluminized Polyamide-H film. Eleven layers of superinsulation were employed. The nozzle extensions were evaluated to determine their effect on performance, systems design, predonning activities, and the AMU development program. Performance tests on extension configurations indicated that extensions could be added without markedly affecting thruster performance. A detailed thermal analysis of the selected design verified that it would solve the heating problem associated with the upper forward-firing thrusters.

Prior to the Gemini IX mission, the decision was made to utilize the Gemini VIII extravehicular glove because its mobility and tactility characteristics were considered superior to the glove which had been developed for the AMU application. Because the Gemini VIII glove was found to afford very little thermal protection from the AMU exhaust plume, shields were incorporated on the AMU controllers. Although temperatures on the gloves were calculated to be significantly lower than those anticipated in the leg areas, the shields utilized the same materials and layup as the modified extravehicular coverall.

8.1.5 AMU Mission Activity Description

8.1.5.1 Flight planning.- The AMU mission on the Gemini IX-A flight was planned to extend over one complete revolution around the earth. The nightside was devoted to checkout and donning activities, and the dayside to the maneuvering evaluation. Electrical power and oxygen and propellant supplies limit the maneuvering or independent operation capability of the AMU to approximately 1 hour. The AMU mission, as planned, consisted of the following activities (see fig. 3.1-6).

At dark, the pilot moves via handholds along the surface of the spacecraft adapter assembly to the interior of the equipment section to check out and don the backpack. Trapeze-like handholds and footbars in the adapter interior are provided to support checkout and donning operations.

The predonning checkout consists of:

(a) Visually checking the oxygen supply system (OSS) and nitrogen tank (used for positive expulsion of the hydrogen peroxide propellant) pressures to verify adequate supply levels

(b) Manually opening the oxygen and nitrogen supply-tank shutoff valves

(c) Turning on the master electrical-power switch

(d) Unstowing the sidearm controllers and arranging the restraint strap, electrical umbilical, and oxygen umbilical in the donning positions.

Assuming that these checks indicate satisfactory systems status, the pilot turns around and backs into the AMU using the handholds and footbars for support. Attachment of the single restraint strap across the front of the chestpack physically unites the chestpack and backpack. The backpack OSS is connected to the chestpack ELSS through a separate connector so that external (to the chestpack) oxygen supply is uninterrupted. To verify satisfactory operation, the command pilot manually shuts off the spacecraft supply prior to the spacecraft oxygen umbilical being disconnected from the chestpack. The spacecraft umbilical electrical connector and the backpack electrical connector both attach to the same chestpack connector. Since voice communications are carried through this chestpack connector, they are interrupted briefly during the changeover operation.

After the checkout, donning, and changeover operations are complete, the command pilot releases the backpack from its mounting in the adapter by firing a guillotine to cut the attachment bolt and propellant servicing lines. The propellant vent line and the electrical cable are equipped with pull-away connections, and cutting of these is not required. After release, the astronaut will return to the spacecraft cabin area via the handrails to detach the spacecraft umbilical and to complete attachment of the AMU tether.

Initial flight checkout activity is performed on the 25-foot section of the lightweight tether in view of the command pilot, who uses the spacecraft attitude control system to keep the extravehicular astronaut in sight. Short translations and rotations about all three axes are made exercising both primary and alternate propulsion systems, and using both stabilized and manual control modes. Following these checks and the performance of sufficient familiarization maneuvers for the pilot to be completely confident of the AMU and his ability to control it, the 25-foot tether hook is detached to permit maneuvers out to the full tether length of 125 feet. Maneuvers will be performed to evaluate control capability, fuel usage in both stabilized and manual control modes, station keeping, and rendezvous.

When the AMU mission is completed, the pilot returns to the spacecraft nose area to retrieve the spacecraft umbilical. The reverse of the donning changeover is performed, the backpack released into space, and a normal ingress performed. If for any reason retrieval of the spacecraft umbilical is not possible or practical, the emergency supply of oxygen in the chestpack can be used for ingress.

8.1.5.2 Actual mission.- The AMU was serviced for flight prior to the initially planned launch date of May 17, 1966. Monitoring of the propellant status after launch cancellation indicated a stable pressure rise of 0.2 psia per hour due to normal active-oxygen loss, which was well below the allowable of 0.6 psia per hour. The decision was made not to reservice the propellant. At launch on June 3, 1966, the pressure had increased to approximately 87 psia, a nominal condition for launch. The oxygen and nitrogen systems, monitored through ground equipment, showed zero leakage. Fresh batteries were installed in the flight unit on May 25, 1966. A telemetry check of the AMU, subsequent to battery replacement, indicated that all systems were operating normally.

Immediately after launch the propellant tank pressure increased to a normal 90.7 psia where it remained until nitrogen pressure was applied to the propulsion system by the pilot during donning. The propellant temperatures measured during this period were normal at 72 to 77° F.

As the pilot entered the spacecraft adapter section to begin the AMU experiment, he found that the left adapter handhold and the umbilical guide were not fully extended and the AMU adapter thermal cover was not completely released. In addition, the left adapter floodlight was not operating. As the pilot pulled on the handhold to enter the adapter, the handhold and umbilical guide moved to the fully-deployed position and the thermal cover was released. The pilot completed AMU inspection and donning activities through the point of connecting the AMU electrical umbilical. These activities included attaching portable penlights, opening nitrogen and oxygen shutoff valves, readout of oxygen and nitrogen pressures, positioning sidearm controllers, positioning umbilicals and AMU restraint harness, attaching AMU tether, turning on AMU electrical power, and changeover to AMU electrical umbilical. The oxygen pressure was 7500 psia, and nitrogen pressure was approximately 3000 psia, both of which were normal. The propulsion system pressure after opening the nitrogen valve was 455 psia – normal AMU operating pressure. Accomplishment of AMU donning activities was more difficult than expected and required a much longer time to complete because of the difficulty in maintaining position in the adapter. The pilot constantly tended to drift away from the work area in the adapter. Extension of the sidearm controller and attachment of the AMU tether were particularly difficult because both hands were required and neither hand could be used to maintain position. AMU communications to the command pilot were "garbled" but were considered acceptable by both pilots. (Some degradation of communications was expected while the AMU was in the adapter.)

Because the pilot's visor began to fog and restrict his vision, the command pilot decided that the AMU experiment could not be completed. The pilot then disconnected the AMU umbilical, connected the ELSS umbilical, and returned to the cockpit for ingress, leaving AMU power on. The AMU remained in the adapter with the systems activated for flight until adapter separation prior to retrofire.

8.1.6 AMU Performance

Termination of the extravehicular activities before the AMU mission could be completed precluded an evaluation of most of the AMU performance capabilities. However, the backpack experienced a Gemini launch and a 2-day orbital-soak period, and most of the functions of checkout and donning were performed prior to aborting the mission. Although the AMU was transmitting telemetry data following power-up during the predonning activity, failure of the Gemini PCM tape recorder precluded a quantitative analysis of the AMU systems performance. Analysis of the AMU systems, therefore, is based primarily on comments made by the flight crew during their debriefing.

8.1.6.1 Prelaunch.- AMU stored gaseous expendables, breathing oxygen and pressurant nitrogen, were serviced prior to May 6, 1966, and were never reserviced or topped off. Both tank pressures were indicating a full charge when they were checked during preflight on May 29, 1966. The AMU was serviced with H_2O_2 on May 15, 1966. Ullage pressure built up from the initial 19.7 psia to 87.9 psia at launch. The final prelaunch pressure could not be precisely determined due to a slow leak in the ground servicing and monitoring equipment. Temperature of the H_2O_2 remained at the ambient temperature of the Gemini white room.

The initial sets of AMU batteries were replaced on May 21, 1966, because they would have been approaching their demonstrated wet stand-time life by the time the mission was conducted. The final sets had been activated on May 5, 1966, and were considered good for rated load until June 13, 1966.

No major problems were encountered in the installation of the AMU in the adapter. Some interference was encountered in installing the AMU thermal cover because of the installation of the AMU sidearm controller thermal shields; however, after evaluation, the interference was judged to be insignificant and the installation was accepted for flight.

Reading of H_2O_2 pressure and temperature were monitored on the cockpit gages and/or by telemetry throughout the prelaunch period. Readings on the cockpit H_2O_2 pressure gage were approximately equivalent to telemetry readings. The cockpit temperature gage read 6° F lower than the telemetry readings. The cockpit H_2O_2 pressure gage had a range of 0 to 500 psia and the cockpit temperature gage had a -100 to +200° F range, both on small dial faces. The problems of instrument accuracy and parallax make these instruments difficult to read closer than ±10 psia and ±10° F.

8.1.6.2 Launch.- The only AMU parameters available to either the crew or the ground monitor during launch were H_2O_2 pressure and temperature. No change in either parameter could be detected by the crew who monitored both parameters periodically during the launch. No temperature change was noted on telemetry during launch. An increase of two PCM counts in H_2O_2 pressure, or approximately 6 psia maximum, was recorded on telemetry at lift-off. No change in pressure was noted during the launch.

8.1.6.3 Orbit.- During an approximate 2-day pre-EVA period, the AMU parameters available for monitoring were H_2O_2 pressure and temperature. These parameters were monitored a minimum of one time per orbit by telemetry and approximately five times by the astronauts using cockpit gages. The relatively few cockpit readouts were due to the low activity of both parameters. During the pre-EVA period, the predicted active oxygen loss (AOL) buildup, based on the system dimensions and previous activity and on the current temperature, was continuously computed and plotted against the recorded AOL buildup.

AOL pressure buildup was much lower than predicted. During the 50 hours 37 minutes before the backpack telemetry switch was changed to BACKPACK (during AMU donning), the total pressure rise was less than one PCM count (3 psi). A rise of 8.5 psi had been predicted based on system activity prior to launch.

Temperature of the H_2O_2 was expected to fall, based on a thermal analysis of the AMU storage area. During the pre-EVA period the temperature varied from 69° F to 78° F. Readings on the cockpit gages during this period were 65° F and 90 psia.

8.1.6.4 Extravehicular activity (EVA).- Ingress to the spacecraft adapter was accomplished at 50:09:00 ground elapsed time (g.e.t.) of the Gemini IX-A mission. Dangling lines or lanyards which had been observed on the Gemini VII/VI-A rendezvous mission were not encountered. The thermal cover had not fully deployed and the left handhold was trapped behind the cover and was not fully extended. The pilot released the cover by hand with no difficulty. It separated from the spacecraft, and the handhold extended normally.

The pilot moved into the adapter, stood on the footbar, and pulled out some of the slack in the umbilical between the cockpit and the adapter umbilical guard. The umbilical guide allowed some movement of the umbilical in the cockpit direction, but the pilot reported adequate umbilical slack for AMU donning.

One of the floodlights in the adapter was not on. The pilot removed and turned on the two penlights stowed in the AMU tether bag. Although one of the lights did not work, he attached both to the handholds. The operating light was mounted on the side opposite the functioning floodlight. This lighting arrangement was considered marginally adequate by the pilot, and night donning activities were considered acceptable. The mirrors were unstowed and positioned.

The pilot examined the AMU for evidence of loose mounting, structural damage, or unstowed gear. No degradation from the prelaunch configuration was noted.

At 50:19:00 g.e.t., the black hook on the tether jumper was attached to the ring on the AMU tether. No difficulty was encountered at this point since the pilot held on to the short piece of tether secured to the tether bag which, in turn, was secured to the AMU seat. This provided a three-point support (both feet and the tether), and left one of his hands free to snap the hook over the ring. Next, the tether bag was unstowed from the AMU with no difficulty and the pilot attempted to attach the small AMU tether hook (25-foot length) and large AMU tether hook (125-foot portion) to the ring on the tether jumper. This operation proved very difficult and the pilot was unable to connect the small hook to the ring. The decision was made to continue donning and either to connect the small hook after donning or to operate completely on the 125-foot tether. During this series of events, the pilot changed to the high flow setting on the ELSS because of a hot spot on his back.

At 50:28:00 g.e.t., the pilot unstowed and checked the sidearm controllers. Unstowing the attitude controller proved difficult under both one and two-handed operations. This difficulty resulted from the combination of the fairly high unstowage forces and the tendency of the pilot's feet to slip out of the stirrups. This tendency of the feet to slip out of the stirrups was noted earlier during the tether hookup and was the principal difficulty experienced in all work activities in the adapter. Unstowing was finally accomplished using both hands for a quick hard pull on the controller. During this period the pilot became aware of fogging on the suit faceplate.

At 50:31:00 g.e.t., the pilot unstowed the oxygen umbilical, electrical umbilical, and restraint harness without difficulty. He experienced some difficulty in checking the VOX switch because it was difficult to reach with the controller in the down (donning) position.

At approximately 50:33:00 g.e.t., the pilot opened the nitrogen and oxygen shutoff valves. The nitrogen valves opened easily, and he could feel the nitrogen flowing into the H_2O_2 tank ullage volume. More than one try was required to open the oxygen valve, and the opening torque was described by the pilot as higher than he had experienced during any simulation. After the initial breakout, the valve opened easily. The pilot could not feel oxygen flowing through this valve. (This system is dead ended at the oxygen umbilical and the downstream volume is small.) After opening the valves, the pilot read both high pressure gages on the front of the backpack. He reported a normal 7500 psia on the oxygen gage but approximately 3000 psia on the nitrogen gage, which is normally 2775 psia. The oxygen gage was fairly easy to read but he reported some difficulty in reading the nitrogen gage. The command pilot reported the H_2O_2 pressure rose to 450 psia immediately after the nitrogen valve was opened and then rose slowly to 455 psia which is the nominal regulated pressure.

At 50:37:00 g.e.t., the pilot released the nozzle extensions which deployed promptly. He switched the H_2O_2 transducer from spacecraft to backpack telemetry and turned on the main power switch. The upper position lights were observed to come on. He again reported fogging of his visor and his suit pressure gage.

At 50:39:00 g.e.t., the pilot turned around to his left, and backed into the AMU. After he turned and had positioned himself in the AMU with his feet on the footbar, he had no difficulty maintaining his position.

At 50:42:00 g.e.t., the pilot changed over to the AMU electrical umbilical. No difficulty was encountered in making the change. The reaction-control-system warning light came on, and the warning tone was heard by both the pilot and the command pilot. The pilot reset his warning tone, and the warning light remained on. At this point the spacecraft onboard voice tape ran out. From this point until the AMU mission was terminated, the pilot and command pilot communicated RF via the AMU transceiver. The pilot reported his reception clear but slightly low in volume, with background noise. He reported no anomalies in the side tone from his transmissions. The command pilot reported the pilot's transmissions to be abnormal after the first few syllables. This anomaly was described as a wavering tone superimposed on the transmission. The command pilot considered it marginally acceptable but noted that extensive training in the donning exercise permitted him to understand the pilot's messages from less than complete transmissions. The pilot used both AMU switch positions (normal VOX and listen mode) during the transmissions, and the command pilot used either the VOX or the push-to-talk mode of the spacecraft communications system during this period.

The pilot checked out the displays and warning lights on the ELSS and determined that they were normal. He reported the H_2O_2 quantity as reading 85 percent (normal for a full load of H_2O_2 after the nitrogen supply is opened to the propellant tank).

The pilot located the restraint harness and assured himself that he could hook it up but delayed the hookup until he could see well enough to visually check his connections. At this point, the pilot reported that the visor fogging had advanced to the point where he had blurred forward vision with no side vision. Because of this condition, the crew decided to forego complete donning until the fogging problem could be alleviated. While resting and awaiting sunrise, the pilot raised both sidearm controllers to the horizontal position and locked them in the operational position. The orientation of the plume shields on the attitude controller did not appear correct, but the control head did appear to be extended to the right length. He tried unsuccessfully to turn the controller head with one hand to assure himself that the controller was in the proper position. Later, when the pilot left the donning station, he twisted the control head and thought he detected a further extension of the arm. He lowered the translation controller with no difficulty.

Sunrise occurred at 50:56:00 g.e.t. and the pilot attempted to use the donning mirrors to check his condition with the AMU. At 51:00:00 g.e.t., he reported that visor fogging had increased to the point that he was unable to use the mirrors.

At 51:03:00 g.e.t., the command pilot declared a no-go condition for the AMU evaluation. In postflight debriefing the pilot reported that the AMU was nominal in all respects and that the decision to abort was based entirely on visor fogging and the schedule condition caused by the fogging. The pilot changed over to the spacecraft electrical umbilical and egressed from the adapter without incident.

8.1.6.5 Post-EVA.- There were no established procedures for conditioning a backpack to be left in the adapter section. After the pilot returned to the cockpit and the cockpit was repressurized, he reported that the backpack condition was as follows:

Main power switch	ON
RCS handles	OFF

Telemetry switch	BACKPACK
Nitrogen valve	OPEN
Oxygen valve	OPEN
Attitude controller	Flight position (horizontal)
Translation controller	Donning position (down)
Electrical umbilical	Stowed on translation controller
Oxygen umbilical	Probably stowed on translation controller
Restraint straps	Stowed on translation controller

An evaluation was conducted to determine whether the AMU should be jettisoned or retained until adapter separation. With the sidearm extended and the restraint and umbilicals restrained only by velcro there was a possibility of the backpack hanging up in the adapter. If the backpack remained, the gyros would exhaust the RCS batteries and probably overpressure the battery cans. Also the OSS heater would cycle without flow for several hours before the 28-volt battery was exhausted. It was no longer possible to monitor H_2O_2 pressure on telemetry, and the range of the cockpit gage (0 to 500 psia) was nearly exceeded when the tank was pressurized to 455 psia during donning.

A decision was made to leave the AMU in the adapter. The basis of this decision was the unknowns associated with jettisoning the backpack in its post-donning configuration and there were no known hazards in leaving the AMU in the adapter in this condition. The AMU batteries are protected from over-pressurization by burst disks. The OSS heater had been tested with oxygen in a continuously-on condition for 40 minutes, at which time no fire or explosion had occurred and the heater had stabilized. The flight history of the H_2O_2 had evidenced very little activity. The telemetry readout of temperature, the temperature and pressure gages in the cockpit, and the cockpit high-pressure warning light were considered adequate monitoring provisions for detecting an unsafe condition.

The temperature of the H_2O_2 was monitored by telemetry continuously until the adapter was separated. The temperature was very stable with a slight rising trend to a final temperature of 70° F. Temperature readout in the cockpit was 60° F. The cockpit pressure gage rose to a reported 500 psia at 66:39:00 g.e.t. At 70:32:00 g.e.t., the crew reported it to be 491 psia.

8.1.7 Conclusions and Recommendations

8.1.7.1 Conclusions.- All AMU systems exercised during the mission were in an acceptable condition for proceeding with the mission when the AMU evaluation was terminated. Some difficulty was experienced in reception of the AMU voice signal by the command pilot, who described the condition as marginal. The AMU transceiver, the spacecraft transceiver, and the conditions in the adapter must be investigated to determine the cause of the problem.

All donning provisions appear practical for use in the orbital environment; however, the donning activities were more difficult to perform than experienced in the one-g and zero-g training exercises. These activities will have to be facilitated by improvement of the restraint hardware at the donning station, and the forces required to unstow the attitude controller should be decreased. Also, the adapter lighting was marginal because one adapter floodlight and one penlight failed. This condition must be corrected.

The crew reported a tendency of the pilot and any loose equipment to move away from the spacecraft. This tendency affects activities at a predefined work station, as experienced on Gemini IX-A. This tendency must be understood qualitatively and quantitatively in planning future extravehicular missions. Night retrieval of the propulsion device from the adapter is satisfactory if artificial lighting is adequate.

8.1.7.2 Recommendations.- For future AMU activities the following general guidelines are recommended:

(a) Acquire and don the propulsion device as early in the mission as possible.

(b) Provide positive body restraint to minimize the work required for maintaining position at a work area. The restraint hardware should be sufficient to allow two-handed operations.

(c) Provide hardware such as hooks, rings, and fittings that can be easily handled in pressure suit gloves and that provide some margin for attachment.

(d) Assure that operations such as unstowing the controller arms and opening the oxygen valve do not require higher forces than can be readily applied.

The following activities must be accomplished prior to the next AMU mission:

(a) Resolve the visor fogging problem experienced on Gemini IX-A.

(b) Identify the problem in jettisoning the AMU thermal curtain and correct it, if necessary.

(c) Review AMU deployment procedures including those related to jettisoning after the controller arms have been released.

(d) Investigate means to reduce the workload associated with lowering the AMU sidearm controllers. Review methods of retaining the side arms in the down position during donning.

(e) Improve astronaut restraint hardware at the donning station.

(f) Identify and correct the communications anomaly experienced on Gemini IX-A.

(g) Provide a larger tether hook for the 25-foot tether and larger rings on the tether jumper and AMU tether.

(h) Improve the lighting conditions at the donning station.

(i) Establish positive procedures to record voice transmission during AMU mission.

(j) Investigate color coding of the AMU oxygen and nitrogen gages.

(k) Re-examine all activities that are currently planned to be accomplished without a prepared work station or a stabilized propulsion unit. These include thruster warmup, reconnection to the spacecraft umbilical, and AMU donning.

TABLE 8.1-I.- BACKPACK WEIGHT

Item	Weight, lb
Structure	33
Propulsion	63
Flight control	13
Oxygen supply	26
Power supply	20
Abort alarm	1
Communications	10
	166

TABLE 8.1-II.- TELEMETRY PARAMETER LIST

System	Parameter	Range	Number of channels	Type
Astronaut	EKG	--	1	Analog
	Respiration rate	--	1	Analog
	Suit inlet temperature	40° F to 90° F	1	Analog
	Suit exhaust pressure	0 to 15 psia	1	Analog
Propulsion	N_2 tank pressure	0 to 4000 psia	1	Analog
	H_2O_2 pressure	0 to 715 psia	1	Analog
	H_2O_2 temperature	0 to 160° F	1	Analog
	H_2O_2 fuel remaining	0 to 100 percent	1	Analog
Flight control	Rate gyro	0.01 to 45 deg/sec	3	Analog
	Control valve	On - off	8	Bilevel
	Control system valve	On - off	2	Bilevel
	Manual switch position	Manual - automatic	1	Bilevel
	Control switch position	On - off	10	Bilevel
Oxygen	O_2 tank pressure	0 to 8000 psia	1	Analog
Power	28 V	0 to 28 V dc	1	Analog
	Primary +16 V	0 to +16 V dc	1	Analog
	Primary -16 V	0 to -16 V dc	1	Analog
	Alternate +16 V	0 to +16 V dc	1	Analog
	Alternate -16 V	0 to -16 V dc	1	Analog
	Transducer excitation	0 to 10 V dc	1	Analog
	Signal conditioner reference	0 to 5 V dc	1	Analog
Abort alarm	Alarm signals	On - off	4	Bilevel
Communications	Voice transceiver AGC	0 to 8 V dc	1	Analog
	Signal conditioner temperature	0 to 160° F	1	Analog

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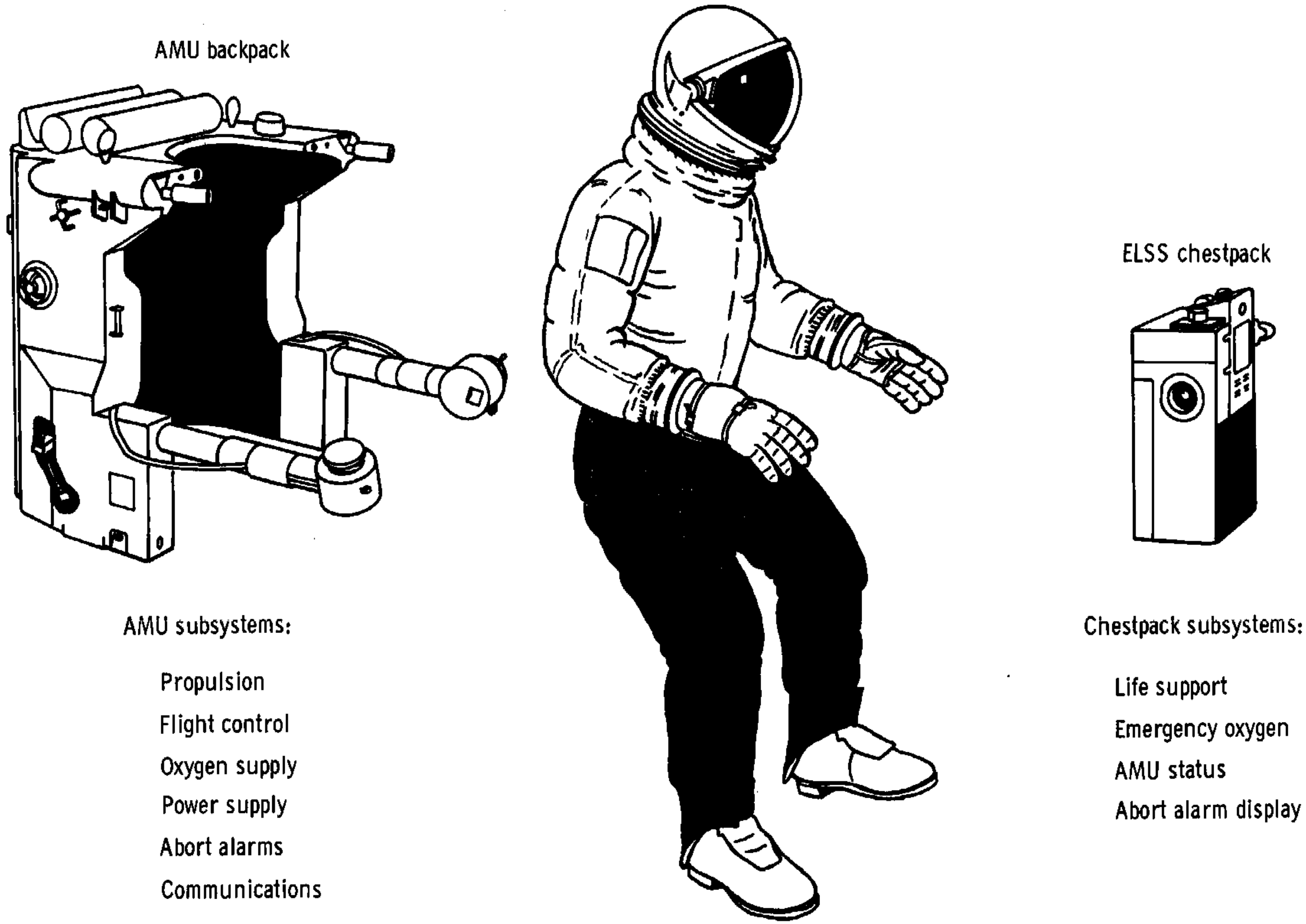

Figure 8.1-1. - Experiment D-12, AMU configuration.

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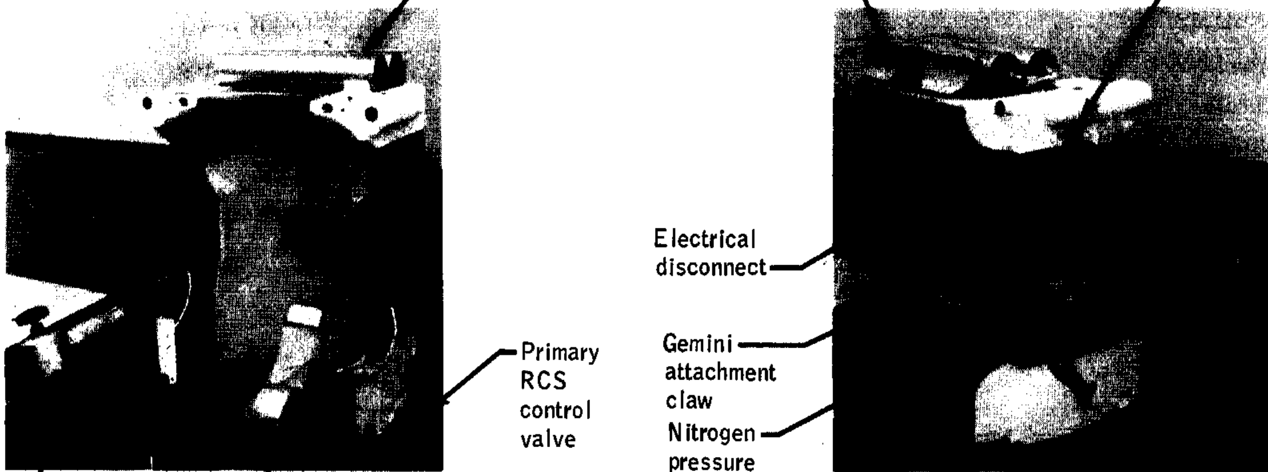

Figure 8.1-2. - Experiment D-12, AMU external structure.

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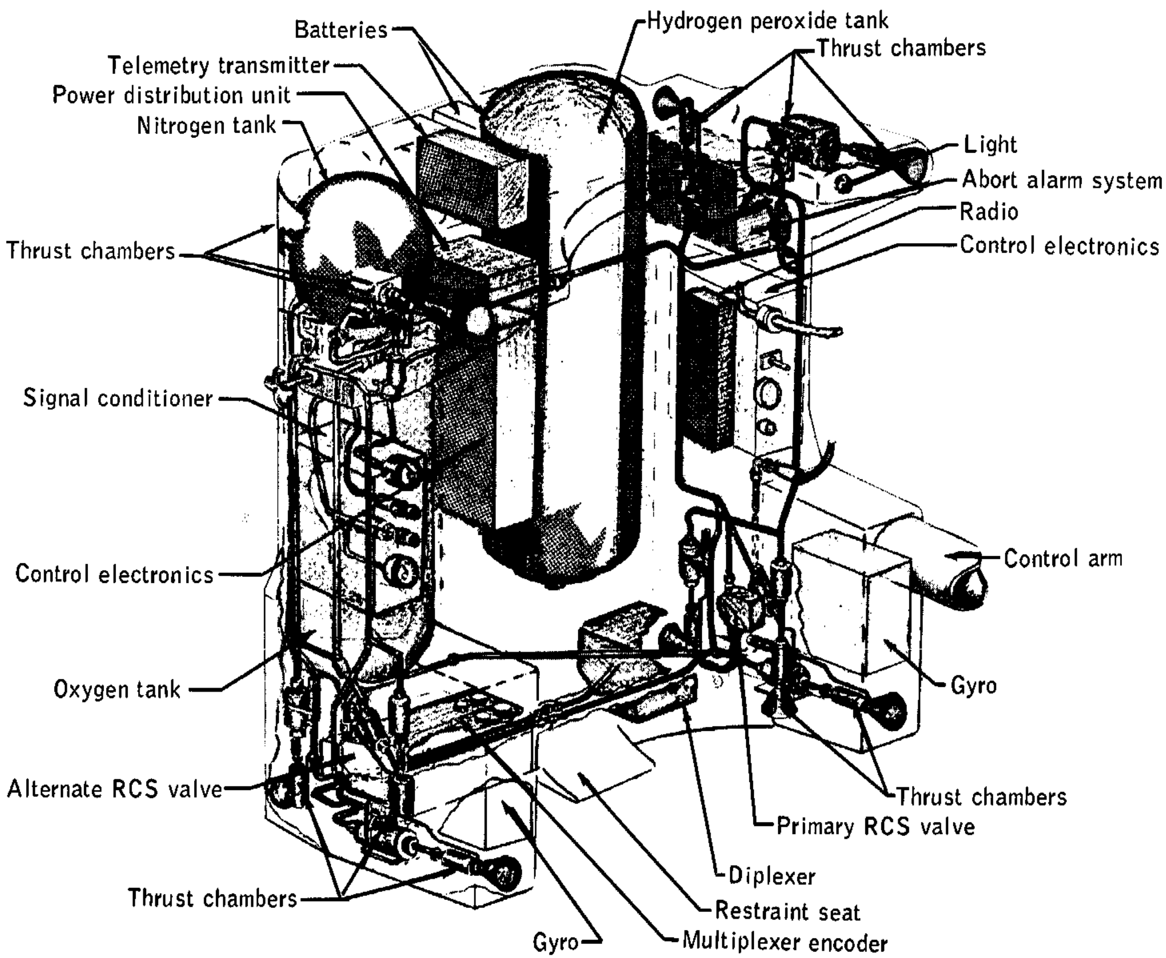

Figure 8.1-3. - Experiment D-12, AMU internal configuration.

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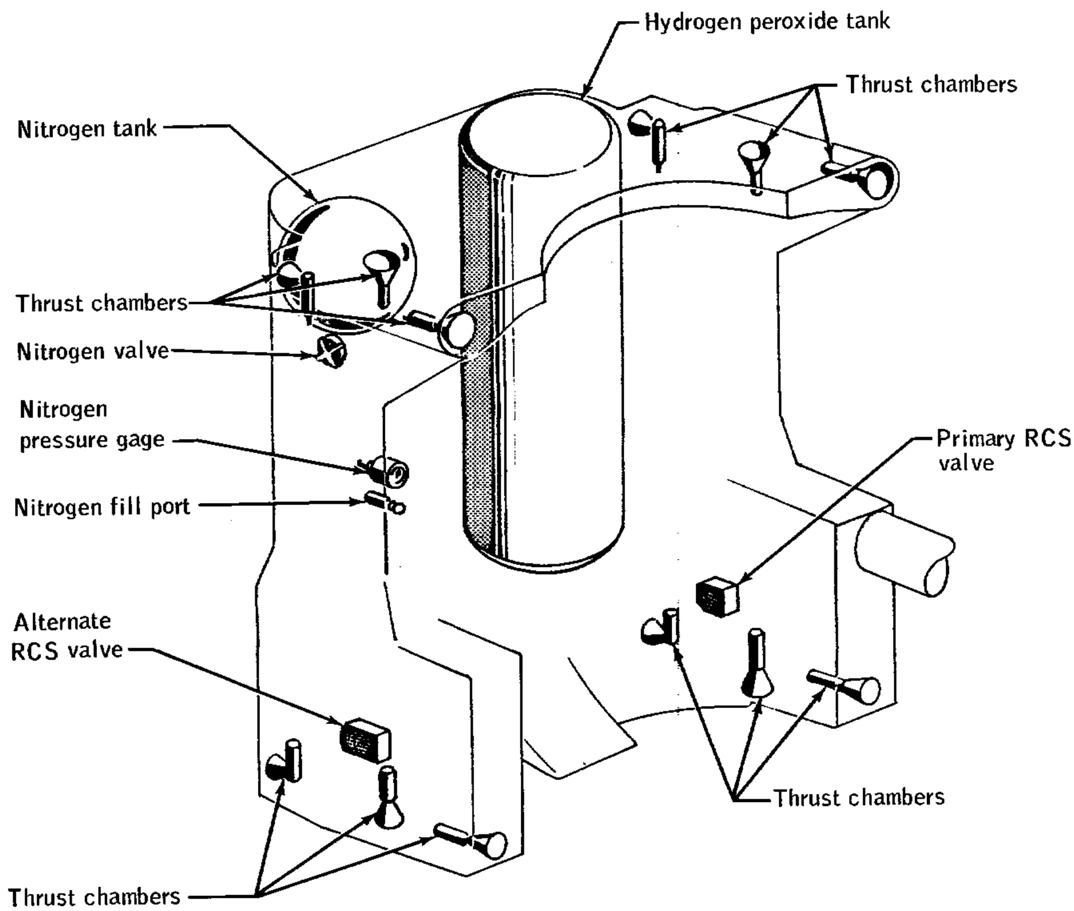

Figure 8.1-4. - Experiment D-12, AMU propulsion system.

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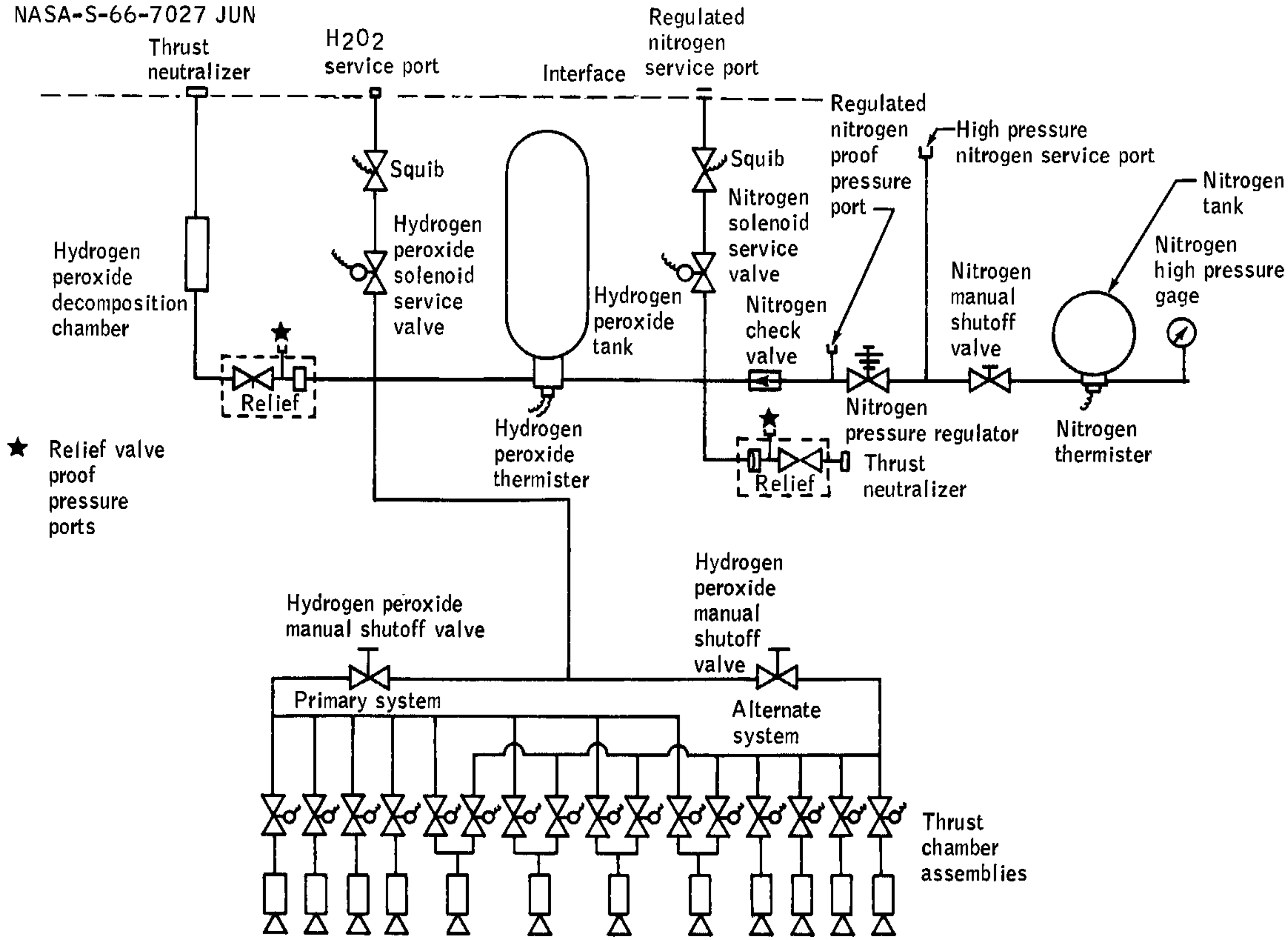

Figure 8.1-5. - Experiment D-12, AMU propulsion system schematic.

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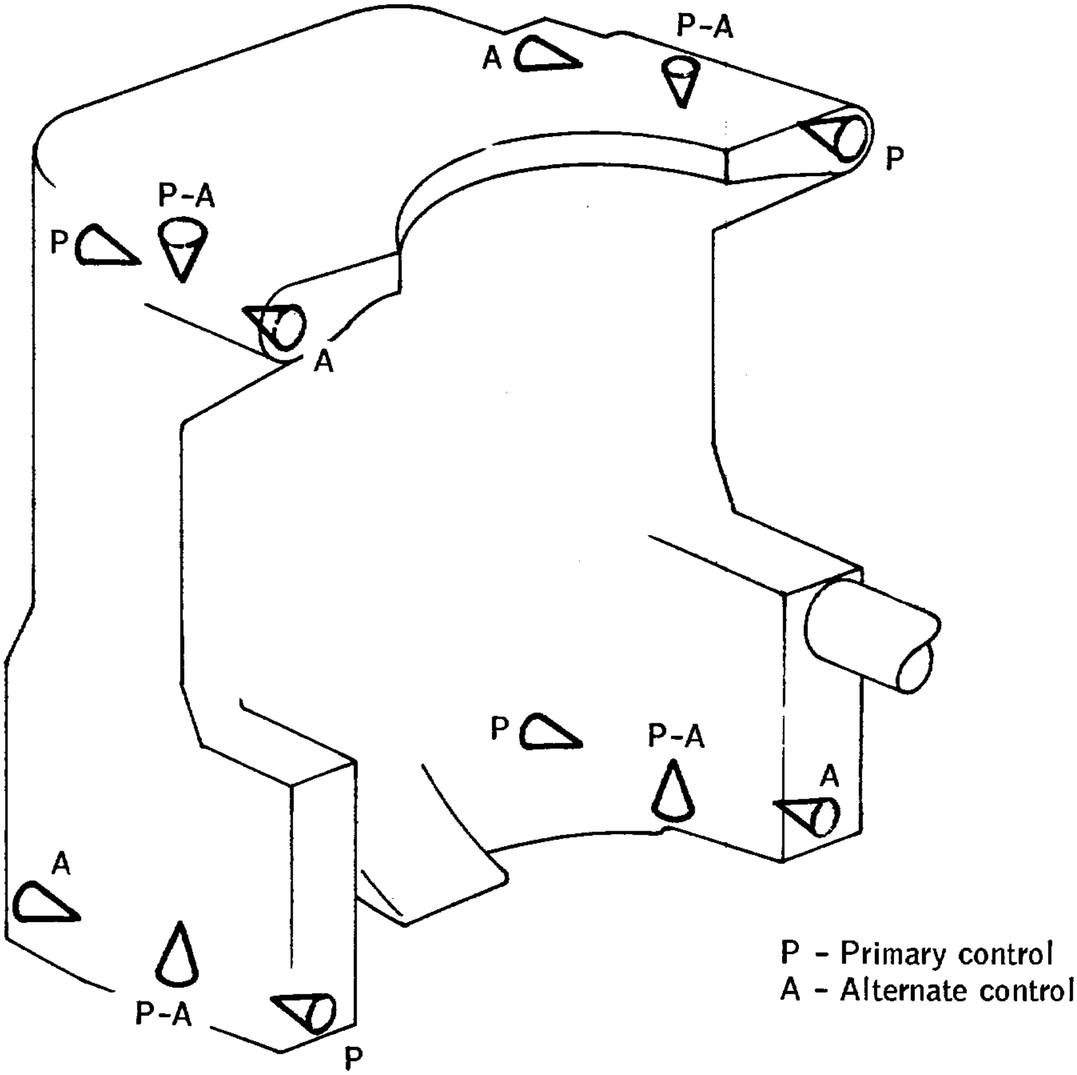

Figure 8.1-6. - Experiment D-12, AMU thruster arrangement.

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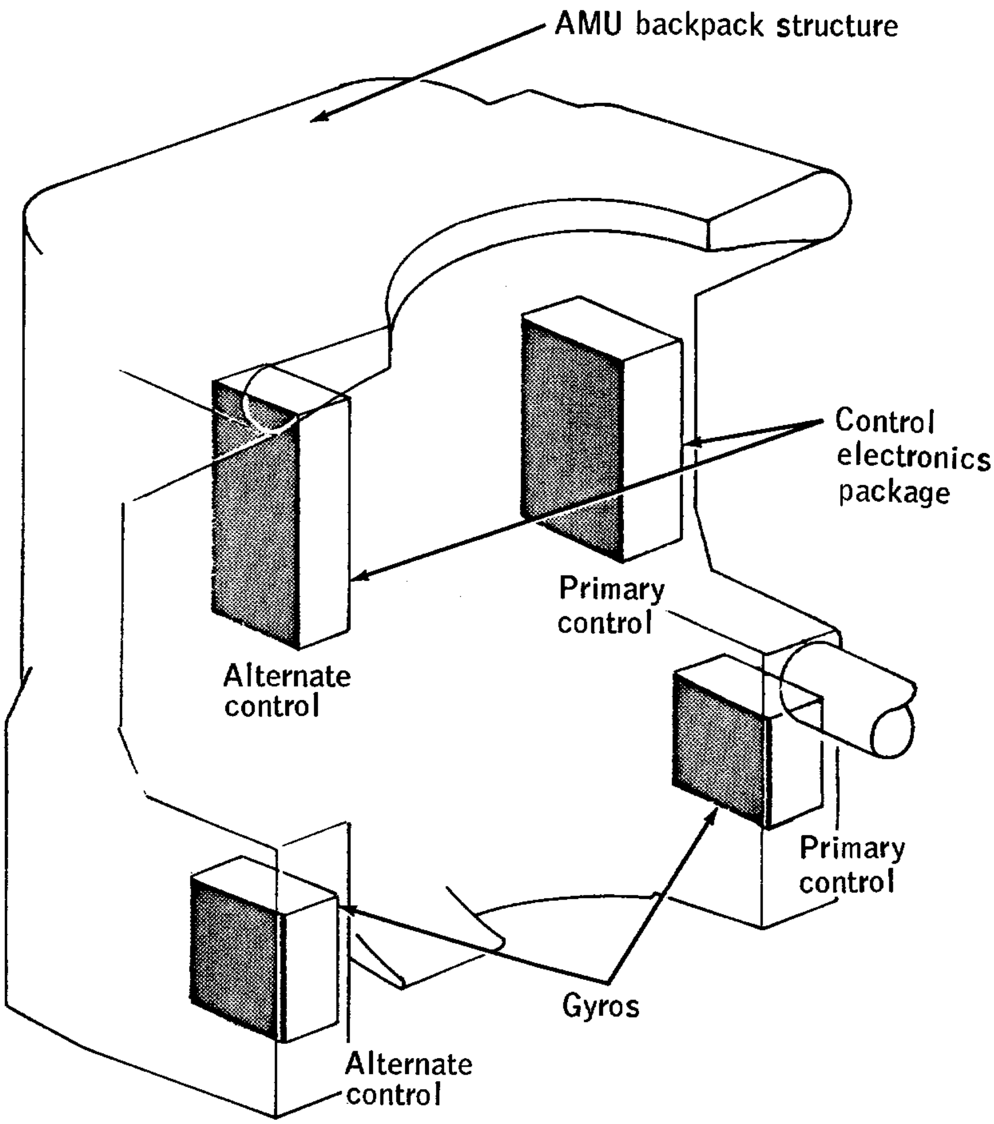

Figure 8.1-7. - Experiment D-12,AMU stabilization and control system.

