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GEMINI PROGRAM MISSION REPORT

GEMINI XII

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MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

JANUARY 1967

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MSC-G-R-67-1

GEMINI PROGRAM MISSION REPORT

GEMINI XII

Prepared by: Gemini Mission Evaluation Team

Approved by:

Charles W. Mathews
Charles W. Mathews
Manager, Gemini Program

George M. Low
George M. Low
Deputy Director

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
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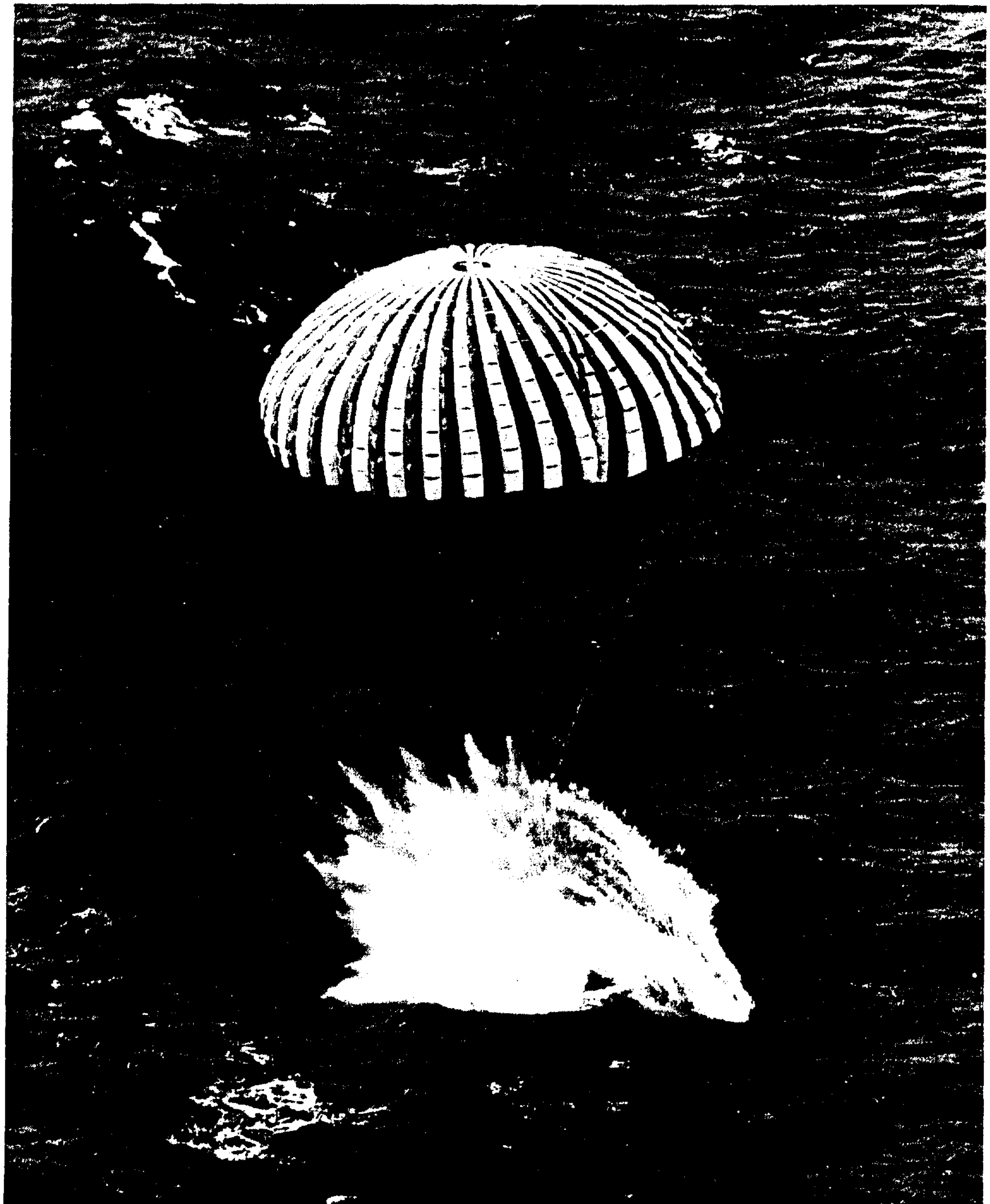

Gemini XII spacecraft at landing.

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1.0 MISSION SUMMARY

Gemini XII was the tenth manned mission and the sixth rendezvous mission of the Gemini Program. The Gemini Atlas-Agena Target Vehicle was launched from Complex 14, Cape Kennedy, Florida, at 2:07:59 p.m. e.s.t. on November 11, 1966. The Gemini Space Vehicle was launched from Complex 19, Cape Kennedy, Florida, at 3:46:33 p.m. e.s.t. on November 11, 1966, with Astronaut James A. Lovell as the Command Pilot and Astronaut Edwin E. Aldrin as the Pilot. The flight was successfully concluded on November 15, 1966, when the spacecraft was landed within three nautical miles of the prime recovery ship, the U.S.S. Wasp, at 94:34:30. (NOTE: All times in this section are spacecraft ground elapsed time (g.e.t.), referenced to lift-off of the Gemini Space Vehicle, unless otherwise specified.) The flight crew elected to be retrieved by helicopter and were on the deck of the prime recovery ship approximately 28 minutes after landing. The crew completed their flight in excellent physical condition and demonstrated full control of the spacecraft and competent management of all aspects of the mission.

The primary objectives of this mission were (1) to rendezvous and dock and (2) to evaluate extravehicular activities, and both were successfully achieved. The secondary objectives were to (1) conduct a tethered-vehicle evaluation, (2) conduct experiments, (3) rendezvous and dock during the third spacecraft revolution, (4) demonstrate automatic reentry, (5) conduct docked maneuvers for a high-apogee excursion, (6) conduct docking practice, (7) conduct system tests, and (8) park the Gemini Agena Target Vehicle. All the secondary objectives were achieved except two: (1) the high-apogee excursion was not attempted because of an anomaly noted during the primary propulsion system firing of the Gemini Agena Target Vehicle during insertion, and (2) the attempt to park the Gemini Agena Target Vehicle after the spacecraft landed was not performed because attitude control gas had been depleted.

The launch of the Gemini Atlas-Agena Target Vehicle was satisfactory and resulted in the Gemini Agena Target Vehicle achieving a nearly circular orbit with an apogee of 163.6 nautical miles and a perigee of 159.0 nautical miles.

Lift-off of the Gemini Space Vehicle occurred approximately 1 hour 38 minutes after the lift-off of the Gemini Atlas-Agena Target Vehicle. The powered flight of the Gemini Space Vehicle was satisfactory in all respects, and the spacecraft was separated from the launch vehicle approximately 23 seconds after second-stage engine cutoff. The Insertion Velocity Adjust Routine of the onboard computer was used to calculate the necessary velocity to be added to achieve the required orbit. The indicated velocity was applied by the Command Pilot, and the spacecraft was placed into an orbit from which a rendezvous during the third revolution could be achieved.

After insertion, nine maneuvers were performed by the crew to effect a third-orbit rendezvous with the Gemini Agena Target Vehicle. Prior to the terminal phase initiate maneuver, the onboard radar malfunctioned; however, the crew used onboard backup procedures, including optical tracking techniques and preprepared backup charts, to calculate the terminal phase maneuvers. The rendezvous was completed at 3 hours 46 minutes, and the Command Pilot docked the spacecraft with the Gemini Agena Target Vehicle at 4 hours 14 minutes.

At 5 hours 44 minutes the flight controller on the Coastal Sentry Quebec tracking ship reported that the fuel-cell oxygen-to-water differential pressure warning lights were on. A few minutes later, the lights went off but came on at approximately 7 hours 30 minutes. The lights continued to illuminate intermittently as the mission progressed, until at approximately 41 hours, they came on and remained on.

Because of the decision not to operate the Gemini Agena Target Vehicle primary propulsion system for the high-apogee excursion, photographing the solar eclipse was scheduled into the flight plan. At 7 hours 5 minutes, a docked maneuver of 43 ft/sec was performed using the secondary propulsion system of the Gemini Agena Target Vehicle to phase the orbit for the eclipse photography. After the first sleep period, a second phasing maneuver, also using the secondary propulsion system, was performed at 15 hours 16 minutes. This maneuver required a velocity change of 15 ft/sec. The crew photographed the solar eclipse but were not able to photograph the shadow of the moon on the earth.

The first of two periods of standup extravehicular activity began at 19 hours 29 minutes. During the 2 hours 29 minutes the pilot was outside the spacecraft, he installed the telescoping handrail between the spacecraft and the Gemini Agena Target Vehicle, performed photographic experiments, and retrieved the micrometeorite collection device which was located on the adapter just aft of the open hatch. The remainder of the second day was spent performing sequences of various experiments, and the second sleep period was started at 29 hours 30 minutes.

The crew was awakened at 36 hours 50 minutes to purge the fuel cells because of poor load sharing. The purge did not correct the problem, and at 37 hours 40 minutes, stack B of fuel cell section 2 failed completely and was removed from the line. During the next two hours, the crew performed several experiments, and at 39 hours 30 minutes they reported that little or no thrust was available from both a pitch-down thruster and a yaw-right thruster.

Preparations for umbilical extravehicular activity were begun at 39 hours 40 minutes, and the hatch was opened at 42 hours 48 minutes. The pilot translated to the Target Docking Adapter and attached a 100-foot tether from the Gemini Agena Target Vehicle to the spacecraft docking bar. He then moved to the area of the micrometeorite collection package mounted on the target vehicle, and, with the aid of two waist tethers and a pip-pin attachment system, he opened the package to expose the collection surfaces to the space environment. The pilot then moved to the spacecraft adapter, where he evaluated several restraint systems and performed various tasks. After completing these tasks, he returned to the target vehicle to evaluate additional restraint systems and aids, including two portable handholds, and to perform a second series of tasks. All tasks during the umbilical extravehicular activity were completed successfully, and the pilot returned to the cockpit and closed the hatch at 44 hours 55 minutes.

At 47 hours 23 minutes, the crew undocked the spacecraft from the Gemini Agena Target Vehicle and began the tether evaluation. The tether tended to remain slack and to tauten only occasionally; however, according to the crew, the two vehicles did slowly attain gravity-gradient stabilization. The tether evaluation continued until 51 hours 51 minutes, at which time the crew jettisoned the docking bar and released the tether. About 23 minutes later the crew performed a maneuver, using the spacecraft propulsion system, to separate the spacecraft from the target vehicle.

After the third sleep period, the crew performed a phase adjust maneuver at 61 hours 48 minutes and began to conduct several experiments. At 62 hours 42 minutes and again at 64 hours 17 minutes, a sodium-cloud rocket was launched from the French launch site in Algeria. Although the crew could not see either cloud, they took photographs of the planned areas.

Because of experiment activities, preparations for the second stand-up extravehicular activity became somewhat hurried and the crew requested a one-revolution delay in the start of the activity. At 66 hours 6 minutes, the hatch was opened for the extravehicular activity and several photographs were taken. The crew performed all planned experiment sequences and the hatch was closed at 67 hours 1 minute.

The crew reported further problems with the spacecraft attitude control thrusters at 68 hours; one yaw-left thruster was apparently inoperative, and the second yaw-left thruster was severely degraded. Prior to the fourth sleep period, the crew performed various sequences of several experiments. Fuel cell stack 1C failed during the sleep period, and the crew were awakened early to turn off a switch to stop the flow of reactants to this stack.

After the sleep period, the crew again performed experiments. A test of the propulsion system, conducted at 88 hours 57 minutes, indicated to the crew that two thrusters were delivering no measurable thrust and two others were degraded. At 89 hours the two remaining stacks—2A and 2C—in fuel cell section 2 were carrying less than one-half their normal share of the load. Because of this, two of the four main batteries had to be placed on the line at 91 hours 7 minutes to permit powering up the computer, and the other two batteries were placed on the line at 92 hours 42 minutes. All load was then removed from section 2 of the fuel cell system.

When the spacecraft Reentry Control System was activated during the last revolution, the regulated pressure in the A-ring was above normal and slowly rising, apparently as a result of a malfunction. To reduce the pressure and prevent the possibility of rupturing a safety burst disc, the crew used the A-ring for controlling spacecraft attitude. Prior to retrofire, the pressure was still slightly higher than normal but was well within safe limits.

Retrofire occurred at 93:59:58, and the crew performed all manual functions to prepare the spacecraft for reentry. At 400K feet, the Command Pilot controlled the spacecraft to the correct attitude and, after guidance initiate, switched to the automatic reentry mode. With the hand controller deactivated, he continued to follow all control commands so that manual control could have been restored in a minimum amount of time if a problem had arisen. The landing point was 2.6 nautical miles from the planned landing point and about three nautical miles from the prime recovery ship, the U.S.S. Wasp. After landing, the crew elected to be retrieved by helicopter and were aboard the ship just 28 minutes later. The spacecraft was hoisted onto the ship at 95 hours 41 minutes.

2.0 INTRODUCTION

A description of the Gemini XII mission and a discussion of the mission results are contained in this report. The report covers the time from the start of the simultaneous countdown of the Gemini Atlas-Agena Target Vehicle and the Gemini Space Vehicle to the date of publication of 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 when it is required for a logical presentation of the subject matter.

Data were reduced from telemetry, onboard records, and ground-based radar tracking but were reduced only in areas of importance. The evaluation of all vehicles consisted of analyzing the flight results and comparing them with expected or predicted results and with results of ground tests and previous missions.

Section 6.1, FLIGHT CONTROL, is based on observations and evaluations made in real time and may not coincide with the results of the postflight analyses; however, this section does present an excellent chronology of the mission, as seen in real time.

Brief descriptions and preliminary results of the experiments flown on this mission are presented in section 8.0.

The primary objectives of the Gemini XII mission were as follows:

(a) To rendezvous and dock

(b) To evaluate extravehicular activity.

The secondary objectives were as follows:

(a) To conduct a tethered-vehicle evaluation

(b) To perform experiments

(c) To rendezvous and dock during the third revolution

(d) To demonstrate automatic reentry

(e) To perform docked maneuvers (high-altitude excursion)

(f) To perform docking practice

(g) To conduct system tests

(h) To park the Gemini XII Agena Target Vehicle.

More detailed analyses of the performance of the launch vehicles and the guidance systems were continuing at the time of publication of this report. 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 12.

3.0 VEHICLE DESCRIPTION

The manned vehicle for the Gemini XII mission consisted of Spacecraft 12 and Gemini Launch Vehicle (GLV) 12. The Gemini Atlas-Agena Target Vehicle (GAATV) consisted of Gemini Agena Target Vehicle (GATV) 5001 and Target Launch Vehicle (TLV) 5307.

The general arrangement and major reference coordinates of the Gemini Space Vehicle are shown in figure 3.0-1. Section 3.1 of this report describes the spacecraft configuration; section 3.2 describes the GLV configuration; and section 3.3 provides the Gemini Space Vehicle weight and balance data. The general arrangement and major reference coordinates of the GAATV are shown in figure 3.0-2. Section 3.4 describes the GATV configuration, including the Target Docking Adapter; section 3.5 describes the TLV configuration; and section 3.6 provides the GAATV weight and balance data.

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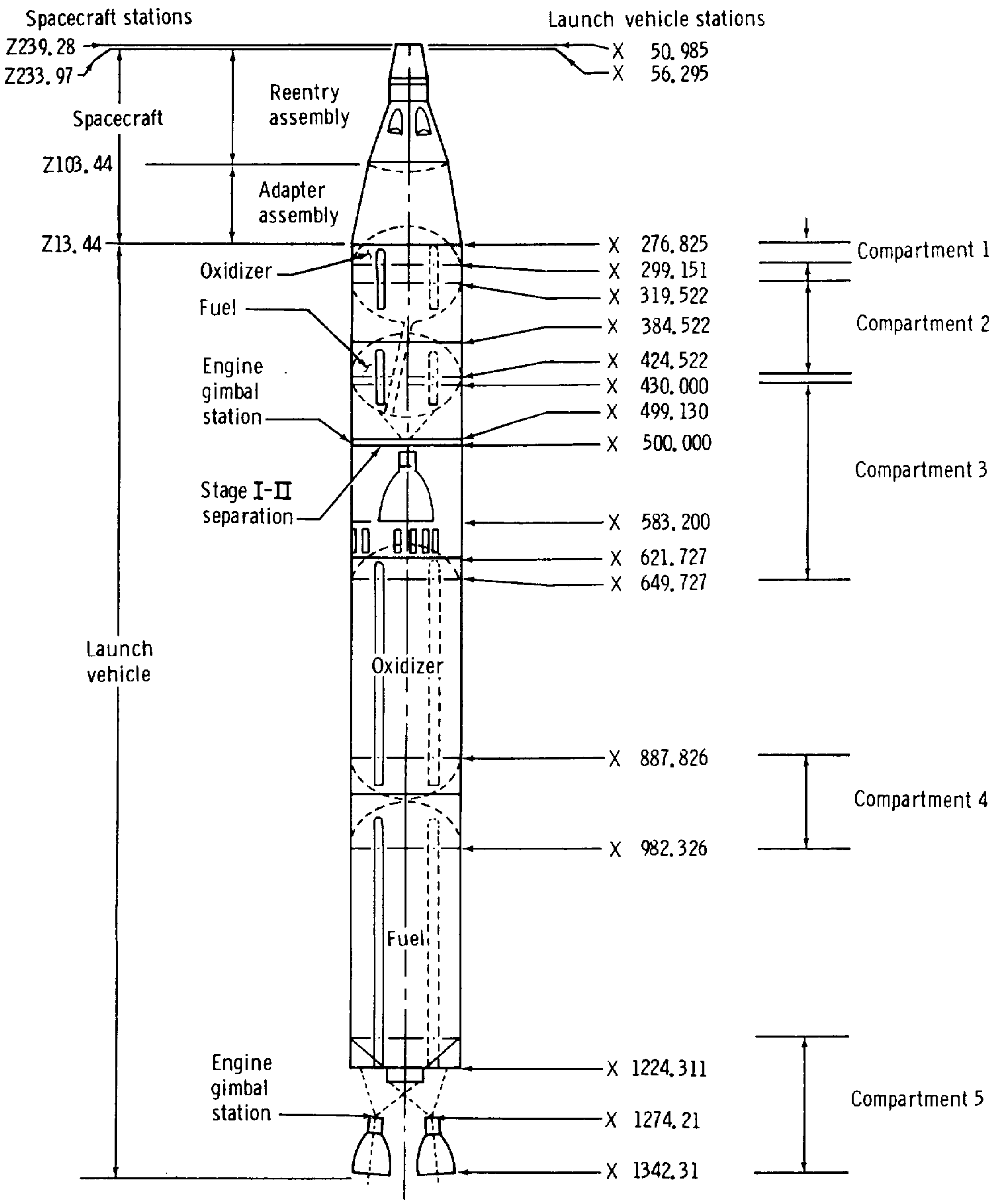

(a) Launch configuration.

Figure 3.0-1. - GLV/spacecraft relationships.

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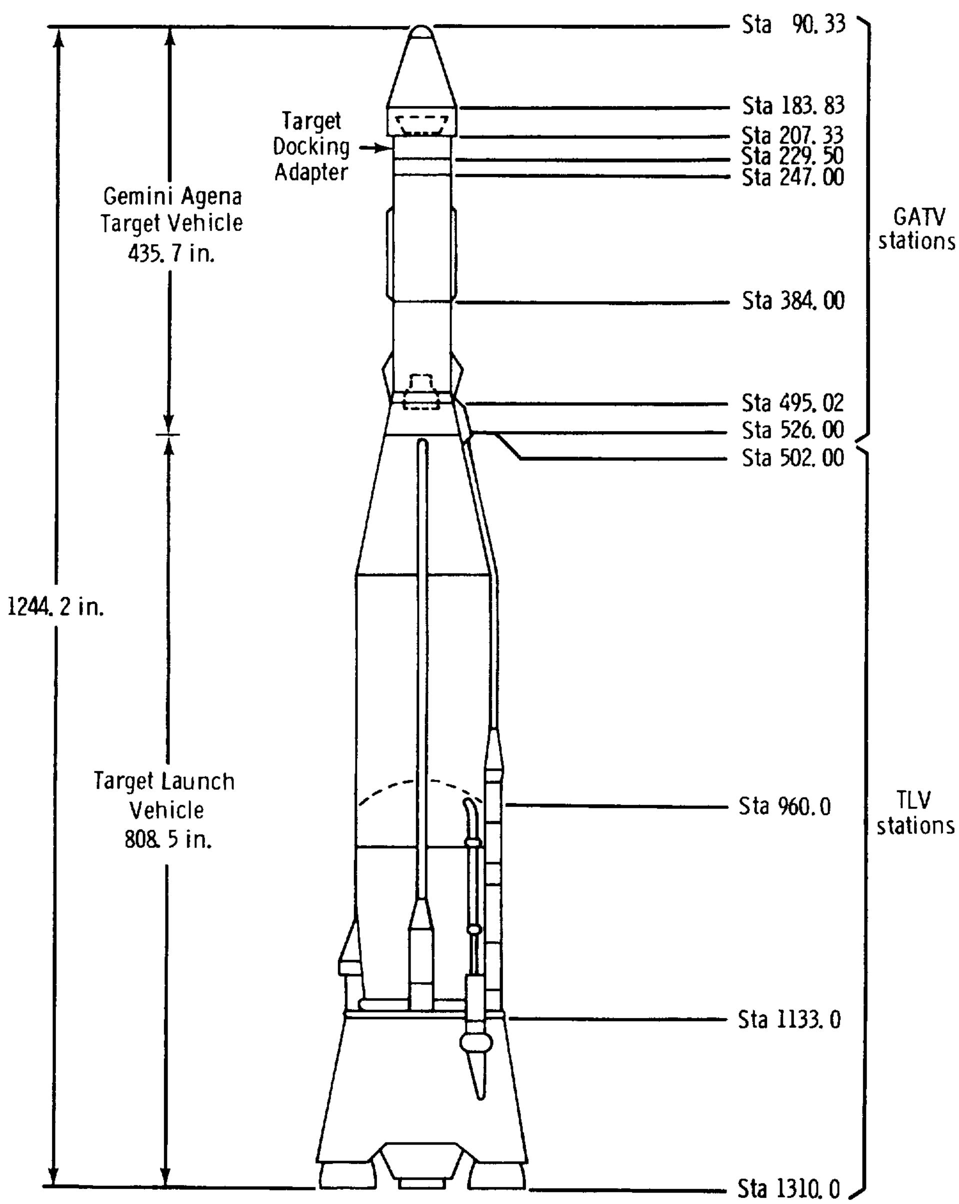
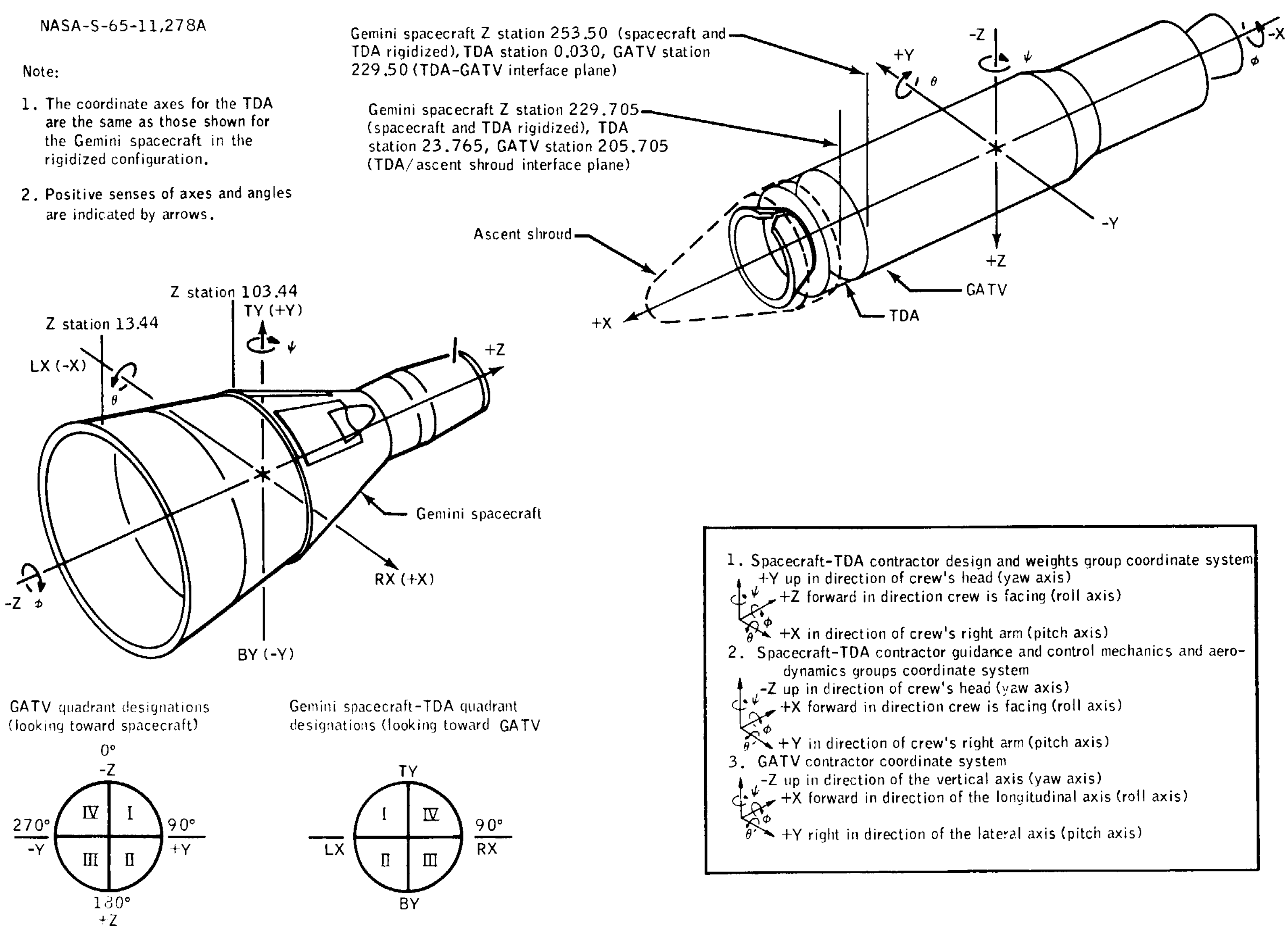


(a) Launch configuration.

Figure 3.0-2. - TLV/GATV relationship.



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

Figure 3.0-2. - Continued.

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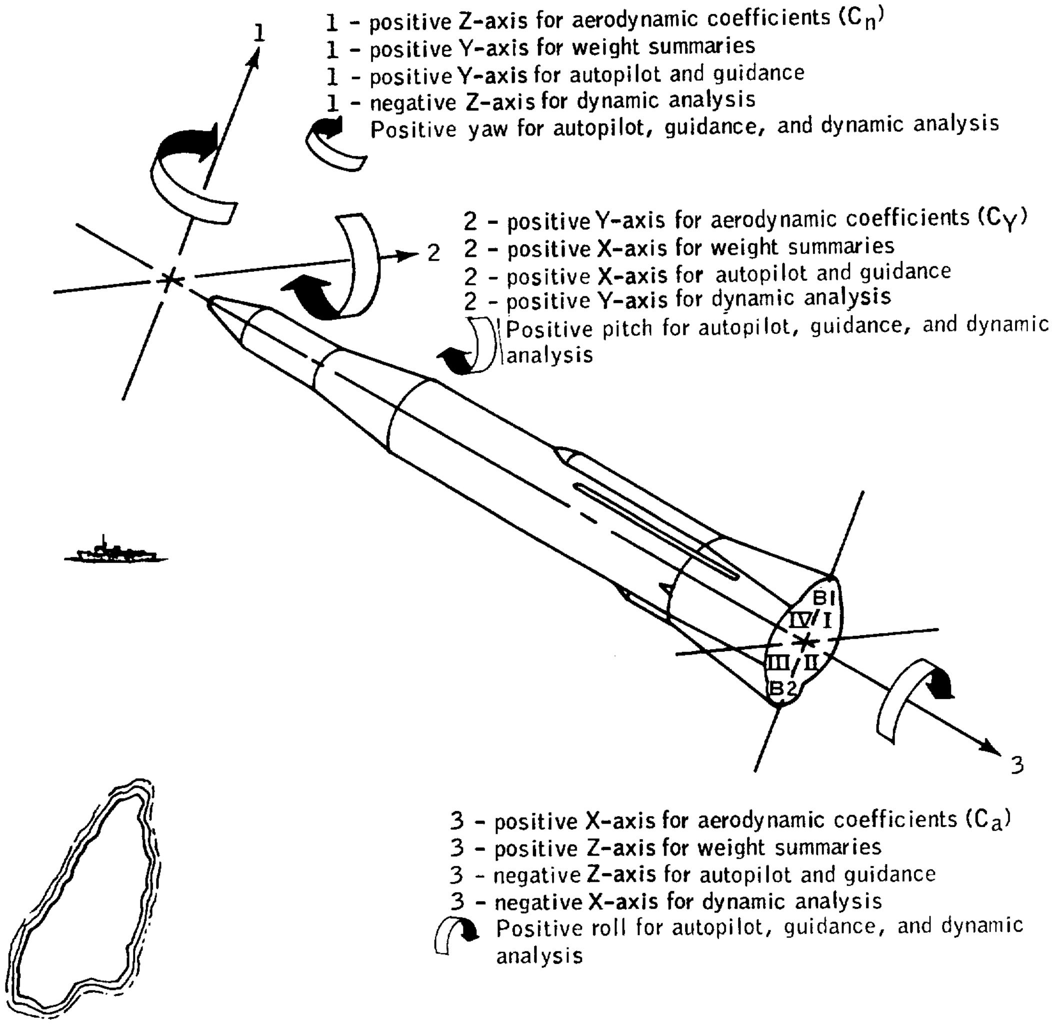

Vehicle shown in flight attitude

(c) Dimensional axes and guidance coordinates, TLV.

Figure 3.0-2. - Concluded.

3.1 GEMINI SPACECRAFT

The structure and major systems of Spacecraft 12 (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 12 describe the modifications incorporated into the subsequent spacecraft. Except for the extravehicular equipment, Spacecraft 12 closely resembled Spacecraft 11 (ref. 12), and only the significant differences (table 3.1-I) between those two spacecraft are included in this report. A detailed description of Spacecraft 12 is contained in reference 13.

3.1.1 Spacecraft Structure

The primary load-bearing structure of Spacecraft 12 was essentially the same as that of Spacecraft 11. The few changes were as follows:

(a) A cap was added to the top of the docking bar to aid in "fly in" attachment (without extravehicular activity) to the GATV tether, and to prevent the tether from slipping off the index bar during the tethered exercise (fig. 3.1-2).

(b) An extravehicular activity (EVA) work station was mounted in place of the Apollo sump tank evaluation equipment and the Hand Held Maneuvering Unit (HHMU) propellant tank in the adapter equipment section.

(c) The Orbital Attitude and Maneuver System (OAMS) reserve oxidizer tank, F-package, and associated lines, wiring, and structural supports were removed from the equipment section of the adapter assembly. This change, which reduced the spacecraft weight by 35.83 pounds, was made to increase the margin between the spacecraft weight and the launch vehicle payload capability.

(d) The retro lines were removed from both the inner and outer surfaces of the pilot's window. These lines were in the field of view when the pilot was using the sextant or taking photographs through the window.

(e) The EVA camera mounting bracket in the adapter section was relocated to the left handhold. This permitted the pilot to have his feet in the fixed foot restraints while he was installing the camera.

3.1.2 Major Systems

No significant changes were made to the following major systems:

(a) Communications

(b) Environmental Control

(c) Guidance and Control

(d) Time Reference

(e) Electrical

(f) Landing

(g) Postlanding and Recovery

3.1.2.1 Instrumentation and Recording System.- The instrumentation system was modified to include telemetry parameters for the reactant supply system oxygen-to-hydrogen pressure differentials (BC05 and BC06), and fuel-cell sections oxygen-purge-valve actuation (BD04 and BE04). The telemetry parameters for the OAMS F-package were deleted.

3.1.2.2 Propulsion System.- The reserve oxidizer tank, component F-package, and associated tubing and electrical components were removed from the OAMS (fig. 3.1-3).

3.1.2.3 Pyrotechnic System.- The OAMS F-package and footrest deploy cartridges were removed from the spacecraft. Other pyrotechnic devices were added as required to support the experiments.

3.1.2.4 Crew-station furnishings and equipment.- In addition to the changes required by the different experiments (section 8.0), the following modifications were incorporated into the crew-station furnishings and equipment.

3.1.2.4.1 Controls and displays: The crew-station controls and displays (fig. 3.1-4) were modified as follows:

(a) The RES/O position nomenclature on the OAMS/RCS pressure/temperature select switch was changed to "O". Also, other nomenclature associated with the OAMS reserve oxidizer were deleted from the applicable controls and displays.

(b) The switch position used for the Apollo sump tank camera activation on Spacecraft 11 was used to control the EVA camera on Spacecraft 12.

3.1.2.4.2 Stowage provisions: The stowage provisions were essentially unchanged. The individual stowage containers are shown in figure 3.1-5, and table 3.1-II lists the major items of equipment stowed in the containers at launch.

3.1.2.4.3 Crew furnishings: An additional sun filter was provided for installation on the left-hand hatch window to provide visual protection for the command pilot during the period when he was aligning the spacecraft on the sun to take photographs of the solar eclipse.

3.1.2.5 Extravehicular equipment.- No significant changes were made to the Extravehicular Life Support System (ELSS), and the 25-foot umbilical was similar to the one provided for the Gemini VIII mission. An EVA work station was mounted in the adapter equipment section, and an additional EVA work station was mounted on the Target Docking Adapter (see section 3.4).

3.1.2.5.1 Structural modifications: The EVA work station (fig. 3.1-6) contained the following equipment for one-hand tasks:

(a) A push-pull type fluid quick disconnect attached to a high-pressure type hose

(b) An electrical wire bundle with three different types of connectors

(c) Six Velcro hook and pile strips (two each 3-inch, 4-inch, and 5-inch strips, three of nylon Velcro and three of steel Velcro)

(d) Portable handholds, with Velcro on the handhold feet for attachment during EVA

(e) Quick-release tether attach pins to hold the portable handholds during launch

(f) Waist-tether attach rings.

The following equipment was installed on the EVA work station for two-handed tasks:

(a) Tether hooks for attachment to D-rings of two sizes were attached by lanyards to the work station.

(b) A debris cutter was stowed in a pouch on the work station.

(c) A raised section on the work station contained a wrench boss and two bolts—the head of one bolt was 0.5-inch high. A torque wrench was stowed in a pouch on the work station.

(d) Two penlights were stowed in individual pouches mounted on the tool pouch.

A telescoping handrail (fig. 3.1-7), for attachment between the spacecraft and the TDA, was stowed on the inside of the right-hand hatch. The handrail was designed to be extended manually and the small end inserted into a funnel hole in the TDA docking core. A fitting mounted on the other end of the handrail mated to a modified shingle bolt between the hatches.

3.1.2.5.2 Space suits: The space suit configuration for the command pilot (G-4C suit with a lightweight coverlayer) was basically the same as the used by the Gemini XI command pilot.

The G-4C space suit for the pilot was the same basic design as that of the command pilot and was fitted with an extravehicular coverlayer and a single-lens sun visor assembly. The space suit coverlayer was a modified version of that used for the Gemini IX-A mission. The heat-protective cloth was replaced with nylon, and four layers of the super-insulation were removed. The coverlayer thermal layup was quilted to the first layer of the micrometeoroid material, using a rectangular pattern over the torso.

TABLE 3.1-I.- SPACECRAFT 12 MODIFICATIONS

System	Significant differences between Spacecraft 12 and Spacecraft 11
Structure	(a) A cap was added to the top of the docking bar. (b) The Apollo sump tank equipment and the propellant tanks for the HHMU were replaced by an EVA work station. (c) The retro lines were removed from the pilot's window. (d) The EVA camera mounting bracket in the adapter section was relocated to the left handhold.
Instrumentation and Recording	Four parameters were added to the fuel cell sections and all OAMS F-package parameters were deleted.
Propulsion	The OAMS reserve oxidizer system was removed.
Pyrotechnic	The OAMS F-package and footrest deploy cartridges were removed.
Crew Station Furnishings and Equipment	(a) The nomenclature was removed from all controls and displays pertaining to the OAMS reserve oxidizer. (b) An EVA camera switch replaced the Apollo sump tank camera switch used on Spacecraft 11. (c) An additional sun filter was provided for the left-hand hatch window.
EVA equipment	(a) The umbilical cable was a 25-foot single umbilical hose instead of a 30-foot dual umbilical. (b) An EVA work station was installed in the adapter equipment section. (c) The HHMU was deleted.

TABLE 3.1-I.- SPACECRAFT 12 MODIFICATIONS - Concluded

System	Significant differences between Spacecraft 12 and Spacecraft 11
EVA equipment - Concluded	(d) A telescoping handrail was stored inside the right-hand hatch for use during EVA for controlled motion to the spacecraft. (e) The pilot's extravehicular coverlayer was modified by quilting the thermal layup to the first layer of the micrometeoroid material.

TABLE 3.1-II.- CREW-STATION STOWAGE LIST

Stowage area (see fig. 3.1-4)	Item	Quantity
Centerline stowage container	18-mm lens, 16-mm camera	1
	16-mm sequence camera with magazine	2
	5-mm lens, 16-mm camera	1
	70-mm camera, superwide angle	1
	70-mm film magazine	1
Left sidewall containers	Personal hygiene towel	1
	Waste container	1
	Defecation device	1
	Voice tape cartridge	5
	Food, one-man meal	2
	Velcro pile, 2 by 6 in.	1
	Velcro hook, 2 by 6 in.	1
	Penlight	1
	Visor anti-fog pads	5
Left aft stowage container	ELSS Umbilical	1
	Hose nozzle interconnectors	2
	Visor anti-fog pads	5
	EVA waist tethers	2
	Food, one-man meal	2
	Dual connectors	2
	ELSS restraints	2
	EVA gloves	1 pr.
	Cable, EVA remote control, 16-mm camera	1

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

Stowage area (see fig. 3.1-4)	Item	Quantity
Left aft stowage container - concluded	ELSS hose, short	1
	ELSS hose, long	1
Left footwell	Food, one-man meal	7
Right sidewall containers	Spot meter and exposure dial	1
	Inflight medical kit	1
	Personal hygiene towel	1
	Waste container	1
	Defecation device	1
	Voice tape cartridge	5
	Velcro pile, 2 by 6 in.	1
	Velcro hook, 2 by 6 in.	1
	Penlight	1
	Glass contamination strips	3
Right aft stowage container	16-mm sequence camera with magazine	1
	16-mm film magazine	10
	Postlanding kit	1
	70-mm general-purpose camera	1
	Inflator, manual, blood pressure	1
	Waste containers	2
	Defecation device	6
	70-mm film magazine	2
	Ultraviolet filter	1
	5-mm lens, 16-mm camera	1

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

Stowage area (see fig. 3.1-4)	Item	Quantity
Right footwell	Celestial display - polar	1
	Food, one-man meal	6
Orbital utility pouch	Standup tether	1
Right and left circuit-breaker fairings	Clamps, urine system	2
	Pouch, roll-on cuff receiver	2
	Latex, roll-on cuffs	8
	Urine receiver, removable cuff	1
	Tape, 1/2 in. by 10 ft	2
	Glareshield, optical sight	1
Center stowage	Mirror mounting bracket	2
	18-mm lens, 16-mm camera	1
	75-mm lens, 16-mm camera	1
	Magazine, film, 16-mm	8
	Ringe viewfinder	1
	70-mm general-purpose camera	1
	f/2.8 lens, general-purpose	1
	70-mm film magazine	1
	ELSS chestpack	1
Water management console	Roll-on cuff receiver, urine system	1
Left and right dry-stowage bags	Window shade reflective	2
	Orbital path display	1
	Celestial display - Mercator	1
	Shade, auxiliary, window	2

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

Stowage area (see fig. 3.1-4)	Item	Quantity
Right hatch	Telescoping handrail	1
	Food, one-man meal	7

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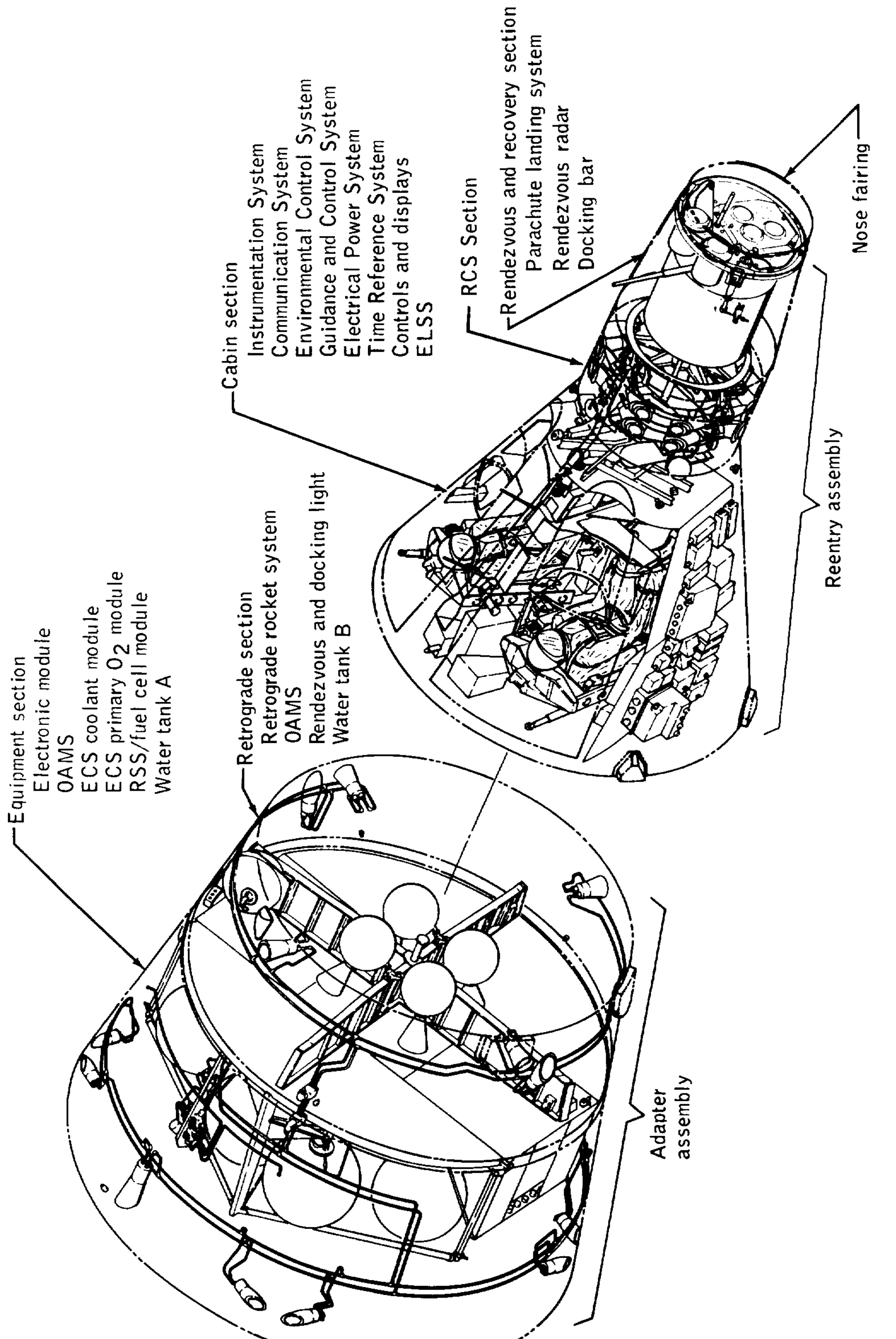

Figure 3.1-1. - Spacecraft arrangement and nomenclature.

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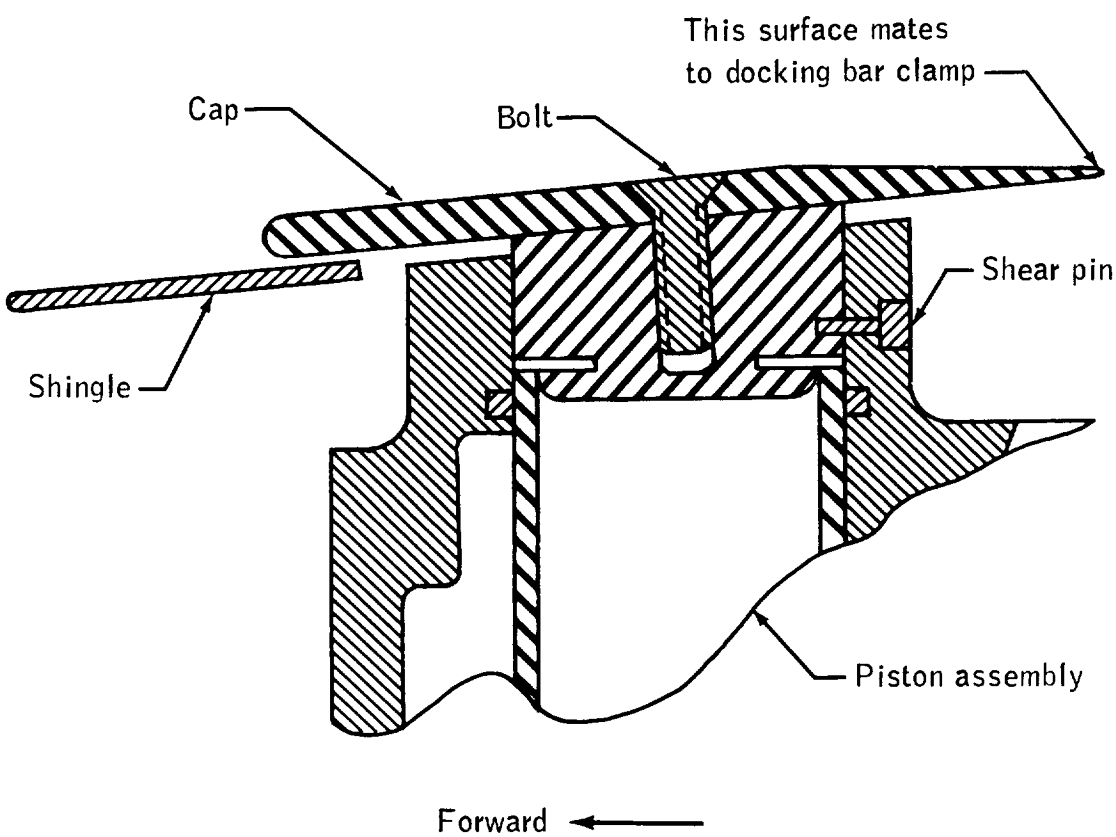

(Docking bar in retracted position)

Figure 3.1-2. - Docking bar cap.

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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

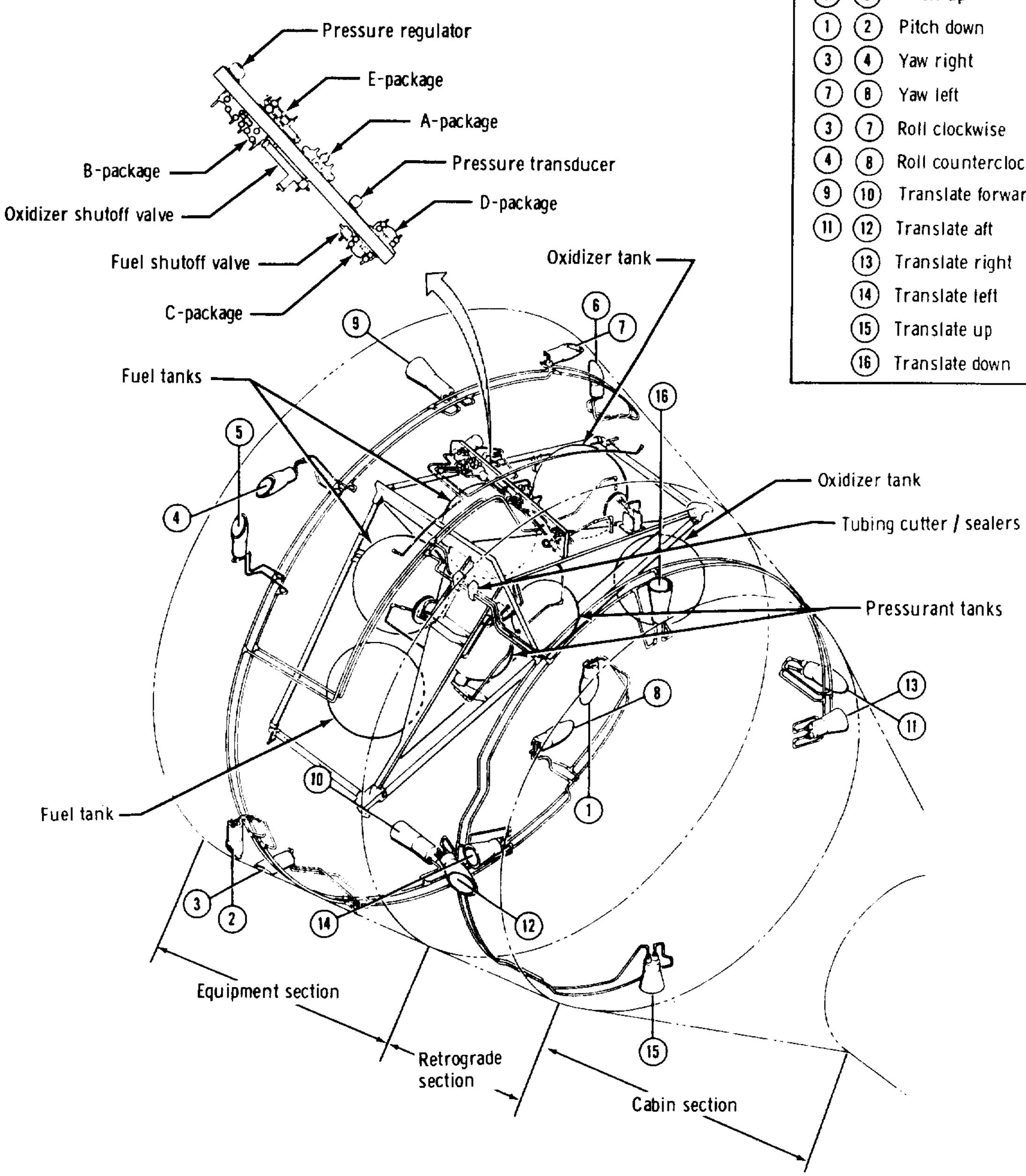

Figure 3.1-3. - Orbital Attitude and Maneuver System.