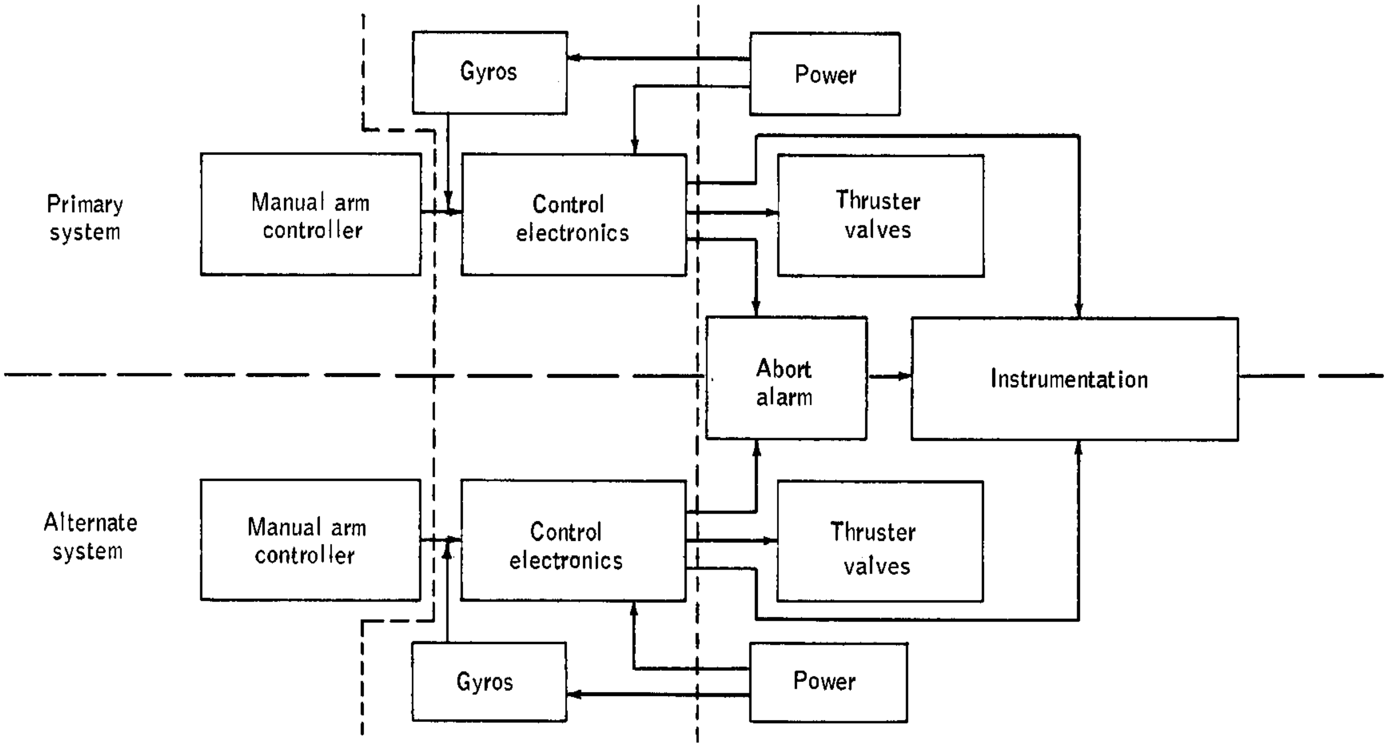

Figure 8.1-8. - Experiment D-12, AMU flight control system.

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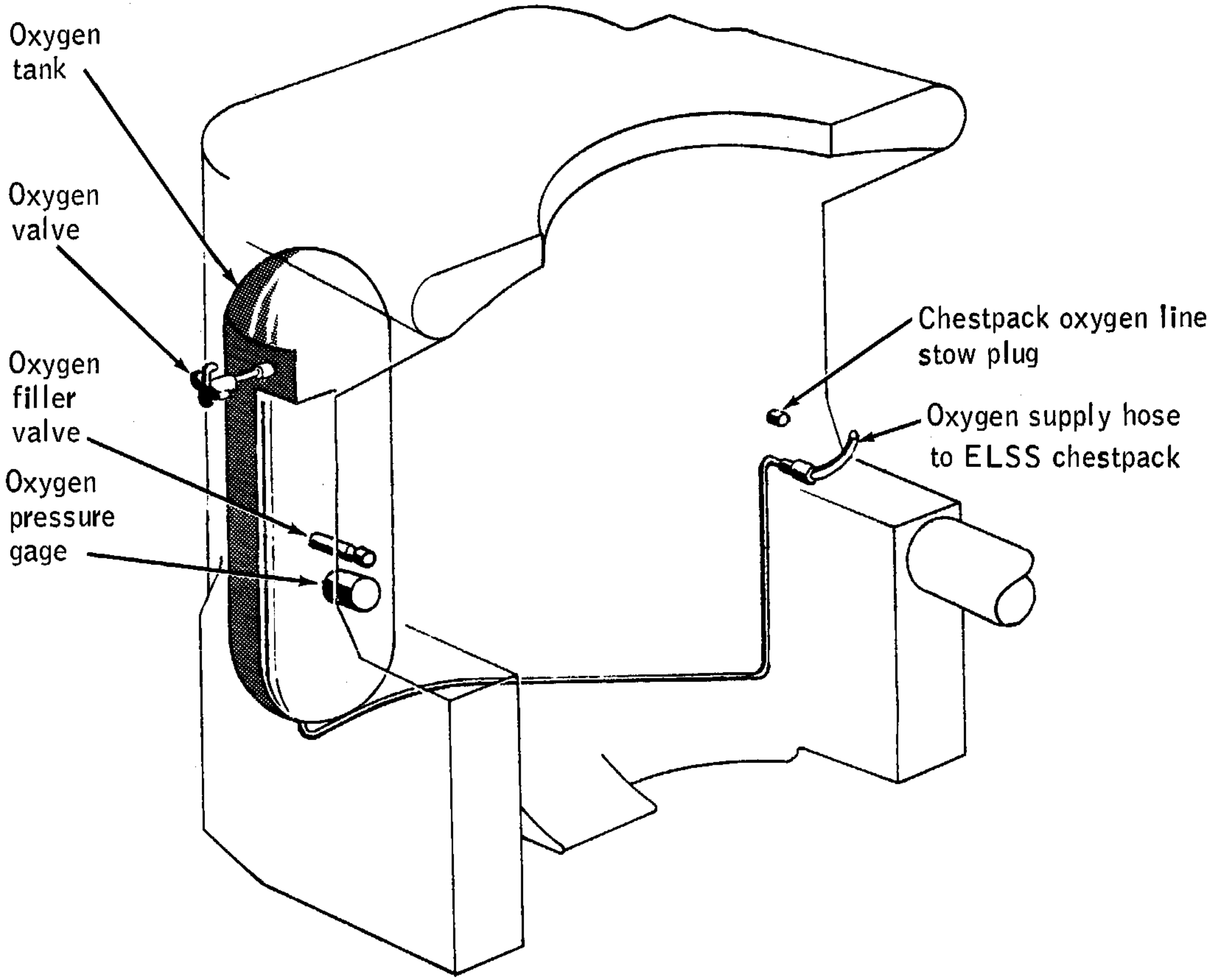

Figure 8.1-9. - Experiment D-12, AMU oxygen supply system to ELSS.

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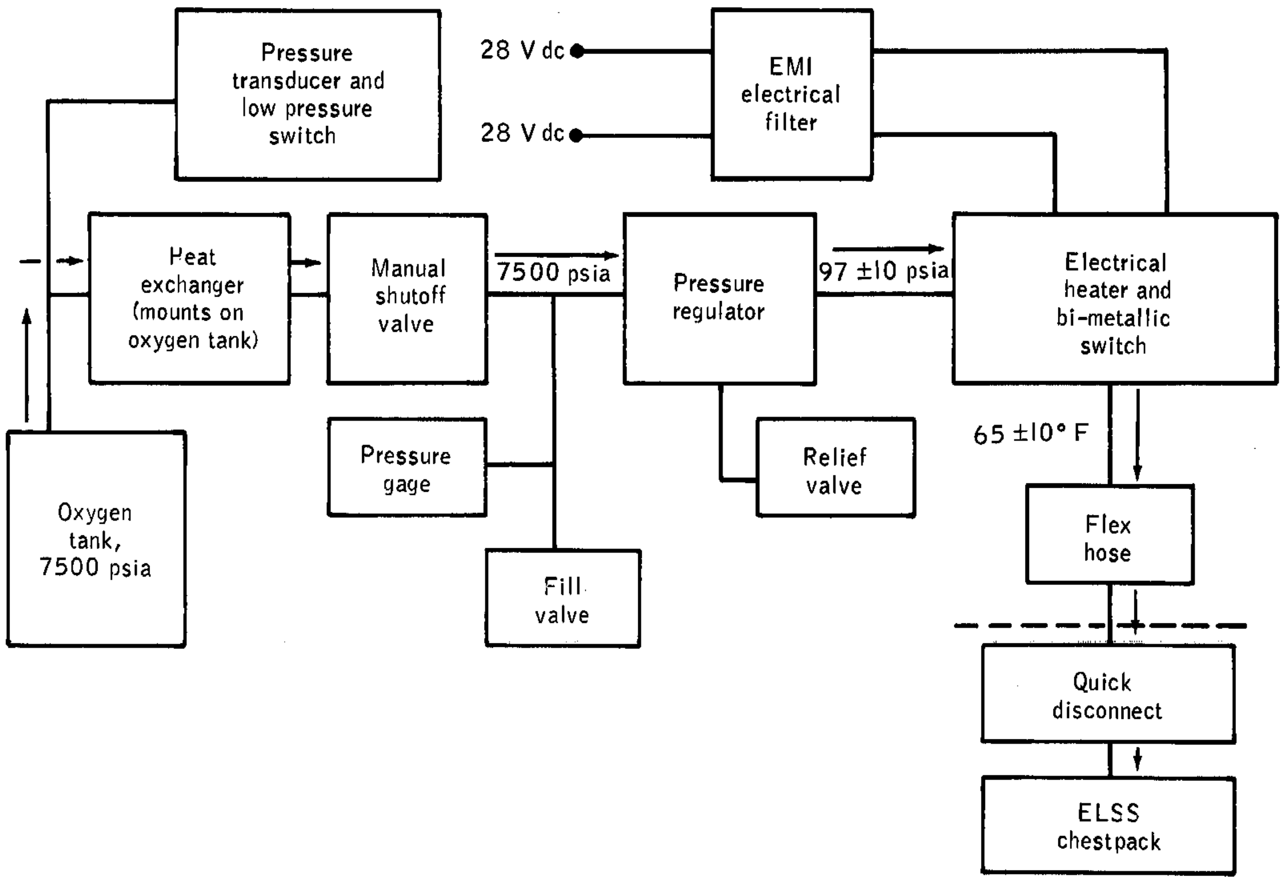

Figure 8.1-10. - Experiment D-12, AMU oxygen supply functional diagram.

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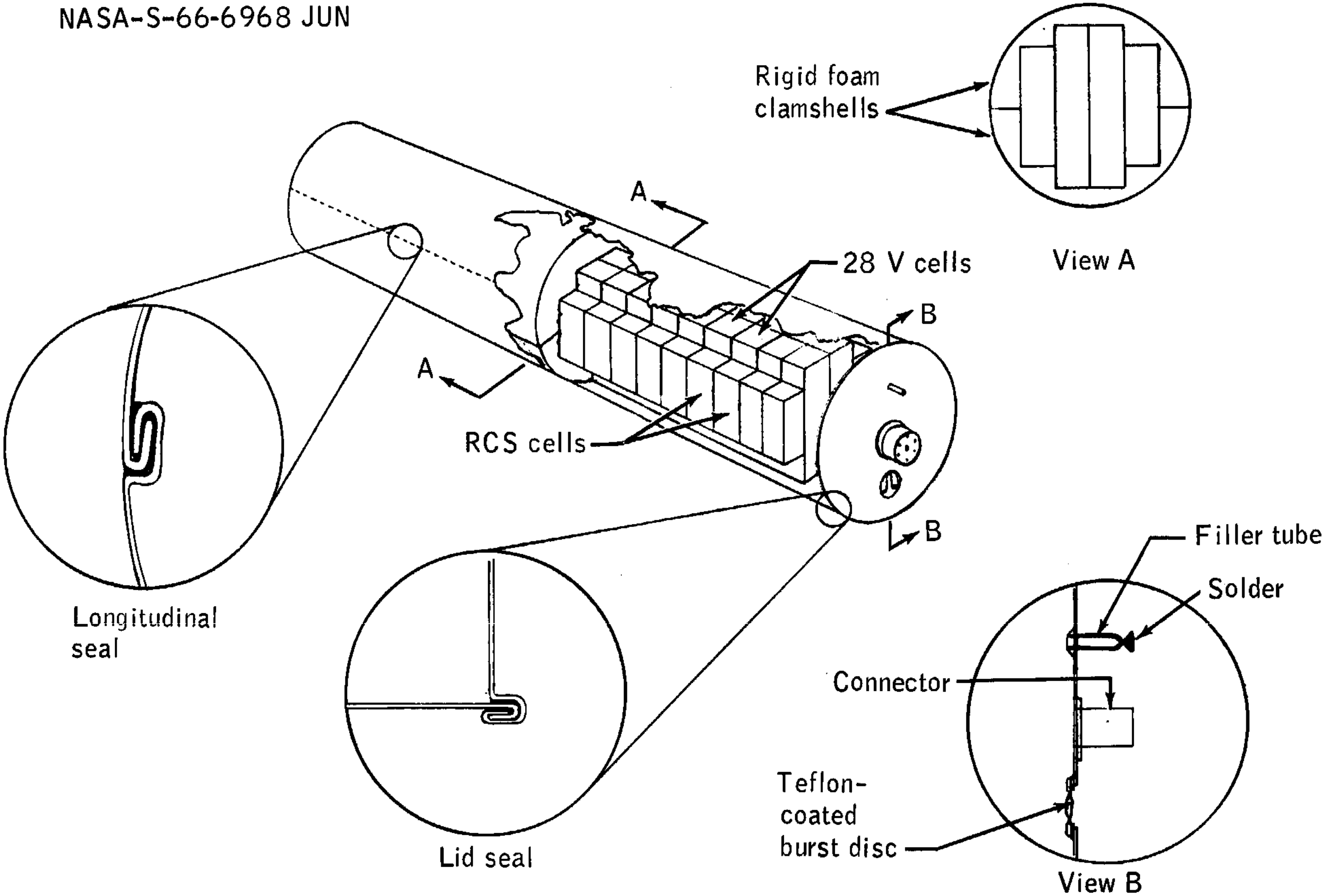

Figure 8.1-11. - Experiment D-12, AMU battery pack assembly.

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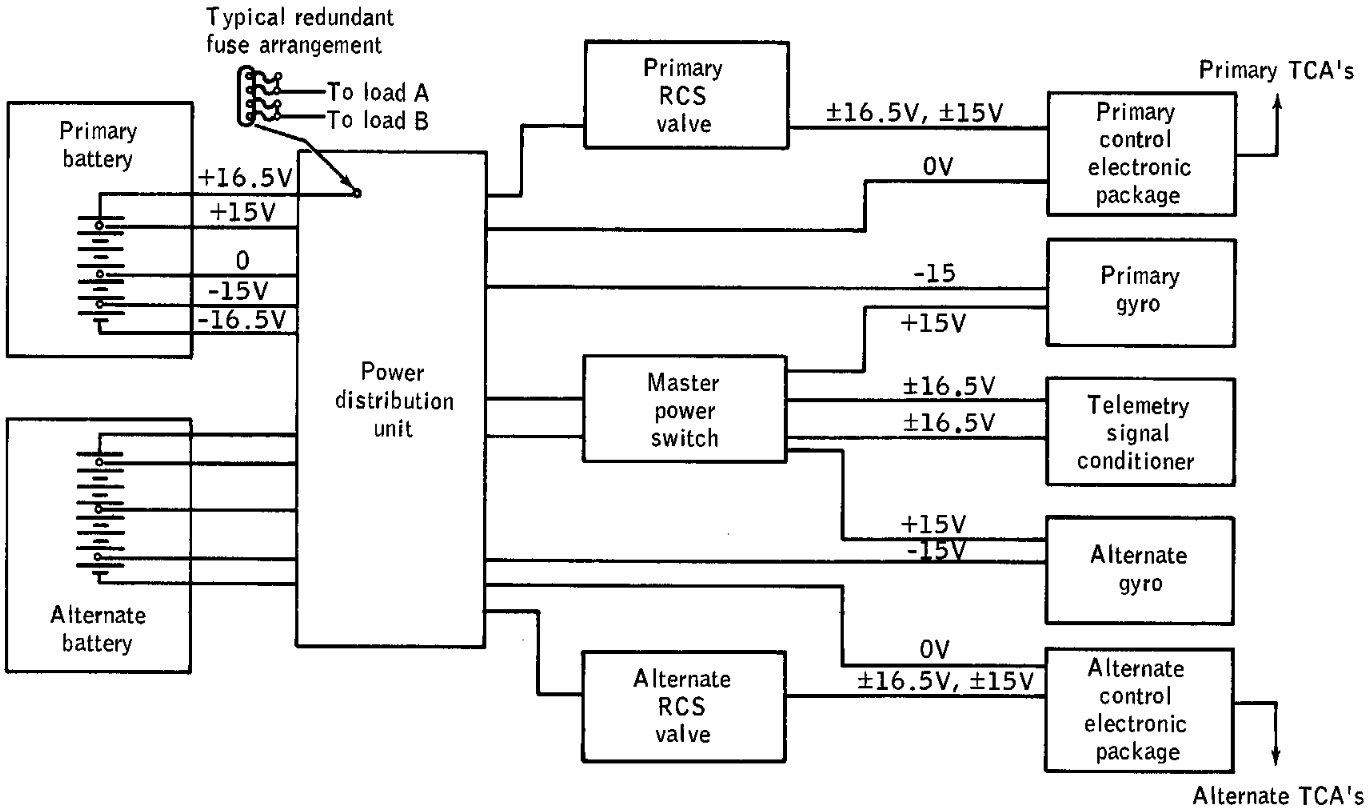

Figure 8.1-12. - Experiment D-12, AMU RCS electrical system.

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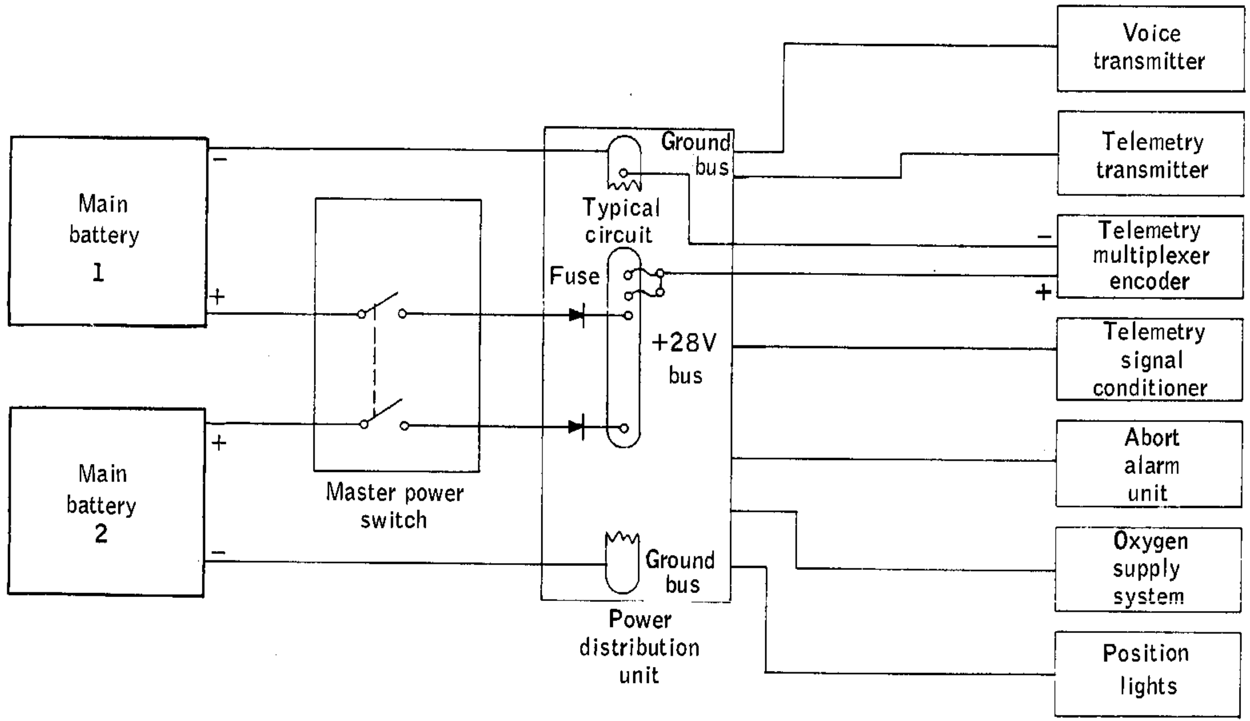

Figure 8.1-13. - Experiment D-12, AMU 28 V dc power supply.

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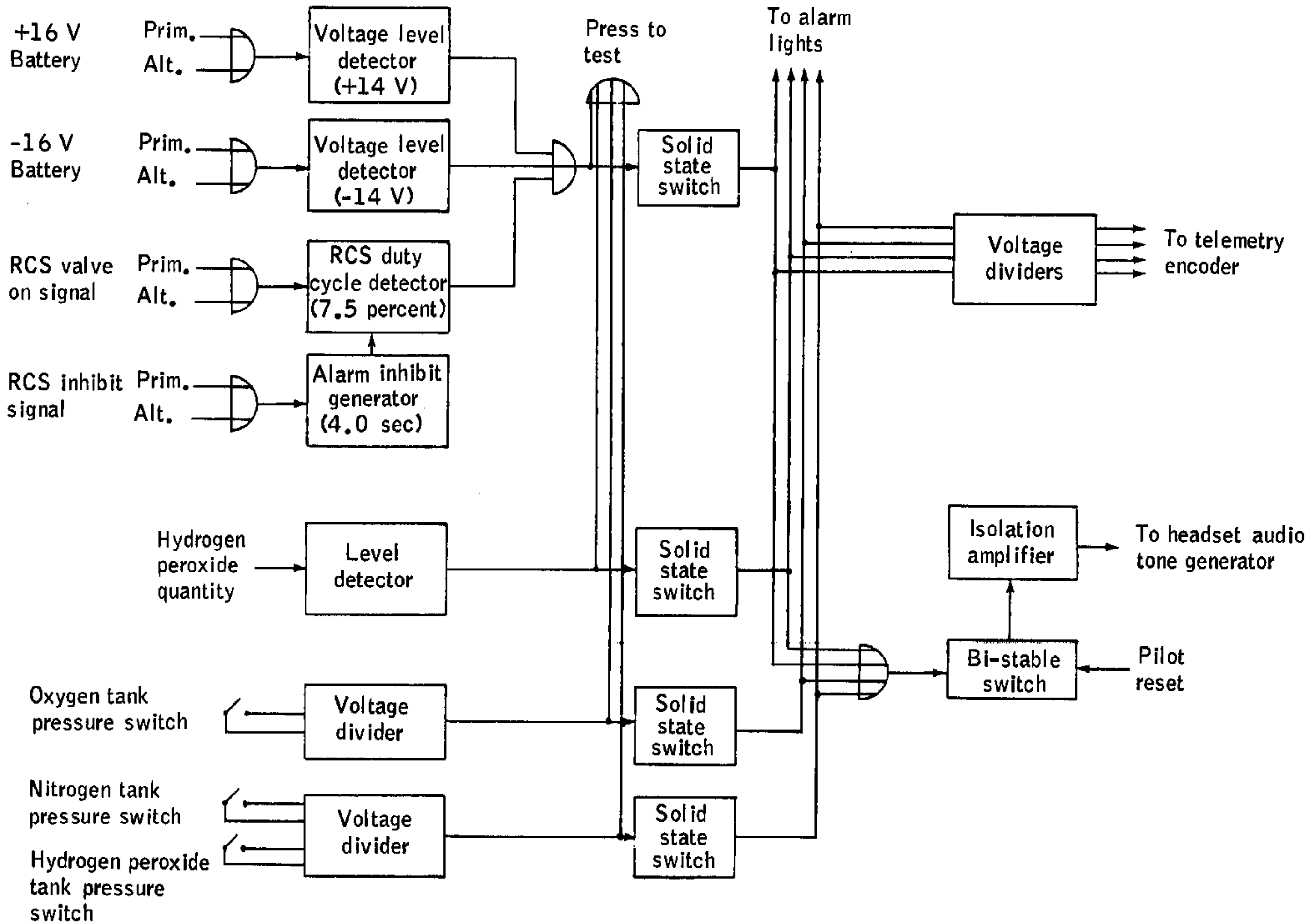

Figure 8.1-14. - Experiment D-12, AMU malfunction detection system functional diagram.

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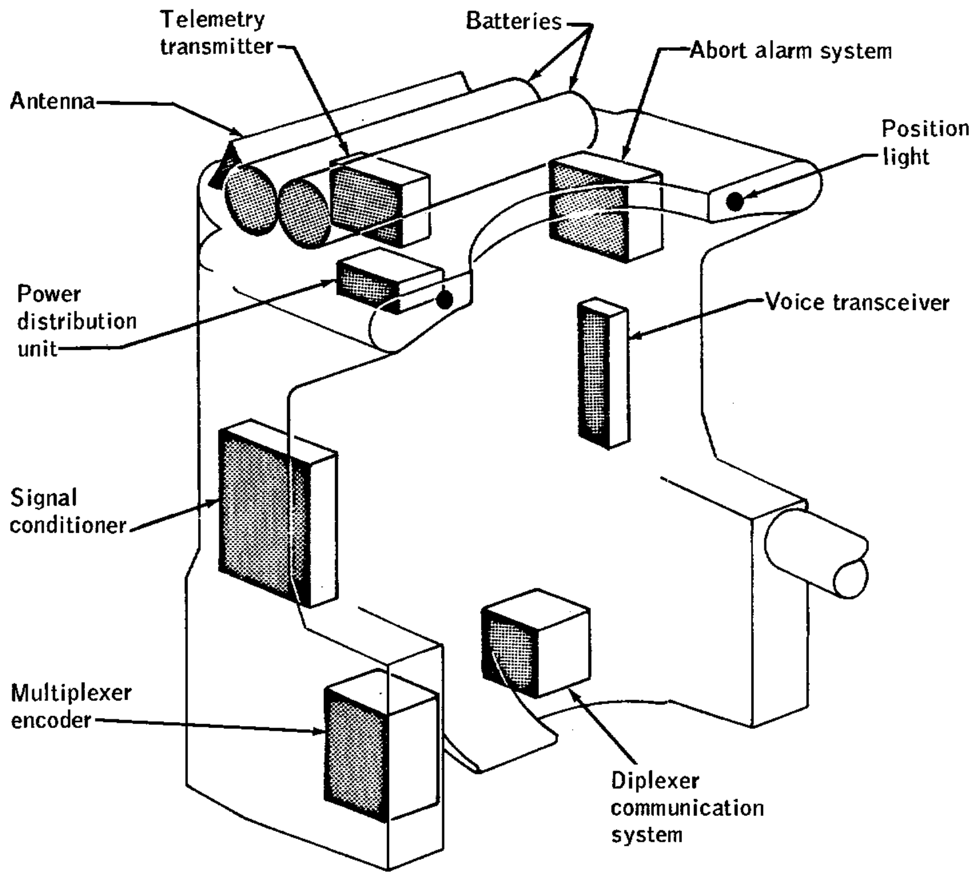

Figure 8.1-15. - Experiment D-12, AMU communications, telemetry, and electrical systems arrangement.

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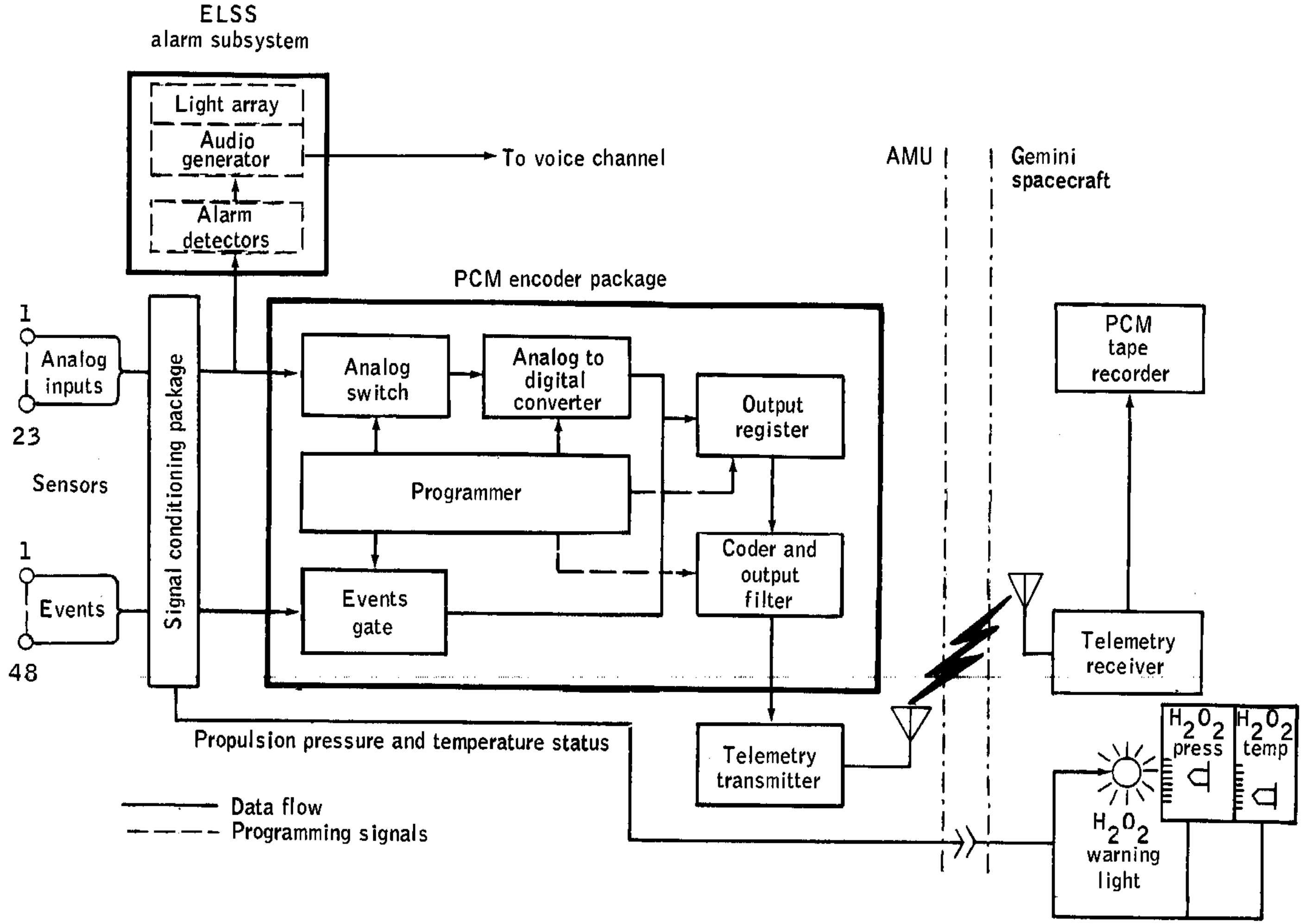

Figure 8.1-16. - Experiment D-12, AMU telemetry system functional schematic.

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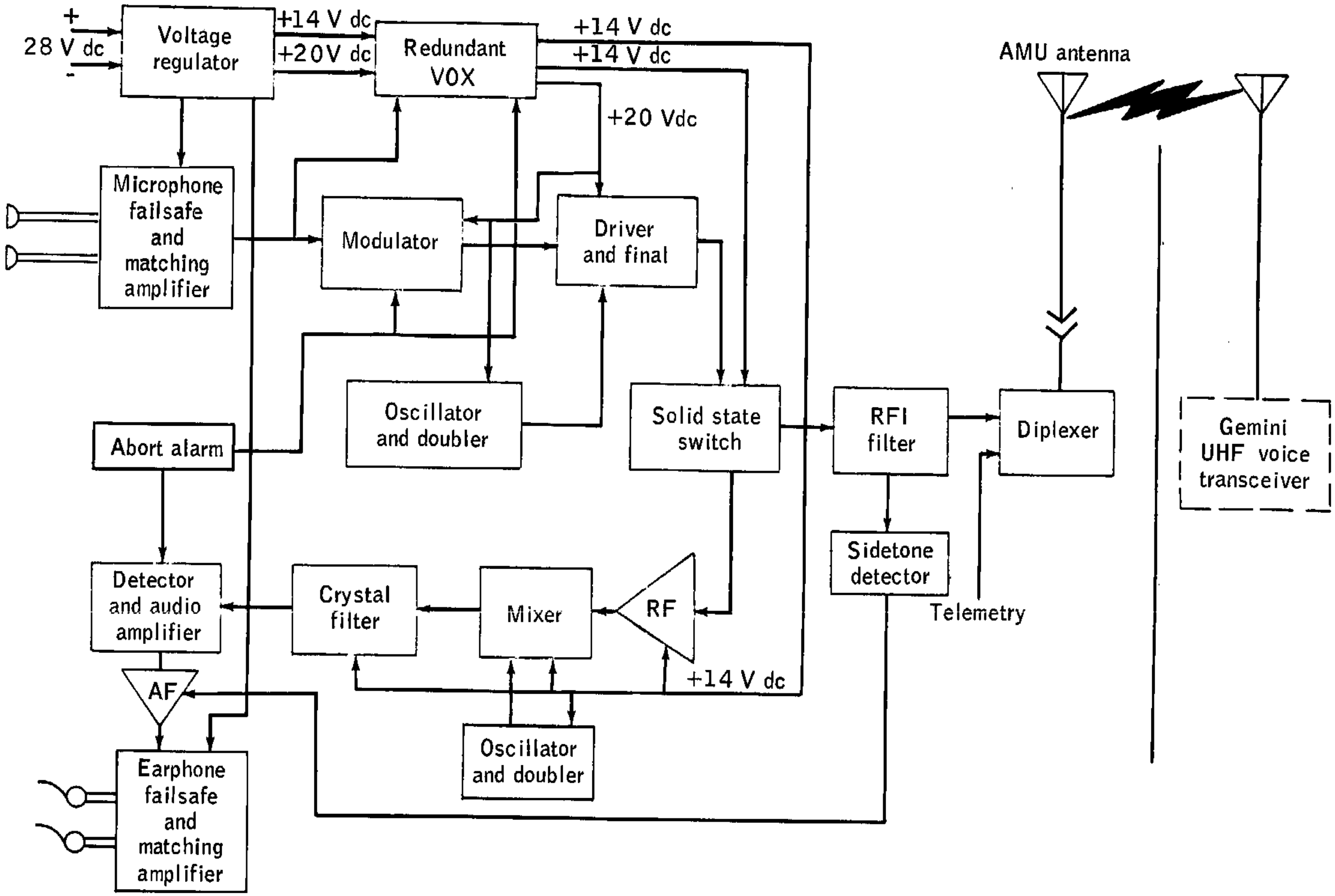

Figure 8.1-17. - Experiment D-12, AMU voice communications system functional diagram.

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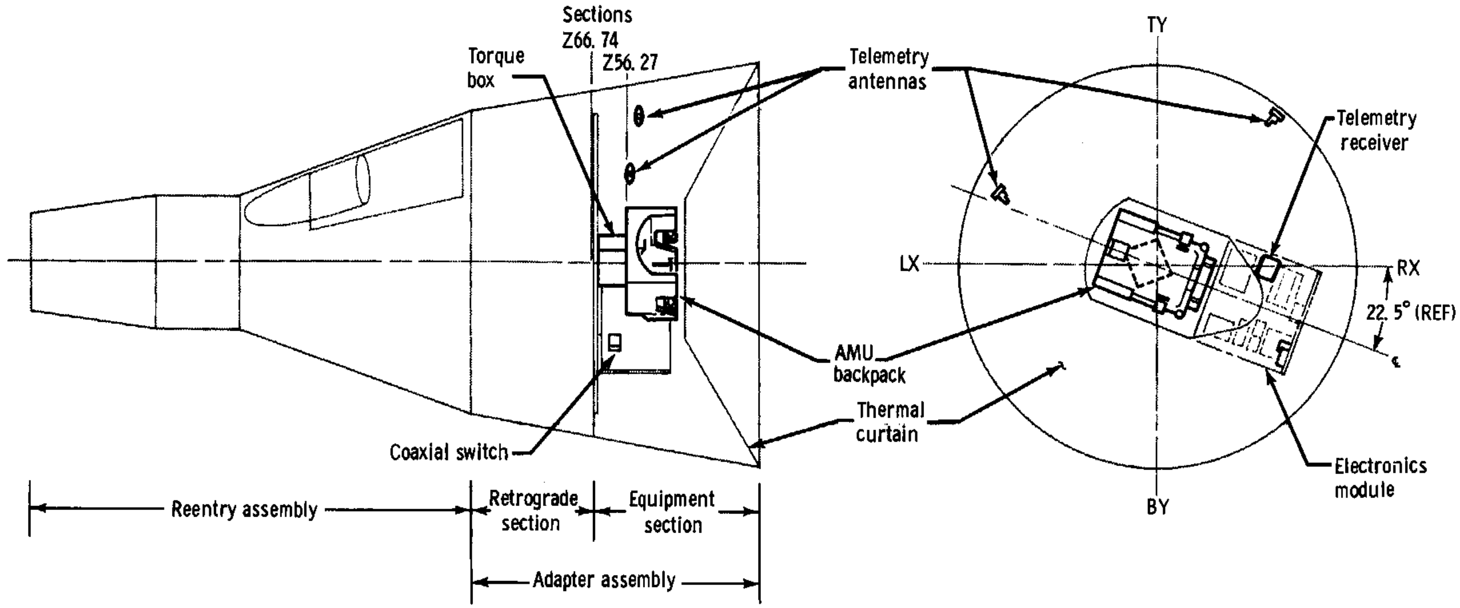

Figure 8.1-18. - Experiment D-12, AMU stowage in adapter assembly.

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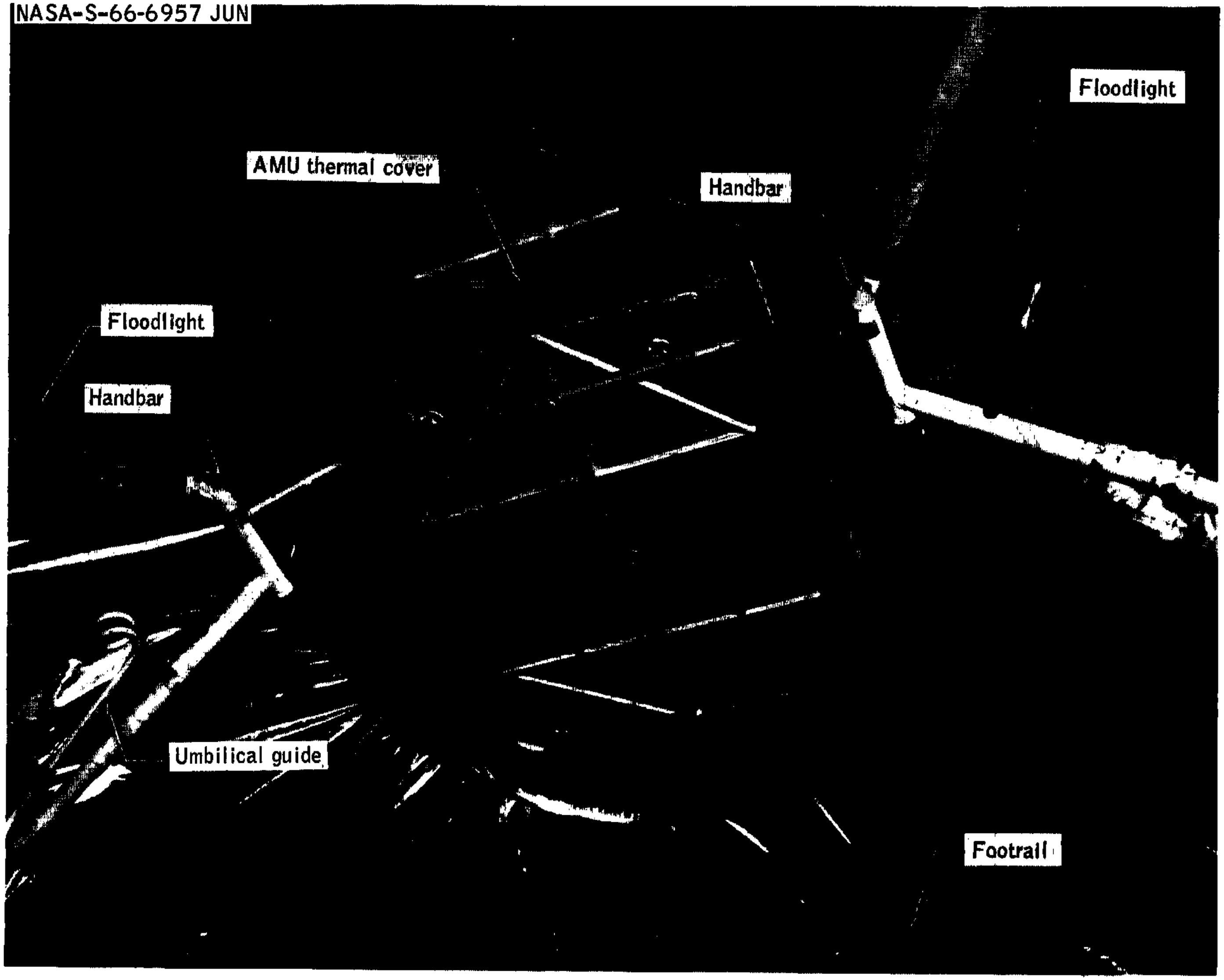

Figure 8.1-19. - Experiment D-12, AMU donning hardware.

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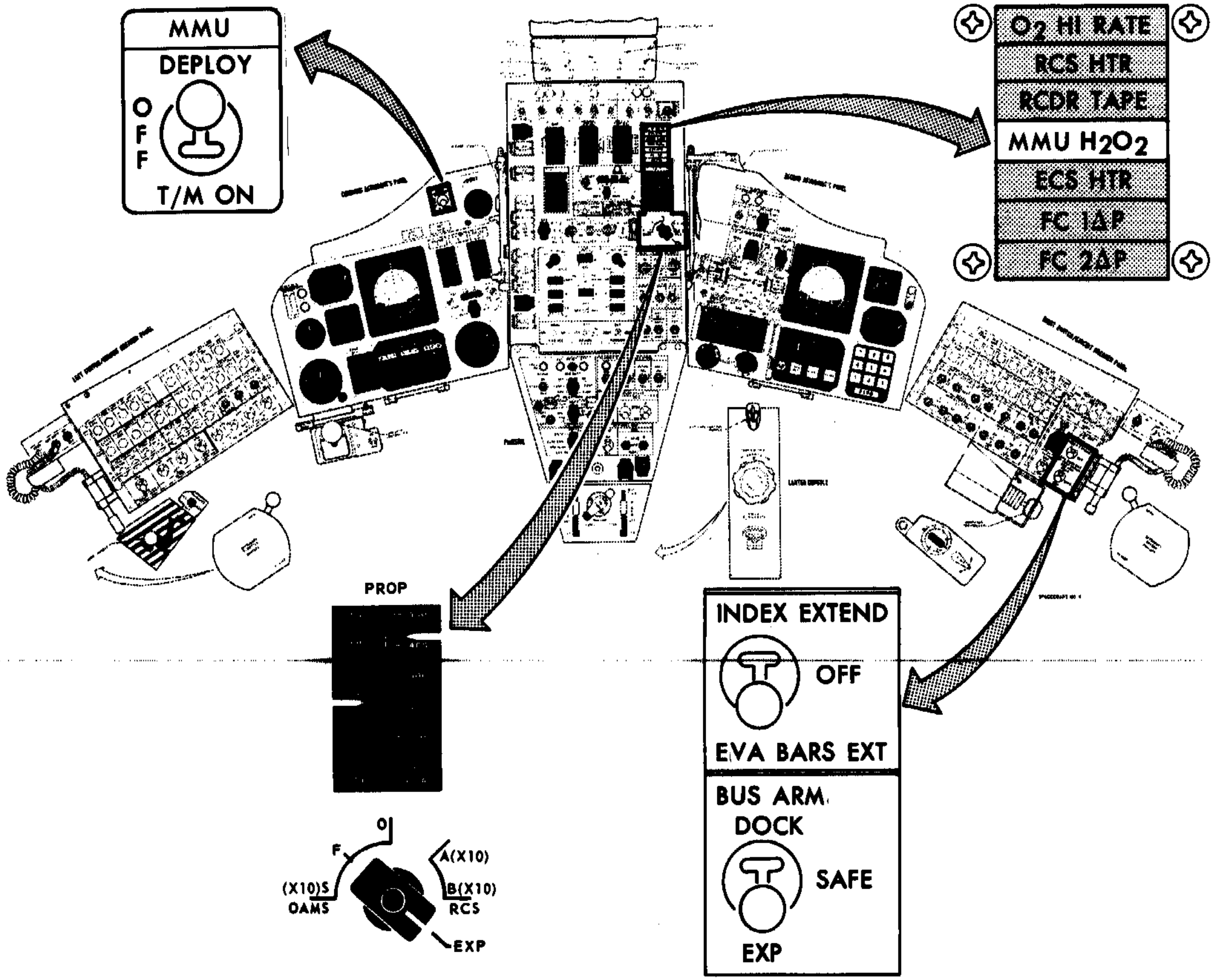

Figure 8.1-20. - Experiment D-12, AMU controls and indicators.