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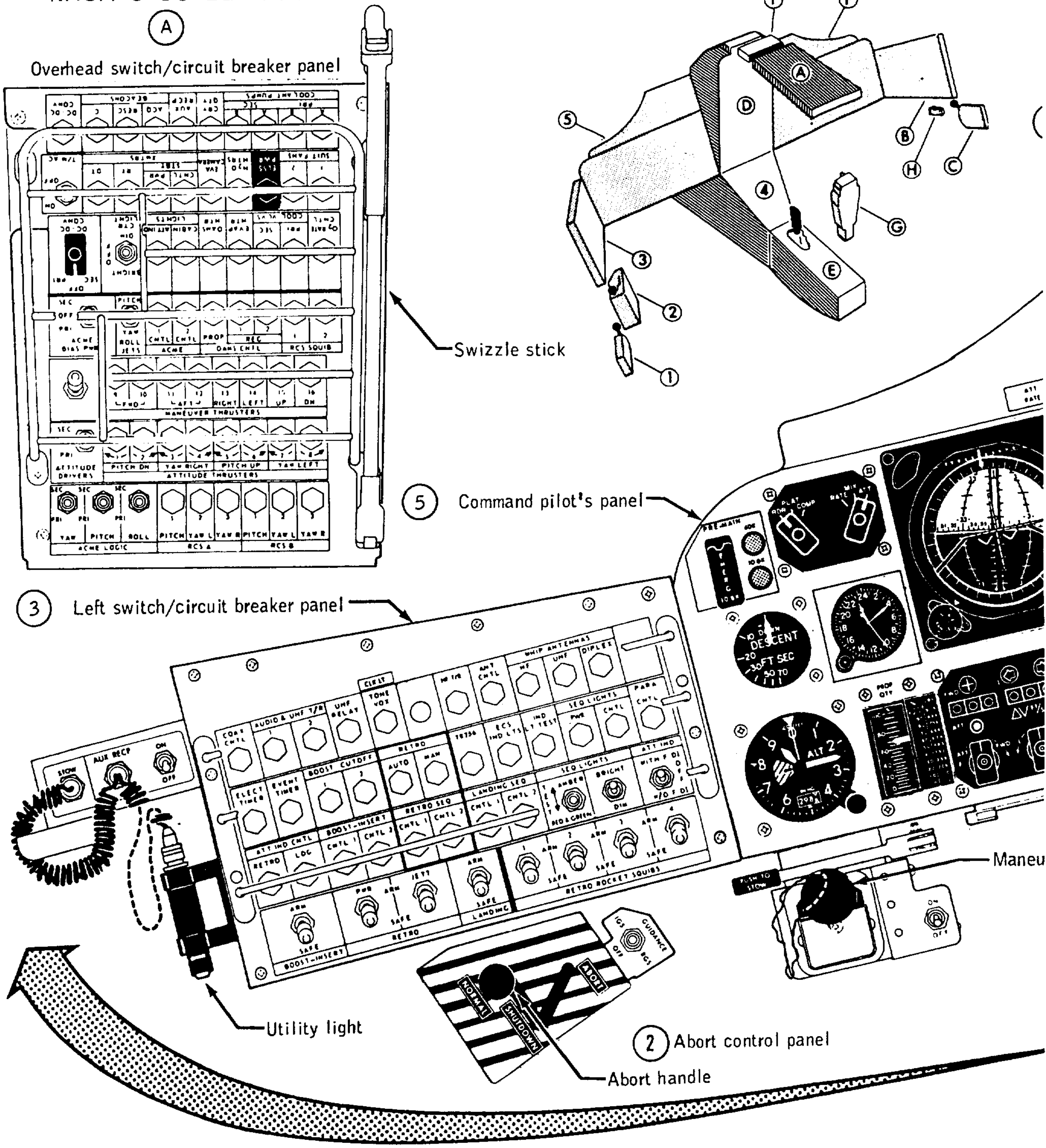

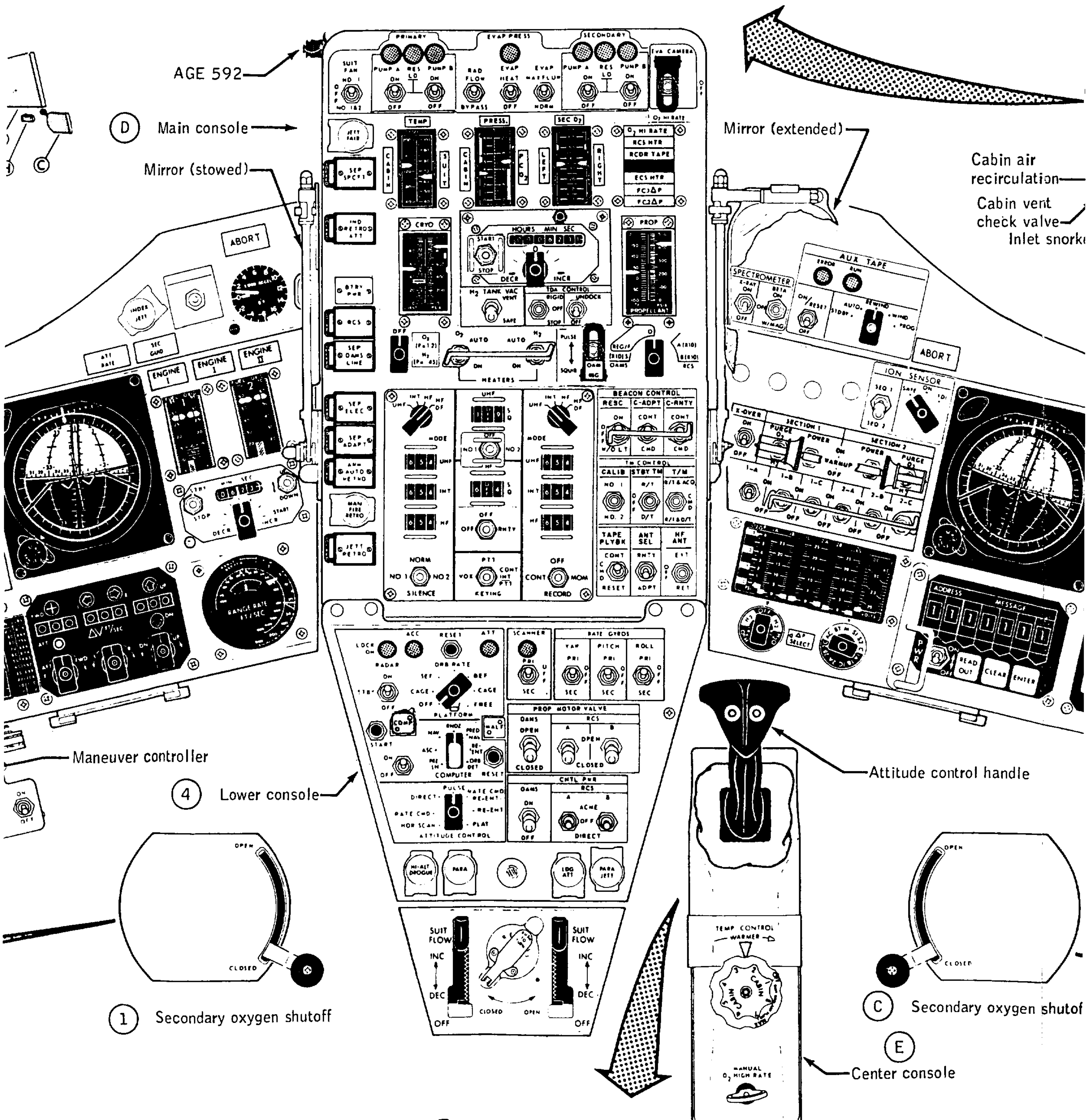
AGE 592
D Main console
Mirror (stowed)
Mirror (extended)
Cabin air recirculation
Cabin vent check valve
Inlet snork
Maneuver controller
4 Lower console
Attitude control handle
1 Secondary oxygen shutoff
C Secondary oxygen shutof
E
Center console

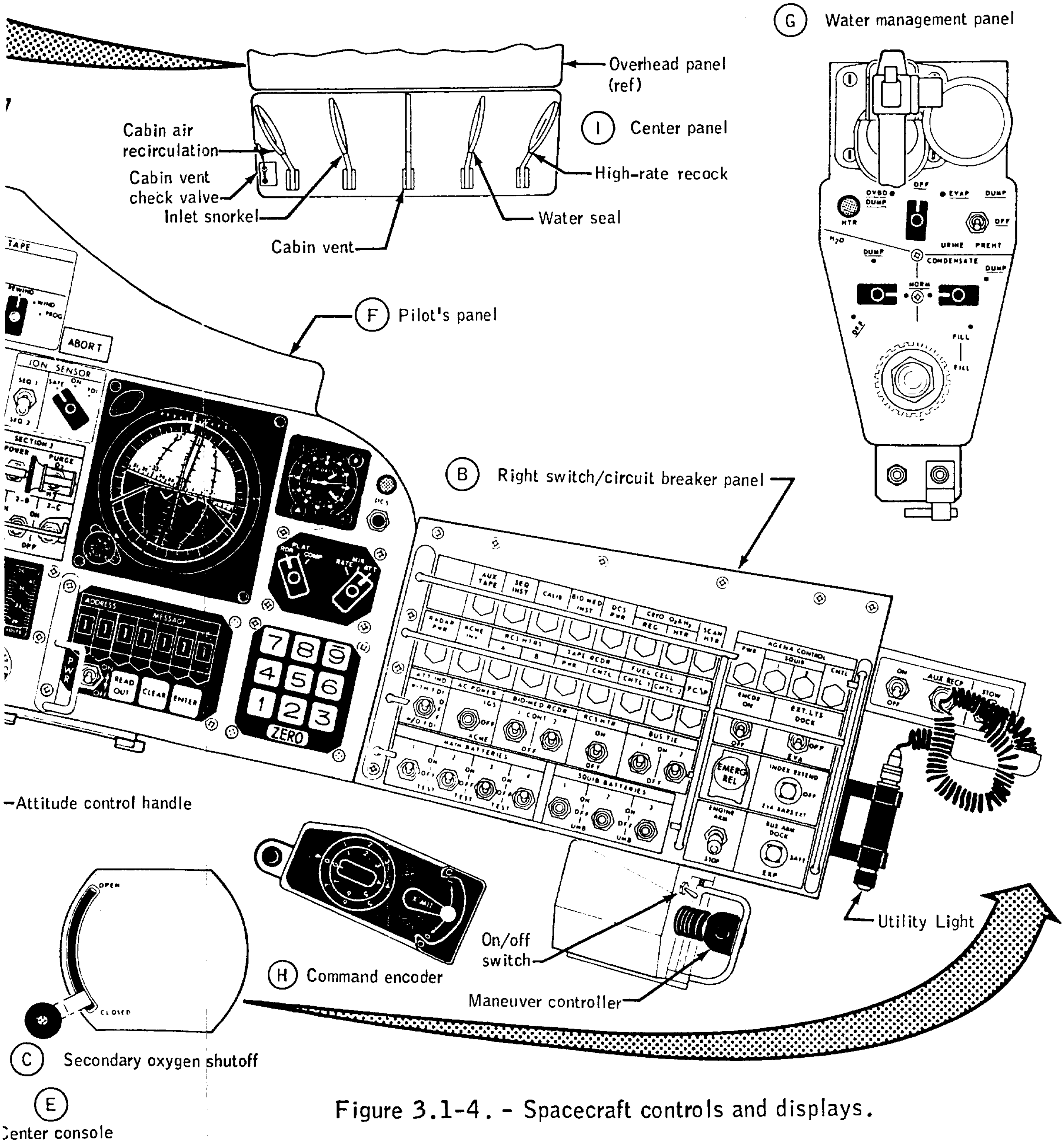

Figure 3.1-4. - Spacecraft controls and displays.

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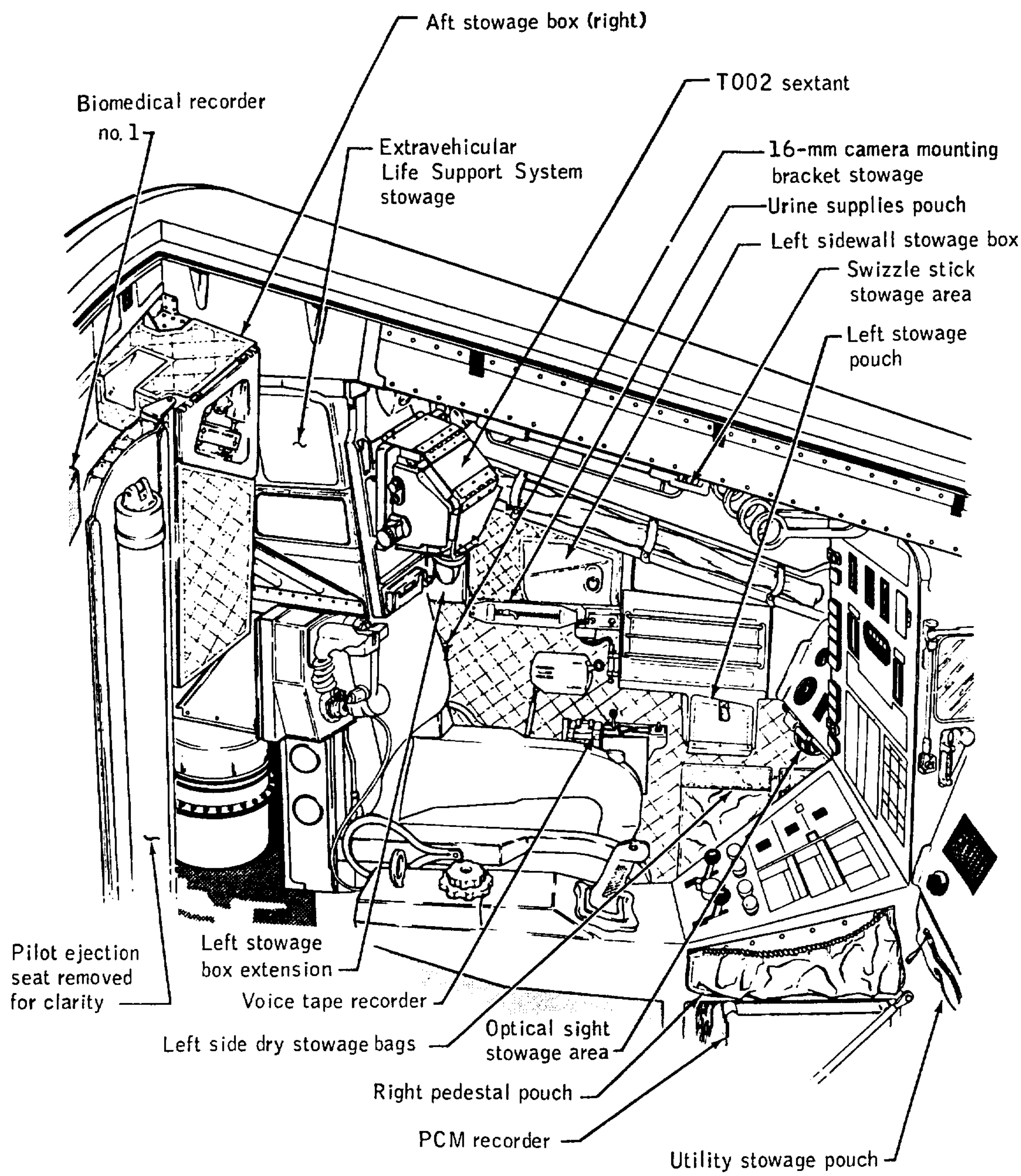

(a) View looking into command pilot's side.

Figure 3.1-5. - Spacecraft interior stowage areas.

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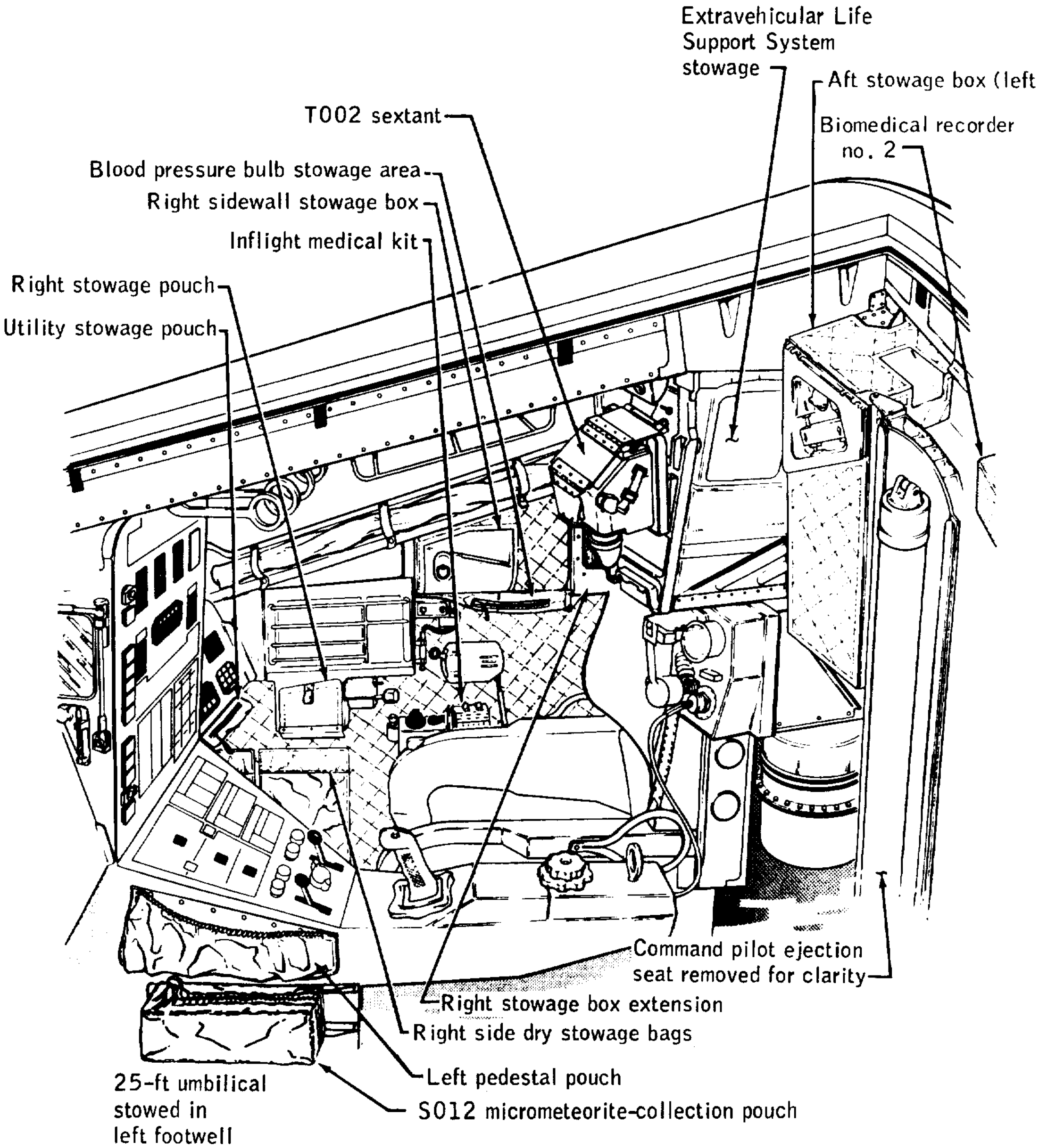

(b) View looking into pilot's side.

Figure 3.1-5. - Concluded.

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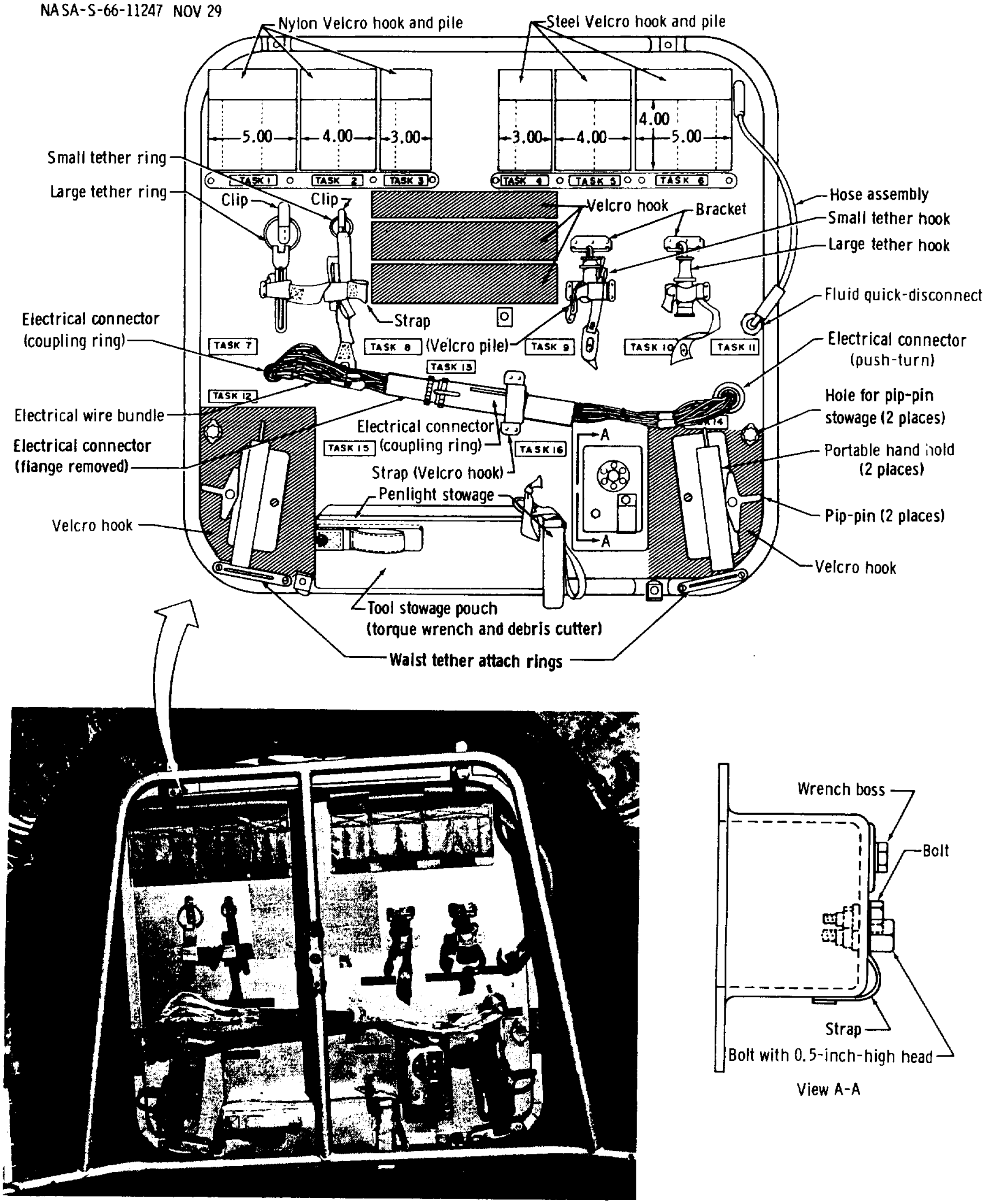

Figure 3.1-6. - Adapter work station.

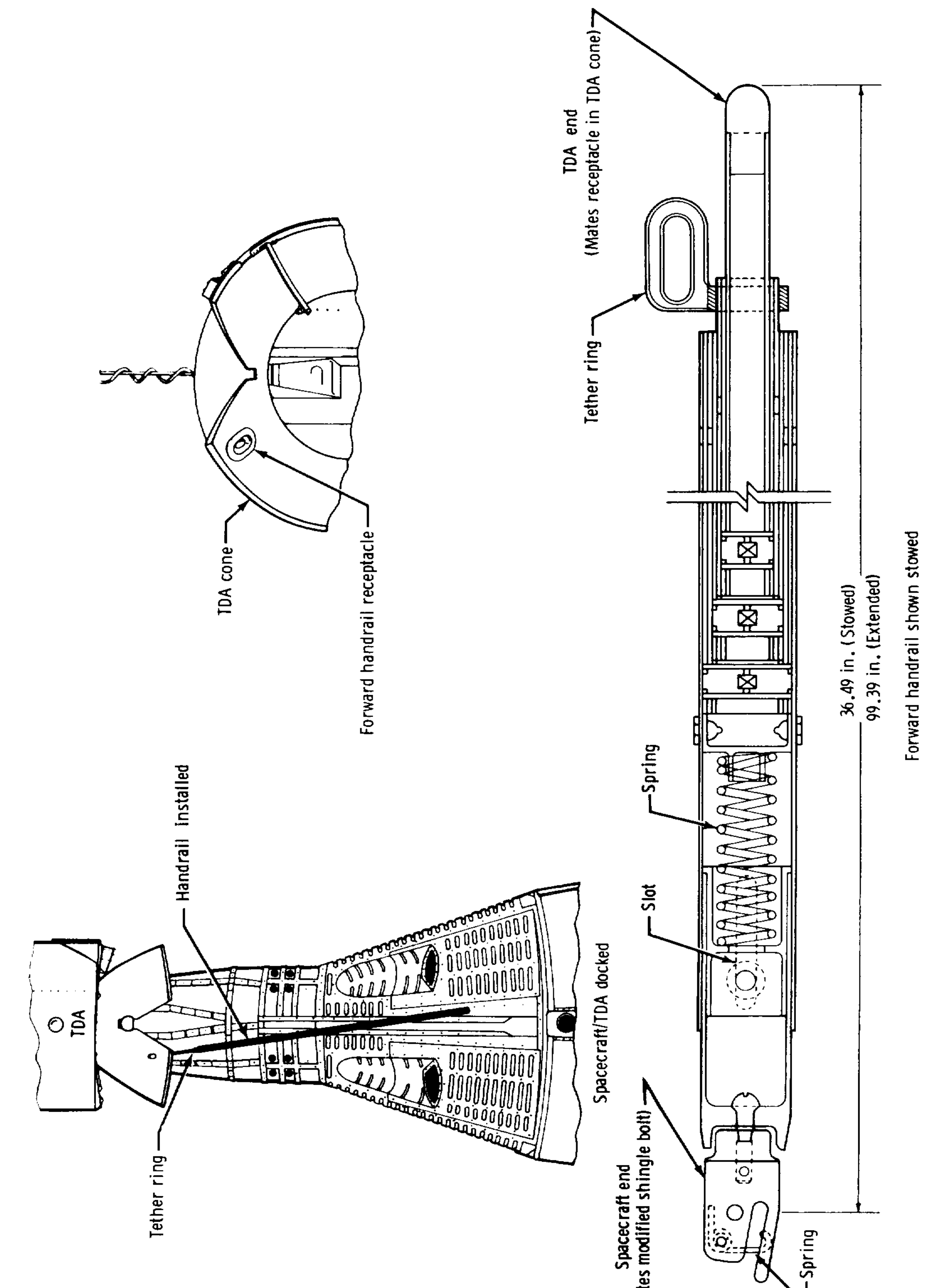

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Figure 3.1-7. - Telescoping handrail.

3.2 GEMINI LAUNCH VEHICLE

Gemini Launch Vehicle (GLV) 12 was of the same basic configuration as the GLV's used on previous Gemini missions, and there were no significant differences between GLV-12 and GLV-11.

3.3 GEMINI SPACE VEHICLE WEIGHT AND BALANCE DATA

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

Condition	Weight (including spacecraft), lb (a)	Center-of-gravity location, in. (a), (b)		
		X	Y	Z
Stage I ignition	345 710	774.4	0.000	59.90
Lift-off	342 092	794.6	0.000	59.90
First-stage engine cutoff (BECO)	85 386	439.1	-0.100	59.90
Second-stage start of steady-state combustion	74 011	343.5	-0.047	59.97
Second-stage engine cutoff (SECO)	14 499	281.0	-0.128	59.93

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

[b]Refer to figure 3.0-1 for the Gemini Space Vehicle coordinate system. Along the X-axis, the center of gravity is referenced to GLV station 0.00. Along the 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 12 weight and balance data are as follows:

Condition	Weight, lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Launch, gross weight	8296.47	-1.32	+1.91	104.84
Retrograde	5627.87	+0.05	-1.15	129.62
Reentry (0.05g)	4802.18	+0.17	-1.61	136.58
Main parachute deployment	4403.74	+0.14	-1.74	129.83
Landing (no parachute)	4292.93	+0.14	-1.80	127.77

[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

Gemini Agena Target Vehicle (GATV) 5001 was of the same configuration as GATV 5006 used for the Gemini XI mission (ref. 12). The only significant differences were in the Target Docking Adapter (TDA) and these were as follows:

(a) A docking bar clamp (fig. 3.4-1), designed for one-handed EVA operation, was stowed on the outboard surface of the tether container which was mounted on the TDA cone. The clamp was attached to the docking bar by the extravehicular pilot. The clamp positioned the GATV tether such that the tether would not apply high mechanical advantage loads at the end of the docking bar.

(b) An EVA work station (fig. 3.4-2) was installed on the left-hand side of the TDA docking cone in view of the EVA camera mounted on the adapter retrograde section. The work station included the following:

(1) An Apollo torque wrench and a fixed bolt

(2) A push-pull type fluid quick disconnect attached to a high-pressure type hose

(3) Electrical cable assembly with a push-turn connector

(4) Three pip-pins for waist tether attachment to the cylindrical section of the TDA. (Two additional pip-pins were stowed with the portable handholds near the work station.)

(c) Two portable handholds were stowed near the work station on the outboard side of the TDA cone, and utilized waist-tether pip-pins for hold-down during the launch phase. The nylon Velcro base plates of the portable handholds were attached by the extravehicular pilot to the polyester Velcro patches added at different locations on the TDA cylindrical section.

(d) Sixteen pip-pins receptacles were located on the TDA cylindrical section for attachment of the extravehicular pilot's waist tether to a pip-pin for work station operations.

(e) Four tether-attach rings were installed on the TDA cone lip and two on the shroud mating ring for waist tether attachment during GATV tether attachment to the docking bar. Two handholds were also attached to the shroud mating ring.

(f) The GATV tether was modified to provide a large tether loop to permit attachment to the docking bar by the extravehicular pilot or by "fly-in".

(g) The entire static discharge device assembly was removed from the TDA.

(h) The TDA docking cone was modified to include the funnel hole for the telescoping handrail.

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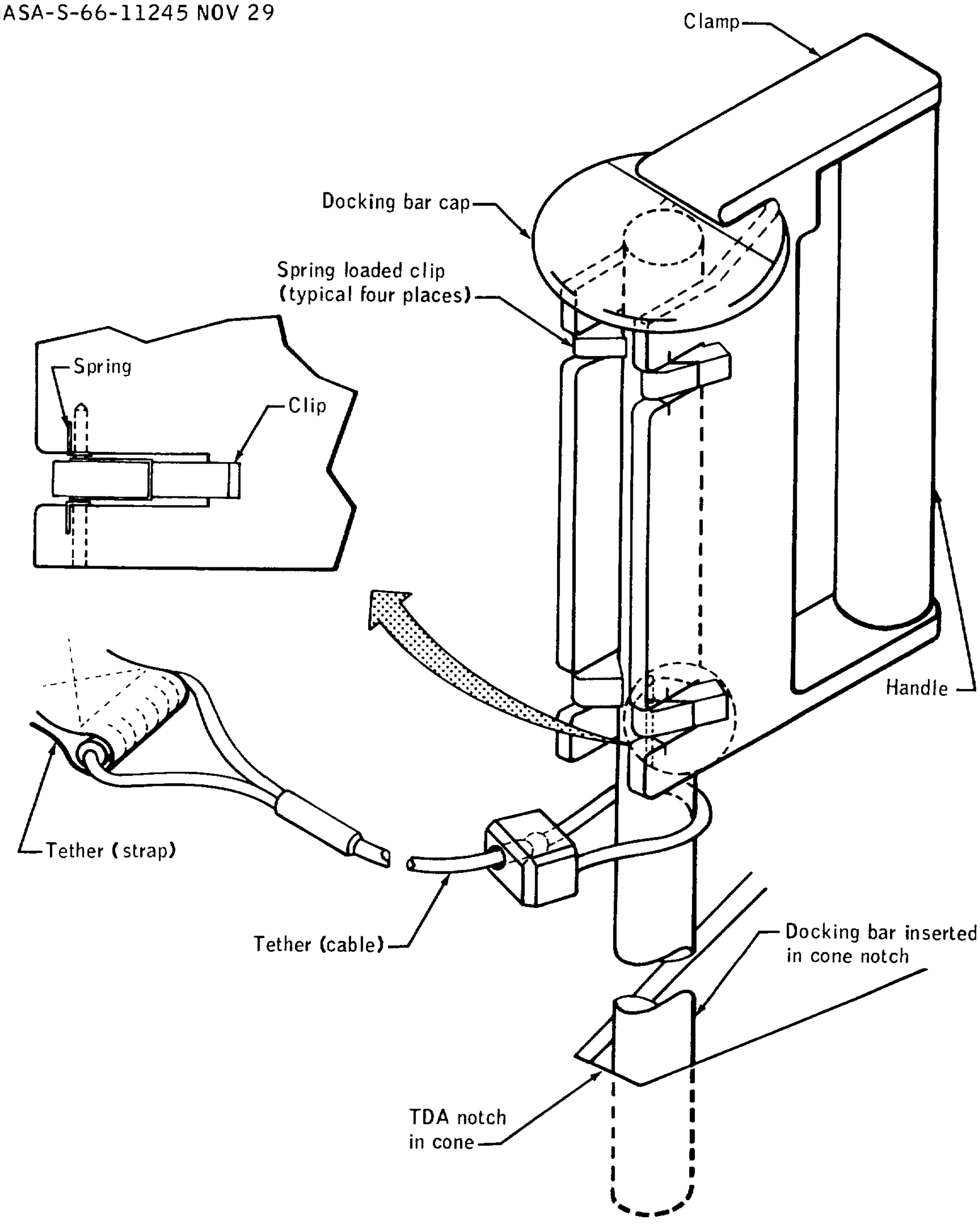

Figure 3.4-1. - Docking bar clamp.

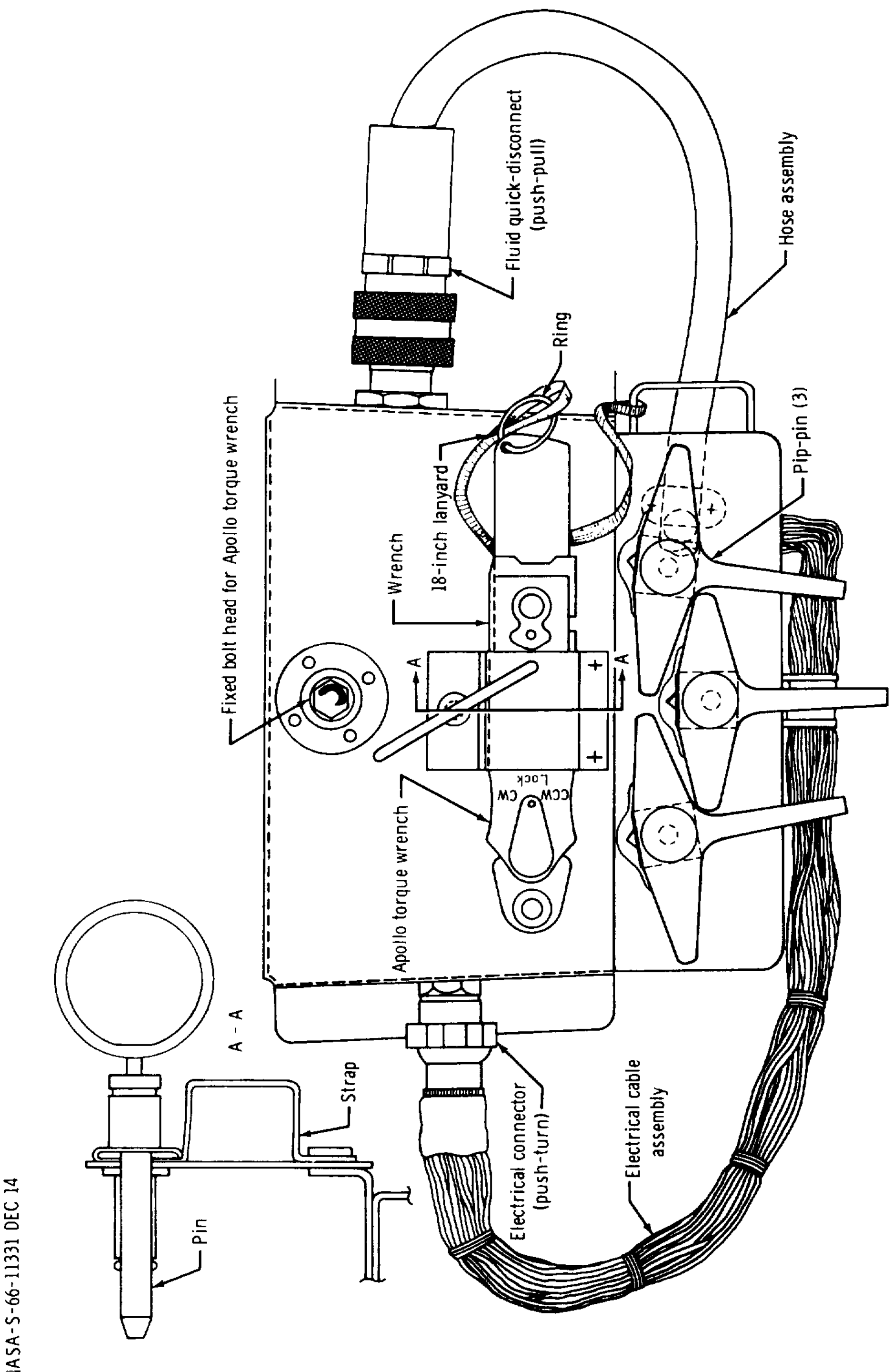

Figure 3.4-2. - TDA EVA work station.

3.5 TARGET LAUNCH VEHICLE

Target Launch Vehicle (TLV) 5307 was an Atlas Standard Launch Vehicle (SLV-3) which was originally configured as Lunar Orbiter Vehicle 5803. This vehicle contained modifications that had not previously been incorporated on Gemini TLV's. Although most of these modifications were not required to support the Gemini XII mission, they were not deleted from the vehicle. Considering these changes, the only significant differences between TLV 5307 and TLV 5306 (used for the Gemini XI mission) were as follows:

(a) Fifteen resistors in the electrical distribution box were replaced by a special type to improve reliability of the electrical system.

(b) The booster half of the fuel staging valve was modified to strengthen the four "spider webs" and increase the thickness of the cylinder walls. The support poppet was also moved forward to provide greater valve opening and to eliminate the need for a spacer.

3.6 GEMINI ATLAS-AGENA TARGET VEHICLE

WEIGHT AND BALANCE DATA

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

Condition	Weight (including GATV), lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Ignition	281 343	--	--	--
Lift-off	279 182	821.1	-0.05	-0.4
Booster engine cutoff (BECO)	73 701	849.5	-1.7	-1.5
Sustainer engine cutoff (SECO)	26 527	549.4	-2.0	-3.3
Vernier engine cutoff (VECO)	26 415	544.1	-2.1	-3.4

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

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

Condition	Weight, lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Launch (including shroud)	18 071	340.0	0.0	-0.1
Separation	17 653	337.2	0.0	-0.1
Insertion weight (after insertion firing)	7 114	344.1	0.0	-0.2

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

4.0 MISSION DESCRIPTION

4.1 ACTUAL MISSION

The Gemini XII mission was initiated when the Gemini Atlas-Agena Target Vehicle (GAATV) lifted off at 19:07:58.688 G.m.t. on November 11, 1966. The Gemini Agena Target Vehicle (GATV) achieved a nearly circular orbit with a perigee of 156 nautical miles and an apogee of 163 nautical miles. One hour 38 minutes 34.731 seconds after the GAATV lift-off, the Gemini Space Vehicle was launched at the start of the 30-second launch window available for an M=3 (third spacecraft revolution) rendezvous with the GATV. The spacecraft was inserted into a satisfactorily phased orbit with a perigee of 87 nautical miles and an apogee of 146 nautical miles, which was acceptable for a rendezvous with the GATV in the third revolution (M=3). The mission is outlined in figure 4.1-1, which shows both the planned and the actual mission activities.

Maneuvers for the M=3 rendezvous were successfully accomplished as planned, and, at 3 hours 45 minutes ground elapsed time (g.e.t.), the crew reported station keeping with the GATV. Initial radar contact with the GATV was achieved at a range of 235.5 nautical miles, prior to the corrective combination maneuver. After the coelliptic maneuver was performed at a range of approximately 65 nautical miles from the GATV, the rendezvous radar lock-on indication and the radar range and range rate became erratic, necessitating the use of onboard backup charts for terminal phase maneuver determination. Because of the intermittent radar data, the computer was placed in the catch-up mode rather than the rendezvous mode since the TPI solution could not be obtained in time for TPI.

The initial docking occurred at 4:13:52 g.e.t. over the Coastal Sentry Quebec. Following this docking, two practice dockings were accomplished. The fuel-cell oxygen-to-water differential-pressure warning lights came on also during this period, and the M408 (Beta Spectrometer), M409 (Bremsstrahlung Spectrometer), and M405 (Tri-Axis Magnetometer) experiments were activated between the second and third dockings.

The crew ate after the final docking. During the eat period, Mode A of Experiment M408 (Beta Spectrometer) was accomplished. The fuel-cell differential-pressure warning lights again came on intermittently; however, extraction of drinking water caused the lights to extinguish. At the conclusion of the eat period, the crew was informed that the GATV primary propulsion system (PPS) maneuver originally scheduled for this time had been canceled because of a possible turbo-pump problem in the PPS. As a result of this anomaly, all planned GATV PPS maneuvers prior to landing of the Gemini spacecraft were canceled. The crew slept after receiving this information. The Experiment S012 (Micrometeorite Collection) door was opened by ground command over the RKV during revolution 6 and closed during revolution 10, again by command.

A decision was made to substitute photography of the solar eclipse for the planned high-apogee PPS maneuver. Two phasing maneuvers were accomplished with the GATV secondary propulsion system (SPS) to place the spacecraft in the correct position to photograph this phenomenon at 16:01:44 g.e.t. Good photographs were obtained of the eclipse.

Following the eclipse photography, the crew ate and then started preparations for the initial standup extravehicular activity (EVA) and turned on the heater switch for Experiment S003 (Frog Egg Growth). The EVA was accomplished nominally and as scheduled, with hatch opening at 19 hours 29 minutes g.e.t. and repressurization beginning at 21 hours 58 minutes g.e.t. During the standup EVA, the pilot conducted planned exercises and Experiment S013 (Ultraviolet Astronomical Camera). Also the Experiment S012 (Micrometeorite Collection) package was retrieved. Following conclusion of the standup EVA, the crew ate.

The crew then conducted Modes A and B of Experiment S011 (Airglow Horizon Photography), took several photographs for Experiment S006 (Synoptic Weather Photography), conducted Experiment S029 (Libration Regions Photography), and performed Mode A of Experiment M408 (Beta Spectrometer). Following the conclusion of these experiments, the crew slept and ate. During the sleep period, the crew were awakened to purge the fuel cells because of a fuel-cell problem. Stack 2B was taken off the line since the purge did not correct the situation. Following conclusion of the sleep and eat periods, several photographs for Experiment S005 (Synoptic Terrain Photography) were taken. While maneuvering to take these photographs the crew determined that thrusters no. 2 and no. 4 of the Orbital Attitude and Maneuver System (OAMS) were considerably degraded.

Preparation for the umbilical EVA was initiated at 39 hours 40 minutes g.e.t. Unit I of Experiment S003 (Frog Egg Growth) was fixed at 41:43:40 g.e.t. The hatch was opened at 42:48:26 g.e.t. The pilot connected the tether between the GATV and the spacecraft and performed all other tasks in a deliberate and calm manner. The umbilical EVA was totally successful, with all objectives accomplished. In addition, the Experiment S010 (Agena Micrometeorite Collection) package was deployed on the GATV. The spacecraft hatch was closed at 44 hours 55 minutes g.e.t.

The tether evaluation was initiated at 47 hours 23 minutes g.e.t., with the third undocking and subsequent deployment of the tether. The tether deployed smoothly and initially tended to remain slack, with tautening occurring only occasionally. The crew reported that the spacecraft was difficult to stabilize because of the failure of thruster no. 8 in addition to thrusters 2 and 4. The spacecraft/GATV tethered configuration finally stabilized, and the crew reported that a gravity-gradient stabilization had been attained. The spacecraft docking bar was jettisoned, releasing the tether, at 51:50:57 g.e.t., and the spacecraft remained near the GATV while the crew performed a platform alignment. At 52:14:27 g.e.t., a posigrade separation maneuver was accomplished after which the crew ate and slept.

The crew had to be awakened early since a slow spacecraft power up was required because of the fuel cell problem. After powering up they performed a platform alignment prior to the phasing maneuver required to stop the opening rate between the spacecraft and GATV. Because of the thruster problems, the platform alignment went slower than expected and the platform was not fully aligned for this maneuver. The retrograde phasing maneuver was accomplished at 61:47:48 g.e.t. and resulted in a noticeable out-of-plane velocity; however, the separation rate was essentially stopped by this maneuver.

Preparation for the second standup EVA was initiated and became somewhat hurried. As a result of this, the crew requested and received permission to delay the EVA one revolution. During the preparation for EVA, the crew accomplished two sequences of Experiment S051 (Daytime Sodium Cloud) and one sequence of Experiment T002 (Manual Navigation Sightings).

The second standup EVA was accomplished as planned, with depressurization and pressurization at 66:05:55 g.e.t. and 67:03:03 g.e.t., respectively. The pilot jettisoned several pouches of excess equipment and waste. During this EVA, the pilot took a number of ultraviolet photographs of possible upper-atmosphere dust clouds.

Following conclusion of the second standup EVA, Modes A, E, and F of Experiment D010 (Ion-Sensing Attitude Control) were accomplished. The crew reported over Carnarvon during revolution 43 that they had difficulty in maneuvering the spacecraft to the attitudes required for this experiment because of the failure of thruster no. 8 in addition to no. 4, canceling roll-left capability (see section 5.1.8).

The crew ate and then conducted sequences of the following experiments:

D010, Ion-Sensing Attitude Control

S005, Synoptic Terrain Photography

S006, Synoptic Weather Photography

S029, Libration Regions Photography

S011, Airglow Horizon Photography

T002, Manual Navigation Sightings

M408, Beta Spectrometer

At the conclusion of these experiments the crew ate and slept. During the sleep period, Mode F of Experiment D010 was conducted. The crew was awakened at 84 hours 46 minutes g.e.t. and requested to turn off fuel cell stack 1C because the current had dropped to zero.

After taking care of the fuel-cell system, the crew ate and then conducted sequences of the following experiments:

S003, Frog Egg Growth

T002, Manual Navigation Sightings

S005, Synoptic Terrain Photography

S006, Synoptic Weather Photography

The drinking water in the adapter tank became depleted during revolution 55, and the crew experienced difficulty in extracting water from the cabin reservoir. This problem was procedural and, after closing the bleed valve of the blood pressure bulb, the crew succeeded in pressurizing the cabin tank and no further difficulty was experienced in extracting water.

At 88 hours 57 minutes g.e.t., a propulsion system test was performed to determine the status of the thrusters. This test indicated that thrusters 4 and 8 were delivering no measurable thrust and thrusters 2 and 7 were severely degraded. All other thrusters appeared normal.

At this time the remaining two stacks in fuel cell section 2 were carrying less than half of their normal share of load. Sequences of the following experiments were then performed:

T002, Manual Navigation Sightings

D010, Ion-Sensing Attitude Control

S005, Synoptic Terrain Photography

S006, Synoptic Weather Photography

Pre-retrofire preparation was initiated at 90 hours 30 minutes g.e.t., with the stowage of loose equipment. Main batteries 1 and 4 were placed on the line to aid the fuel cells in handling the additional load of the platform.

At activation of the Reentry Control System (RCS), regulated pressure in the A-ring was higher than normal and continued to rise above its normal range. In an attempt to lower this pressure, the A-ring was used to align the platform. Immediately prior to the final orbit, batteries 2 and 3 were added to the main bus and all load was removed from fuel cell section 2. Immediately prior to retrofire, over Carnarvon, the RCS pressure had decreased to 335 psi as a result of deliberately using propellants to lower this pressure. Retrofire was initiated automatically and occurred at the planned time of 93:59:58 g.e.t. over the Canton Island station. Reentry was nominal using automatic control of the RCS, and the spacecraft landed 2.6 nautical miles from the planned landing point and within 3 nautical miles of the prime recovery ship, the U.S.S. Wasp. Landing was harder than expected, and, as a result, a spacecraft shingle was bent. The crew elected to be brought aboard the U.S.S. Wasp by helicopter, and 28 minutes after landing they were on the deck of the ship.

After spacecraft landing and recovery, a 20-second firing of the GATV PPS was attempted to check the turbo-pump anomaly noted during the PPS insertion firing. (A PPS firing was originally planned to deplete the PPS fuel and place the vehicle in a circular parking orbit but was not attempted because of the lack of attitude control gas.) A turbine overspeed caused a shutdown of the PPS. The GATV remained in an elliptical orbit with a perigee of 138 nautical miles and an apogee of 158 nautical miles at the termination of the mission.

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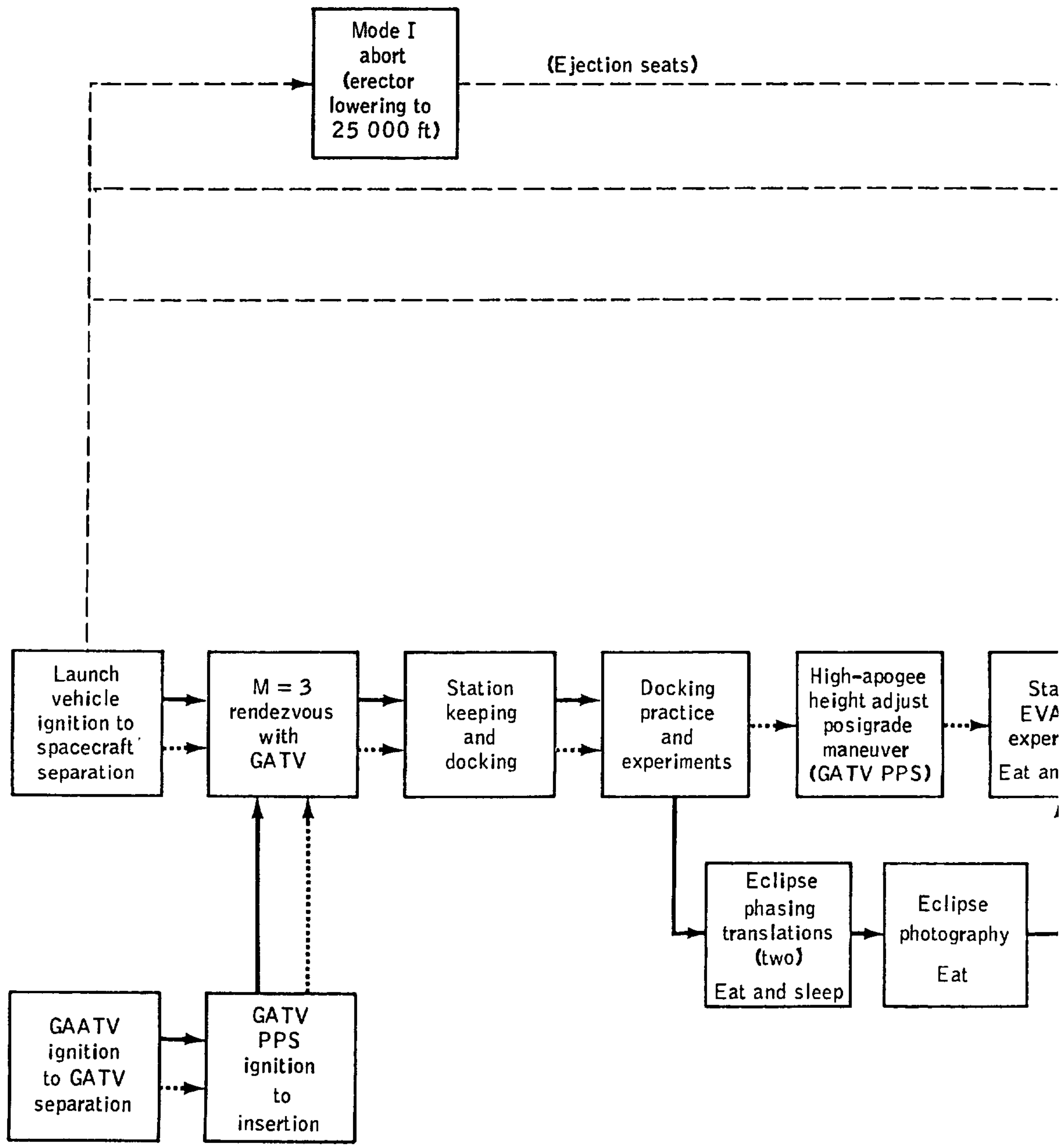

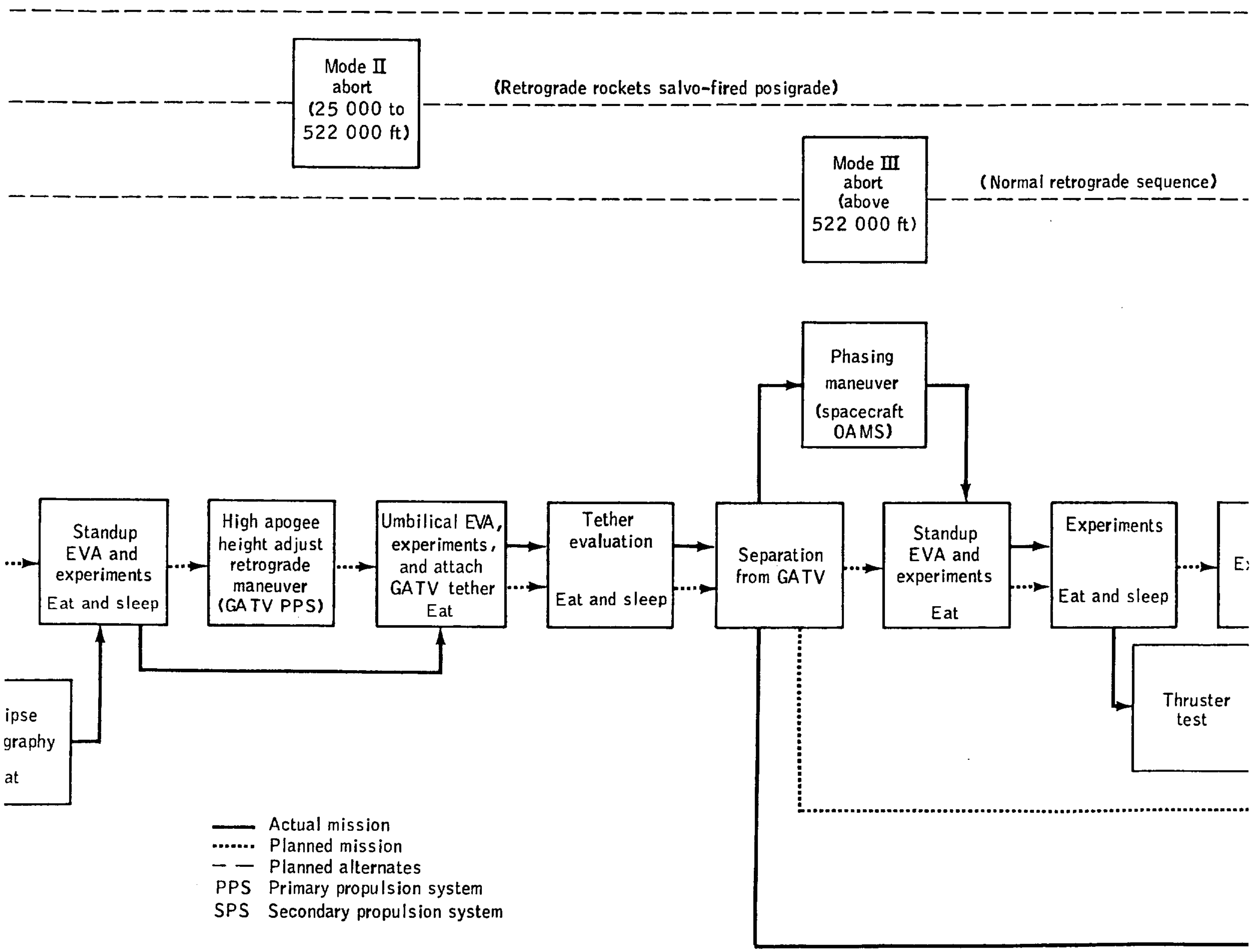

Mode II abort (25 000 to 522 000 ft)
(Retrograde rockets salvo-fired posigrade)
Mode III abort (above 522 000 ft)
(Normal retrograde sequence)
Phasing maneuver (spacecraft OAMS)
Standup EVA and experiments
Eat and sleep
High apogee height adjust retrograde maneuver (GATV PPS)
Umbilical EVA, experiments, and attach GATV tether
Eat
Tether evaluation
Eat and sleep
Separation from GATV
Standup EVA and experiments
Eat
Experiments
Eat and sleep
Thruster test
ipse
graphy
at
Actual mission
Planned mission
Planned alternates
PPS Primary propulsion system
SPS Secondary propulsion system

Figure 4

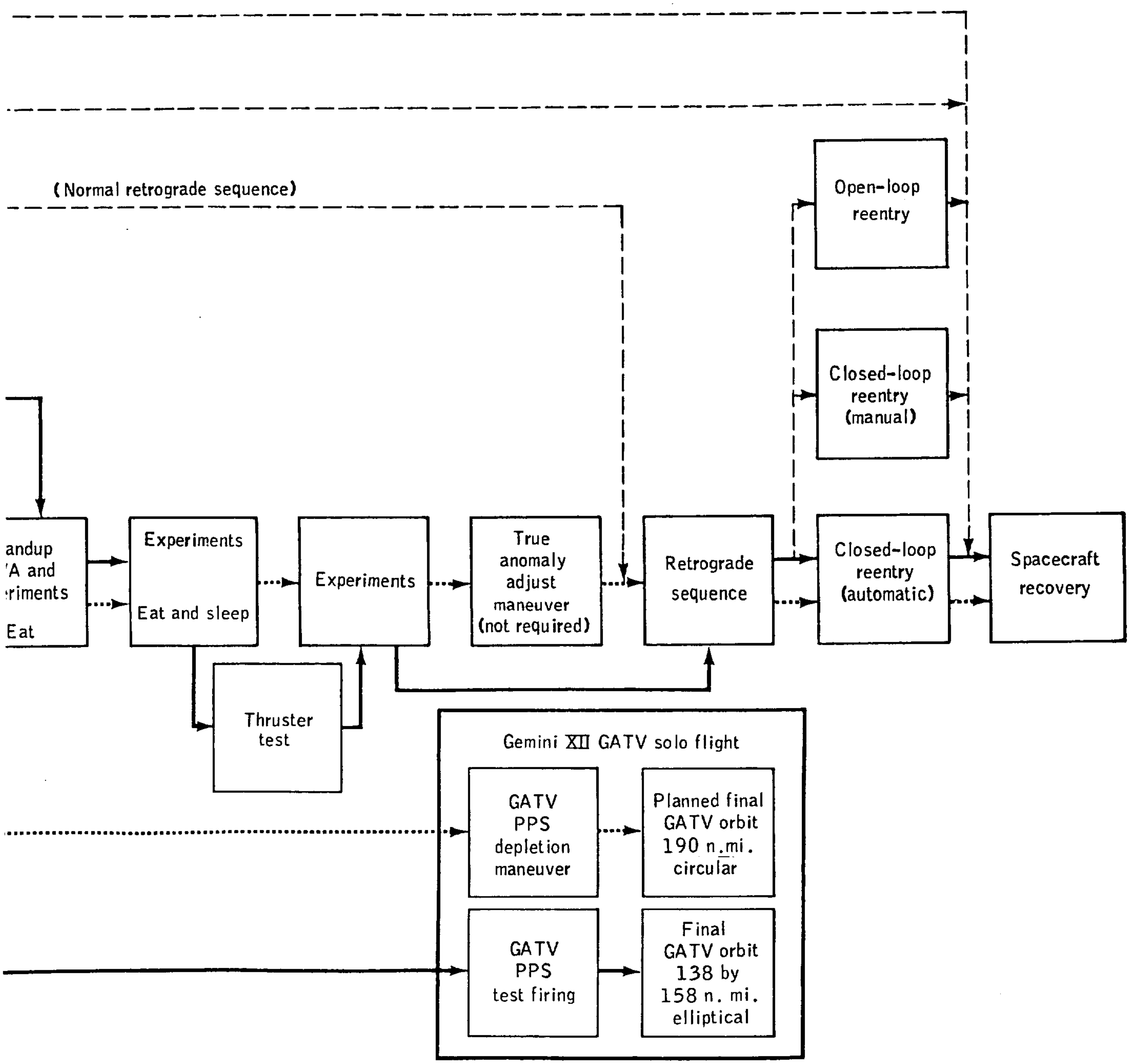

Figure 4.1-1. - Planned and actual Gemini XII mission with planned alternates included.

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 and in tables 4.2-III and 4.2-IV for the Gemini Atlas-Agena Target Vehicle.