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8.2 EXPERIMENT D-14, UHF/VHF POLARIZATION MEASUREMENTS

8.2.1 Objectives

The objective of Experiment D-14, UHF/VHF Polarization Measurements, was to measure the electron content of the ionosphere below the spacecraft and in particular the electron content inhomogeneities which exist along the orbital path of the spacecraft. Such measurements increase the understanding of the structure of the low ionosphere and its temporal variations.

This experiment was conducted by measuring the degree of Faraday rotation of two continuous wave (cw) signals, one at 133.9 mc and the other at 401.7 mc, transmitted from the spacecraft and received at a station on the ground. A computer program was set up to calculate the electron content from the measurement of the Faraday rotation angle. Two frequencies were used to provide a nonambiguous measurement of angle (401.7 mc) and a vernier measurement of angle (133.9 mc) to permit the desired experiment accuracy.

8.2.2 Equipment

The equipment necessary to conduct Experiment D-14 included both equipment mounted in the spacecraft and equipment in the ground stations. The transmitting system, a block diagram of which is shown in figure 8.2-1, as mounted in the spacecraft, consisted of the following units:

(a) Dual frequency transmitter

(b) Diplexer

(c) Colinear dipole antenna

(d) Antenna boom.

The transmitter and diplexer were mounted in the adapter equipment section and the antenna and antenna boom were mounted in the adapter retrograde section.

Two cw signals were generated by the transmitter shown in figure 8.2-2 and fed into a diplexer where they were combined into a single 50-ohm output. The two signals from the diplexer were fed through coaxial cables to the colinear dipole antenna from which they were radiated with a linear polarization.

The colinear dipole antenna was mounted on an extendable boom assembly, shown in figure 8.2-3, so that it could be extended about 7 feet beyond the outside surface of the spacecraft. The upper element of the antenna was hinged in two places (fig. 8.2-4) so that it could be entirely confined within the spacecraft prior to and during launch.

Two ground receiving stations, a block diagram of which is shown in figure 8.2-5, were used in Experiment D-14. One of these stations was located at Kauai, Hawaii, and the other station was located at Antigua, West Indies. The receiving and data recording systems at each of these stations, a composite photograph of which is shown in fig- 8.2-6, were composed of the following equipment:

(a) A 28-foot-diameter dish antenna with cross-polarized feed

(b) An antenna pedestal and tower

(c) A 133.9-mc dual channel receiver

(d) A 401.7-mc dual channel receiver

(e) A 14-channel magnetic tape recorder

(f) A 4-channel chart recorder

(g) Two X-Y oscilloscopes

(h) Two 35-mm frame-by-frame cameras

(i) A camera-control unit

(j) An RF-attenuator control unit

(k) An antenna-control unit.

Signals that were transmitted from the spacecraft were intercepted by the ground antenna and separated into their vertical and horizontal components by the antenna feed system. The two components of each frequency were fed into a dual-channel receiver tuned to that frequency. In the receivers, the signal components were amplified and heterodyned to an IF of 120 kc with little or no phase shift between components of the same carrier frequency. The 120-kc vertical and horizontal signal components were fed into X-Y oscilloscopes, one for each RF, to produce a line display whose angle of rotation from 90 degrees, for a vertically polarized signal, was the Faraday rotation angle. These camera displays were recorded by the 35-mm cameras at a rate of about 2 frames per second. At the same time the 120-kc signals were fed into the X-Y oscilloscopes they were also fed into the tape recorder along with time signals. The 120-kc IF signals were also detected and envelopes of the IF signals were recorded on the four-channel paper recorder. In addition to recording data, the chart recorder also provided a visual monitor for the operators.

8.2.3 Flight Procedures

In order to conduct Experiment D-14 it was necessary for the crew to perform several operations. The first operation was to activate the antenna boom mechanism and deploy the antenna. Each time the experiment was performed over a ground station it was necessary for the crew to turn on the dual-frequency transmitter and control the spacecraft attitude in such a manner as to point the colinear dipole antenna toward the center of the earth.

Because of the physical location of the antenna on the spacecraft, a roll of approximately 158 degrees to the right and a nose-down pitch of 17 degrees was required to position the antenna for the experiment. In order to obtain the best antenna patterns with the lowest cross polarization component, a 90-degree yaw with the nose of the spacecraft away from the ground station was also required.

After the spacecraft passed beyond the line of slight or over the radio horizon of the ground station, the crew was required to turn off the dual-frequency transmitter. The last operation required of the crew for Experiment D-14 was to retract the antenna prior to reentry of the spacecraft.

8.2.4 Results

During the Gemini IX-A mission, Experiment D-14 was performed six times. Experiment data were received and recorded one time at Antigua during revolution 18 and five times at Hawaii during revolutions 17, 18, 19, 20, and 21. The spacecraft attitude was controlled during each of these operations by using the platform and attitude indicator. On the first pass over Hawaii, the receiving antenna became locked in the gimbal limits by tracking-radar positioning signals. This was corrected halfway through the pass, resulting in good data.

Three additional experiment runs were scheduled for the Hawaii site, but the colinear dipole antenna was broken during EVA and the experiment was terminated. The pilot reported that the antenna break occurred about eight inches from the antenna base leaving a jagged edge. The antenna was constructed of 6061T aluminum tubing. Tests of this material at temperatures of -320° F do not indicate a brittle state for this material.

During each of the passes for Experiment D-14, the signals received were 20 to 30 db lower than had been anticipated. As a result, the chart recordings do not appear, after a cursory examination, to contain much useful information.

The photographs of the Faraday rotation oscilloscope displays have not as yet been processed. The magnetic tape recordings of the signals used to produce the scope displays were played back into an X-Y oscilloscope and the Faraday rotation angles of each experiment pass were displayed. The direction of rotation and reversals in direction of rotation of each of the displays were as predicted from the computation of the magnetic field parameters for the receiving sites.

Although the signal levels were low, it would appear that useful results will be obtained from the data recorded for Experiment D-14.

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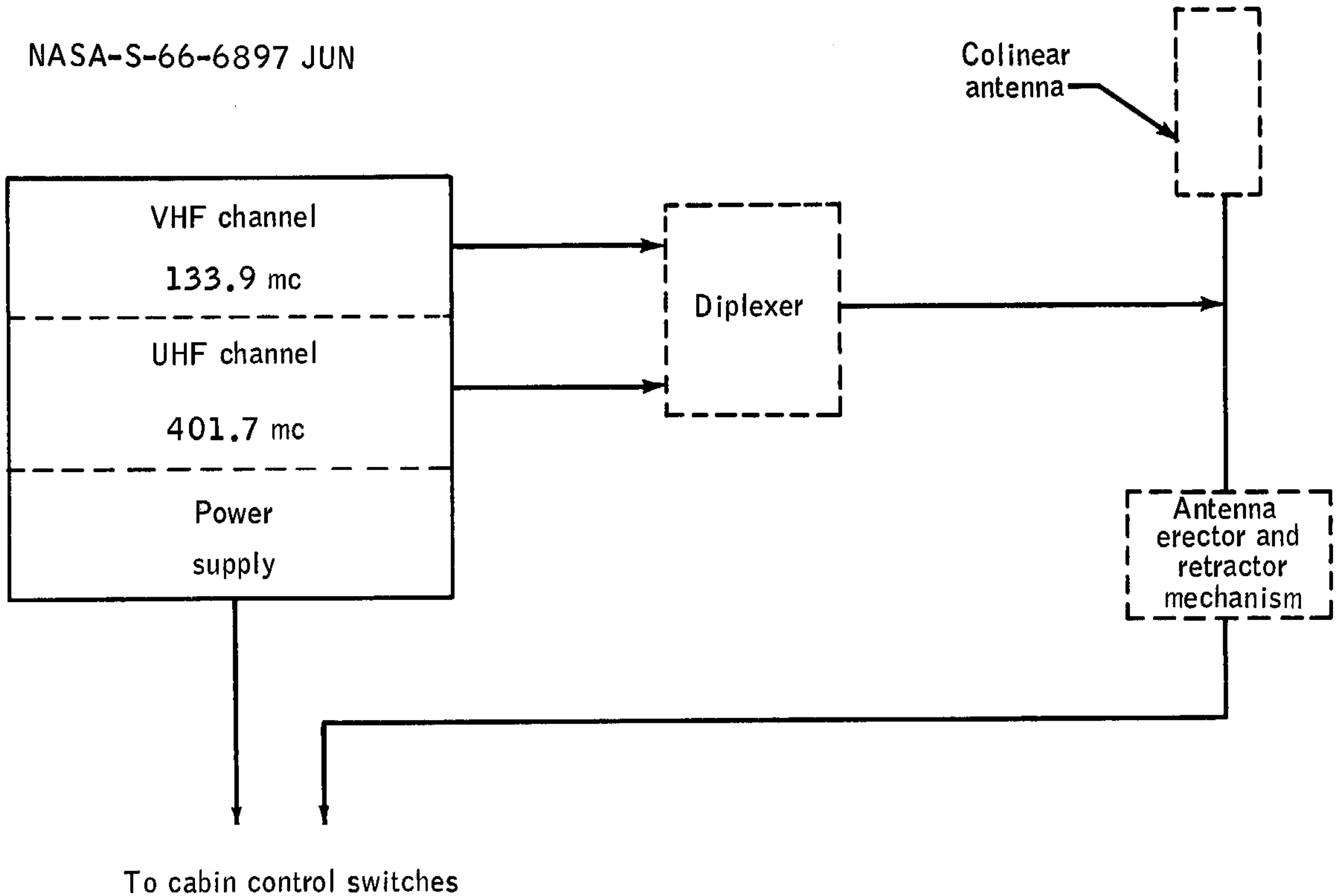

Figure 8.2-1. - Experiment D-14, onboard Gemini transmitter block diagram.

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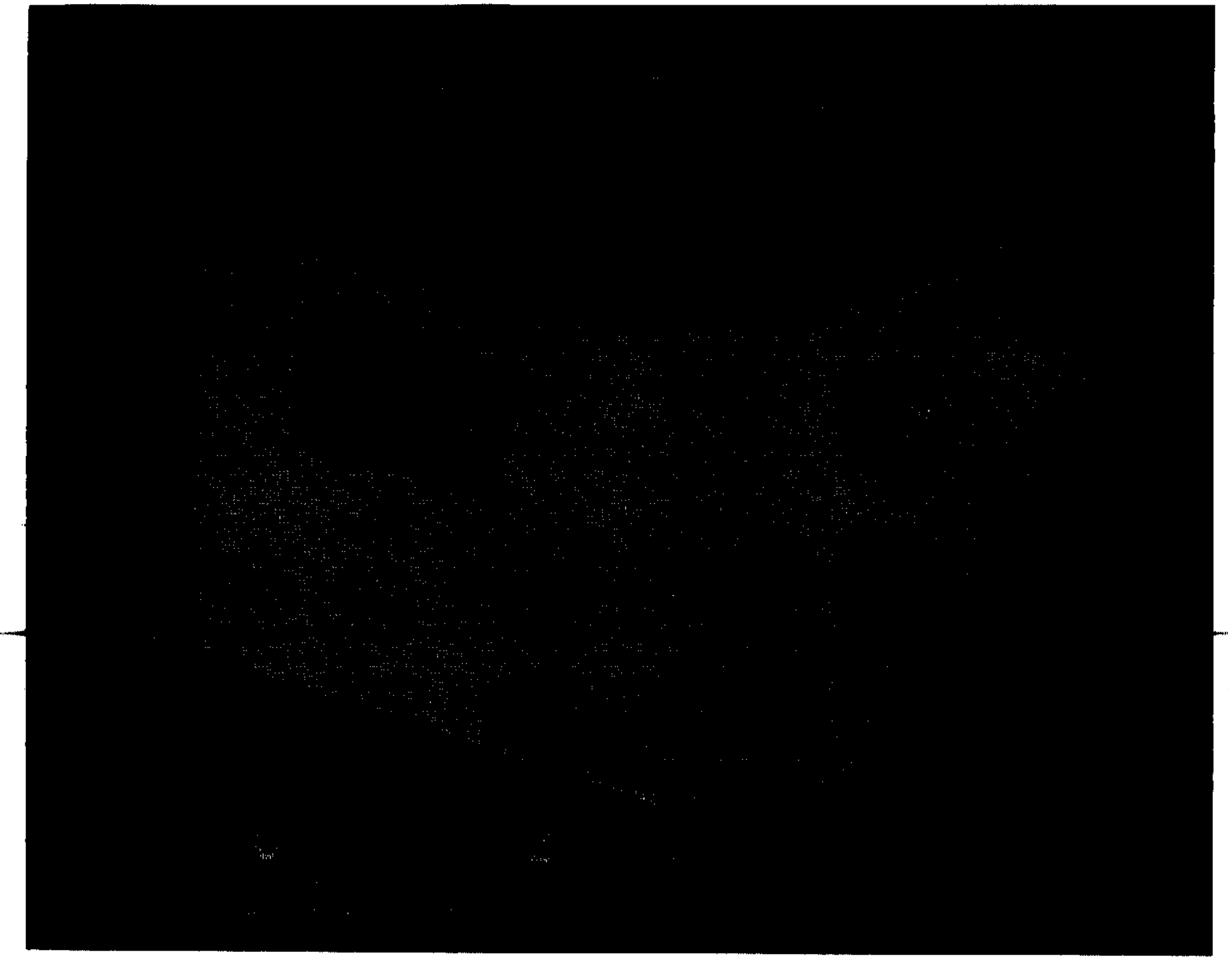

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Figure 8.2-2. - Experiment D-14, onboard Gemini transmitter.

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Figure 8.2-3. - Experiment D-14, colinear dipole antenna, extended position.

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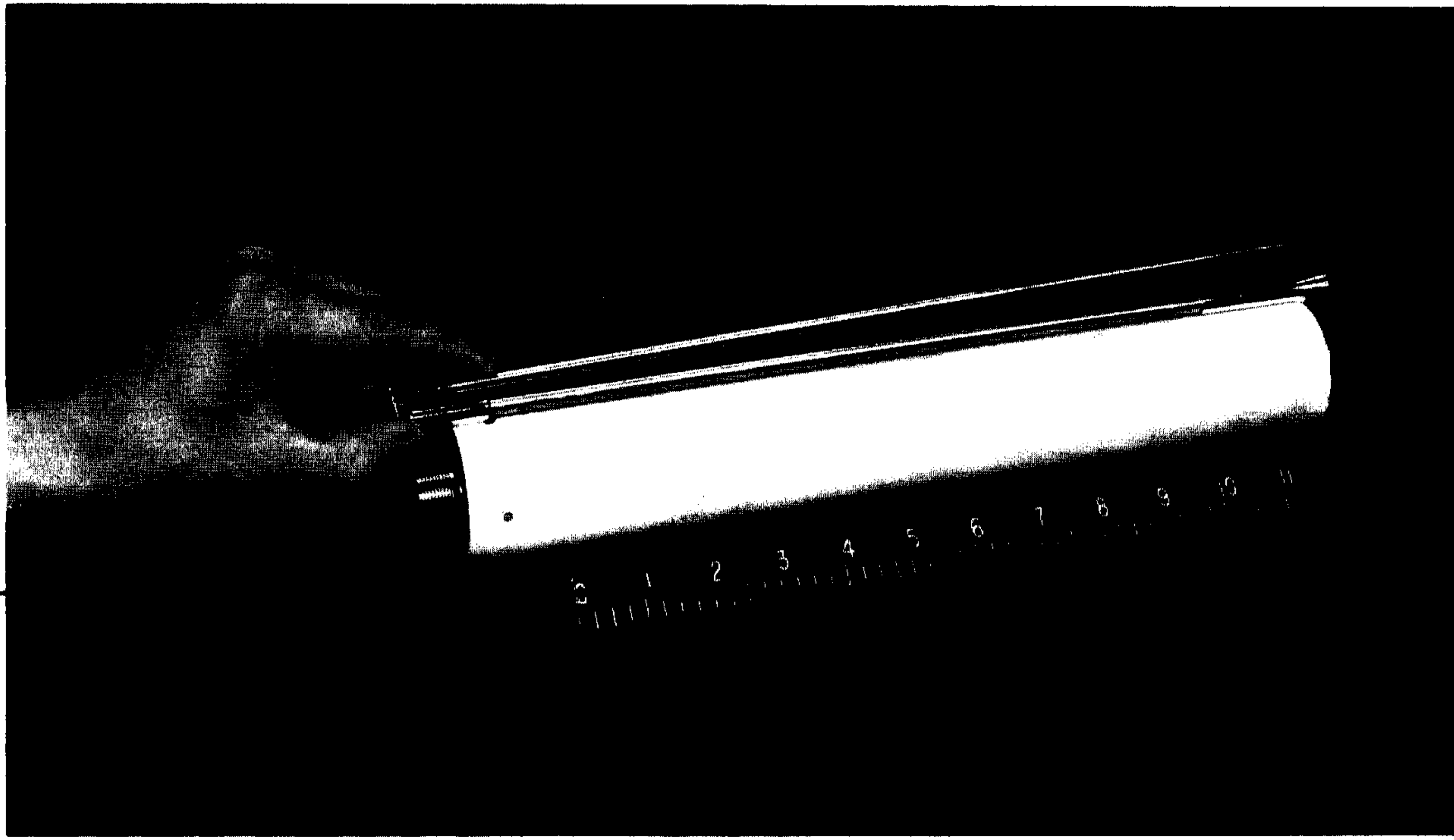

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Figure 8.2-4. - Experiment D-14, colinear dipole antenna, folded position.

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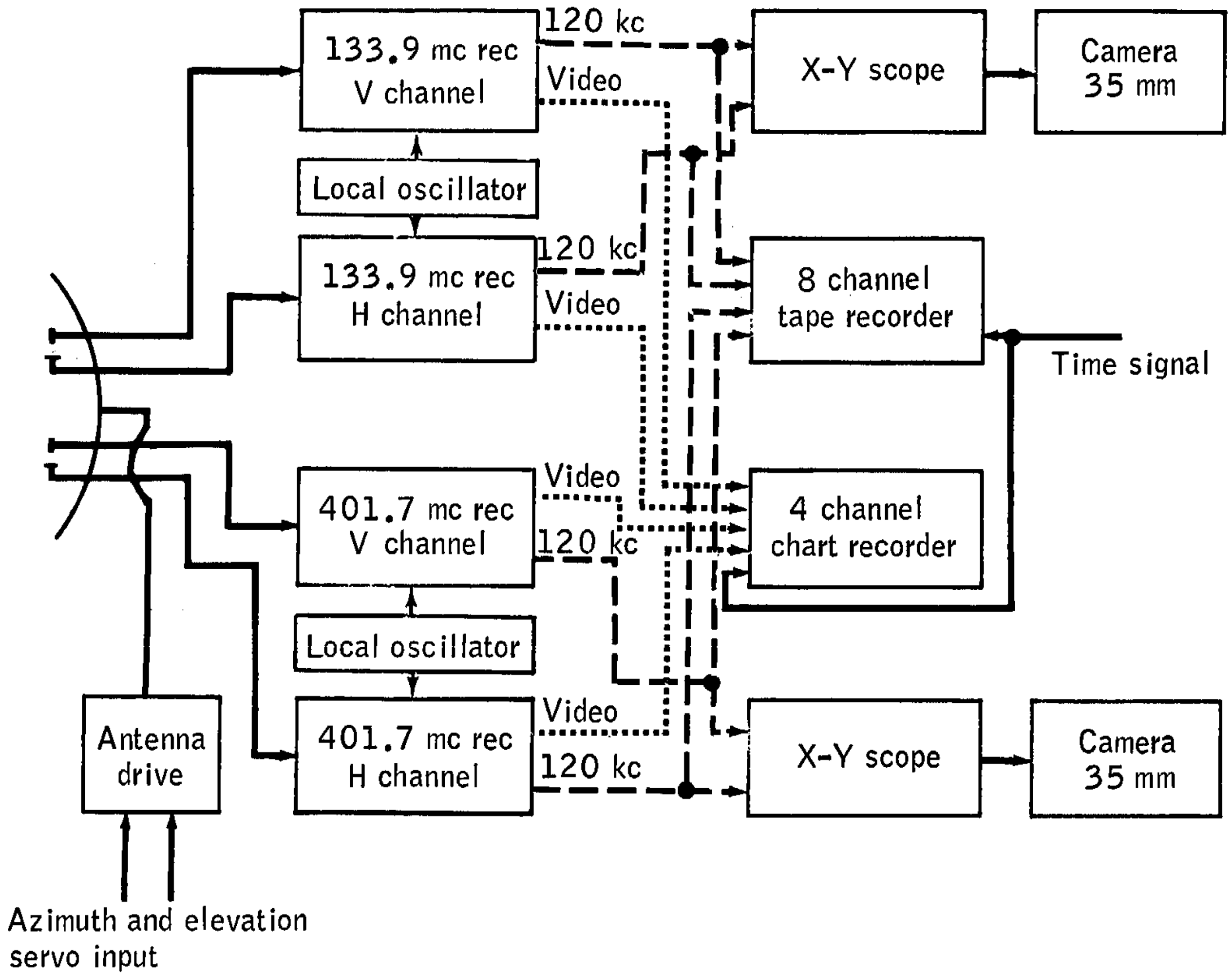

Figure 8.2-5. - Experiment D-14, functional diagram of ground receiving equipment.

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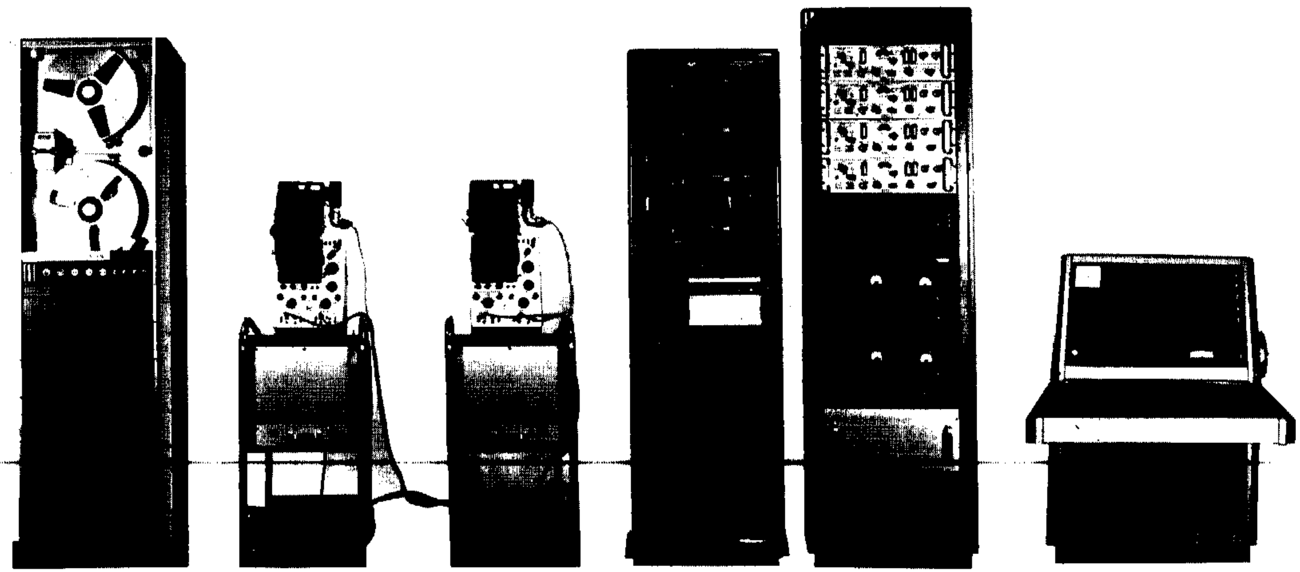

Figure 8.2-6. - Experiment D-14, ground receiving equipment.

8.3 EXPERIMENT M-5, BIO-ASSAYS OF BODY FLUIDS

8.3.1 Objectives

The objectives for Experiment M-5, Bio-Assays of Body Fluids, were to determine the flight crew reaction to the stress requirements of space by the use of hormonic assays.

8.3.2 Equipment

During the mission, urine was sampled with a volume and sample measuring device which consisted of a valve with a tritiated water injector, a mixing bag, and 24 sample bags. Figure 8.3-1 shows the sampling equipment used aboard the spacecraft.

8.3.3 Procedures

Prior to and following the Gemini IX-A mission, at least two plasma and timed urine samples were obtained daily for baseline data. Urine samples were collected during the mission and stored in a preservative for postflight analysis. The crew recorded the time and volume of each sample and indicated whether the tracer pump was passed more than once per sample.

Prior to urination, a precise volume of tritiated water was injected into the valve lines by a positive displacement pump located within the valve. Urine washed the tritium into a mixing bag. A sample of the urine-tritium mixture was then transferred through the valve from the mixing bag to the sample bag. The self-sealing sample bag was removed from the valve mechanism and stored for postflight use.

Special handling procedures were required soon after recovery. Aboard the recovery ship, the urine samples were frozen immediately after removal from spacecraft stowage, packed with frozen postflight blood samples in dry ice, and returned to the Manned Spacecraft Center. The total volume of each urination was then determined by measuring the dilution of the tritium isotope.

8.3.4 Results and Conclusions

Preliminary analyses of the preflight and postflight blood and urine samples are 95 percent complete. The major analyses are the hormones aldosterone, epinephrine, and norepinephrine, and the antidiuretic hormone.

All 16 urine samples tritiated in flight have been assayed to determine the total voided volume of each micturition. The calculated volume determined by the assays have an extreme range of 315 to 12 326 milliliters. Of the 16 measured samples, 13 samples exceeded the physiological limits of bladder capacity, and 12 exceeded the physical capacity of the collection bags of the sampling system. The valve and pump of the sampling system have been rechecked and found to be in proper working condition. It must be concluded that incorrect volume measurements resulted from inadequate mixing of the tritiated water and the urine in the collection bag.

In this flight, as in previous flights, a distinct retention of electrolytes and water was observed immediately postflight. This retention is consistent with the inflight weight losses and postflight recoveries of the crew.

No conclusions may be drawn about the mechanism of the water loss without correct inflight sample volumes. The chemical analyses of the inflight urine samples are still in process and results are not available for this report.

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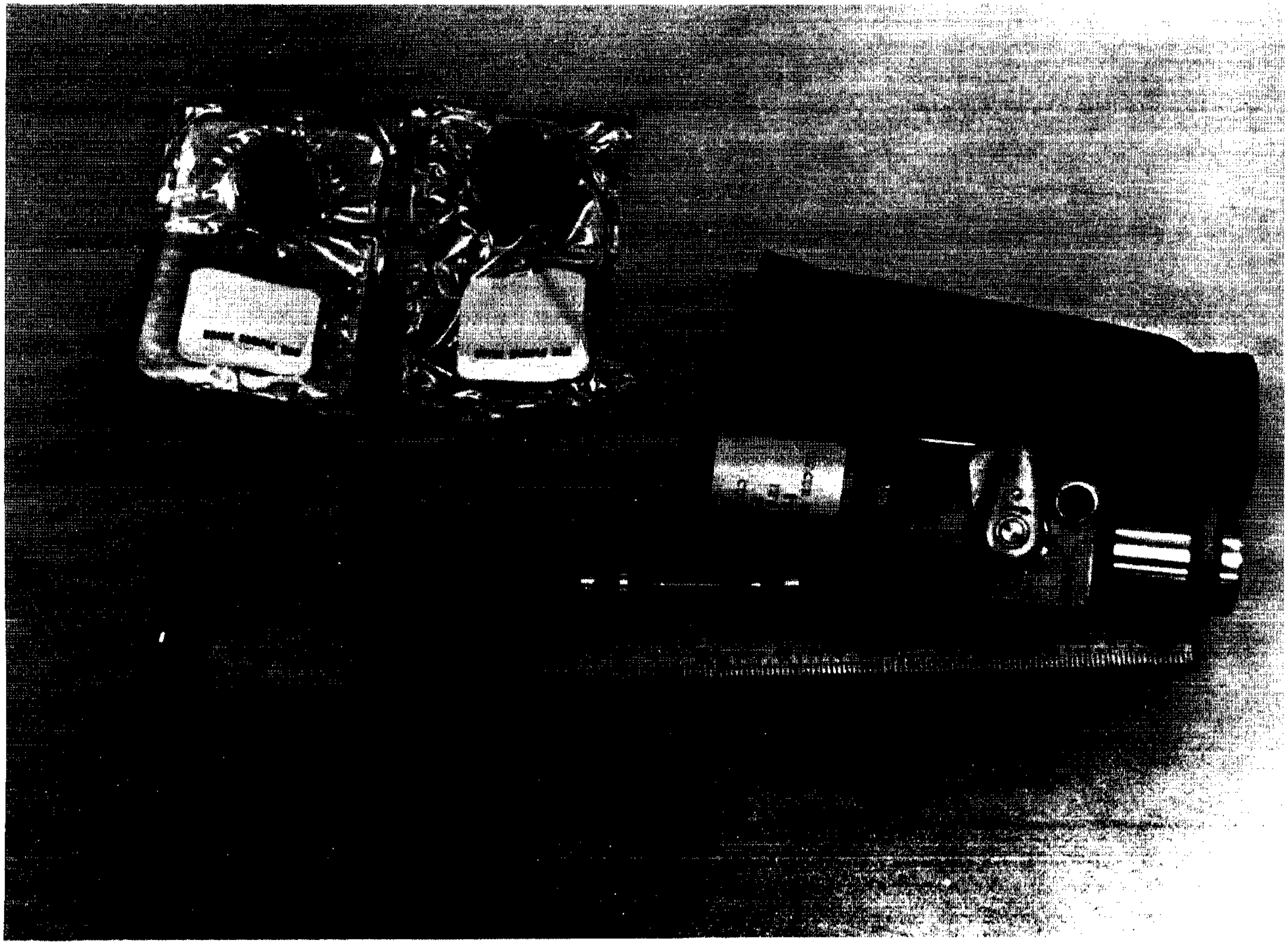

Figure 8.3-1. - Experiment M-5, onboard equipment.

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8.4 EXPERIMENT S-1, ZODIACAL LIGHT PHOTOGRAPHY

8.4.1 Objectives

The purpose of Experiment S-1, Zodiacal Light Photography, was to obtain 30-second exposures at f/1 of several objects of astronomical interest. These include the airglow (viewed in profile from above), the zodiacal light, and the Milky Way.

8.4.2 Equipment

The camera was designed to view a wide-angle field (approximately 50 by 130 degrees). Mechanically, it was the same kind of camera as the one flown on the Gemini V and Gemini VIII missions. The exposure sequence was automatic and alternated 30-second exposures with 10-second off periods. During the off period, thrusters could be fired without exposing the film. The film was 35-mm black and white with a speed of 400 ASA.

8.4.3 Procedure

The original flight plan called for the camera to be handheld during extravehicular activity (EVA) and to be used on the nightside pass just before ingress. However, because of the visor fogging difficulties, the extravehicular operation of the camera was abandoned. Subsequently, the crew carried out the experiment from inside the spacecraft, photographing through the pilot's window. The pilot held the camera in the window during the exposures, sighting past the camera and directing the command pilot to maneuver to the appropriate position. The pilot turned the camera off between successive exposures. The astronomical objects were not in the command pilot's view, and his role was to null the spacecraft rates.

8.4.4 Results

The procedure adopted by the crew resulted in obtaining 17 very good photographs. They are tabulated in the following table:

Exposure number	Orientation	Object
1	North	Horizon airglow
2, 3	West	Horizon airglow
4, 5	South	Horizon airglow and aurora
6, 7	East	Horizon airglow
8, 9, 10, 11	Northeast	Milky Way
12, 13, 14	East	Horizon airglow
15, 16, 17	East	Airglow, zodiacal light, twilight

Exposure 9 is shown in figure 8.4-1. This figure is a copy of a photograph of the Milky Way with Cygrus in the center. The airglow layer shows to the right of the spacecraft. The bright spot at the upper right was caused by moonlight. Exposure 15 is shown in figure 8.4-2. This figure is a copy of an overexposed print to emphasize the zodiacal light, airglow, and stars.

8.4.5 Conclusions

The negatives are being studied using an isodensitracer to produce intensity isophotes. The following data will be obtained:

(a) Intensity distribution of the zodiacal light, both morning and evening

(b) The height and intensity of the airglow at various geographic positions

(c) Intensity distribution of the Milky Way in the region of the sky near Cygnus.

In addition to the above, a previously unreported phenomenon was discovered. This phenomenon appears as an upward extension of the normal 90-kilometer airglow layer. The extension is in the form of wisps or plumes about 5 degrees wide and extending upward about 5 degrees.

The plumes appear in the east and are almost certainly not due to aurora. The phenomenon is believed to be truly in the sky, but a number of tests will be made to determine this point.

The experiment is considered an unqualified success, and the crew must be given credit for making an excellent program for the experiment after the EVA performance became impossible.

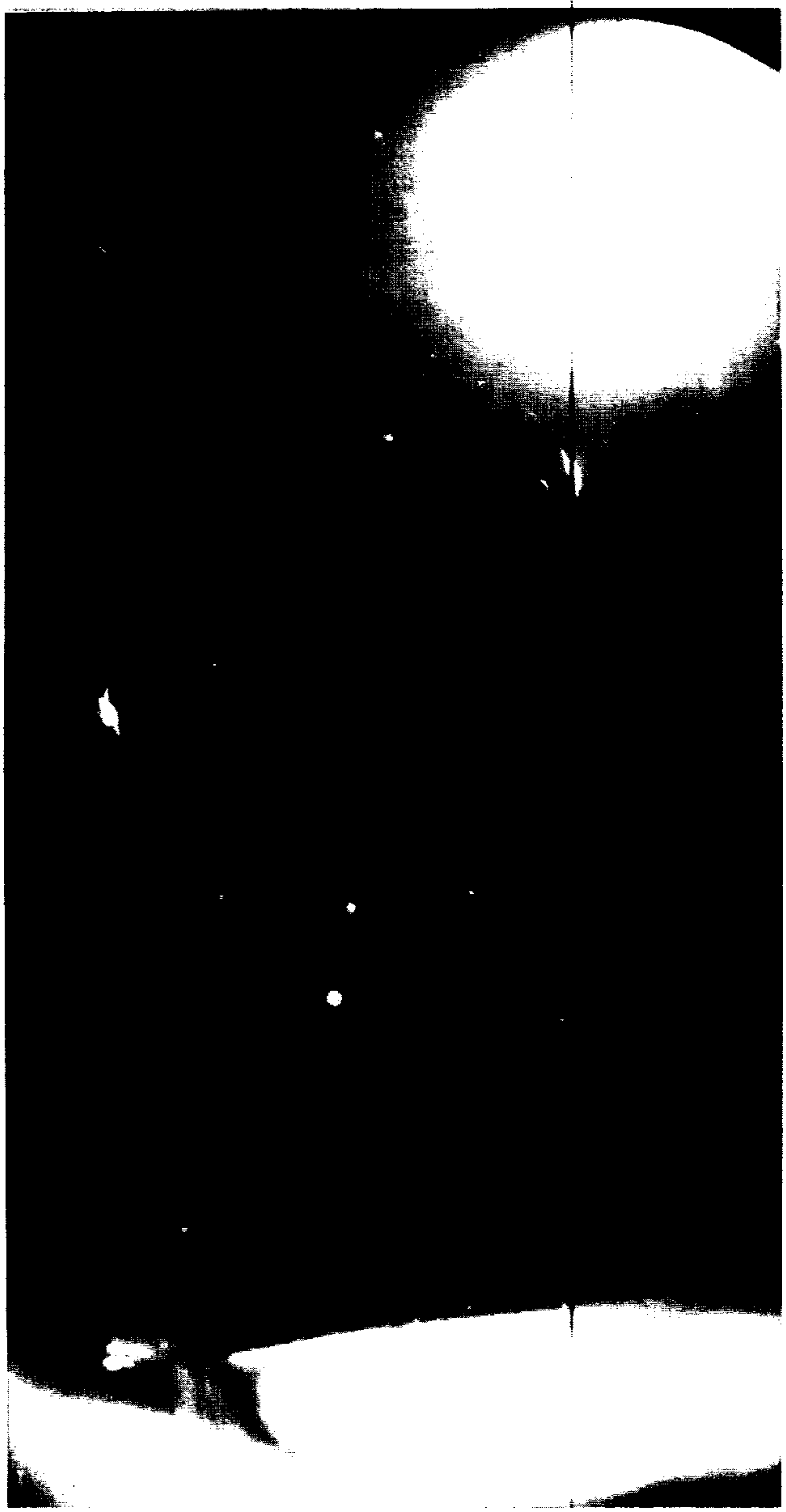

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Figure 8.4-1. - Experiment S-1, photograph of Milky Way.

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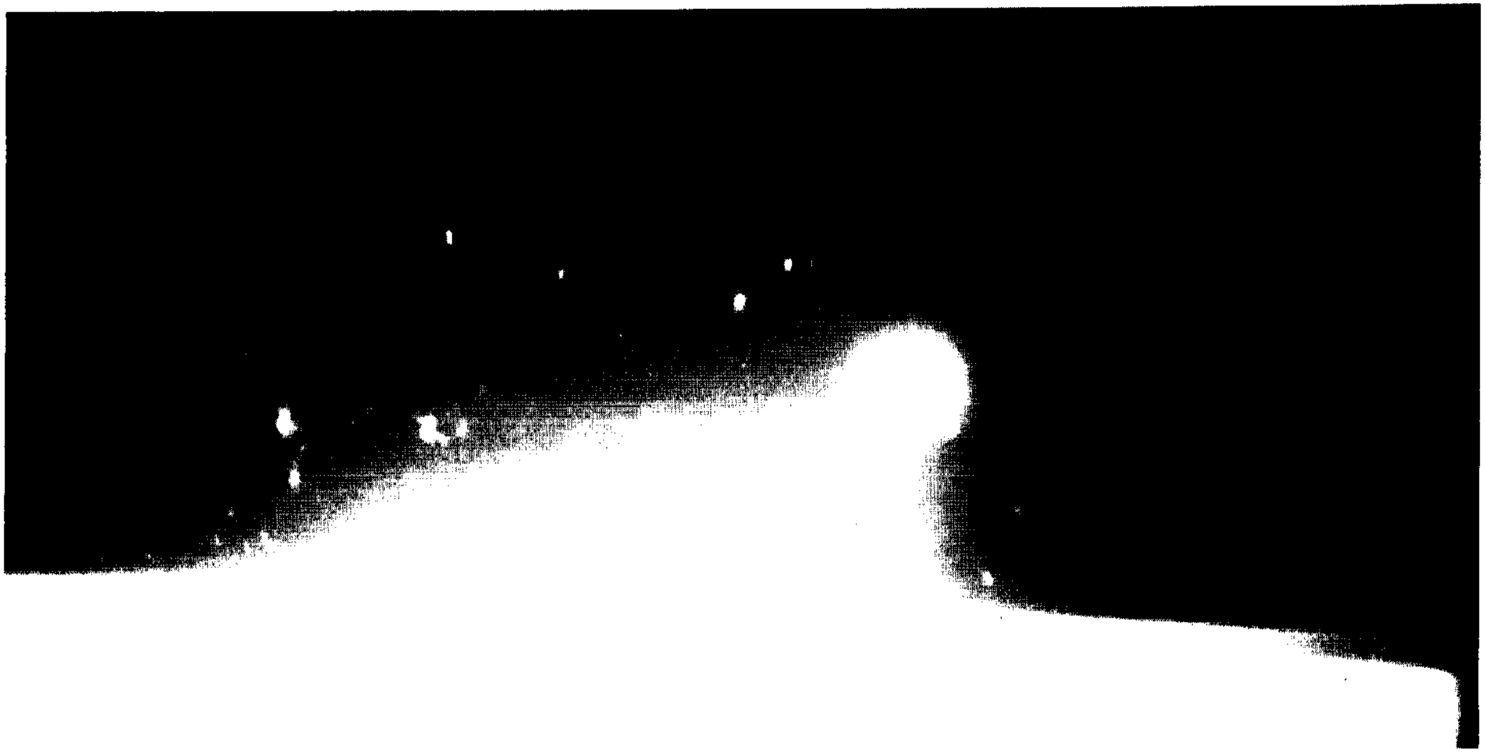

Figure 8.4-2. - Experiment S-1, photograph of zodiacal light.

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8.5 EXPERIMENT S-10, AGENA MICROMETEORITE COLLECTION

8.5.1 Objective

The basic scientific objective of Experiment S-10, Agena Micrometeorite Collection, was to study the micrometeorite content of the upper atmosphere and near-earth space environment. This was to be accomplished by (1) exposing polished metal and plastic surfaces to the particle flux for later study of the resulting impact craters, (2) exposing highly polished sections of meteorite material to the particle flux for obtaining direct measurement of meteor erosion rates, (3) exposing optically polished glass surfaces to the particle flux for determining the deterioration of optical surface properties, (4) exposing thin films to the particle flux to observe thin-film penetration, and (5) exposing extremely clean surfaces to the particle environment in an attempt to collect ultra-small particles.

8.5.2 Equipment

The hardware configuration consisted of an aluminum structure designed to provide a mounting platform for the polished plates and collection surfaces. The device was interfaced with the Augmented Target Docking Adapter (ATDA) by a mounting plate which allowed detachment of the experiment hardware from the vehicle. Cratering samples were installed on the outside surface of the aluminum structure. During powered flight and the insertion phase of the mission, these external surfaces were protected from direct impact of airborne particles by a fairing which directed airflow over the mounting. During extravehicular activity the pilot was to have removed this fairing cover. Figure 8.5-1 is a diagram of the S-10 hardware in both the closed and open positions, with a photograph of the experiment hardware location taken during the Gemini IX-A rendezvous with the ATDA.

8.5.3 Procedures

During extravehicular activities (EVA) and while the spacecraft was docked with the ATDA, the extravehicular pilot was to have activated the S-10 micrometeorite experiment hardware, thereby exposing the inner collection surfaces to the outside environment. The flight plan scheduled this to occur at 21:00 hours ground elapsed time (g.e.t.). The procedures listed in the flight plan were as follows:

(a) Remove forward fairing and discard

(b) Attach wrist tether to collector handle

(c) Remove collector from bracket

(d) Attach collector to Velcro (do not touch prepared surfaces)

(e) Detach wrist tether

(f) Record g.e.t. of collection activation

(g) Retrieve S-10 from ATDA after 80-foot out-of-plane rendezvous.

8.5.4 Results and Conclusions

Extravehicular activities were postponed until the third day of the mission. At this time the Gemini spacecraft was not near the ATDA, and retrieval of the material specimens within the hardware was not possible. Consequently, data were not collected.

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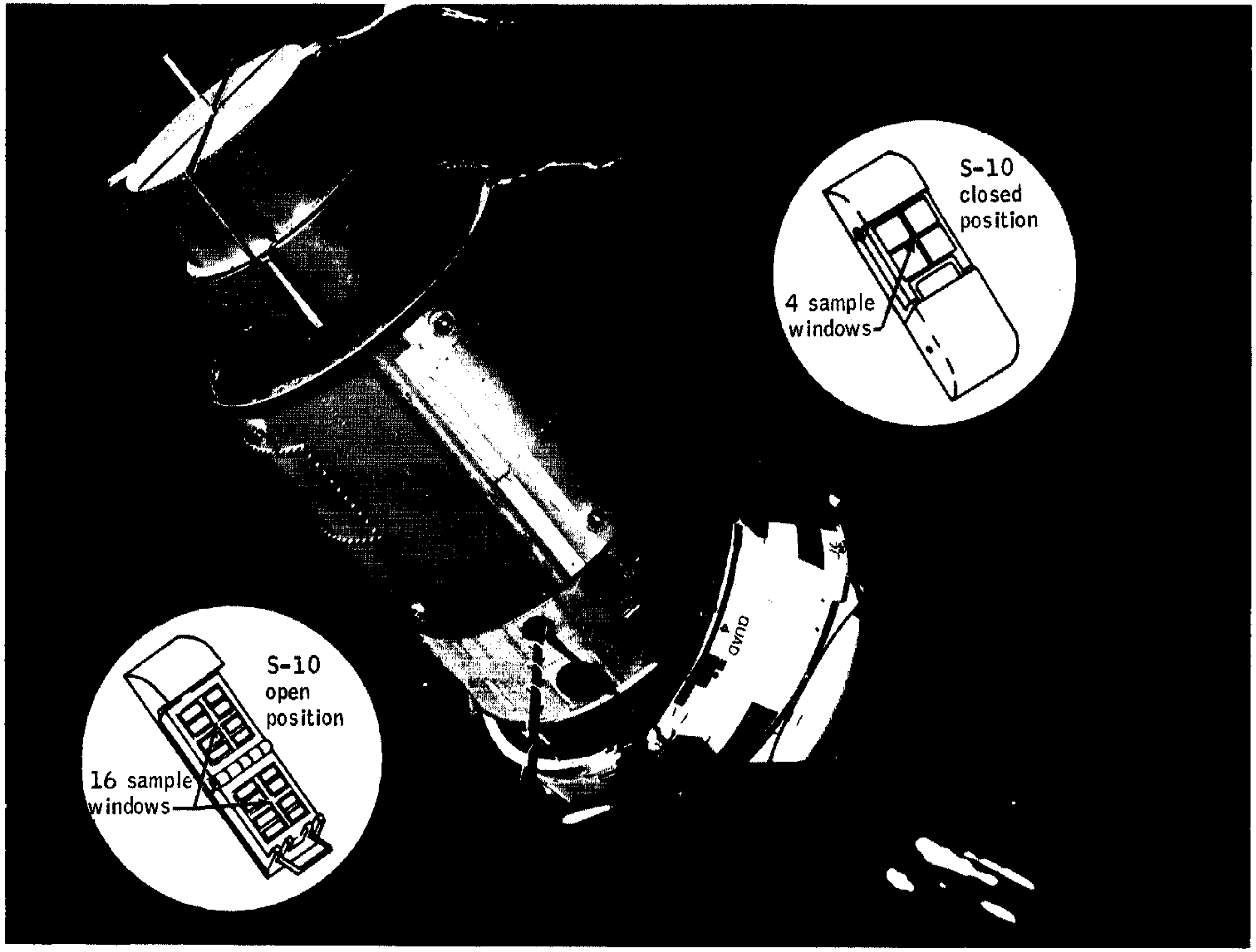

Figure 8.5-1. - Experiment S-10, sample collection device.

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8.6 EXPERIMENT S-11, AIRGLOW HORIZON PHOTOGRAPHY

8.6.1 Objective

The objective of Experiment S-11, Airglow Horizon Photography, was to obtain data on a global scale for the study of the airglows in the upper atmosphere.

The experiment was designed to photograph the night airglow layer which is visible as a bright band lying above the nighttime terrestrial horizon. However, some photographs of the twilight horizon were taken to record the day airglow layer. The camera was optically filtered to photograph airglows in visible bands centered at 5577Å and 5893Å, where prominent airglow emissions occur due to atomic oxygen and atomic sodium, respectively. Horizon photographs taken during the nightside recorded local intensities and altitudes of the emissions.

8.6.2 Equipment

The following equipment was used:

(a) A 70-mm general-purpose camera with a f/0.95 lens including a film magazine containing local-plane optical filters and black and white high-sensitivity film

(b) Illuminated camera sight

(c) Adjustable camera bracket

(d) Camera lens filter.

8.6.3 Procedure

The camera was mounted to the pilot's window by means of a bracket which was adjustable in pitch. The line of sight of the camera was perpendicular to the window. During a nightside pass, the pilot used the bracket and illuminated sight to point the camera at the airglow layer.

Photographs were taken in azimuths East, North, West, and South. For each of these directions the pilot took one time exposure with the camera lens filter on in order to optically isolate the 5577Å and 5893Å visible wavelengths. Additionally, at each of the four azimuths, two-second and five-second exposures were taken with the lens filter removed.

Fifty percent of the camera frame then admitted light of 5800Å wavelength and longer, while the other 50 percent admitted the entire visible wavelength spectrum except for a narrow band centered at the yellow sodium light at 5893Å. This same procedure was repeated for all four directions. When the vehicle approached sunrise, the pilot took 5-second and 10-second exposures with the lens filter on to photograph the day airglow layer.

8.6.4 Results

Forty-four airglow photographs were obtained, including three of the day airglow. Nearly all are of excellent quality, indicating a stable camera during the exposure activation times. Some 20-second exposures were blurred but contain useful data. Many photographs show star images to the sixth magnitude.

Typical results are shown in figures 8.6-1 through 8.6-3 which are the 20-second, 5-second, and 2-second time exposures taken in the easterly direction. In figure 8.6-1 the strip to the left of the central vertical line is recorded in the sodium wavelength band centered at 5893Å with a 300Å half-width. The strip to the right of the central dividing line is a 270Å half-width band centered at the oxygen green-emission band at 5577Å. Figures 8.6-2 and 8.6-3 are 5-second and 2-second exposures made with the camera lens filter removed.

8.6.5 Conclusions

The photographs show clear variations in altitude and intensity of the night airglow layer. Further conclusions cannot be reached until densitometry measurements are completed and spacecraft altitudes are known for each exposure.

8.6.6 Recommendations

The crew indicated that the orientation of the camera produced operational difficulties, such as pointing the camera at the horizon. Spacecraft rate damping in the axis corresponding to the camera line-of-sight was extremely difficult. The crew strongly recommended that the camera be boresighted to the roll axis of the spacecraft or to the spacecraft vertical plane.

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Figure 8.6-1. - Experiment S-11, airglow horizon 20-second exposure with filter.

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Figure 8.6-2. - Experiment S-11, airglow horizon 5- second exposure without filter.

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Figure 8.6-3. - Experiment S-11, airglow horizon 2-second exposure without filter.

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8.7 EXPERIMENT S-12, MICROMETEORITE COLLECTION

8.7.1 Objectives

The objectives of Experiment S-12, Micrometeorite Collection, were to determine the micrometeorite activity in a near-earth space environment and to study the effect of the environment on biological microorganisms.

8.7.2 Equipment

The basic objectives were obtained by exposing polished metal and plastic surfaces to the environment outside the Gemini spacecraft. Experimental data include the particulate material collected, holes and craters in the specially prepared surfaces, and numbers of viable microorganisms remaining on the biological exposure plates. The microorganisms used were ubiquitous agents which are absolutely harmless to man, more so than the bacteria found on human skin. The organisms used were:

(a) T-bacteriophage (an E. coliphage)

(b) Penicillium Roquefort mold spores

(c) Bacillus stearothermophilus spores

(d) Bacillus subtiles spores

(d) Tobacco mosaic virus.

Laboratory tests have shown these organisms to be resistant to adverse conditions, hence their selection for space studies.

Special sterile surfaces for the collection of microorganisms were also included within the sample holders. There was a possibility of collecting microorganisms which form some type of cloud around the Gemini spacecraft and which presumably come from the spacecraft or from ejected materials.

All material specimens were returned to earth by stowage in the Gemini spacecraft for postflight examination and analysis at special laboratories.

The micrometeorite collection hardware consisted of an aluminum structure mounted outside the spacecraft on the retrograde section of the adapter. Mounting spaces were designed for 24 surfaces, materials, or specimens. A cover door was provided for exposing the experiment samples during orbital time periods as desired. Figures 8.7-1 and 8.7-2 show the hardware configuration in both the closed and open positions. The collector cover door could be remotely controlled by the spacecraft crew, thereby allowing opening or closing of the cover as required.

During recovery three biological swabs were taken from preselected locations in the Gemini cabin. The swabs were contained in sterile plastic containers for later analysis.

8.7.3 Procedures

The cover door of the micrometeorite collection remained in the closed position until just prior to the sleep period for the crew. This activation time was required to prevent exposing the sample surfaces to particles caused by thruster firings, fuel-cell purges, or overboard dumps of liquid. The collector door was left open for two successive periods of 7 hours 41 minutes and 9 hours 6 minutes duration. The S-12 hardware was retrieved during the egress part of EVA at 49:27:00 g.e.t. and then stowed in the spacecraft.

8.7.4 Results

The preliminary study of the data indicates that the collector hardware functioned perfectly from a mechanical standpoint. The complete evaluation of the samples will take several months, but indications show that micrometeorites were collected. The major effort to date has been in scanning with the electron microscope to detect holes in the substrates and to look for particles. Two general types of holes have been found. One is a long break in the support film, and the other is a small hole with a large amount of material around the hole, as seen in figures 8.7-3 and 8.7-4.

There were some stains on the copper screening used to support the collection surfaces, which may indicate some low-velocity corrosive contamination from the spacecraft. This is a localized effect, and the samples are being studied to be sure that it does not affect all the collection surfaces. The impact-detection experiments are not effected by surface contamination.

The preliminary results from the biological samples show that some penicillium, T_1 bacteriophage, and tobacco mosaic virus survived. The one organism which failed to survive was the stearothermophilus. Analysis of the cabin swabs also shows survival organisms. Quantitative survival rates will be available after further analysis.

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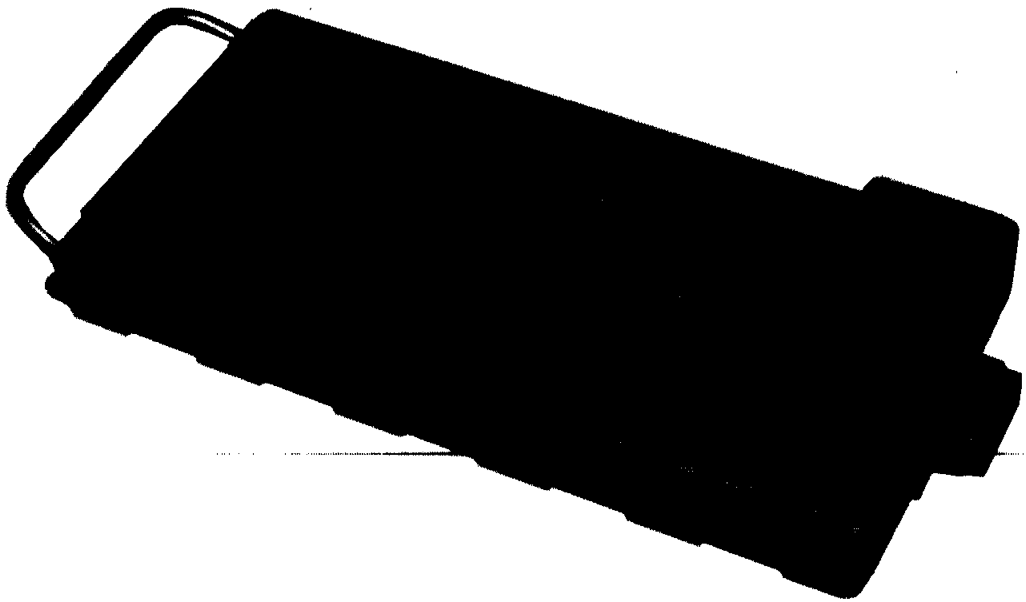

Figure 8.7-1. - Experiment S-12, micrometeorite collection equipment.

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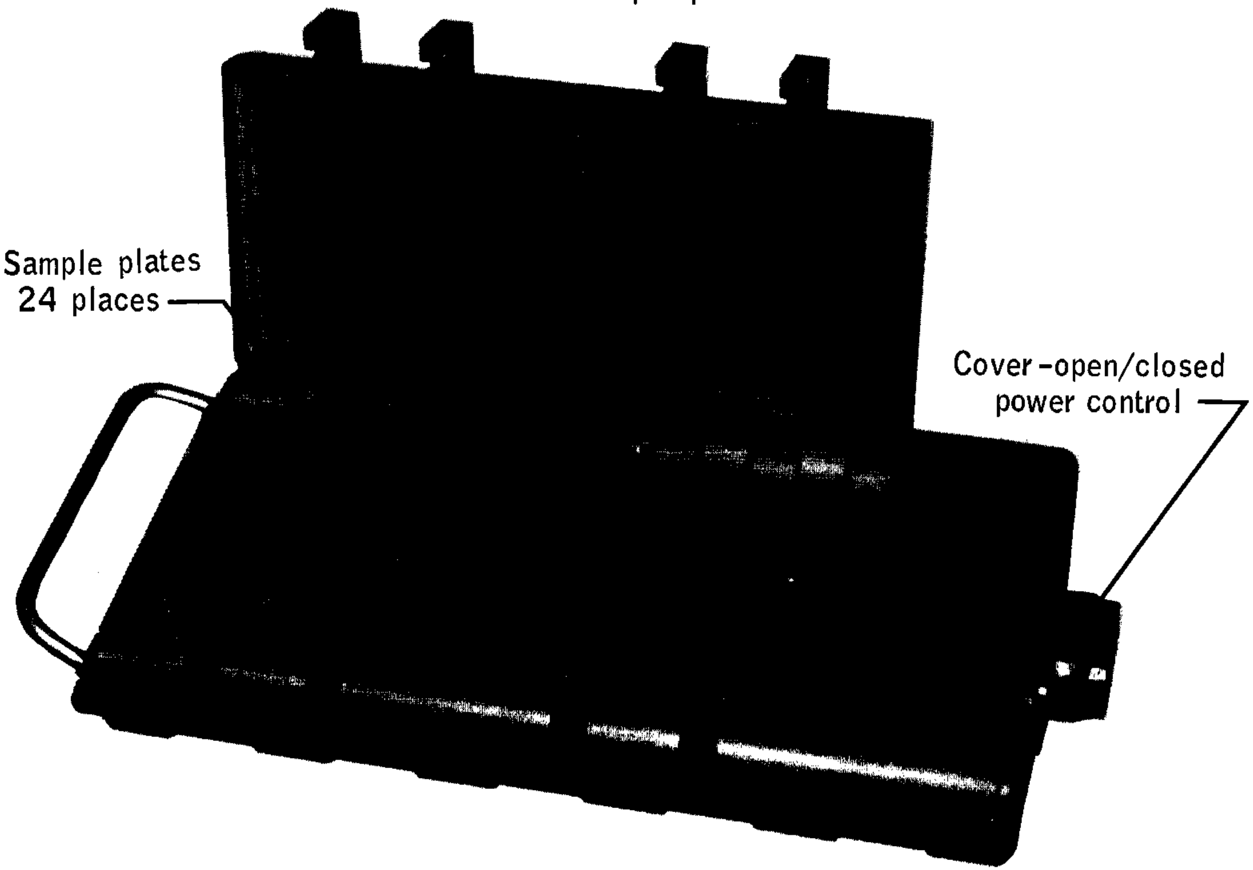

Figure 8.7-2. - Experiment S-12, micrometeorite collection equipment configuration.

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Figure 8.7-3. - Experiment S-12, micrometeorite impact tear hole, magnified 35 000 times.

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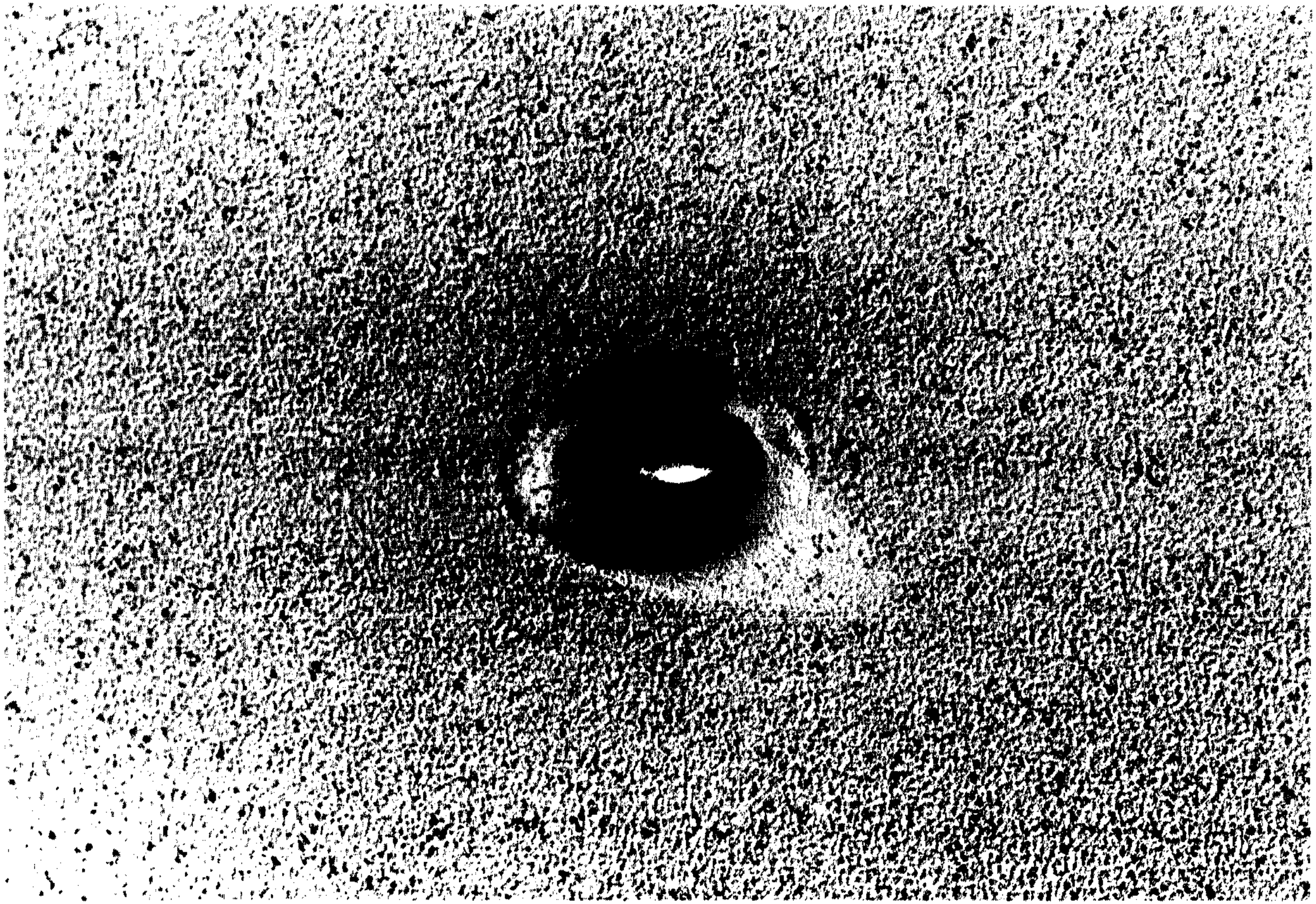

Figure 8.7-4. - Experiment S-12, micrometeorite impact hole, magnified 35 000 times.

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9.0 CONCLUSIONS

The overall performance of the two launch vehicles, the spacecraft, the flight crew, and the mission support was satisfactory for all phases of the Gemini IX-A mission. The Augmented Target Docking Adapter performance was satisfactory for rendezvous but docking could not be accomplished because the shroud covering the Target Docking Adapter did not jettison. The flight contributed to the knowledge of manned space flight, especially in the areas of rendezvous, extravehicular activities, and controlled reentry.

The following conclusions were obtained from data evaluation and crew observations:

1. The three rendezvous operations scheduled in the Gemini IX-A flight plan were accomplished without difficulty. The rendezvous in the third revolution was accomplished as planned, using the Inertial Guidance System to determine the correct velocities to be added at insertion, and at first apogee. The coelliptic maneuver was then initiated one revolution earlier than on the Gemini VI-A and Gemini VIII missions, and the rendezvous was complete at 4 hours and 15 minutes after lift-off. The rendezvous from an equi-period orbit was visually accomplished, using the platform for attitude indications, and onboard charts to determine the necessary rendezvous maneuvers, but without the aid of the radar and computer. The rendezvous from above, simulating an Apollo Lunar Module abort, proved to be difficult during the terminal phase when looking down at the target against the sunlit earth background. The crew concluded that an optical rendezvous would have been very difficult in the case encountered, if not impossible, and that the rendezvous radar was essential to successfully complete the terminal phase maneuvers. The propellant required for the first two rendezvous operations was very close to nominal; however, the propellant required for the rendezvous from above was substantially higher than planned because of the difficulty in maintaining line of sight with the target.

2. The extravehicular activity of the Gemini IX-A mission was conducted during both dayside and nightside conditions for a total period of 2 hours and 5 minutes, and proved the capability of man and his life support equipment to operate in the space environment. The pilot reported that using the tether for maneuvering control is not satisfactory except as an aid for returning to the point of attachment to the spacecraft. Work was found to be much harder in space than in a one-g environment, revealing that the simulation of these work tasks in one-g and short-term (20 seconds) zero-g environments had not been completely

representative of the actual tasks. The overall workloads were much higher than anticipated, due in part to the lack of proper restraint mechanisms for the pilot's use in each work area.

3. The Gemini IX-A reentry was the most accurate of all United States manned space flight. The reentry trajectory, reconstructed from final tracking and Inertial Guidance System data, indicates that the spacecraft was 0.38 nautical mile from the planned trajectory when the drogue parachute was deployed and would probably land within one mile of the planned landing point. This was accomplished even though the retrorockets had produced 1.06 percent more total impulse than nominal, moving the landing point for a ballistic trajectory 46 nautical miles uprange. The reentry rate-command mode was utilized continuously after guidance initiate, and only three-fourths of the propellants loaded into the Reentry Control System were required to stabilize and control the spacecraft during retrofire and reentry.

4. During powered flight, the command pilot's capability to read instrument panel displays of critical launch vehicle and spacecraft systems performance was drastically reduced when the sun directly impinged on the spacecraft window.

5. The PCM tape recorder failed to operate properly in the playback mode after the fifth revolution. This failure was caused by tape coming off a guide roller, apparently as a result of a metallic chip momentarily jamming one of the negator-spring gear trains.

6. To optimize the rendezvous maneuvers, the crew based their choice of a solution on a comparison of onboard radar and computer information with ground-calculated values, polar plots, and backup solutions. This procedure proved to be effective; however, complete reliance upon the onboard solution would have also achieved rendezvous with close to nominal propellant consumption.

7. The Orbital Attitude and Maneuver System operated satisfactorily during extravehicular activity, and the platform and rate-command modes were determined to be the most desirable for attitude control.

8. As a result of an improper installation procedure, the Augmented Target Docking Adapter shroud failed to release, thus preventing the docking phases of the mission from being completed. Four pyrotechnic wire-bundle release-lanyards were not anchored to structure because an adequate procedure to insure correct assembly had not been used.

9. The Augmented Target Docking Adapter stabilization system maintained acceptable attitude rates; however, the system commanded

excessively high thruster activity which resulted in rapid propellant depletion. This condition may have been caused partly by the shroud, which remained attached to the vehicle.

10. During extravehicular activities, the crew reported that the force required to open and close the right-hand hatch was greater than anticipated. This greater force was evident during the movement of the hatch through the mid-travel point rather than at final closing as experienced on Gemini IV. Postflight testing of the right-hand hatch revealed its closing forces to be somewhat greater than the specification value; however, the loss of the crew's mechanical advantage at the mid-travel point of the hatch was the primary cause of the closing difficulties.

11. During extravehicular activities, air-to-ground communications received from the command pilot were clipped and very difficult to understand. The VOX mode was being used, and the voice transmissions were clipped and dropping out. During postflight testing, the sensitivity adjustment was found at a low setting which accounts for the poor performance.

12. The total workload during extravehicular activities caused a high metabolic rate which resulted in the moisture removal capability of the Extravehicular Life Support System being exceeded and the pilot's pressure visor became fogged. However, the pilot reported that he was reasonably comfortable throughout the extravehicular activities until ingress.

13. Immediately after spacecraft landing, water appeared in the command pilot's footwell. This water apparently either accumulated as a result of a slow leak in the cabin drinking-water tank, or entered the cabin as a result of leaving the water seal open until after landing which may have allowed a small amount of sea water to enter through the pressure relief valve.

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10.0 RECOMMENDATIONS

The following recommendations were made as a result of engineering analyses and crew observations of the Gemini IX-A mission:

1. The final VOX sensitivity adjustment for the communications system should be made as near launch time as practical. It should be adjusted to accommodate the member of the flight crew with the lower voice intensity. If use of the VOX mode is contemplated during flight, the VOX operation should be checked as soon as possible after insertion and readjusted by the flight crew if necessary.

2. A review should be performed of the procedures used for testing batteries in-flight to ensure that the tests are adequate to determine the actual status of the batteries.

3. The platform and rate-command control modes should be used for all attitude control during extravehicular activity. Use of the pulse and direct modes should be avoided.

4. Cabin hardware, pyrotechnic devices, and the location of stowed items should be reviewed again to ensure that the elevated temperatures and direct sunlight will not cause damage during extravehicular activities with the hatch open.

5. Before launch, the sun angle and flight-path angle should be studied to determine if the sun could cause interference with the crew's ability to see critical instrument panel displays during the powered phase of flight. If the condition can exist, protective devices must be provided to attenuate the sunlight.

6. The installation of the adapter general-illumination light should be reviewed to determine if protection for the light is necessary during the launch and spacecraft separation phases of flight.

7. Extravehicular activity should be carefully planned with regard to the work effort and forces required for each task, and for the total activity. Adequate restraints must be provided where necessary to allow the astronaut to perform each assigned task. These restraints must be easy to apply and remove; however, they must also be fail-safe for quick release. Two-handed operations must be avoided where restraint devices cannot be utilized. All tasks should be kept as simple as possible, and prolonged activities requiring high levels of work should be avoided.

8. A study should be made to determine other methods of providing extended zero-g simulations, such as underwater simulations, which can provide longer periods in the weightless state. These simulations should enable the participant to become more familiar with the overall workload and the positioning problems associated with long periods in zero-g conditions.

9. Exercise studies should be performed to measure workload level and total workload capability. The data from these studies should provide increased confidence in correlating heart rates and other physiological measurements with workloads under specific conditions.

10. The shroud problem emphasizes the requirement for detailed procedures which have been reviewed by design and operations engineers, technicians, and other associated personnel, and which have been checked out by a complete demonstration. Also, the incident highlights the point that all deviations from a procedure must be documented.

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12. NASA Manned Spacecraft Center: Memorandum Trajectory Information for the Gemini IX/ATDA Mission. MSC Internal Note 66-FM-43, May 24, 1966.

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18. McDonnell Aircraft Corp.: Postflight Evaluation Procedures for Spacecraft 9. SEDR F499-9, May 4, 1966.

19. McDonnell Aircraft Corp.: Corrosion Control Procedures for Recovered Spacecraft. Procedural Specification 186, Aug. 1965.

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 on 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 normal GLV launch-preparation activities.

12.1.3 Augmented Target Docking Adapter History

The history of the Augmented Target Docking Adapter (ATDA) at the contractor's facility is summarized in figure 12.1-7. The ATDA history at Cape Kennedy is shown in 12.1-8 with emphasis on ATDA testing and prelaunch preparation.

12.1.4 Target Launch Vehicle Histories

The Target Launch Vehicle (TLV) history at the contractor's facility in San Diego, California, is shown in figure 12.1-9, and the TLV history at Cape Kennedy is presented in figure 12.1-10. Both figures include systems testing and concurrent problems.

12.1.5 Astronaut Maneuvering Unit and Extravehicular Life Support System Histories

Figure 12.1-11 is a summary of the histories of the Astronaut Maneuvering Unit (AMU) and the Extravehicular Life Support System (ELSS).

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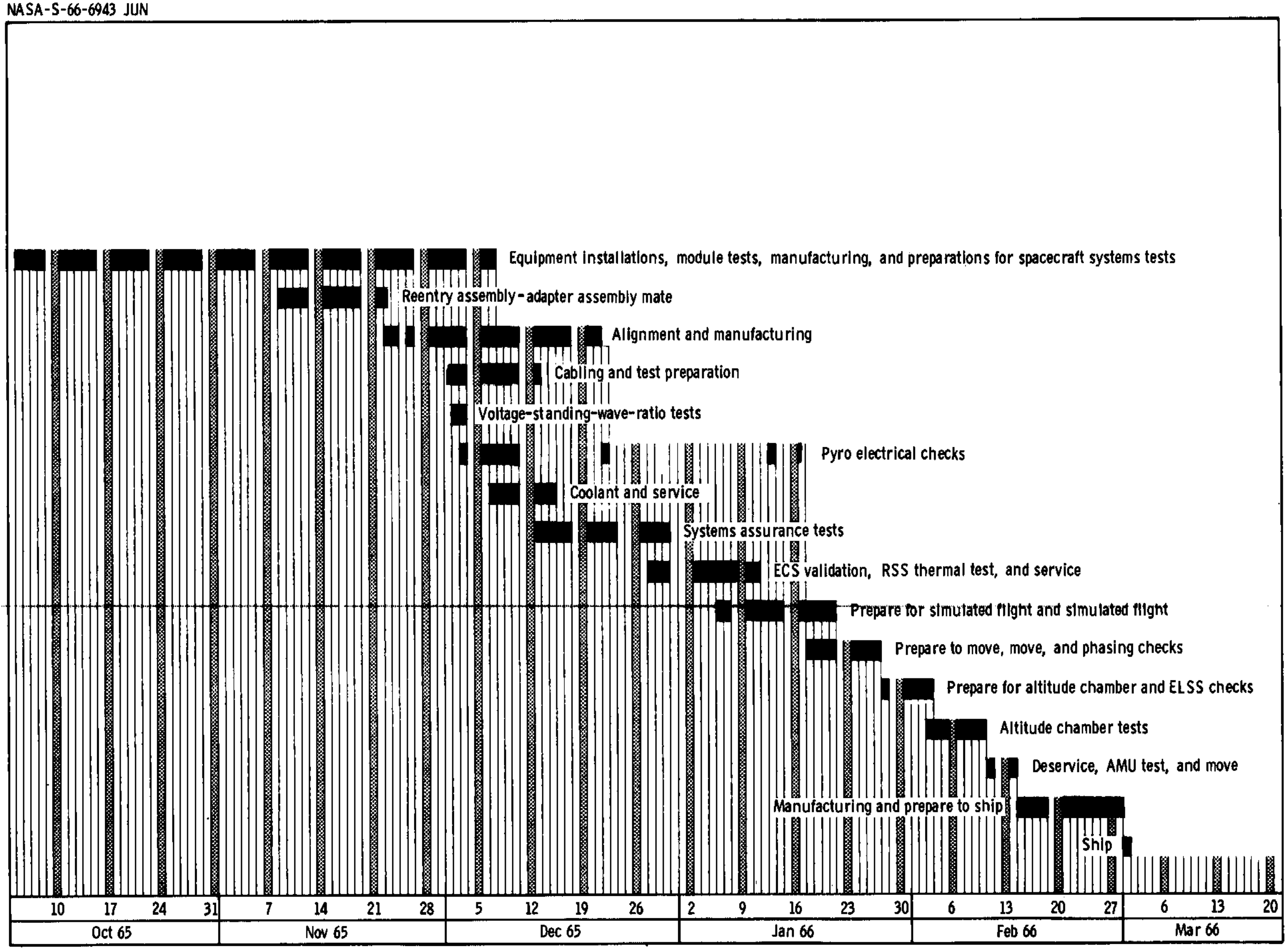

Figure 12.1-1. - Spacecraft 9 test history at contractor facility.

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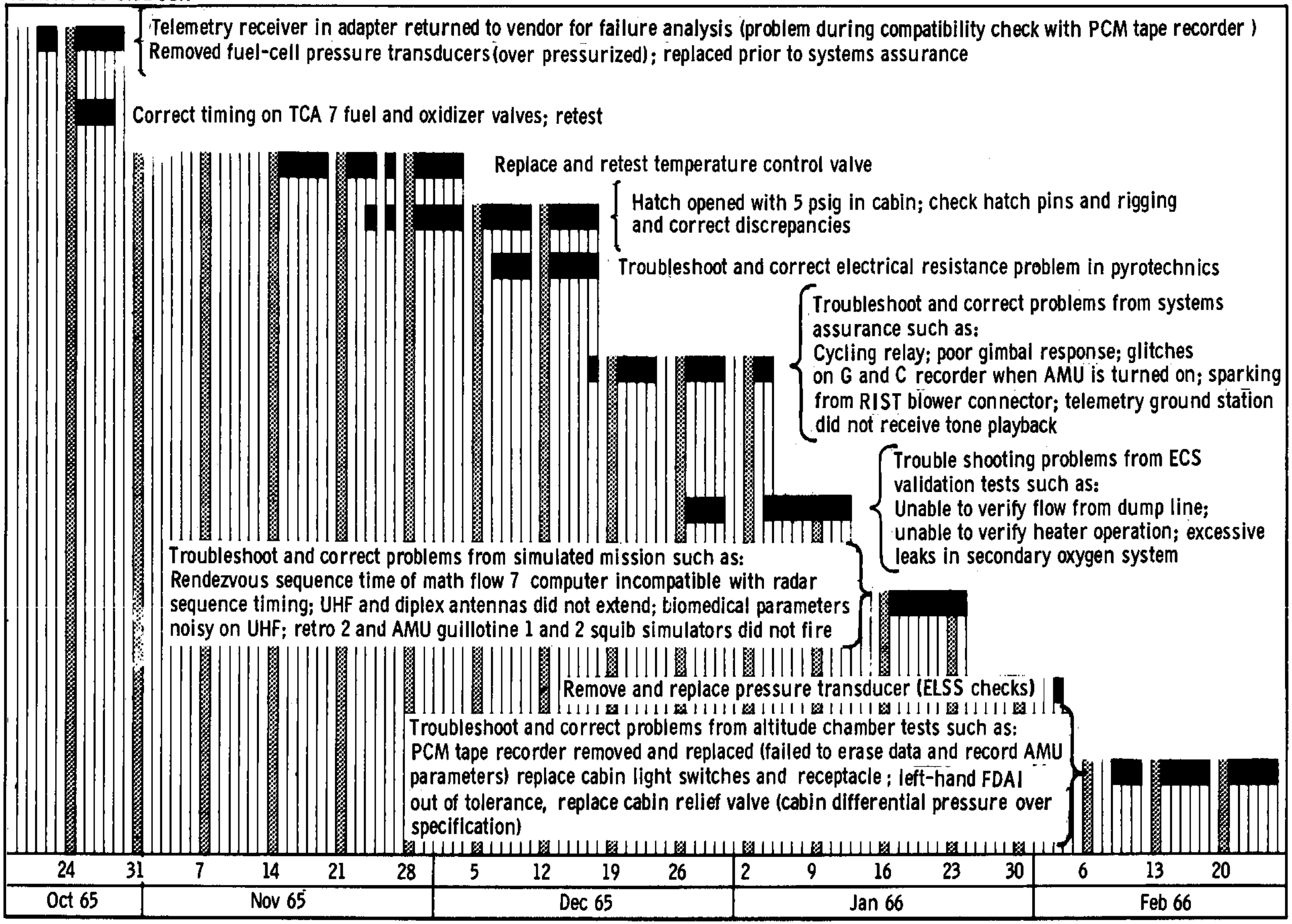

Figure 12.1-2. - Spacecraft 9 problem areas at contractor facility.

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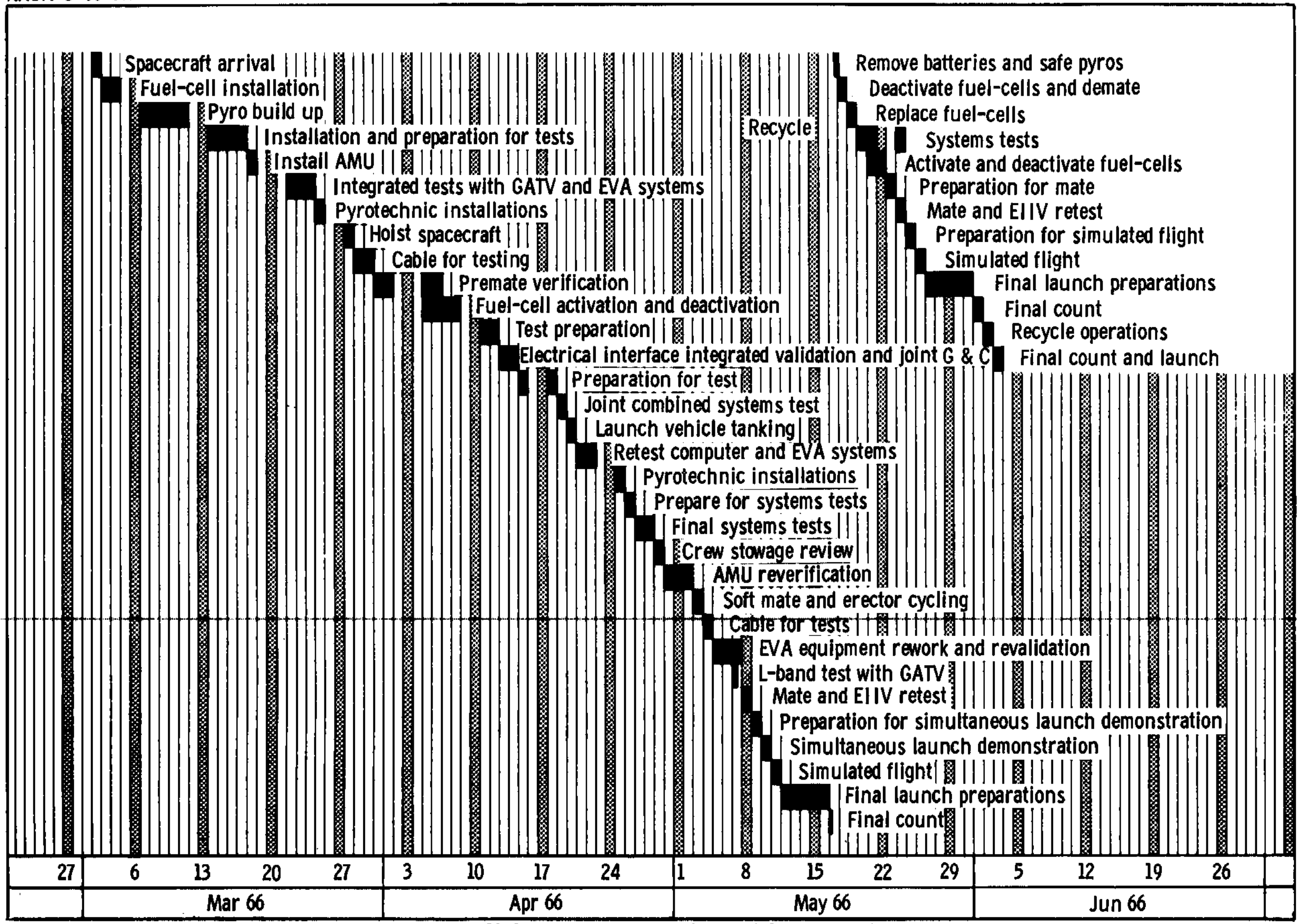

Figure 12.1-3. - Spacecraft 9 history at Cape Kennedy.

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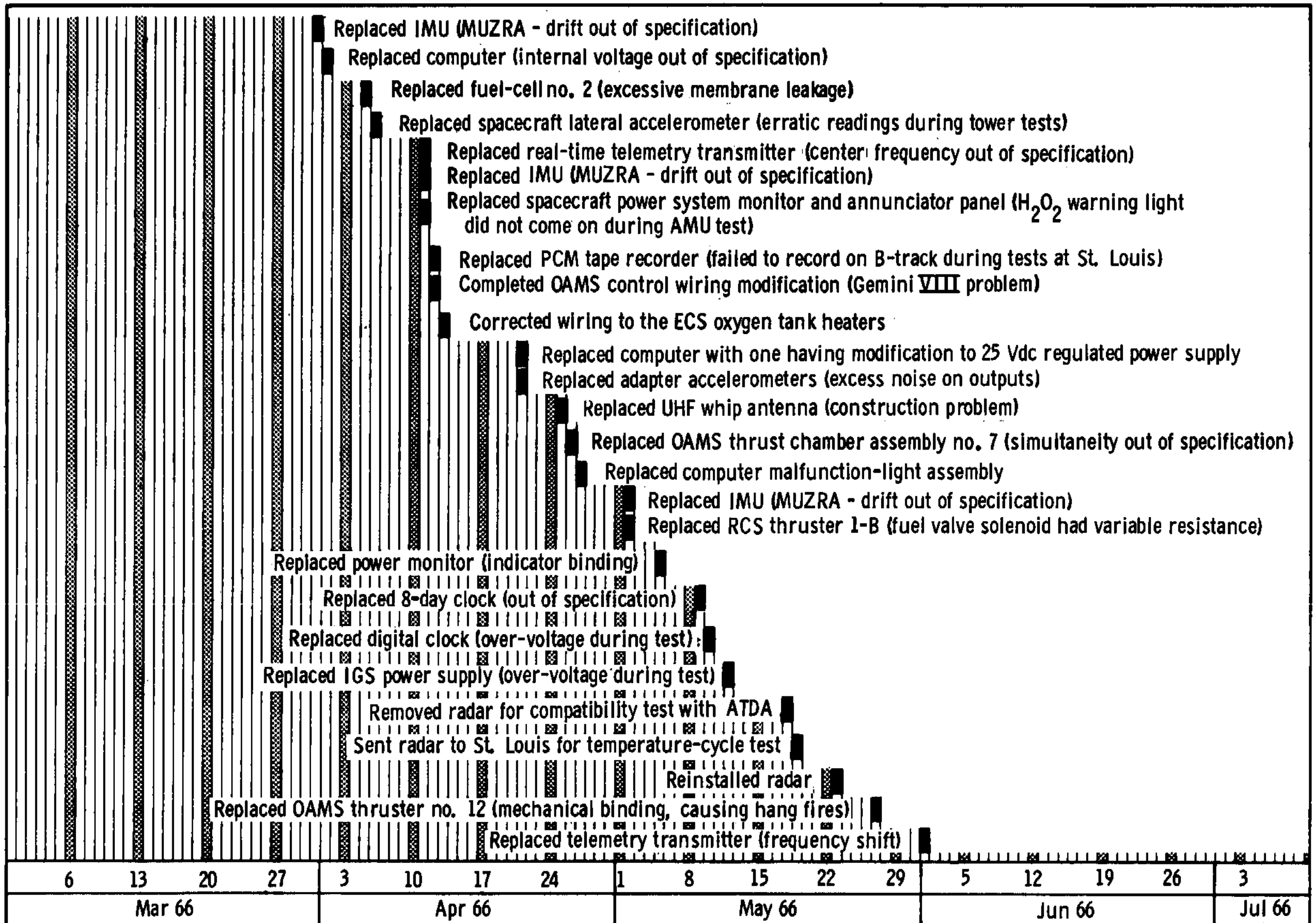

Figure 12.1-4. - Spacecraft 9 significant problems at Cape Kennedy.

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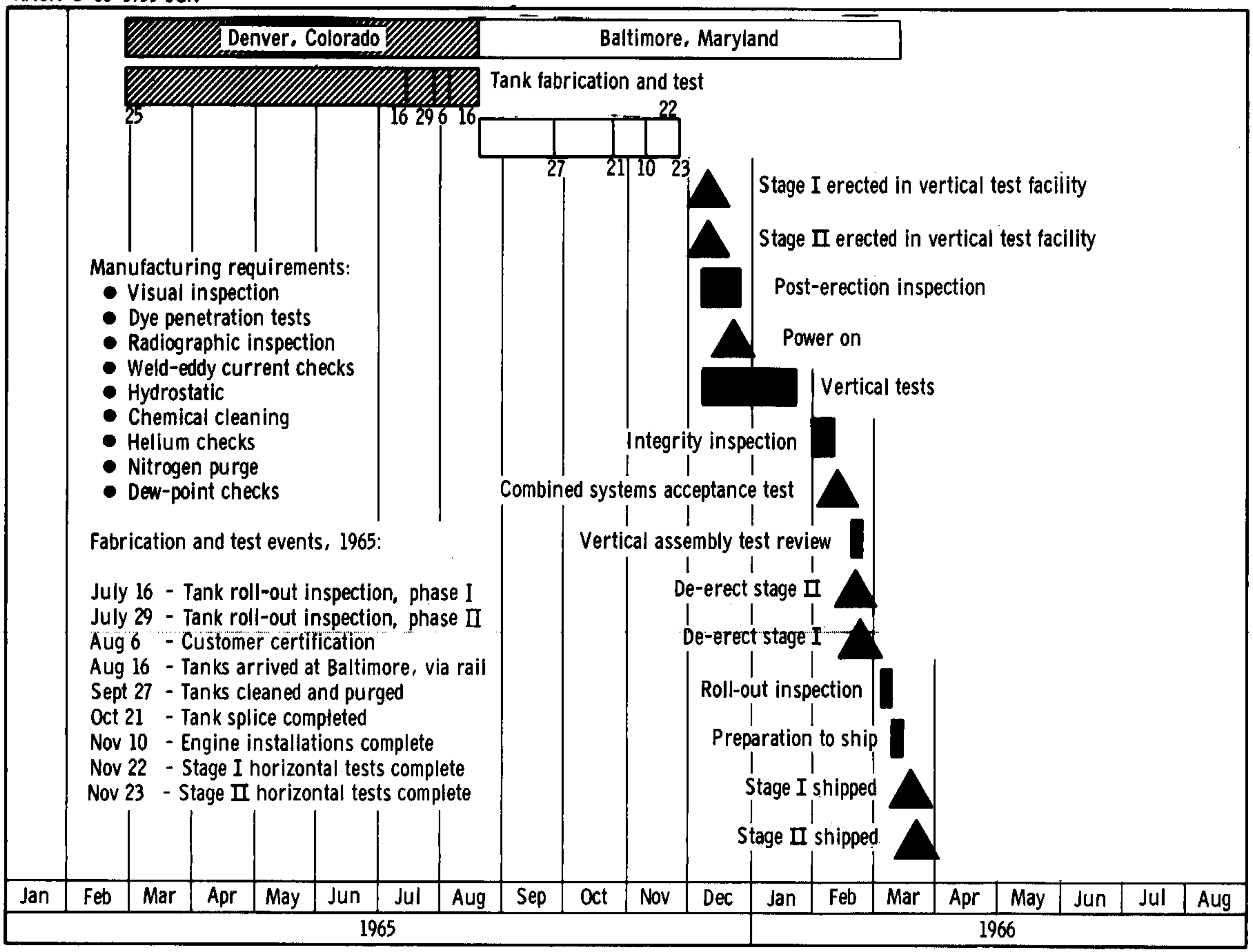

Figure 12.1-5. - GLV-9 history at Denver and Baltimore.