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	-3.23	+0.17
Stage I MDTCPS makes, subassembly 1	-2.30	-2.20	+0.10
Stage I MDTCPS makes, subassembly 2	-2.30	-2.20	+0.10
Shutdown lockout (backup)	-0.10	-0.09	+0.01
Lift-off (pad disconnect separation)	20:46:33.419 G.m.t.		
Roll program start (launch azimuth = 100.6 deg)	8.00	7.99	-0.01
Roll program end	20.48	20.47	-0.01
Pitch program rate no. 1 start	23.04	23.05	+0.01
Pitch program rate no. 1 end, no. 2 start	88.32	88.26	+0.06
First IGS update initiated	100.00	100.00	0.00
Control system gain change no. 1	109.96	109.68	-0.28
Pitch program rate no. 2 end, no. 3 start	119.04	118.96	-0.08
Second IGS update initiated	140.00	140.00	0.00
Stage I engine shutdown circuitry armed	144.64	144.54	-0.10
Stage I MDTCPS unmake	153.30	154.71	+1.41
BECO (Stage I engine shutdown (87FS2))	153.38	154.75	+1.37
Staging switches actuate	153.38	154.75	+1.37
Signals from Stage I rate gyro package to Flight Control System discontinued	153.38	154.75	+1.37
Hydraulic switchover lockout	153.38	154.75	+1.37
Telemetry ceases, Stage I	153.38	154.75	+1.37
Staging nuts detonate	153.38	154.75	+1.37
Stage II engine ignition signal (91FS1)	153.38	154.75	+1.37
Control system gain change	153.38	154.75	+1.37
Stage separation begin	154.68	155.45	+0.77
Stage II engine MDFJPS make	154.88	155.48	+0.60
Pitch program rate no. 3 end	162.56	162.41	-0.15
RGS guidance enable	162.56	162.40	-0.16
First guidance command signal received by TARS	169.00	168.01	-0.99
Stage II engine shutdown circuitry armed	317.44	317.10	-0.34
SECO (Stage II engine shutdown (91FS2))	339.76	343.54	+3.78
Redundant Stage II shutdown	339.76	343.59	+3.83
Stage II MDFJPS break	340.06	343.68	+3.62
Spacecraft separation (shape charge fired)	359.76	366.72	+6.96
OAMS on	359.76	366.50	+6.74
OAMS off (final)[a]	389.76	439.77	+50.01

[a]During a 73-second time interval, several maneuvers were made—a separation maneuver of 3 seconds, an Insertion Velocity Adjust Routine (IVAR) maneuver of 34 seconds, and a lateral maneuver of 22 seconds.

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

ORBITAL AND REENTRY PHASES

Event	Ground elapsed time, hr:min:sec		Difference, sec
	Planned[a]	Actual	
Phase adjust maneuver	0:49:40	0:49:40	0
Plane change maneuver	1:14:22	1:14:22	0
Corrective combination maneuver	1:47:52	1:47:52	0
Coelliptic maneuver	2:22:54	2:22:55	+1
Terminal phase initiate maneuver	3:05:48	3:05:47	-1
First midcourse correction	--	3:11:14	--
Second midcourse correction	--	3:17:07	--
Third midcourse correction	--	3:23:46	--
Fourth midcourse correction	--	3:29:05	--
Terminal phase finalize (braking) maneuver	3:38:16	3:32:36	-340
Eclipse phasing maneuver no. 1 (GATV SPS) (docked)	7:05:06	7:05:06	0
Eclipse phasing maneuver no. 2 (GATV SPS) (docked)	15:16:18	15:16:18	0
Separation maneuver	52:14:27	52:14:27	0
Phasing maneuver	61:47:47	61:47:48	+1
Equipment adapter separation	93:58:58	93:59:03	+5
Retrofire initiation	93:59:58	93:59:58	0
Begin blackout	94:22:29	94:22:04	-25
End blackout	94:27:11	94:27:25	+14
Drogue deployment	94:28:59	94:29:10	+11
Pilot parachute deployment, main parachute initiation	94:30:29	94:30:35	+6
Landing	94:39:29	94:34:30	+1

[a]The planned values for the orbital phase are the latest information forwarded to the crew prior to each maneuver.

TABLE 4.2-III.- SEQUENCE OF EVENTS FOR GAATV LAUNCH PHASE

Event	Time from lift-off, sec		Difference, sec
	Planned	Actual	
Lift-off	19:07:58.688 G.m.t.		
Booster engine cutoff (BECO)	131.50	131.11	-0.39
Booster engine separation (BECO + 3.0 sec)	134.50	134.66	+0.16
Primary sequencer (D-timer) start	277.60	277.58	-0.02
Sustainer engine cutoff (SECO)	280.73	280.12	-0.61
Vernier engine cutoff (VECO)	299.03	298.06	-0.97
TLV/GATV separation (retrorocket fire)	301.50	300.30	-1.20
Initiate horizon sensor roll control	304.00	302.70	-1.30
Start 90 deg/min pitch-down	338.60	338.60	0.00
Stop 90 deg/min pitch-down	351.60	351.60	0.00
Start 3.99 deg/min orbital pitch rate	351.60	351.83	+0.23
SPS ignition	353.60	353.66	+0.06
PPS ignition (90-percent chamber pressure)	372.10	372.51	+0.41
SPS thrust cutoff	373.60	373.65	+0.05
Fire jettison nose shroud squibs	381.60	381.72	+0.12
Velocity meter cutoff	553.86	555.85	+1.99

TABLE 4.2-IV.- SEQUENCE OF EVENTS FOR GATV ORBITAL PHASE

Event	Ground elapsed time, hr:min:sec		Difference, sec
	Planned[a]	Actual	
Eclipse phasing maneuver no. 1 (SPS) (docked)	7:05:06	7:05:06	0
Eclipse phasing maneuver no. 2 (SPS) (docked)	15:16:18	15:16:18	0
Height adjust maneuver (PPS)	98:50:27	98:50:27	0

4.3 FLIGHT TRAJECTORIES

In this section, the launch and orbital trajectories referred to as planned are either preflight calculated nominal trajectories (refs. 14 through 17) 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 by airborne instrumentation. For all trajectories except the 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 three revolutions, the eclipse revolution, and the period from retrofire to landing are shown in figure 4.3-1. Gemini Space Vehicle launch, orbit, rendezvous, and reentry trajectory curves are presented in figures 4.3-2 through 4.3-5. Gemini Atlas-Agena Target Vehicle (GAATV) launch curves are presented in figure 4.3-6.

4.3.1 Gemini Spacecraft

4.3.1.1 Launch.- The Gemini Space Vehicle was launched on a rendezvous launch azimuth of 100.6 degrees. The flight-controller plotboards indicated a launch trajectory that was satisfactory in every respect.

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

Facility	Time from lift-off, sec
IP 3600 (FPS-16)	0 to 36
GMCF (GE MOD III)	36 to 480

The actual launch trajectory, as compared with the planned launch trajectory in figure 4.3-2, was essentially nominal in velocity and slightly low in flight-path angle and altitude during Stage I powered flight. At first-stage engine cutoff (BECO), the velocity was high by 5 ft/sec, and flight-path angle and altitude were low by 0.25 of a degree and 948 feet, respectively. After BECO, the Radio Guidance System (RGS) had very little error to correct in order to guide the second stage to a satisfactory spacecraft insertion. At second-stage engine cutoff (SECO), the velocity, flight-path angle, and altitude were low by 10 ft/sec, 0.01 of a degree, and 952 feet, respectively.

At spacecraft separation, the actual velocity and altitude were low by 15 ft/sec and 988 feet, respectively, and the flight-path angle, measured to the nearest one-hundredth of a degree, was zero as planned. The second-stage tail-off ΔV was 5 ft/sec less than predicted. Table 4.3-I contains a comparison of planned and actual conditions at BECO, SECO, and spacecraft separation. The actual conditions at BECO were obtained from the GMCF tracking data. The preliminary conditions at SECO and spacecraft separation were obtained from MISTRAM and Grand Turk tracking data integrated back through the preliminary Insertion Velocity Adjust Routine (IVAR) maneuver. The final conditions were obtained by integrating the best-estimated-trajectory* orbital fit back through the IVAR and separation maneuvers, and the tail-off impulse as determined from telemetry records of Inertial Guidance System (IGS) and accelerometer 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 12 ft/sec and high by 0.08 of a 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 1 ft/sec and low by 0.04 of a degree, respectively, when compared with the later ephemeris data.

4.3.1.2 Orbit.- Table 4.3-II and figure 4.3-3 show the planned and actual orbital elements after each maneuver, and table 4.3-III shows the orbital elements for selected revolutions from insertion to retrofire. The planned elements shown in these tables were obtained from Gemini tracking network data as calculated in real time by the RTCC. The actual elements were obtained by integrating, after each midcourse and terminal phase maneuver, the Gemini tracking network vectors.

*This best estimated trajectory was based on tracking data obtained during the complete first revolution.

The planned trajectory and the actual trajectory for the rendezvous in the third spacecraft revolution (M=3) are presented in figure 4.3-4. The planned, ground-commanded, and actual maneuvers are presented in table 4.3-IV. The planned trajectory for the rendezvous was obtained from the real-time solution based on the vector from the Bermuda station for GATV revolution 2 and on the vector from the Grand Turk Island tracking station for spacecraft revolution 1. The ground-commanded maneuvers were determined from spacecraft and GATV vectors using the planned maneuvers which were updated after each actual maneuver. The actual trajectory during the initial rendezvous was reconstructed utilizing anchor vectors (obtained from the best estimated trajectory) and the actual maneuvers (derived from the Inertial Guidance System (IGS) postflight analysis) applied as instantaneous changes in velocity.

After spacecraft orbital insertion, ground computations indicated a nominal situation for obtaining a third-orbit rendezvous. At spacecraft insertion, the range between Spacecraft 12 and the Gemini XII GATV was approximately 500 nautical miles, and the out-of-plane velocity error resulting from the spacecraft launch-vehicle ascent yaw steering was about 8 ft/sec.

At 49 minutes 40 seconds g.e.t., a phase-adjust maneuver was initiated near first apogee. A horizontal posigrade ΔV of 66.7 ft/sec was applied with the aft-firing thrusters. The resultant altitude at perigee was about 124 nautical miles, and at apogee about 146 nautical miles. At 1:14:22 g.e.t., a plane change maneuver was initiated with a ΔV of 8.5 ft/sec to the south. This maneuver was computed onboard by the pilot and differed from the ground solution by 1 ft/sec and had about a 3-minute time difference.

At 1:47:52 g.e.t., a corrective combination maneuver was initiated. The actual ΔV of 8.2 ft/sec was applied with the aft-firing thrusters at a pitch attitude of 27.3 degrees and a yaw left of 7.7 degrees.

A coelliptic maneuver was initiated at 2:22:55 g.e.t. and was performed nominally with the aft-firing thrusters. The actual ΔV of 49.9 ft/sec was applied at a pitch-up attitude of 7.4 degrees and a yaw-right of 0.2 of a degree. This maneuver was computed onboard by the pilot and agreed very closely with the ground-computed solution. The ground-computed coelliptic maneuver was ΔV_X of plus 49.8 ft/sec, ΔV_Y of plus 3.5 ft/sec, ΔV_Z of minus 0.7 ft/sec. The maneuver computed by the pilot was ΔV_X of plus 49.5 ft/sec, ΔV_Y of minus 6.5 ft/sec, and ΔV_Z of minus 0.1 ft/sec. The resultant orbit was about 149 by 153 nautical miles and the differential altitude (Δh) between the spacecraft and GATV orbits varied between about 9.2 and 10.5 nautical miles. The Δh varied from

9.2 nautical miles at the time of the coelliptic maneuver to 10.2 nautical miles at the time of the terminal phase initiate (TPI) maneuver. This ellipticity is attributed to a slight excess velocity applied during the corrective combination maneuver, the application of 6.5 ft/sec up during the coelliptic maneuver instead of the ground-computed 3.5 ft/sec, and the fact that a small amount of ellipticity may result from a precisely performed maneuver that was incorrectly computed because of tracking errors.

The TPI maneuver was initiated at 3:05:47 g.e.t. when the elevation angle to the GATV was approximately 27.0 degrees, and the range was about 22.1 nautical miles. A total ΔV of 21.8 ft/sec was applied. In computer coordinates, the actual ΔV applied resuled in a ΔV_X of plus 19.3 ft/sec, a ΔV_Y of minus 10.1 ft/sec, and a ΔV_Z of plus 0.5 ft/sec. This agreed with the onboard-computed backup solution of 22 ft/sec forward. The ground-commanded TPI solution indicated that TPI should occur at 3:05:46 g.e.t. with a ΔV of 23.2 ft/sec to be applied. In computer coordinates, the ground-commanded ΔV resulted in a ΔV_X of plus 18.9 ft/sec, ΔV_Y of plus 13.2 ft/sec, and a ΔV_Z of minus 2.6 ft/sec. This resolved into 22.8 ft/sec forward, 3.2 ft/sec up, and 2.7 ft/sec right. It should be noted that the onboard radar was not working properly; thus the pilot was computing the terminal phase maneuvers using his onboard backup charts.

For the first midcourse correction, the pilot computed 1.5 ft/sec up. The actual maneuver applied, in computer coordinates, was ΔV_X of minus 0.1 ft/sec, ΔV_Y of minus 1.7 ft/sec, and ΔV_Z of plus 0.1 ft/sec. This resolved into 1.0 ft/sec forward, 1.4 ft/sec up, and 0.1 ft/sec left, considering the spacecraft boresighted on the target. This correction was initiated at 3:11:14 g.e.t.

The second midcourse correction, to be applied at 3:17:07 g.e.t., resulted in a ΔV_X of minus 1.5 ft/sec, ΔV_Y of plus 2.7 ft/sec, and ΔV_Z of minus 0.1 ft/sec, which resolves into 1.2 ft/sec forward, 2.9 ft/sec up, and 0.1 ft/sec right. The pilot's onboard computation was 2 ft/sec up.

The third correction, to be applied at 3:23:46 g.e.t., resulted in a ΔV_X of plus 0.3 ft/sec, ΔV_Y of plus 0.5 ft/sec, and ΔV_Z of plus 1.2 ft/sec. This resolves into 0.4 ft/sec aft, 0.3 ft/sec down, and 1.3 ft/sec left. The pilot's onboard computation was zero; however, from line of sight, the command pilot decided to apply 1.0 ft/sec left.

The fourth midcourse correction, applied at 3:29:05 g.e.t., resulted in a ΔV_X of plus 1.1 ft/sec, ΔV_Y of plus 4.8 ft/sec, and ΔV_Z of minus 0.2 ft/sec. This resolves into 4.8 ft/sec aft, 1.1 ft/sec down, and 0.2 ft/sec right.

The terminal pahse finalize (TPF) maneuver was initiated at 3:32:36 g.e.t., and braking thrusts were applied intermittently over the next 13 minutes. An effective resultant velocity of about 20 ft/sec was added to the spacecraft orbit. At 3 hours 46 minutes g.e.t., the spacecraft was less than 50 feet from the GATV and station keeping had been initiated.

The total propellant used for the M=3 rendezvous was about 280 pounds, which includes about 55 pounds for the IVAR maneuver at insertion. Approximately 65 pounds of propellant were used for TPI through braking. This is considered very efficient considering that the onboard radar was not working properly.

At 7:05:06 g.e.t., a retrograde phasing maneuver of 44 ft/sec was applied to the docked vehicles with the GATV secondary propulsion system (SPS) to allow the crew to photograph the solar eclipse on November 12, 1966. This maneuver was not performed accurately and another phasing maneuver was performed at 15:16:18 g.e.t. A ΔV of 17 ft/sec was applied to the docked vehicles with the SPS. The execution of these two maneuvers provided the desired results, and the crew reported that they had passed through the total eclipse starting at 16:01:44 g.e.t. According to post-flight calculations, the spacecraft passed within about three nautical miles of the center of the 15-nautical-mile-radius umbra core.

These two phasing maneuvers, the spacecraft separation maneuver from the GATV, and a phasing maneuver for a possible re-rendezvous are shown in table 4.3-IV.

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 Woomera vector in revolution 58 through the planned retrofire sequences determined by the RTCC, and then using the Math Flow 7 reentry guidance scheme described in reference 17. The Woomera vector taken one revolution before retrofire was selected because the retrofire time transmitted to the spacecraft was based on that solution.

The actual trajectory was obtained by integrating the postretrofire White Sands vector back to retrofire, then integrating forward to landing through the Math Flow 7 reentry guidance scheme.

The times of reconstructed reentry trajectory events agree very well with the times of the actual reentry events. The roll initiate command agrees with the actual event, communication blackout times agree within 25 seconds of actual blackout, maximum acceleration loads compare with telemetry within 0.2g at analogous times, and parachute deployment altitudes at recorded sequence times are in accord with those reported in section 5.1.11. Table 4.3-I contains a comparison of reentry dynamic parameters and landing points. The final landing point was 2.6 nautical miles from the planned landing point. (See section 5.1.5 for a more detailed description of the landing coordinates.)

4.3.2 Gemini Atlas-Agena Target Vehicle

4.3.2.1 Launch.- The Gemini Atlas-Agena Target Vehicle (GAATV) was launched from an initial flight azimuth of 105 degrees to a final flight azimuth of 83.32 degrees. Sustainer steering was used to obtain the desired longitude of the ascending node and inclination angle. No booster steering was required. The flight-controller and range-safety plotboards all indicated a satisfactory Target Launch Vehicle (TLV) flight.

The Gemini Agena Target Vehicle (GATV) performed as planned, executing the 90 deg/min pitch-down rate after separation and continuing this rate until the D-timer started the minus 3.99 deg/min orbital geocentric pitch rate. The GATV achieved a nearly circular orbit with a perigee of 156.4 nautical miles and an apogee of 162.7 nautical miles.

The launch trajectory data presented in figure 4.3-6 are based on the real time output of the GMCF, the IP 3600, and the Bermuda (BDA) tracking radar. Data from these tracking facilities were used during the time periods listed in the following table:

Facility	Time from lift-off, sec
GMCF (GE MOD III)	0 to 339
IP 3600 (FPS-16, TPQ-18)	339 to 397
IP 3600, BDA (FPS-16)	397 to 608

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

4.3.2.2 Orbit.- The GATV was placed into the desired orbit for the planned Gemini Space Vehicle launch and rendezvous. Table 4.3-V contains a comparison of the planned and actual insertion conditions of the GATV. The actual conditions were obtained by integrating the Antigua tracking solution in the first revolution back to the GATV primary propulsion system (PPS) cutoff obtained from telemetry records.

After rendezvous and docking, the GATV SPS was used to perform two maneuvers to place the spacecraft in phase with the solar eclipse on revolution 10. Table 4.3-II shows the orbital elements for these maneuvers. Table 4.3-IV contains the maneuvers performed by the GATV, and table 4.3-VII contains the orbital parameters for every 12th revolution after insertion until spacecraft retrofire and includes the attempted firing of the PPS. Table 4.3-VIII shows the results of the attempted PPS firing. Because the GATV could not be attitude controlled, the ΔV is the only meaningful parameter; however, after the SPS ullage ignition, the PPS failed to ignite and the firing was terminated. (See section 5.4.2 for a description of this event.)

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 130.0 and 86.5 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 its 22-hour orbital lifetime. The Goddard Space Flight Center predicted reentry in revolution 15, with a predicted impact point off the western coast of Africa.

 [redacted]

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

Condition	Planned[a]	Actual	
		Preliminary	Final
BECO			
Time from lift-off, sec	153.38	154.75	154.75
Geodetic latitude, deg north	28.35	28.35	28.35
Longitude, deg west	79.64	79.62	79.62
Altitude, ft	205 384	204 436	204 436
Altitude, n. mi.	33.8	33.6	33.6
Range, n. mi.	49.4	50.5	50.5
Space-fixed velocity, ft/sec	9875	9880	9880
Space-fixed flight-path angle, deg	18.93	18.68	18.68
Space-fixed heading angle, deg east of north	99.73	99.54	99.54
SECO			
Time from lift-off, sec	339.76	343.54	343.54
Geodetic latitude, deg north	26.98	26.96	26.96
Longitude, deg west	71.98	71.85	71.88
Altitude, ft	527 147	526 884	526 195
Altitude, n. mi.	86.7	86.7	86.6
Range, n. mi.	465.2	472.7	472.3
Space-fixed velocity, ft/sec	25 647	25 636	25 637
Space-fixed flight-path angle, deg	0.01	0.02	0.0
Space-fixed heading angle, deg east of north	101.09	101.18	101.16
Spacecraft separation			
Time from lift-off, sec	360.00	366.72	366.72
Geodetic latitude, deg north	26.71	26.64	26.65
Longitude, deg west	70.56	70.24	70.24
Altitude, ft	526 845	525 883	525 857
Altitude, n. mi.	86.7	86.5	86.5
Range, n. mi.	543.4	562.8	562.8
Space-fixed velocity, ft/sec	25 728	25 712	25 713
Space-fixed flight-path angle, deg	0.0	0.05	0.0
Space-fixed heading angle, deg east of north	101.76	101.91	101.94

[a]For preflight-calculated nominal trajectories.

[redacted]

[redacted]

TABLE 4.3-I.- PLANNED AND ACTUAL GEMINI SPACE

VEHICLE AND SPACECRAFT TRAJECTORY PARAMETERS - Concluded

Condition	Planned[a]	Actual	
		Preliminary	Final
Spacecraft insertion			
Time from lift-off, sec	387.00	439.8	439.8
Geodetic latitude, deg north	26.31	25.50	25.50
Longitude, deg west	69.45	65.09	65.09
Altitude, ft	526 539	526 809	525 630
Altitude, n. mi.	86.6	86.7	86.5
Range, n. mi.	649.8	848.9	848.9
Space-fixed velocity, ft/sec	25 748	25 740	25 742
Space-fixed flight-path angle, deg	0.01	0.06	0.07
Space-fixed heading angle, deg east of north	102.67	104.30	104.31
Maximum conditions			
Altitude, statute mi.	460.0	187.2	187.2
Altitude, n. mi.	400.0	162.7	162.7
Space-fixed velocity, ft/sec	25 777	25 740	25 740
Earth-fixed velocity, ft/sec	24 411	24 373	24 373
Exit acceleration, g	7.2	7.1	7.1
Exit dynamic pressure, lb/ft^2	743	730	730
Reentry deceleration, g (tracking data)	6.3	6.4	6.4
Reentry deceleration, g (telemetry data)	Not applicable	6.2	6.2
Reentry dynamic pressure, lb/ft^2	416	425	425
Landing point			
Latitude, north	24 deg 35 min	[b]24 deg 37 min	[c]24 deg 35 min
Longitude, west	70 deg 00 min	[b]69 deg 56 min	[c]69 deg 57 min

[a]For preflight-calculated nominal trajectories.

[b]Landing point based on determinations made onboard the recovery ship.

[c]Landing point based on best estimated trajectory from radar tracking data.

[redacted]

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

Maneuver	Condition	Before maneuver		After maneuver	
		Planned[a]	Actual[b]	Planned[a]	Actual[b]
Phase adjust (N_{C1})	Apogee, n. mi.	149.8	146.1	149.8	145.9
	Perigee, n. mi.	86.7	86.8	123.9	123.4
	Inclination, deg	28.89	28.87	28.90	28.90
	Period, min	88.94	88.87	89.64	89.55
Plane change (N_{PC})	Apogee, n. mi.	146.1	145.9	146.1	145.9
	Perigee, n. mi.	123.9	123.4	123.9	123.4
	Inclination, deg	28.90	28.90	28.90	28.88
	Period, min	89.57	89.55	89.57	89.55
Corrective combination (N_{CC})	Apogee, n. mi.	146.1	145.9	150.0	151.7
	Perigee, n. mi.	123.9	123.4	124.7	124.2
	Inclination, deg	28.90	28.88	28.89	28.89
	Period, min	89.57	89.55	89.66	89.68
Coelliptic (N_{SR})	Apogee, n. mi.	150.0	151.7	150.2	151.7
	Perigee, n. mi.	124.7	124.2	148.8	146.8
	Inclination, deg	28.89	28.89	28.89	28.88
	Period, min	89.66	89.68	90.12	90.11
Terminal phase initiate (TPI)	Apogee, n. mi.	150.2	151.7	162.4	162.4
	Perigee, n. mi.	148.8	146.8	150.3	151.7
	Inclination, deg	28.89	28.88	28.89	28.87
	Period, min	90.12	90.11	90.38	90.41
Terminal phase finalize (TPF)	Apogee, n. mi.	162.4	162.4	164.2	162.7
	Perigee, n. mi.	150.3	151.7	158.8	156.4
	Inclination, deg	28.89	28.87	28.87	28.87
	Period, min	90.38	90.41	90.58	90.50
Eclipse phasing no. 1 (SPS) (docked)	Apogee, n. mi.	163.9	161.4	155.3	151.5
	Perigee, n. mi.	155.6	153.0	139.6	136.8
	Inclination, deg	28.86	28.87	28.87	28.88
	Period, min	90.52	90.41	90.04	89.92

[a]Planned elements are calculated in real time by the Real Time Computer Complex except first apogee and perigee at insertion. Apogee and perigee altitudes are measured over a spherical earth with Launch Complex 19 earth radius. Inclination osculates ±0.04 of a degree.

[b]Actual elements are calculated over an oblate earth referenced to the Fischer ellipsoid earth model of 1960. Inclination osculates ±0.04 of a degree.

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

Maneuver	Condition	Before maneuver		After maneuver	
		Planned[a]	Actual[b]	Planned[a]	Actual[b]
Eclipse phasing no. 2 (SPS) (docked)	Apogee, n. mi.	155.3	151.3	162.2	160.3
	Perigee, n. mi.	139.6	136.5	139.1	136.6
	Inclination, deg	28.87	28.88	28.88	28.88
	Period, min	90.04	89.92	90.17	90.08
Separation	Apogee, n. mi.	162.2	159.3	162.1	160.0
	Perigee, n. mi.	139.1	137.3	142.8	140.3
	Inclination, deg	28.88	28.88	28.88	28.88
	Period, min	90.17	90.07	90.23	90.15
Phasing	Apogee, n. mi.	162.1	159.5	159.4	156.0
	Perigee, n. mi.	142.8	140.3	142.9	140.8
	Inclination, deg	28.88	28.88	28.88	28.89
	Period, min	90.23	90.14	90.19	90.08

[a]Planned elements are calculated in real time by the Real Time Computer Complex except first apogee and perigee at insertion. Apogee and perigee altitudes are measured over a spherical earth with Launch Complex 19 earth radius. Inclination osculates ±0.04 of a degree.

[b]Actual elements are calculated over an oblate earth referenced to the Fischer ellipsoid earth model of 1960. Inclination osculates ±0.04 of a degree.

TABLE 4.3-III.- SPACECRAFT ORBITAL ELEMENTS

Revolution	Condition	Real-time planned[a]	Actual[b]
1 (Insertion)	Apogee, n. mi.	149.8	146.1
	Perigee, n. mi.	86.7	86.8
	Inclination, deg	28.89	28.87
	Period, min	88.94	88.87
3 (Before rendezvous)	Apogee, n. mi.	150.2	151.7
	Perigee, n. mi.	148.8	146.8
	Inclination, deg	28.89	28.88
	Period, min	90.12	90.11
3 (After rendezvous)	Apogee, n. mi.	164.2	162.7
	Perigee, n. mi.	158.8	156.4
	Inclination, deg	28.87	28.87
	Period, min	90.58	90.50
24	Apogee, n. mi.	162.2	160.3
	Perigee, n. mi.	139.1	136.5
	Inclination, deg	28.88	28.89
	Period, min	90.17	90.08
24	Apogee, n. mi.	162.1	159.4
	Perigee, n. mi.	139.5	137.0
	Inclination, deg	28.88	28.89
	Period, min	90.17	90.07
36	Apogee, n. mi.	162.1	159.5
	Perigee, n. mi.	142.8	140.3
	Inclination, deg	28.88	28.88
	Period, min	90.23	90.14

[a]Planned elements are calculated in real time by the Real Time Computer Complex except insertion, which is preflight nominal. Apogee and perigee altitudes are measured over a spherical earth with Launch Complex 19 earth radius. Inclination osculates ±0.04 of a degree.

[b]Actual elements are calculated over an oblate earth referenced to the Fischer ellipsoid earth model of 1960. Inclination osculates ±0.04 of a degree.

TABLE 4.3-III.- SPACECRAFT ORBITAL ELEMENTS - Concluded

Revolution	Condition	Real-time planned[a]	Actual[b]
48	Apogee, n. mi.	157.7	155.5
	Perigee, n. mi.	142.9	140.5
	Inclination, deg	28.86	28.87
	Period, min	90.15	90.06
59 (Retrofire)	Apogee, n. mi.	156.8	155.0
	Perigee, n. mi.	142.7	140.8
	Inclination, deg	28.88	28.87
	Period, min	90.13	90.06

[a]Planned elements are calculated in real time by the Real Time Computer Complex except insertion, which is preflight nominal. Apogee and perigee altitudes are measured over a spherical earth with Launch Complex 19 earth radius. Inclination osculates ±0.04 of a degree.

[b]Actual elements are calculated over an oblate earth referenced to the Fischer ellipsoid earth model of 1960. Inclination osculated ±0.04 of a degree.

TABLE 4.3-IV.- SPACECRAFT AND GATV MANEUVERS

Maneuver	Planned[a]	Ground-commanded[b]	Actual
Phase adjust			
Initiate time, g.e.t. . .	0:49:40	0:49:40	0:49:40
ΔV, ft/sec	66.6	66.6	66.7
Pitch, deg	0.0	0.0	0.2
Yaw, deg	0.0	0.0	-0.2
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .	+66.6, 0.0, 0.0	+66.6, 0.0, 0.0	+66.7, -0.3, +0.2
Δt, sec	88.0	88.0	[d]88.6
Plane change			
Initiate time, g.e.t. . .	1:11:14	1:11:14	1:14:22
ΔV, ft/sec	7.4	7.4	8.5
Pitch, deg	0.0	0.0	-1.8
Yaw, deg	90.0	90.0	91.1
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .	0.0, 0.0, -7.4	0.0, 0.0, -7.4	-0.3, +0.3, -8.4
Δt, sec	10.0	10.0	[d]36.0
Corrective combination			
Initiate time, g.e.t. . .	1:47:51	1:47:52	1:47:52
ΔV, ft/sec	9.3	7.6	8.42
Pitch, deg	33.4	27.3	27.3
Yaw, deg	-2.4	9.6	7.7
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .	+7.7, -5.1, +0.3	+6.7, -3.5, -1.1	+7.4, -3.9, -1.0
Δt, sec	12.0	10.0	[d]17.0

[a]Planned maneuvers were obtained in real time from the RTCC.

[b]Ground-commanded maneuvers were refinements of planned values and represent the latest information passed to the crew.

[c]ΔV_X, ΔV_Y, and ΔV_Z are the velocity vector components in computer coordinates. V_X is positive in the direction of motion, V_Y is positive towards the center of the earth, and V_Z is positive to the left of the orbit path (North).

[d]The time interval (Δt) indicated here is the amount of time that was taken to perform the maneuver, which includes the zeroing of the IVI.

TABLE 4.3-IV.- SPACECRAFT AND GATV MANEUVERS - Continued

Maneuver	Planned[a]	Ground-commanded[b]	Actual
Coelliptic			
Initiate time, g.e.t.	2:23:43	2:22:54	2:22:55
ΔV, ft/sec	50.5	49.9	49.9
Pitch, deg	12.4	4.0	7.4
Yaw, deg	-0.9	0.8	0.2
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec	+49.3, -10.8, +0.8	+49.8, -3.5, -0.7	+49.5, -6.4, -0.2
Δt, sec	68.0	66.0	[d]60.0
Terminal phase initiate			
Initiate time, g.e.t.	3:00:44	3:05:51	3:05:47
ΔV, ft/sec	21.2	23.2	21.8
Pitch, deg	25.6	34.7	27.5
Yaw, deg	-2.2	+7.7	-1.4
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec	+19.1, -9.2, +0.7	+18.9, -13.2, -2.6	+19.4, -10.1, +0.5
ΔV_{body}[e], ft/sec	21.2 fwd, 0.5 dn, 0.8 lt	22.8 fwd, 3.2 up, 2.7 rt	21.8 fwd, 0.1 dn, 0.6 lt
Δt, sec	29.0	30.0	[d]29.0

[a]Planned maneuvers were obtained in real time from the RTCC.

[b]Ground-commanded maneuvers were refinements of planned values and represent the latest information passed to the crew.

[c]ΔV_X, ΔV_Y, and ΔV_Z are the velocity vector components in computer coordinates. V_X is positive in the direction of motion, V_Y is positive towards the center of the earth, and V_Z is positive to the left of the orbit path (North).

[d]The time interval (Δt) indicated here is the amount of time that was taken to perform the maneuver, which includes the zeroing of the IVI.

[e]Delta velocity applied along the body axis, with the spacecraft boresighted on the target; fwd-aft is along line-of-sight, up-dn is normal to the line-of-sight, lt-rt is perpendicular to the line-of-sight plane.

TABLE 4.3-IV.- SPACECRAFT AND GATV MANEUVERS - Continued

Maneuver	Planned[a]	Ground-commanded[b]	Actual
First midcourse correction			
Initiate time, g.e.t. . .	Not computed	Not sent	3:11:14
ΔV, ft/sec	Not computed	Not sent	1.7
Pitch, deg			[f]36.7
Yaw, deg			[f]1.2
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .			-0.1, -1.7, +0.1
ΔV_{body}[e], ft/sec			1.0 fwd, 1.4 up, 0.1 lt
Δt, sec			[d]6.0
Second midcourse correction			
Initiate time, g.e.t. . .	Not computed	Not sent	3:17:07
ΔV, ft/sec	Not computed	Not sent	3.1
Pitch, deg			[f]50.6
Yaw, deg			[f]2.7
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .			-1.4, -2.8, -0.1
ΔV_{body}[e], ft/sec			1.2 fwd, 2.9 up, 0.1 rt
Δt, sec			[d]8.0

[a]Planned maneuvers were obtained in real time from the RTCC.

[b]Ground-commanded maneuvers were refinements of planned values and represent the latest information passed to the crew.

[c]ΔV_X, ΔV_Y, and ΔV_Z are the velocity vector components in computer coordinates. V_X is positive in the direction of motion, V_Y is positive towards the center of the earth, and V_Z is positive to the left of the orbit path (North).

[d]The time interval (Δt) indicated here is the amount of time that was taken to perform the maneuver, which includes the zeroing of the IVI.

[e]Delta velocity applied along the body axis, with the spacecraft boresighted on the target; fwd-aft is along line-of-sight, up-dn is normal to the line-of-sight, lt-rt is perpendicular to the line-of-sight plane.

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

TABLE 4.3-IV.- SPACECRAFT AND GATV MANEUVERS - Continued

Maneuver	Planned[a]	Ground-commanded[b]	Actual
Third midcourse correction			
Initiate time, g.e.t. . .	Not computed	Not sent	3:23:46
ΔV, ft/sec	Not computed	Not sent	1.4
Pitch, deg			[f]73.8
Yaw, deg			[f]5.8
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .			0.3, 0.5, 1.2
ΔV_{body}[e], ft/sec			0.4 aft, 0.3 dn, 1.2 lt
Δt, sec			[d]4.0
Fourth midcourse correction			
Initiate time, g.e.t. . .	Not computed	Not sent	3:29:05
ΔV, ft/sec	Not computed	Not sent	4.9
Pitch, deg			[f]102.4
Yaw, deg			[f]10.9
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .			+1.1, +4.8, -0.2
ΔV_{body}[e], ft/sec			4.8 aft, 1.1 dn, 0.2 rt
Δt, sec			[d]6.0

[a]Planned maneuvers were obtained in real time from the RTCC.

[b]Ground-commanded maneuvers were refinements of planned values and represent the latest information passed to the crew.

[c]ΔV_X, ΔV_Y, and ΔV_Z are the velocity vector components in computer coordinates. V_X is positive in the direction of motion, V_Y is positive towards the center of the earth, and V_Z is positive to the left of the orbit path (North).

[d]The time interval (Δt) indicated here is the amount of time that was taken to perform the maneuver, which includes the zeroing of the IVI.

[e]Delta velocity applied along the body axis, with the spacecraft boresighted on the target; fwd-aft is along the line-of-sight, up-dn is normal to the line-of-sight, lt-rt is perpendicular to the line-of-sight plane.

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

TABLE 4.3-IV.- SPACECRAFT AND GATV MANEUVERS - Continued

Maneuver	Planned[a]	Ground-commanded[b]	Actual
Terminal phase finalize (TPF) (braking)			
Initiate time, g.e.t. . .	Not computed	[g]3:38:12	3:32:36
ΔV, ft/sec	Not computed	28.4	[h]19.7
Pitch, deg		55.6	
Yaw, deg		-176.3	
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .		+16.1, +23.5, -1.0	
ΔV_{body}[e], ft/sec		28.4 aft, 0.3 up, 0.1 lt	
Δt, sec		50.0	[i]780
Eclipse phasing no. 1 (SPS)			
Initiate time, g.e.t. . .	Not computed	7:05:06	7:05:06
ΔV, ft/sec	Not computed	43.0	43.8
Pitch, deg		0	-2.5
Yaw, deg		180	-176.6
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .		-43, 0, 0	-43.7, +1.9, +2.6
Δt, sec		51	51

[a]Planned maneuvers were obtained in real time from the RTCC.

[b]Ground-commanded maneuvers were refinements of planned values and represent the latest information passed to the crew.

[c]ΔV_X, ΔV_Y, and ΔV_Z are the velocity vector components in computer coordinates. V_X is positive in the direction of motion, V_Y is positive towards the center of the earth, and V_Z is positive to the left of the orbit path (North).

[e]Delta velocity applied along the body axis, with the spacecraft boresighted on the target; fwd-aft is along line-of-sight, up-dn is normal to the line-of-sight, lt-rt is perpendicular to the line-of-sight plane.

[g]This is the theoretical TPF maneuver computed on the ground but was not sent to the crew.

[h]This is the resultant ΔV applied during braking; however, the total ΔV expended during the semi-optical approach was about 30 ft/sec.

[i]Braking lasted intermittently for about 13 minutes.

TABLE 4.3-IV.- SPACECRAFT AND GATV MANEUVERS - Concluded

Manuever	Planned[a]	Ground-commanded[b]	Actual
Eclipse phasing no. 2 (SPS)[j]			
Initiate time, g.e.t. . .	Not computed	15:16:18	15:16:18
ΔV, ft/sec	Not computed	15.0	16.7
Pitch, deg		0	3.9
Yaw, deg		0	2.0
ΔV_X, ΔV_Y, ΔV_Z		+15, 0, 0	+16.6, -1.1, -1.0
Δt, sec		18	18
Separation			
Initiate time, g.e.t. . .	Not computed	52:14:27	52:14:27
ΔV, ft/sec	Not computed	6.0	7.1
Pitch, deg		0	-1.3
Yaw, deg		0	1.0
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .		6, 0, 0	+7.1, +0.2, -0.1
Δt, sec		8.2	8.7
Phasing			
Initiate time, g.e.t. . .	Not computed	61:47:47	61:47:48
ΔV, ft/sec	Not computed	5.5	5.2
Pitch, deg		0	2.7
Yaw, deg		0	7.0
ΔV_X, ΔV_Y, ΔV_Z[c], ft/sec . .		-5.5, 0, 0	-5.1, +0.2, +0.6
Δt, sec		9	[d]64

[a]Planned maneuvers were obtained in real time from the RTCC.

[b]Ground-commanded maneuvers were refinements of planned values and represent the latest information passed to the crew.

[c]ΔV_X, ΔV_Y, and ΔV_Z are the velocity vector components in computer coordinates. V_X is positive in the direction of motion, V_Y is positive towards the center of the earth, and V_Z is positive to the left of the orbit path (North).

[d]The time interval (Δt) indicated here is the amount of time that was taken to perform the maneuver, which includes the zeroing of the IVI.

[j]This maneuver was determined from orbital radar tracking data because the platform was not on and the accelerometer data were not available.

TABLE 4.3-V.- PLANNED AND ACTUAL TLV AND GATV TRAJECTORY PARAMETERS

Condition	Planned[a]	Actual	
		Preliminary	Final
BECO			
Time from lift-off, sec	131.50	131.11	131.11
Geodetic latitude, deg North	28.58	28.57	28.57
Longitude, deg West	79.73	79.74	79.74
Altitude, ft	197 517	195 195	194 716
Altitude, n. mi.	32.5	32.2	32.0
Range, n. mi.	43.2	42.9	42.9
Space-fixed velocity, ft/sec	9842	9803	9805
Space-fixed flight-path angle, deg	21.21	21.30	21.21
Space-fixed heading angle, deg east of North	84.50	84.55	84.65
SECO			
Time from lift-off, sec	280.73	280.12	280.12
Geodetic latitude, deg North	28.93	28.91	28.91
Longitude, deg West	74.64	74.64	74.62
Altitude, ft	656 258	655 811	654 501
Altitude, n. mi.	108.0	108.1	107.7
Range, n. mi.	313.8	313.5	313.6
Space-fixed velocity, ft/sec	17 637	17 636	17 630
Space-fixed flight-path angle, deg	10.21	10.26	10.23
Space-fixed heading angle, deg east of North	87.61	87.65	87.60
VECO			
Time from lift-off, sec	299.02	298.06	298.06
Geodetic latitude, deg North	28.96	28.94	28.94
Longitude, deg West	73.73	73.85	73.75

[a]For preflight-calculated nominal trajectories.

TABLE 4.3-V.- PLANNED AND ACTUAL TLV AND GATV TRAJECTORY PARAMETERS - Continued

Condition	Planned[a]	Actual	
		Preliminary	Final
VECO - Concluded			
Altitude, ft	710 826	704 200	708 556
Altitude, n. mi.	117.0	115.9	116.6
Range, n. mi.	360.4	353.7	359.2
Space-fixed velocity, ft/sec . .	17 568	17 571	17 571
Space-fixed flight-path angle, deg	9.29	9.35	9.35
Space-fixed heading angle, deg east of North	88.07	88.06	88.06
PPS ignition			
Time from lift-off, sec	372.10	372.51	372.51
Geodetic latitude, deg North . .	29.02	29.00	29.00
Longitude, deg West	70.24	70.26	70.29
Altitude, ft	874 908	875 900	875 900
Altitude, n. mi.	144.0	144.1	144.1
Range, n. mi.	544.2	541.6	541.6
Space-fixed velocity, ft/sec . .	17 288	17 292	17 292
Space-fixed flight-path angle, deg	5.50	5.60	5.60
Space-fixed heading angle, deg east of North	89.92	89.62	89.62
Insertion			
Time from lift-off, sec	554.06	556.05	556.05
Geodetic latitude, deg North . .	28.55	28.52	28.52
Longitude, deg West	59.79	59.63	59.63

[a]For preflight-calculated nominal trajectories.

TABLE 4.3-V.- PLANNED AND ACTUAL TLV AND GATV TRAJECTORY PARAMETERS - Concluded

Condition	Planned[a]	Actual	
		Preliminary	Final
Insertion - Concluded			
Altitude, ft	979 995	981 195	981 195
Altitude, n. mi.	161.3	161.5	161.5
Range, n. mi.	1095.6	1106.5	1106.5
Space-fixed velocity, ft/sec	25 367	25 365	25 365
Space-fixed flight-path angle, deg	0.00	0.04	0.04
Space-fixed heading angle, deg east of North	95.51	95.54	95.54
Maximum conditions			
Altitude, statute mi.	460	187.1	187.1
Altitude, n. mi.	400	162.7	162.7
Space-fixed velocity, ft/sec	25 777	25 740	25 740
Earth-fixed velocity, ft/sec	24 411	24 373	24 373
Exit acceleration, g	6.27	6.34	6.34
Exit dynamic pressure, lb/ft^2	944	937	937

[a]For preflight-calculated nominal trajectories.

TABLE 4.3-VI.- PLANNED AND ACTUAL TLV CUTOFF

AND GATV INSERTION CONDITIONS

Condition	Planned[a]	Actual	Difference
VECO targeting parameters			
Semimajor axis, n. mi.	2330.7	2330.0	-0.7
Eccentricity	0.5436	0.5438	+0.0002
Inclination, deg	28.86	28.86	0.00
Inertial ascending node, deg	129.30	129.20	-0.10
VECO osculating elements			
Apogee altitude, n. mi.	158.07	157.50	-0.57
Perigee altitude, n. mi.	-2376.97	-2377.77	-0.80
Period, min	47.07	47.05	-0.02
Inclination, deg	28.86	28.86	0.00
True anomaly, deg	172.01	171.97	-0.04
Argument of perigee, deg	-85.52	-85.49	+0.03
Insertion osculating elements			
Semimajor axis, n. mi.	3603.2	3603.0	-0.2
Eccentricity	0.0007	0.0009	+0.0002
Inclination, deg	28.88	28.86	-0.02
Inertial ascending node, deg	128.84	128.94	+0.10
Apogee altitude, n. mi.	161.12	163.6	+2.48
Perigee altitude, n. mi.	159.14	159.0	-0.14
Period, min	90.52	90.56	+0.04
True anomaly, deg[b]	0.00	50.94	+50.94
Argument of perigee, deg[b]	100.07	49.19	-50.88

[a]For preflight-calculated nominal trajectories.

[b]These elements are not well defined for near-circular orbits.

TABLE 4.3-VII.- GATV ORBITAL ELEMENTS

Revolution	Condition	Planned[a]	Actual[b]
1 (Insertion)	Apogee, n. mi.	161.1	162.7
	Perigee, n. mi.	159.1	156.4
	Inclination, deg	28.88	28.87
	Period, min	90.52	90.50
12	Apogee, n. mi.	162.2	160.3
	Perigee, n. mi.	139.1	136.5
	Inclination, deg	28.88	28.87
	Period, min	90.17	90.08
24	Apogee, n. mi.	162.1	159.4
	Perigee, n. mi.	139.5	137.0
	Inclination, deg	28.88	28.89
	Period, min	90.17	90.07
36	Apogee, n. mi.	161.6	159.2
	Perigee, n. mi.	139.9	137.4
	Inclination, deg	28.87	28.88
	Period, min	90.17	90.08

[a]Planned elements are calculated in real time by the Real Time Computer Complex except insertion, which is preflight nominal. Apogee and perigee altitudes are measured over a spherical earth with Launch Complex 19 earth radius. Inclination osculated ±0.04 of a degree.

[b]Actual elements are calculated over an oblate earth referenced to the Fischer ellipsoid earth model of 1960. Inclination osculated ±0.04 of a degree.

TABLE 4.3-VII.- GATV ORBITAL ELEMENTS - Concluded

Revolution	Condition	Planned[a]	Actual[b]
48	Apogee, n. mi.	160.9	158.6
	Perigee, n. mi.	139.8	137.6
	Inclination, deg	28.88	28.88
	Period, min	90.15	90.07
59 (Spacecraft retrofire)	Apogee, n. mi.	160.1	158.5
	Perigee, n. mi.	139.8	137.9
	Inclination, deg	28.86	28.89
	Period, min	90.14	90.07
62 (PPS firing attempt)	Apogee, n. mi.	Not applicable	[c]158.1
	Perigee, n. mi.		137.8
	Inclination, deg		28.89
	Period, min		90.02

[a]Planned elements are calculated in real time by the Real Time Computer Complex except insertion, which is preflight nominal. Apogee and perigee altitudes are measured over a spherical earth with Launch Complex 19 earth radius. Inclination osculates ±0.04 of a degree.

[b]Actual elements are calculated over an oblate earth referenced to the Fischer ellipsoid earth model of 1960. Inclination osculates ±0.04 of a degree.

[c]Due to GATV attitude control problems, the following tolerances apply to these elements: apogee and perigee altitudes, ±1.8 n. mi.; period, ±0.03 min.

TABLE 4.3-VIII.- GATV MANEUVERS

Maneuver	Ground-commanded[a]	Actual
Eclipse phasing no. 1 (SPS)		
Initiate time, g.e.t.	7:05:06	7:05:06
ΔV, ft/sec	43.0	43.8
Pitch, deg	0	-2.5
Yaw, deg	180	-176.6
ΔV_X, ΔV_Y, ΔV_Z[b], ft/sec	-43, 0, 0	-43.7, +1.9, +2.6
Δt, sec	51	51
Eclipse phasing no. 2 (SPS)		
Initiate time, g.e.t.	15:16:18	[c]15:16:18
ΔV, ft/sec	15.0	16.7
Pitch, deg	0	3.9
Yaw, deg	0	2.0
ΔV_X, ΔV_Y, ΔV_Z	+15, 0, 0	+16.6, -1.1, -1.0
Δt, sec	18	18
PPS firing attempt		
Initiate time, g.e.t.	94:50:27	94:50:27
ΔV, ft/sec	1550	[d]3.3
Pitch, deg	0	Unknown
Yaw, deg	0	Unknown
Δt, sec	41	[d]22

[a]Ground-commanded maneuvers were refinements of planned values and represent the latest information passed to the crew.

[b]ΔV_X, ΔV_Y, ΔV_Z are the velocity vector components in computer coordinates. V_X is positive in the direction of motion, V_Y is positive towards the center of the earth, and V_Z is positive to the left of the orbit path (North).

[c]This maneuver was determined from orbital radar tracking data because the platform was not on and the accelerometer data were not available.

[d]These values were determined emperically based on the SPS ullage firing prior to PPS main engine ignition.

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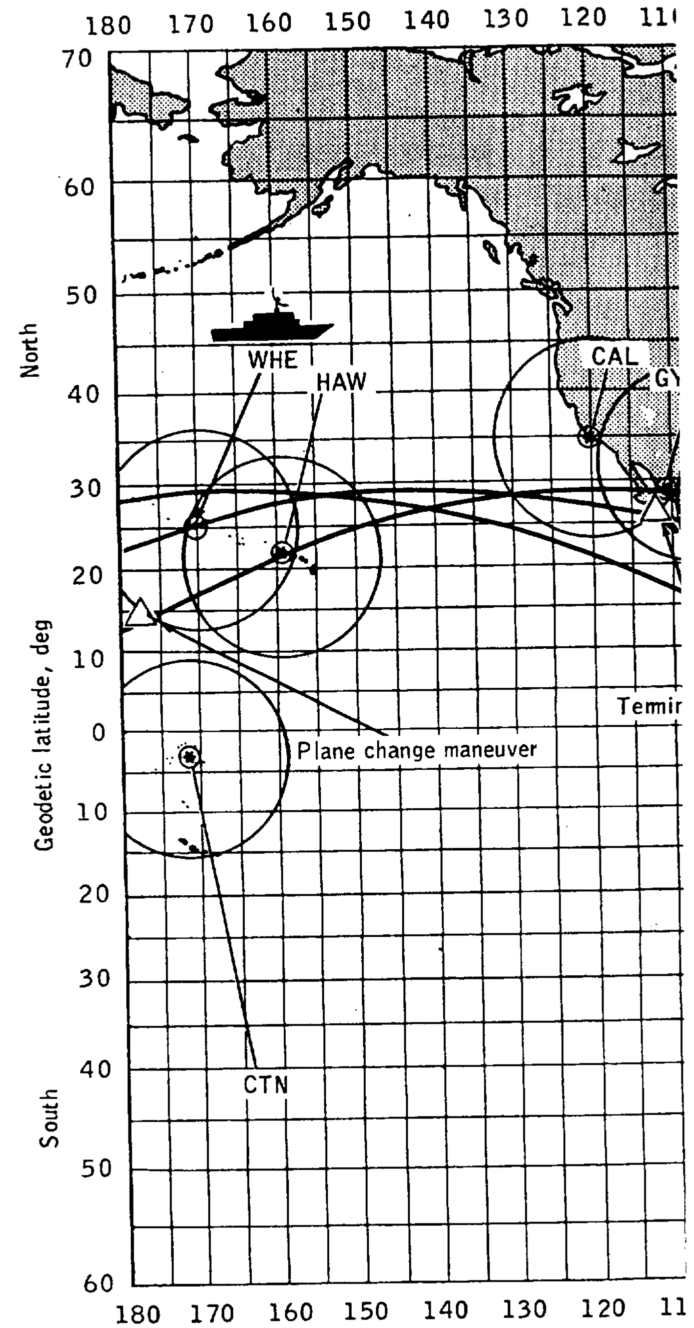
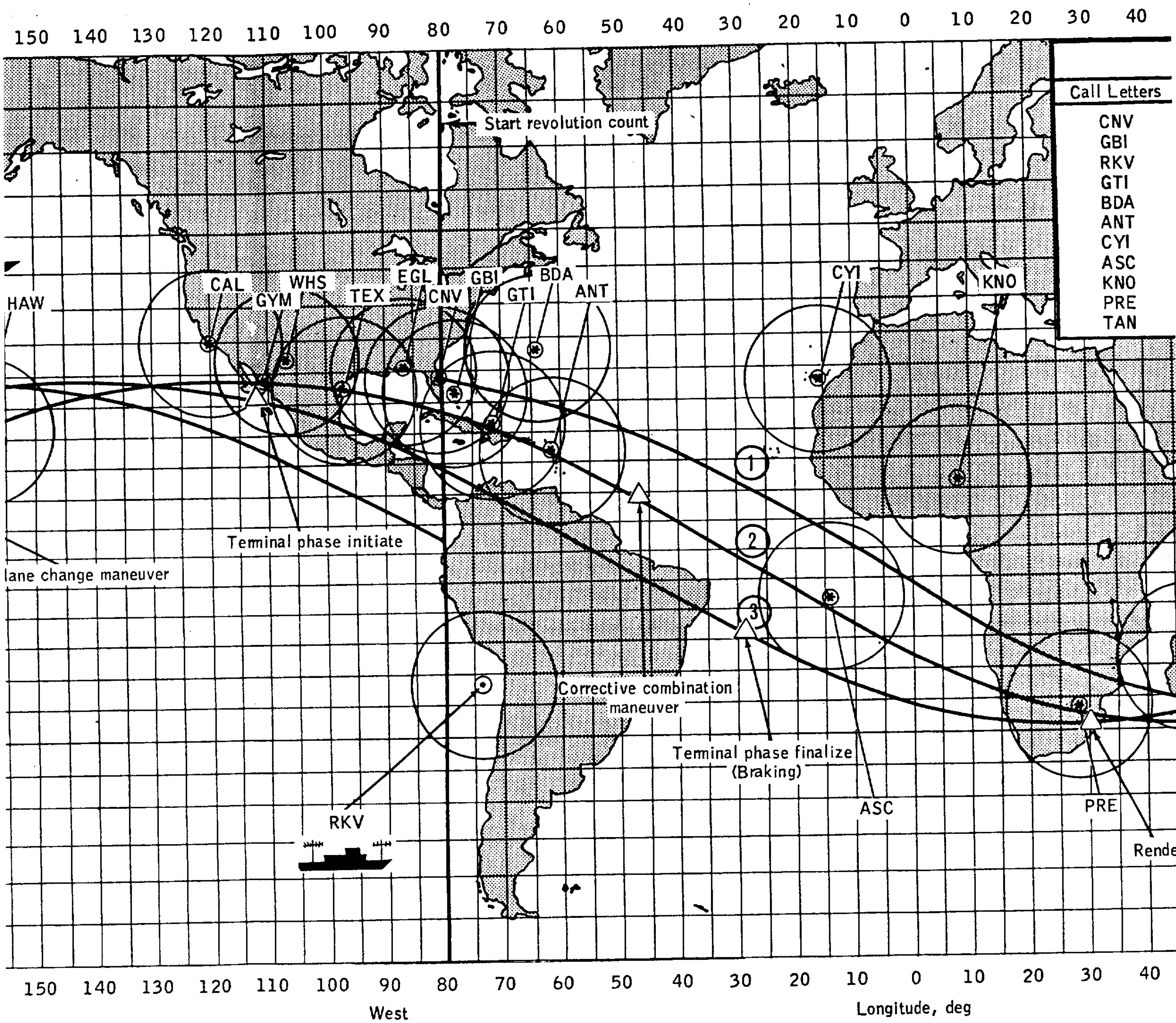
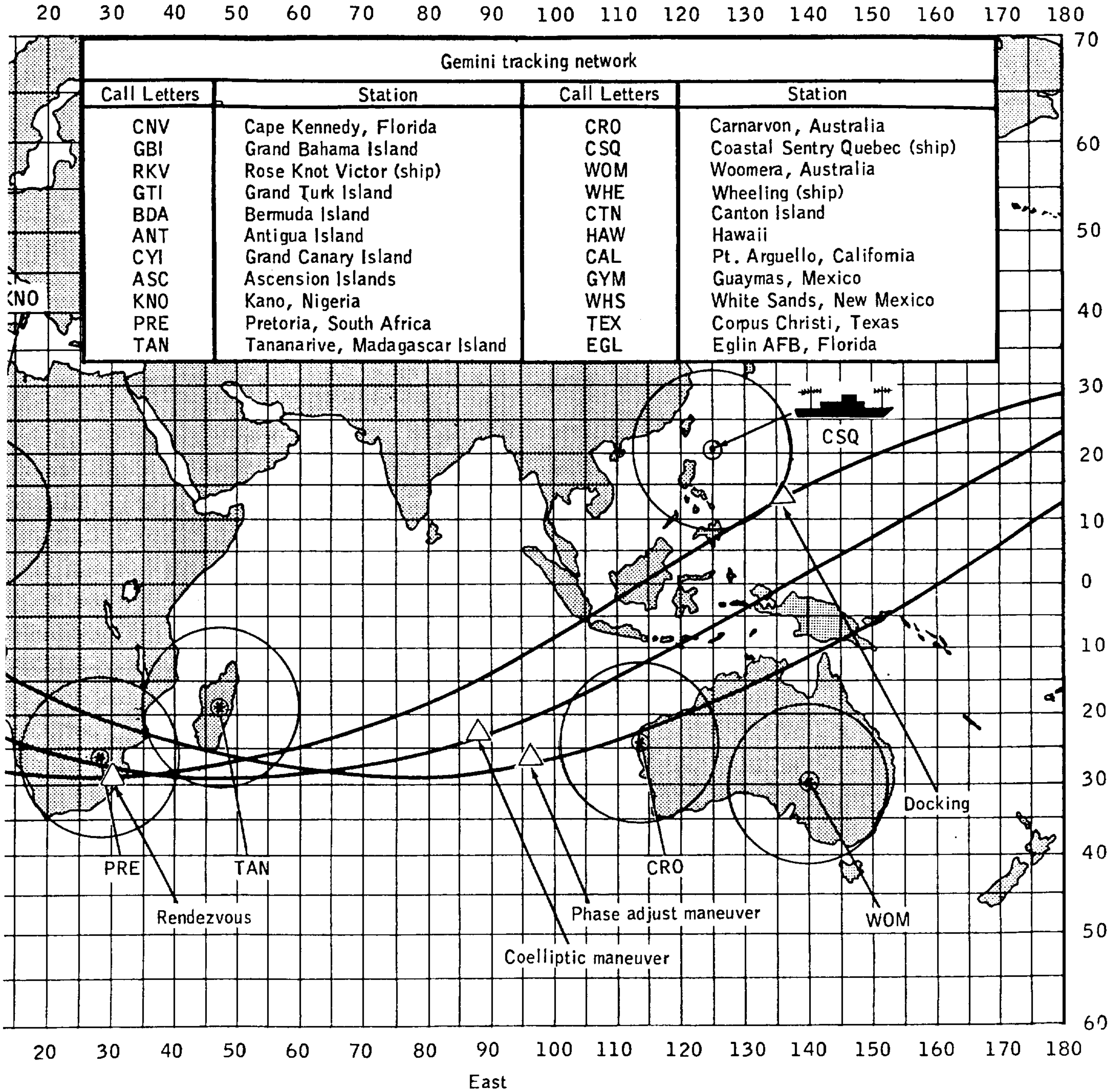




150 140 130 120 110 100 90 80 70 60 50 40 30 20 10 0 10 20 30 40
Call Letters
CNV
GBI
RKV
GTI
BDA
ANT
CYI
ASC
KNO
PRE
TAN
Start revolution count
HAW
CAL
GYM
WHS
TEX
EGL
CNV
GBI
GTI
BDA
ANT
CYI
KNO
1
2
3
Terminal phase initiate
lane change maneuver
Corrective combination maneuver
Terminal phase finalize (Braking)
RKV
ASC
PRE
Rende
West
Longitude, deg

2



(a) Revolutions 1 through 3.