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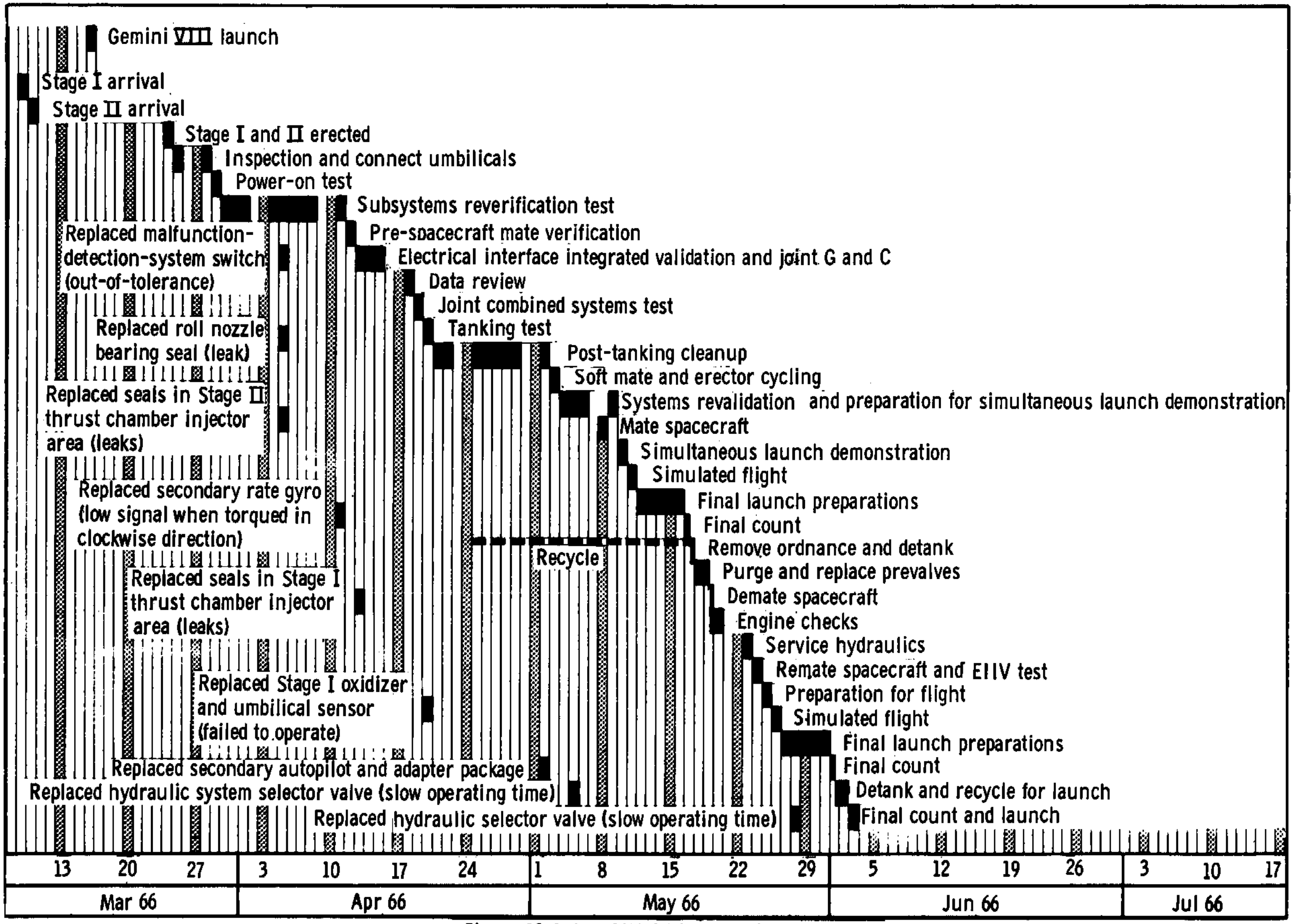

Figure 12.1-6. - GLV-9 history at Cape Kennedy.

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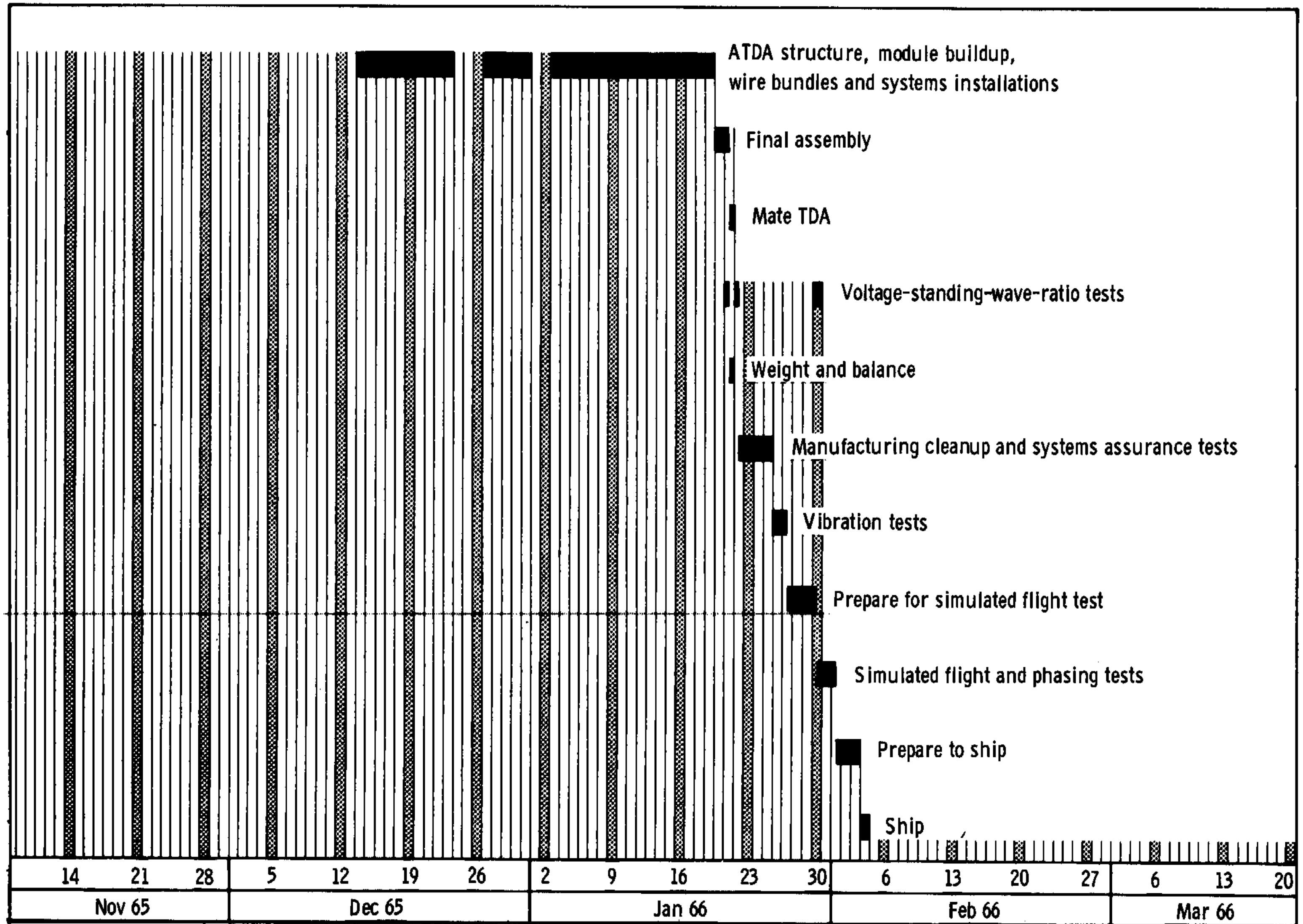

Figure 12.1-7. - ATDA manufacturing and systems test at contractor facility.

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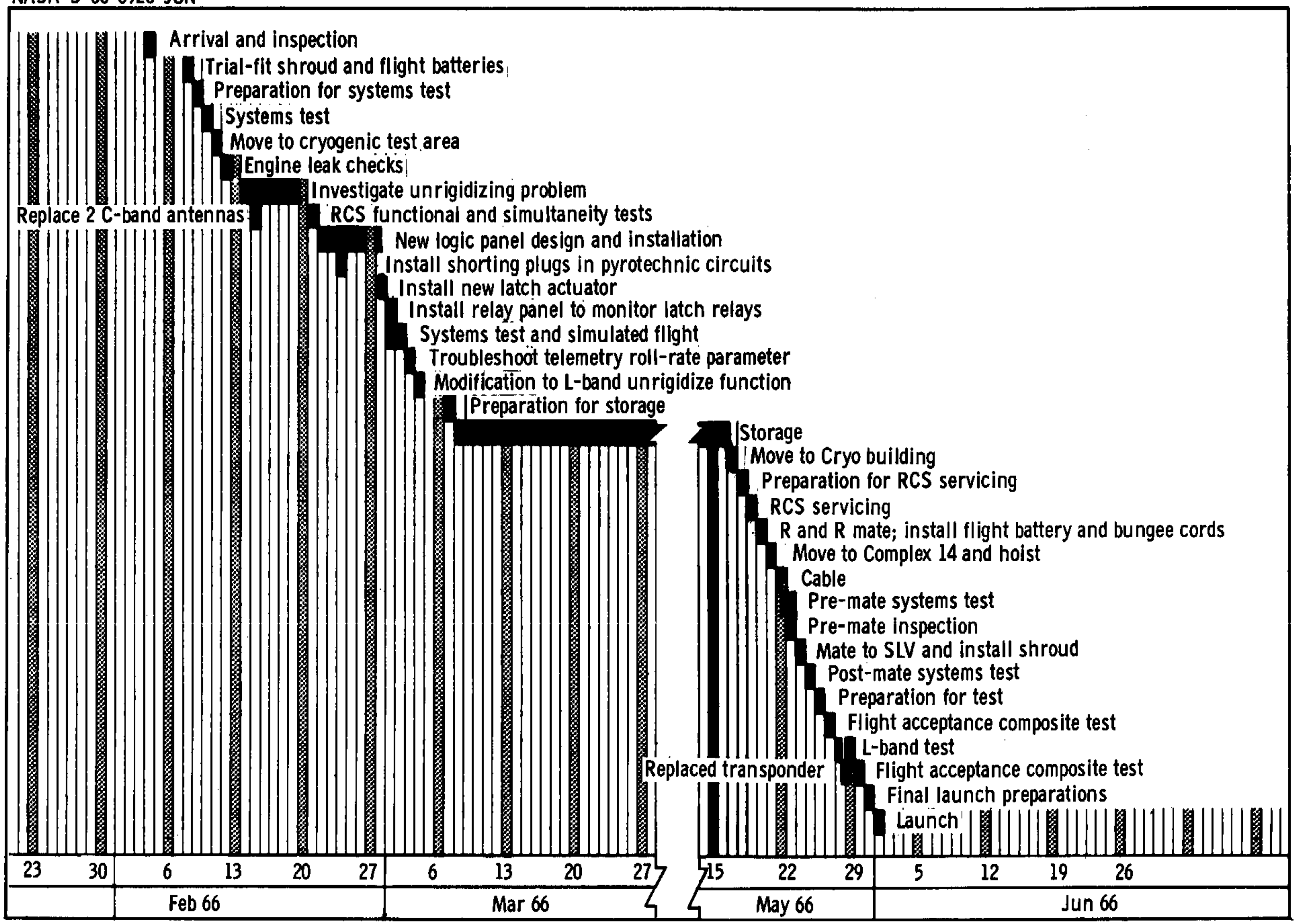

Figure 12.1-8. - ATDA history at Cape Kennedy.

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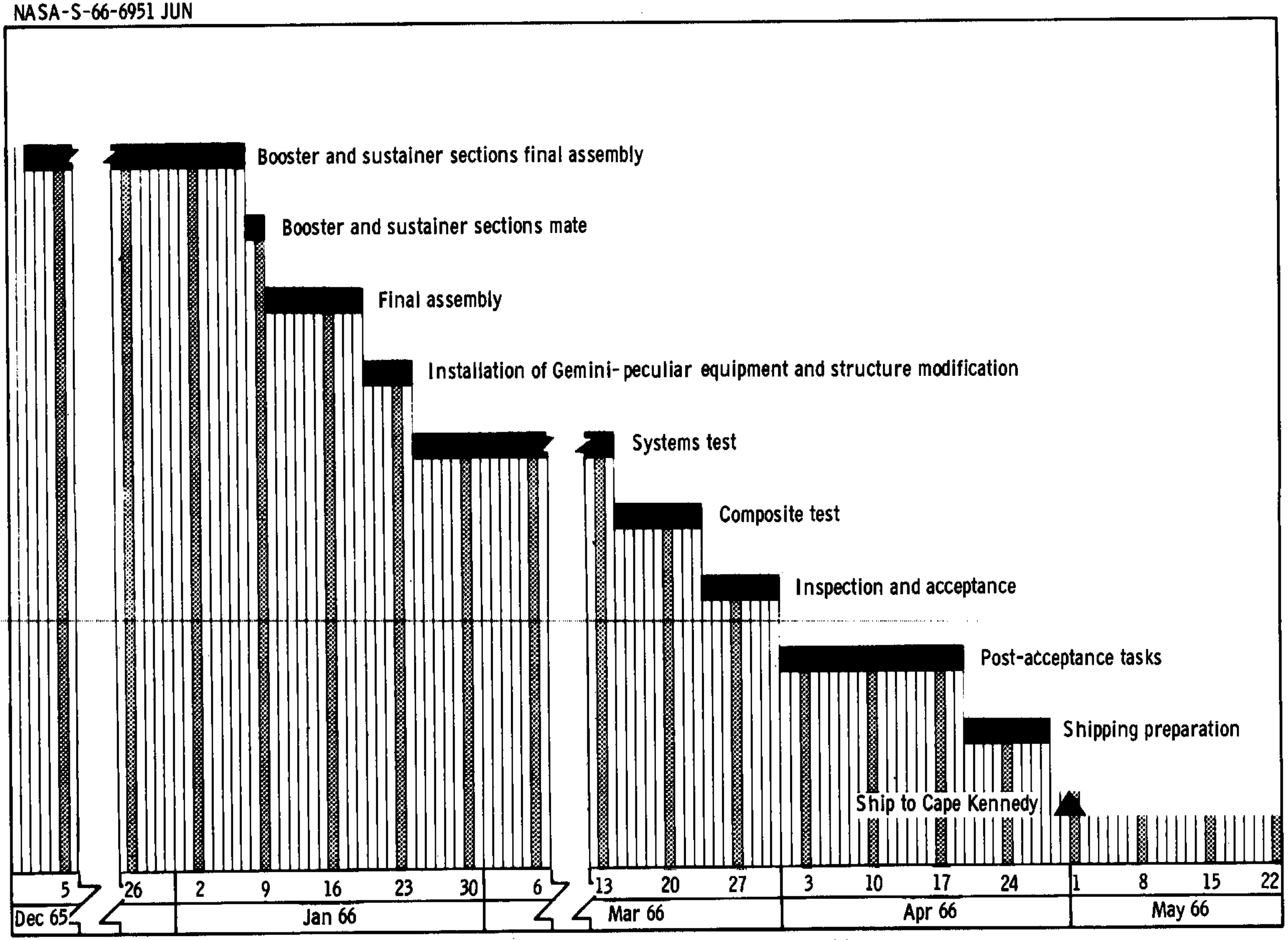

Figure 12.1-9. - SLV 5304 history at contractor facility.

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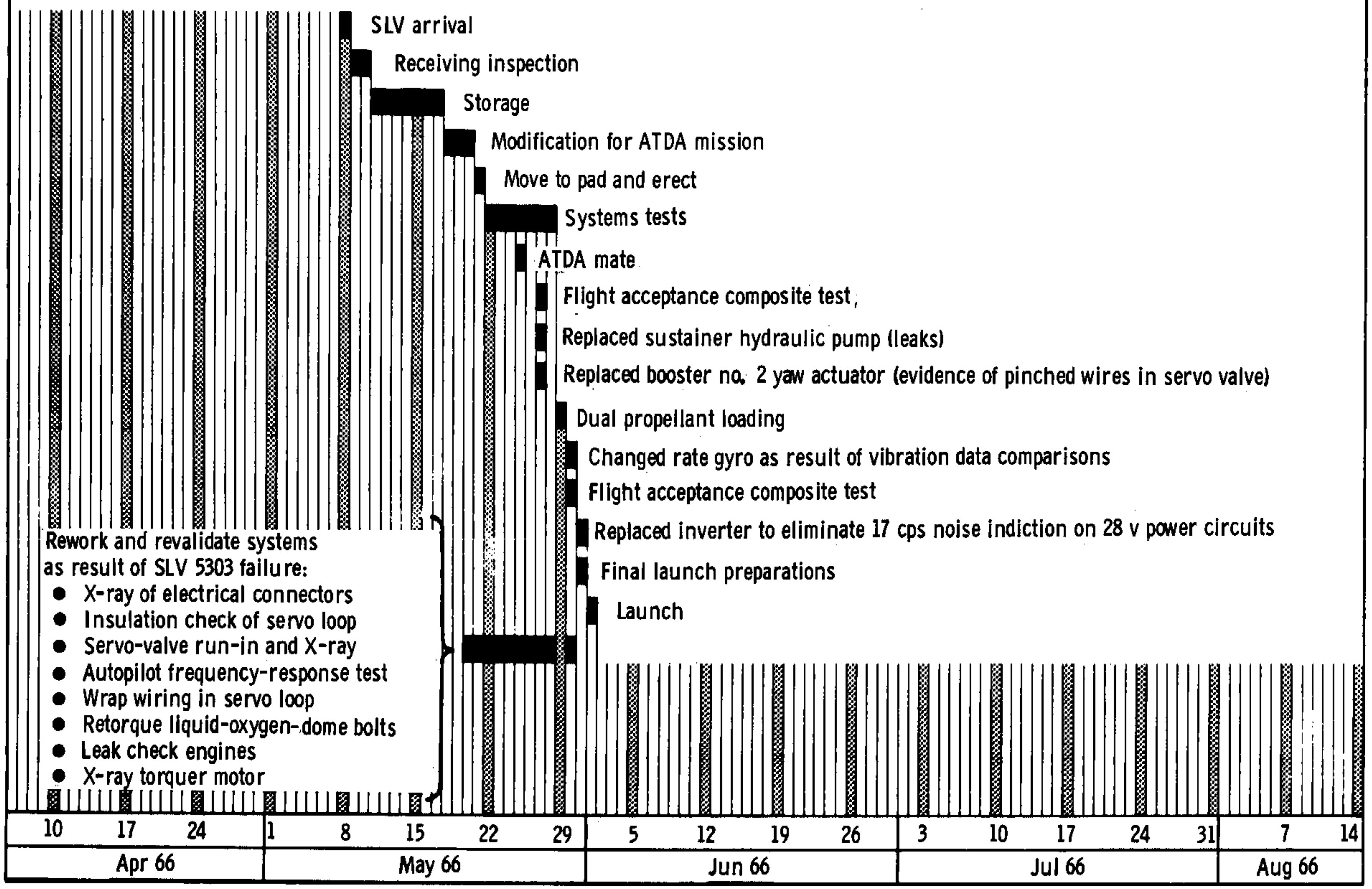

Figure 12.1-10. - SLV-3 5304 history at Cape Kennedy.

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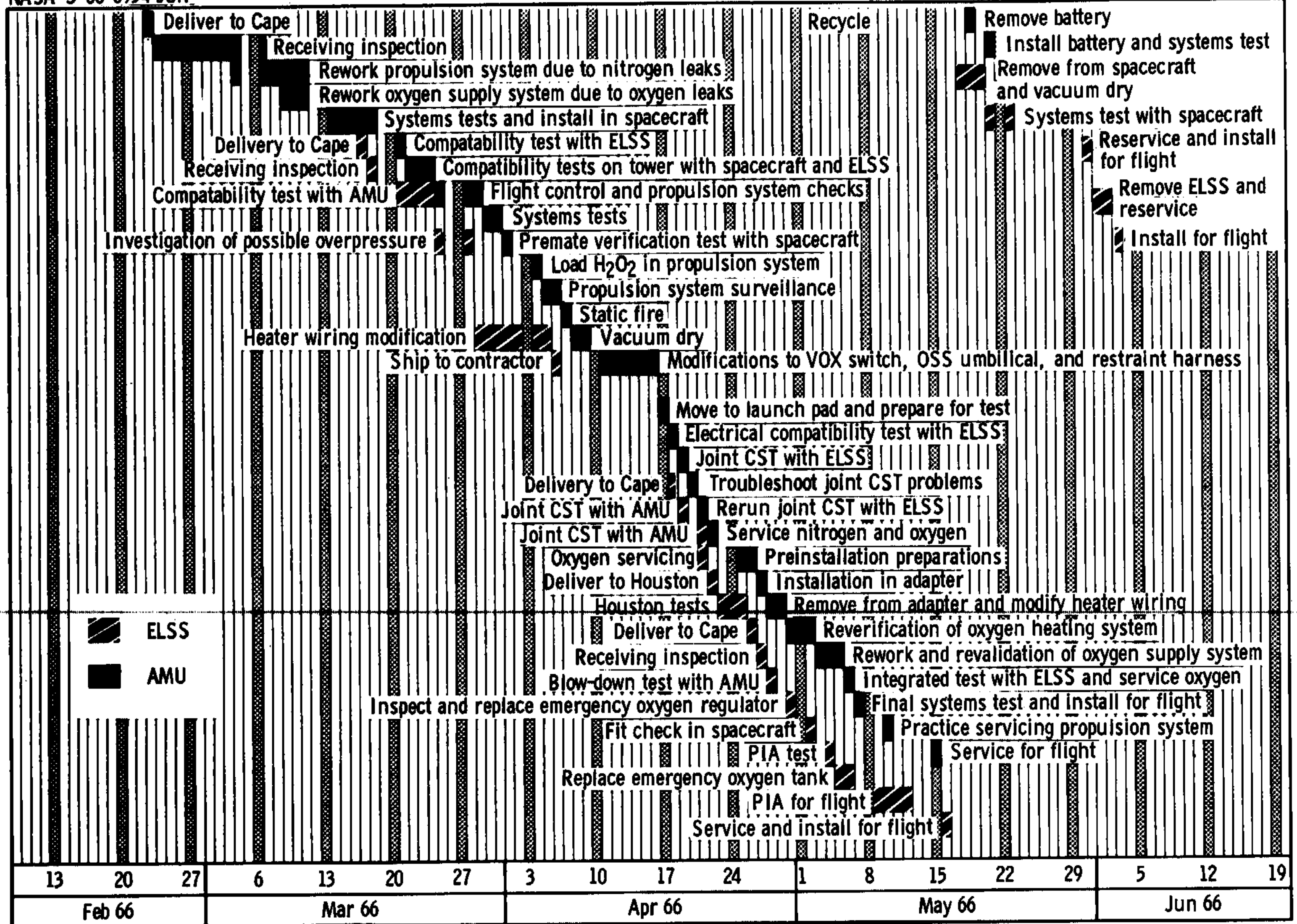

Figure 12.1-11. - AMU and ELSS history at Cape Kennedy.

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, June 1, 1966. Surface weather observations in the launch area at 10:00 a.m. e.s.t. were as follows:

Cloud coverage	Scattered clouds, 1200 feet; broken clouds, 9500 feet; broken clouds, 14 000 feet; overcast, 20 000 feet
Wind direction, deg from north	360
Wind velocity, knots	4
Visibility, miles	7
Pressure, in. Hg	29.95
Temperature, °F	79
Dew point, °F	72
Relative humidity, percent	80

Because of a problem in the ground equipment, the launch of the Gemini Space Vehicle was postponed until June 3, 1966. Surface weather conditions in the launch area at 8:40 a.m. e.s.t. were as follows:

Cloud coverage	Scattered clouds, 3500 feet; stratocumulus, 2/10 covered; haze aloft
Wind direction, deg from north	80
Wind velocity, knots	16
Visibility, miles	10
Pressure, in. Hg	30.23
Temperature, °F	77
Dew point, °F	58
Relative humidity, percent	52

The prime recovery ship for the Gemini IX-A mission was the U.S.S. Wasp, which was stationed at latitude 27°52' north, longitude 74°58.9' west, on June 6, 1966. Weather conditions observed in that area at approximately 14:00 G.m.t. were as follows:

Cloud coverage	2/10 cumulus, 2000 feet; 7/10 altocumulus, 9000 feet; 1/10 cirrus
Wind direction, deg from north	80
Wind velocity, knots	13
Visibility, miles	10
Temperature, °F	76
Dew point, °F	70
Relative humidity, percent	82
Sea temperature, °F	77
Sea state	2-foot to 4-foot waves

TABLE 12.2-I.- LAUNCH AREA ATMOSPHERIC CONDITIONS FOR THE TARGET LAUNCH VEHICLE/AUGMENTED TARGET DOCKING ADAPTER AT 15:28 G.m.t., JUNE 1, 1966

Altitude, ft (a)	Temperature, °F (a)	Pressure, lb/ft^2 (a)	Density, $slugs/ft^3$ (a)
0×10^3	78.8	2116.7	2267.0×10^{-6}
5	59.7	1775.5	1982.0
10	41.4	1478.9	1714.8
15	27.86	1224.3	1462.6
20	7.7	1007.1	1254.8
25	-7.6	822.3	1059.2
30	-25.8	666.0	894.3
35	-43.0	534.5	747.4
40	-66.6	424.6	629.2
45	-76.4	332.9	506.2
50	-85.2	260.0	404.7
55	-87.0	202.6	216.9
60	-77.3	158.1	241.0
65	-76.7	123.9	188.6

[a]The accuracy of the readings is indicated at the end of the table.

TABLE 12.2-I.- LAUNCH AREA ATMOSPHERIC CONDITIONS FOR THE TARGET LAUNCH VEHICLE/AUGMENTED TARGET DOCKING ADAPTER AT 15:28 G.m.t., JUNE 1, 1966 - Concluded

Altitude, ft (a)	Temperature, °F (a)	Pressure, lb/ft^2 (a)	Density, $slugs/ft^3$ (a)
70×10^3	-70.8	97.5	145.9×10^{-6}
75	-66.1	76.9	113.7
80	-58.4	60.8	88.1
85	-53.0	48.2	69.1
90	-47.7	38.4	54.3
95	-42.7	30.7	42.9

[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	0.8

TABLE 12.2-II.- LAUNCH AREA ATMOSPHERIC CONDITIONS

FOR GEMINI LAUNCH VEHICLE

AT 12:00 G.m.t., JUNE 3, 1966

Altitude, ft (a)	Temperature, °F (a)	Pressure, lb/ft^2 (a)	Density, $slugs/ft^3$ (a)
0×10^3	36.3	2135.3	2310.5×10^{-6}
5	51.6	1787.4	2028.8
10	47.8	1488.7	1707.7
15	27.1	1234.1	1477.0
20	12.2	1015.9	1254.2
25	-6.2	830.0	1066.4
30	-28.0	672.1	907.3
35	-48.6	538.6	763.5
40	-69.2	426.7	636.6
45	-83.4	334.4	517.9
50	-84.6	260.7	404.9
55	-89.9	202.8	319.6
60	-79.4	158.1	242.3
65	-74.7	124.1	187.6
70	-71.1	97.5	146.1
75	-67.0	76.9	113.9
80	-59.4	60.8	88.3
85	-53.1	48.7	69.9
90	-50.3	38.9	55.3
95	-43.8	30.9	43.5
100	-32.1	24.9	34.0
105	-25.6	20.1	27.0

[a]The accuracy of the readings is indicated at the end of the table.

TABLE 12.2-II.- LAUNCH AREA ATMOSPHERIC CONDITIONS

FOR GEMINI LAUNCH VEHICLE

AT 12:00 G.m.t., JUNE 3, 1966 - Concluded

Altitude, ft (a)	Temperature, °F (a)	Pressure, lb/ft^2 (a)	Density, $slugs/ft^3$ (a)
110×10^3	-27.2	16.3	21.7×10^{-6}
115	-15.9	13.2	17.3
120	-14.8	10.7	14.0
125	-14.1	9.6	11.3
130	-6.7	7.1	9.1
135	-5.0	5.9	7.2
140	24.1	4.8	5.8
145	27.1	4.0	4.7
150	30.6	3.3	3.9
155	36.5	2.7	3.1

[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	0.8
120 to 165	4	1.5	1.0

TABLE 12.2-III.- REENTRY AREA ATMOSPHERIC CONDITIONS

AT 14:25 G.m.t., JUNE 6, 1966

Altitude, ft (a)	Temperature, °F (a)	Pressure, lb/ft^2 (a)	Density, $slugs/ft^3$ (a)
0×10^3	79.3	2127.4	2276.4×10^{-6}
5	58.6	1785.1	1994.2
10	43.9	1487.9	1715.2
15	28.6	1232.9	1468.4
20	13.5	1015.7	1250.5
25	-6.0	830.2	1066.0
30	-26.9	672.7	905.5
35	-44.9	540.1	758.5
40	-60.0	429.2	625.8
45	-79.8	337.9	518.5
50	-89.0	263.6	414.1
55	-93.3	204.7	325.4
60	-79.6	159.4	244.1
65	-77.6	124.9	190.3
70	-70.2	98.2	146.7
75	-65.7	77.3	114.3
80	-59.8	29.3	89.1
85	-52.8	48.7	69.7
90	-48.3	38.9	54.9
95	-47.0	30.9	43.7
100	-41.4	24.9	34.5
105	-35.9	19.8	27.4
110	-28.1	16.1	21.5
115	-23.8	13.0	17.3

[a] The accuracy of the readings is indicated at the end of the table.

TABLE 12.2-III.- REENTRY AREA ATMOSPHERIC CONDITIONS

AT 14:25 G.m.t., JUNE 3, 1966 - Concluded

Altitude, ft (a)	Temperature, °F (a)	Pressure, lb/ft^2 (a)	Density, $slugs/ft^3$ (a)
120×10^3	-19.1	10.4	13.8×10^{-6}
125	-5.3	8.6	10.9
130	3.2	6.9	8.7
135	10.2	5.6	7.2
140	13.1	4.6	5.8
145	16.0	4.0	4.7
150	24.8	3.1	3.9
155	28.8	2.7	3.1
160	34.0	2.1	2.5
165	36.7	1.9	2.1
170	32.7	1.5	1.8
175	29.3	1.3	1.6
180	22.1	1.0	1.2
185	14.2	0.8	1.0
190	11.0	0.6	0.8
195	5.2	0.6	0.8

[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	0.8
120 to 165	4	1.5	1.0
165 to 200	6	1.5	1.5

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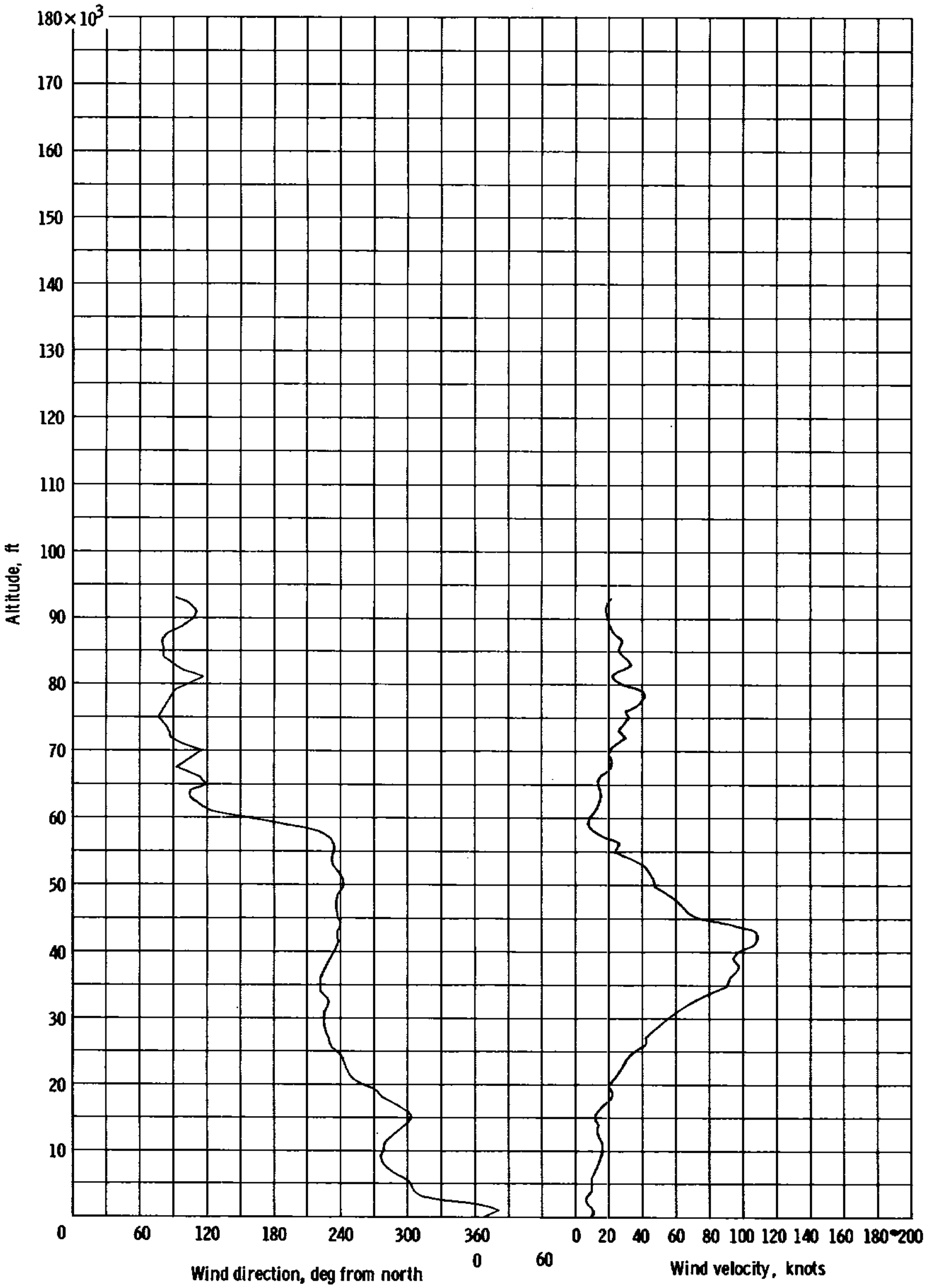

Figure 12.2-1. - Variation of wind direction and velocity with altitude for the TLV/ATDA at 15:28 G.m.t., June 1, 1966.

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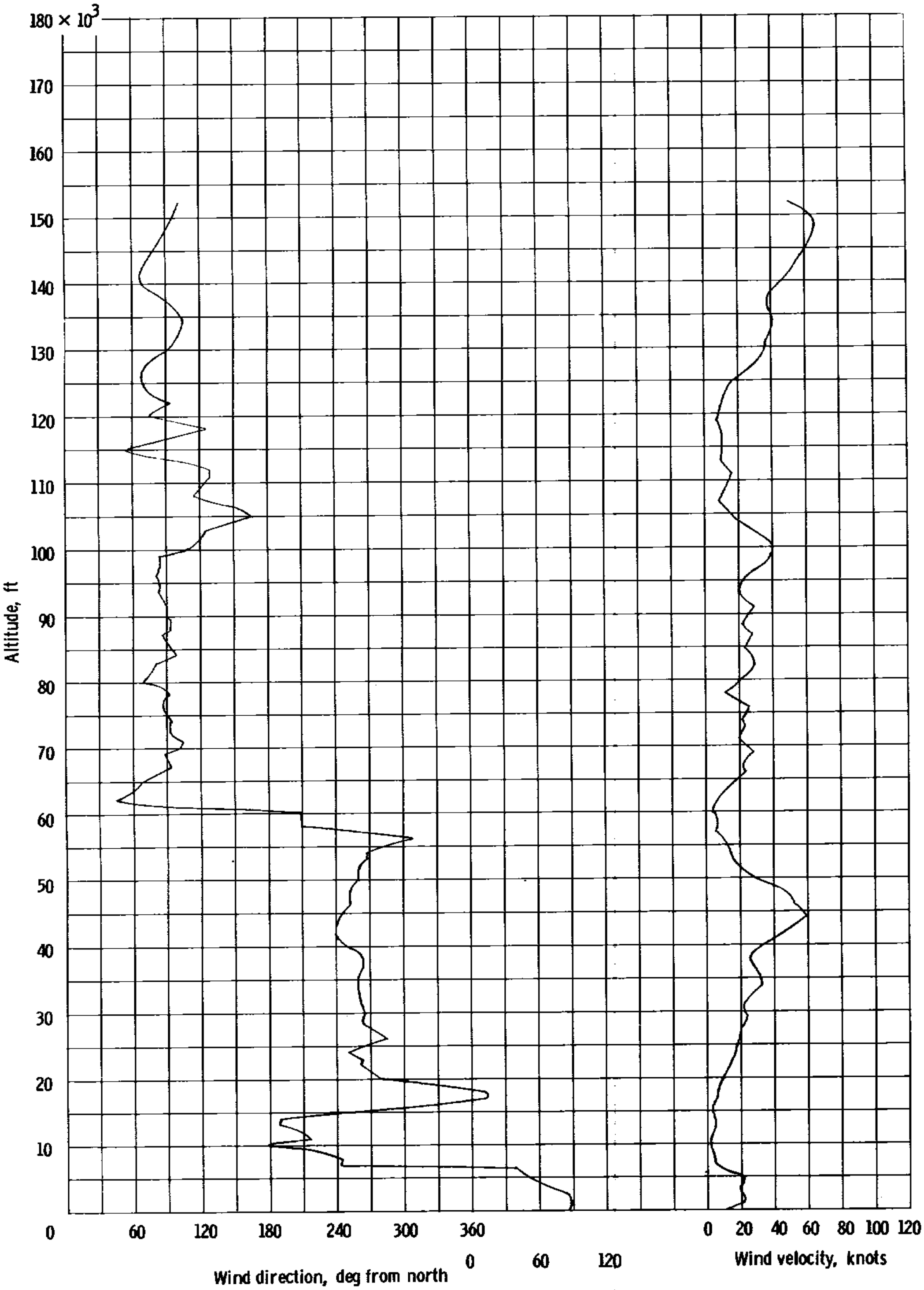

Figure 12.2-2. - Variation of wind direction and velocity with altitude for the Gemini Space Vehicle at 12:00 G.m.t., June 3, 1966.

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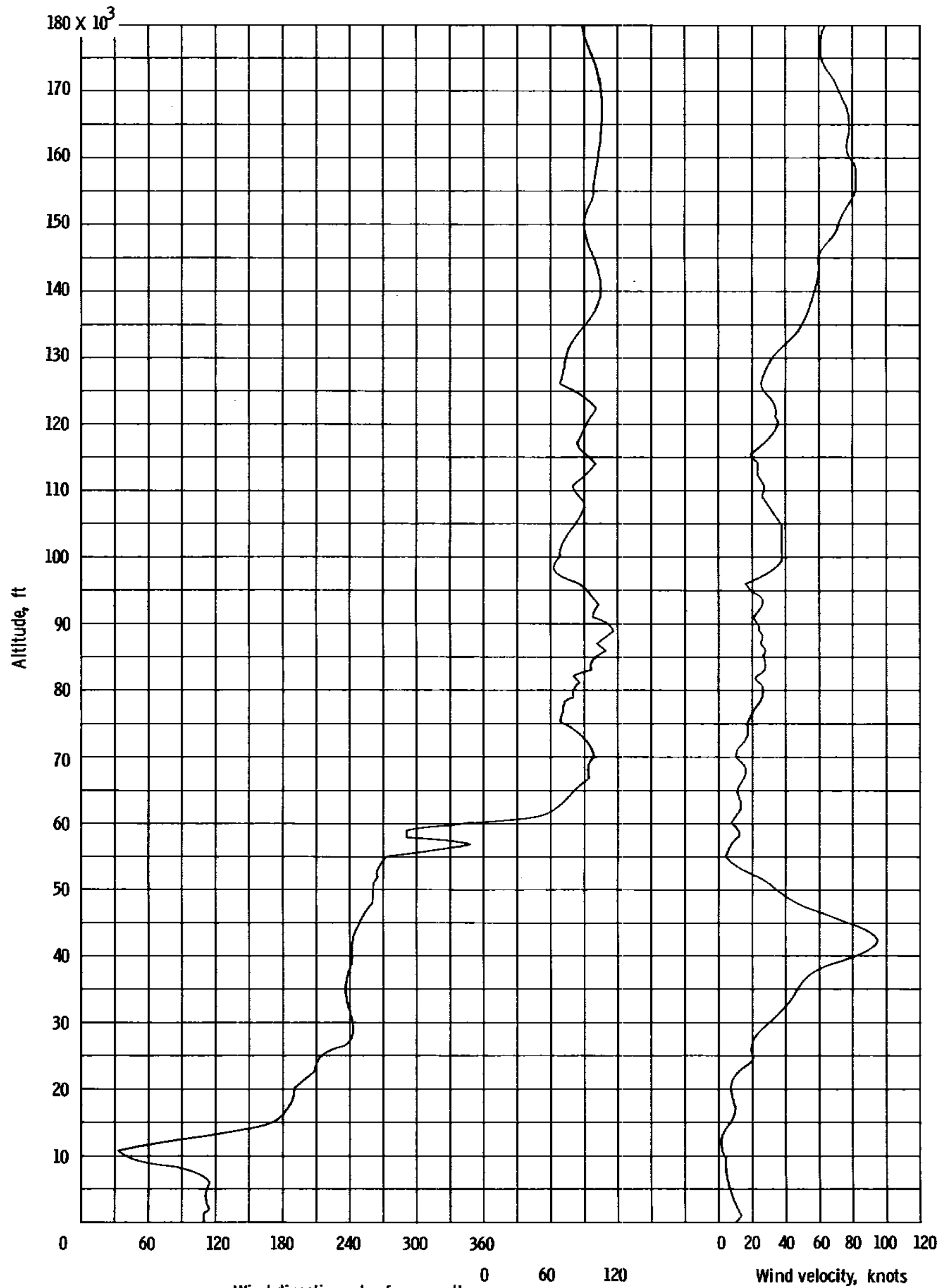

Figure 12.2-3. - Variation of wind direction and velocity with altitude for the Gemini IX-A reentry area at 14:25 G.m.t., June 6, 1966.

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12.3 FLIGHT SAFETY REVIEWS

The flight readiness of the launch vehicles, spacecraft, Gemini Agena Target Vehicle, Augmented Target Docking Adapter, and all support elements for the accomplishment of the Gemini IX and IX-A missions was determined at the review meetings noted below.

12.3.1 Gemini IX Mission

12.3.1.1 Spacecraft Readiness Review.- The Flight Readiness Review of Spacecraft 9 was held on May 4, 1966, at the Kennedy Space Center. The following action items were to be accomplished prior to the launch:

(a) Procedure to be established for updating the computer through the Mission Control Center at Cape Kennedy to be reviewed by KSC to prevent recurrence of the problem of Burroughs ground station being unable to update the spacecraft computer during ascent.

(b) Information on the failure modes of the Reentry Control System (RCS) helium regulator transducer and on the leak rate in the RCS as a function of size of leak to be provided by the contractor.

(c) Complete failure analyses on the RCS thrust chamber assembly (TCA) 1 B and the Orbital Attitude and Maneuver System (OAMS) TCA 7 to be provided by the contractor.

(d) Pulse code modulation (PCM) tape-recorder drivebelt to be inspected, and a complete set of back-up telemetry components (multiplexers and programmer) to be inspected and tested.

(e) Complete failure analyses on the power system monitor and on the telelight panel to be provided by the contractor.

(f) Explanation of problem regarding the attitude controller in the Hybrid Simulator to be provided by the contractor.

(g) Study of the Environmental Control System (ECS) primary oxygen supply accomplished by the contractor. The study was to include two-phase operation, venting during EVA, and oxygen-heater duty cycle.