Figure 4.3-1.- Ground track for the Gemini XII orbital mission.

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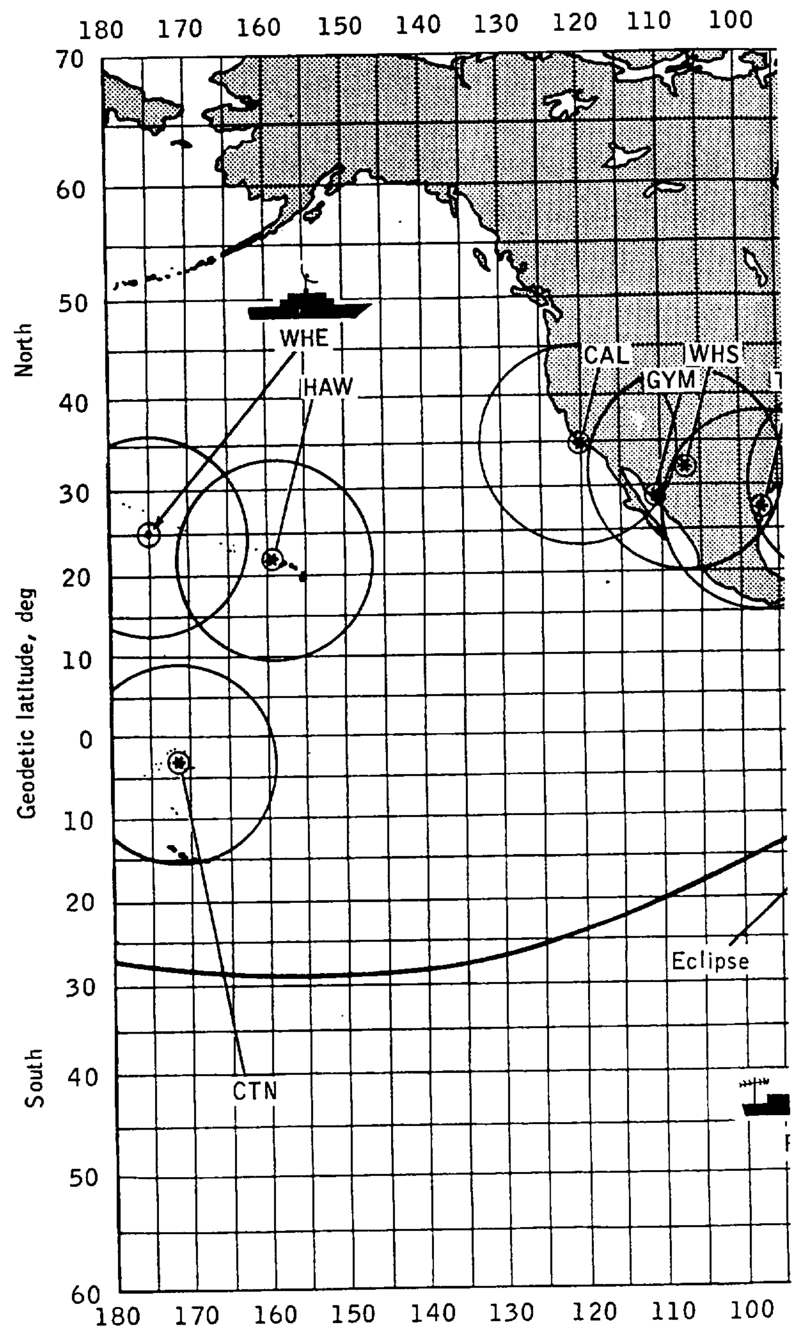
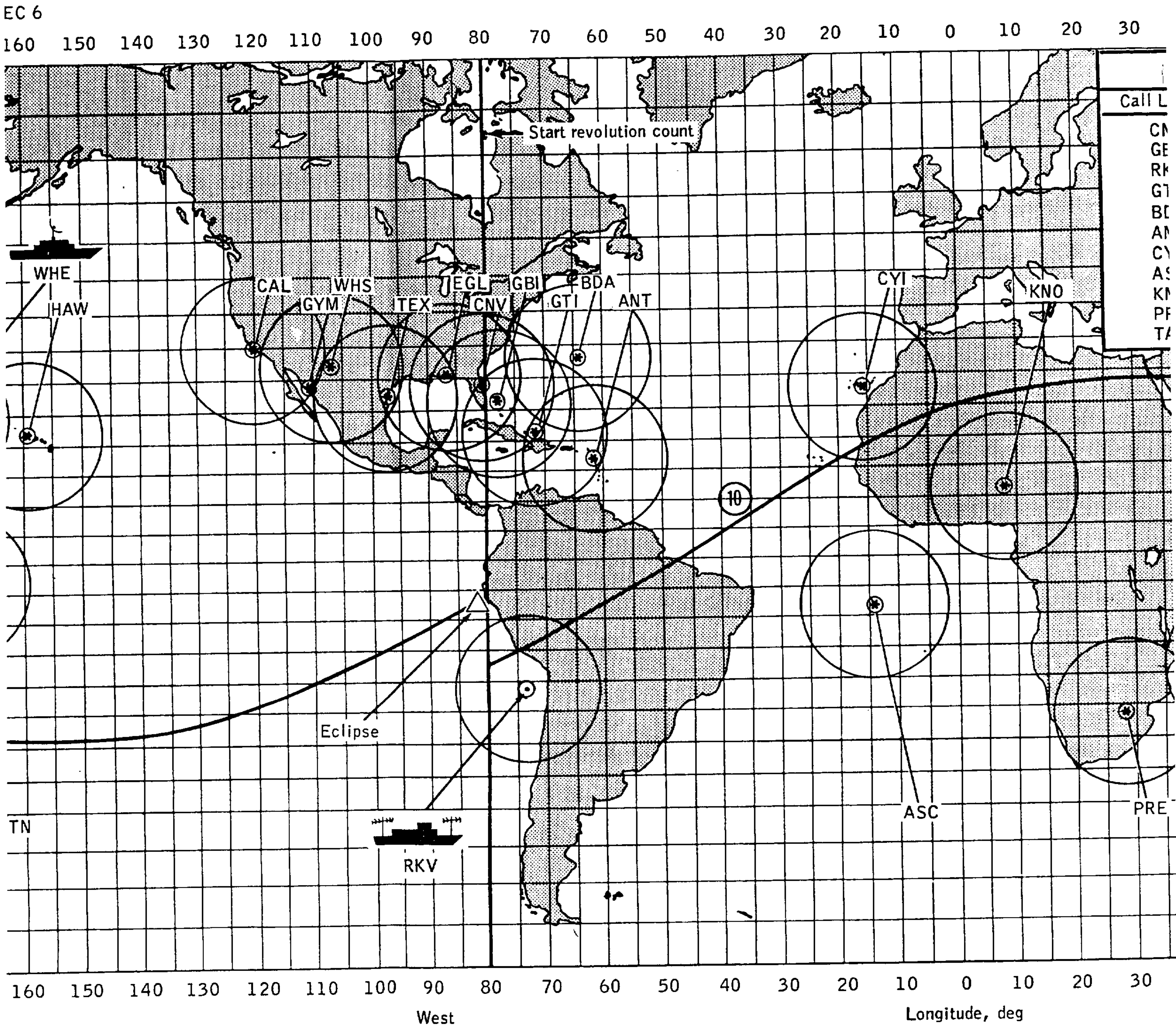


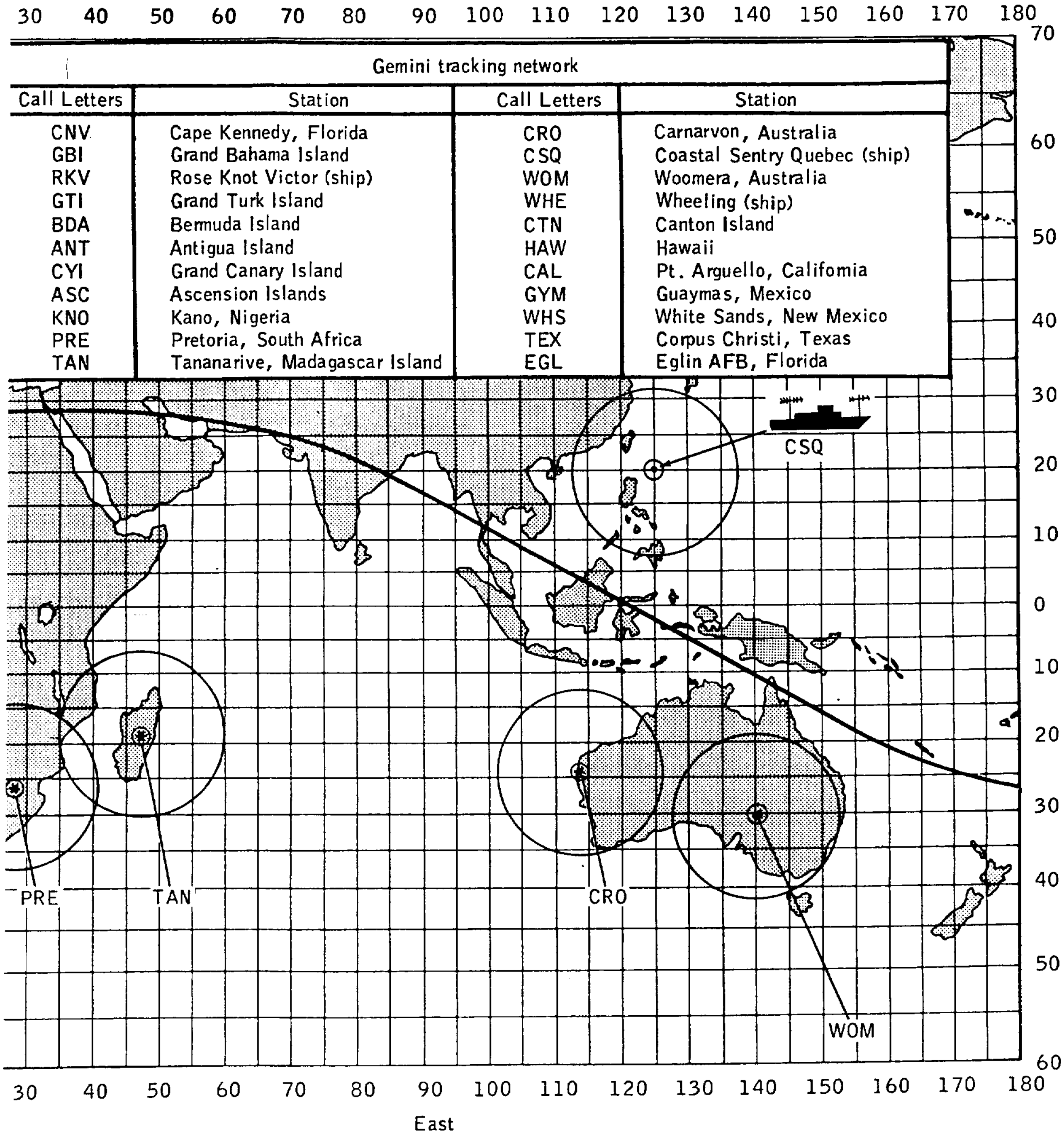


EC 6
Start revolution count
Call L
WHE
HAW
CAL
GYM
WHS
TEX
EGL
CNV
GBI
GTI
BDA
ANT
CYI
KNO
10
Eclipse
RKV
ASC
PRE
TN
160 150 140 130 120 110 100 90 80 70 60 50 40 30 20 10 0 10 20 30
West
Longitude, deg

2

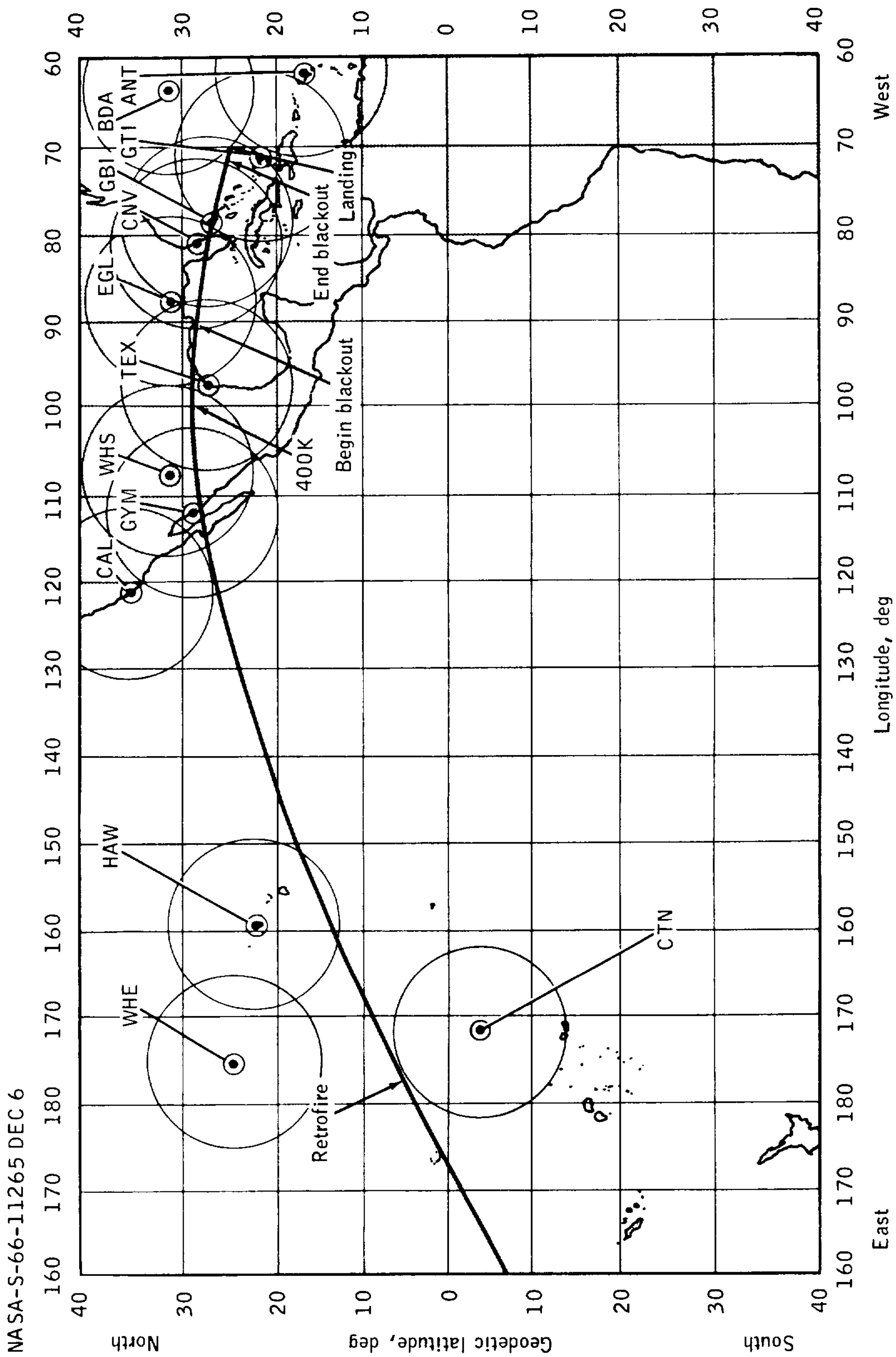
Gemini tracking network

Call Letters	Station	Call Letters	Station
CNV	Cape Kennedy, Florida	CRO	Carnarvon, Australia
GBI	Grand Bahama Island	CSQ	Coastal Sentry Quebec (ship)
RKV	Rose Knot Victor (ship)	WOM	Woomera, Australia
GTI	Grand Turk Island	WHE	Wheeling (ship)
BDA	Bermuda Island	CTN	Canton Island
ANT	Antigua Island	HAW	Hawaii
CYI	Grand Canary Island	CAL	Pt. Arguello, California
ASC	Ascension Islands	GYM	Guaymas, Mexico
KNO	Kano, Nigeria	WHS	White Sands, New Mexico
PRE	Pretoria, South Africa	TEX	Corpus Christi, Texas
TAN	Tananarive, Madagascar Island	EGL	Eglin AFB, Florida



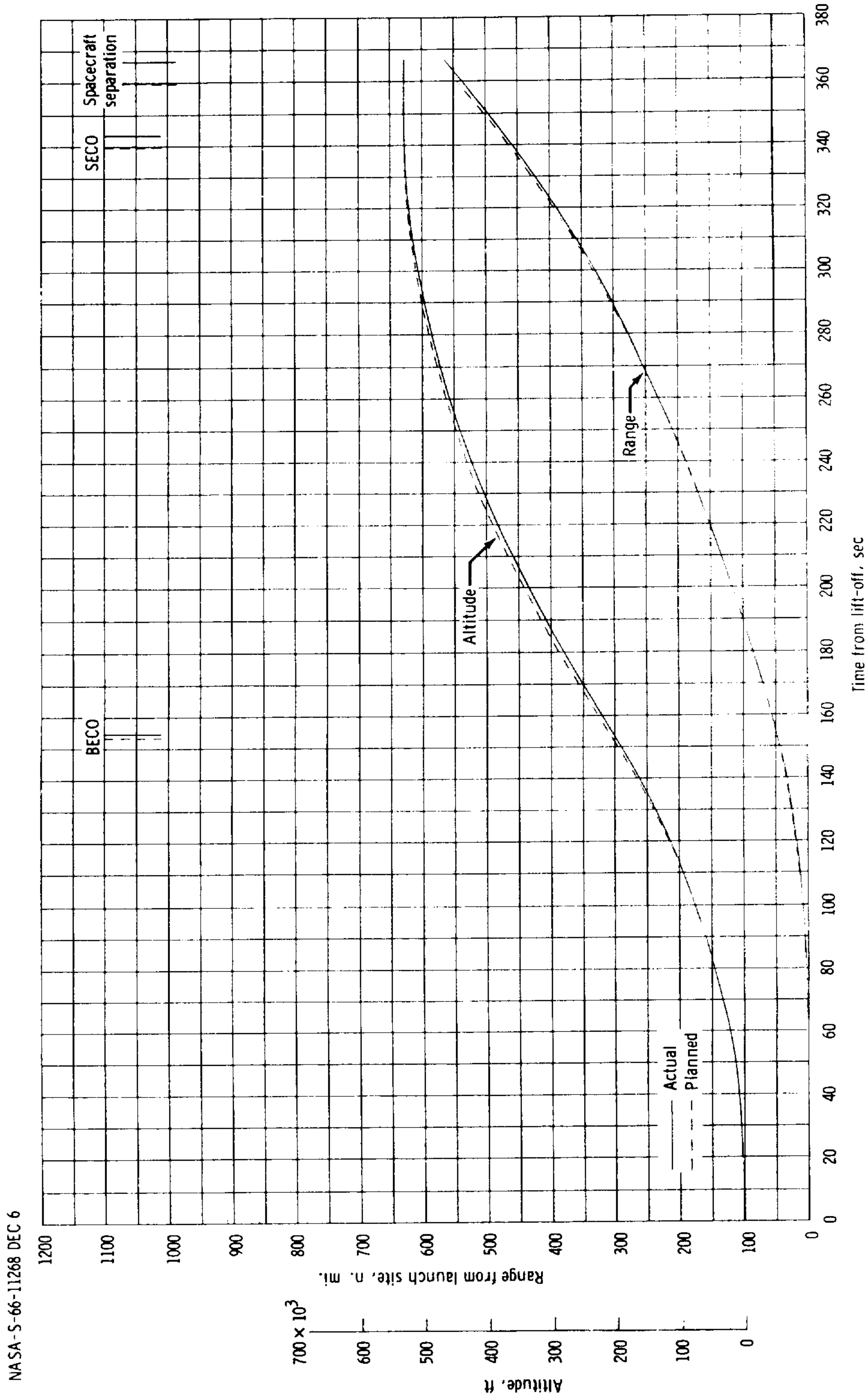
(b) Revolution 10.

Figure 4.3-1.- Continued.



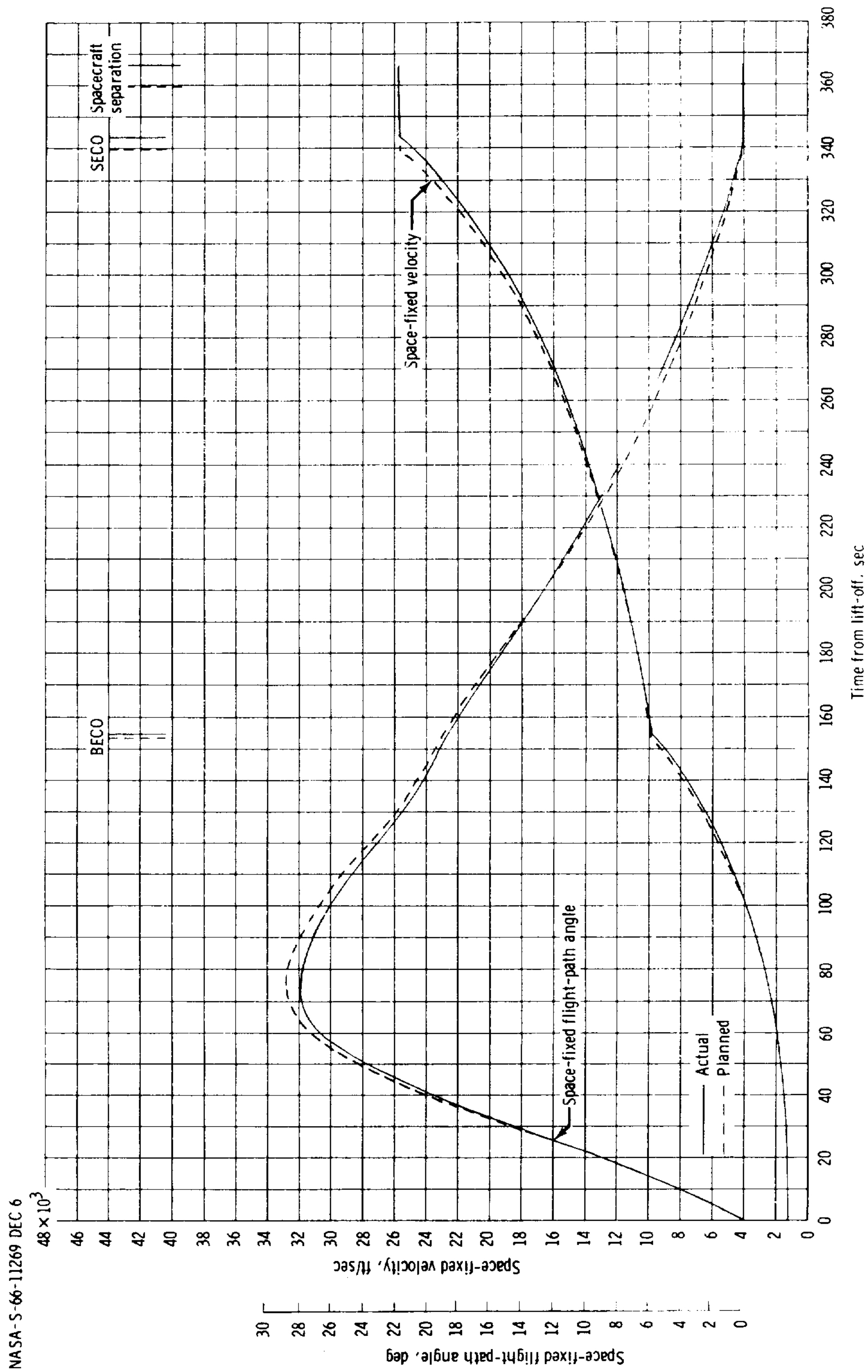
(c) Reentry.

Figure 4.3-1.- Concluded.



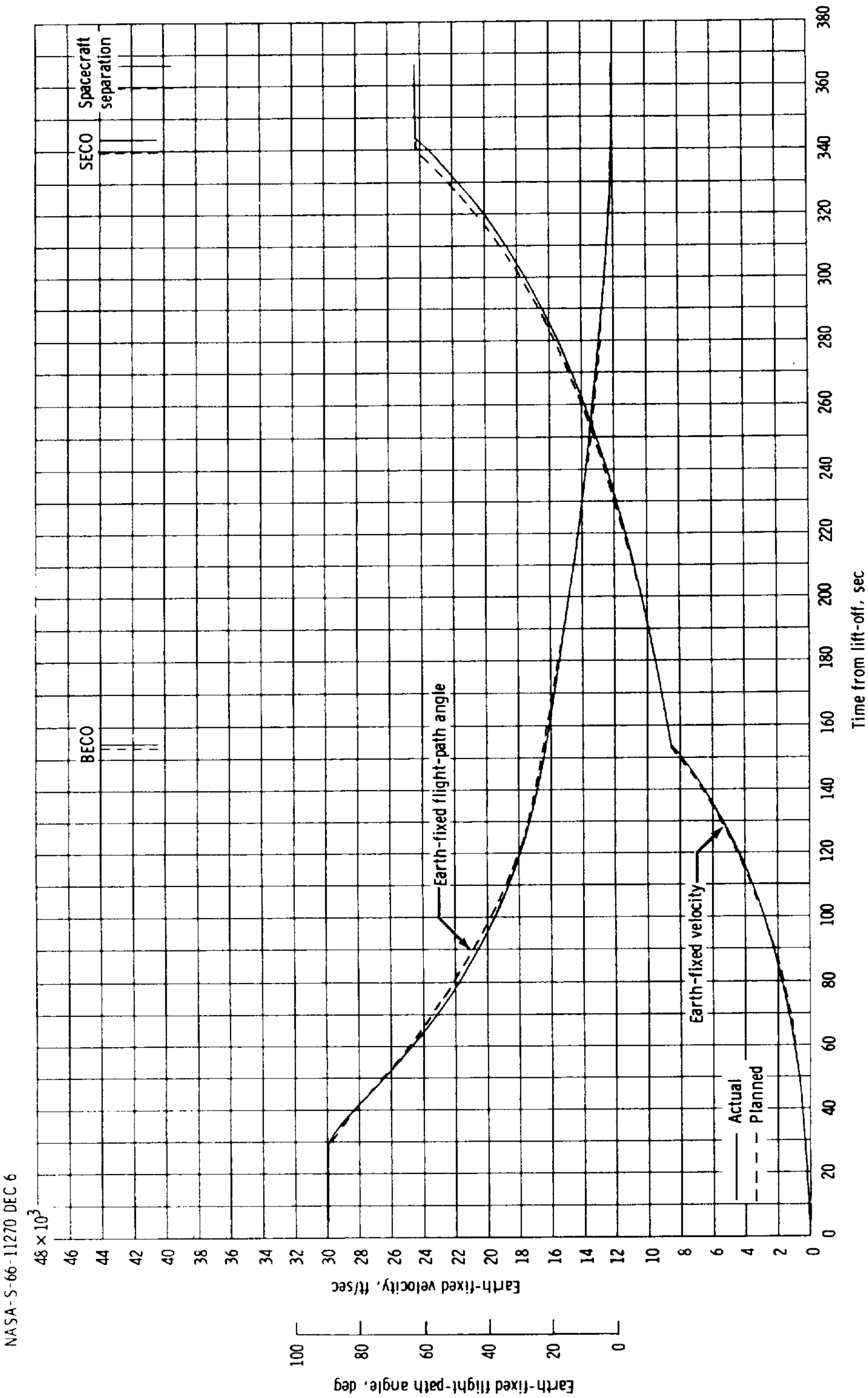
(a) Altitude and range.

Figure 4.3-2. - Trajectory parameters for Gemini Space Vehicle launch phase.



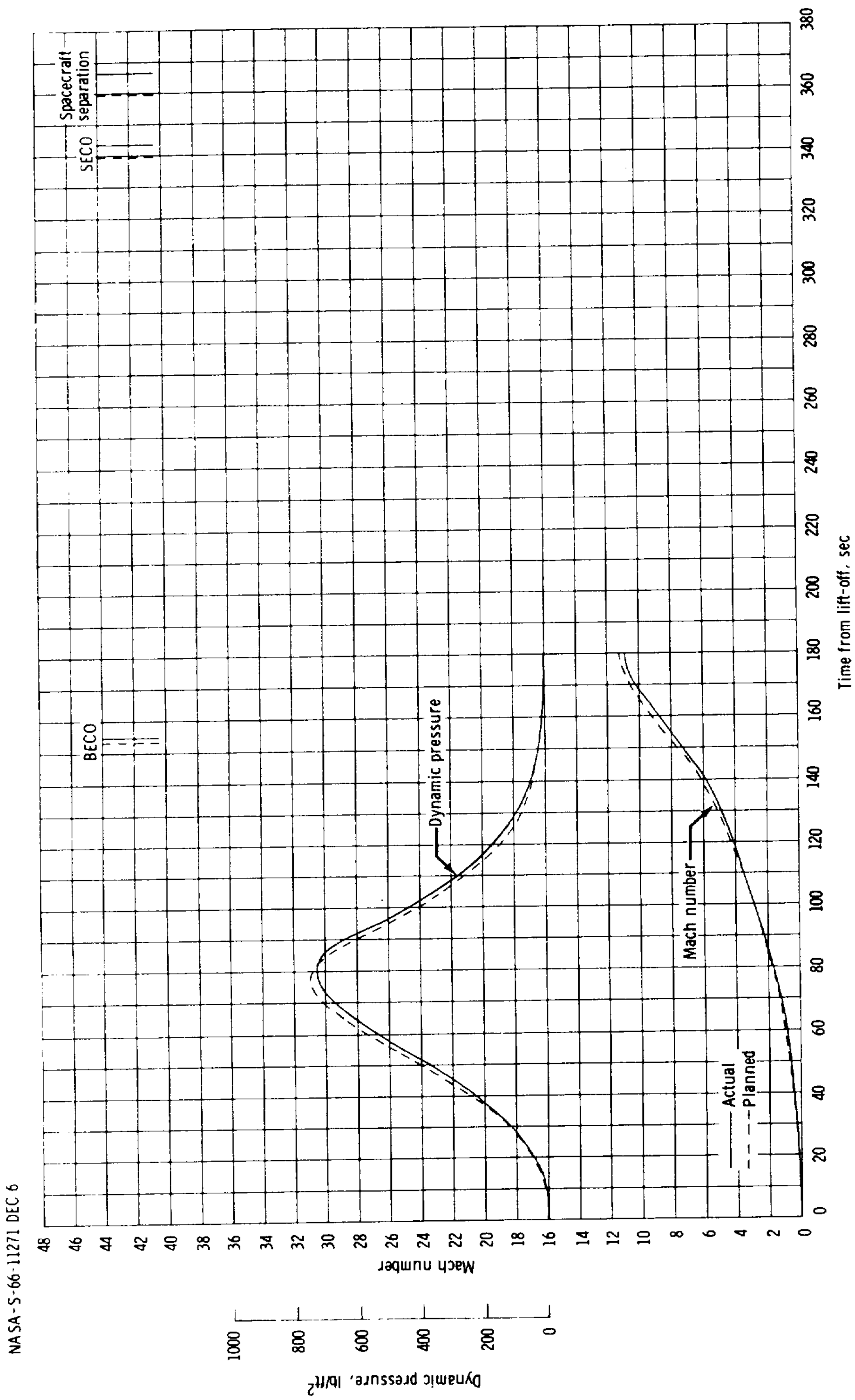
(b) Space-fixed velocity and flight-path angle.

Figure 4.3-2.- Continued.



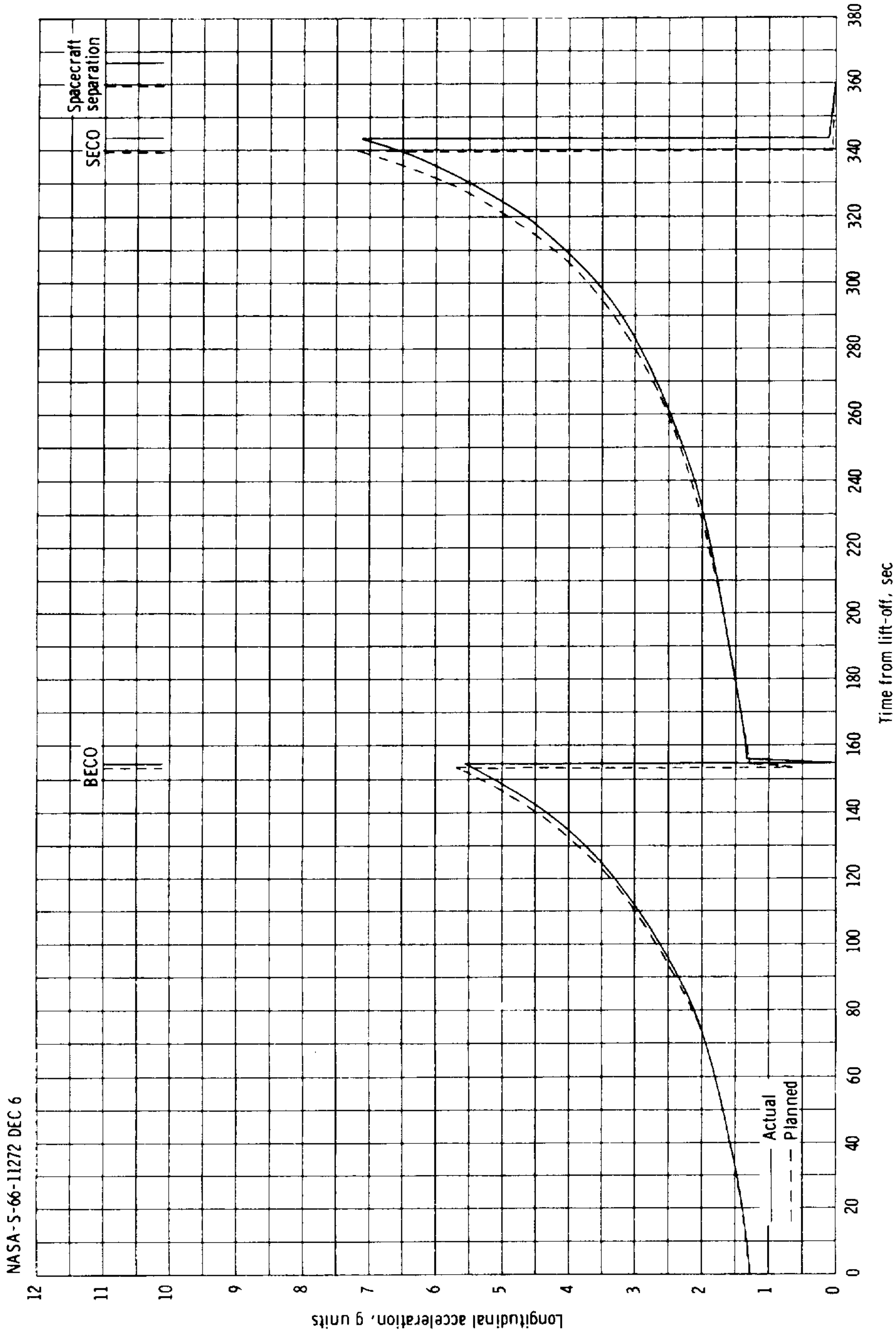
(c) Earth-fixed velocity and flight-path angle.

Figure 4.3-2. - Continued.



(d) Dynamic pressure and Mach number

Figure 4.3-2.- Continued.



(e) Longitudinal acceleration.

Figure 4.3-2.- Concluded.

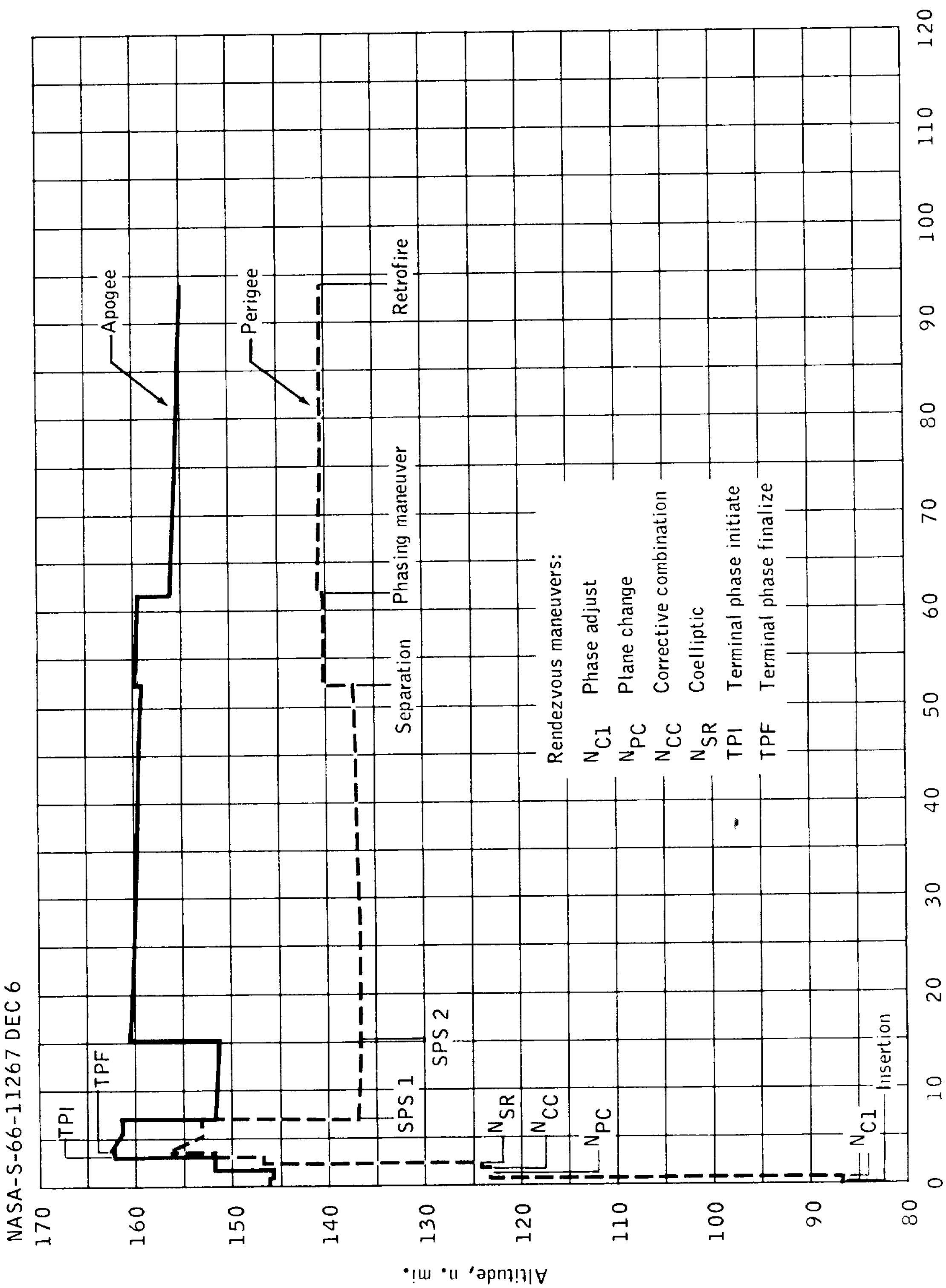


Figure 4.3-3. - Apogee and perigee altitudes for the Gemini XII mission.

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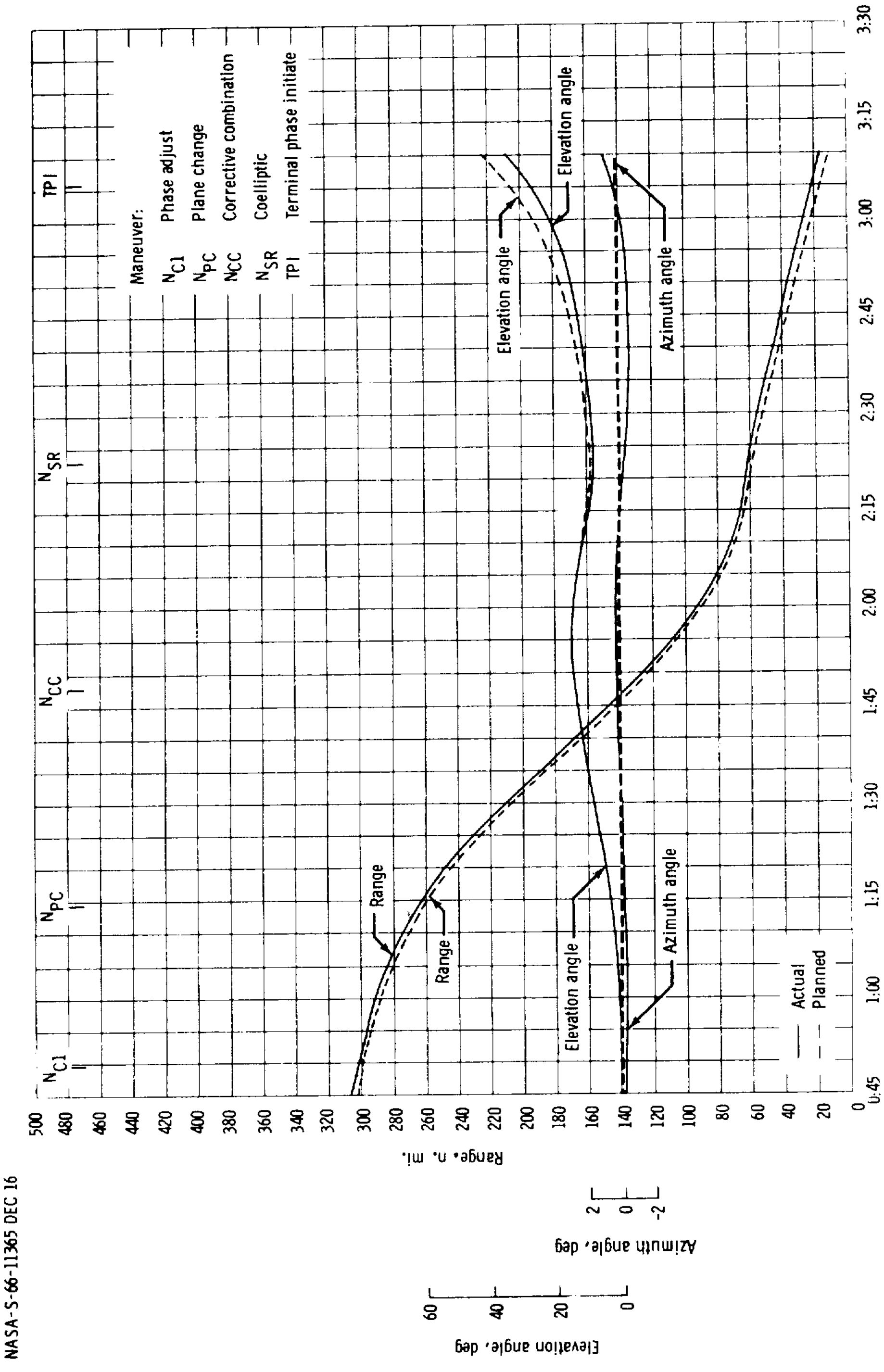

(a) Relative range, azimuth, and elevation from Spacecraft 12 to Gemini XII GATV during pretransfer maneuvers.

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

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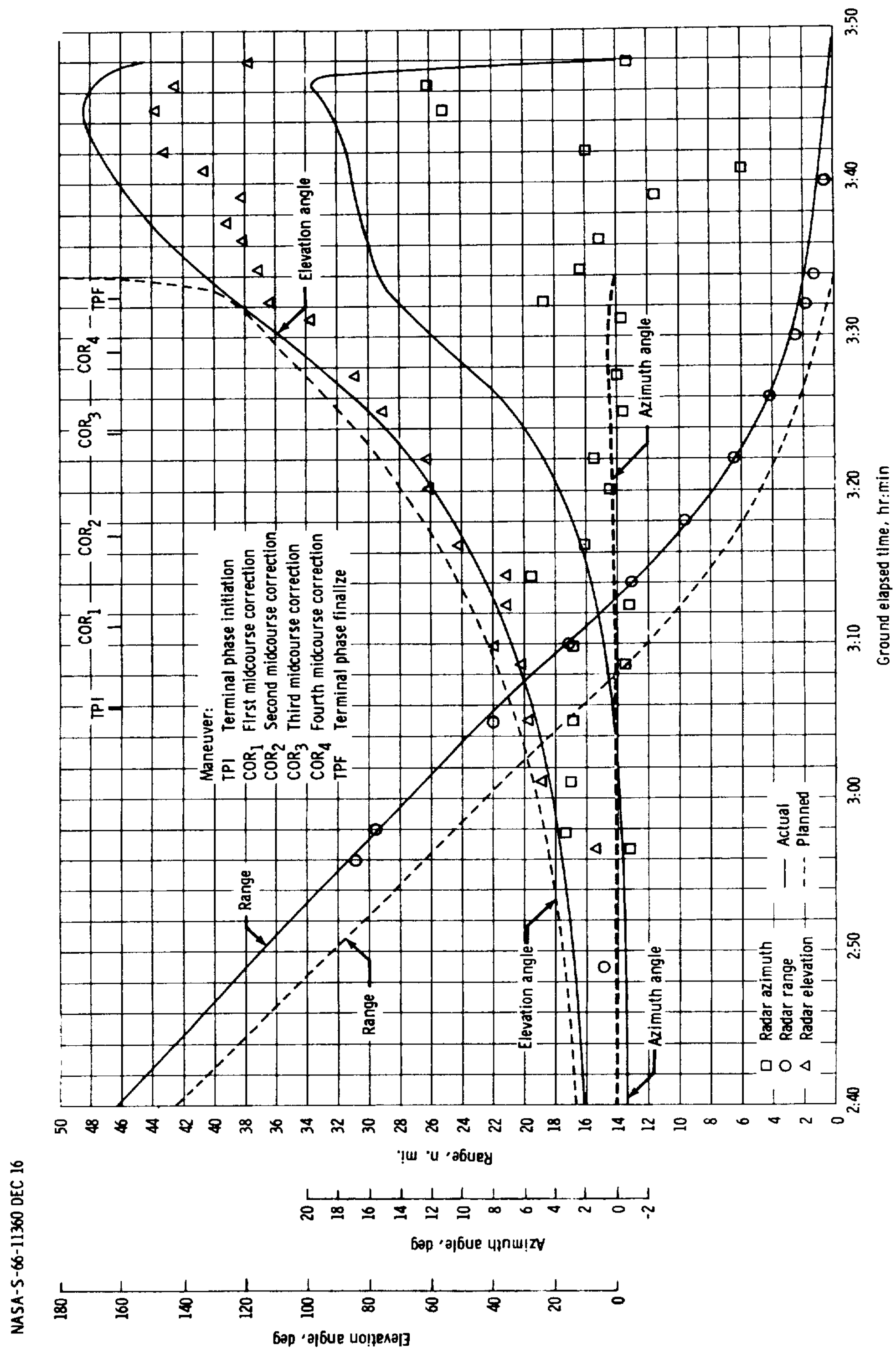

(b) Relative range, azimuth, and elevation from Spacecraft 12 to Gemini XII GATV during terminal phase maneuvers.

Figure 4.3-4. - Continued.

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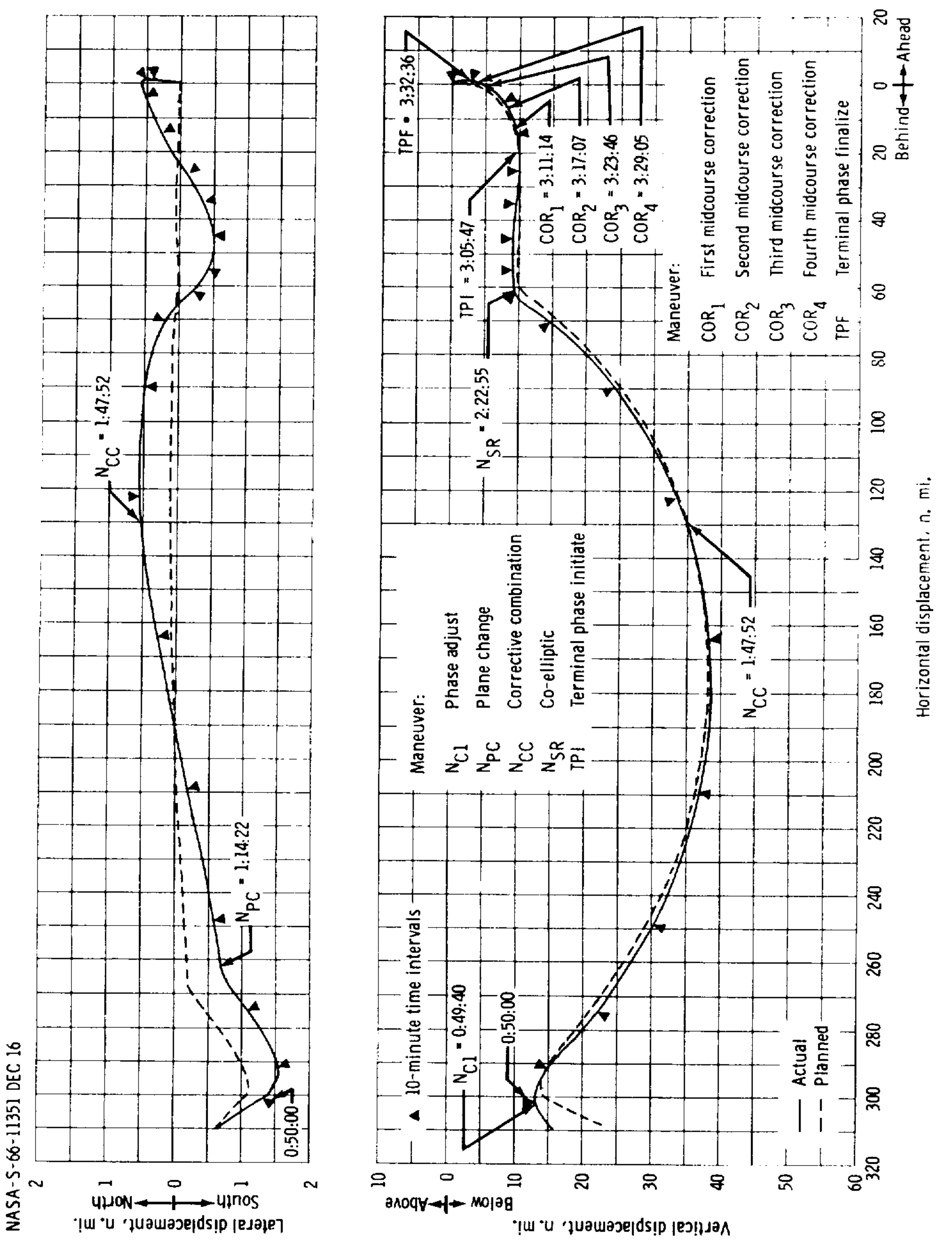
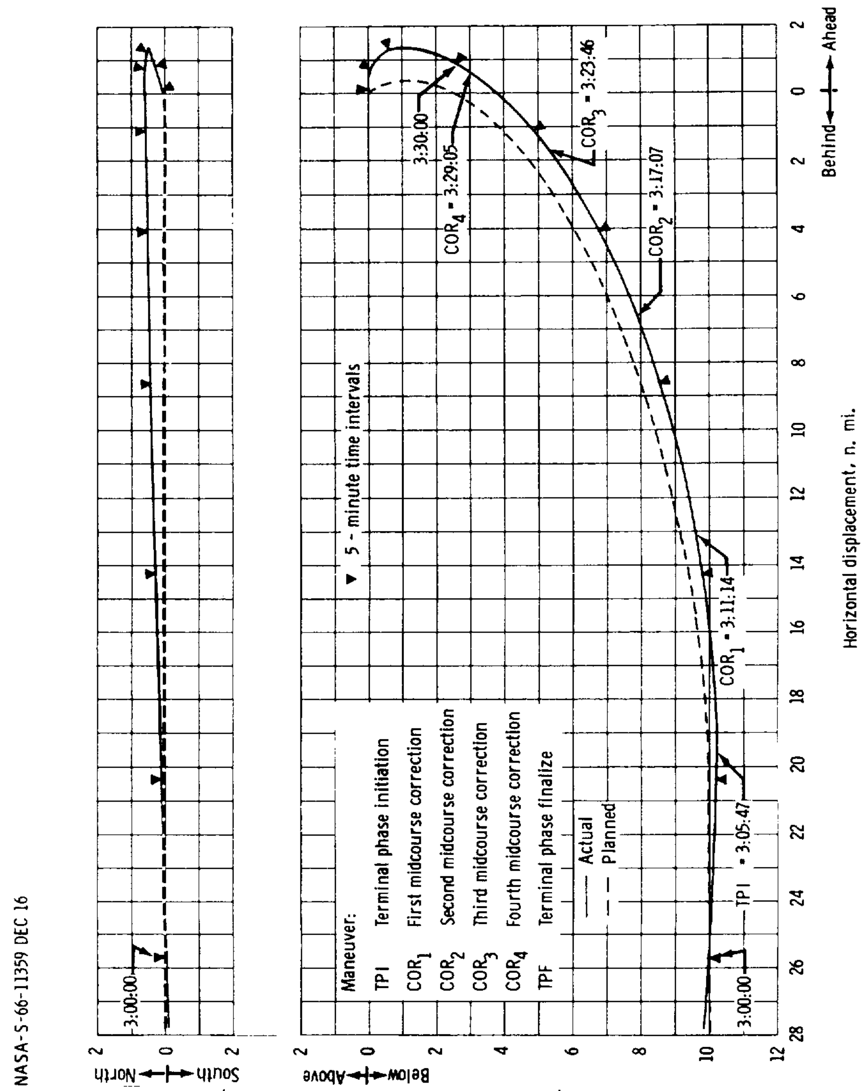


(c) Relative trajectory profile as measured from Gemini XII GATV to Spacecraft 12 in curvilinear coordinate system.

Figure 4.3-4. - Continued.



(d) Relative trajectory profile from TPI to TPF as measured from Gemini XII GATV to Spacecraft 12 in curvilinear coordinate system.

Figure 4.3-4. - Concluded.

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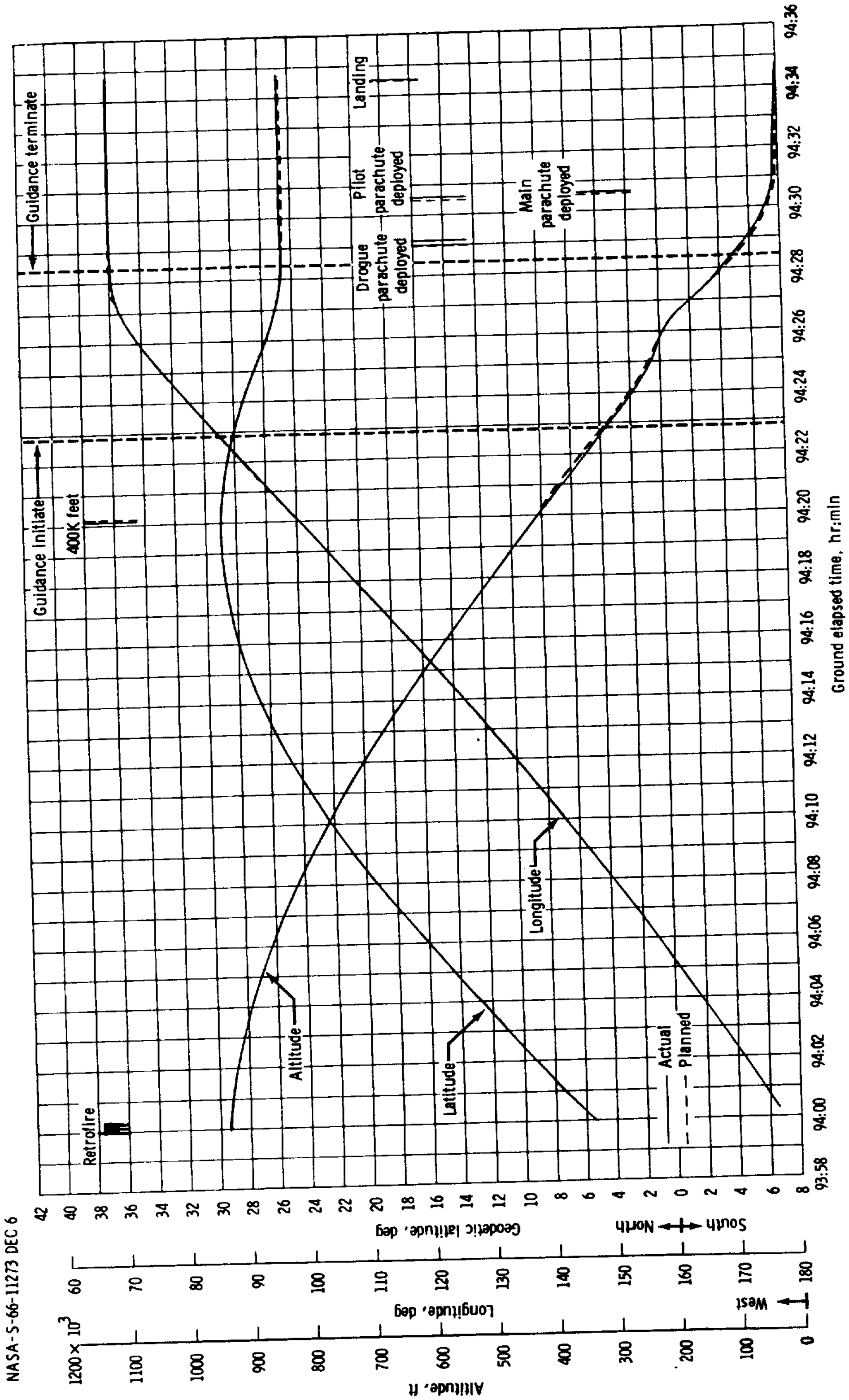

(a) Latitude, longitude, and altitude.

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

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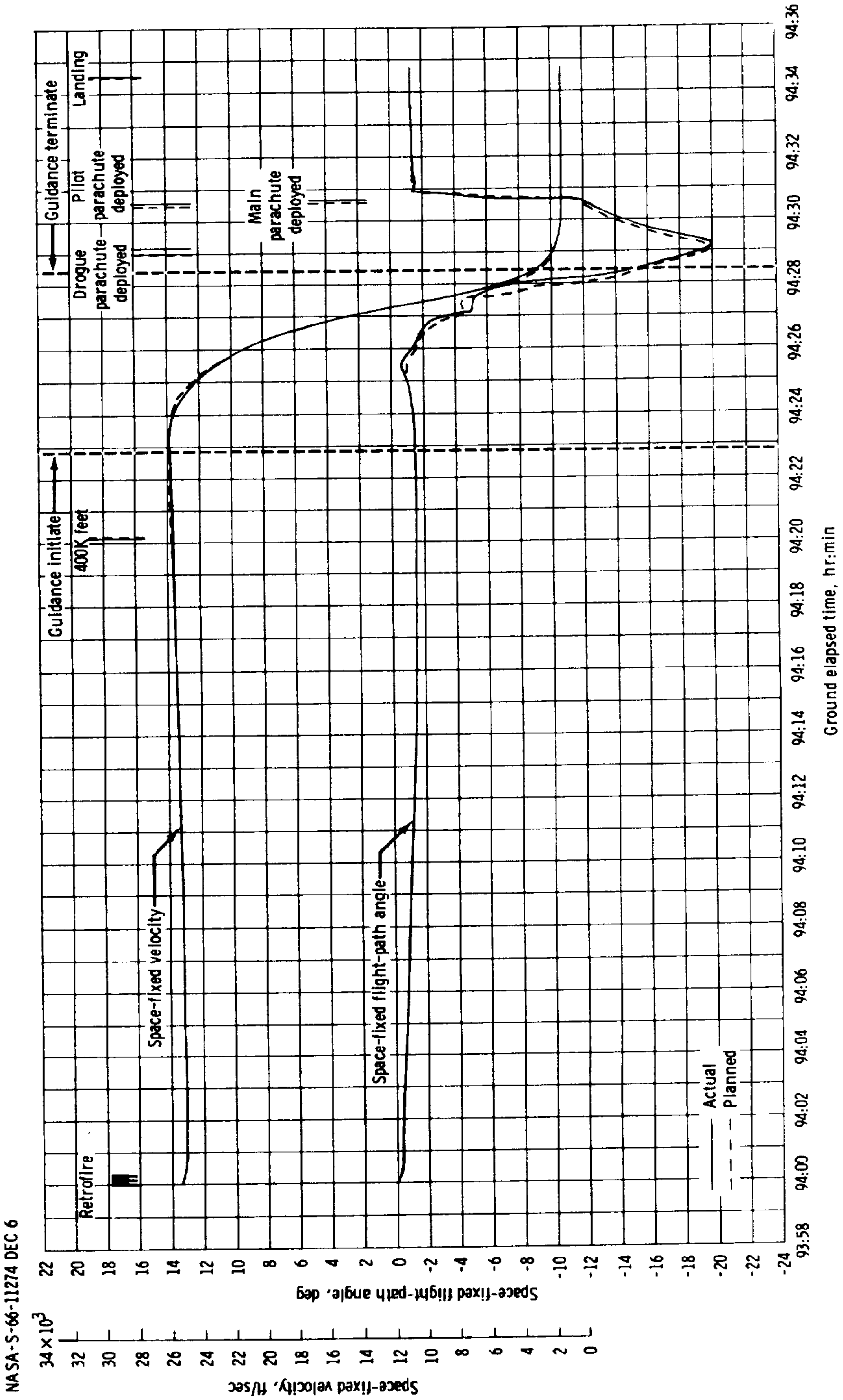

(b) Space-fixed velocity and flight-path angle.

Figure 4.3-5. - Continued.

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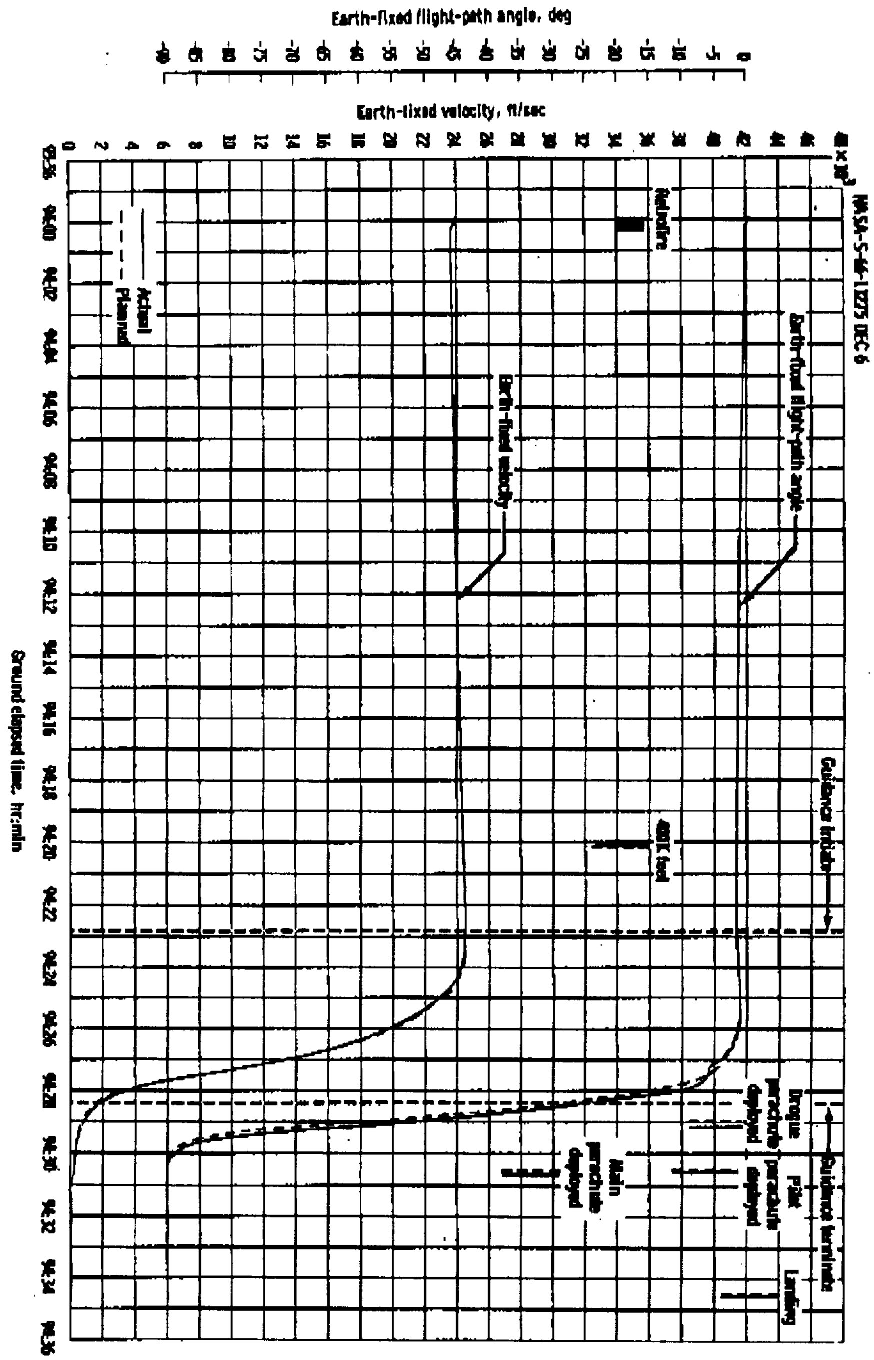

(c) Earth-fixed velocity and flight-path angle.

Figure 4.3-5.- Continued.

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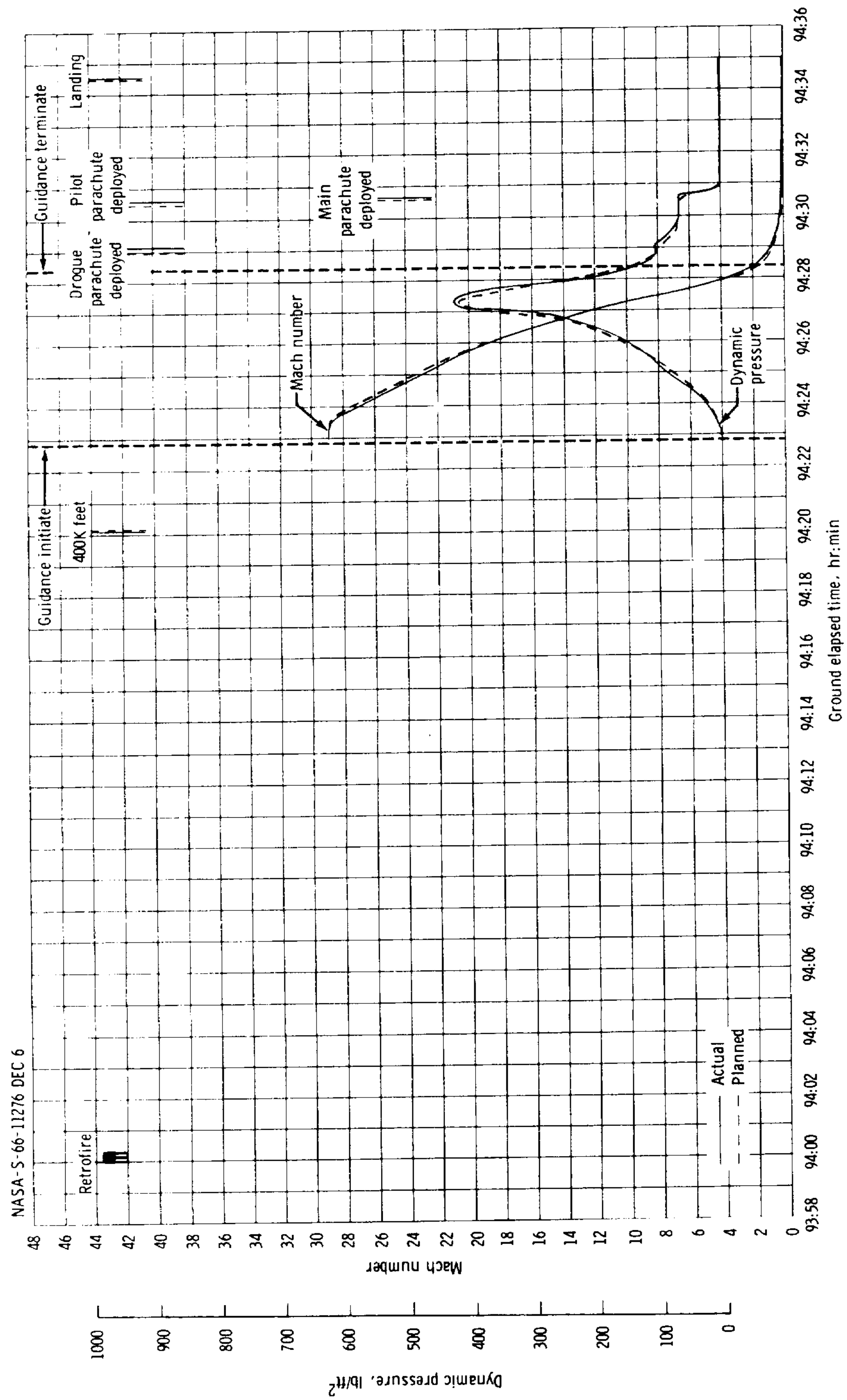
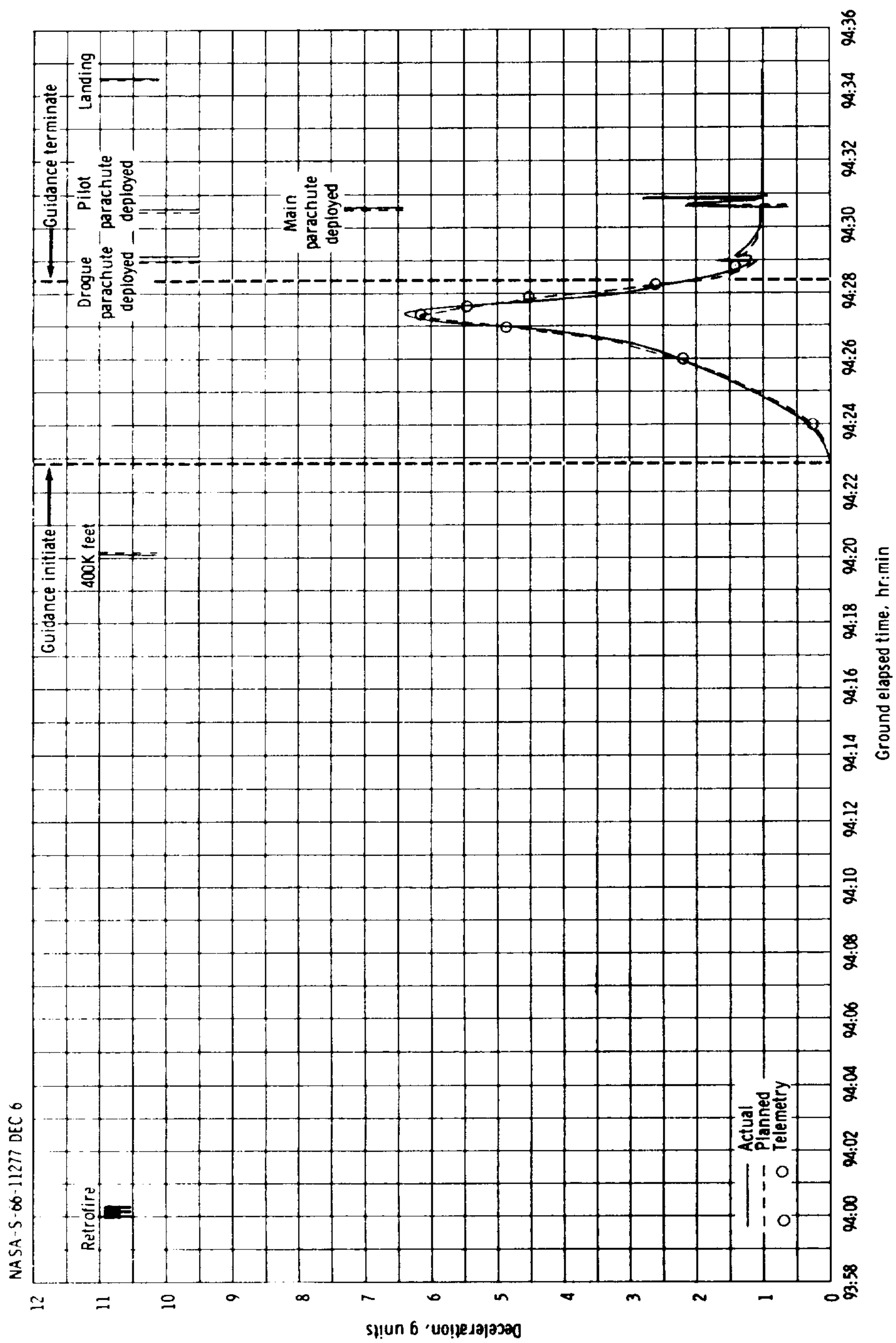


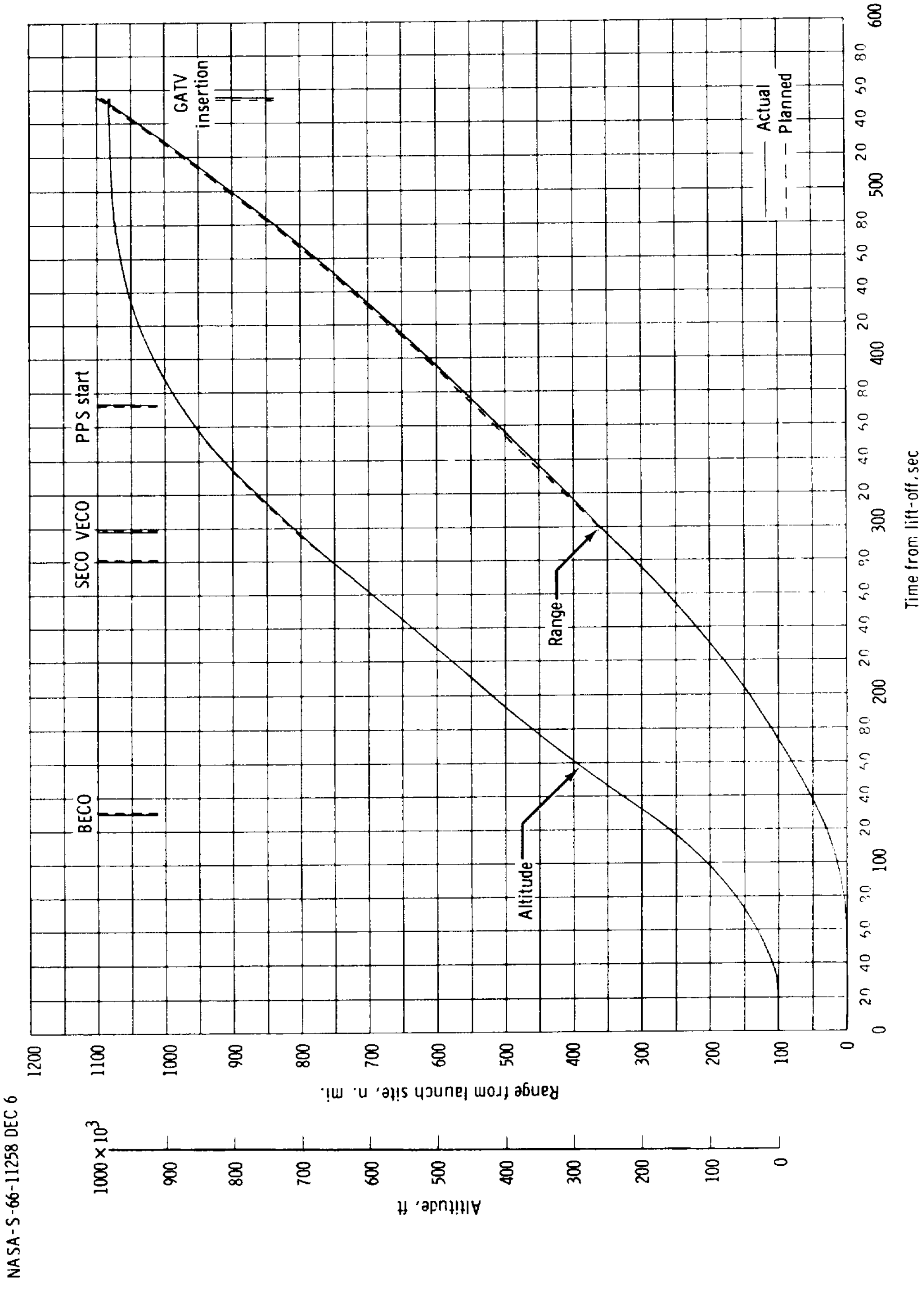
(d) Dynamic pressure and Mach number.

Figure 4.3-5. - Continued.



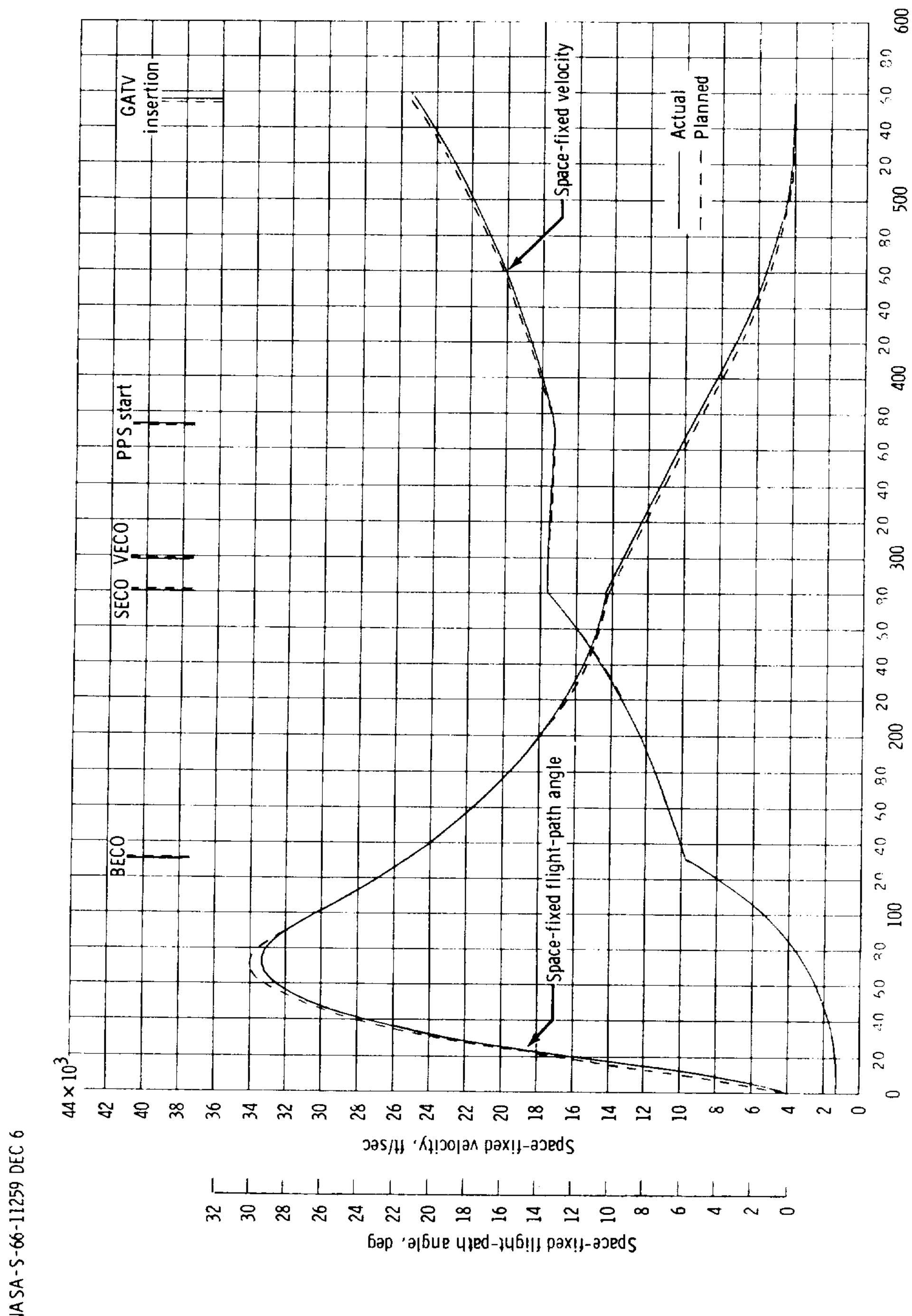
(e) Longitudinal deceleration.

Figure 4.3-5. - Concluded.



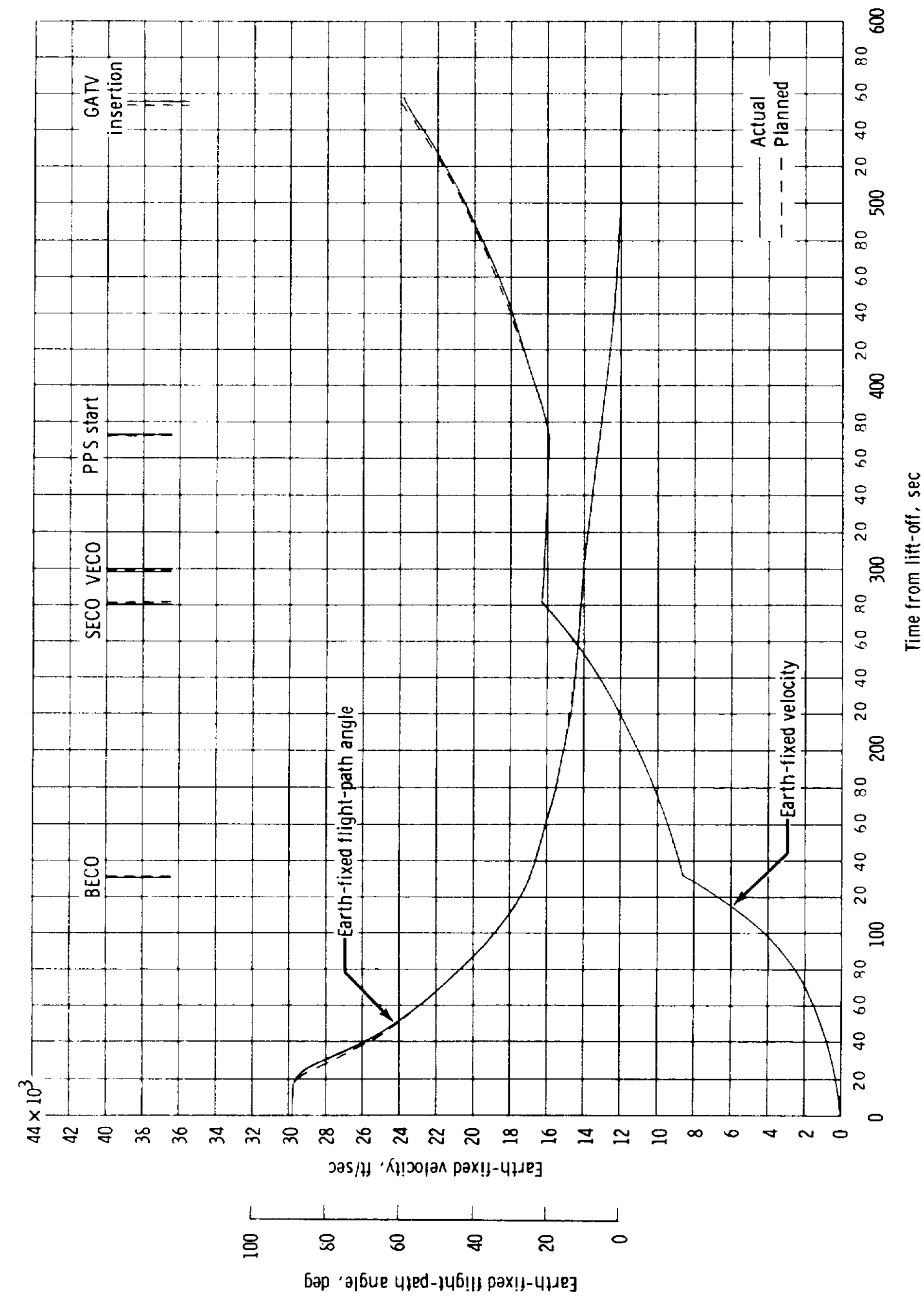
(a) Altitude and range.

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



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

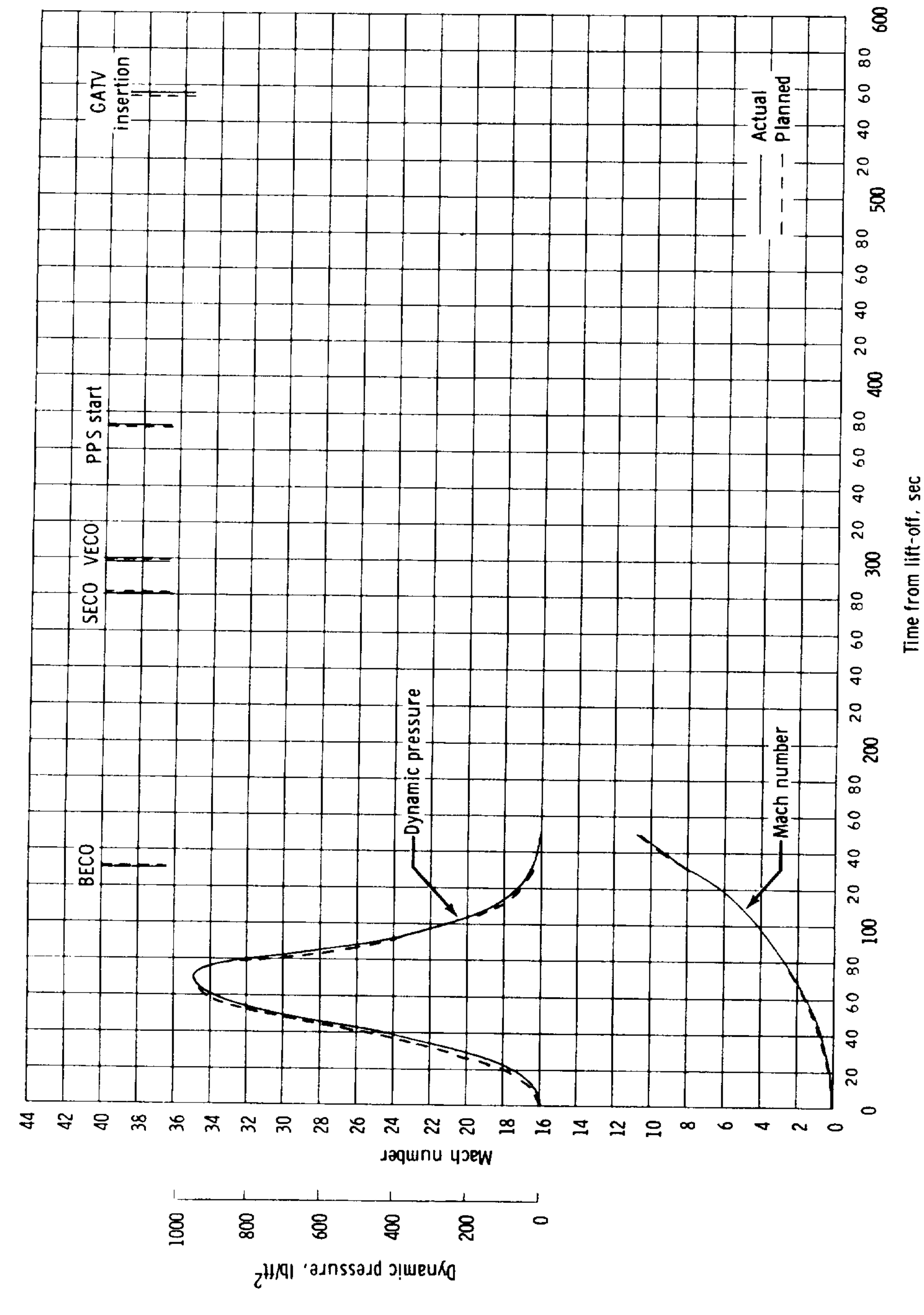
Figure 4.3-6.- Continued.



NASA-S-66-11260 DEC 6

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

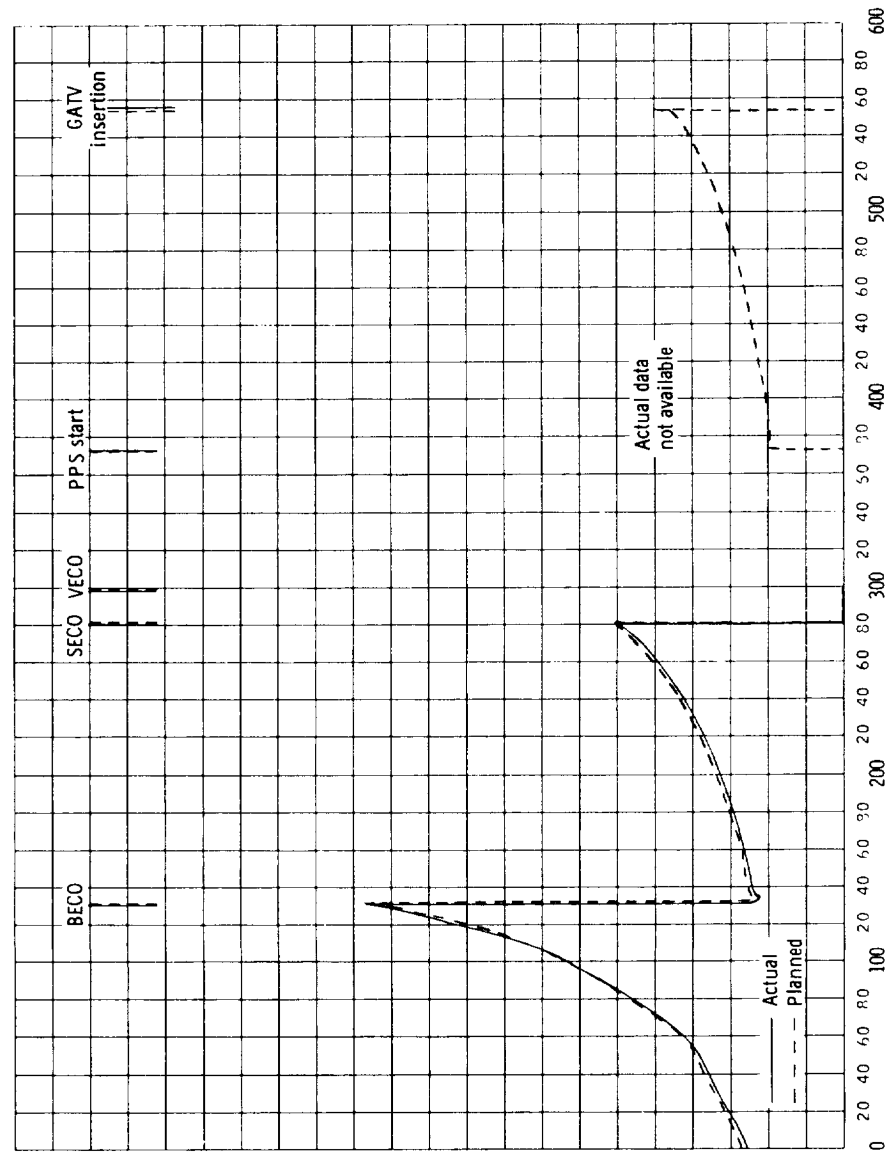
Figure 4.3-6.- Continued.



NASA-S-66-11261 DEC 6

(d) Dynamic pressure and Mach number.

Figure 4.3-6.- Continued.



NASA-S-66-11262 DEC 6

(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 satisfactorily sustained the loading and environment of the mission. During the postflight inspection, a bent shingle and a slightly deformed equipment-bay door were noted. This is the second time during the twelve Gemini flights that a shingle was bent and indicates a relatively hard landing, although well within the structural limits of the spacecraft. A particular combination of wave slope, swing on the parachute, and drift will produce this type of landing, and some secondary structural damage may be expected.

Some difficulty was reported by the crew in making the second docking, apparently as a result of insufficient closing velocity. All available evidence indicates that contact between the spacecraft and the Target Docking Adapter (TDA) caused no damage to either vehicle. Refer to section 5.7 for more details.

The windows on the Gemini XII spacecraft appeared cleaner after reentry than previous spacecraft windows. A qualitative examination revealed contamination on the outer surfaces that was very similar in content to that found on previous windows. There was no visible contamination between the window assemblies, indicating that the corrective action taken on this spacecraft was effective. This correction, minimizing the outgassing from nonmetallic materials in the window area, was incorporated as a result of finding a substantial amount of silicon between the window assemblies of Spacecraft 11. During the umbilical extravehicular activity (EVA), the pilot cleaned the left-hand window with a cloth and virtually all deposits were removed. Analysis of the cloth used to wipe the command pilot's window revealed no contaminant that could be traced to the window itself. This wiping was not specifically for the purpose of obtaining a sample for analysis, and the subsequent environment of the cloth was such that it would heavily mask the orbital contaminants.

Based on the available evidence, which is a compilation of ground tests, crew debriefings, and various corrective actions taken throughout the Gemini Program, the following events indicate the major factors contributing to the window contamination. The launch temperatures in certain areas are such that nonmetallic materials will outgas and collect as a film on the window surfaces. The ablative nose cover is probably the prime contributor. Under specific conditions, the staging operation also deposits contaminants on the window surface. Outgassing and window deposition, primarily in the local window area, continue in the orbital environment, although to a much lesser degree than that of the launch environment, with a significant percentage of deposition occurring in the first 24 hours. The reentry, another prime contributor, is characterized by a deposition from the heat shield early in the reentry and a subsequent burning as the heating rate builds to its peak. On Gemini XII, the left-hand window was clean just prior to reentry, but the film and subsequent burning occurred on both windows as it has in past missions. This charred substance was subjected to detailed optical and chemical examination. In summary, the windows require protection from contamination by some means, such as a protective cover, during launch and the first 24 hours of orbit.

As with the previous flights, the reentry trim angle of attack and lift-to-drag ratio were well within the expected ranges. The spacecraft heating was normal, with a maximum heating rate of 46.7 $Btu/ft^2/sec$ and a total heat load of 9600 Btu/ft^2. The stagnation point was measured to be 20 inches below the axis of symmetry.

Figure 5.1.1-1 presents a comparison summary of flight stagnation-point measurements with the trim angle stagnation-point distances obtained from wind tunnel pressure data. The figure, presented in previous mission reports, has been modified in the following manner based on further analysis:

(a) The band of uncertainty, illustrated by the dashed lines, represents the uncertainty in determining the stagnation point from wind tunnel pressure data.

(b) The flight stagnation point as measured on the heat shield was re-evaluated.

(c) The trim angle of attack was redesignated by virtue of the solidification conditions (convective heating rate $\approx$17.5 $Btu/ft^2/sec$) rather than the previously assumed Mach number of 15.

Although these data present rough correlation with the trim angle of attack as derived from flight data and wind tunnel measurements, it is of limited technical value because of the uncertainties in the data as quoted above.

NASA-S-66-11330 DEC 14

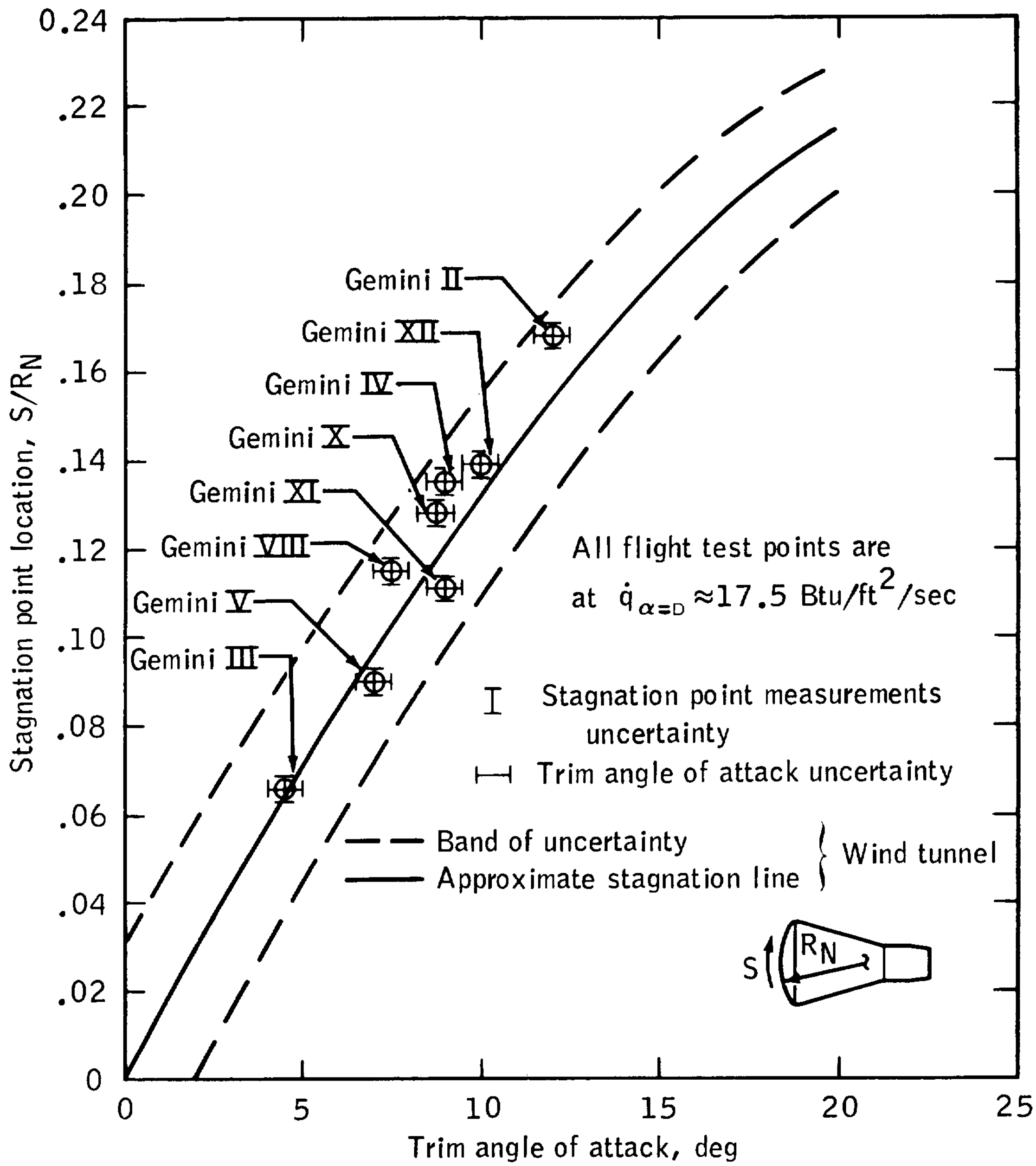


Figure 5.1.1-1. - Gemini stagnation point location.

5.1.2 Communications System

The spacecraft communications equipment provided satisfactory support for all phases of mission operations. Minor areas of concern were noted and investigated during postmission debriefings and data analyses.

The voice tape recorder recorded eleven tapes of satisfactory quality. The recorder used on Spacecraft 11 had operated intermittently, possibly because of small particles floating at random (under zero-gravity conditions) and entering the capstan drive mechanism. On Spacecraft 12, special precautions—careful cleaning and handling, and use of a plastic cover—had been taken during the prelaunch test period to prevent the entry of foreign particles into the recorder drive mechanism.

The pilot reported that during EVA preparation he discovered that the microphone on the right side of his helmet appeared to be inoperative. The audio circuits involved are redundant, including dual helmet microphones, wiring, and preamplifiers; therefore, no voice communications problem occurred. Postflight tests revealed that the problem was external to the helmet or suit harness and was caused by a broken wire at a connector in the spacecraft wiring.

During the launch phase, the Ascension tracking station was given an acquisition time which was in error by approximately three minutes. As a result, the station did not acquire or track the spacecraft, and there was a complete loss of telemetry, radar tracking, and voice communications. All communications were normal approximately ten minutes later over Tananarive, the next tracking station, and the effect on mission operations was insignificant.

The communications blackout interval caused by reentry plasma effects was as follows:

Signal loss began . . . . 94:21:59 g.e.t.

Signal loss complete . . . . 94:22:04 g.e.t.

Signal return began . . . . 94:27:20 g.e.t.

Signal return complete . . . . 94:27:25 g.e.t.

The blackout times were determined from real-time telemetry signal strengths recorded at the Texas and Grand Turk Island stations.

In contrast to several previous missions, there were few, if any, spacecraft-to-ground voice transmissions with poor signal-to-noise quality attributable to improper microphone positions or to lower-than-normal voice levels.

5.1.2.1 Ultrahigh frequency voice communications.- UHF voice communications were satisfactory during all phases of the mission except during the normal reentry communications blackout period. Orbital spacecraft-to-ground voice transmissions were described by several monitors as superior in fidelity to those of any previous mission. UHF voice traffic in the recovery area was so heavy that the crew, except for a few transmissions, chose to contact the Capsule Communicator in Houston via the HF voice link through the Cape Kennedy tracking station. Voice operated transmitter (VOX) keying was used extensively with excellent results during the extravehicular activities. No instances were noted of accidental VOX keying caused by heavy breathing.

5.1.2.2 High frequency voice communications.- The HF voice communications equipment was included in the Gemini spacecraft for emergency purposes during orbital flight and to aid in locating the spacecraft after landing. The flight crew reported that during the orbital mission phase the HF link was used to monitor music over the Cape Kennedy tracking station but no attempt was made to contact the station. The HF link was used to contact the Capsule Communicator through the Cape Kennedy tracking station during the postlanding phase and HF direction-finding (HF-DF) signals were also transmitted and received.

5.1.2.3 Radar transponder.- The operation of both C-band radar transponders was very satisfactory, as evidenced by the excellent tracking information supplied by the network stations. Beacon-sharing operations by ground radars were satisfactory, and C-band tracking during reentry was also satisfactory.

5.1.2.4 Digital Command System.- The performance of the Digital Command System (DCS) was satisfactory throughout the mission. Flight control personnel reported that all commands sent to the spacecraft were validated.

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

5.1.2.6 Antenna systems.- All antenna deployed and operated properly during the mission. The HF antenna installed on the adapter assembly was extended in orbit, and the HF whip antenna installed on the reentry assembly was deployed and retracted during the postlanding phase of the mission.

5.1.2.7 Recovery aids.- UHF voice communications between the spacecraft and the recovery forces were excellent. The flashing light extended normally but was not required and was not turned on by the crew. During the recovery phase of the mission and prior to the hatches being opened, communications between the swimmers and the flight crew were also excellent. The UHF recovery beacon operated normally, and signals were received at a distance of approximately 200 nautical miles. Postlanding HF-DF signals were transmitted by the spacecraft and received by several Department of Defense stations which then plotted direction bearings to the spacecraft. The operation of spacecraft recovery aids is further discussed in section 6.3.3.

5.1.3 Instrumentation and Recording System

Very good performance was obtained from the Instrumentation and Recording System throughout the mission. The PCM tape recorder functioned continuously from prior to lift-off until it was powered down on the deck of the recovery ship at 95 hours 45 minutes g.e.t. The quality of the delayed-time data was outstanding, with over 99 percent usable for the mission.

5.1.3.1 System performance.- A total of 239 parameters were monitored during this mission, with satisfactory performance obtained from each parameter except the carbon-dioxide partial pressure sensor. This measurement rose abnormally twice near the end of the mission. Further discussion regarding this parameter may be found in section 5.1.4.

5.1.3.2 Real-time data quality.- Since the delayed-time PCM tape recorder operated properly, computer processing of the real-time telemetry data was minimal. The usable data obtained represent normal operation of the real-time telemetry data link. From the real-time computer-processed time edits, the following usable data percentages were obtained:

Station	Phase or revolution	Usable data, percent
Cape Kennedy	Launch, 1/2, 2/3, 13/14, 16/17, 27/28, 29/30	99.751
Hawaii	2, 5, 49	97.695

5.1.3.3 Delayed-time data quality.- The delayed-time data reception at the Cape Kennedy, Hawaii, Grand Canary Island, Rose Knot Victor, Antigua, and Bermuda stations, as well as the data recovered from the onboard PCM tape recorder, are summarized in table 5.1.3-I. Data from 45 of the 58 delayed-time data playbacks, as well as the last revolution and reentry data from the onboard PCM recorder, are shown in the table. From the computer-processed data edits, the delayed-time data summary shown in table 5.1.3-I reveals that an average of 99.279 percent usable data was obtained through the recorder system.

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	1, 12, 13, 14, 15, 16, 27, 28, 29, 30, 31, 43, 44, 45, 46, 54, 56, 57, 58	26:48:53	965 329	6 153	0.637	99.363
Hawaii	2, 3, 4, 5, 17, 18, 32, 33, 34, 47, 48, 49	16:32:53	595 731	4 367	0.733	99.267
Grand Canary Island	10, 24, 39, 40, 41	8:39:01	311 414	1 239	0.398	99.602
Rose Knot Victor	23, 50, 52, 53	6:19:40	227 790	3 425	1.504	98.496
Antigua	11, 25, 26	4:02:39	145 590	2 153	1.479	98.521
Bermuda	54, 55	1:53:19	67 986	55	0.081	99.919
Onboard recorder	59, reentry, postlanding	3:19:31	119 706	161	0.135	99.865
Summation		67:35:56	2 433 546	17 553	0.721	99.279

5.1.4 Environmental Control System

The performance of the Environmental Control System (ECS) was satisfactory throughout the mission. The crew reported that they were reasonably comfortable during the mission. A reduction of suit flow was required by both crewmen during the sleep periods to keep from being uncomfortably cold. The pilot was usually slightly warmer than the command pilot, primarily because of the added insulation on the pilot's suit. Another contributing factor, as noticed by the crew, was that the usual attitude of the spacecraft was one which allowed the sun to shine more on the right side of the spacecraft than on the left.

The drinking water in the adapter tank was depleted during the morning of the final day of flight. The adapter storage tank capacity was slightly less than the average amount required for a 2-man, 4-day mission.

The command pilot reported eye irritation at the end of the umbilical EVA period. The command pilot's analysis at the time indicated that the irritation had been caused by perspiration which had collected around his eyes and entered them when he blinked. After repressurization of the spacecraft, he wiped his eyes as soon as he opened the visor and the irritation immediately disappeared.

A failure of the carbon-dioxide partial pressure sensor was noted as having started at retrofire. The telemetered value of carbon-dioxide partial pressure started increasing at retrofire to a maximum of 15.1 mm at six minutes after retrofire. The value then decreased slowly to 0.4 mm at landing, at which time another increase was noted to 12 mm in four minutes. The reading then decreased to 1.4 mm over the next 12 minutes. These characteristics have not been observed before. The sensor was removed from the spacecraft for failure analysis; however, the anomaly could not be repeated.

5.1.5 Guidance and Control System

5.1.5.1 Summary.- The performance of the Guidance and Control System was excellent throughout the mission with the exception of a partial failure of the radar transponder which occurred just after the coelliptic maneuver, prior to the start of the closed-loop rendezvous phase. Ascent backup guidance was satisfactory, resulting in only small navigation errors at second stage engine cutoff (SECO). The Insertion Velocity Adjust Routine (IVAR) solution was applied and resulted in near-nominal insertion conditions. Because of the radar transponder anomaly, the rendezvous

 [redacted]

was performed using onboard backup solutions for the terminal phase. Radar range, although intermittent, was used as backup information and for braking. The Auxiliary Tape Memory Unit (ATMU) was utilized to augment the onboard computer memory and performed properly both times it was required. Reentry was performed using the automatic mode of control and resulted in a landing within 2.6 nautical miles of the planned landing point, with the vehicle visible from the recovery ship during descent on the main parachute. The control system performed properly, although degraded thruster performance (reported in section 5.1.8) reduced control effectiveness late in the mission. Table 5.1.5-I contains a summary of events significant to the Guidance and Control System.

5.1.5.2 <u>Inertial Guidance System performance evaluation</u>.-

5.1.5.2.1 Ascent phase: The Inertial Guidance System (IGS) roll, pitch, and yaw steering command deviations are presented in figure 5.1.5-1. Superimposed on the IGS steering quantities are the steering signals representative of the primary guidance system—Radio Guidance System (RGS)—along with the zero-wind upper and lower IGS attitude-error limit lines. The nominal Gemini XI steering-signal limit lines were considered to be valid for the Gemini XII mission and are included in the figure. Analog time histories of predicted pitch and yaw attitude errors for lift-off minus 3-hour winds are shown for the first 90 seconds of flight. No noticeable differences in error signals caused by guidance anomalies were observed between the two systems, and the IGS steering commands agreed very closely with those delivered by the primary system. Deviations between the primary and secondary attitude signals never exceeded 1.9 degress except at guidance initiation when the IGS pitch and yaw errors saturated to their respective limit values of plus six degrees, indicating a normal response to closed-loop steering commands. The primary guidance system initiated a 100-percent pitch-down command at the proper time, whereas on the previous two missions the primary-system pitch-validation test lasted about four seconds longer than nominal. The IGS attitude-error time histories were within predicted preflight 3-sigma boundaries and the IGS indicated agreement with the primary system. As in the past, there were minor deviations between the two systems due to known programmer and timing differences, initial engine misalignment, and drifts in the primary Three Axis Reference System.

If guidance switchover had occurred during early second stage flight, the vehicle would have achieved an insertion vector with a flight-path angle within 0.005 of a degree of nominal, an in-plane velocity error from 0 to 6 ft/sec, an out-of-plane velocity error of minus 1.0 ft/sec, and an altitude error of approximately 70 feet. The IGS SECO discrete was delivered about 11 milliseconds after the primary SECO signal,

[redacted]

verifying the comparison between primary and secondary guidance systems. Following the IVAR correction, the resultant orbit would have been the same as the actually achieved since the IVAR was followed for insertion on this mission.