12.3.1.2 EVA Equipment Review.- A Gemini Program Office review of the EVA equipment was held on May 2, 1966, at KSC. The following action items resulted from this review:

(a) Have putty-type clearance inspection made between the launch vehicle dome and the adapter equipment.

(b) Develop mission rule regarding the Astronaut Maneuvering Unit (AMU) "propellant remaining" indication system.

(c) Determine whether AMU has aluminum "B" nuts and establish their susceptibility to stress corrosion.

(d) Determine that Extravehicular Life Support System (ELSS) emergency oxygen-heater preinstallation acceptance (PIA) tests are realistic and will detect any discrepancies as well as verify proper heater operation.

12.3.1.3 Gemini Launch Vehicle Technical and Preflight Reviews.- On April 20, 1966, a Technical Review on the Gemini Launch Vehicle (GLV) was held in Houston, Texas. On May 13, 1966, a Preflight Readiness Review was held at Cape Kennedy. Items discussed at this latter meeting included the umbilical drop configuration, spares status, and the malfunction-detection thrust-chamber pressure switches.

12.3.1.4 Gemini Launch Vehicle Flight Safety Review Board.- The Air Force Space Systems Division (AFSSD) Flight Safety Review Board (FSRB) met on May 16, 1966, at Cape Kennedy and, with all items resolved, recommended to the Mission Director that the Gemini Launch Vehicle be committed to flight.

12.3.1.5 Gemini Atlas-Agena Target Vehicle Technical and Preflight Reviews.- On April 20, 1966, a Technical Review of the Atlas Standard Launch Vehicle (SLV-5303) and the Gemini Agena Target Vehicle (5004) was held in Houston. On May 12, 1966, a Preflight Readiness Review on the Gemini Atlas-Agena Target Vehicle (GAATV) was held at Cape Kennedy. Items discussed at this meeting were the changes and repairs to both vehicles since delivery to the Air Force Eastern Test Range (ETR), open failure analyses, review of unqualified hardware, propellant contamination found in the SLV liquid-oxygen system, and the spares status.

12.3.1.6 GAATV Flight Safety Review Board.- The Air Force SSD FSRB met on May 16, 1966, at Cape Kennedy and recommended to the Mission Director that the GAATV be committed to flight, as all ground and airborne systems were in readiness. It was noted that eight flight systems onboard the TLV had not been tested in the Combined Systems Test at the contractor's facility in San Diego, California. Project

Sure-Fire, to assure proper operation of the Gemini Agena Target Vehicle propulsion system, had been completed satisfactorily at Tullahoma, Tennessee.

12.3.1.7 Mission Briefing.- The Mission Director conducted the Gemini IX Mission Briefing on May 14, 1966, at KSC. All elements reviewed their status and were found in readiness to support the mission.

12.3.2 Gemini IX-A Mission

Because of the Atlas SLV-5303 failure on the Gemini IX Mission, the Gemini Mission Failure Investigation Plan was initiated, and investigation was turned over to the Air Force SSD. A Gemini IX-A Mission utilizing the GLV and Spacecraft 9, but using SLV-5304 carrying an Augmented Target Docking Adapter (ATDA), was scheduled for June 1, 1966.

12.3.2.1 Air Force SSD Flight Safety Review Board.- The investigation of the SLV-5303 failure was conducted under the Air Force SSD FSRB, which had the following meetings:

(a) The analysis of flight data was discussed on May 17, 1966, at Cape Kennedy.

(b) The failure was pinpointed to the electrical system in the B-2 engine servovalve wiring. All elements presented their review and conclusions of the flight data on May 24, 1966, at SSD Headquarters, Los Angeles.

(c) Two meetings were held on May 31, 1966, at Cape Kennedy, to review the status of the corrective action on SLV-5304 to preclude a similar failure.

Investigation teams of the NASA, General Dynamics/Convair, McDonnell, the Air Force, and Aerospace did extensive reviewing of all past problems, test procedures, Government-Furnished Equipment (GFE), Aerospace Ground Equipment (AGE), flight hardware (including changes), qualification status, sequential events and backups, and test operations on both the SLV-5304 and the ATDA to assure satisfactory completion of the mission.

12.3.2.2 Spacecraft Readiness Review.- The Spacecraft 9, EVA equipment, and ATDA Flight Readiness Review was held at J. F. Kennedy Space Center on May 27, 1966. The following action items resulted:

(a) KSC to correct the AMU-AGE servicing-line problem.

(b) Contractor to provide failure analysis on OAMS thruster 12 oxidizer valve.

(c) Contractor to review ATDA countdown to verify that all possible tests are accomplished through the command link.

12.3.2.3 Gemini SLV-5304 Technical Review.- An Atlas SLV Technical Review meeting was held at SSD Headquarters, Los Angeles, on May 24, 1966. Corrective action items as a result of the SLV-5303 failure were presented. All systems were reviewed, and satisfactory explanations were made for the booster-engine skirt transit damage and for the history of hydraulic-actuator failures.

12.3.2.4 Gemini SLV-5304 Preflight Review.- An Atlas SLV Preflight Review was held on May 28, 1966, at Cape Kennedy. A review of the system status was made, including hardware changes, anomalous test conditions, and the SLV-5303 corrective action items. Open items included the dual propellant tanking with its accompanying propulsion checks and tests, and some retesting of the autopilot.

12.3.2.5 Mission Briefing.- The Mission Director held a Mission Briefing at which the status of the SLV-5304 and the ATDA was presented. All other elements were still in a position to support the mission.

12.3.2.6 Launch Vehicles Flight Safety Review Board.- As mentioned in paragraph 12.3.2.1(c), the Air Force SSD Flight Safety Review Board met on May 31, 1966, at Cape Kennedy. All flight and ground systems were reviewed and found satisfactory. The recommendation was made to the Mission Director that the launch vehicles be committed to flight for the Gemini IX-A mission.

12.4 SUPPLEMENTAL REPORTS

Supplemental reports for the Gemini IX-A 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 page as being a Gemini IX-A 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 IX-A SUPPLEMENTAL REPORTS

Number	Report title	Responsible organization	Completion due date
1	Launch Vehicle Flight Evaluation Report - NASA Mission Gemini/Titan GT-9-A	Aerospace Corp.	August 5, 1966
2	Launch Vehicle No. 9 Flight Evaluation	Martin Co.	July 21, 1966
3	Manned Space Flight Network Performance Analysis for GT-9-A Mission	Goddard Space Flight Center	August 5, 1966
4	Gemini GT-9-A IGS Evaluation Trajectory Reconstruction	TRW Systems	August 5, 1966
5	GT-9-A Postflight Analysis Report	International Business Machines Corp.	July 21, 1966
6	Gemini Agena Target Vehicle 5004 Systems Test Evaluation	Lockheed Missiles and Space Co.	July 21, 1966
7	Atlas SLV-3 Space Launch Vehicle Flight Evaluation Report SLV-3 5303	General Dynamics Corp.	June 30, 1966
8	Atlas SLV-3 Space Launch Vehicle Flight Evaluation Report SLV-3 5304	General Dynamics Corp.	July 21, 1966

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12.5 DATA AVAILABILITY

Delayed-time data are available through revolution 4 only because of the problem with the spacecraft onboard tape recorder. After recovery, the onboard recorder contained data for the following time intervals:

Interval, ground elapsed time, hr:min
6:08 to 6:38
7:45 to 9:13
9:19 to 10:36
48:59 to 49:33

The above data were reduced and used during the evaluation.

Because of the tape recorder failure, real-time telemetry data from 90 passes of network stations were reduced. Of these, Kennedy Space Center reduced approximately 22. All Augmented Target Docking Adapter (ATDA) data are from real-time telemetry data. ATDA data from approximately 47 passes of network stations were reduced, of which 23 were reduced by KSC.

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 of selected parameters for launch, revolutions 1, 2, 3, and selected real-time site passes for revolutions 27 to 36, 42 to 45, and reentry

GLV telemetry measurements (launch)

Telemetry signal-strength recordings

MCC-H and range safety plotboards (Confidential)

Radar data

IP-3600 trajectory data (Confidential)

MISTRAM (Confidential)

Natural coordinate system

Final reduced

C-band (Confidential)

Final reduced (launch)

Natural coordinate system (reentry)

Trajectory data processed at MSC

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

Bandpass tabulations
Selected time history tabulations
Time history plots (KSC)

Orbital phase

Bandpass tabulations for revolutions 1, 2, 3, and 4

Time history tabulations for selected parameters for revolutions 1, 2, 3, 4 and selected real-time site passes for revolutions 5, 6, 12, 13, 14, 15, 17, 29 to 36, 42 to 45, and reentry

Time history plots (KSC) for revolutions 1, 2, 3, and selected real-time site passes for revolutions 12, 13, 15, 17, 18, 30, 33, 34, 44, and 45

Reentry phase

Plots and tabulations of all system parameters for the following approximate ground elapsed times

From	To
71:51:13	71:51:16
71:51:25	71:56:35
72:01:25	72:01:43
72:02:05	72:08:28
72:13:53	72:17:12
72:17:12	72:25:36
	(Tabulations only)

Event Tabulations

Sequence-of-event tabulations versus time (including thruster firings) for ascent, revolutions 1, 2, 3, 4, and selected real-time site passes for revolutions 5, 6, 12, 13, 14, 15, 17, 18, 29 to 36, 42 to 45, and reentry

Special computations

Ascent phase

IGS computer-word flow tag corrections (Confidential)

Special aerodynamic and guidance-parameter calculations (Confidential)

TABLE 12.5-I.- SUMMARY OF INSTRUMENTATION DATA AVAILABILITY - Concluded

Data description	
Ascent phase - continued Steering-deviation calculation (Confidential) MISTRAM versus IGS velocity comparison (Confidential) MOD III RGS versus IGS velocity comparison (Confidential) Orbital phase IGS Computer-word flow-tag corrections for revolution 1, 2, 3, 4, and the data found on the onboard recorder after recovery OAMS propellant-remaining computations for revolutions 1, 2, 3, 4, and 44 OAMS-thruster activity computations for revolutions 1, 2, 3, and 4 OAMS thruster valve program for revolutions 1, 2, 3, and 4 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 <u>Target Launch Vehicle (TLV) and Augmented Target Docking Adapter (ATDA) data</u> <u>Paper recordings</u> ATDA telemetry measurements for selected real-time site passes for launch and revolutions 2, 3, 5, 12, 13, 14, 15, 16, 17, 31, 32, 33, 43, 44, 45, 46, 47, 48, 57, 58, 59, 60, 61, and 63	<u>Paper recordings</u> - Continued TLV telemetry measurements (launch) MCC-H and Range Safety Plotboards (Confidential) <u>Radar data</u> IP-3600 trajectory data (Confidential) C-band overlapping trajectory (Confidential) Final reduced, coordinate systems 2 and 3 Trajectory data processed at MSC <u>ATDA reduced telemetry data</u> <u>Engineering units versus time</u> <u>Ascent phase</u> Time-history tabulations Time-history plots <u>Orbital phase</u> Time history tabulations for selected real-time site passes for revolutions 2, 17, 32, 43, 44, 46, 57, 58, and 63 Time-history plots for selected real-time site passes for revolutions 2, 3, 30, 43, 44, and 63 <u>Event tabulation</u> Sequence-of-event tabulations versus time (including thruster firings) for selected real-time sites for revolutions 2, 17, 30, 32, 43, 44, 46, 57, 58, and 63 <u>Special computations</u> Reaction control system thruster duration (launch) Reaction control system thruster valve program (launch)

TABLE 12.5-II.- SUMMARY OF PHOTOGRAPHIC DATA AVAILABILITY

Category	Number of still photographs	Motion picture film, feet
Launch		
TLV/ATDA	(a)	[b]1850
GLV-spacecraft	(a)	[b]7004
Recovery		
Spacecraft in water	40	475
Loading of spacecraft on carrier	35	900
Inspection of spacecraft	58	--
Boston, Massachusetts		
General activities	65	--
Inspection of spacecraft	48	1100
Postflight inspection	36	--
Inflight photography		
Augmented Target Docking Adapter	110	792
Extravehicular Activities	45	88
Reentry	--	176
General purpose	174	320

[a]Still launch-photography is not normally used for evaluation purposes.
[b]Engineering sequential film only.

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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	306
1.2-2	16	50-foot tower, 19-9	GLV possible fuel leakage	314
1.2-3	16	50-foot tower, 19-4A	GLV possible fuel leakage	306
1.2-9	16	50-foot tower, 19-1	GLV launch	170
1.2-10	16	50-foot tower, 19-5	GLV launch	180
1.2-11	16	50-foot tower, 19-7A	GLV launch	170
1.2-12	16	50-foot tower, 19-2	Spacecraft launch	73
1.2-13	16	50-foot tower, 19-7A	Spacecraft launch	69
1.2-14	16	Umbilical tower, second level	GLV Stage II umbilical	115
1.2-15	16	50-foot tower, 19-7A	GLV, engine observation	120
1.2-16	16	East launcher	GLV, possible fuel leakage	125
1.2-17	16	West launcher	GLV, possible fuel leakage	125
1.2-18	16	North launcher	GLV, engine observation	130
1.2-20	16	Umbilical tower, first level	GLV, umbilical disconnect	115
1.2-21	16	Umbilical tower, second level	GLV, umbilical disconnect	66
1.2-22	16	Umbilical tower, fourth level	GLV, umbilical disconnect	115
1.2-23	16	Umbilical tower, fifth level	GLV, umbilical disconnect	125
1.2-24	16	Umbilical tower, sixth level	GLV, umbilical disconnect	135
1.2-25	16	Umbilical tower, sixth level	GLV, umbilical disconnect	207

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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	140
1.2-27	16	Umbilical tower, top level, no. 2	J-bars and lanyard observation	100
1.2-28	16	50-foot tower, east side	Spacecraft umbilical	220
1.2-29	70	South of Pad 19	Spacecraft launch	45
1.2-31	16	North of Pad 19	Tracking	396
1.2-32	16	West of Pad 19	Tracking	380
1.2-33	16	South of Pad 19	Tracking	334
1.2-34	16	South of Pad 19	Tracking	240
1.2-35	16	South of Pad 19	Tracking	240
1.2-36	35	South of Pad 19	Tracking	175
1.2-37	35	South of Pad 19	Tracking	212
1.2-38	35	Northwest of Pad 19	Tracking	295
1.2-39	70	Northwest of Pad 19	Tracking	121
1.2-41	70	Melbourne Beach, Florida	Tracking, ROTI	40
1.2-42	35	Patrick AFB, Florida	Tracking, IGOR	91
1.2-46	16	Umbilical tower, sixth level	GLV, cable-cutter action	210

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TABLE 12.5-III.- LAUNCH PHASE ENGINEERING SEQUENTIAL CAMERA DATA AVAILABILITY - Concluded

(b) TLV and ATDA

Sequential film coverage, item	Size, mm	Location	Presentation	Total length of film, ft
1.2-4	16	East of Pad 14	Atlas engine observation	140
1.2-5	16	West of Pad 14	Atlas engine observation	150
1.2-6	16	Northwest of Pad 14	Atlas engine observation	75
1.2-7	16	Ramp, south of Pad 14	Atlas engine observation	130
1.2-8	16	West of Pad 14	Atlas launch	70
1.2-9	16	Northwest of Pad 14	Atlas launch	50
1.2-10	16	Northwest of Pad 14	Atlas vernier-engine heat shield	140
1.2-11	16	Southeast of Pad 14	Atlas vernier-engine heat shield	120
1.2-17	16	South-southwest of Pad 14	Tracking	130
1.2-18	35	West of Pad 14	Tracking	174
1.2-19	35	Patrick Air Force Base	Tracking, IGOR	345
1.2-20	70	Northwest of Pad 14	Tracking	85
1.2-21	70	Cocoa Beach, Florida	Tracking, ROTI	50
1.2-22	70	Melbourne Beach, Florida	Tracking, ROTI	191

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12.6 POSTFLIGHT INSPECTION

The postflight inspection of the spacecraft 9 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 June 9, 1966 to June 31, 1966. The rendezvous and recovery (R and R) section was not recovered. The main parachute was recovered and dispositioned to Cape Kennedy for washing, drying, and damage charting. The crew station items defined in Spacecraft Test Request (STR) 9000 had been removed and dispositioned on the prime recovery ship. In addition, several items were removed from the equipment bays aboard the prime recovery ship and treated in accordance with reference 19.

The reentry assembly was received in good condition at the contractor's facility in St. Louis. 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. The water stain on the Environmental Control System (ECS) door indicated the presence of 1 to 2 pints of water in the ECS well.

c. The right-hand gooseneck mirror was cracked.

d. A shingle on the left-hand skid-well door had a curled edge, and one hold-down washer for this shingle was missing.

e. The flashing-recovery-light door-hinge half on the spacecraft did not have a roll pin in the hinge pin to secure it.

f. A wire in the main-trunk wire bundle had two areas that appeared to have a fractured conductor or a manufacturing defect.

g. A 3/16-inch bolt attaching the left-hand side of the ECS door had the wrong grip length, resulting in a gap under the bolt head.

h. A pyrotechnic cartridge in the hoist-loop door-release mechanism was determined not to have detonated.

i. The check valve on the drinking-water tank leaked.

j. Water was found in two of the Aerospace Ground Equipment (AGE) test-point connectors.

k. A resistance of approximately 1000 ohms was measured in the main bus to ground electrical check.

l. A leak was detected in the check-valve boss of the drinking-water tank.

m. Aluminum flaresavers in the drinking water system were severely corroded.

n. The UHF descent antenna was bent approximately 30 degrees aft from the normal position.

o. The left-hand dual utility cord had a mechanical break in a wire adjacent to the connector-pin solder connection.

p. The paint on the right-hand ejection-seat egress-kit contour was blistered in places.

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, and the stagnation point was located on the vertical centerline, 19.6 inches below the horizontal centerline. The heat shield was removed and dried with the reentry assembly. The wet weight of the heat shield was 323.68 pounds without the insulation blankets. The dry weight of the heat shield in the same configuration was 314.09 pounds.

Residue similar to that found on the windows of previous spacecraft was noted, and an investigation to determine the adequacy of the protective window covers was initiated (STR 9015A). The water stain on the ECS door indicated the presence of 1 to 2 pints of water in the ECS well. A shingle on the left-hand corner of the skid-well door was curled forward as if damaged upon landing. The insulation beneath the defect was not heat damaged. The retainer washer had been sheared off, but the retaining screw was in place. The hinge half in the spacecraft for the flashing recovery light did not have a roll pin through the hinge pin to secure it. A 3/16-inch bolt attaching the left-hand side of the ECS door had the wrong grip length, resulting in a gap under the bolt head. Torques of 250 and 300 inch-pounds applied at the external hatch sockets were required to open the left-hand and right-hand hatches, respectively. The heat shield and the heat-affected areas of the exterior surface appeared similar to that of previous spacecraft after reentry. The magnitudes of the hatch closing and opening forces and the hatch actuator axial forces were measured (STR 9013). The cabin was pressurized to 5.1 psi, and the postflight leakage rate was determined to be 425 scc/min, well below the specified maximum of 1000 scc/min (STR 9014).

12.6.1.2 Environmental Control System.- The drinking water was removed and dispositioned for analysis. The results of this analysis were compared with the specification for the water (STR 9511). The total water remaining in the system was 1.96 pounds. The lithium hydroxide cartridge was removed from the ECS package and weighed. The cartridge weighed 112.71 pounds with a center-of-gravity 7.81 inches from the bottom. The cartridge was dispositioned for reuse (STR 9509). The secondary oxygen system was deserviced, and residual pressures of 36 psi for the left-hand bottle and 4692 psi for the right-hand bottle were measured.

The ECS handles were actuated in accordance with reference 18 and the maximum handle force recorded was 24 pounds on the left-hand secondary-oxygen shut-off valve.

The check valve on the drinking-water tank leaked, and a leak was detected in the check-valve boss of the drinking-water tank (STR 9029). In addition, the flaresavers in the drinking-water system were severely corroded.

The left-hand suit-inlet temperature sensor and circuitry were investigated (STR 9018). A preinstallation acceptance test of the cabin pressure regulator was performed (STR 9019). The cabin pressure relief valve was evaluated (STR 9031). The carbon dioxide partial-pressure indicator was investigated (STR 9510).

12.6.1.3 Communications System.- The external appearance of all communications equipment was good. The UHF descent antenna was inspected for normal deployment (STR 9025). The antenna was bent aft at approximately 30 degrees from the vertical.

The VOX circuit and the voice control center were tested (STR 9020A). The swimmer's interphone circuit in the spacecraft was checked out (STR 9021). The onboard voice tape recorder was tested for proper operation (STR 9028).

12.6.1.4 Guidance and Control System.- The computer and Inertial Measurement Unit (IMU) were removed onboard the prime recovery ship and dispositioned to the vendor representative in Boston, Massachusetts (STR's 9003, 9005). The Auxiliary Control Power Unit (ACPU), Attitude Control Maneuver Electronics (ACME) system, and horizon sensor electronics were removed on the prime recovery ship, returned to

St. Louis, and dispositioned to the vendors (STR's 9006, 9007, and 9008). An investigation of the power supply, computer, START COMP switch, and circuit wiring was conducted (STR 9022A).

12.6.1.5 Pyrotechnic system.- Pyrotechnic resistance checks were performed on all electrically initiated pyrotechnic devices in the re-entry assembly in accordance with reference 18. Two pyrotechnic devices had bridge resistances which were near the unfired range and were removed for visual inspection. It was determined that one of the two had fired, but the hoist-loop and flashing-light cartridge had not denotated. STR 9505 was initiated to investigate the hoist-loop door circuitry. The cartridge was removed and detonated under controlled conditions (STR 9508).

The postflight visual inspection of the wire-bundle guillotines, parachute bridle-release mechanisms, and other pyrotechnics disclosed that all appeared to have functioned normally.

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 appeared to have 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) programmer and multiplexers were removed on the prime recovery ship and dispositioned to the vendor representative at Boston, Massachusetts (STR 9001). The instrumentation package 2 was removed on the prime recovery ship and returned with the spacecraft to St. Louis (STR 9002A). The PCM tape recorder was checked out in St. Louis and returned to the vendor for failure analysis (STR 9010).

The dc-to-dc converters were removed on the prime recovery ship and returned to St. Louis (STR 9500). The telemetry transmitters were removed during the spacecraft evaluation at St. Louis and placed in bonded storage (STR 9500).

The biomedical tape recorders were removed on the prime recovery ship and immediately flown to the Manned Spacecraft Center (MSC) for data processing (STR 9000):

12.6.1.7 Electrical System.- The main batteries and the 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 batteries still delivering the current specified in reference 18.

Main battery	Discharge, A-hr	Squib battery	Discharge, A-hr
1	35.40	1	5.68
2	34.15	2	5.00
3	37.25	3	6.00
4	41.00		

The main and squib batteries were recharged and placed in bonded storage for use in ground tests.

A wire in the main -257 trunk wire bundle at Z140.50 had two areas that appeared to have conductor fracture or a manufacturing defect.

The AGE test-point inspection was conducted per reference 18. Water was found in AGE test points 13 and 148 behind access doors 32 and 39, respectively.

A resistance of approximately 1000 ohms was measured when the main battery switches were actuated during the electrical check to determine current leakage due to salt-water immersion. The investigation was narrowed to the 52-77610 relay panel, and the panel was removed for pre-installation acceptance testing (STR 9507).

An investigation was conducted to determine whether the load sharing and variations in test readings of the main batteries were due to wiring (9502). The open-circuit voltage of each individual main battery cell was measured (STR 9504).

Both dual utility cords were removed and placed in the failure analysis lab (STR 9503). The left-hand dual utility cord had a mechanical break in a wire adjacent to the connector pin solder connection. The right-hand cord checked out satisfactorily.

12.6.1.8 Crew station furnishings and equipment.- The appearance of the cabin interior was good. The switch positions and instrument panels were photographed in accordance with reference 18. The ejection seats were removed and deactivated in accordance with reference 18. The backboard contours, pelvic blocks, and lap belts were placed in bonded storage at the contractor's plant in St. Louis. The seat ballast was shipped to the Kennedy Space Center (KSC) for reuse. The survival kits and suit-hose interconnects were shipped to MSC per STR 9000.

The right-hand gooseneck mirror was cracked horizontally across the mirror.

The Velcro bonded to the exterior surface of the spacecraft forward of the right-hand window had melted and run in the aft direction.

The water-metering dispenser and astronaut retractable pencils were removed and dispositioned to MSC (STR 9000).

An investigation of the difficulty encountered in inserting the safety pin for the ejection control mechanism of the right-hand ejection seat was conducted (STR 9030). A point on the right-hand ejection-seat egress kit contour was blistered as if from direct heating.

12.6.1.9 Propulsion System.- The Reentry Control System (RCS) thrust chamber assemblies appeared normal. The RCS section was deactivated at Boston, Massachusetts. The A-ring propellant tanks were empty; however, propellant was obtained from the B-ring, and this was transported to KSC for analysis. Purge gas samples were also sent to KSC for analysis, and the results were recorded in reference 18.

The performance of the motor-operated shutoff valves was evaluated, and the valves functioned normally with no leakage (STR 9027). The RCS section was removed from the spacecraft, and a system gas flow of all engines in the B-ring was performed (STR 9032). Thrusters 3A, 6A, 4B, and 8B were removed and tested (STR 9023). Thrusters 8A and 4B were placed in the failure analysis lab and the electrical riser tubes sectioned to allow visual inspection of the solder joints and lead wires (STR 9501).

12.6.1.10 Landing System.- The main parachute was returned to the KSC for washing, drying, and damage charting. The parachute will be returned to MSC for further analysis (STR 9012). The R and R section was not recovered.

12.6.1.11 Postlanding recovery aids.- The flashing recovery light and the hoist-loop door appeared to have functioned normally. The sea dye marker was removed on the prime recovery ship and returned to St. Louis as a loose piece.

12.6.1.12 Experiments.- No effort was expended on experiments during the postflight evaluation at the contractor's facility in St. Louis.

12.6.2 Continuing Evaluation

The following is a list of STR's that were approved for the post-flight evaluation of reported spacecraft anomalies.

STR number	System	Purpose
9009A	Crew Station	To evaluate performance of Extra-vehicular Life Support System
9010	Instrumentation and Recording	To determine the malfunction of PCM tape recorder.
9011	Crew Station	To perform evaluation of pilot's space suit.
9013	Structure	To determine the magnitude of the hatch opening and closing loads and compare with preflight values.
9015A	Structure	To determine the adequacy of the protective window covers.
9017	Crew Station	To perform transmittance evaluation of pilot's pressure visor.
9018	Environmental Control	To investigate cause of failure of left-hand suit-inlet temperature sensor or circuitry.
9019	Environmental Control	To investigate cause of cabin-pressure decay and subsequent build-up to a higher-than-normal level.

STR number	System	Purpose
9020A	Communications	To test VOX circuitry and voice control center because of poor communications during extravehicular activity.
9021	Communications	To investigate cause of intermittent voice communications between the swimmers and the flight crew.
9022	Guidance and Control	To determine the cause of inadvertent appearance of the computer start compute discrete.
9023	Propulsion	To determine postflight performance of RCS thruster fuel valves.
9507	Electrical	To investigate 1000-ohm resistance reading from main bus to spacecraft ground.
9508	Pyrotechnic; Electrical	To fire a hoist-loop door guillotine cartridge found undetonated.
9510	Instrumentation	To investigate possible anomaly in carbon dioxide partial-pressure sensor.

12.7 TARGET LAUNCH VEHICLE 5303 FLIGHT EVALUATION

This section contains excerpts from the General Dynamics/Convair Division Space Launch Vehicle Flight Evaluation Report, SLV-3 5303, GDC/BFK66-029. Only those portions of the report that describe the problem area are included in this section.

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FLIGHT DATE: 17 MAY 1966
ISSUE DATE: 27 JUNE 1966

SLV 3

GENERAL DYNAMICS
Convair Division

GDC/BKF66-029
ATLAS SLV-3
SPACE LAUNCH VEHICLE
FLIGHT EVALUATION REPORT

SLV-3 5303 (U)

CONTRACT AF04(694)-240

PREPARED BY TEST EVALUATION GROUP

APPROVED BY [signature]
A. KALITINSKY — ASSISTANT PROGRAM DIRECTOR, ENGINEERING - LAUNCH VEHICLES

APPROVED BY [signature]
G. L. HANSEN — VICE-PRESIDENT, LAUNCH VEHICLE PROGRAMS

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SECTION 1

INTRODUCTION

Atlas 5303, the twenty-third standardized-Atlas space launch vehicle (SLV-3) flight article, was launched from the Air Force Eastern Test Range (AFETR) Complex 14 at 1015 hours Eastern Standard Time (EST) on 17 May 1966. This was the first attempt to launch this vehicle.

The primary Atlas objective of placing the upper stage vehicle into the specified coast ellipse was not accomplished, due to the loss of pitch control of the B2 thrust chamber at 120.63 seconds. Cryogenic leakage in the thrust section may be related to the loss of pitch control.

This report was prepared for the Air Force Space Systems Division (AFSSD) produced under the AF04(694)-240 Standardized Space Launch Vehicle Contract, and summarizes the analysis of the Atlas SLV-3 boost vehicle operation only. This report also presents, as applicable, the current status of an integrated analysis of the associate contractor systems (General Electric, NAA Rocketdyne, and Acoustica Associates). This task was accomplished as a systems integration responsibility under the AF04(695)-710 SLV-3 Space Boosters Contract.

The Convair test number for this operation was P4-701-00-5303; and the AFETR designation was test number 2398. All times in this report are referred to 2-inch vehicle motion (liftoff) as zero time. This occurred at 1015:03.422 hours EST, as determined at range control.

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SECTION 2

SUMMARY

Atlas SLV-3 5303, the third Atlas boost vehicle to be used in support of this particular Atlas/Agena program, was launched from Complex 14, AFETR, on 17 May 1966. This was the seventieth Atlas to be employed in the Atlas/Agena programs. The flight was not successful and the mission was not satisfied. The primary objective of placing the upper stage Agena vehicle into the specified coast ellipse was not satisfied due to the loss of B2 pitch control after 120.63 seconds. Although the exact reason for the loss of B2 pitch control could not be isolated, analysis and laboratory testing has shown that the most probable cause of the flight failure was a low impedance (100-ohm) short to ground of the B2 servo amplifier (-) output command signal. This short to ground may have been caused by cryogenic leakage in the thrust section.

Tracking film showed that after the loss of pitch stability, the vehicle pitched downward in excess of 180 degrees, and changed in azimuth towards the left (northward). Flight control data substantiated that the vehicle pitched downward, and extrapolated and integrated data indicated that the vehicle pitched down 216 degrees from the 67 degree reference at 120.6 seconds. Radar data from the Grand Bahama Island station at 436 seconds, approximately 136 seconds after VECO, placed the vehicle about 103.4 nautical miles from the launch site, headed northerly at 97,000 feet altitude and descending. This data correlates well with a set of radar impact coordinates which place vehicle impact 107 nautical miles from the launch site in a north-easterly direction.

SYSTEMS OPERATION

Flight Control. Operation of the flight control system was not satisfactory. At 120.628 seconds the B2 thrust chamber began an uncontrolled movement toward the negative gimbal limit at varying rates of from 7.7 degrees per second to greater than 17.9 °/sec. After reaching the gimbal limit, the B2 thrust chamber remained there through staging. The B1 thrust chamber, acting on proper flight control commands, was not able to completely correct for the pitch-down vehicle attitude and the actual vehicle trajectory diverged from the required trajectory. At the same time that the B2 thrust chamber began the uncontrolled pitch movement, small negative null shifts were observed in the vernier pitch and yaw channels. It is believed that the booster and sustainer engines also had experienced slight null shifts, however, the very low engine position gains of the sustainer and booster thrust chambers would have made small null shifts extremely difficult to detect in the telemetered data. These null shifts were caused by excessive loading of the -60 vdc power supply and were the key factor in identifying a low impedance short to ground as the most probable failure mode.

At BECO the B2 pitch position indicated the usual relaxation of thrust loading by a slight relative positive pitch position shift. This structural shift does not change the actuator position, but is recorded by the telemetry position transducer as a relative movement between the thrust chamber and the gimbal mount. Also, as the sustainer engine moved in pitch after activation at BECO, the B2 thrust chamber displayed no pitch motion. Both the small relaxation motion and the lack of movement in response to sustainer pitch control verified B2 pitch actuator integrity.

The flight data showed that the vernier engine null shifts were no longer evident after jettison of the booster package. This indicates that the source of the null shift was in the booster portion of the flight control system or in an area significantly affected by the booster jettison environment.

The vehicle nose-down pitch acceleration was stopped at 137.24 seconds after activation of the sustainer and vernier engines in pitch and yaw at BECO. Good stability was regained in all channels by 150 seconds. At this time, the vehicle had pitched down a total of 231 degrees from the 67 degree reference at 120.6 seconds. The pitch angle then backed off 15 degrees to the displacement gyro null. The vehicle pitch angle, measured from the inertial vertical at liftoff, was approximately 283 degrees by 150 seconds.

An investigation team was organized to direct the 5303 failure investigation effort. The major areas of study were an examination of the vehicle preflight history, review and analysis of all flight data, a comparison of flight data with previous flight failures, the development of a test plan, and implementation of the tests.

The test program, resulting from the investigation team effort, accomplished the following component testing:

a. Actuator cryogenic impingement.
b. Actuator cryogenic soak tests.
c. Hydraulic line cryogenic soak/freeze tests.
d. Blocked actuator pull tests.
e. Harness cryogenic vibration tests.
f. Feedback transducer failure tests.
g. B2 pitch servoamplifier output shorting tests.
h. Harness short or open testing.
i. Booster servo valve short or open testing.

The most probable failure mode isolated during the special post-flight testing and analysis is an electrical short, which might also be related to cryogenic leakage. This hypothesis is supported by the flight data which indicates that a null shift occurred in the vernier engines at the same time that the B2 pitch control was lost; and by the belief that this same shift occurred in the B1 and sustainer systems.

Laboratory testing duplicated the flight failure mode with a low impedance short-to-ground of the negative (-) servo amplifier output command signal to the servo valve torque motor. The most probable location of the short was determined to be in the electrical circuit between the booster side of the staging disconnect plug and side A of hydraulic servo valve coil. However, there is the possibility that the short occurred in the sustainer side of the disconnect plug and was corrected at BECO/staging.

The exact cause of the short cannot be determined. However, cryogenic leakage in the thrust section during flight was conclusively demonstrated by the ambient temperature on the Quad IV jettison rail support (P671T) and the sustainer fuel pump discharge pressure measurement (P330P). P671T began a temperature decay from 61°F at 65 seconds and reached -50°F at 85.5 seconds. The temperature remained below -50°F (the lower band limit) until 228 seconds when it rose from -50°F to -17°F by SECO. P330P indicated a frozen sense line at 116 seconds and never, thereafter, displayed normal pump discharge pressure trends. The source of this lox leakage has not been specifically identified, and could be in either Convair or Rocketdyne components.

This cryogenic leakage assumes significance for this flight failure when compared with results from some of the special tests. During laboratory cryogenic impingement tests of servo cylinder specimens, anomalous reactions were noted. In one test the actuator lost control momentarily several times but recovered before the actuator could extend; in another test the servo cylinder extended fully for one second. In both of these cases LN2 was impinging on the servo cylinder body just below a standard flight-type connector. One explanation for this behavior is a short in the harness/connector, generated by the cryogenic environment.

As a result of the investigation, corrective action as outlined below was taken to preclude recurrence of this failure mode.

a. X-ray servo valve, feedback transducer plug, and servo valve torque motor wiring. (Surveys 38-66 and 40-66)

b. Manual flex test on servo amplifier and excitation transformer connectors with hydraulic pressure up and autopilot on. (TWX memo)

c. Perform megger checks of certain specified components and plugs. (TWX memo)

d. Fiberglass wrapping of certain harnesses. (ECPs 3681, 3337, 55-1525; ECP 3650)

e. Retorque lox dome bolts of sustainer engine (R/D FEB R 66-17) and perform 10 psi lox dome leak check, with main propellant valves closed and throat plugs in (Procedure 69-92050 change).

f. Perform a 1000 cycle low frequency autopilot gimbaling check at a low amplitude (0.25 to 0.50 cps). (Parameters document change)

The results of this failure investigation are documented under separate cover in Convair Report BKF66-041, Failure Investigation Report-Vehicle 5303.

Table 2-1 presents events of interest during the flight of vehicle 5303.

TABLE 2-1. ATLAS SLV-3 5303 FLIGHT SEQUENCE OF EVENTS

Time(sec)	Event
00.0	Vehicle liftoff, 1015:03.422 hours EST.
54.5	Guidance rate lock established.
58.5	Guidance track lock established.
65.	Ambient temperature in Quad IV (P671T) starts decaying from 61°F, at 6°F per second. This cooling trend is not reflected by the other thrust section temperature instrumentation.
75.	Ambient temperature at the sustainer instrumentation panel in Quad I (A743T) reaches a minimum value of 43°F, then increases gradually.
85.5	Ambient temperature in Quad IV reaches lower instrumentation limit of -50°F.
97.3	Booster pitch steering initiated.
101.8	Booster pitch steering terminated.
111.5	Ambient temperature at the sustainer instrumentation panel (A743T) starts rise from 52°F.
111.5	Ground guidance station reports signal strength fluctuations. Signal strength at vehicle shows a small but not abnormal, decrease.
116.	Sustainer fuel pump discharge pressure (P330P) starts increase from 910 psia.
120.0	Ambient temperature at the sustainer instrumentation panel (A743T) reaches 79°F. Temperature then slowly rises to 82°F by BECO.
120.5	Sustainer fuel pump discharge pressure reaches 1050 psia.
120.63	B2 pitch started negative, ~ 1% shift.
120.69	Vehicle CCW roll transient starts. Maximum roll was 0.5°.

TABLE 2-1. ATLAS SLV-3 5303 FLIGHT SEQUENCE OF EVENTS (CONTD)

Time(sec)	Event
120.73	B2 Pitch indicates a larger negative step, ~ 16% shift at 17.9°/sec.
120.75	V1 Yaw and V2 Yaw positions slow ~ 1% negative null shift, 0.5°.
120.73	Vehicle starts nose-down acceleration at 8.1°/sec^2.
120.86	B1 Pitch starts moving positive in proper response to gyro signals.
120.96	Booster hydraulic pump discharge pressure and accumulator pressure start to reflect large demands due to gimbaling of the booster engine thrust chambers.
121.0	Guidance track signal strength starts decay, lock becomes intermittent.
121.04	Recovered from roll transient.
121.27	B2 Pitch reaches telemetry band limit.
121.35	Pitch acceleration decreases.
121.46	B1 Pitch reaches positive limit.
121.82	Pitch displacement gyro reaches telemetry limit, nose-down.
124.0	Sustainer fuel pump discharge pressure starts decay from 1050 psia.
130.977	BECO by backup accelerometer.
131.17	Verniers activated.
131.2	Sustainer hydraulic pressure shows drop from 3080 psia in response to gimbaling demands of sustainer engine at enable, recovers within one-half second.
131.218	Booster thrust decay starts.

TABLE 2-1. ATLAS SLV-3 5303 FLIGHT SEQUENCE OF EVENTS (CONTD)

Time(sec)	Event
131.35	B2 Pitch moves 0.6° positive.
131.42	Vehicle starts CW roll acceleration at 5.27°/sec^2.
131.7	Guidance rate and track lost, not reacquired throughout flight.
132.68	Roll displacement gyro reaches CW telemetry bandlimit of 3.74°.
134.072	Conax valve command.
134.094	Booster jettison on U101A.
134.1	Ambient temperature at the sustainer instrumentation panel drops abruptly from 82°F to 10°F. Data starts rising immediately, reaching 58°F by 140 seconds.
134.15	Booster position measurements drop out due to separation of instrumentation staging disconnect plugs.
135.0	Sustainer fuel pump discharge pressure temporarily stabilizes at 880 psia.
137.84	Roll displacement gyro comes off stop. Calculations indicate 1 degree loss of reference.
138.5	BECO discrete generated by ground guidance, not received at vehicle.
138.8	Sustainer fuel pump discharge pressure decay resumed from 880 psia.
190.0	Sustainer fuel pump discharge pressure reaches 300 psia on continuing decay curve.
222.53	Station 6 Acoustica fuel sensor uncovers.
224.41	Station 6 Acoustica lox sensor uncovers.

TABLE 2-1. ATLAS SLV-3 5303 FLIGHT SEQUENCE OF EVENTS (CONTD)

Time(sec)	Event
228.	Ambient temperature in Quad IV rises from -50°F; and has recovered to -17°F by SECO. Temperature data displays normal characteristics throughout the remainder of recorded flight.
240.8	Sustainer fuel pump discharge pressure increases from 250 psia.
245.0	Sustainer fuel pump discharge pressure reaches 595 psia and stabilizes.
265.35	Lox head sensing port uncovers.
269.55	Fuel head sensing port uncovers.
273.86	Sustainer engine thrust decay starts (lox depletion).
274.282	SECO by lox depletion.
	Extrapolated ΔP data indicates 38 lbs useable lox residual. Extrapolated Station 6 indicates 916 lbs useable lox residual.
277.7	Sustainer fuel pump discharge pressure initiates a decay from 595 psia.
281.3	Sustainer fuel pump discharge pressure reaches 225 psia and stabilizes.
299.245	VECO by programmer backup command (SECO plus 25 seconds).
299.357	Vernier engines thrust decay starts.
301.736	Agena separation by programmer backup (SECO plus 27.5 seconds).
320.0	Sustainer fuel pump discharge pressure initiates a very slow increase from 225 psia.
432.0	Sustainer fuel pump discharge pressure initiates a rapid increase from 255 psia.

TABLE 2-1. ATLAS SLV-3 5303 FLIGHT SEQUENCE OF EVENTS (CONTD)

Time(sec)	Event
459.0	Sustainer fuel pump discharge pressure goes above the upper instrumentation bandlimit (1500 psia).
470.0	Sustainer fuel pump discharge pressure re-enters the 1500 psia bandlimit in a step-decrease and stabilizes at 90 psia until the loss of telemetry.

Propulsion. Propulsion system operation was satisfactory during engine start, thrust buildup, transition to mainstage, and flight. Normal engine operating characteristics were reflected in all telemetered data through flight, with exception of the sustainer fuel pump discharge pressure measurement (P330P) which presented characteristics indicative of a frozen transducer sensing line. The source of the lox leakage has not been isolated to either the Convair or Rocketdyne hardware.

Starting at 116 seconds the P330P data started a gradual pressure increase from 910 psia. The pressure reached 1050 psia by 121 seconds. At 124 seconds the pressure began a decay from 1050 psia, and did not exhibit proper sustainer fuel pump discharge pressure characteristics throughout the remainder of telemetered flight. The frozen sense line hypothesis is supported by the ambient temperature on the Quad IV jettison rail support (P671T) which indicated a temperature of less than minus 50°F between 85.5 seconds and 228 seconds. The frozen P330P sense line anomaly has been noted on twelve previous Atlas flight vehicles (11D, 211D, 232D, 243D, 248D, 285D, 289D, 297D, 5301, 7115, 7118 and 67E) and corrective ECP action has been initiated to rectify the source of the problem. The critical engine control lines of this vehicle had been wrapped per APIN 66. (APIN 66 will be superceeded on downstream vehicles by ECPs MA5-213 and MA5-214.)

Propellant Controls. Operation of the Acoustica Associates (AA) propellant utilization system was satisfactory. The actual and predicted propellant residuals are presented in Table 2-2.

TABLE 2-2. USEABLE PROPELLANT RESIDUALS AT SECO

		Lox (lb)	Fuel (lb)	Total (lb)	Additional Burn Time(sec)	Excess Propellant at Depletion(lb)
Predicted		892	520	1412	Not Available	Not Available
Actual	ΔP	38	436	474	(1)	436 Fuel
Actual	Sta.6	916	432	1348	5.07	26 Fuel

Notes: Both propellant head pressure instrumentation ports uncovered. The predicted values were based on the LMSC preflight simulation, LMSC/A811727, dated 29 April 1966.

(1) Since cutoff was by lox depletion, there was no additional burn time and the fuel outage is the useable fuel residual, 436 pounds.

The reason for the large lox residual error using the Station 6 residual calculation method is under investigation. A number of previous SLV-3 flights have had large disagreements between Acoustica Station 6 and head sensing port lox residuals. Although no other flight has had as large a lox residual difference as 5303, the following vehicles have had discrepancies in excess of 300 pounds: 5301, 5302, 5303, 7106, 7108, 7113, 7114, 7115, 7117, and 7118. The exact reason for the discrepancy is not presently known, although late dry indications of either or both lox sensors 5 and 6 are suspected as contributing factors. All SLV-3 flight PU data is being reviewed in an effort to resolve the lox residual anomaly. Tanking test data is also being investigated. This is the first flight where an external source (lox propellant depletion as sensed by the propulsion system) has substantiated the validity of the Convair head sensing port residual calculation method.

Pneumatics. Operation of the ground and vehicleborne pneumatic systems was satisfactory throughout the countdown, launch, and flight. All propellant tanks pressurization and pneumatic control functions were properly performed.

Hydraulics. Hydraulic systems operation was satisfactory. Telemetered and landline data indicated that proper hydraulic pressures were maintained within the booster and sustainer/vernier subsystems throughout the launch countdown and powered flight. At 120.9 seconds the booster hydraulic system reflected large demands due to booster thrust chamber gimbaling; and at 131 seconds the sustainer system showed large demands due to gimbaling of the sustainer engine. All demands were satisfied.