The IVAR correction was applied during this mission with a resultant orbit of 86.8 by 146.1 nautical miles, indicating that both the IVAR solution and the application were accurate. The Incremental Velocity Indicator (IVI) display, as actually computed by the onboard IVAR, was reconstructed using IGS navigational and gimbal-angle data. At spacecraft separation, the reconstructed IVI display read 29.7 ft/sec forward, 6.4 ft/sec right, and 7.1 ft/sec down, displaying a 30.2 ft/sec in-plane and a 5.9 ft/sec out-of-plane velocity correction in component form. About 25 seconds later, after separation and a roll maneuver to the heads-up attitude, the reconstructed IVI display read 25.8 ft/sec forward, 13.4 ft/sec left, and 3 ft/sec down, confirming the crew-reported readings of 25 ft/sec forward, 13 ft/sec left, and 3 ft/sec down. After the pitch-attitude error indications were nulled, the reconstructed IVI display read 27.1 ft/sec forward and 8.5 ft/sec left, confirming crew-reported readings of 28.4 ft/sec forward and 9 ft/sec left. Following the 36-second IVAR maneuver, the reconstructed IVI readings were zero fore and aft, 1.2 ft/sec left, and 0.4 ft/sec up. The perigee correction to be applied at apogee, as computed in the IVAR, was minus 0.1 ft/sec. The values of the reconstructed IVAR parameters in the final computation, as compared with the actual values obtained from telemetry, verify that the orbit insertion equations and computer/IVI interface operated properly.

A preliminary estimate of Inertial Measurement Unit (IMU) component errors was obtained by comparing ground tracking measurements with guidance velocity data. The external tracking data used for these comparisons were final data from the MOD III system and the Missile Trajectory Measurement (MISTRAM) system, using 100K-foot legs. The tracking data were corrected after the flight. The difference in orbital parameters obtained from the real-time MISTRAM and MOD III data and from the postflight MISTRAM and MOD III data shown in table 5.1.5-II indicates the extent of postflight corrections to the data. The MOD III refraction correction amounted to minus 5 ft/sec, minus 220 ft/sec, and minus 340 ft/sec in the vertical direction at BECO, SECO, and SECO plus 20 seconds, respectively. Table 5.1.5-II contains an estimate of orbital injection parameters at SECO + 20 seconds as determined from the IGS, the real-time tracking data, and the postflight corrected data. The tracking data agree within the accuracy expected in the downrange (X) and the cross-range (Z) axes. The Y-axis (vertical) velocity residuals indicate a disagreement of greater than 1 ft/sec between sets of tracking data during the time interval between lift-off + 220 seconds and SECO. MISTRAM may have had a p and/or q bias error as noted on some of the previous flights.

[redacted]

Assuming that these MISTRAM errors exist, the velocity residuals obtained using MOD III data were used to estimate component errors which could account for the velocity error propagations along the computer X, Y, and Z axes (see figure 5.1.5-2). The accelerometer telemetry data acquired during ascent had no significant dropouts and were excellent for analysis.

On this flight, compensations for the gyro drift terms were made in addition to the normal accelerometer compensations made on the Gemini XI flight. The preflight test data, predicted drift values, postflight ascent values, and drift values used in the computer are shown in figure 5.1.5-3.

The velocity error along the X axis during the time interval from 80 to 300 seconds after lift-off was larger in magnitude than that noted during previous flights since Gemini III. At the present time, there is no term in the IMU error model that can adequately describe the error. The coupling of the vertical velocity into the X-axis can induce errors; however, the Y-axis error does not indicate a pitch drift or misalignment of the magnitude necessary to explain the X velocity error. The inertial component errors are assumed in the analysis program to be statistically independent and constant during the thrusting period. The X velocity error could have been caused by a condition which is a function of time and/or acceleration and therefore cannot be recovered directly. An X accelerometer scale-factor shift of 1470 ppm at lift-off which shifted downward 250 ppm at BECO might partially explain the trend of the data. X-axis accelerometer scale-factor shifts were noted in the preflight data but were not of sufficient magnitude to fit the flight data. An adequate fit of the data could not be obtained using error coefficients which were consistent with preflight data or engineering judgment.

Most of the gyro terms which can induce velocity errors along the Y and Z axes appear to have been adequately compensated. The major error contributor along the Y axis could be a misalignment of the Z-accelerometer input axis toward X. This error would propagate much like a pitch gyro drift, which also may have been the cause, but the pitch-drift-induced velocity error would couple into X. This coupling effect would be difficult to see on this particular flight because of the magnitude of the X error. Preflight data indicated that the pitch drift terms were stable. The Z velocity errors appear to have been caused by errors of the magnitude of those noted in Gemini XI.

The component errors which could have caused the velocity errors along the computer axes are shown in table 5.1.5-III. In addition, sensor and tracker errors obtained from a preliminary Error Coefficient Recovery Program (ECRP) run are presented.

[redacted]

The present best estimates of the guidance position and velocity errors at insertion are given in table 5.1.5-IV. These quantities were obtained from position and velocity comparisons using the best estimates of the tracker reference trajectory. In this table, the IMU error consists of sensor-induced state errors, whereas navigation errors result from various approximations within the airborne computer. Three estimates of the velocity error at SECO + 20 seconds were computed by backward integration of orbit data. The values varied from 3.9 ft/sec to 8.3 ft/sec along X, 5.0 ft/sec to 7.5 ft/sec along Y, and minus 3.2 ft/sec to plus 2.9 ft/sec along Z.

5.1.5.2.2 Orbital phase: Table 5.1.5-V summarizes the translation maneuvers performed during the mission, and a close agreement between planned and actual values is indicated for those maneuvers for which an attempt was made to be precise. The crew reported no trouble in zeroing the residuals for these maneuvers.

An analysis of the platform alignments prior to the terminal phase initiate (TPI) maneuver and prior to retrofire indicates accuracies within 0.3 of a degree in both pitch and roll—values commensurate with those noted during previous missions.

The ATMU operation was normal and was utilized twice during the mission—first, to load the catch-up/rendezvous module, and second, to load the touchdown-predict/reentry module.

Figure 5.1.5-4 contains relative trajectories for the M=3 rendezvous, as reconstructed from onboard radar data and from the best available state vectors. Because of the radar transponder malfunction, onboard backup procedures were used to compute the terminal phase maneuvers. Table 5.1.5-VI contains the rendezvous maneuvers computed using the various onboard and ground sources. The onboard computer was switched to the catch-up mode when the radar malfunction occurred, thereby inhibiting a computer solution for the rendezvous maneuvers. The values for the onboard computer solution in the table are those reconstructed postflight using telemetered radar data and a computer program simulation, and indicate the values which would have been available to the crew had the rendezvous mode been used. Figure 5.1.5-5 is a time history of radar and platform data during the rendezvous. Figure 5.1.5-6 is a time history of applied ΔV's, gimbal angles, and translation thruster firings prior to the first docking. The total ΔV expenditure from TPI to docking was 77.6 ft/sec. The ΔV expended for braking and line-of-sight control after the last midcourse correction was 34.4 ft/sec, including all station keeping between rendezvous and docking.

5.1.5.2.3 Reentry phase: The IGS operated properly throughout the retrofire and reentry phases of the mission. The total velocity change as a result of firing the retrorockets was 0.9 ft/sec lower than predicted. A comparison of the actual and planned velocity components can be found in table 5.1.5-V. The pitch attitude was held within 1.2 degrees, yaw within 3.4 degrees, and roll within 2.8 degrees. The total footprint shift due to equipment section jettison, pressure bleed-off, retrorocket deviation from preflight impulse prediction, and retrograde section jettison was approximately 22.4 nautical miles, as shown in figure 5.1.5-7.

From retrofire to an altitude of 400K feet, a 10-degree bank angle toward the south was flown as planned. At 94:20:12 g.e.t., the computer commanded a zero-degree bank angle. This indicated proper spacecraft navigation to the 400K-foot level when compared with the time of arrival at 400K feet as computed on the ground. From 400K feet until guidance initiate, the backup angle of 45 degrees toward the south was flown as planned. At 94:22:50 g.e.t., the spacecraft acceleration passed through a level of 1.0 ft/sec^2 or 0.03g (density-altitude factor of 8.73289) and the computer began to calculate the bank-angle commands necessary to guide the spacecraft to the target.

By 94:23:05 g.e.t., the crew had determined that the primary guidance system was functioning properly and had oriented the spacecraft to the attitude commanded by the computer. At 94:23:06 g.e.t., the control system was switched to REENTRY, and from this time until guidance termination the computer-commanded bank angles were automatically held by the control system to within two degrees. Guidance termination occurred at 94:28:24 g.e.t., which corresponded to the proper density-altitude factor of 4.025.

Time histories of bank angle commands and actual bank angles, crossrange error, and downrange error are presented in figure 5.1.5-8.

At guidance termination (80K feet), the IGS calculated position was 0.54 of a nautical mile southeast of the planned landing point. Radar tracking data placed the spacecraft at 70 degrees 1.4 minutes west longitude and 24 degrees 37.3 minutes north latitude at this time. This indicates a navigation error (between radar and IGS data) of 2.0 nautical miles. The guidance error was 2.3 nautical miles. The recovery forces reported that the position of the spacecraft at landing was 3.8 nautical miles from the planned landing point, while the miss distance computed from radar data was 2.6 nautical miles. Figure 5.1.5-7 shows these points relative to the reentry footprint. Table 5.1.5-VII contains a comparison of IGS and radar data and a breakdown of contributors to the miss distance.

5.1.5.3 Control system performance evaluation.-

5.1.5.3.1 Attitude control and maneuvering system: Performance of the attitude control and maneuvering system was satisfactory throughout the mission. The thruster problem encountered in the propulsion system degraded control capability somewhat and required the use of translation thrusters in some instances to maintain attitude control.

During the second docking attempt at 4 hours 49 minutes g.e.t., the rigidizing mechanism did not function. The crew attempted to complete the rigidizing sequence and, when unable to do so, they separated from the docking cone and imparted rates of about 3 deg/sec to the spacecraft in all three axes. This apparently led the crew to report that they may have had a control system problem. An analysis of the telemetry data showed that thrust chamber assembly (TCA) no. 4 was low in thrust before the docking. This caused the spacecraft to roll right and yaw right anytime that a yaw-right command was generated, and roll left and yaw left when a roll-left command was generated. However, this did not contribute to the rigidizing problem, since rates were nulled before docking and TCA no. 4 was not used in the docking sequence.

After separating the spacecraft from the docking cone, the rate command mode was selected, and, in attempting to maintain attitude control, yaw-right and roll-right commands were generated. These commands increased the roll rate to approximately 4 deg/sec. When the roll-right command was removed, the rate command mode responded correctly by commanding roll-left thrusters on, and the roll rate was approaching null when the crew switched to the direct mode. While in direct, the roll rate increased to approximately 10 deg/sec as the result of some yaw-right and roll-right commands generated by the crew. The crew then selected pitch-thruster roll logic and regained control of the spacecraft in the direct mode and subsequently docked at 5 hours 7 minutes g.e.t.

The control system functioned properly in all attitude control modes and it is believed that the low relative velocity caused a partial latching. Darkness and unawareness that TCA no. 4 was low in thrust led the crew to believe that they had a control system problem.

During the phasing maneuver at 61:47:48 g.e.t., the crew reported that the spacecraft attitudes and rates were divergent while the automatic rate-command attitude control mode was being used. An analysis of the telemetry data confirmed that rates and attitudes did diverge slightly, but control system performance was adequate within itself and the divergence was caused by a combination of the circuit-breaker configuration and the degradation of certain TCA's. The crew was able to adequately control the spacecraft in the direct mode. Figure 5.1.5-9 presents a plot of spacecraft rates and attitudes, hand controller activation, and thruster activity during this period, which has been divided into four basic time intervals for discussion. Periods of divergent rates and attitudes can be seen in the second and fourth basic intervals.

The initial conditions were as follows:

(a) TCA's no. 2 and no. 4 circuit breakers open because of no thrust from these thrusters

(b) TCA no. 7 operating at a low level of thrust

(c) Pitch logic selected, which uses TCA's no. 1 and no. 5 for roll right and TCA's no. 2 and no. 6 for roll left

(d) TCA no. 11 circuit breaker open and TCA no. 12 being used for yaw-right attitude control.

It can be seen in the figure that TCA no. 12 was used to obtain a yaw-right rate during intervals one and three. Each time it fired, the yaw rate increased since TCA no. 7 (yaw right and roll right) was low. When TCA no. 12 stopped firing, TCA's no. 7 and no. 8 continued to fire because the yaw hand controller was commanding yaw left and the yaw rate had exceeded the plus yaw deadband of 0.2 deg/sec. TCA's no. 1, no. 5, and no. 6 were pulsing on and off because of the roll-left and pitch-down disturbance torques caused by TCA's no. 7 and no. 8 unbalance and TCA no. 12 firing. Some activity on TCA no. 3 was also present but it was proper when spacecraft rates and yaw hand controller commands are considered.

During the second and fourth intervals, rates can be seen to be divergent in roll. This was caused by the following sequence of events: Initially there was a yaw-left rate which was less than that being commanded; therefore, TCA's no. 7 and no. 8 were firing. Suddenly, the yaw-left command was removed. This called for yaw-right thrust and TCA's no. 7 and no. 8 (yaw-left) stopped firing and TCA's no. 3 and no. 4 (yaw-right) were commanded on. However, the TCA no. 4 circuit breaker was open and only TCA no. 3 fired. This caused yaw-right and roll-right accelerations. TCA's no. 1 and no. 5 (roll right) stopped firing because of the roll right from TCA no. 3. When the roll rate reached the positive roll deadband, TCA's no. 2 and no. 6 (roll left) were commanded on, but the TCA no. 2 circuit breaker was open; therefore, only TCA no. 6 fired. This caused roll-left and pitch-up accelerations. When the pitch rate reached the positive deadband, TCA's no. 1 and no. 2 (pitch down) were commanded on, but the TCA no. 2 circuit breaker was open and only TCA no. 1 fired. This resulted in three TCA's—no. 1, no. 3, and no. 6—firing steadily, two of which (no. 1 and no. 3) are roll-right thrusters. Consequently, the roll rate continued to increase. Eventually, the yaw rate reached the negative deadband and TCA no. 3 should have stopped firing; however, a yaw-left disturbance was caused by both TCA's no. 1 and no. 6, and TCA no. 3 continued to fire (probably at a duty cycle of at least 10 percent) to compensate for this.

The net result was that the roll rate continued to increase until a stable condition was reached at some high roll-right rate. Commanding roll left produced no improvement since the only roll-left thruster available (TCA no. 6) was firing. Commanding roll in the direction of the roll divergence while allowing pitch to stabilize, then removing the roll command would have provided rate control in all axes. Another method would have been to alternate between pitch and yaw logic with the hand controller at null. The rate-command switching logic was operating properly, and the divergence was caused by a combination of certain thrusters being turned off, low thrust from some of the thrusters being used, and selection of pitch roll logic.

Retrofire was made in rate command with both the A-ring and the B-ring on. Following retrofire, the control mode was switched to pulse and the crew reported turning off the B-ring. Just before guidance initiate, the rate command mode was utilized. Reentry mode was used throughout the active guidance phase. The crew reported turning on the B-ring during this phase and leaving both rings on until powering down. After guidance termination, rate command mode was selected and left on until the spacecraft was powered down. The maximum rates experienced by the spacecraft prior to drogue deployment were approximately 4 deg/sec in pitch and 6 deg/sec in yaw, comparable to the rates observed on previous missions. Figure 5.1.5-10 is a time history of control parameters during the reentry phase.

5.1.5.3.2 Gravity-gradient stabilization: During the umbilical EVA, the pilot attached a 100-foot tether between the spacecraft and the GATV. At 46 hours 55 minutes g.e.t., the docked vehicles were pitched down to a local-vertical attitude using the GATV attitude control system. At 47:23:17 g.e.t., the spacecraft was undocked and backed away vertically from the GATV, deploying the tether. Figure 5.1.5-11 is a time history of the spacecraft attitudes, rates, and control system usage for most of the tether evaluation, beginning with the pitch down to the local vertical and ending with the tether jettison at 51:50:57 g.e.t.

The GATV was stabilized by the attitude control system during the initial part of the tether evaluation. The crew attempted to establish a position above the GATV and on the GATV radius vector, with an angular orbital rate equal to that of the GATV. Preflight analyses had indicated that the initial angular velocity of the tethered system required to establish gravity-gradient stabilization would be between minus 0.113 and plus 0.113 deg/sec with respect to the local vertical, with a closing rate of less than 0.2 ft/sec between the vehicles.

Because of problems with the spacecraft attitude control thrusters, the crew was hampered in establishing the desired conditions. By selective use of the spacecraft attitude and translation thrusters, the crew was successful in establishing conditions considered by them to be satisfactory for remaining on the tether during the darkness period beginning at 48 hours 14 minutes g.e.t., and possibly adequate to establish capture. The spacecraft thruster activity used in establishing the initial conditions and in making adjustments to the conditions is shown in figure 5.1.5-11. By about 48 hours 45 minutes g.e.t., the crew decided that capture had not been effected on the initial attempt, in view of the large amplitude of the libration angle, and again tried to establish the desired conditions. By this time, the crew was more familiar with the available spacecraft control capability and succeeded in establishing conditions closer to those desired. The last adjustment to the conditions was made at about 49 hours 10 minutes g.e.t. The GATV attitude control system was turned off at about 49 hours 25 minutes g.e.t. A disturbance in GATV attitude was noted by the crew as the vehicle adjusted to its preferred attitude with respect to the tether. This disturbance may have led indirectly to the large spacecraft pitch attitude excursion of about 300 degrees shown in figure 5.1.5-11, beginning at about 49 hours 25 minutes g.e.t. This spacecraft attitude excursion is probably attributable to temporary slack in the tether as the GATV rotated slightly from its previously controlled attitude, and is not indicative of an excursion of the tethered system. No GATV attitude data are available to substantiate this conclusion, but the conclusion is consistent with crew observations and is supported by the absence of spacecraft oscillations, which would have been present had the system been rotating.

Figure 5.1.5-11 shows that the spacecraft began stable oscillations about the tether at about 49 hours 42 minutes g.e.t. The period of the spacecraft oscillations was about eight minutes, and the amplitude was approximately ±30 degrees. This motion, which was sustained until 50 hours 18 minutes g.e.t., is superimposed on an oscillation of the tethered system. If the assumption is made that the mean value of the spacecraft motion corresponds to the tethered system attitude, it can be seen from the figure that the system motion had a period of about 60 minutes, and the amplitude was approximately ±50 degrees. These conditions were disturbed at 50 hours 18 minutes g.e.t. and it is probably not coincidental that a fuel cell purge was started at about that time and a urine dump was also performed. Both of these operations impart energy to the system.

Figure 5.1.5-11 shows that the spacecraft rotated through about 410 degrees in pitch at a slowly increasing rate, beginning at 50 hours 18 minutes g.e.t. and ending at about 50 hours 38 minutes g.e.t. At that time, a reduction of about 0.5 deg/sec in the spacecraft pitch rate, accompanied by smaller changes in roll and yaw rates, is shown by the figure. These rate changes are an apparent result of a tautening of the tether following the 410-degree rotation of the spacecraft, removing most of the rotational velocity imparted to the spacecraft during the fuel cell purge. Again, the large spacecraft attitude excursion does not represent motion of the tethered system. This conclusion is consistent with observations of the crew and with the absence of spacecraft oscillations.

The motion of both the spacecraft and the tethered system was stable subsequent to the spacecraft rate transients at 50 hours 38 minutes g.e.t. and remained so until tether jettison. However, the characteristics of the motions were changed slightly from those existing prior to 50 hours 18 minutes g.e.t. The period of the spacecraft angular motion remained at about eight minutes, but the amplitude was reduced from about ±30 degrees to about ±20 degrees. Assuming that the motion of the tethered system can be deduced from the mean values of the spacecraft motion, figure 5.1.5-11 shows that the period of the tethered system motion remained at about 60 minutes, but the amplitude was reduced to about ±25 degrees.

The exact mechanism by which the spacecraft attitude excursions at 49 hours 25 minutes g.e.t. and 50 hours 18 minutes g.e.t. were induced is not completely understood, nor is the system behavior during and immediately following these excursions. Analysis of these areas, and of the gravity-gradient stabilization evaluation in general, is continuing. The gravity-gradient-stabilization tether evaluation was terminated at 51:50:57 g.e.t., when the crew jettisoned the docking bar.

5.1.5.3.3 Horizon sensors: The horizon sensors performed satisfactorily during the mission. The data indicate that the horizon sensors provided a stable spacecraft attitude measurement for platform alignment. Several losses of track did occur due to known causes, which included (1) exceeding attitude limits, (2) spacecraft rendezvous and docking with the GATV, and (3) the sun falling within the horizon sensor field-of-view at sunset and sunrise.

The qualitative comparison of the primary horizon sensor (wide-band optics) to the secondary horizon sensor (narrow-band optics) was in agreement with the previous performance evaluation of similar horizon sensors used on Spacecraft 9. The narrow-band horizon sensor provided an expected reduction of 40 percent in the time the sun affected the horizon sensor attitude measurement. In addition, the narrow band horizon sensor provided increased attitude measurement stability in the presence of earth/space gradient distortion by atmospheric conditions.

5.1.5.4 Rendezvous radar transponder anomaly.- The rendezvous radar was switched from STBY to ON at 1:23:44 g.e.t. Lock-on was obtained immediately at a range of 235 nautical miles, and smooth tracking ensued until 2:19:28 g.e.t. (at a range of 64 nautical miles). At this time the radar lost lock and no radar data were accepted by the onboard computer for the next 21 minutes. Figure 5.1.5-5 contains a time history of radar and transponder parameters for this period. At 2:41:17 g.e.t., the onboard computer again began to accept radar data although the crew reported that analog displays in the cockpit continued to be erratic. This condition remained until rendezvous. Range data, read from the onboard computer, is utilized throughout the final phases of the rendezvous as backup information and to assist in braking. Throughout this period, GATV commands sent via the radar were received and executed; however, no message acceptance pulses (MAP's) were received in the spacecraft.

The radar was operated again during the tether evaluation in an attempt to provide additional data concerning the problem. The transponder RF power monitor indicated normal operation from 47 hours 39 minutes g.e.t. until 47 hours 45 minutes g.e.t., when the power dropped and the pre-rendezvous indications returned. As in the rendezvous phase, the computer accepted some radar data, radar commands were executed by the GATV, and no MAP's were received in the spacecraft.

Normal operation of the radar system is as follows: Upon interrogation by an RF pulse of sufficient amplitude from the rendezvous radar, the transponder replies with another RF pulse which is frequency shifted and delayed from the interrogating pulse. Prior to interrogation, the transponder is normally in the dipole antenna position and the high voltage power supply is off. Upon being interrogated, and when the sufficient-amplitude-detector threshold is exceeded, the high voltage power supply is turned on, the modulator is enabled, and the transponder then replies to each received pulse. If, after being received, a signal level falls below the sufficient-amplitude-detector threshold, the high voltage remains on and the receiver alternately searches the dipole and spiral antennas for a sufficient signal. The radar may also transmit messages to the transponder by pulse-position modulating the radar RF transmitted pulse. Upon receipt, the transponder processes the signal in its sub-bit detector and supplies the message as an input to the GATV programmer. The GATV programmer in turn supplies a MAP signal to the transponder. The transponder codes the reply signal to the radar by pulse-width modulating its transmitted response pulse. The transponder return is normally a 6-microsecond RF pulse. When indicating a message acceptance pulse to the radar, this return pulse is increased to 10 microseconds. The radar receiver, therefore, determines message acceptance by looking for a transponder return pulse in excess of six microseconds. The radar angle tracking circuitry is mechanized in such a way that the spacecraft-to-GATV elevation angle is determined by interrogating the 4 to 6-microsecond period of the transponder return pulse. Azimuth angle is determined by interrogating the 2 to 4-microsecond period of the return pulse, while the leading edge supplies range and range rate information.

The fact that the GATV properly executed radar transmitted commands (similar to the anomaly during the Gemini XI mission) tends to absolve the radar transmitter and the transponder receiver. The transponder RF power indication, coupled with the fact that range data were received by the spacecraft, tends to absolve the radar receiver and implicate the transponder transmitter. The inability to receive MAP's and the noisy angle data are indicative of improper transmitted pulse characteristics such as those which would result from arcing in the transponder transmitter assembly and cables. The most likely cause of the problem is in the modulator, the transmitter assembly, the RF cable connecting the transmitter to the isolator, the isolator, or the high voltage section. Loss in pressure of a sealed component to critical arcing conditions was the probable cause.

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

Ground elapsed time, hr:min:sec	Event	Component status					Remarks
		ACME	Computer	IMU	Horizon sensor	Radar	
0:00:00	Lift-off	Rate command	Ascent	Free	Primary (covers on)	Off	20:46:33 G.m.t.
0:05:43.5	SECO	Rate command	Ascent	Free	Primary (covers on)	Off	91FS2 indication
0:06:06.6	Spacecraft/ GLV separation	Direct	Ascent	Free	Primary (covers on)	Off	Spacecraft separation bilevel. Aft thrusters fired at 0:06:06.4 g.e.t. for 2.7 seconds
0:06:07.8	Jettison horizon sensor and radar covers	Direct	Ascent	Free	Primary	Off	
0:06:43.6	IVAR maneuver	Rate command	Ascent	Free	Primary	Off	ΔV = 30.6 ft/sec
0:08:25	Align platform	Platform	Ascent	SEF	Primary	Off	
0:09:28	Load Module III-A	Platform	Prelaunch	SEF	Primary	Off	
0:49:39.9	Phase adjust maneuver	Platform	Catch-up	Orbit rate	Primary	Off	ΔV = 66.7 ft/sec
1:14:22.3	Plane change maneuver	Rate Command	Catch-up	Orbit rate	Primary	Off	ΔV = 8.5 ft/sec
1:23:45	Radar lock-on	Pulse	Catch-up	Orbit rate	Primary	On	Range = 235 nautical miles
1:47:52.3	Corrective combination maneuver	Platform	Catch-up	Orbit rate	Secondary	On	ΔV = 8.4 ft/sec Range = 135.4 nautical miles

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

Ground elapsed time, hr:min:sec	Event	Component status					Remarks
		ACME	Computer	IMU	Horizon sensor	Radar	
2:19:27.6	Loss of radar lock-on	Platform	Rendezvous	Orbit rate	Secondary	On	RF power monitor shows variations
2:22:54.5	Coelliptic maneuver	Platform	Catch-up	Orbit rate	Secondary	On	ΔV = 49.9 ft/sec Range = 64.5 nautical miles
3:05:47.3	Terminal phase initiate	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 21.8 ft/sec Range = 21.8 nautical miles
3:11:14.3	First correction	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 1.7 ft/sec
3:17:07.1	Second correction	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 3.1 ft/sec
3:23:45.9	Third correction	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 1.4 ft/sec
3:29:05.2	Fourth correction	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 4.9 ft/sec
3:32:36	Terminal phase finalize and braking	Rate command	Catch-up	Orbit rate	Secondary	On	ΔV = 19.7 ft/sec
3:46:--	Station keeping	Rate command and pulse	Catch-up	Orbit rate	Secondary	On	Crew air-to-ground reported time
4:13:29.4	First docking	Rate command	Catch-up	Orbit rate	Off	Off	Contact with docking cone; transient seen in spacecraft rate gyro signals
4:31:22.2	First undocking	Pulse	Catch-up	Orbit rate	Off	Off	GATV data

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

Ground elapsed time, hr:min:sec	Event	Component status					Remarks
		ACME	Computer	IMU	Horizon sensor	Radar	
4:49:43.8	Second docking attempted	Rate command	Off	Orbit rate	Off	Off	Contact made but spacecraft not fully latched
5:07:14.1	Second docking	Direct	Off	Orbit rate	Off	Off	Contact with docking cone; transient in spacecraft rates
5:10:--	Crew reported a possible control system problem	Pulse	Off	Orbit rate	Off	Off	
6:03:04.1	Second undocking	Direct	Off	Orbit rate	Off	Off	Undocking time from GATV data
6:07:54.7	Third docking	Rate command	Off	Orbit rate	Off	Off	Contact with docking cone; transient in spacecraft rates
7:05:06	Eclipse phasing no. 1 (SPS)	Rate command (OAMS off)	Catch-up	Orbit rate	Off	Off	ΔV = 43.8 ft/sec
7:45:--	Power down spacecraft	Off	Off	Off	Off	Off	Approximate time
15:16:18	Eclipse phasing no. 2 (SPS)	Off	Off	Off	Off	Off	
19:22:20	OAMS power on	Pulse	Off	Off	Off	Off	
19:29:--	Standup EVA	Pulse, direct, and rate command	Off	Off	Off	Off	19:29 to 21:58
21:37:48	OAMS power off	Rate command	Off	Off	Off	Off	

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

Ground elapsed time, hr:min:sec	Event	Component status					Remarks
		ACME	Computer	IMU	Horizon sensor	Radar	
22:36:56	Power down	Off	Off	Off	Off	Off	
38:53:--	Crew checks thrusters 2 and 4	Direct	Off	Off	Off	Off	Thrusters 2 and 4 give no thrust
42:48:--	Umbilical EVA	Off	Off	Orbit rate	Off	Off	42:48 to 44:54
44:48:07	Pilot visually observed OAMS thruster firing	Direct	Off	Orbit rate	Off	Off	Observed difference in plumes from thrusters 4 and 5
47:23:16.6	Third undocking and begin tether evaluation	Direct	Catch-up	Orbit rate	Off	On	
51:50:57	Tether jettison	Pulse	Off	Orbit rate	Off	Off	
52:08:00	Align platform	Platform	Rendezvous and catch-up	SEF	Secondary	Off	
52:14:27	Spacecraft/GATV separation	Platform	Catch-up	Orbit rate	Secondary	Off	ΔV = 7.1 ft/sec
52:29:00	Power down	Off	Off	Off	Off	Off	Air-to-ground time
61:42:40	Align platform	Platform, pulse, and direct	Catch-up	SEF	Secondary	Off	
61:47:47.9	Phasing maneuver	Rate command	Catch-up	Orbit rate	Secondary	Off	ΔV = 5.2 ft/sec

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

Ground elapsed time, hr:min:sec	Event	Component status					Remarks
		ACME	Computer	IMU	Horizon sensor	Radar	
66:05:--	Standup EVA	Platform	Off	Orbit rate	Secondary	Off	66:05: to 67:01
68:01:00	Crew checked thruster no. 8	Pulse	Off	Orbit rate	Primary	Off	Thruster no. 8 gives no thrust
68:50:--	Retrofire update	Pulse	Prelaunch	Orbit rate	Primary	Off	
75:30:--	Power down	Off	Off	Off	Off	Off	Air-to-ground time
88:20:12	OAMS test	Direct and pulse	Off	Off	Off	Off	Thrusters 2 and 7 give low thrust; 4 and 8 not tested properly
91:47:00	Platform alignment	Platform	Off	BEF	Secondary	Off	
92:02:--	Accelerometer bias check	Platform	Prelaunch	BEF	Secondary	Off	
93:56:00	Preretrofire alignment complete	Pulse	Reentry	BEF	Secondary	Off	
93:59:57.7	Retrofire	Rate command	Reentry	Free	Secondary	Off	ΔV = 321.5 ft/sec
94:20:11.9	400K feet	Pulse	Reentry	Free	Off	Off	
94:22:04	Begin black-out	Pulse	Reentry	Free	Off	Off	
94:22:50.3	Guidance initiation	Reentry rate command	Reentry	Free	Off	Off	
94:23:05.6	Select automatic reentry mode	Reentry	Reentry	Free	Off	Off	

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

Ground elapsed time, hr:min:sec	Event	Component status					Remarks
		ACME	Computer	IMU	Horizon sensor	Radar	
94:27:25	End blackout	Reentry	Reentry	Free	Off	Off	
94:28:24.4	Guidance termination	Reentry	Reentry	Free	Off	Off	
94:29:10	Drogue deploy	Rate command	Reentry	Free	Off	Off	
94:30:38	Main para-chute deploy	Rate command	Reentry	Free	Off	Off	ACME off at 94:31:50, computer off at 94:31:56
94:34:30	Landing	Off	Off	Off	Off	Off	

[redacted]

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

Data source	Inertial velocity, ft/sec	Inertial flight-path angle, deg	Inertial velocity components (computer coordinates), ft/sec		
			X	Y	Z
Flight plan	25 728	0.000			
IGS	25 717	+0.025	25 287	4682	-156
Preliminary best estimate trajectory	25 713	+0.028	25 284	4677	-155
MISTRAM 100K	25 710	+0.022	25 281	4679	-155
GE MOD III (final)	25 710	+0.034	25 282	4674	-155
GE/Burroughs	25 711	+0.026	25 282	4677	-155
GE MOD III (real time)	25 701	+0.08			
MISTRAM IP	25 714	+0.04			

[redacted]

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

Error source	Specification value	Engineering estimates				TRW's 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		N				(a)			
Y_p-gyro		N				(a)			
Z_p-gyro		0.036			0.5	(a)			
g-sensitive drift	0.5 deg/hr/g					deg/hr/g			
X_p-gyro spin-axis unbalance		N				(a)			
Y_p-gyro spin-axis unbalance		N				(a)			
Z_p-gyro spin-axis unbalance		0.16			-1.3	(a)			
X_p-gyro input-axis unbalance		0.15			-1.8	(a)			
Y_p-gyro input-axis unbalance		N				0.4 ±0.23	1.4	11.0	
Z_p-gyro input-axis unbalance		-0.07			-2.4	-0.183 ±0.26			-6.3
Acceleration bias	300 ppm	ppm				ppm			
X_p		N				8 ±10	N	0.2	
Y_p		N							
Z_p		N				54 ±10		0.6	

[a]Not included in TRW model.

N = negligible.

[redacted]

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

Error source	Specification value	Engineering estimates				TRW's Error Coefficient Recovery Program estimates			
		Error	Velocity error, ft/sec			Error	Velocity error, ft/sec		
Accelerometer scale factor	360 ppm	ppm	X	Y	Z	ppm	X	Y	Z
X_p		(Undetermined)				1200 ±200 (Pre-BECO) -300 ±100 (Post-BECO)	10.0 -7.4		
Y_p		N				(a)			
Z_p		-70		0.5		-60 ±20	0.4		
Misalignments		Arc seconds				Arc seconds			
Azimuth misalignment	60 arc seconds	34			4.1	35 ±96			4.2
Pitch misalignment	100 arc seconds	40		4.8		-24.6 ±71		-3.0	
Initialization error (data processing error)		N				(a)			
Time bias		N				-6.4 ±4.2	-1.3	-0.4	
IGS time scale factor	50 ppm	-70	-5.3	-1.4		(a)			

External tracker errors

System	Range bias, ft	P-bias, ft	Q-bias, ft	Azimuth, radians	Elevation, radians	Refraction, n units
GE MOD III (final)	-27 ±11	N/A	N/A	4.4×10^{-6} ±20×10^{-6}	4×10^{-6} ±53×10^{-6}	2 + 1
MISTRAM 100K		-1 (Pre-BECO) 0 (Post-BECO)	1.8 (Pre-BECO) 4.0 (Post-BECO)			

[a] Not included in TRW model.

N = negligible.

N/A = not applicable.

[redacted]

[redacted]

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

Error	Position, ft			Velocity, ft/sec		
	X	Y	Z	X	Y	Z
IMU	2250 ±200	850 ±200	215 ±50	3.0 ±3.0	5.3 ±3.0	-0.9 ±1.0
Navigation	200	-50	35	-0.7	-0.3	-0.2
Total guidance	2450 ±200	800 ±200	250 ±50	2.3 ±3.0	-5.0 ±3.0	-1.1 ±1.0

[redacted]

TABLE 5.1.5-V.- TRANSLATION MANEUVERS

Maneuver	Ground elapsed time, hr:min:sec	ΔV_X, ft/sec	ΔV_Y, ft/sec	ΔV_Z, ft/sec	ΔV total, ft/sec	ΔV real-time planned, ft/sec
Tail-off	0:06:02.1	+67.2	+15.8	-3.1	69.0	
Separation	0:06:06.6	+3.2	+0.1	-0.2	3.2	
IVAR	0:06:43.6	+28.2	-7.1	+9.6	30.6	
M=3 rendezvous						
Phase adjust maneuver	0:49:39.9	+66.7	-0.3	+0.2	66.7	66.6
Plane change maneuver	1:14:22.3	-0.2	+0.3	-8.4	8.5	7.4
Corrective combination maneuver	1:47:52.3	+7.4	-3.9	-1.0	8.4	7.6
Coelliptic maneuver	2:22:54.5	+49.5	-6.4	-0.2	49.9	49.9
Terminal phase initiate maneuver	3:05:47.3	+19.4	-10.1	+0.5	21.8	23.2
First midcourse correction	3:11:14.3	-0.1	-1.7	+0.1	1.7	
Second mid-course correction	3:17:07.1	-1.4	-2.8	-0.1	3.1	
Third midcourse correction	3:23:45.9	+0.3	+0.5	+1.2	1.4	
Fourth mid-course correction	3:29:05.2	+1.1	+4.8	-0.2	4.9	

TABLE 5.1.5-V.- TRANSLATION MANEUVERS - Concluded

Maneuver	Ground elapsed time, hr:min:sec	ΔV_X, ft/sec	ΔV_Y, ft/sec	ΔV_Z, ft/sec	ΔV total, ft/sec	ΔV real-time planned, ft/sec
Terminal phase finalize	3:32:35.9	+11.2	-15.9	-2.3	19.7	28.4
Eclipse phasing no. 1 (docked) (GATV SPS)	7:05:06	-43.7	+1.9	+2.6	43.8	43.0
Eclipse phasing no. 2 (docked) (GATV SPS)	15:16:18	Computer and platform were off				15.0
Separation	52:14:27.0	+7.1	+0.2	-0.1	7.1	6.0
Phasing maneuver	61:47:47.9	-5.1	+0.2	+0.6	5.2	5.5
Retrofire	93:59:57.7	-300.8	-113.5	+3.6	321.5	

TABLE 5.1.5-VI.- RENDEZVOUS MANEUVERS

[All values in ft/sec]

Maneuvers	Ground backup	Computer closed loop[a]	Onboard backup	Intended to apply	Actually applied
Terminal phase initiate (TPI)	22.8 fwd 3.2 dn 2.7 rt	18 fwd 5 dn 8 lt	22.0 fwd 3.0 up 0.0 rt/lt	22.0 fwd 0.0 up 0.0 rt/lt	21.8 fwd 1.3 up 0.5 lt
First midcourse correction	Not computed	2 aft 4 dn 4 lt	0.0 fwd/aft 1.5 up 0.0 rt/lt	0.0 fwd/aft 1.0 up 0.0 rt/lt	1.0 aft 1.4 up 0.1 lt
Second midcourse correction	Not computed		0.0 fwd/aft 2.0 up 0.0 rt/lt	0.0 fwd/aft 2.0 up 0.0 rt/lt	1.2 fwd 3.1 up 0.1 rt
Third midcourse correction	Not computed	9 aft 3 dn 1 lt	0.0 fwd/aft 0.0 up/dn 1.0 lt	0.0 fwd/aft 0.0 up/dn 1.0 lt	0.3 aft 0.5 dn 1.2 lt
Fourth midcourse correction	Not computed		5.0 aft 1.0 dn 0.0 rt/lt	5.0 aft 1.0 dn 0.0 rt/lt	4.8 aft 1.1 dn 0.2 rt

[a]Solutions reconstructed postflight, assuming that the computer START had been depressed at 3:05:44 g.e.t. TPI would have been applied at 3:09:44 g.e.t., and the two corrections would have been applied at approximately 3:21:44 and 3:33:44 g.e.t.

TABLE 5.1.5-VII.- COMPARISON OF IGS AND RADAR DATA

(a) Spacecraft position data during reentry

Event	Planned		Actual (radar BET)		Miss distance (planned minus actual), n. mi.	IGS		Navigation error (IGS minus actual), n. mi.
	Longitude, deg	Latitude, deg	Longitude, deg	Latitude, deg		Longitude, deg	Latitude, deg	
Retrofire	-177.646	5.367	-177.588	5.402	3.86	-177.646	5.367	3.86
Guidance initiation	-87.319	28.069	-87.704	28.118	26.59	-87.710	28.117	0.33
Guidance termination	-70.050	24.598	-70.023	24.621	2.99	-70.009	24.579	2.01
Drogue deploy	-70.000	24.583	-69.952	24.591	2.68	-69.944	24.530	2.44

(b) State vectors used for comparison at retrofire

(Earth-centered inertial coordinates)

	BET (TRW)	IGS (RTCC)	IGS minus BET (initial-condition error)
X, ft	-6 193 335.4	-6 997 200	3 864.6
Y, ft	20 609 907	20 608 200	-1 707
Z, ft	2 042 140.3	2 043 600	1 459.7
V_X, ft/sec	-20 768.3	-20 767.2	-1.1
V_Y, ft/sec	-8 223.7	-8 228.1	4.4
V_Z, ft/sec	12 026.7	12 026.4	-0.3

(c) Contributors to IGS/BET difference at guidance termination

	Latitude, n. mi.	Longitude, n. mi.	Total
Initial alignment error at retrofire			
X (correlated with Z)			
Y = 0.19 deg	Negligible	0.14	
X and Z = 0.75 deg	-0.9	Negligible	
Total			
Update initialization	Negligible	Negligible	
Total, alignment and initialization	-0.9	0.14	0.91
Other (gyro, accelerometer, and timing)	-0.6	+0.19	0.62
Total	-1.5	0.33	1.53

[redacted]

NASA-S-66-11291 DEC 8

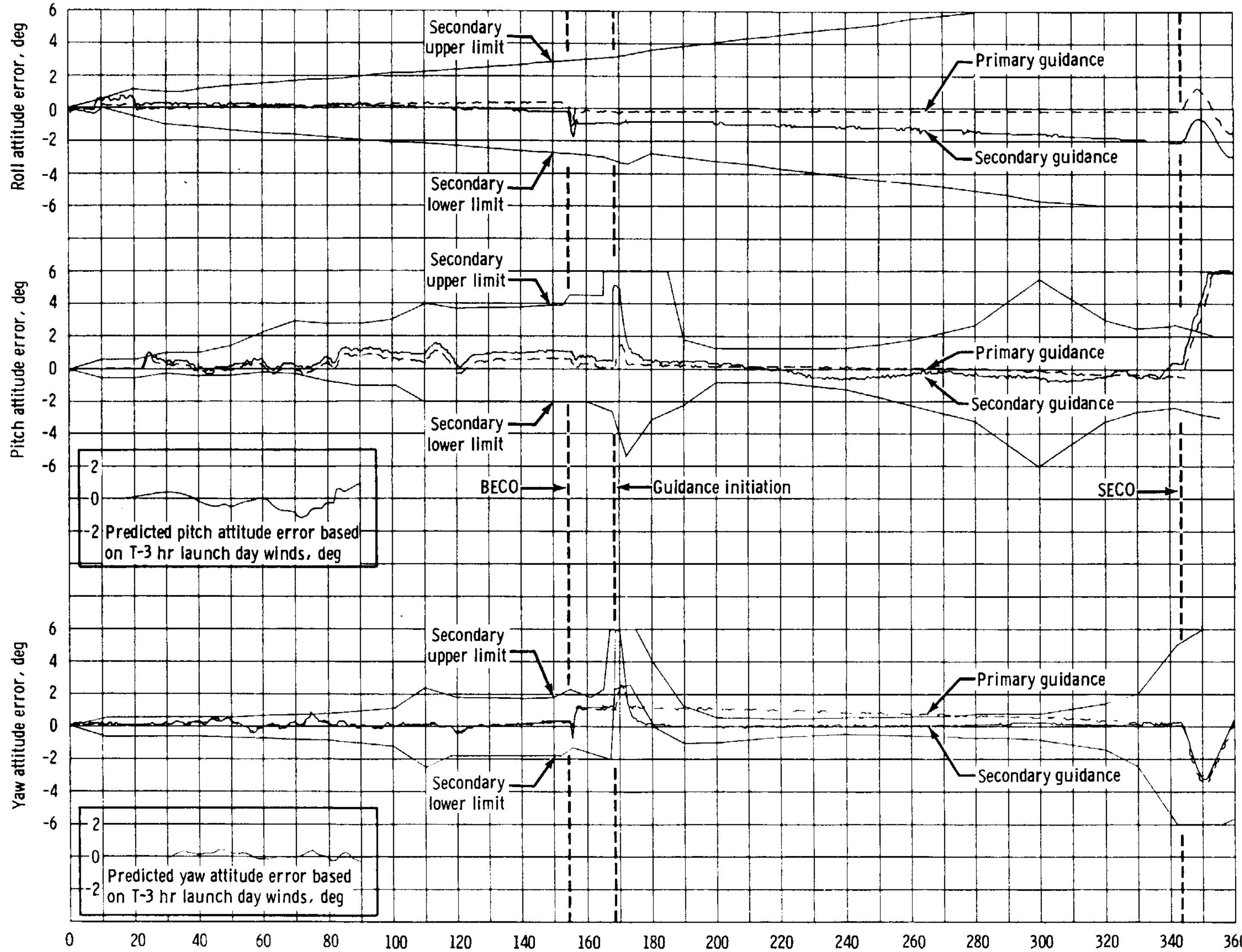
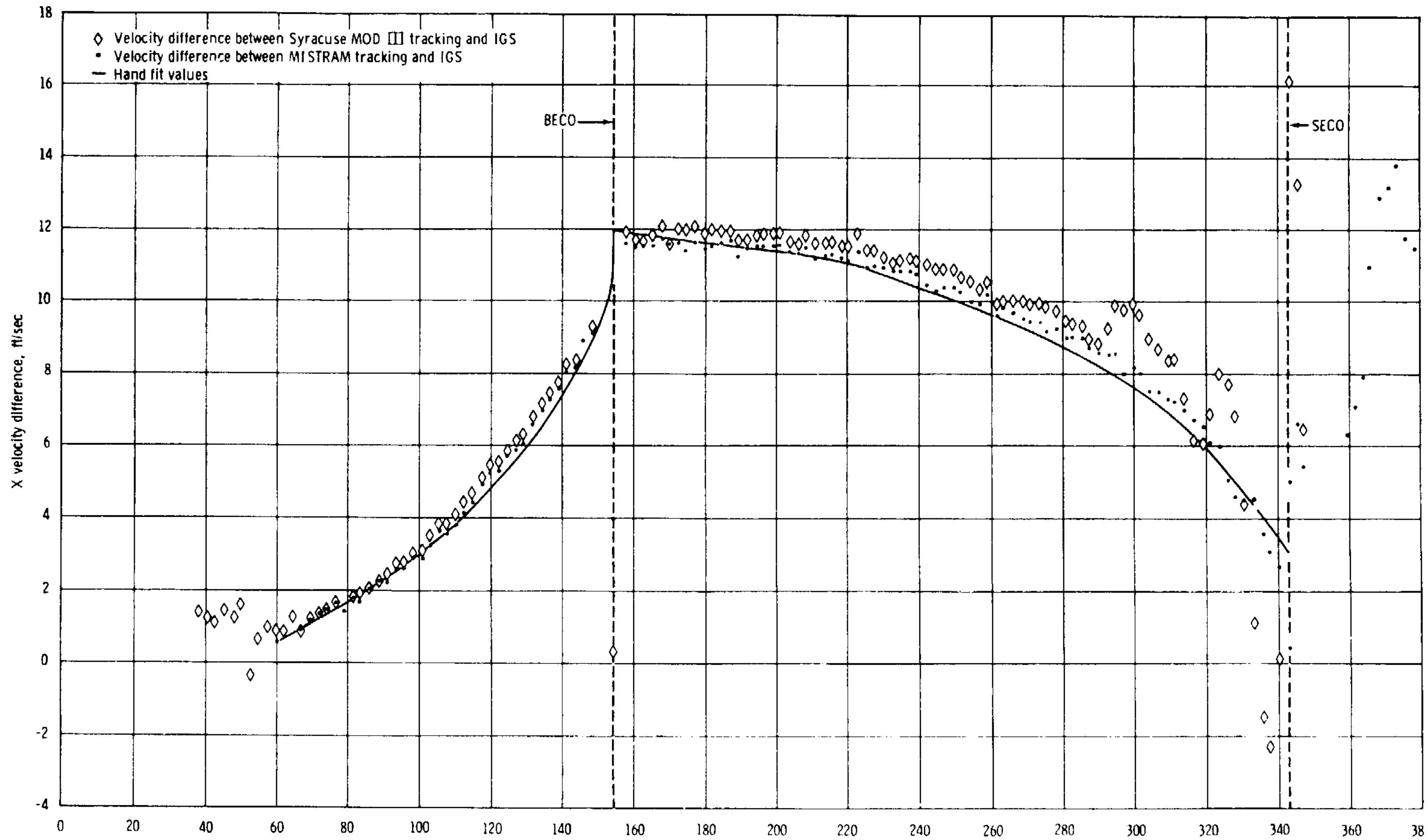


Figure 5.1.5-1.- Comparisons of launch vehicle and spacecraft steering error.

[redacted]

[redacted]

NASA-S-66-11341 DEC 16

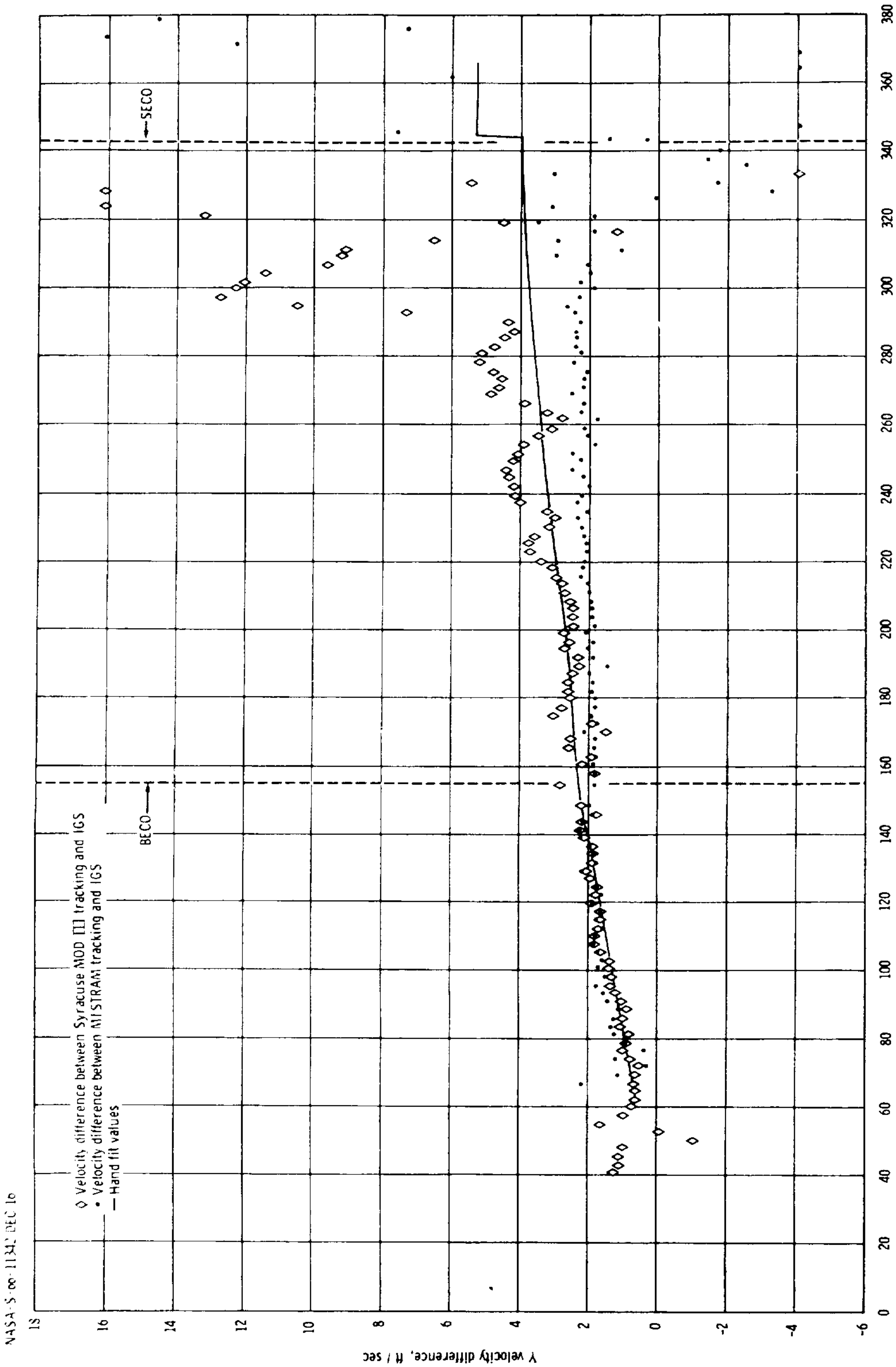


(a) Downrange velocity.

Figure 5.1.5-2. - Comparisons of spacecraft IGS and radar tracking velocities.

[redacted]

[redacted]

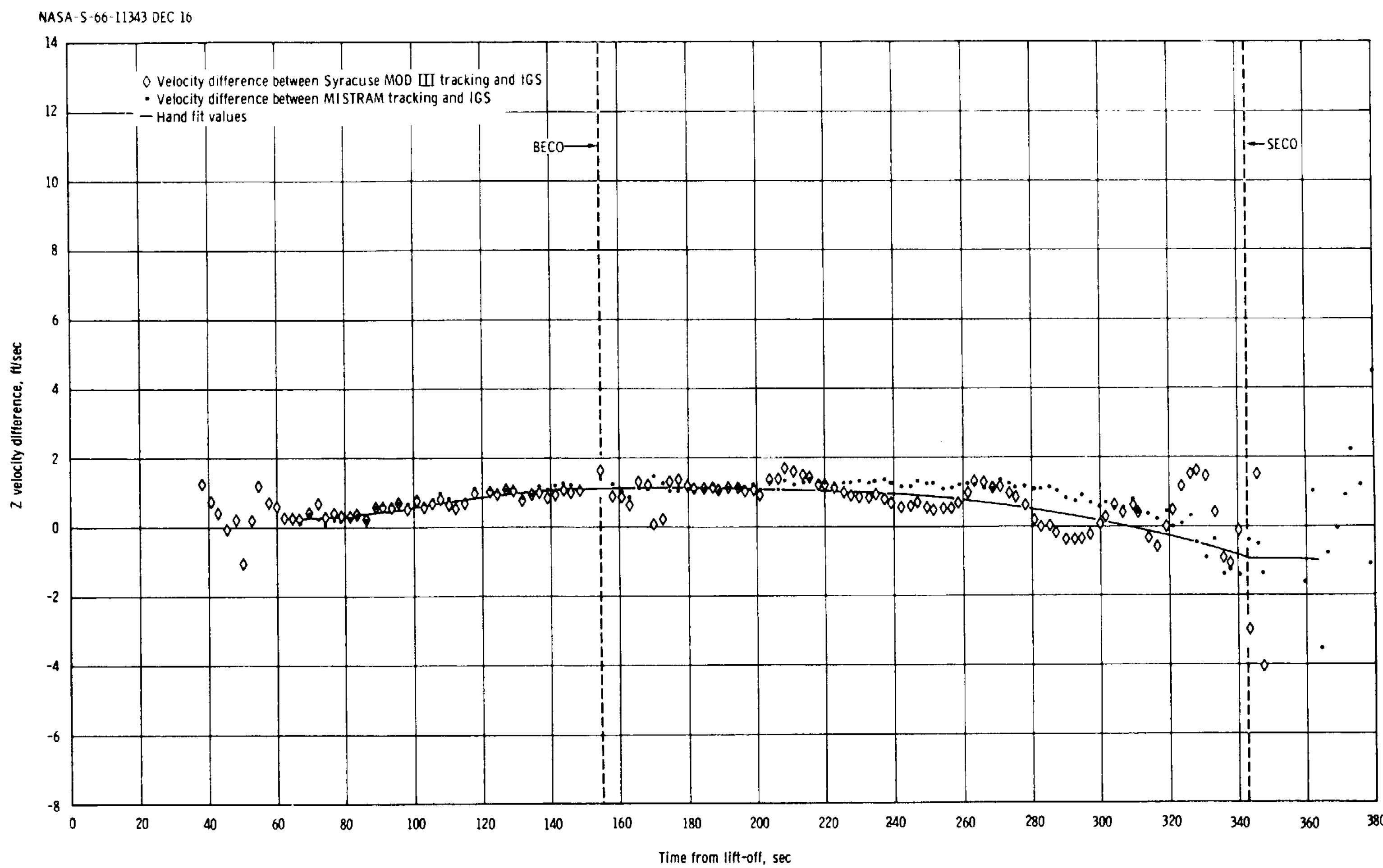


(b) Vertical velocity.