Airframe. Vehicle structural integrity was satisfactorily maintained throughout power flight and beyond Atlas/Agena separation. Motion picture film and flight control system data verified that booster section jettison was accomplished under adverse vehicle pitch maneuvers. The engine compartment was satisfactory with a maximum temperature of 106°F recorded by A745T in the sustainer fuel pump inlet area at BECO.

The engine compartment temperature data from P671T, on the Quad IV jettison rail support, indicated that abnormal temperature conditions existed during the flight. The temperature in this area remained essentially constant at 61°F until approximately 65 seconds when a temperature decrease was initiated. The temperature was decaying at a rate of 6°F per second when it exceeded the lower instrumentation band limit of minus 50°F at 85.5 seconds. The temperature remained below minus 50°F until 228 seconds, and had increased to minus 17°F by SECO. Effects of the cryogenic environment were noted in the propulsion system where the sensing line for the sustainer fuel pump discharge pressure transducer (P330P) froze.

Guidance. Operation of the vehicleborne guidance system was satisfactory. However, due to the loss of vehicle pitch control at 120.6 seconds the received signal strength at the vehicle and the ground station began to attenuate at 121 seconds and rate and track lock were were lost at 131.7 seconds. Corrective booster-phase steering commands had been properly generated prior to loss of lock.

As a result of loss of lock, BECO was generated by the staging backup accelerometer, SECO was generated by the propellant depletion system (lox depletion), and VECO and Agena separation were generated by flight control programmer backup functions. These times are summarized in Table 2-3. Ground guidance did generate a BECO/staging discrete at 138.5 seconds from extrapolated data; however, the discrete was not received at the vehicle due to loss of lock.

TABLE 2-3. SUMMARY OF DISCRETE COMMANDS

Guidance Discrete	Preflight Nominal(1)	Decoder Output	Engine Relay Activation	Axial Accelerometer Indication
BECO/Staging	130.500	(2)	(7)	131.218
Start Agena Timer	278.538	(3)	N/A	N/A
SECO	279.969	(4)	274.286	273.86(8)
VECO	299.500	(5)	299.245	299.357
Agena Separation	302.000	(6)	N/A	301.747

Notes:
(1) Preflight nominals taken from LMSC preflight simulation, LMSC/A811727, dated 29 April 1966.
(2) BECO generated by backup accelerometer at 130.977 seconds. Discrete was generated by ground guidance at 138.5 seconds, but was not received at vehicle.
(3) Guidance discrete was not generated.
(4) SECO generated by propellant (lox) depletion at 274.282 seconds. Guidance discrete was not generated.
(5) VECO generated by programmer backup at 299.246 ± 0.017 seconds (SECO + 25 seconds). Guidance discrete was not generated.
(6) Agena separation generated by programmer backup at 301.736 seconds (SECO + 27.5 seconds). Guidance discrete was not generated.

(7) Time not recoverable from data.
(8) First indication of sustainer engine shutdown occurred prior to generation of lox depletion cutoff.
All times in seconds from two-inch motion, 1015:03.422 hours EST.
N/A Not Applicable.

When the BECO/staging discrete was generated by the backup staging accelerometer, the U101A, axial accelerometer indicated 6.74 g. This value is slightly above the 6.50 ± 0.2 g nominal setting of the backup accelerometer. Moreover, this accelerometer has been tested prior to flight and activated at 6.48 g in two out of three tests. The reason for the difference between the laboratory trip point of the staging accelerometer and the U101A accelerometer value at BECO is under investigation.

Range Safety Command. Operation of the range safety command system was satisfactory. Sufficient signal strength was received at the vehicle to maintain proper operation throughout the flight. No range safety functions were planned, required, transmitted, or received during the flight.

Electrical. Operation of the Atlas electrical system was satisfactory. The main vehicle battery voltage and inverter frequency and voltage were within specification throughout launch and flight operations. Analysis of all electrical measurements indicated no abnormal transients throughout flight. However, after 444 seconds the ac and dc voltage telemetry measurements were lost, probably as a result of component or harness damage resulting from adverse reentry conditions.

SUPPORTING SYSTEM OPERATION

Film Analysis. All available motion picture coverage verified a satisfactory release sequence and vehicle liftoff. All observable functions of the launcher system were satisfactory. Film items from stationary cameras showed that the nacelle doors in all four quadrants bounced before closing; however, the doors closed properly after bouncing. One film item showed that the LN2 drain duct closest to the nacelle in Quad I broke in half before it was separated by the lanyard. Tracking film from Melbourne Beach showed the abnormal vehicle motion prior to and after staging due to the loss of vehicle pitch stability.

Instrumentation. Operation of the telemetry and landline instrumentation systems was satisfactory. All of the 110 instrumented telemetry measurements and 47 analog landline measurements of vehicle functions provided valid data during the periods of interest.

Countdown. Vehicle 5303 was successfully launched on the first attempt. The planned 680 minute integrated range countdown was initiated at 0015 hours EST on 17 May 1966. One unscheduled hold of 15 minutes was called at minus 150 minutes (0905 hours EST) to allow completion of Agena oxidizer tanking. The countdown was resumed at 0920 hours EST, and proceeded to vehicle liftoff (at T-95 minutes in the range countdown) at 1015:03.422 hours EST.

TABLE 2-4. SLV-3 5303 FLIGHT OBJECTIVES

The following presents a tabulation of the Atlas flight objectives which were scheduled for the flight of Atlas SLV-3 5303, and against which data were obtained and evaluated. These test objectives are defined in the program Flight Test Plan, GDC Report No. BKF65-002.

Objective Number	Description	Priority	Satisfied
M01	Demonstrate that the SLV-3 Vehicle boosts the upper stage vehicle into the proper coast ellipse as defined by the trajectory and guidance equations.	1	No
M02	Demonstrate that the SLV-3 Vehicle properly initiates or relays commands as required for separation of the upper stage, and to start the Agena ("D") timer.	1	Yes
T01	Determine SLV-3 Vehicle systems performance utilizing telemetry data.	1	Yes
SPECIAL FLIGHT OBJECTIVE			
A09	Determine the transient shock and oscillatory pnenomena at booster and sustainer engine shutdown.	3	Yes

SECTION 3

TRAJECTORY DATA

The trajectory mission of SLV-3 5303 was not accomplished. Loss of vehicle control ten seconds prior to booster engine cutoff resulted in radical departures from the nominal trajectory. FPS-16 radar data from the Grand Bahama Island station at 436 seconds placed the vehicle about 103.4 nautical miles from the launch site headed northerly at 97,000 feet altitude and descending. This data correlates with a set of radar impact coordinates which place vehicle impact 107 nautical miles from the launch site in a northeasterly direction. A separate set of impact coordinates placing impact at 298 nautical miles east of Cape Kennedy apparently apply to the jettisoned booster section. Range safety was not jeopardized during the flight, so that no commands to terminate the flight or destroy the vehicle were required.

Prior to initial loss of pitch control at 120.63 seconds the trajectory parameters closely conformed to the preflight plan except for a slight deviation to the right of the nominal flight path as shown in Figure 3-1B.

Booster engine cutoff normally would be generated by the guidance system when 6.28 g of acceleration were attained. Because the guidance system track lock became intermittent when the vehicle pitched over, normal generation of BECO was precluded. Actual BECO occurred through generation of the backup signal when 6.5 ±0.2 g were sensed by the backup vehicleborne staging accelerometer. Axial acceleration at the start of thrust decay was 6.76 g.

Earth-relative velocity, which is shown together with axial acceleration in Figure 3-2B, reached a peak value of 7,935 feet per second at 127.98 seconds, declining to 7,762 feet per second at BECO. This loss of velocity prior to BECO is explained by the fact that the vertical and downrange components of velocity began to decline at 122.48 and 129.08 seconds, respectively, as the vehicle pitched over.

It should be noted that the acceleration shown in Figure 3-2B is sensed by an airborne instrument essentially as the thrust-to-weight ratio along the longitudinal axis of the vehicle regardless of what direction the vehicle is heading. The earth-relative resultant velocity, however, reflects rate of change of position of the vehicle with respect to the earth-fixed coordinate system. If, then, the attitude of the vehicle changes so that engine thrust vector is uprange, the net earth-fixed resultant velocity will decrease while the accelerometer may register an increase. Such is the case for SLV-3 5303 illustrated by the data in Figure 3-2B.

An indication of the actual flight path may be obtained by correlation of parts A and B of Figure 3-1. Part A represents movement in the vertical-downrange plane while part B is a plan view of movement in the horizontal or crossrange-downrange plane. In reading these plots it should be remembered that they represent the locus of the vehicle without regard to attitude. These plots show that during sustainer phase the center of gravity of 5303 described a climbing turn to the left, i.e., north. Downrange progress ceased completely at 262 seconds, becoming mostly northerly (+Y) and slightly uprange (decreasing X). Altitude began to decrease after VECO as shown in Figure 3-1A and in Figure 3-2A, which is a plot of the vertical position coordinate as a function of time.

Referring again to the plot of resultant velocity magnitude in Figure 3-2B it will be seen that the minimum velocity of 945 feet per second corresponds to the reversal of direction of net progress from downrange to uprange. A gradual buildup of the X component of velocity followed until loss of sustainer thrust at SECO. The slight increase in velocity, starting about 320 seconds, reflects the beginning of downward movement and gravitational acceleration.

Mach number and dynamic pressure were very close to nominal. The only abnormality is the abrupt decline of Mach number at 128 seconds when vehicle velocity began to decrease. These data are shown in Figure 3-3A. Plots of the upper air soundings of temperature, pressure and wind velocity used in calculating Mach number and dynamic pressure are given in Figures 3-3B and 3-3C.

Table 3-1 is a summary of Atlas trajectory performance.

TABLE 3-1. ATLAS SLV-3 5303 TRAJECTORY PERFORMANCE DATA SUMMARY

Parameter	Nominal(1)	Actual(2)
AT BOOSTER CUTOFF		
Time (seconds)	130.500	131.218
Vertical coordinate (Z)(feet)	194,530	197,950
Altitude above earth (feet)	196,175	199,740
Downrange coordinate (X)(feet)	259,730	271,125
Range along earth (n mi)	42.35	44.18
Crossrange coordinate (Y)(feet)	565	-4,405
Position azimuth from launcher (deg True)	83.726	84.780
Earth-relative velocity (ft/sec)	8,536	7,762
Flight path angle (degrees)	24.78	17.73
Axial acceleration (g)	6.28	6.76

TABLE 3-1. ATLAS SLV-3 5303 TRAJECTORY PERFORMANCE DATA SUMMARY (Contd)

Parameter	Nominal(1)	Actual(2)
AT MAXIMUM DYNAMIC PRESSURE		
Time (seconds)	66	64
Maximum dynamic pressure (lbs/ft^2)	961	973
Mach number	1.82	1.76
AT SUSTAINER CUTOFF		
Time (seconds)	279.969	273.86
Vertical coordinate (Z)(feet)	568,220	354,360
Altitude above earth (feet)	656,865	369,820
Downrange coordinate (X)(feet)	1,954,230	809,725
Range along earth (n mi)	312.19	130.96
Earth-relative velocity (ft/sec)	16,280	1,498
Flight path angle (degrees)	11.06	32.10
Axial acceleration (g)	3.08	3.15
AT VERNIER CUTOFF		
Time (seconds)	299.500	299.357
Vertical coordinate (Z)(feet)	595,475	366,910
Altitude above earth (feet)	714,850	381,305
Downrange coordinate (X)(feet)	2,270,200	780,525
Range along earth (n mi)	361.86	126.17
Earth-relative velocity (ft/sec)	16,200	1,374
Flight path angle (degrees)	10.00	2.03

Note: 1. Nominal values quoted in table and used for reference curves in this section were obtained from the preflight reference trajectory prepared by Lockheed Missile and Space Co.; LMSC/A811727, dated 22 April 1966.

2. All actual data except acceleration obtained or calculated from AFETR radar data, Stations 19.18 and 3.16. Acceleration obtained from accelerometer data, telemetry measurement U101A.

REFERENCE INFORMATION

Launch Data

Launch site	Complex 14, AFETR
Launcher coordinates	28.49131 deg N, 80.54690 deg W
Launcher orientation	104.977 deg True
Pitchover azimuth	83.850 deg True
Desired roll program	21.127 degrees, positive
Time of two-inch motion	1015:03.422 EST

Surface Weather at the Launch Site

Time recorded	1015 EST
Ambient pressure	14.75 psia
Ambient temperature	81°F
Dew point	69°F
Relative humidity	66 percent
Cloud cover	Scattered at 1800 feet, broken at 15,000 feet. No percentage of coverage given
Visibility	10 nautical miles
Wind velocity	11 knots from 190 deg True
Maximum upper wind	42 knots from 333 deg True at 44,000 feet

Reference Coordinate System

Origin	Center of launcher at plane of launch pins
Positive X axis	Downrange along pitchover aximuth, 83.850 deg True
Positive Y axis	353.85 deg True. To right of observer facing downrange
XY plane	Tangent to reference ellipsoid at elevation of origin
Z axis	Perpendicular to XY plane at origin, positive upward
Reference ellipsoid	Fischer spheroid of 1960

SECTION 11

FLIGHT CONTROL SYSTEM

Operation of the flight control system was unsatisfactory. Control of the B2 thrust chamber, in the pitch plane, was lost at 120.63 seconds due to electrical shorting of the circuitry associated with servoamplifier control of the B2 pitch servovalve. As a result of this loss of engine control, the vehicle inertial pitch reference was reoriented 216 degrees, (nose-down from the original reference at the time of the problem) before stability was regained at approximately 150 seconds. Vehicle powered flight, at the new inertial pitch reference, continued through Agena separation/retrorockets fire sequences. Acquisition of the vehicle, by guidance, was lost prior to the expected booster staging discrete time. This discrete was, therefore, initiated by the backup accelerometer at approximately 6.7 g (nominal setting is 6.5±0.2 g). Sustainer cutoff was initiated by the propellant depletion sensor (due to lox depletion), with VECO and Agena separation occurring by programmer backup commands.

FLIGHT CONTROL CONFIGURATION

The flight control system configuration for Vehicle 5303 was identical to that of Vehicle 5302, with the following functional difference:

ECP 3519 changed the vernier yaw bias angle from 50 degrees to 45 degrees.

MAINSTAGE AND LIFTOFF

Telemetered engine position shifts at mainstage were normal with a maximum of +0.33 degree experienced by the sustainer engine in the pitch plane.

Vehicle transients at liftoff were moderate and were quickly damped following autopilot activation at 42-inch motion; 42-inch motion was indicated by initial engine movements at 0.76 second. The liftoff roll transient was in the clockwise direction reaching a maximum displacement of 0.14 degree at a peak rate of 1.92 degrees/second. Rate gyros were ungrounded prior to engine ignition, as planned, and indicated the usual vibrations due to engine start.

The usual bending at liftoff consisted of a third mode in yaw at a frequency of 10.5 cps and a second mode in pitch at a frequency of 6.1 cps. Third mode bending was observed from liftoff to approximately 10 seconds, reaching a maximum peak-to-peak amplitude of 0.45 degree/second. Second mode bending was observed only between 1 second and 3 seconds. The maximum peak-to-peak amplitude was 0.40 degree/second.

BOOSTER PHASE

Roll and Pitch Programs. A pre-set clockwise roll program of 21.127 degrees was utilized to effect vehicle roll from the launcher heading of 104.977 degrees (True) to the desired pitchover azimuth of 83.85 degrees (True). Calculations using vehicle velocity vectors obtained from external radar tracking data indicated an error of 1.0 degree to the right of planned (deficient roll) at 100 seconds. This error is within expected system tolerances and tracking data resolution.

Table 11-1 presents the nominal pitch program profile utilized for this flight. This profile, in conjunction with the 0.388 degree per volt-second slaving sensitivity (PAF 0.97), yielded a nominal pitch angle of -58.9 degrees at 100 seconds. The actual vehicle pitch angle, computed using vehicle velocity vectors obtained from external radar tracking data, adjusted -2.7 degrees for nominal angle of attack and -0.4 degree for earth rotation, was -58.5 degrees. The indicated error was 0.4 degree deficient pitchover.

Small variations in phase A voltage and frequency, and pitch gyro slaving sensitivity will result in pitchover errors even though these parameters are within specified tolerances.

TABLE 11-1. VEHICLE 5303 NOMINAL PITCH PROGRAM

BOOSTER PHASE

Time (sec)	Programmer Output (volts)	Programmer Output Integral (volts-sec)	Rate (deg/sec)	Vehicle Angle (degrees)
15.0	2.4	0.0	-0.931	0.0
35.0	2.0	48.0	-0.776	-18.62
45.0	1.2	68.0	-0.466	-26.38
58.0	1.6	83.6	-0.621	-32.44
70.0	1.9	102.8	-0.737	-39.89
82.0	1.6	125.6	-0.621	-48.73
91.0	1.3	140.0	-0.504	-54.32
105.0	0.9	158.2	-0.349	-61.38
120.0	0.6	171.7	-0.233	-66.62
131.08	0.0	178.35	0.0	-69.20

TABLE 11-1. VEHICLE 5303 NOMINAL PITCH PROGRAM (Contd)

Time (sec)	Programmer Output (volts)	Programmer Output Integral (volts-sec)	Rate (deg/sec)	Vehicle Angle (degrees)
		SUSTAINER PHASE		
140.98	0.2	0.0	-0.078	-69.20
274.28	0.0	26.66	0.0	-79.54

Note: The pitch program is based upon a gyro torquing gain of 0.400 degree per volt-second, with an attenuation factor (PAF) of 0.97 which gives a nominal torquing gain of 0.388 degree per volt-second.

The booster pitch program ends 0.1 second after the BECO discrete (S236X) or the "staging backup" acceleration switch signal (S359X), whichever occurs first.

The sustainer pitch program of 0.078 degree per second was utilized from BECO discrete +10 seconds to SECO discrete.

The error contribution of each of these parameters, based upon theoretical computations using the values of voltage and frequency obtained during flight, and preflight calibration of pitch gyro slaving sensitivity, are presented in Table 11-2.

TABLE 11-2. PARAMETER ERROR CONTRIBUTION

Parameter	Direction from Nominal	Error Contribution at 100 seconds
Phase A voltage	High	0.7 deg excess
Phase A frequency	Nominal	0.0
Slaving sensitivity	High	0.3 deg excess

Net accountable error: 1.0 deg excess

Vehicle Dynamics. Low amplitude rigid body oscillations at a frequency of 0.43 cps began in pitch and yaw at approximately 50 seconds. These oscillations reached their maximum amplitudes as the vehicle passed through the region of maximum dynamic pressure (approximately 65 seconds). Maximum peak-to-peak amplitudes were:

Pitch	0.8 degree/second
Yaw	0.6 degree/second

Rigid body oscillations were completely damped by 82 seconds.

Small amplitude propellant slosh oscillations were observed from 80 seconds to 110 seconds. Slosh amplitudes did not exceed 0.2 degree/second peak-to-peak in both pitch and yaw. Associated booster engine movements, due to slosh, were negligible.

First mode bending, in pitch and yaw, was observed during the interval from 103 seconds to BECO. There was no buildup of this bending mode and maximum peak-to-peak amplitudes were less than 0.2 degree/second. Bending frequency increased as expected, from 3.9 cps at 103 seconds to 4.1 cps at BECO.

The maximum booster engine positive pitch deflections, to counteract the effects of aerodynamic forces, occurred at 75.5 seconds with B1 and B2 thrust chambers reaching an average of 0.8 degree (from post-liftoff null levels).

Phase I Guidance Steering. The programmer enabled guidance steering at 80.0 seconds; however, no steering commands were transmitted until 97.5 seconds. Guidance steering commands provided a pitch nose-down correction of less than one degree.

LOSS OF BOOSTER 2 PITCH CONTROL

Engine Movement and Null Shifts. The B2 thrust chamber began uncontrolled movement toward the negative pitch limit stop at 120.628 seconds. The maximum initial rate of movement was in excess of 17.9 degrees/second followed by a decreasing rate averaging 7.7 degrees/second. The rate again increased, during the final 0.1 second of movement, to 15.6 degrees/second. The total duration from initial movement to final movement was 0.6 second. At the same time that the B2 thrust chamber began uncontrolled movement, small negative null shifts were observed in the vernier pitch and yaw channels. V1 and V2 indicated a -1.4 degree pitch shift and a -1.1 degree yaw shift.

Although null shifts were not observed on the booster and sustainer engines, it is most probable these engines did experience small changes in null outputs of their respective servoamplifiers. The very low engine position gains of the sustainer and boosters, when compared to the high gain of the verniers, would necessarily make these null shifts extremely difficult to detect on the booster and sustainer telemetered position transducer outputs.

A series of tests were performed on Vehicle 5802, which closely duplicated the B2 pitch movements and associated null shifts of other engines. These tests consisted of placing several low impedance loads between the B2 pitch servoamplifier output and ground. The engine position transducers of all booster and vernier channels were monitored via telemetry and the respective actuator feedback transducers were monitored on landline recorders. Sustainer positions were also monitored via telemetry but were omitted from landline measurements due to availability of recorder channels.

A resistance value of 100 ohms between the out-of-phase output side of the servoamplifier and ground resulted in B2 pitch movements very nearly identical to those obtained on Vehicle 5303 during flight. The corresponding negative null shifts of the verniers were also very similar. Landline data of the other booster actuator feedback transducers also indicated these engine positions experienced very slight null changes. The telemetered position data of boosters and sustainer did not record these shifts, as expected, due to lack of sensitivity from the position transducers. Vernier null shifts were obtained on telemetered data as well as on the landline data.

The shift in null or operating position of other engines occurs as a result of coupling through the minus 60 volt power supply common to all servoamplifiers. When a short circuit or low impedance to ground is placed on a servoamplifier output, an abnormally high current flows out of the power supply every other half cycle of the 400 cycle ac supply. This occurs because the servoamplifier demodulator effectively removes the short when the series transistor is turned off. Since the short is "chopped", a 400 cycle voltage is induced in the output of the minus 60 volt power supply. The phase of the induced voltage is dependent on which side of the servoamplifier output is shorted. The ac voltage on the power supply causes a small change in the null output of all other servoamplifiers.

<u>Vehicle Motion</u>. When B2 pitch control was lost, the B1 thrust chamber began responding to the gyro errors by moving toward the positive physical limit stop of +5.0 degrees. This response slightly lagged the B2 pitch movement since a very short time was required for gyro errors to

buildup. The initial B2 pitch movements resulted in a small counterclockwise roll acceleration which was stopped at a peak rate of 1.9 degrees/second as the B1 thrust chamber and verniers (in roll only) began to respond to the gyro errors. This roll transient was a result of the cross products of inertia. Counterclockwise roll will result (under nose-down acceleration) from the mass unbalance on the left side of the vehicle. This mass unbalance is due to the location of the lox line and booster turbopump.

The vehicle pitch nose-down acceleration, before the B1 thrust chamber began responding, was 14.4 degrees/sec/sec. This acceleration leveled off the 8.3 degrees/sec/sec as B1 was in transit to the positive limit stop. With B1 at the positive limit stop, the acceleration was 1.0 degree/sec/sec and was slightly increasing due to increasing aerodynamic loading as the vehicle angle of attack became larger. The vehicle continued to accelerate nose-down with B2 at the negative limit in pitch and B1 at the positive limit in pitch. This acceleration was finally stopped by activation of the sustainer and vernier engines at BECO. The continuing pitch acceleration, with B1 at the positive stop and B2 at the negative stop, will result from pitch torque due to:

a. Actual booster limits with respect to vehicle centerline.

b. Center of gravity offset from vehicle centerline.

c. Sustainer engine alignment and drift (during booster phase) with respect to vehicle centerline.

d. Aerodynamic loading with a large negative angle of attack.

The booster cutoff discrete, from the staging backup accelerometer, was received at 130.977 seconds. Extrapolation and integration of rate gyro data at BECO, indicated the vehicle had pitched down an additional 156 degrees from the initial 67 degree pitch reference at 120.6 seconds. The B2 thrust chamber remained at the negative pitch limit throughout the booster cutoff and booster jettison sequences.

When booster thrust decayed, the B2 pitch position indicated the usual relaxation of thrust loading by a relative slight positive pitch position shift. This is the usual structural shift which does not change the actuator position (that is, the actuator remained at the physical extended limit) but is recorded by the telemetered position transducer as a relative movement between the thrust chamber and the gimbal mount. This slight relaxation indicated that the B2 thrust chamber was not beyond its physical limit due to a broken actuator. This was not apparent before the relaxation since the engine limit with thrust was also the limit of the telemetered information band. Further verification of actuator integrity was received when the B2 thrust chamber did not move upward as the sustainer engine, which was activated at BECO, moved to its positive pitch limit and accelerated the aft end of the vehicle downward.

The sustainer and vernier engines were activated in pitch and yaw at BECO and were commanded to move to their respective positive limits in pitch, by the large gyro errors. These engines were commanded at flow limiting rates. The sustainer engine movement resulted in a clockwise roll acceleration as expected, due to the cross products of inertia. Since the vernier engines were being commanded to the pitch limit stops at flow limit rates, these engines could not respond in roll until the pitch error signal was reduced or the roll error signal became sufficiently large to command a differential pitch motion. As the roll acceleration continued, the roll error signal reached a level which allowed V2 to move off the stop to a position of 50.2 degrees. Calculation of gyro signal amplifier limits at this time indicated the commands to V1 and V2 were for 95 degrees of pitch and 44.8 degrees of roll. The proper differential between V1 and V2 could not be obtained due to 70 degree pitch limit stops. Therefore, roll control by the verniers was only partially effective. The roll transient reached a peak rate of 6.7 degrees/second in the clockwise direction.

At 137.24 seconds, the pitch nose-down error signal was reduced by the sustainer engine and a pitch up rate caused the sustainer engine to move toward the negative pitch limit stop. Vernier 2 also moved negative in pitch at flow limit rate while V1 remained at the positive stop due to the roll signal. The verniers were again ineffective in roll while the cross products of inertia due to the sustainer accelerating the vehicle nose-up, drove the vehicle sharply counterclockwise. The peak roll rate reached was 25 degrees/second. Complete roll control was regained at BECO + 6.7 seconds when the verniers were deactivated in pitch, allowing them to control fully in roll. Good stability was regained in all channels by 150 seconds. At this time, the vehicle had pitched down an additional 231 degrees from the 67 degree reference at 120.7 seconds. The pitch angle then backed off 15 degrees to the displacement gyro null. The vehicle pitch angle, measured from the inertial vertical at liftoff, was approximately 283 degrees by 150 seconds.

The vehicle was also re-oriented approximately 1 degree clockwise in roll, with respect to the original reference, due to exceeding the roll displacement gyro physical limit during the roll transient following sustainer activation.

Vernier engine null shifts were no longer present after jettison of the booster section, the original problem was in the booster control system only.

Cause of B2 Pitch Control Loss. As a result of the post flight investigation of possible causes of the loss of B2 pitch control it was concluded that the most probable cause of the failure was a low impedance short-to-ground of the negative (-) Servo Amplifier Output Command signal to the servo valve torque motor. The electrical circuit between the booster side of the staging disconnect plug and side A of the B2 hydraulic servo valve coil is suspected of shorting. Although, the cause of the short could not be isolated, the thrust section cryogenic leakage on this flight (as indicated by P671T and P330P) could have been a contributing factor.

In addition to the shorts associated with the servo valve amplifier described previously, several other failure modes were investigated but were considered unlikely due to test results obtained. These tests included:

a. Actuator cryogenic impingement.

b. Actuator cryogenic soak tests.

c. Hydraulic line cryogenic soak/freeze tests.

d. Blocked actuator pull tests.

e. Harness cryogenic vibration tests.

f. Feedback transducer failure tests.

For details on these additional tests, refer to Convair Report BKF66-041, Failure Investigation Report-Vehicle 5303.

STABLE VEHICLE MOTION

After the vehicle had regained stability, the usual rigid body oscillations were observed in both pitch and yaw. These oscillations were of low amplitude, not exceeding 0.4 degree/second peak-to-peak in both pitch and yaw. These oscillations were completely damped in yaw by 190 seconds and continued in pitch up to sustainer cutoff.

Sustainer cutoff was initiated by oxidizer depletion at 274.282 seconds. There were no vernier phase guidance steering corrections due to loss of guidance track prior to BECO. A low amplitude roll limit cycle was evident throughout vernier phase. Maximum peak-to-peak amplitude of these oscillations was 0.31 degree/second. Limit cycle oscillations of this type have been observed on previous vehicles and are in no way detrimental to vehicle stability. These oscillations are due to small control system "dead zones".

Vernier cutoff and Agena separation/retrofire sequences were initiated by programmer backup signals at SECO + 25 and SECO + 27.5 seconds, respectively. These events appeared entirely normal.

All expected programmer switching functions were generated and properly executed.

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TABLE 11-3. ATLAS SLV-3 5303 FLIGHT EVENTS AND PROGRAMMER SWITCHING FUNCTIONS

Programmer Switch or Initiated by	Function	Reference Time	Design Time(3)	2-Inch Motion Flight Time(1)	Actuation From Reference(Flight)
--	Flight programmer start (S1235X)	2-inch motion	+0.0	0.18	0.18
11	High roll torquer excitation	2-inch motion	+2.0	Indeterminate	-
10	Enable roll program (S61D)	2-inch motion	+2.0	2.10	2.10
10	Disable roll program (S61D)	2-inch motion	+15.0	15.08	15.08
11	Low roll gyro torquer excitation	2-inch motion	+15.0	Indeterminate	-
Time Slot	Start booster pitch program (S190V)	2-inch motion	+15.0	15.03	15.03
6	Replace 4.7 cps with 13.7 cps filter section (P&Y)	2-inch motion	+24.0	Indeterminate	-
7	Guidance enable pitch and yaw (S190V)	2-inch motion	+80.0	Indeterminate	-
Time Slot	Enable staging discrete	2-inch motion	+122.0	Not Measured	-
Guidance	Staging discrete (S236X, Programmer Input) (T1)	2-inch motion	--	130.997	130.997
--	Staging backup acceleration switch (S359X)	2-inch motion	--	130.997	130.997
12	Booster cutoff signal/PVT (S291X, Programmer Output)	T1	+0.1	131.060-131.093	0.063-0.096
Time Slot	End booster phase pitch program (S190V)	T1	+0.1	131.06	0.063
2	Activate sustainer pitch and yaw (S256D)	T1	+0.1	131.16	0.163
3	Activate verniers pitch and yaw (S260D)	T1	+0.1	131.09	0.093
--	Booster flight lock-in relay (P616X)	T1	--	Not Recovered	-
--	Axial acceleration indicated BECO (U101A)	T1	--	131.218	0.221
1	Zero booster (pitch, yaw, and roll) (S253D)	T1	+1.0	132.02	1.023
5	Null integrator pitch and yaw	T1	+1.0	Indeterminate	-
2	Zero sustainer pitch and yaw (S256D)	T1	+3.0	134.09	3.093
13	Conax valve signal (booster section jettison signal - M32X)	T1	+3.1	134.072	3.075
13	Booster jettison (Programmer Output, S290X)	T1	+3.1	134.067-134.102	3.070-3.105
--	Axial acceleration indicated jettison (U101A)	T1	--	134.094	3.097
2	Reactivate sustainer pitch and yaw	T1	+3.7	134.69	3.693
Time Slot	Start sustainer pitch program (S190V)	T1	+10.0	141.02	10.023
3	Disable verniers in pitch and yaw (S258D)	T1	+6.7	137.64	6.643
4	Bias verniers to 45-degree (yaw) (S260D)	T1	+6.7	137.69	6.693
5	Reactivate integrator pitch and yaw	T1	+6.7	Indeterminate	-
Time Slot	Enable SECO and VECO	T1	+56.0	186.898-186.932	55.901-55.935
16	Enable ISS, SDT, RDT, Jettison Shroud, and Fuel depletion switch (S290X, Programmer Output)	T1	+56.0	186.898-186.932	55.901-55.935

Notes: (1) Engine activation and nulling times, enable and disable booster roll and pitch programs, and programmer outputs times are necessarily approximate due to commutation rate of telemetry channel and readability. Times have been adjusted for decommutation filter delays.

(3) Nominal design times received from Autopilot Requirements and Constraints - Drawing No. 2-00091.

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TABLE 11-3. ATLAS SLV-3 5303 FLIGHT EVENTS AND PROGRAMMER SWITCHING FUNCTIONS - Concluded

Programmer Switch or Initiated by	Function	Reference Time	Design Time(3)	2-Inch Motion Flight Time(1)	Actuation From Reference(Flight)
Guidance	Sustainer cutoff discrete (S241X, Programmer Input) (T2)	T1	--	274.282	143.285
20	Sustainer cutoff signal (S291X, Programmer Output)	T2	+0.0	274.269-274.303	-0.013 to +0.021
20	Disarm Agena "Premature Separation Destruct (PSD)"	T2	+0.0	274.269-274.303	-0.013 to +0.021
--	Sustainer cutoff relay (P347X)	T2	--	274.286	0.004
3	Reactivate verniers in pitch and yaw (S260D)	T2	+0.0	274.27	-0.012
Time Slot	End sustainer pitch program (S190V)	T2	+0.0	274.33	0.048
5	Null integrator pitch and yaw	T2	+0.0	Indeterminate	-
--	Axial acceleration indicated SECO (U101A)	T2	--	273.86	-0.422
Guidance	Agena "Start D Timer" discrete (Y41X)	T2	--	Not Sent	-
Guidance	Vernier cutoff discrete (S245X, Programmer Input)	T2	--	Not Activated	-
19	Vernier cutoff signal (S291X, Programmer Output) (T3)	T3	+0.0	299.244 (5)	299.244
19	Uncage Agena gyros, arm Agena separation, disarm PSD, eject horizon sensor fairing	T3	+0.0	299.229-299.263	-0.015 to +0.019
9	Rate gain change	T3	+0.0	Indeterminate	-
--	Vernier cutoff relay (P77X)	T3	--	299.245	0.001
--	Axial acceleration indicated VECO (M79A)	T3	--	299.360	0.116
Time Slot	Vernier cutoff backup	T2	+25.0	299.229-299.263	-0.015 to +0.019
Guidance	Agena "ISS" discrete (S248X)	T3	--	301.736	2.492
--	Agena separation/retrocockets fire (U101A)	T3	--	301.747	2.503
21	Agena "ISS" backup (S248X)	T2	+27.5	301.736	2.492

Notes: (1) Engine activation and nulling times, enable and disable booster roll and pitch programs, and programmer outputs times are necessarily approximate due to commutation rate of telemetry channel and readability. Times have been adjusted for decommutation filter delays.
(2) Airborne decoder output times. All discretes were superimposed blips.
(3) Nominal design times received from Autopilot Requirements and Constraints - Drawing No. 2-00091.
(4) Programmer backup.
(5) Calculated from known programmer delay time and inverter frequency.

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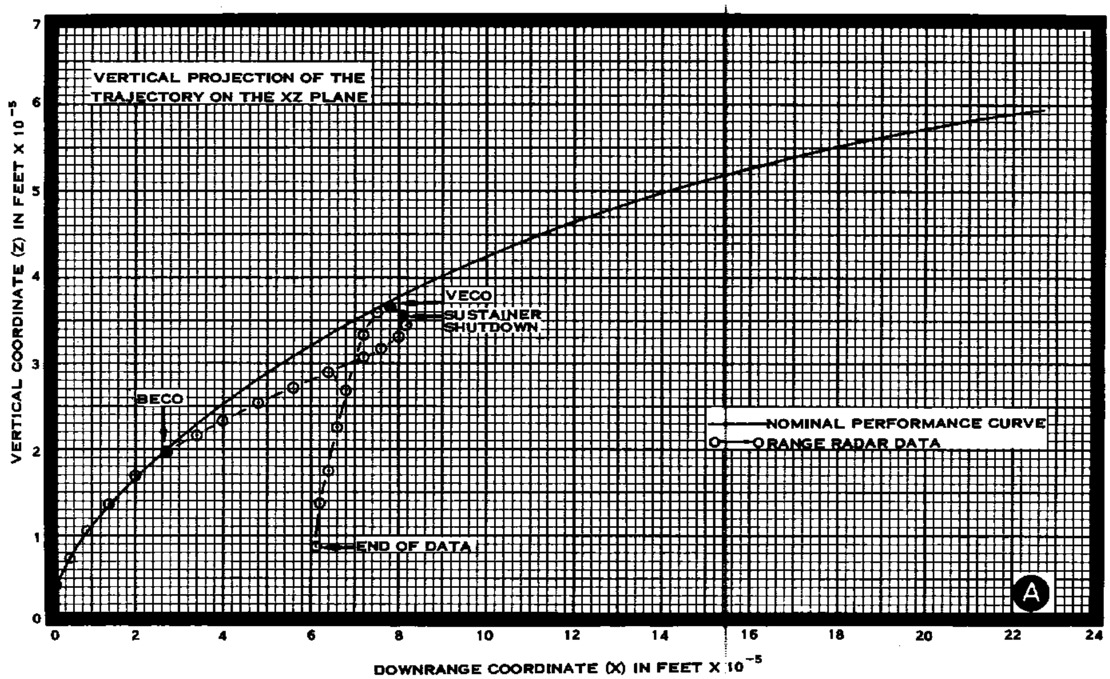
VERTICAL PROJECTION OF THE TRAJECTORY ON THE XZ PLANE
VECO
SUSTAINER SHUTDOWN
BECO
END OF DATA
NOMINAL PERFORMANCE CURVE
ORANGE RADAR DATA
A
VERTICAL COORDINATE (Z) IN FEET X 10^-5
DOWNRANGE COORDINATE (X) IN FEET X 10^-5

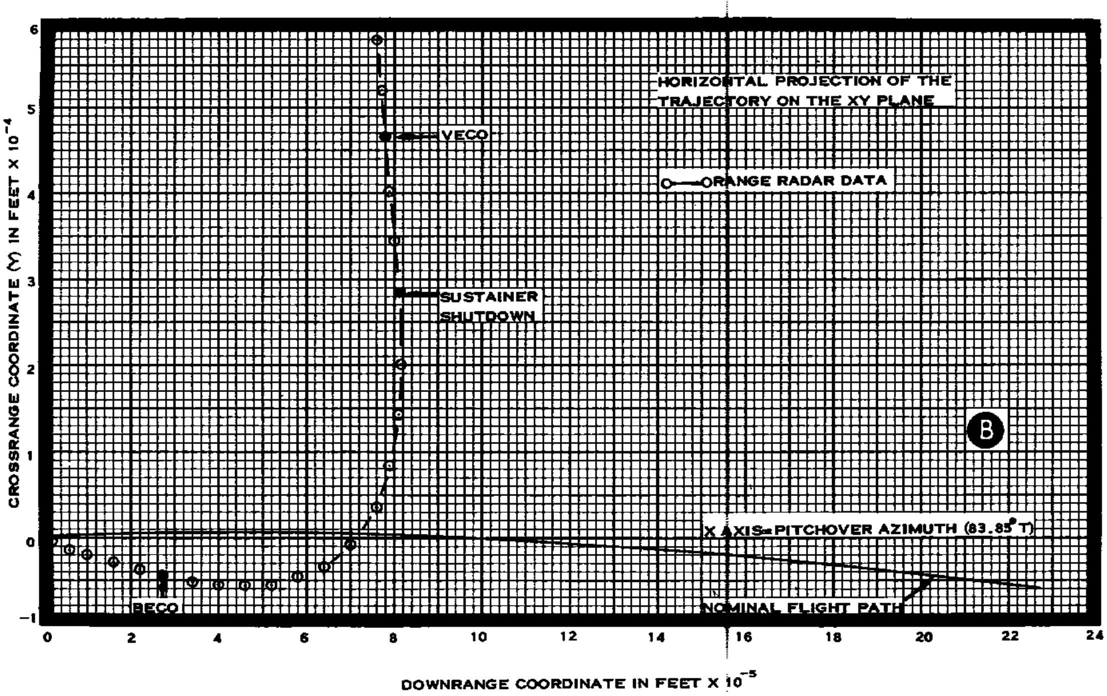
HORIZONTAL PROJECTION OF THE TRAJECTORY ON THE XY PLANE
VECO
ORANGE RADAR DATA
SUSTAINER SHUTDOWN
B
X AXIS-PITCHOVER AZIMUTH (83.85° T)
BECO
NOMINAL FLIGHT PATH
CROSSRANGE COORDINATE (Y) IN FEET X 10^-4
DOWNRANGE COORDINATE IN FEET X 10^-5

FIGURE 3-1.

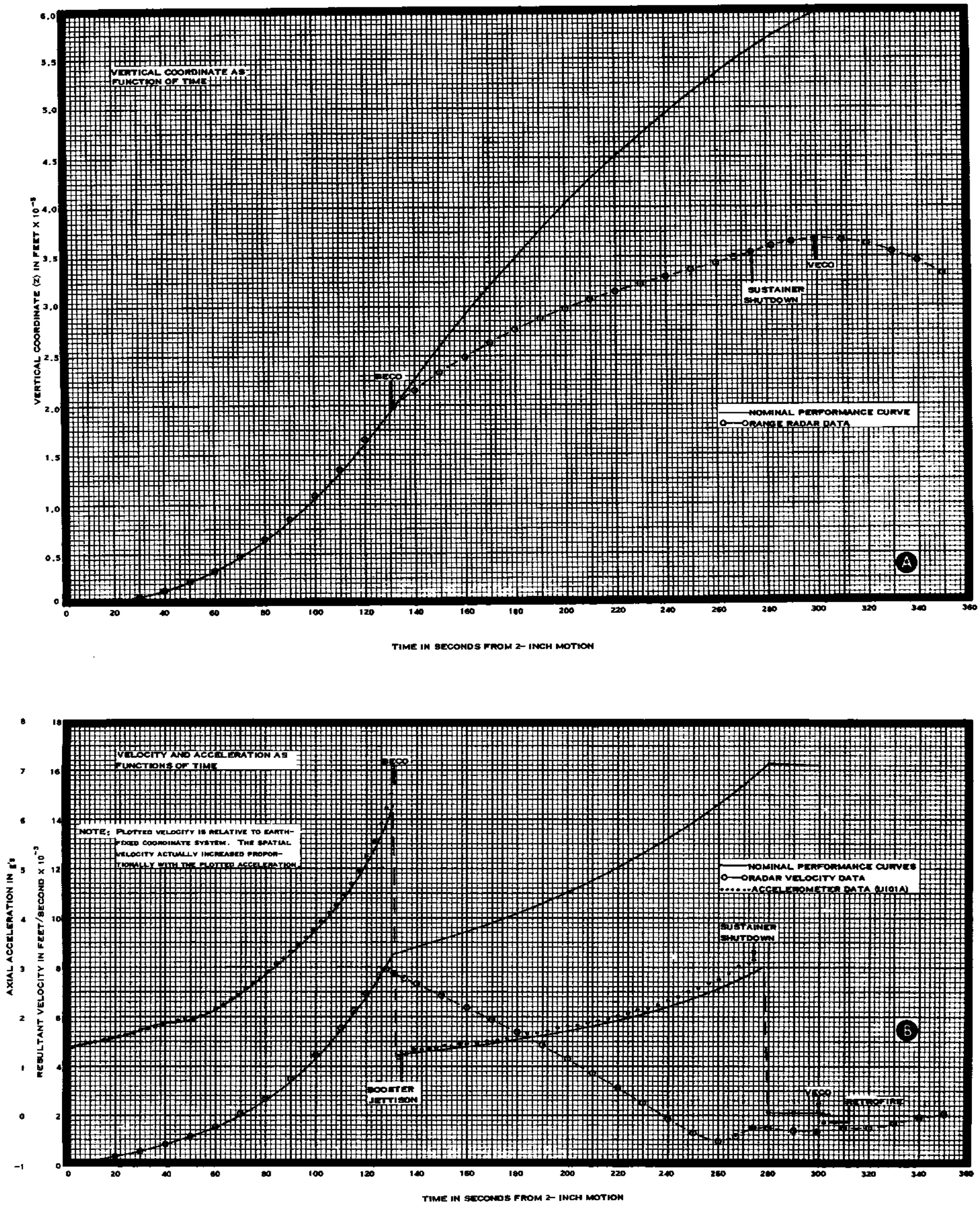
VERTICAL COORDINATE AS FUNCTION OF TIME
VERTICAL COORDINATE (Z) IN FEET X 10^{-3}
BECO
SUSTAINER SHUTDOWN
VECO
NOMINAL PERFORMANCE CURVE
ORANGE RADAR DATA
A
TIME IN SECONDS FROM 2- INCH MOTION
VELOCITY AND ACCELERATION AS FUNCTIONS OF TIME
NOTE: PLOTTED VELOCITY IS RELATIVE TO EARTH-FIXED COORDINATE SYSTEM. THE SPATIAL VELOCITY ACTUALLY INCREASED PROPORTIONALLY WITH THE PLOTTED ACCELERATION.
BECO
BOOSTER JETTISON
NOMINAL PERFORMANCE CURVES
RADAR VELOCITY DATA
ACCELEROMETER DATA (U101A)
SUSTAINER SHUTDOWN
VECO
RETROFIRE
B
AXIAL ACCELERATION IN g's
RESULTANT VELOCITY IN FEET/SECOND X 10^{-3}
TIME IN SECONDS FROM 2- INCH MOTION

FIGURE 3-2.