Figure 5.1.5-2. - Continued.

[redacted]

[redacted]

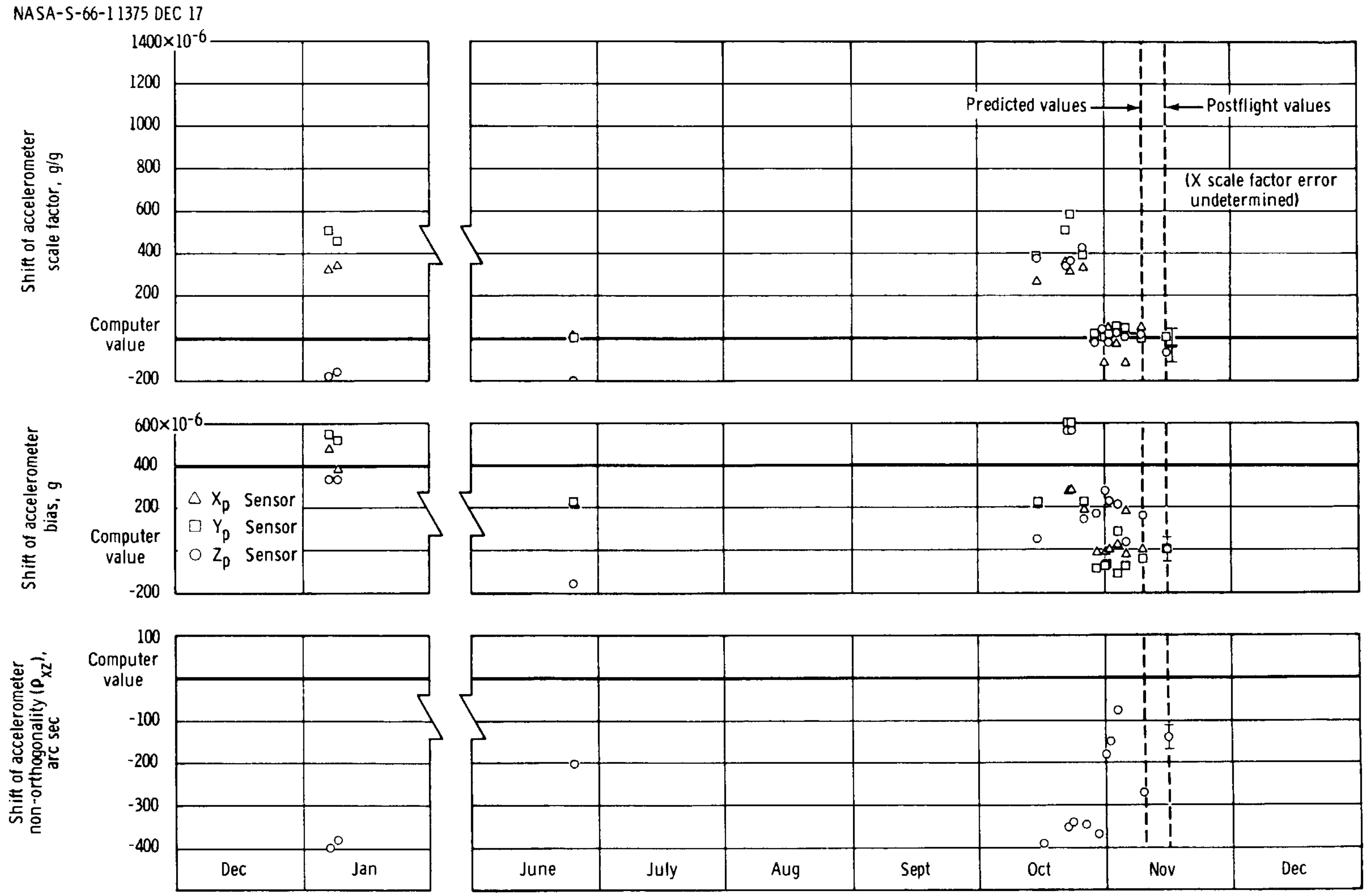


(c) Crossrange velocity.

Figure 5.1.5-2. - Concluded.

[redacted]

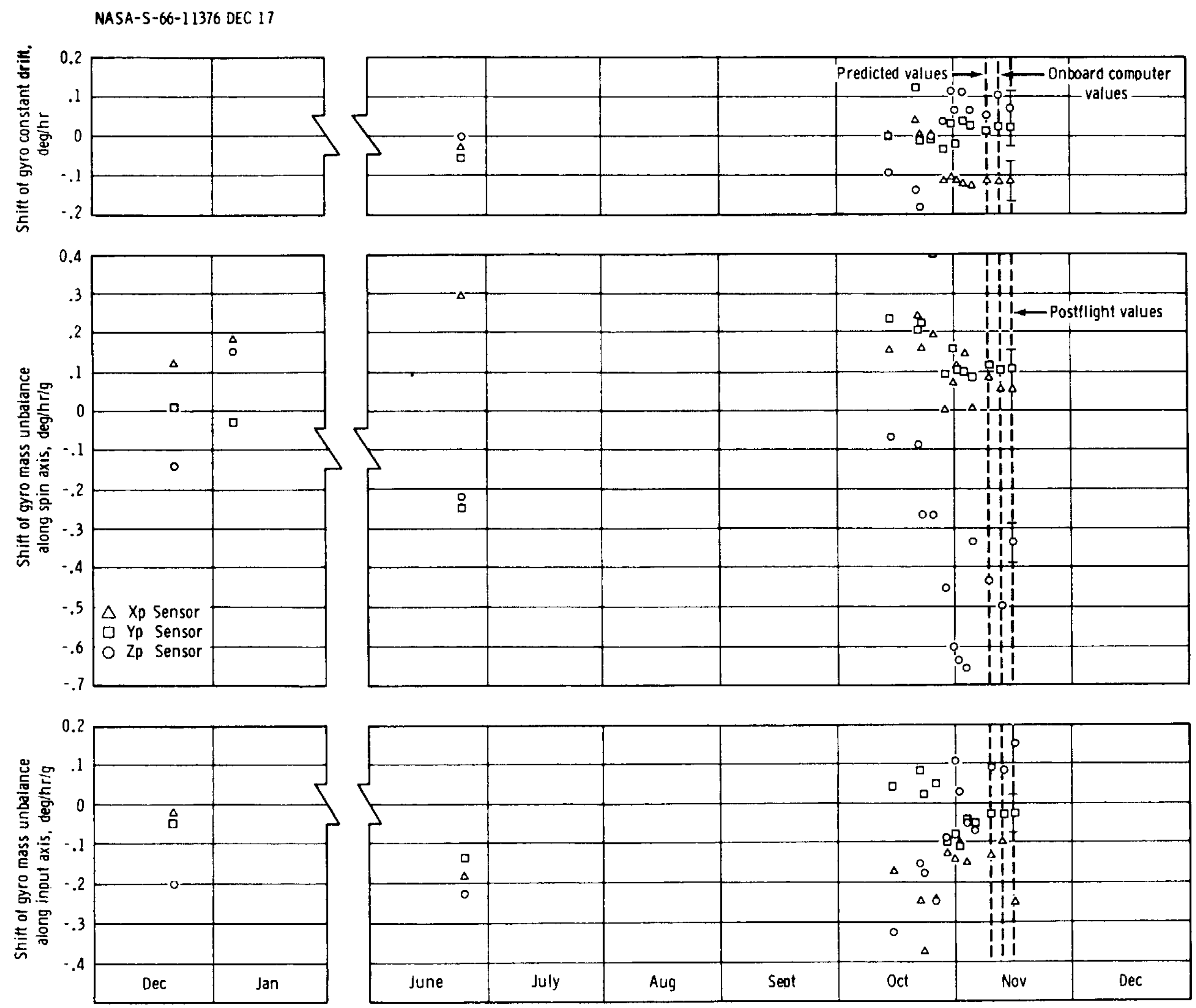
[redacted]



(a) Shift of accelerometer biases and scale factors.

Figure 5.1.5-3. - IMU error coefficient history.

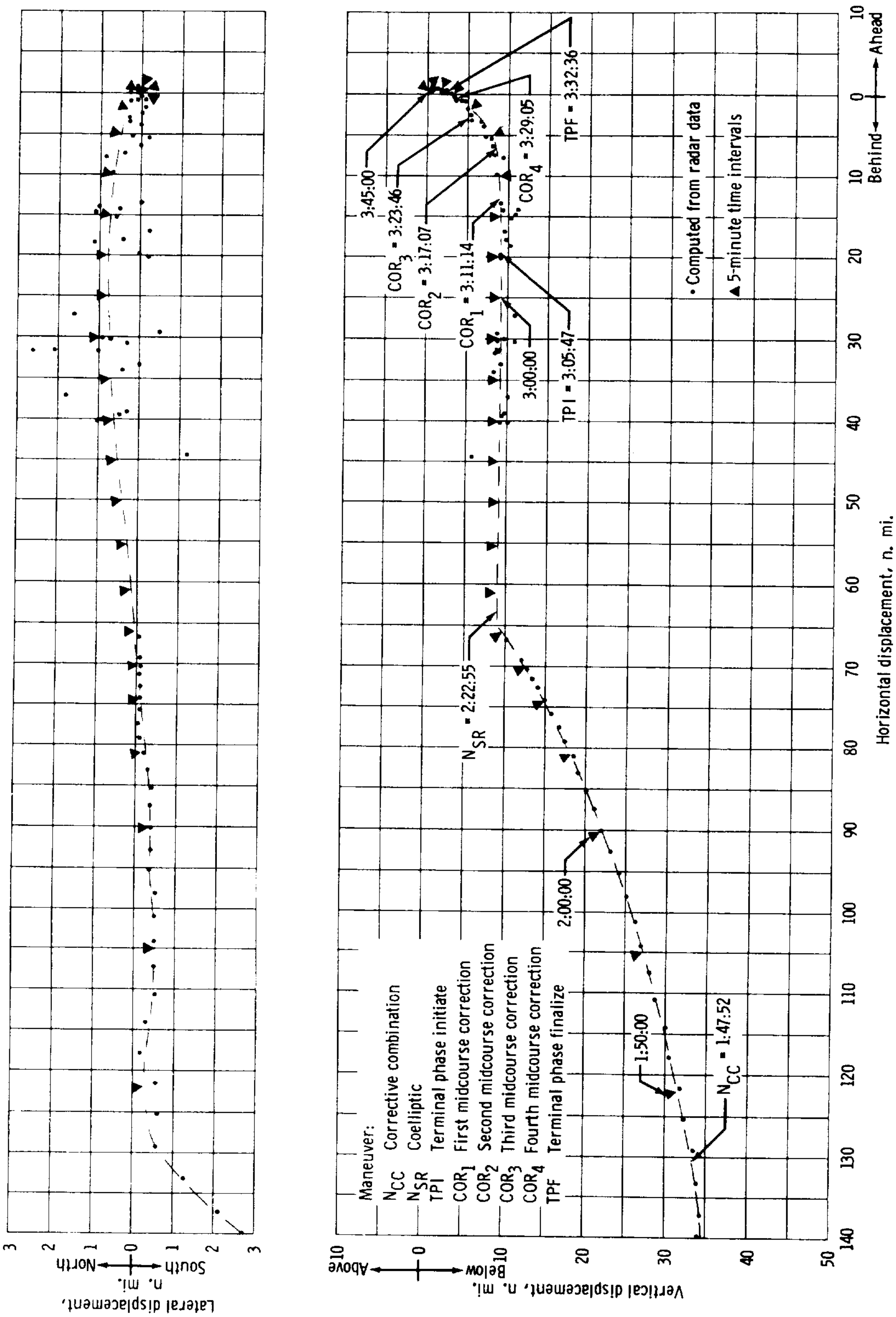
[redacted]



(b) Shift of gyro mass unbalances and constant drifts.

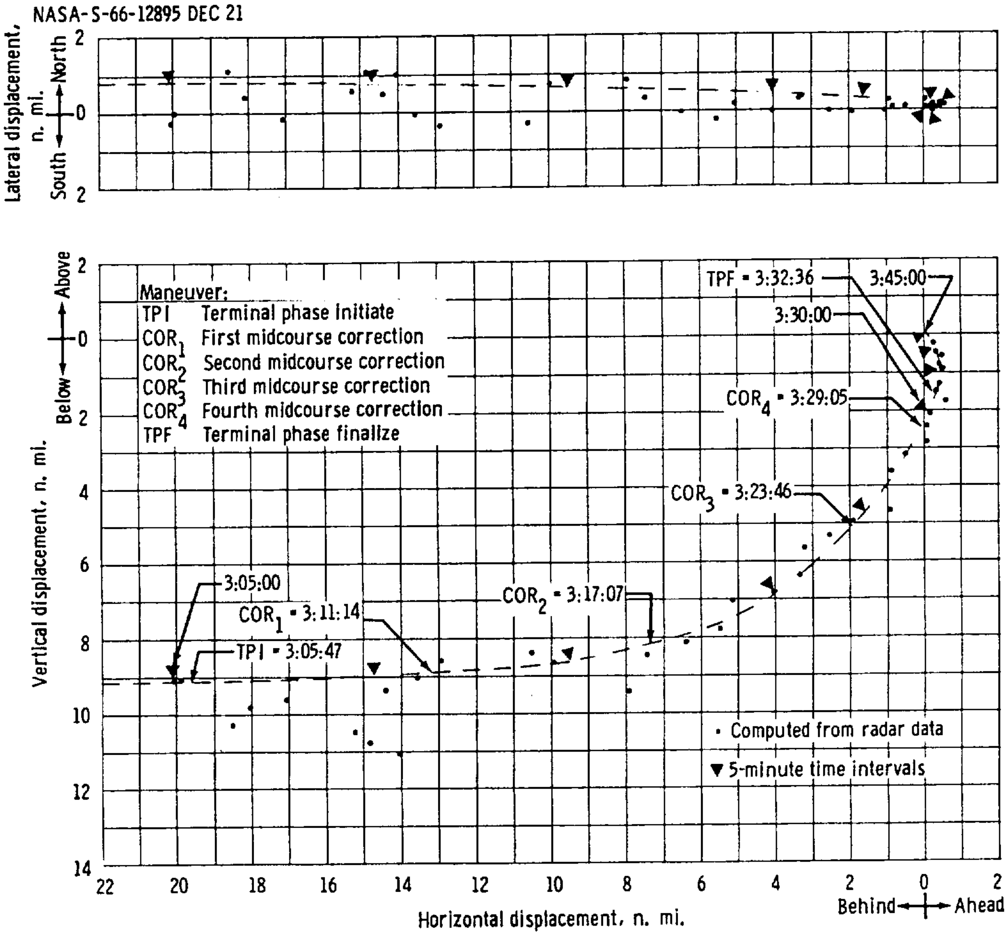
Figure 5.1.5-3. - Concluded.

NASA-S-66-12894 DEC 21

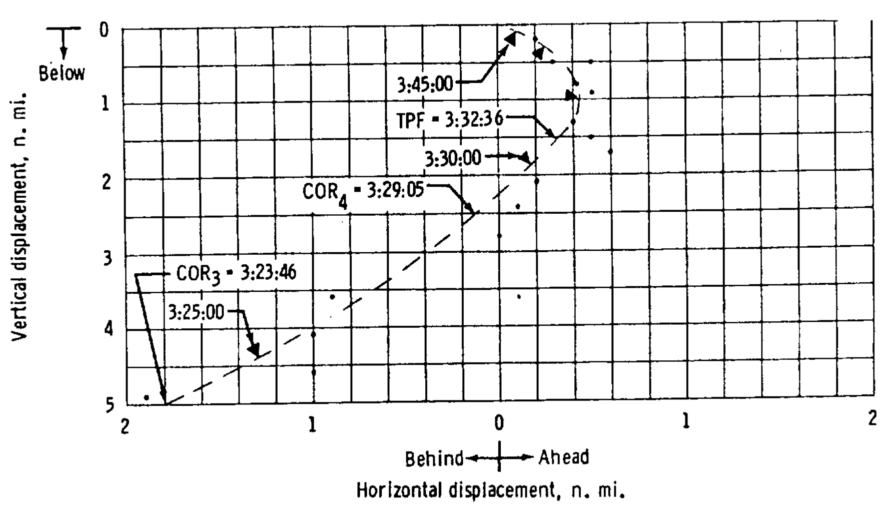


(a) N_{CC} through TPF.

NASA-S-66-12895 DEC 21

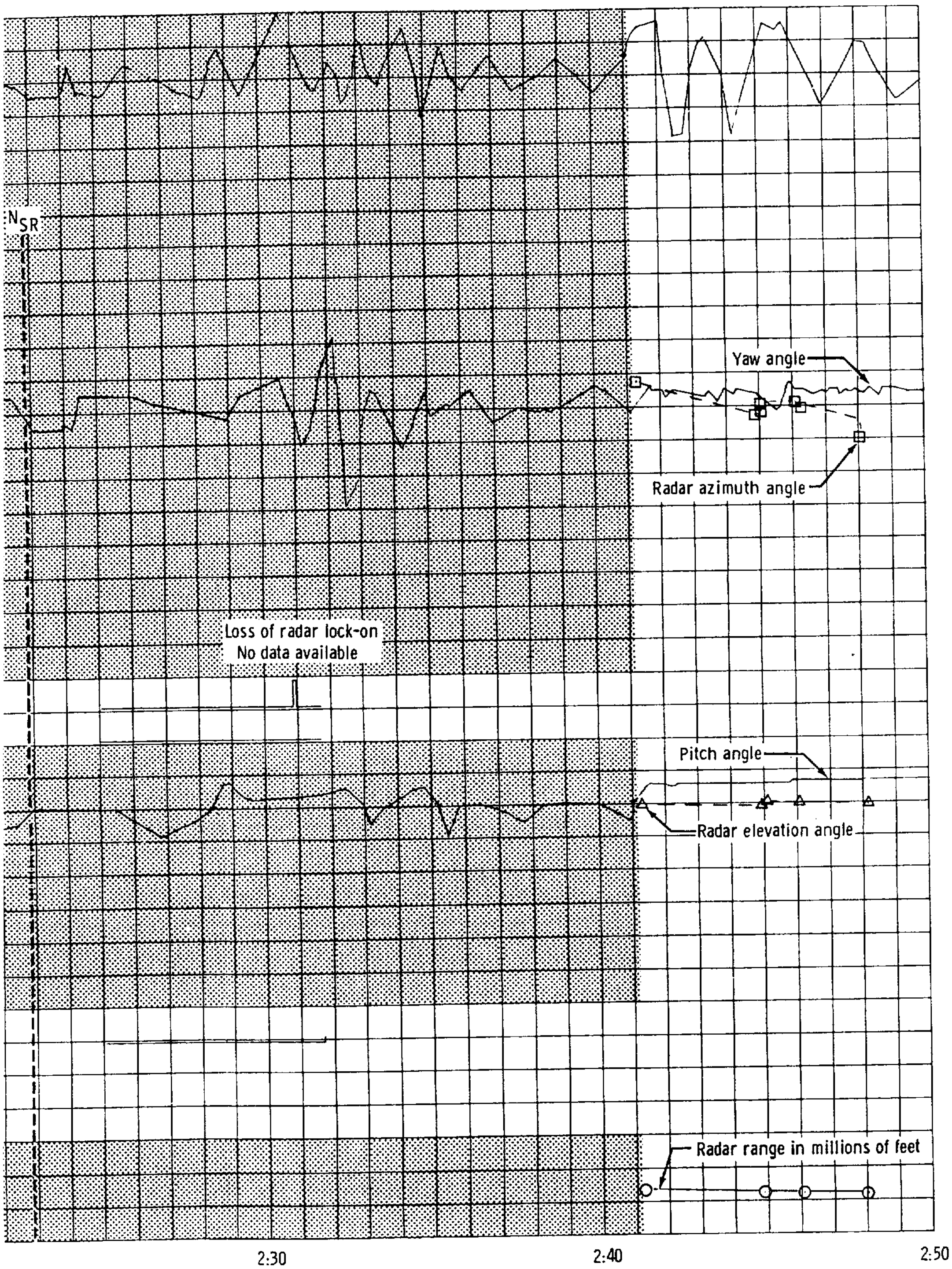


(b) TPI through TPF.

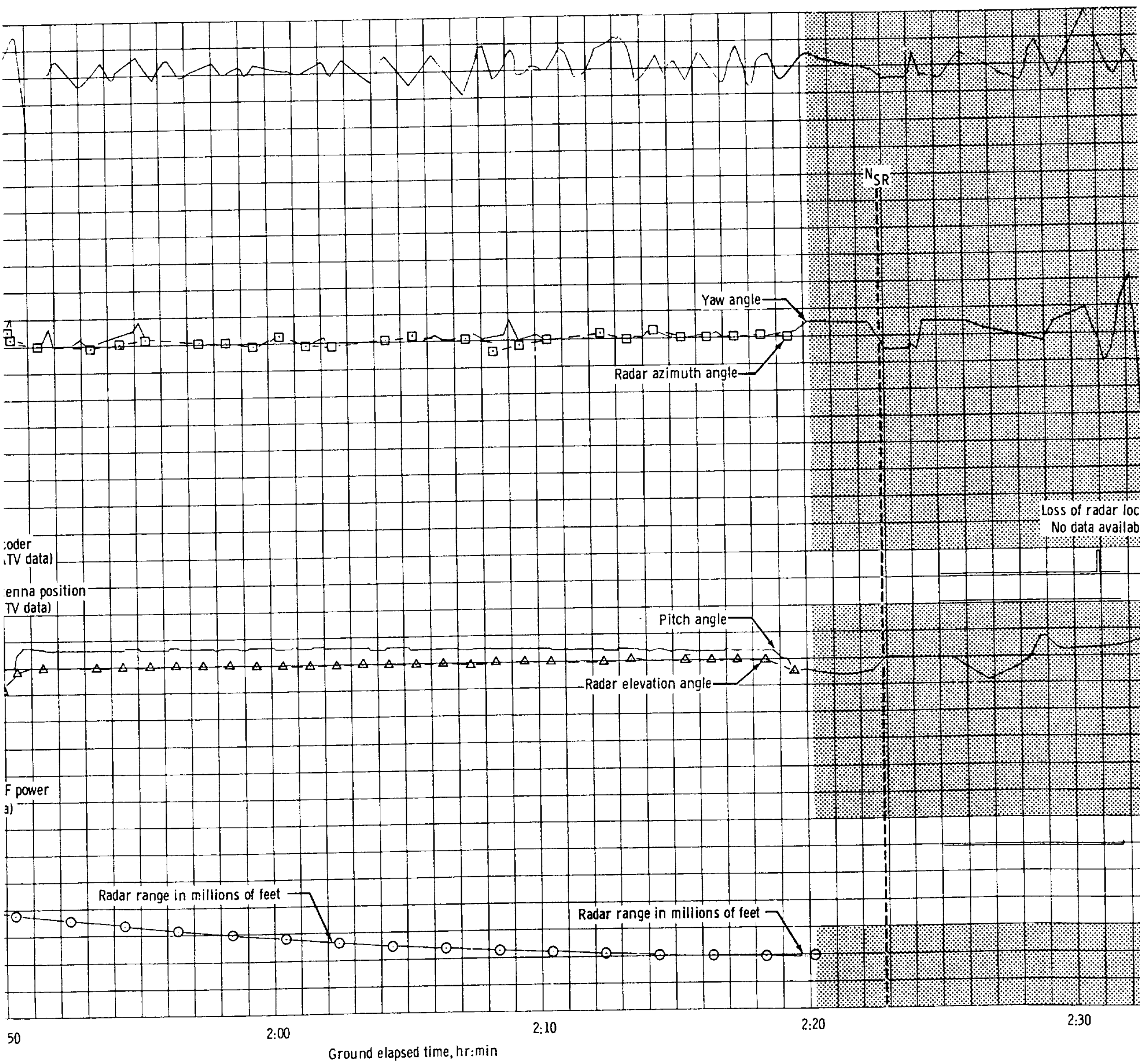


(c) COR_3 through TPF.

Figure 5.1.5-4. - Concluded.



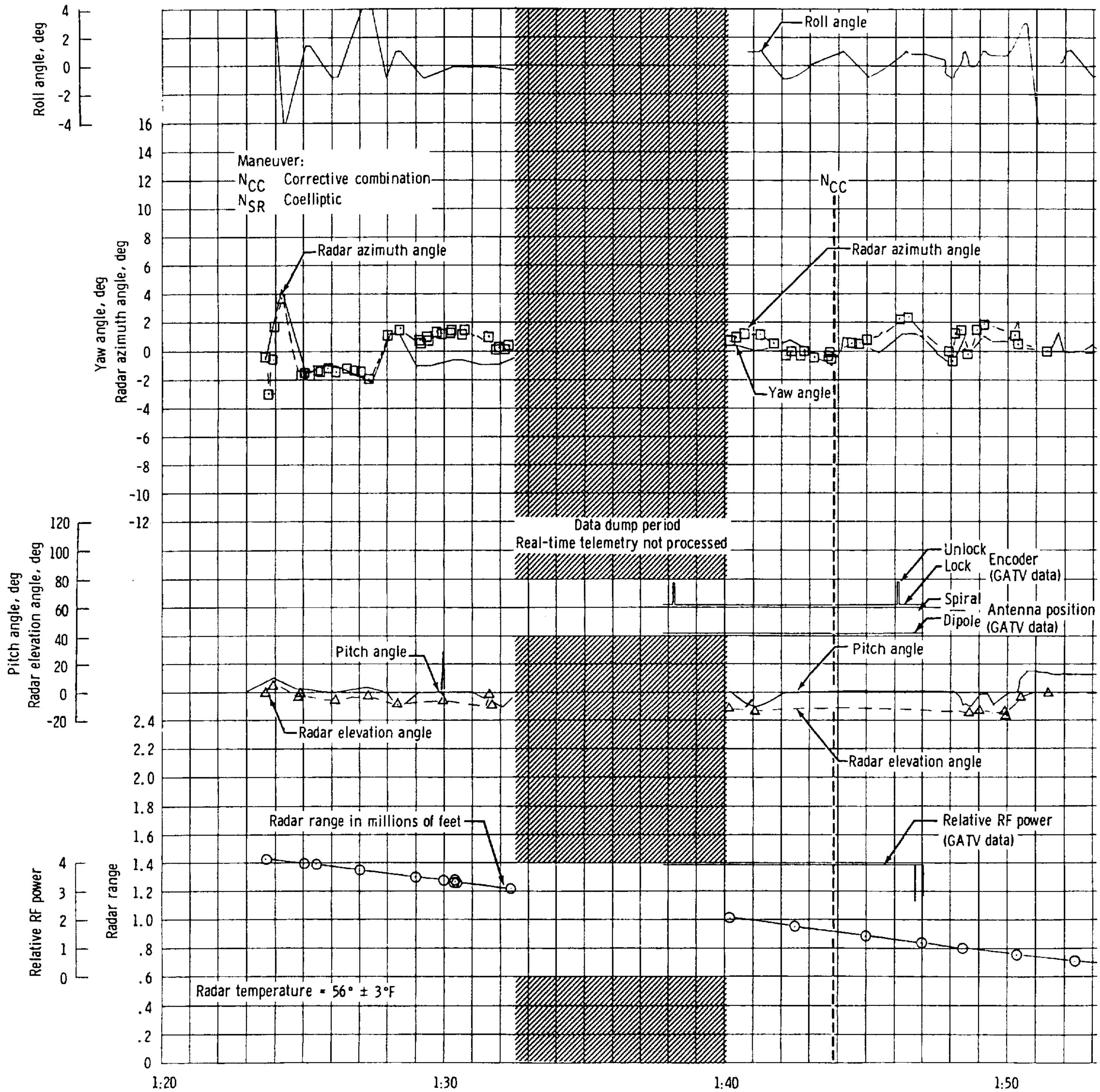
NSR
Yaw angle
Radar azimuth angle
Loss of radar lock-on
No data available
Pitch angle
Radar elevation angle
Radar range in millions of feet
2:30
2:40
2:50



Yaw angle
Radar azimuth angle
N_{SR}
Loss of radar loc
No data availab
:oder
TV data)
enna position
TV data)
Pitch angle
Radar elevation angle
F power
a)
Radar range in millions of feet
Radar range in millions of feet
50
2:00
2:10
2:20
2:30
Ground elapsed time, hr:min

2

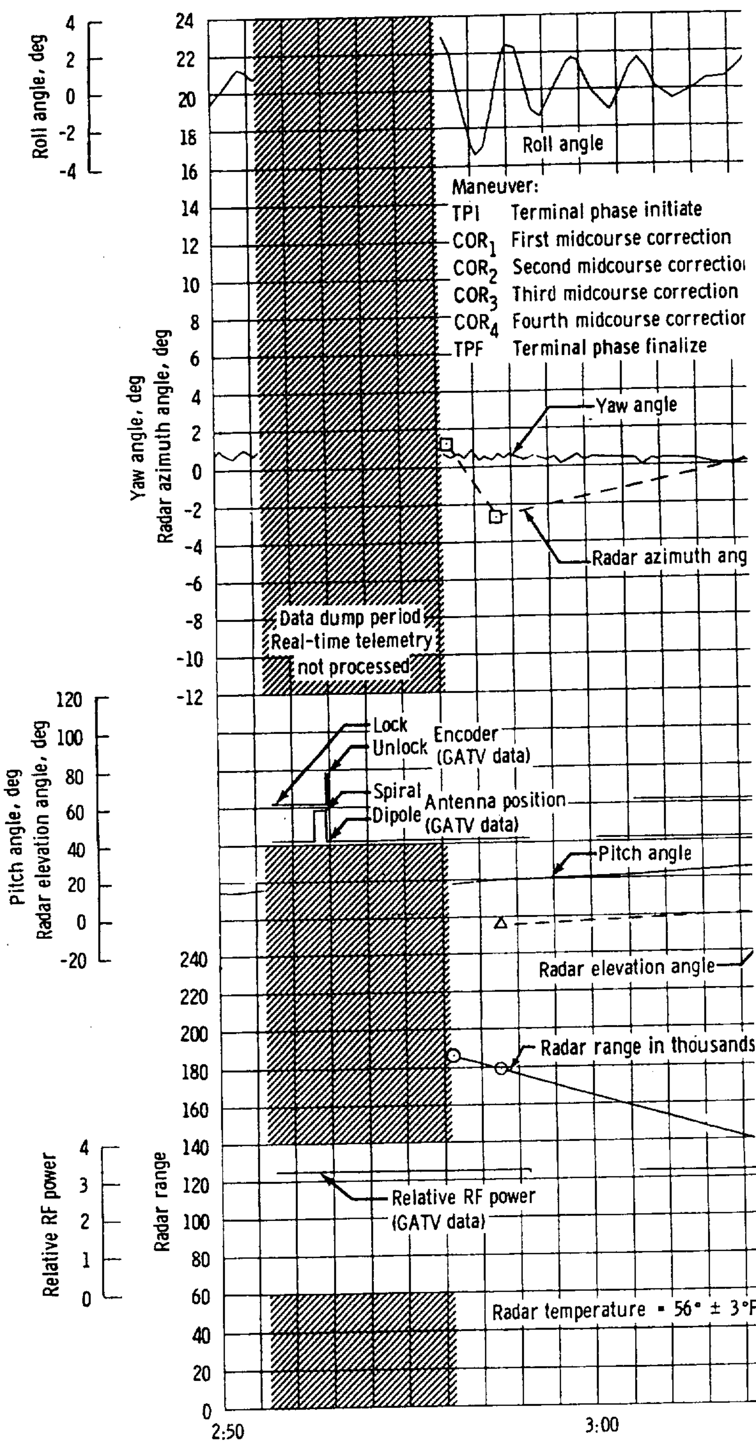
NASA-S-66-11379 DEC 19

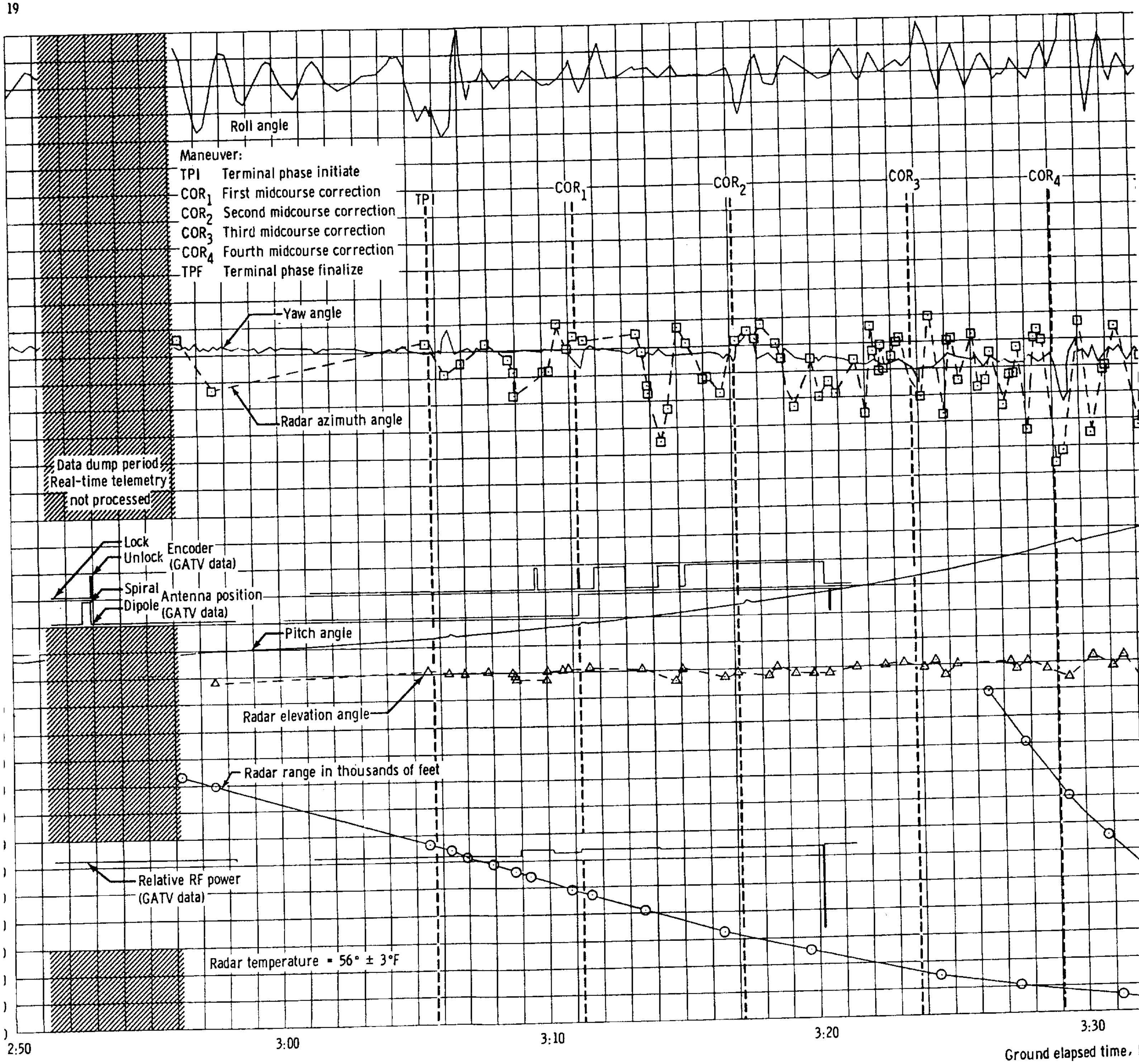


(a) 1 hr 20 min to 2 hr 50 min g. e. t.

Figure 5.1.5-5. - Computer data for the closed-loop rendezvous.

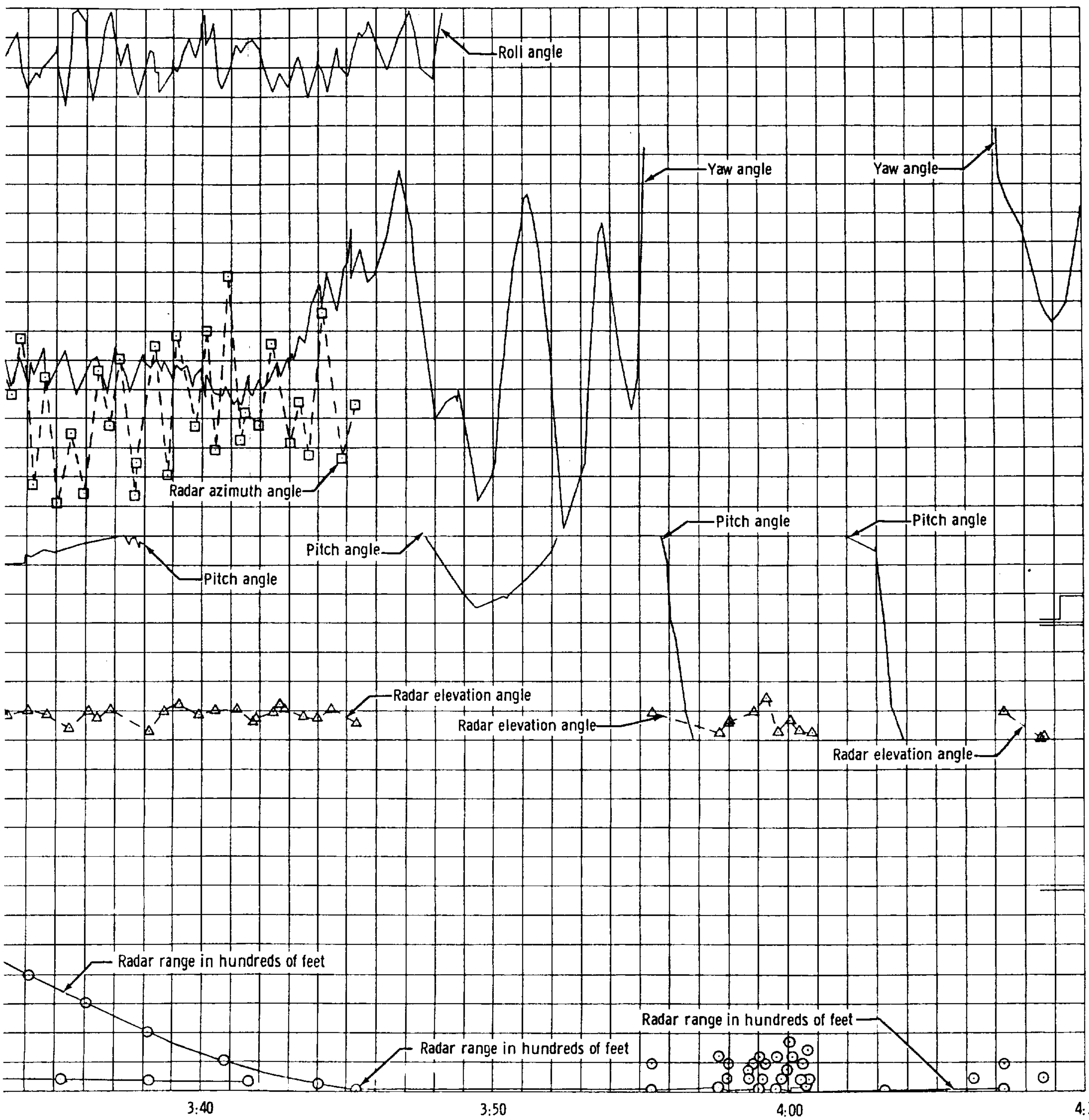
NASA-S-66-11380 DEC 19





Roll angle
Maneuver:
TPI Terminal phase initiate
COR1 First midcourse correction
COR2 Second midcourse correction
COR3 Third midcourse correction
COR4 Fourth midcourse correction
TPF Terminal phase finalize
TPI
COR1
COR2
COR3
COR4
Yaw angle
Radar azimuth angle
Data dump period
Real-time telemetry
not processed
Lock
Unlock
Encoder
(GATV data)
Spiral
Dipole
Antenna position
(GATV data)
Pitch angle
Radar elevation angle
Radar range in thousands of feet
Relative RF power
(GATV data)
Radar temperature = 56° ± 3°F
2:50
3:00
3:10
3:20
3:30
Ground elapsed time, h

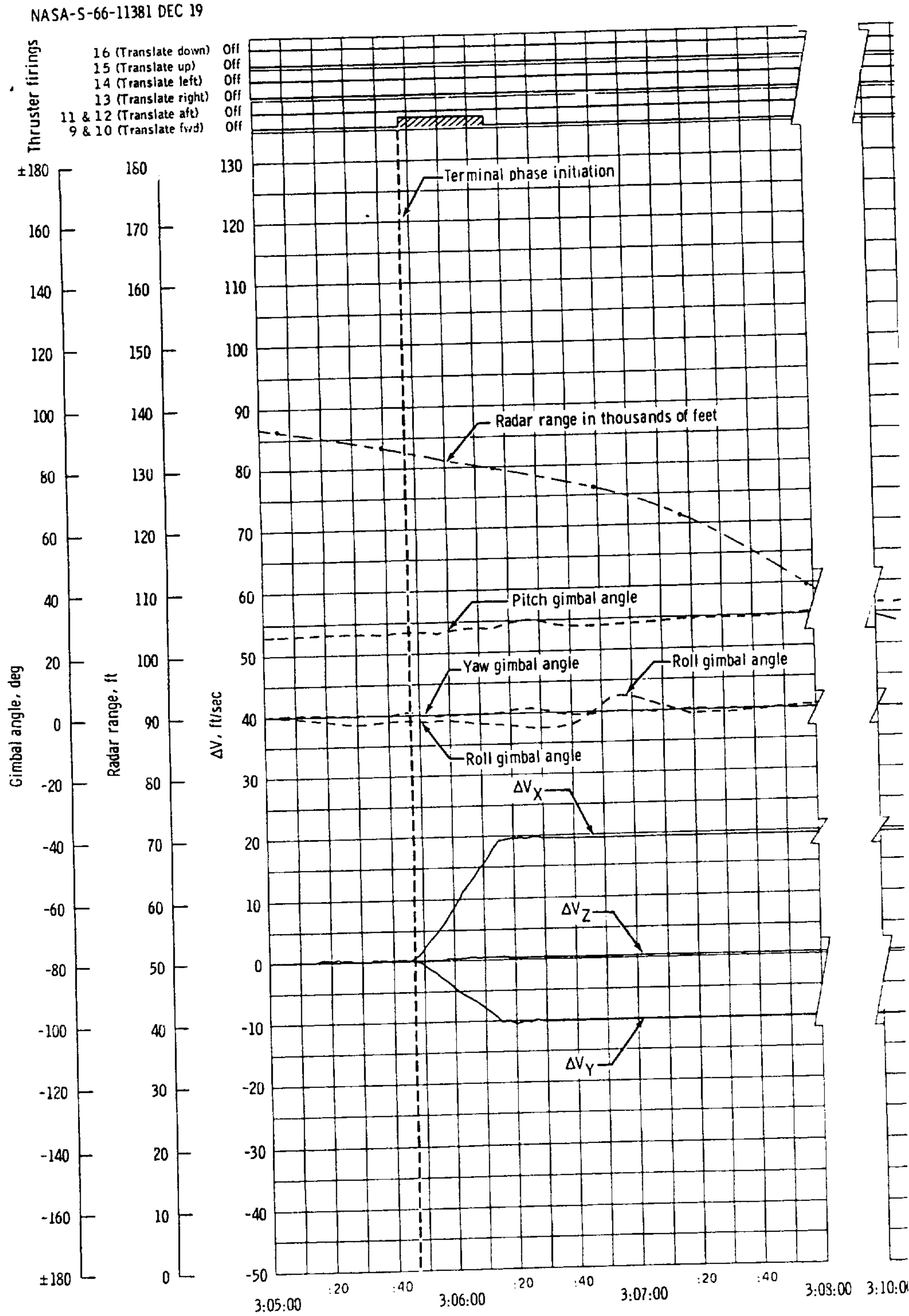
2

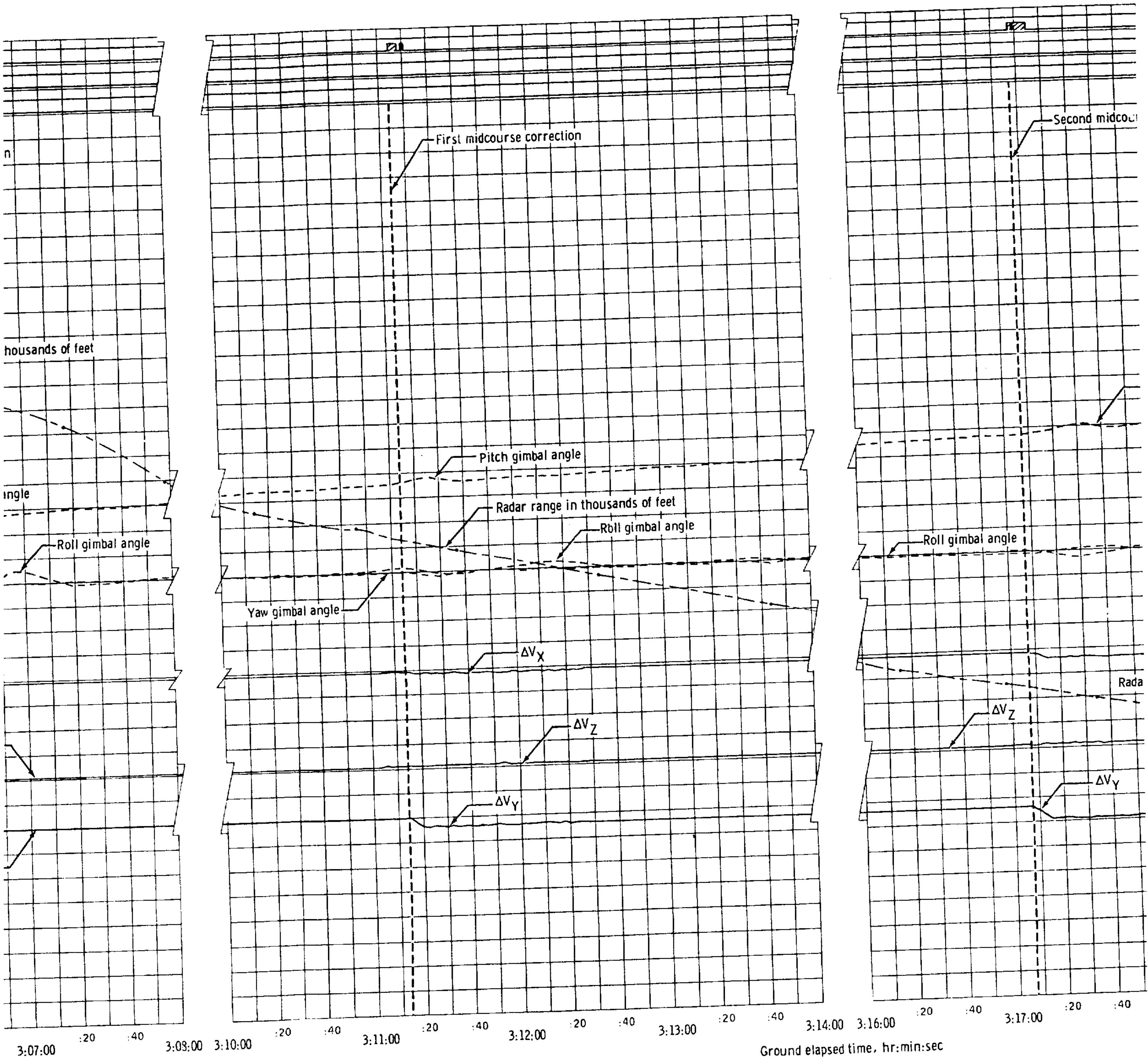


(b) 2 hr 50 min to 4 hr 10 min g.e.t.

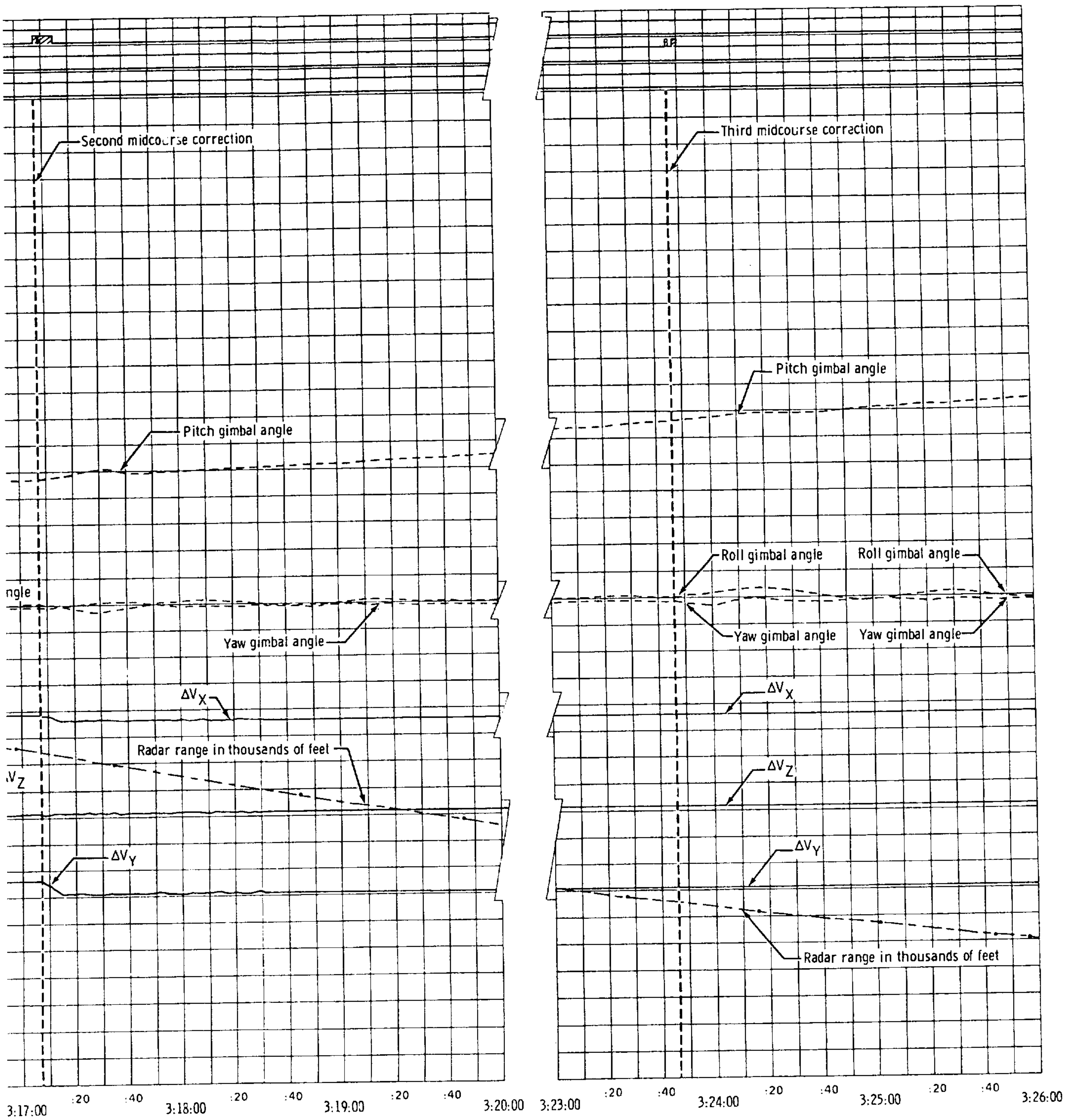
Figure 5.1.5-5 - Concluded.

NASA-S-66-11381 DEC 19



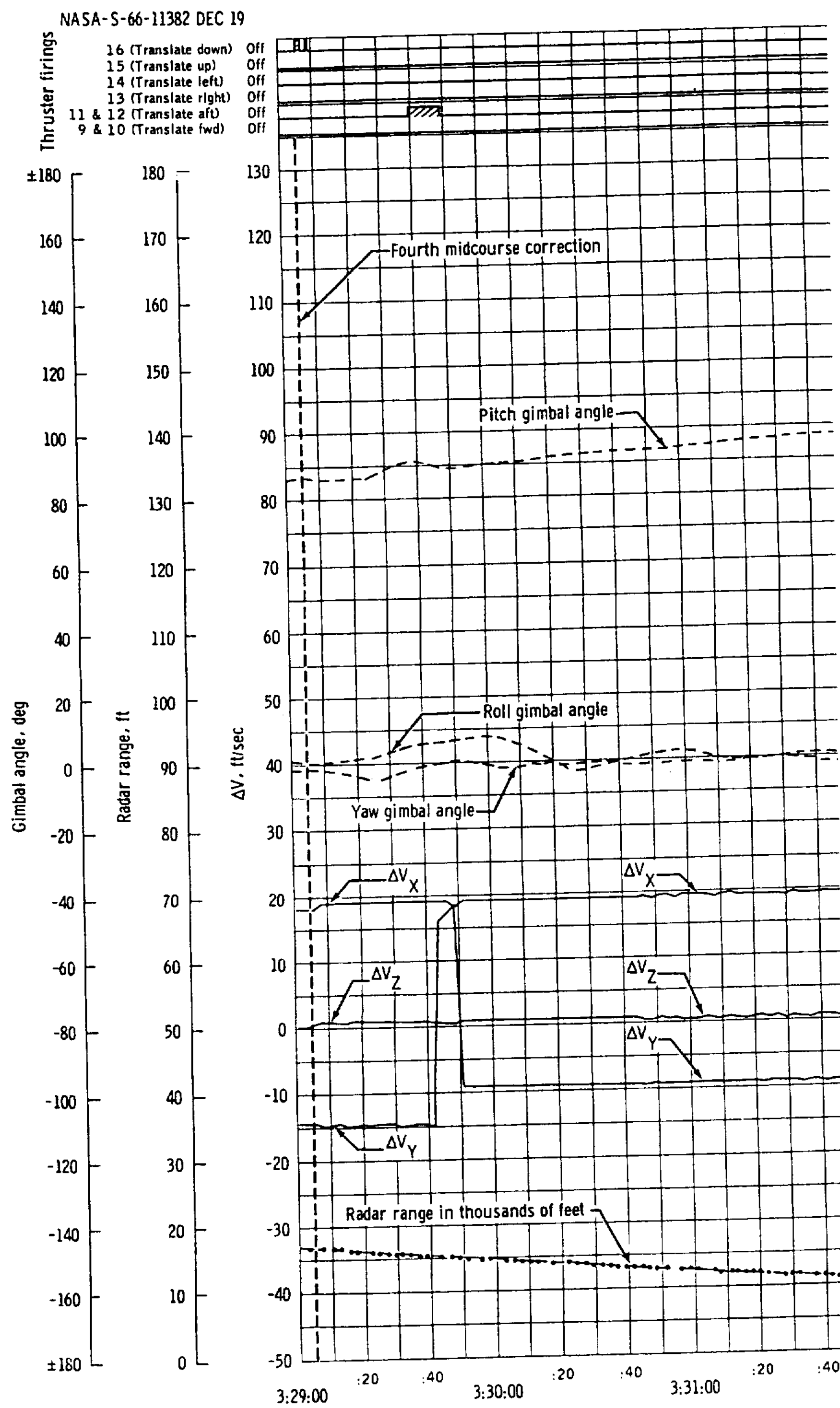


First midcourse correction
Second midcou
thousands of feet
angle
Roll gimbal angle
Pitch gimbal angle
Radar range in thousands of feet
Roll gimbal angle
Yaw gimbal angle
Roll gimbal angle
ΔV_X
ΔV_Z
ΔV_Y
Rada
ΔV_Z
ΔV_Y
3:07:00
:20
:40
3:08:00
3:10:00
:20
:40
3:11:00
:20
:40
3:12:00
:20
:40
3:13:00
:20
:40
3:14:00
3:16:00
:20
:40
3:17:00
:20
:40
Ground elapsed time, hr:min:sec

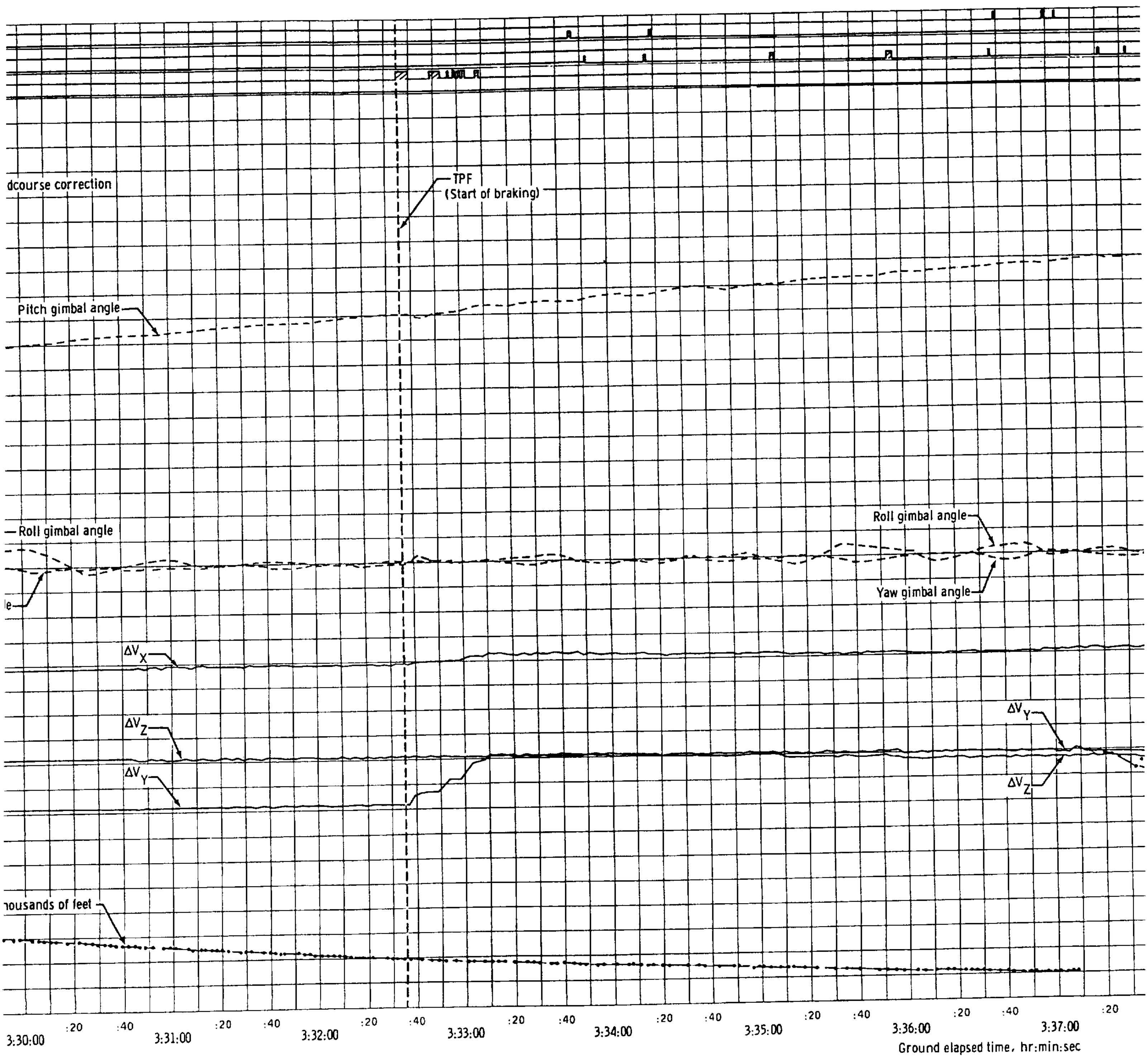


(a) 3:05:00 to 3:26:00 g. e. t.