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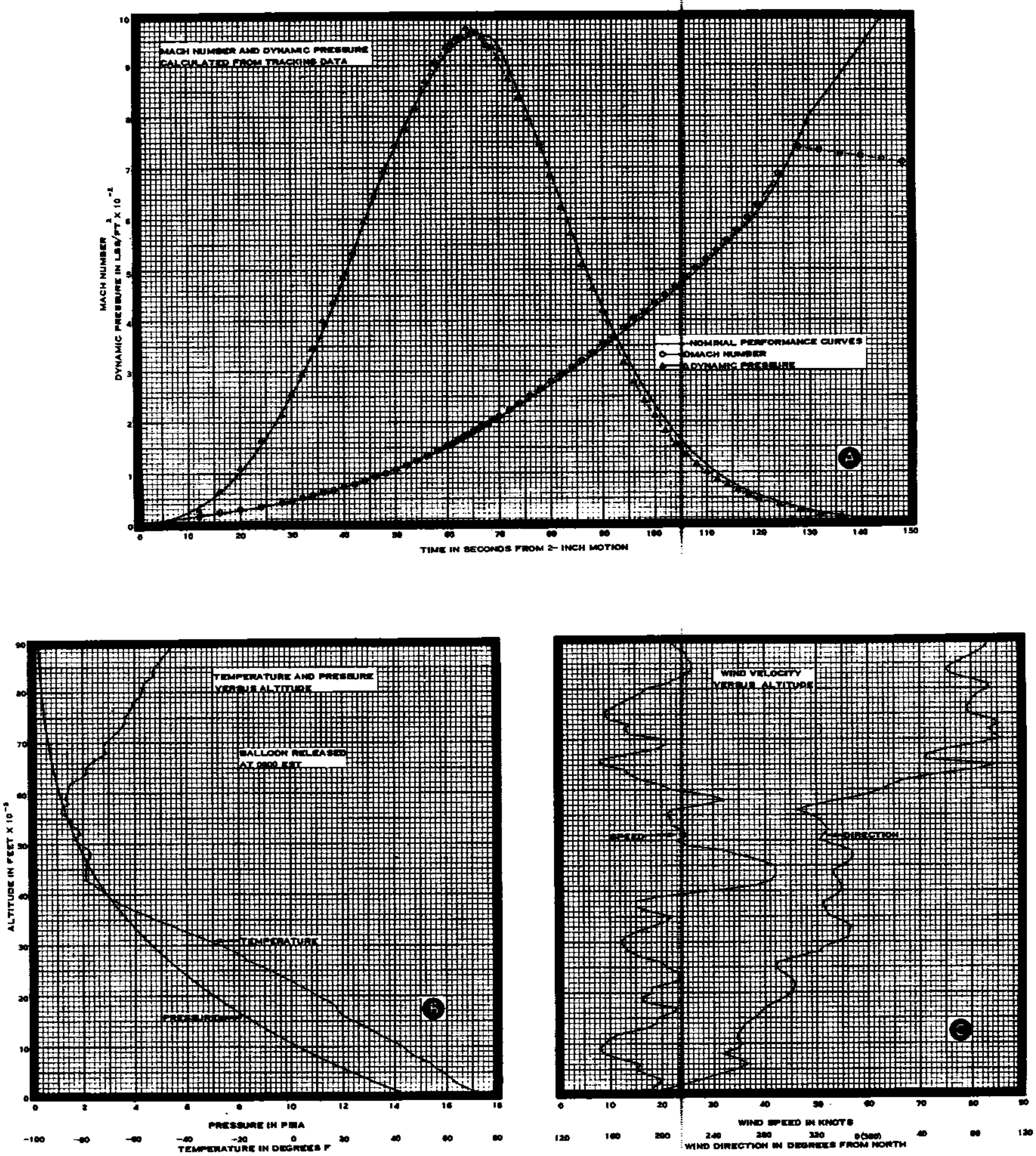

FIGURE 3-3.

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THE POLARITY CONVENTION FOR FLIGHT CONTROL SYSTEM TELEMETERED TRACES IS REFERENCED TO THE SKETCH BELOW. POSITIVE DEFLECTION DESIGNATES RATE AND DISPLACEMENT GYRO OUTPUTS AND ENGINE MOVEMENTS WHICH CAUSE THE VEHICLE TO ACCELERATE NOSE-UP IN PITCH, NOSE-RIGHT IN YAW, AND CLOCKWISE IN ROLL AS VIEWED FROM AFT. IN-PHASE SIGNALS CAUSE POSITIVE MOVEMENT FOR ALL ENGINES.

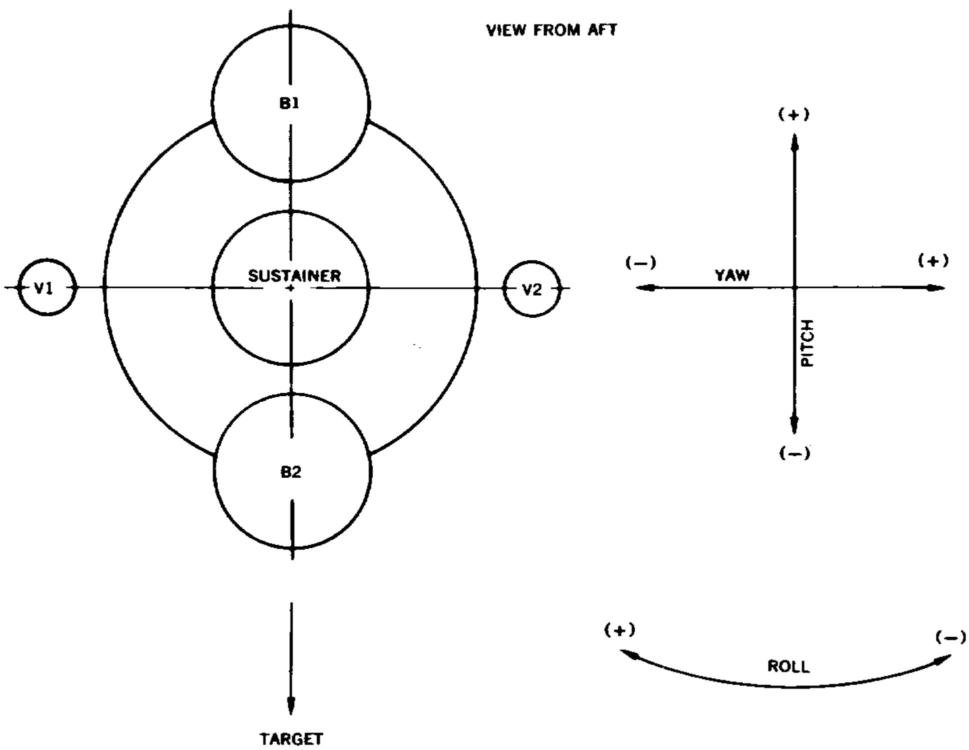

FIGURE 11-1. POLARITY CONVENTION

UNCLASSIFIED

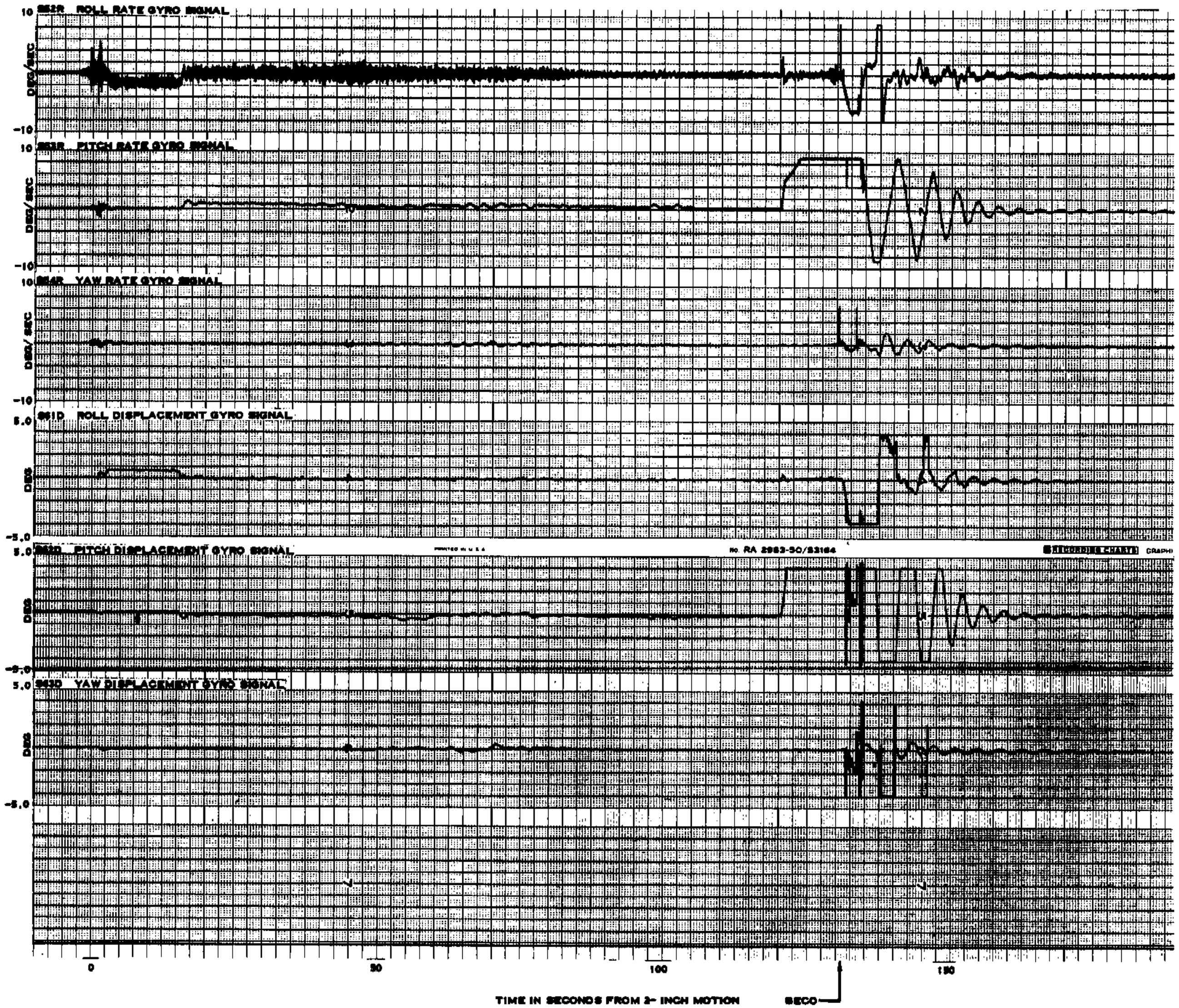

FIGURE 11-2A. TELEMETRY DATA

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UNCLASSIFIED

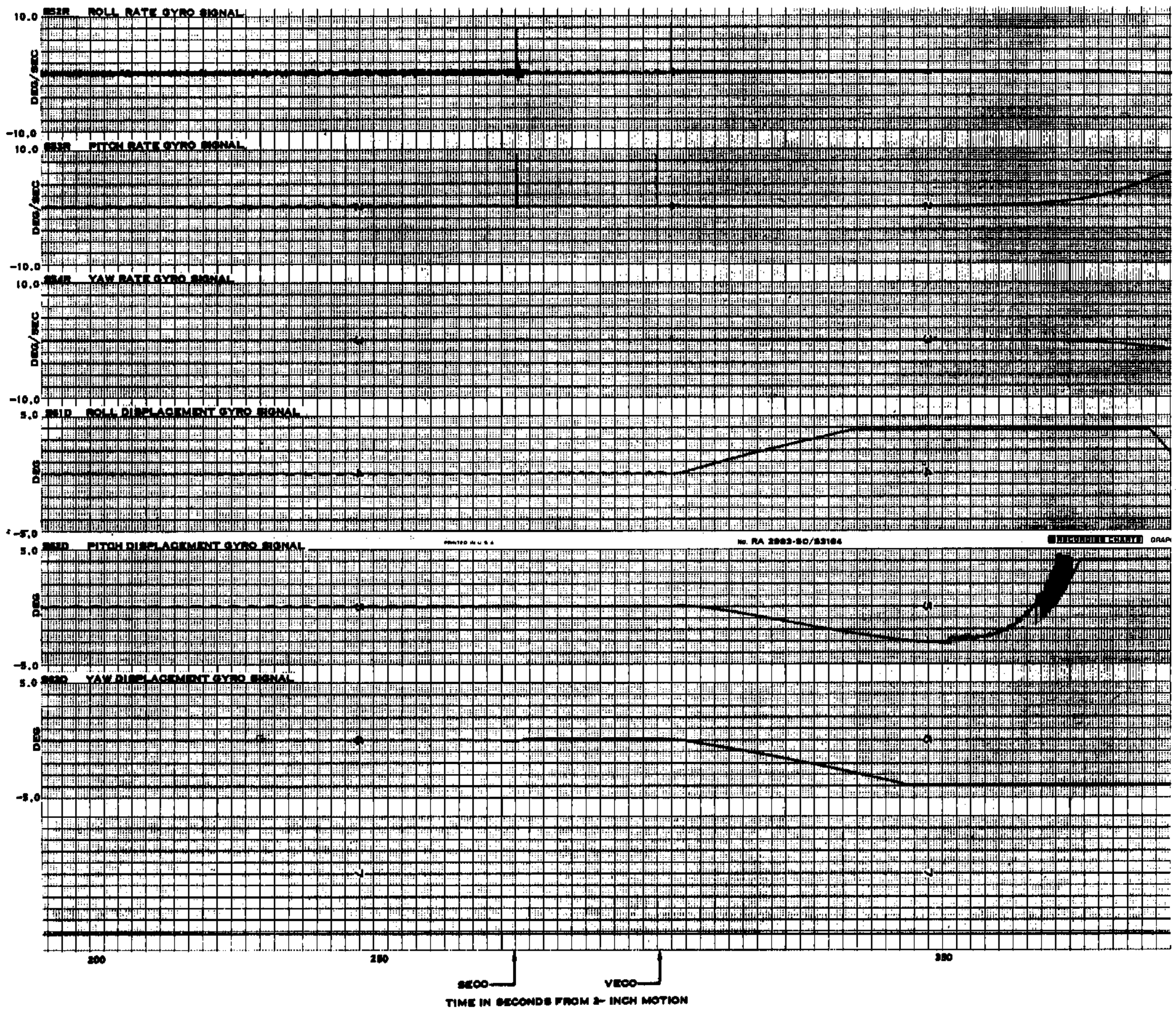

FIGURE 11-2B. TELEMETRY DATA

UNCLASSIFIED

UNCLASSIFIED

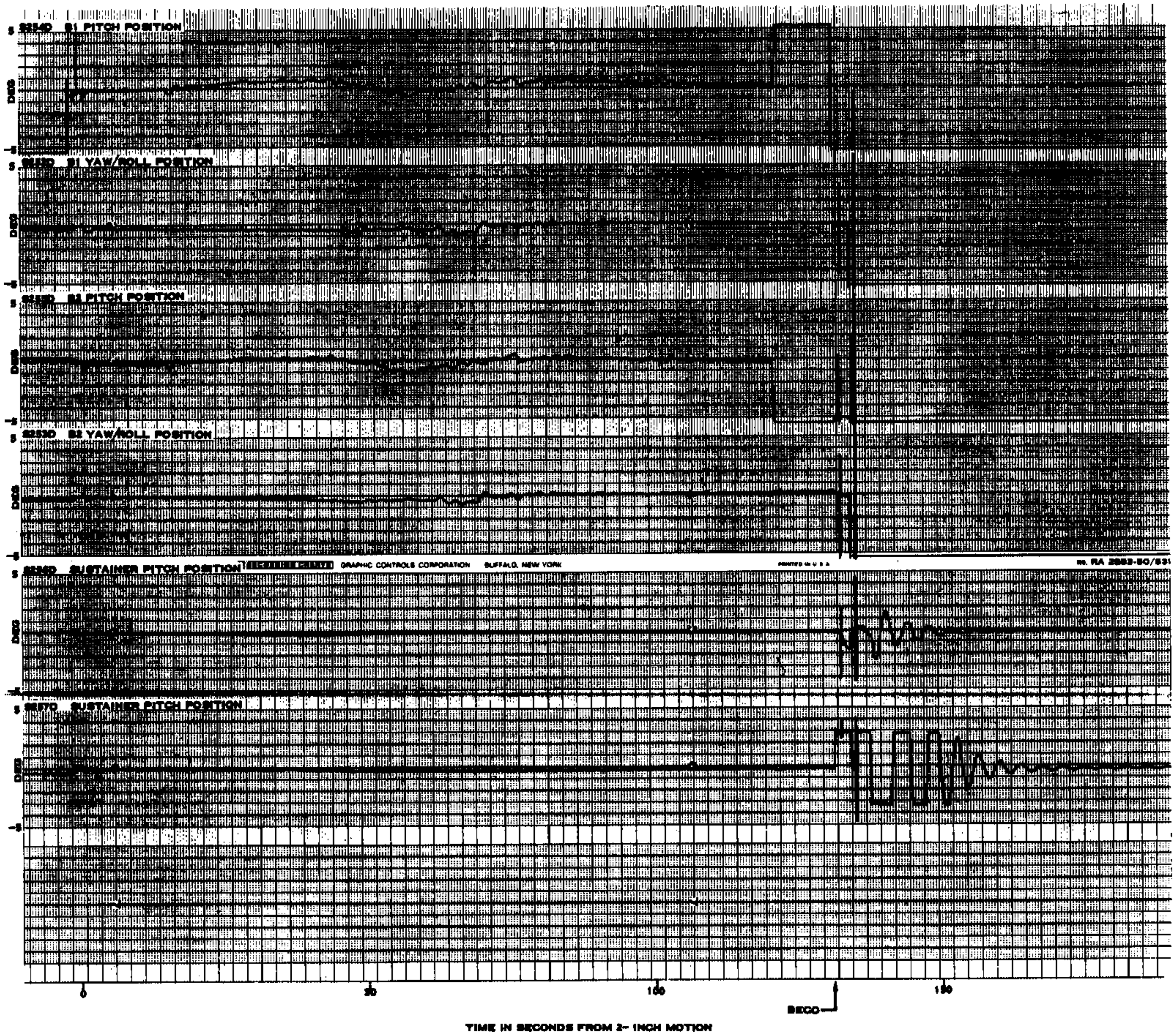

FIGURE 11-3A. TELEMETRY DATA

UNCLASSIFIED

UNCLASSIFIED

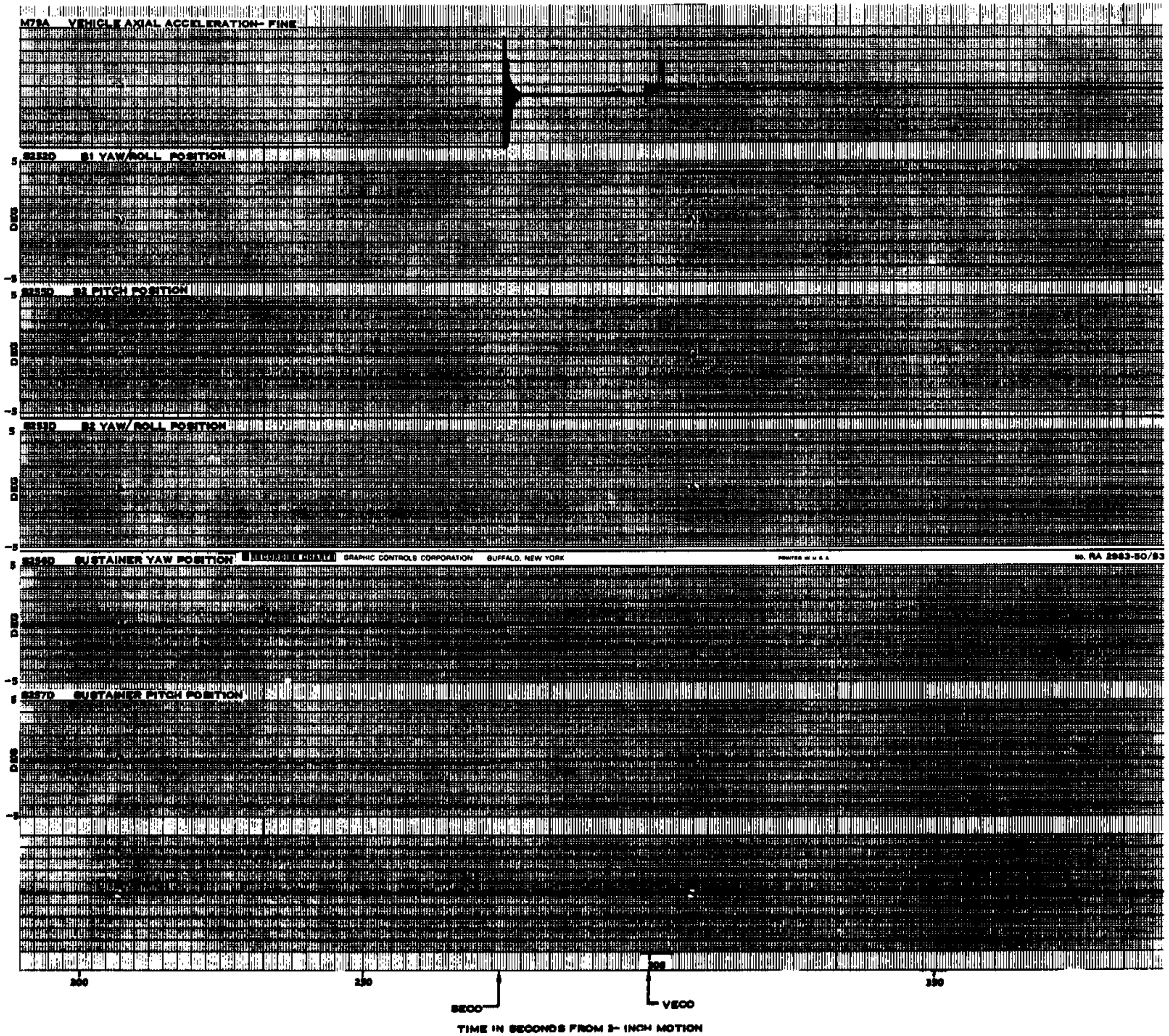

FIGURE 11-3B. TELEMETRY DATA

UNCLASSIFIED

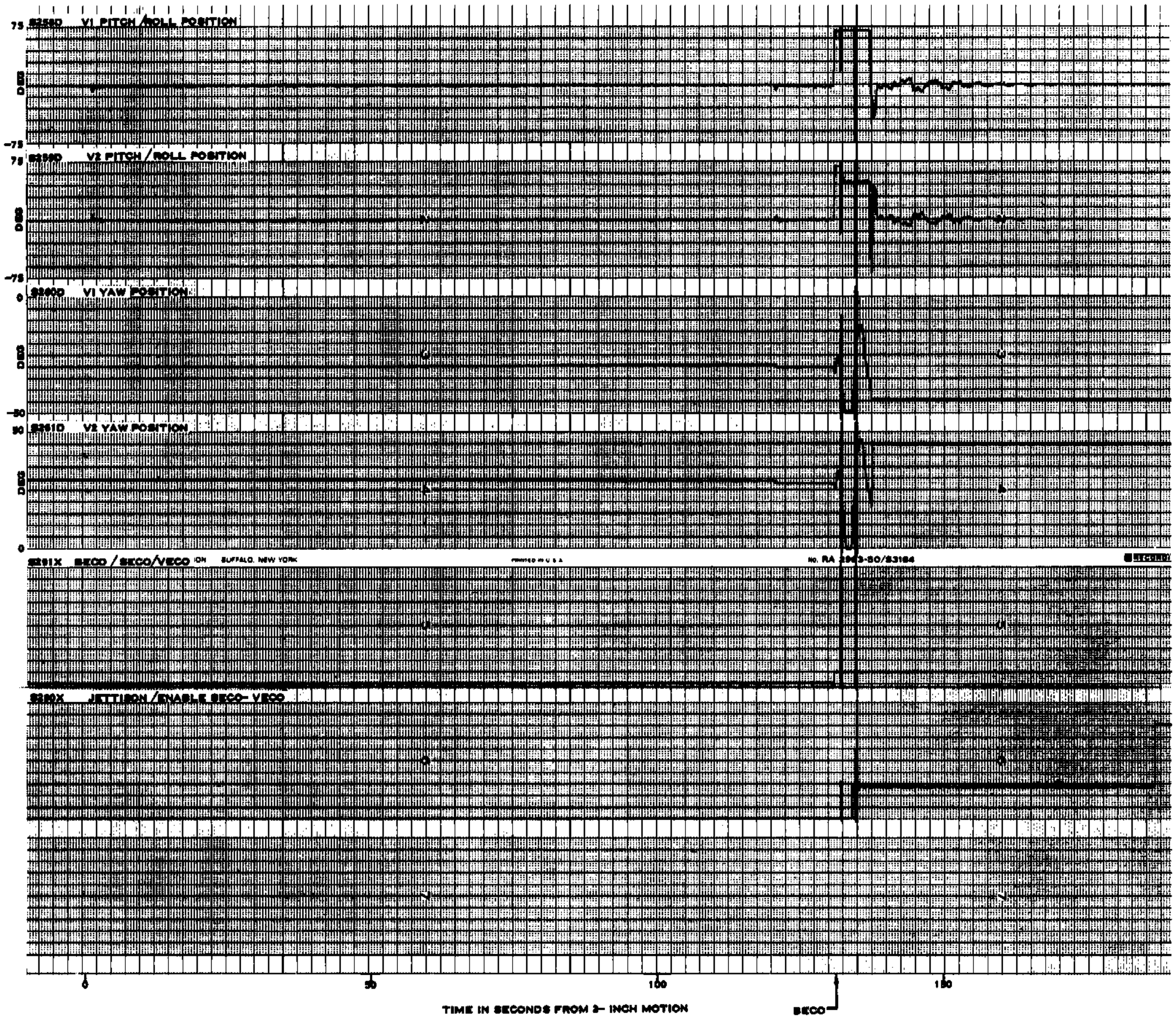

FIGURE 11-4A. TELEMETRY DATA

UNCLASSIFIED

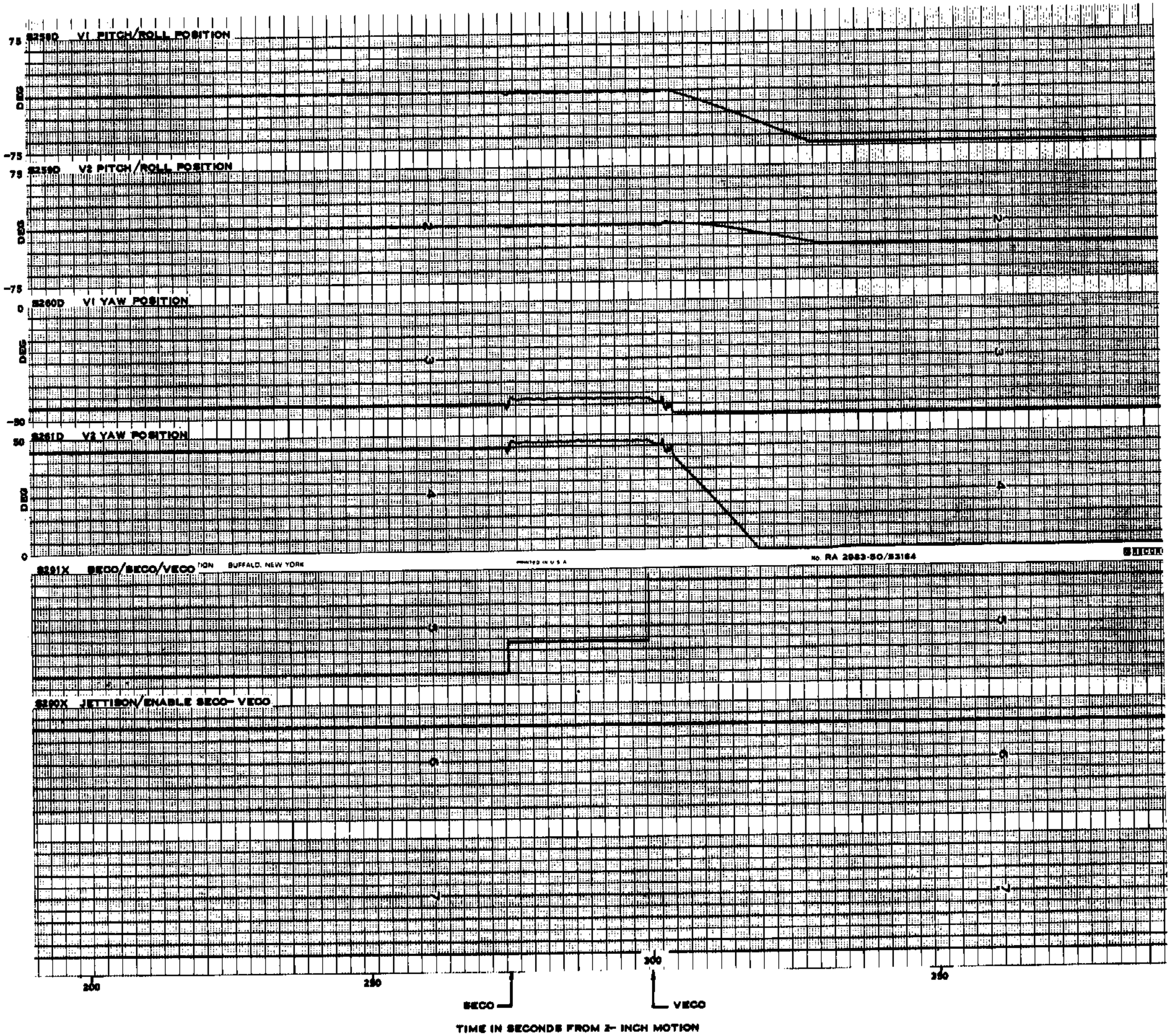

FIGURE 11-4B. TELEMETRY DATA

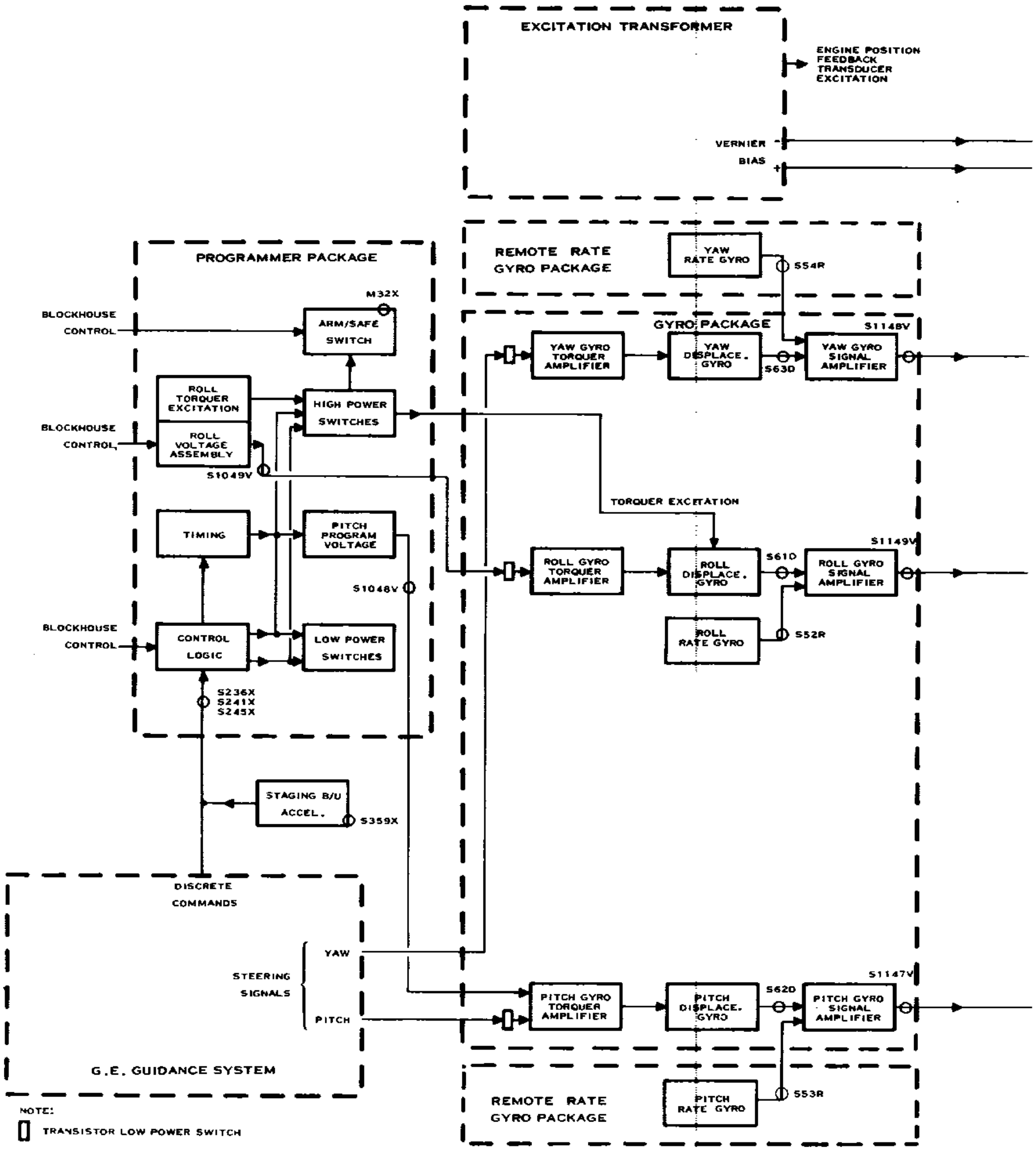

FIGURE 11-5A. FLIGHT CONTROL SYSTEM, SCHEMATIC

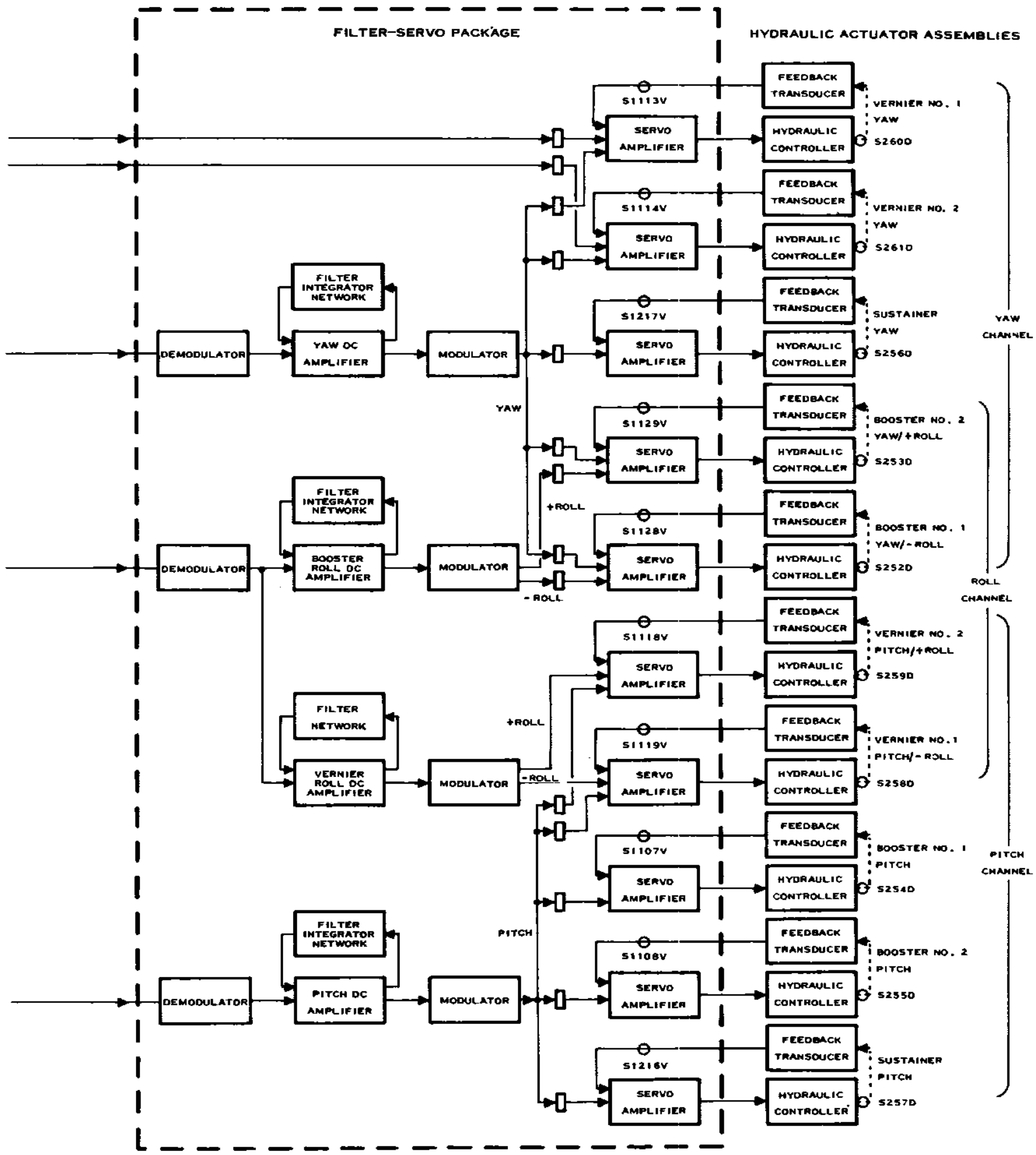

FIGURE 11-5B. FLIGHT CONTROL SYSTEM, SCHEMATIC

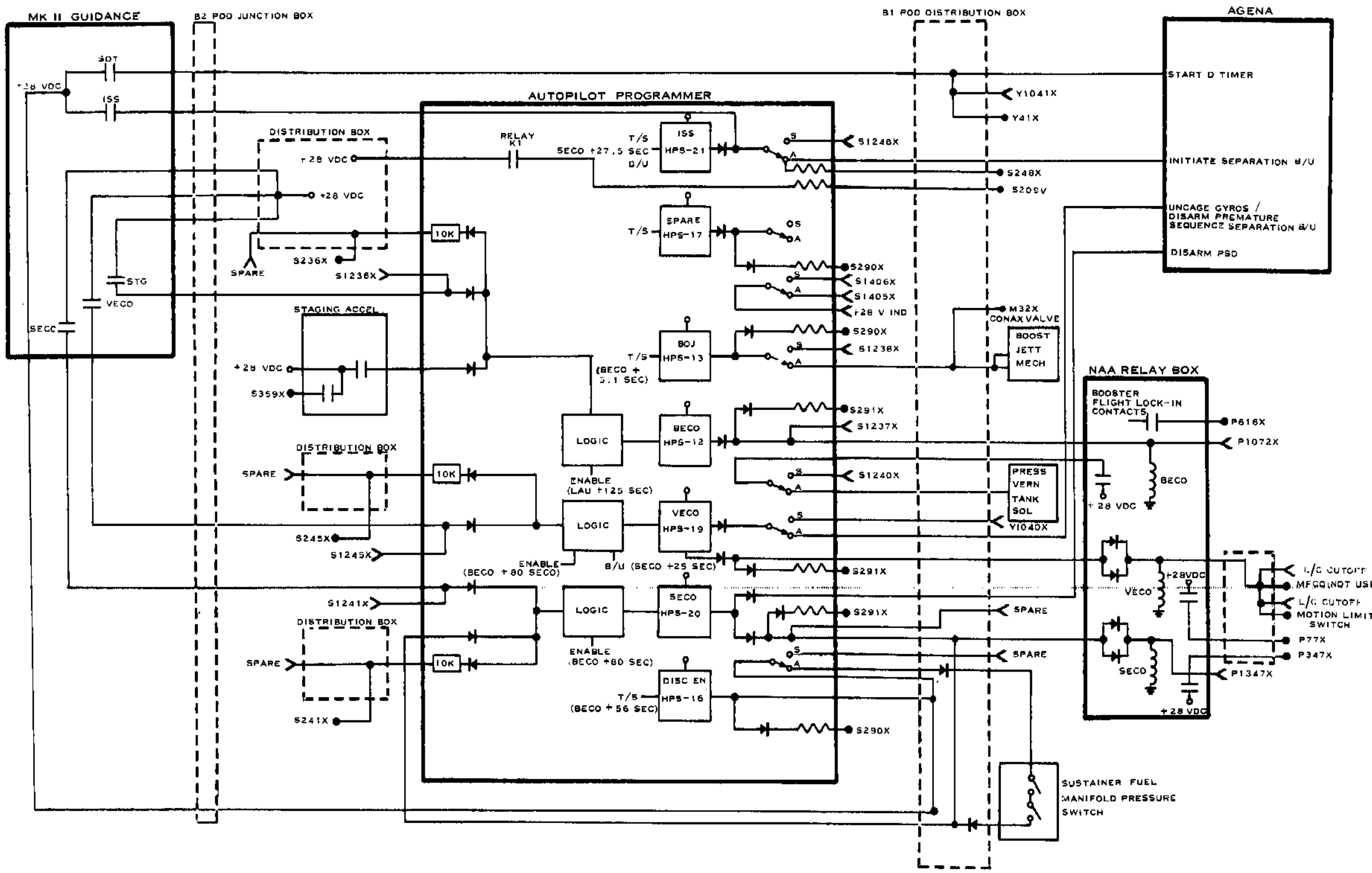

FIGURE 11-6. DISCRETE SCHEMATIC

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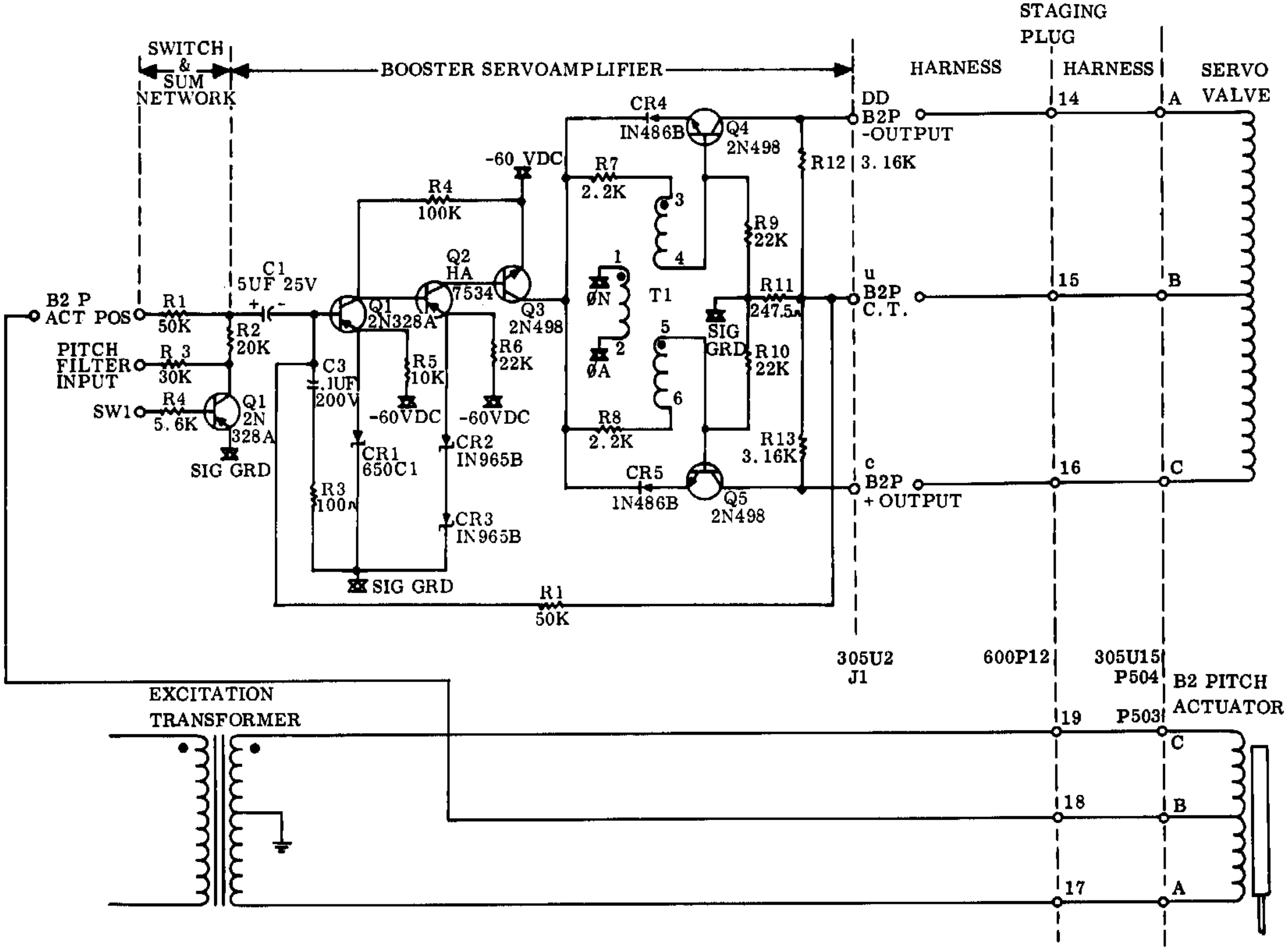

FIGURE 11-7. B2 PITCH CIRCUITRY

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