Figure 5.1.5-6. - Spacecraft applied ΔV's, gimbal angles, and radar range, from TPI to first docking.

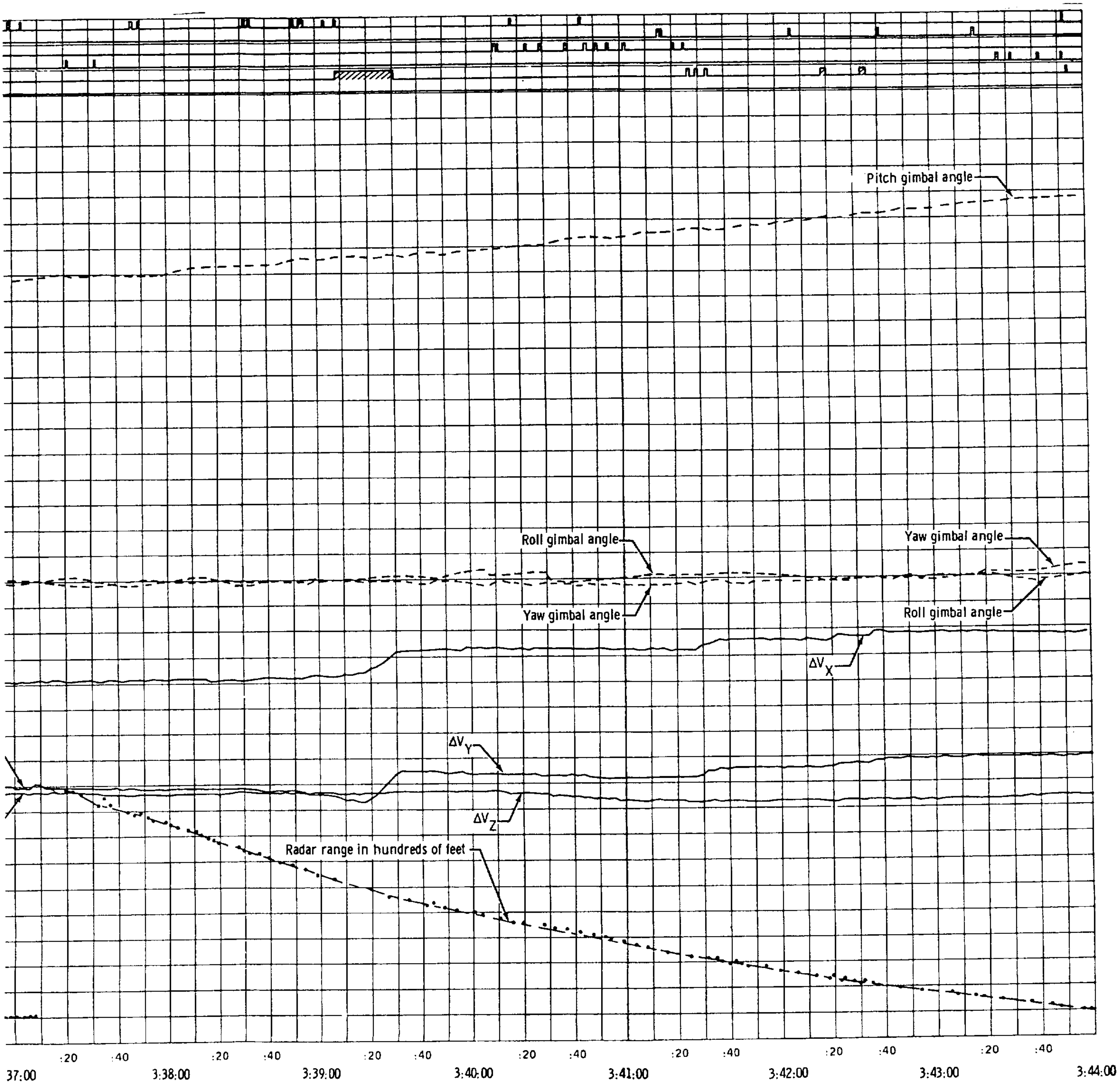


NASA-S-66-11382 DEC 19
Thruster firings
16 (Translate down) Off
15 (Translate up) Off
14 (Translate left) Off
13 (Translate right) Off
11 & 12 (Translate aft) Off
9 & 10 (Translate fwd) Off
Fourth midcourse correction
Pitch gimbal angle
Roll gimbal angle
Yaw gimbal angle
ΔVX
ΔVZ
ΔVY
Radar range in thousands of feet
Gimbal angle, deg
Radar range, ft
ΔV, ft/sec
3:29:00
3:30:00
3:31:00



dcourse correction
TPF
(Start of braking)
Pitch gimbal angle
Roll gimbal angle
Roll gimbal angle
Yaw gimbal angle
le
ΔV_X
ΔV_Z
ΔV_Y
ΔV_Y
ΔV_Z
housands of feet
3:30:00
:20
:40
3:31:00
:20
:40
3:32:00
:20
:40
3:33:00
:20
:40
3:34:00
:20
:40
3:35:00
:20
:40
3:36:00
:20
:40
3:37:00
:20
Ground elapsed time, hr:min:sec

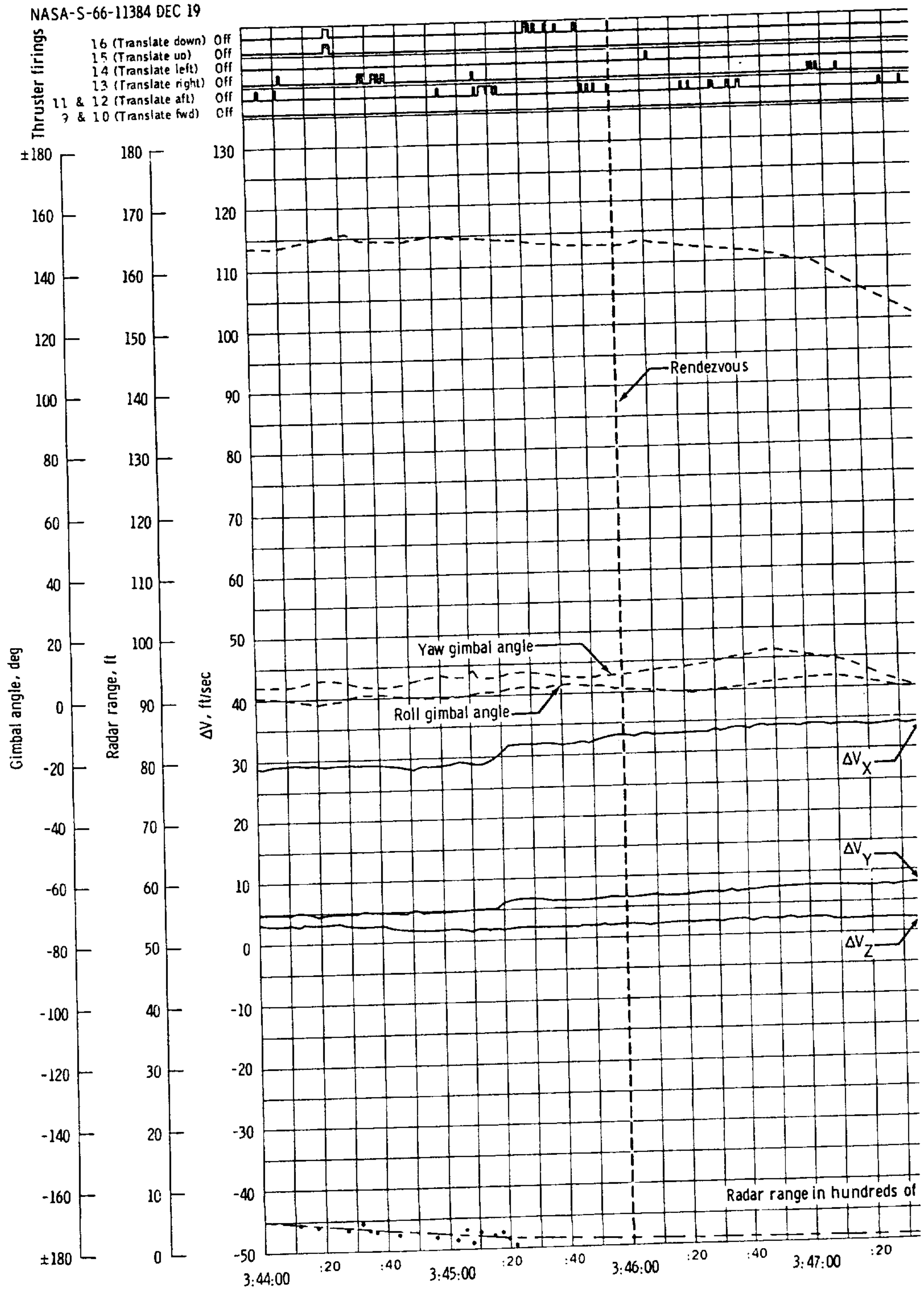
2

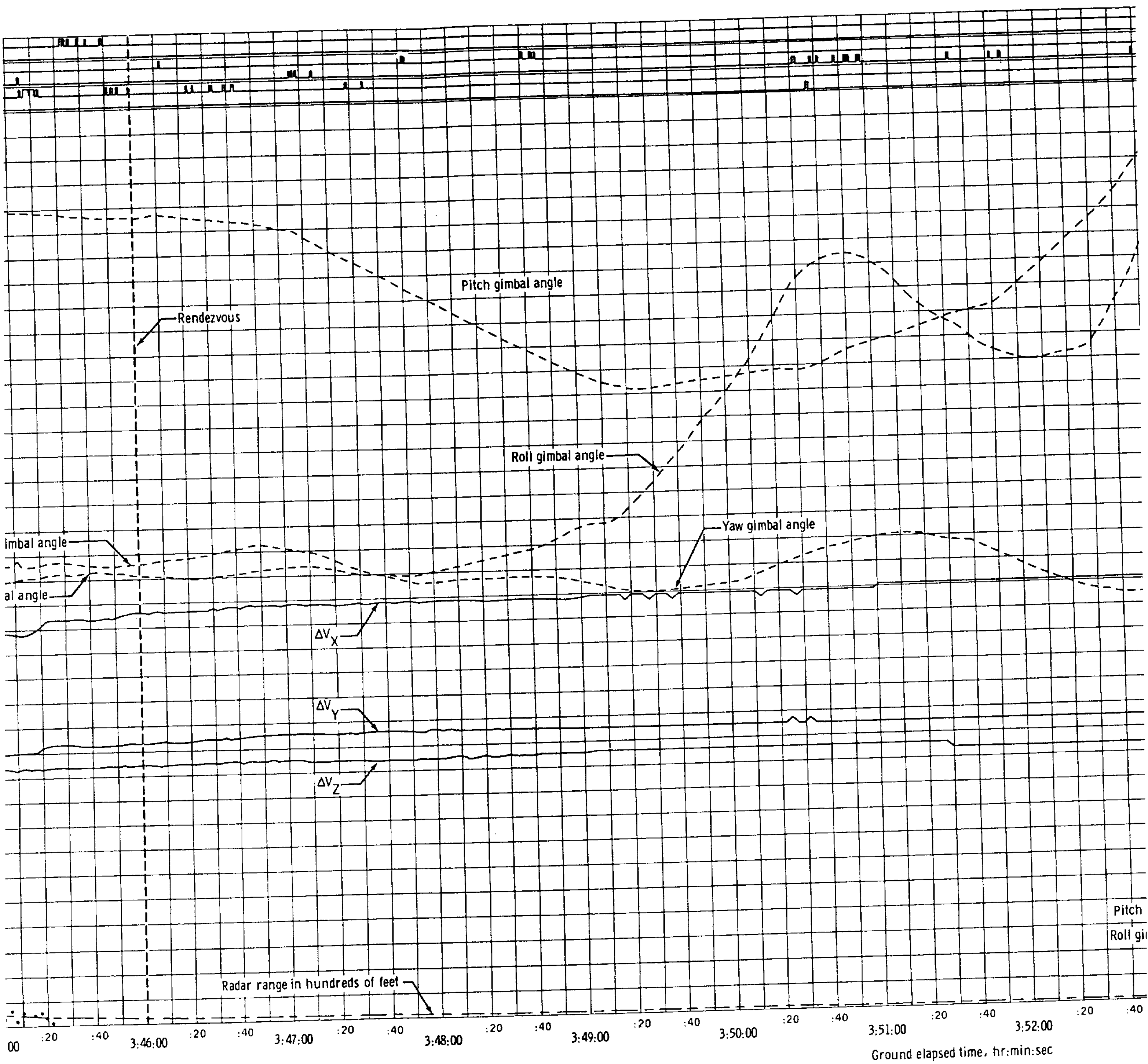


(b) 3:29:00 to 3:44:00 g. e. t.

Figure 5.1.5-6. - Continued.

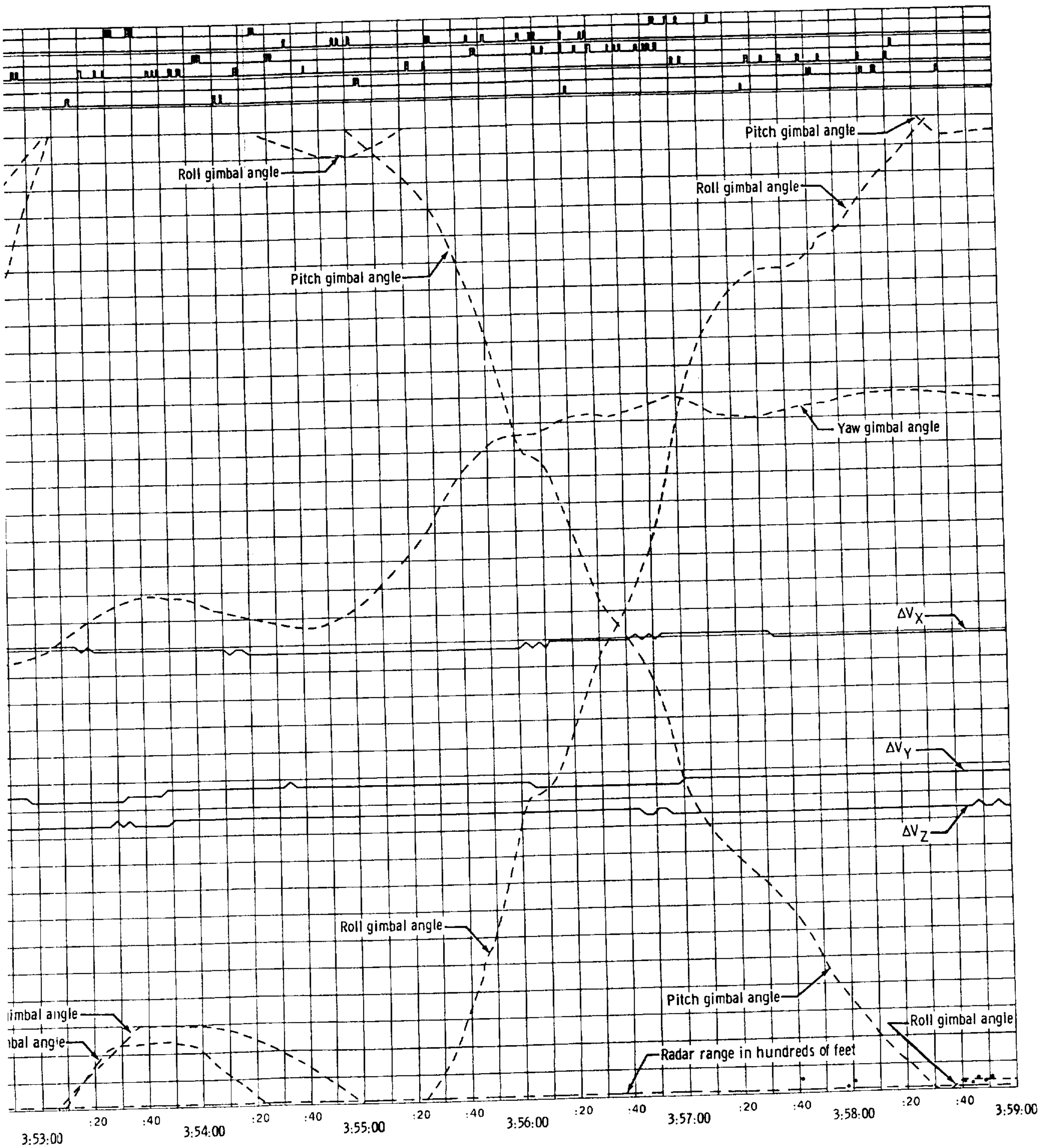
NASA-S-66-11384 DEC 19





Pitch gimbal angle
Rendezvous
Roll gimbal angle
Yaw gimbal angle
imbal angle
al angle
ΔVX
ΔVY
ΔVZ
Radar range in hundreds of feet
Pitch
Roll gi
00
:20
:40
3:46:00
:20
:40
3:47:00
:20
:40
3:48:00
:20
:40
3:49:00
:20
:40
3:50:00
:20
:40
3:51:00
:20
:40
3:52:00
:20
:40
Ground elapsed time, hr:min:sec

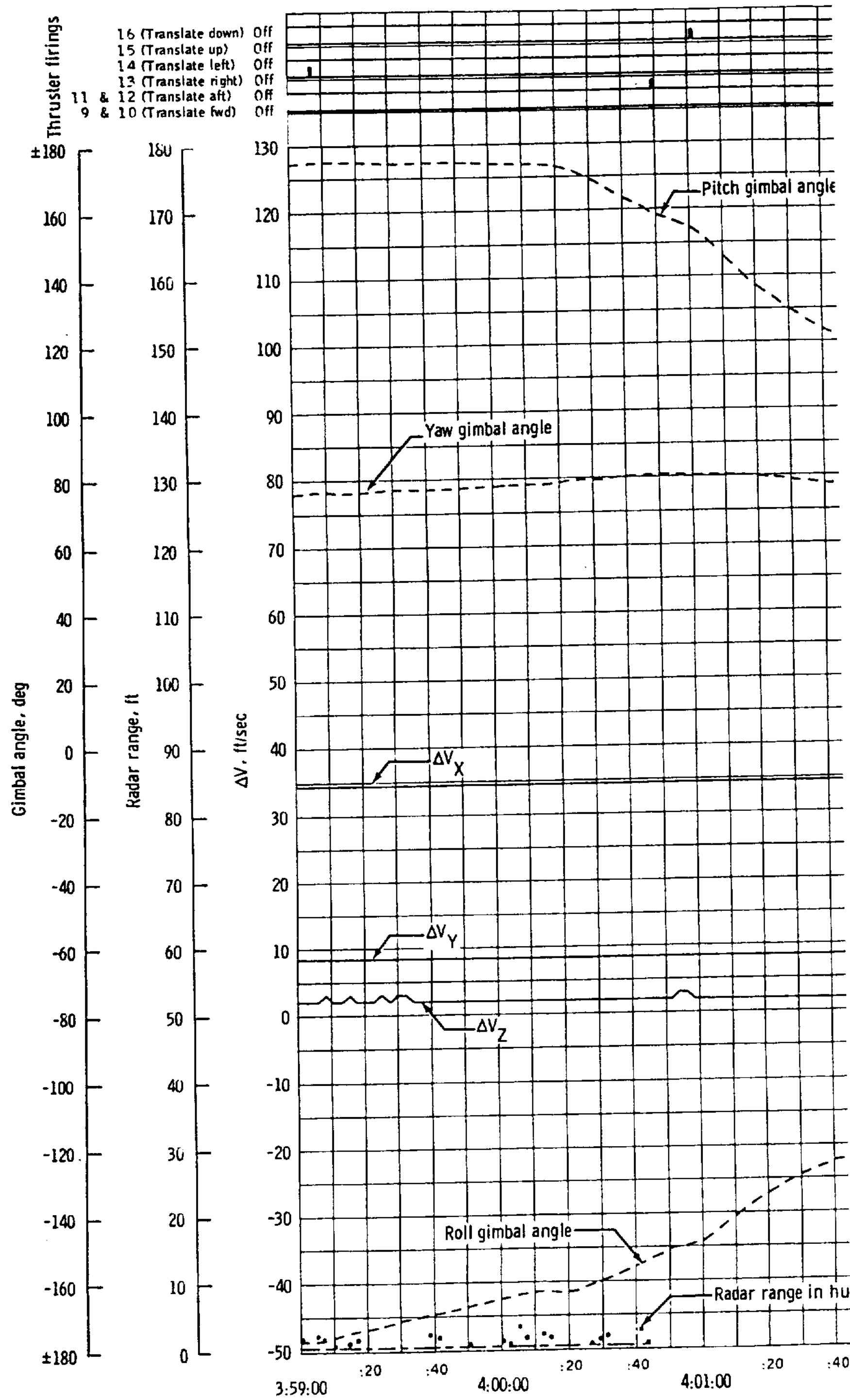
2

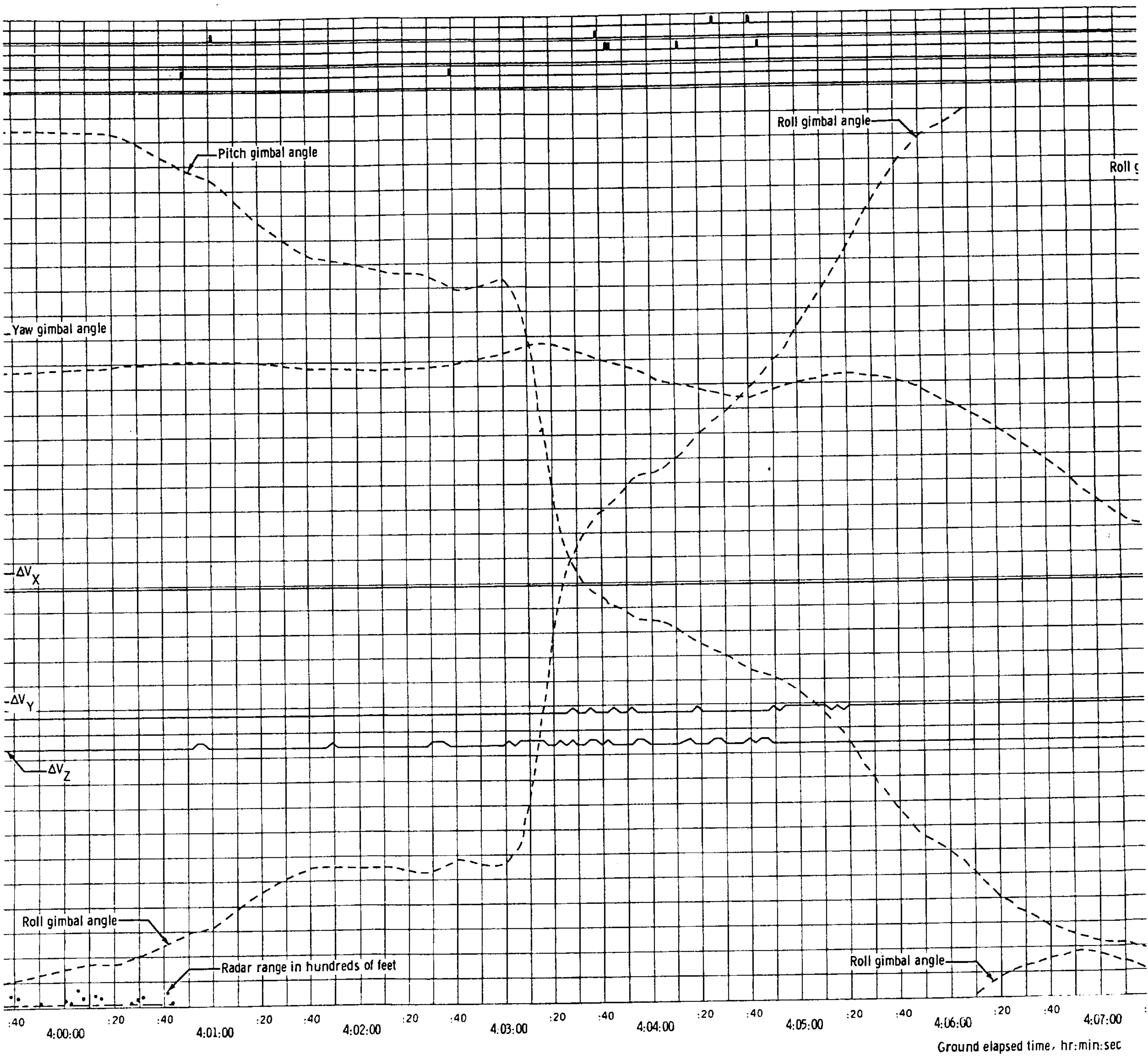


(c) 3:44:00 to 3:59:00 g. e. t.

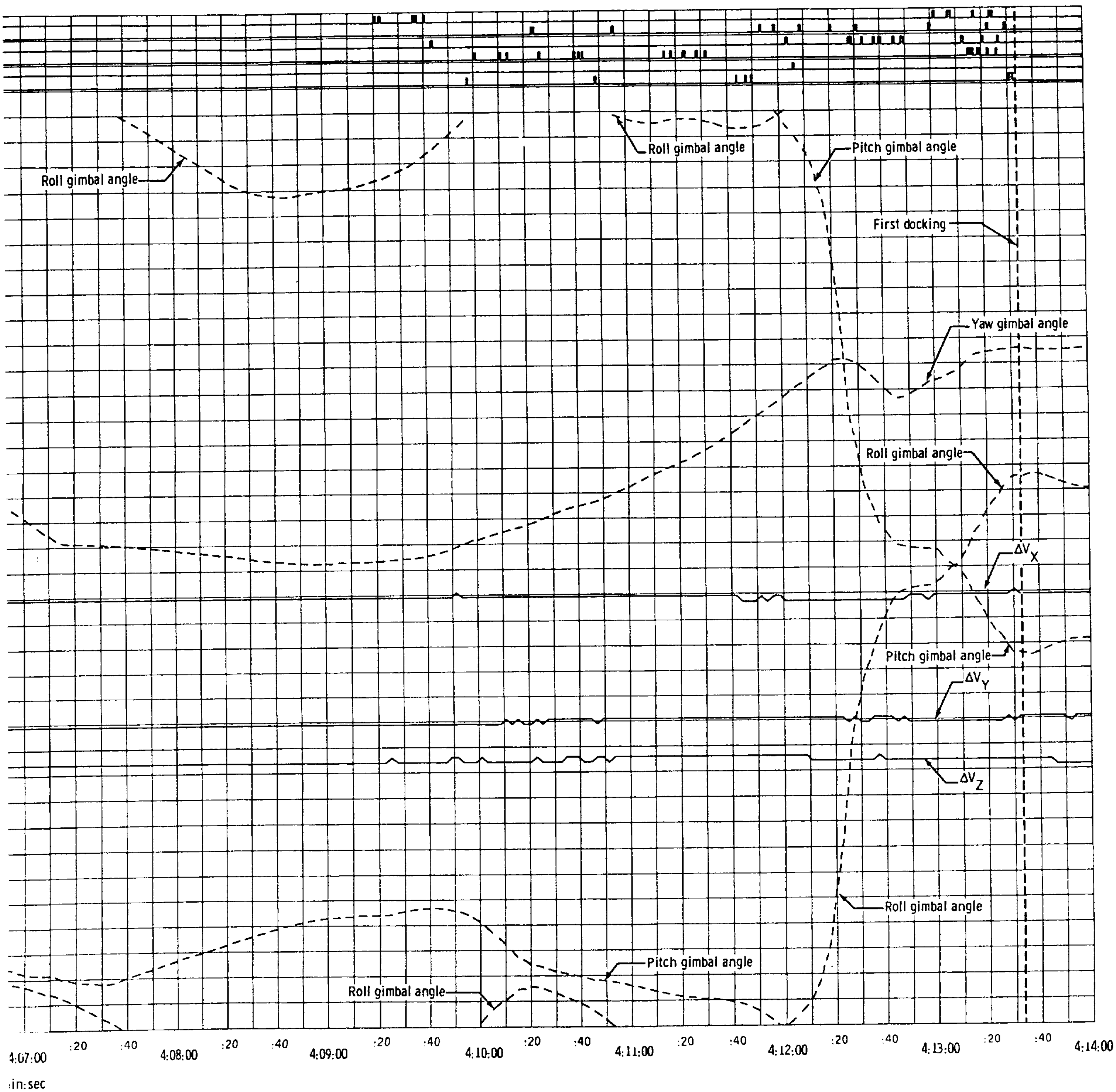
Figure 5.1.5-6. - Continued.

NASA-S-66-11383 DEC 19





Roll gimbal angle
Pitch gimbal angle
Roll g
Yaw gimbal angle
ΔV X
ΔV Y
ΔV Z
Roll gimbal angle
Radar range in hundreds of feet
Roll gimbal angle
:40
4:00:00
:20
:40
4:01:00
:20
:40
4:02:00
:20
:40
4:03:00
:20
:40
4:04:00
:20
:40
4:05:00
:20
:40
4:06:00
:20
:40
4:07:00
Ground elapsed time, hr:min:sec



(d) 3:59:00 to 4:14:00 g.e.t.

Figure 5.1.5-6. - Concluded.

NASA-S-66-11392 DEC 19

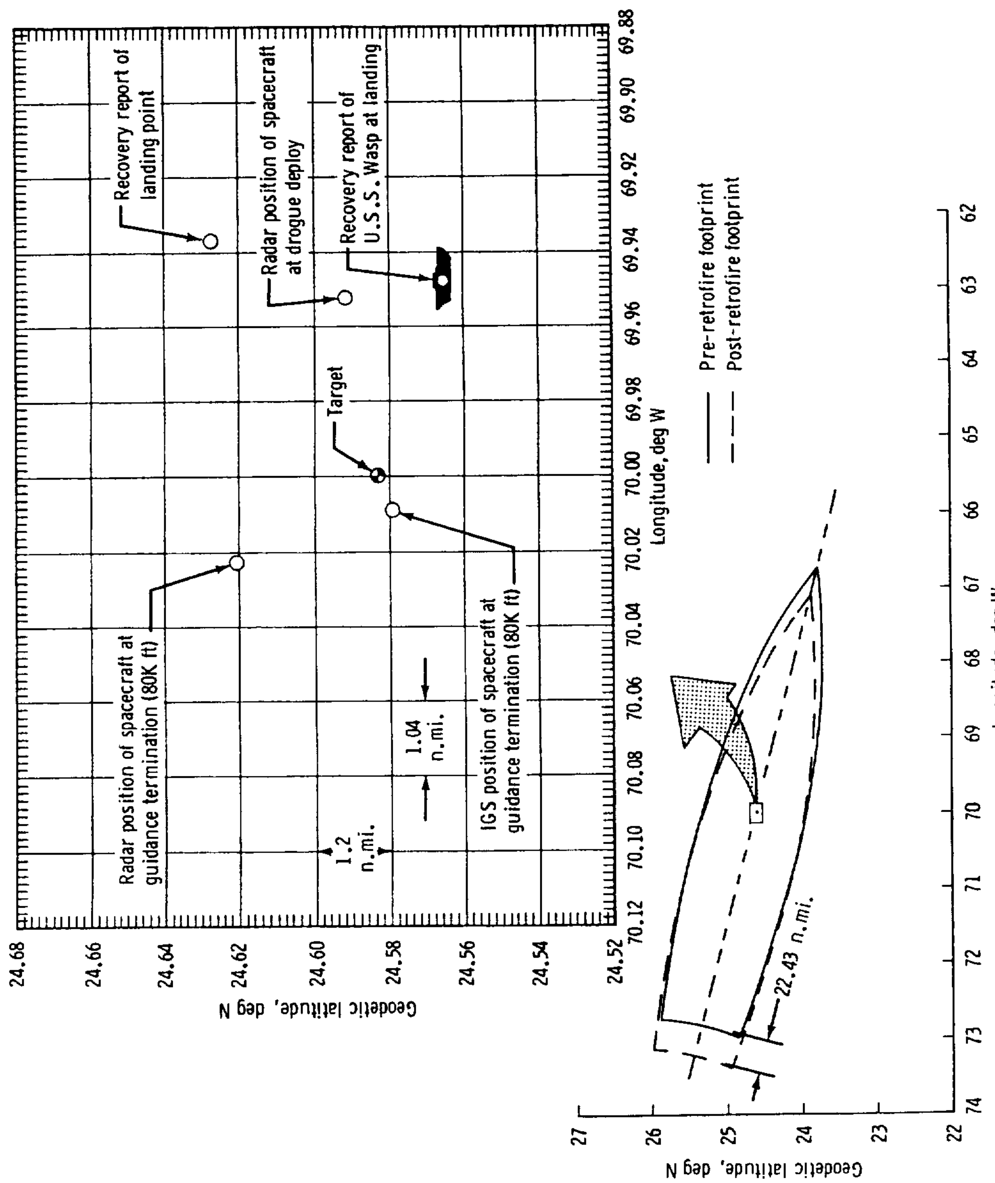


Figure 5.1.5-7 - Touchdown comparisons.

NASA-S-66-11385 DEC 19

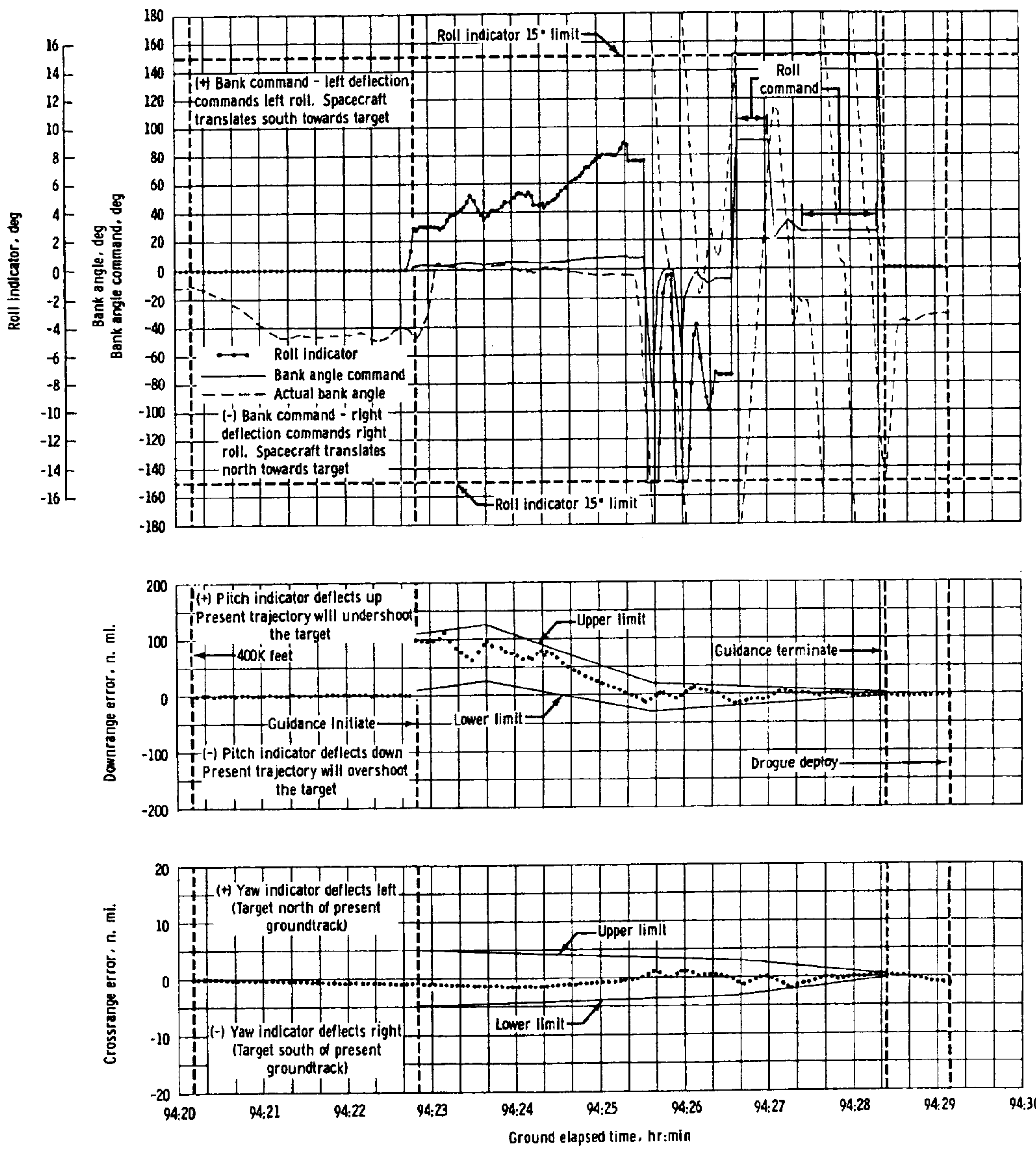
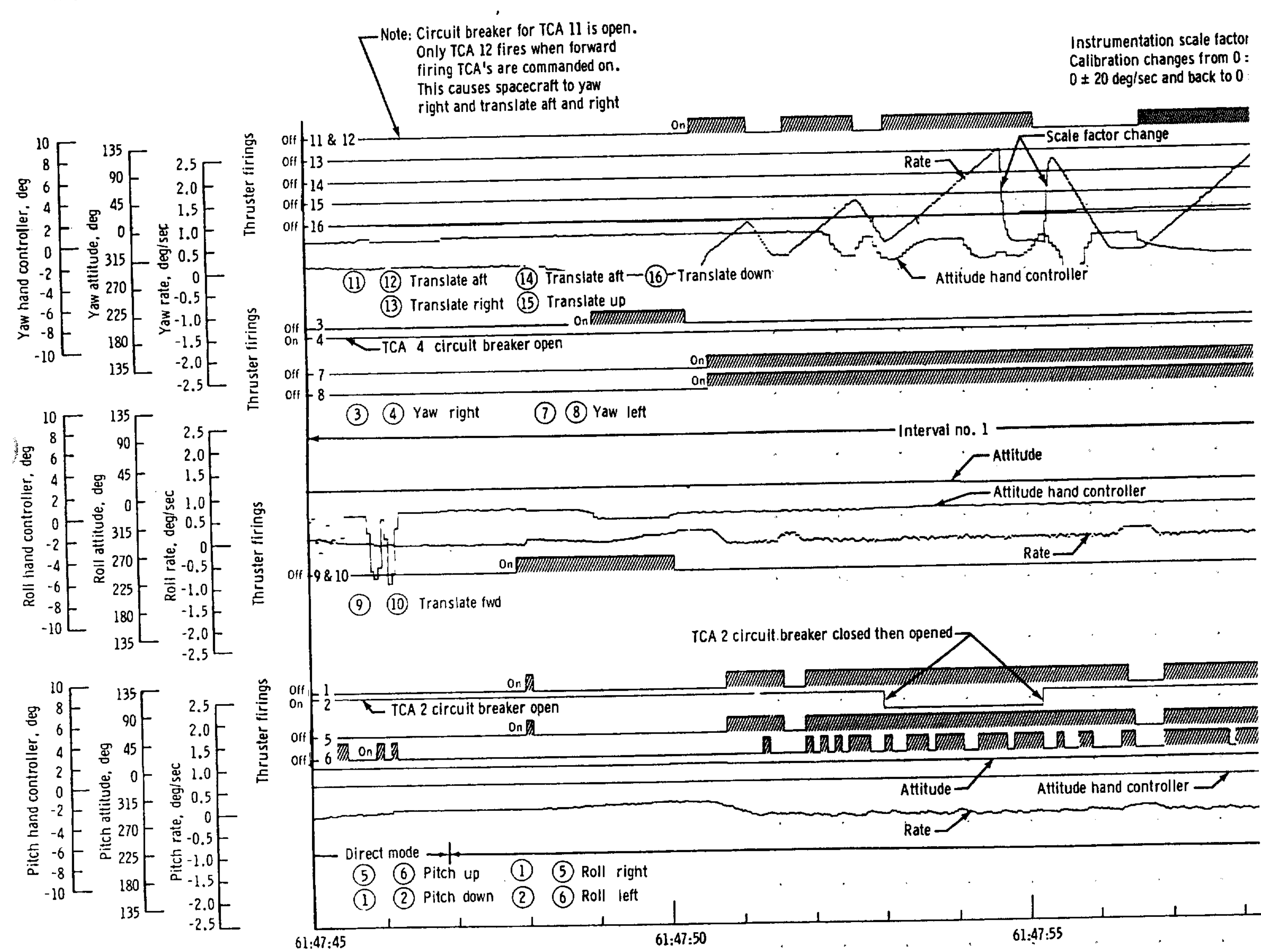


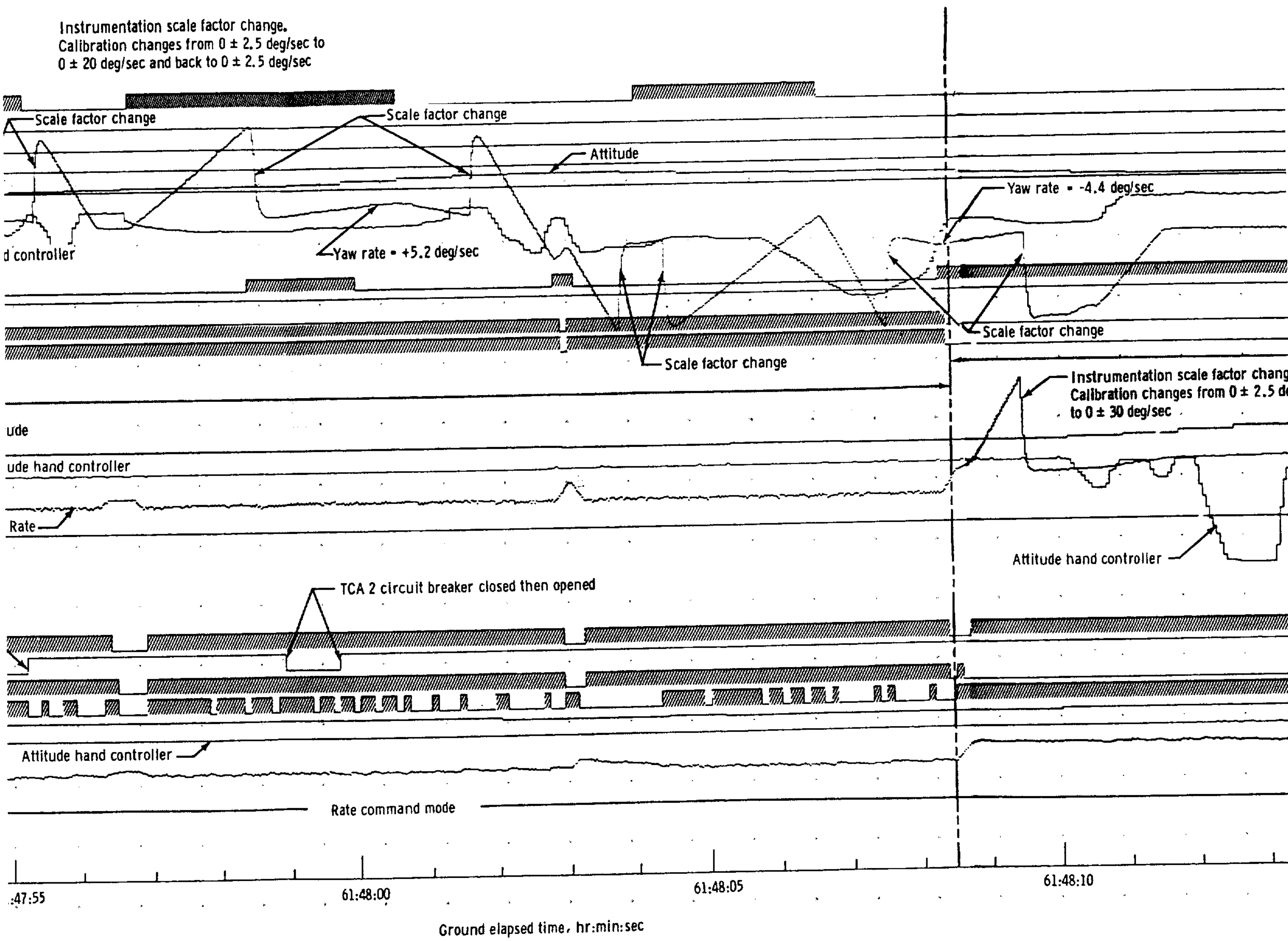
Figure 5.1.5-8. - Reentry guidance parameters.

NASA-S-66-11377 DEC 17

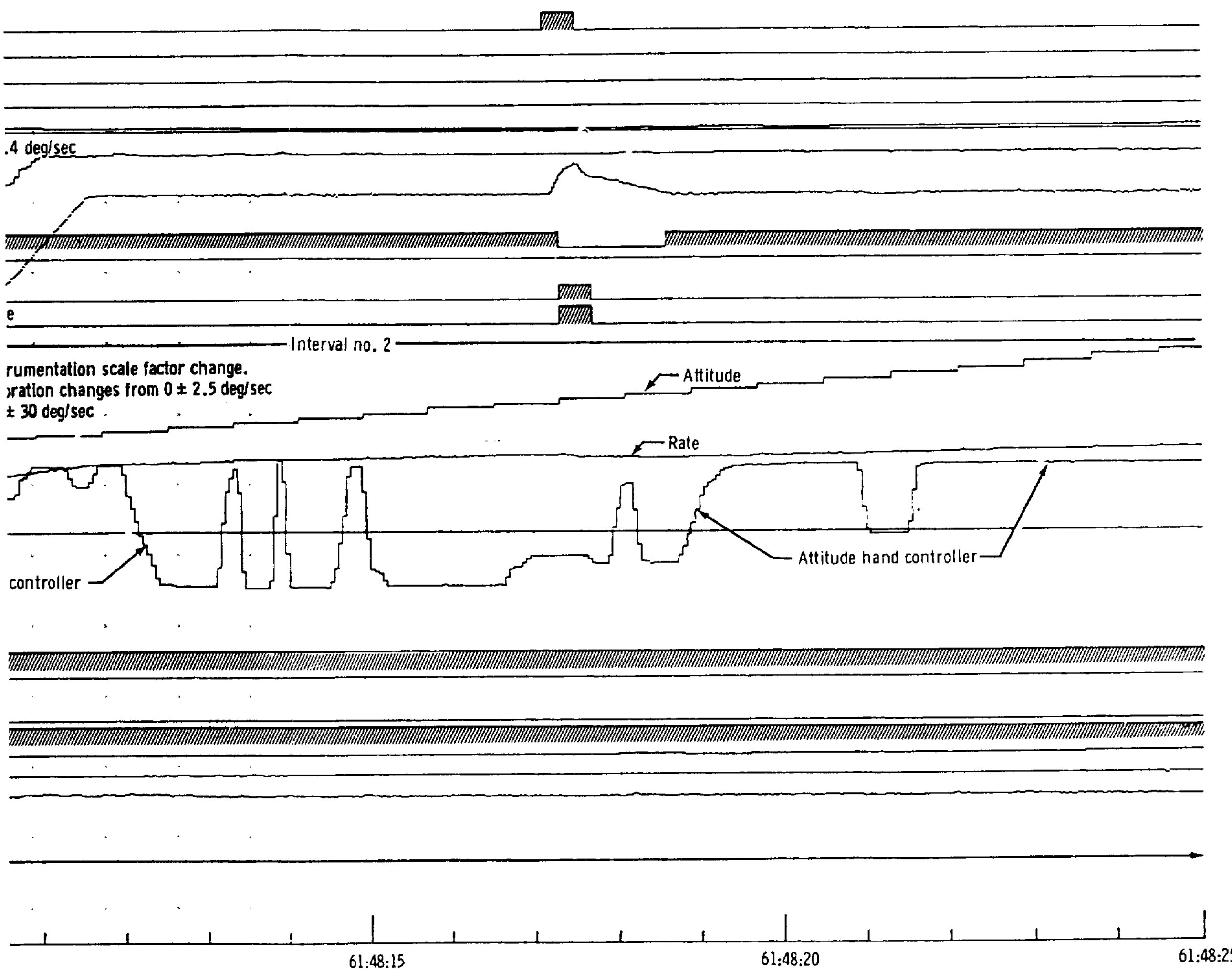


(a) 61:47:45 to 61:48:25, g.e.t.

Figure 5.1.5-9. - Rate command performance during phase adjust maneuver.


Instrumentation scale factor change.
Calibration changes from 0 ± 2.5 deg/sec to
0 ± 20 deg/sec and back to 0 ± 2.5 deg/sec
Scale factor change
Scale factor change
Attitude
Yaw rate = -4.4 deg/sec
d controller
Yaw rate = +5.2 deg/sec
Scale factor change
Scale factor change
Instrumentation scale factor chang
Calibration changes from 0 ± 2.5 de
to 0 ± 30 deg/sec
ude
ude hand controller
Rate
Attitude hand controller
TCA 2 circuit breaker closed then opened
Attitude hand controller
Rate command mode
:47:55
61:48:00
61:48:05
61:48:10
Ground elapsed time, hr:min:sec

2



.4 deg/sec
e
Interval no. 2
rumentation scale factor change.
ration changes from 0 ± 2.5 deg/sec
± 30 deg/sec
Attitude
Rate
controller
Attitude hand controller
61:48:15
61:48:20
61:48:25

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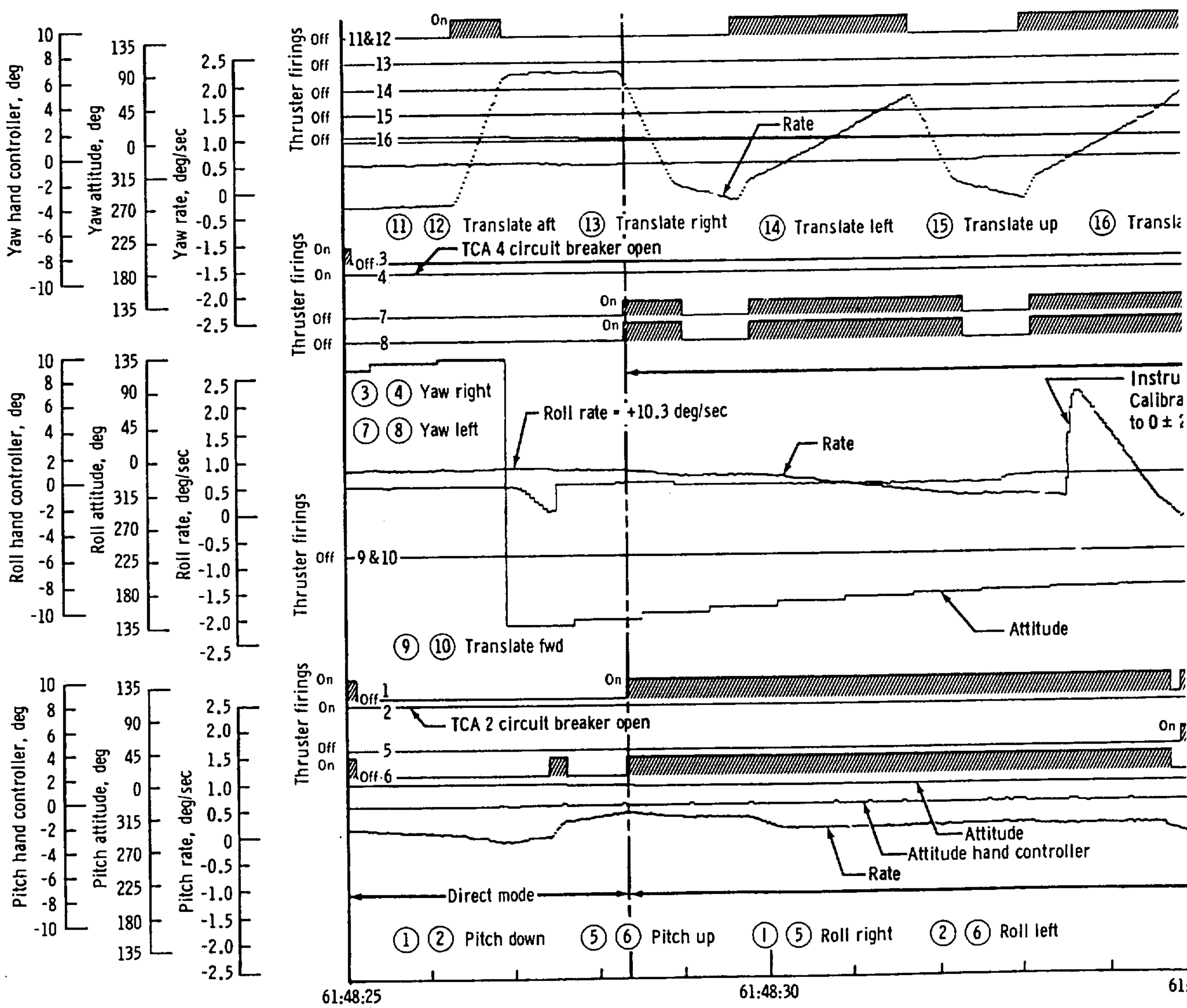
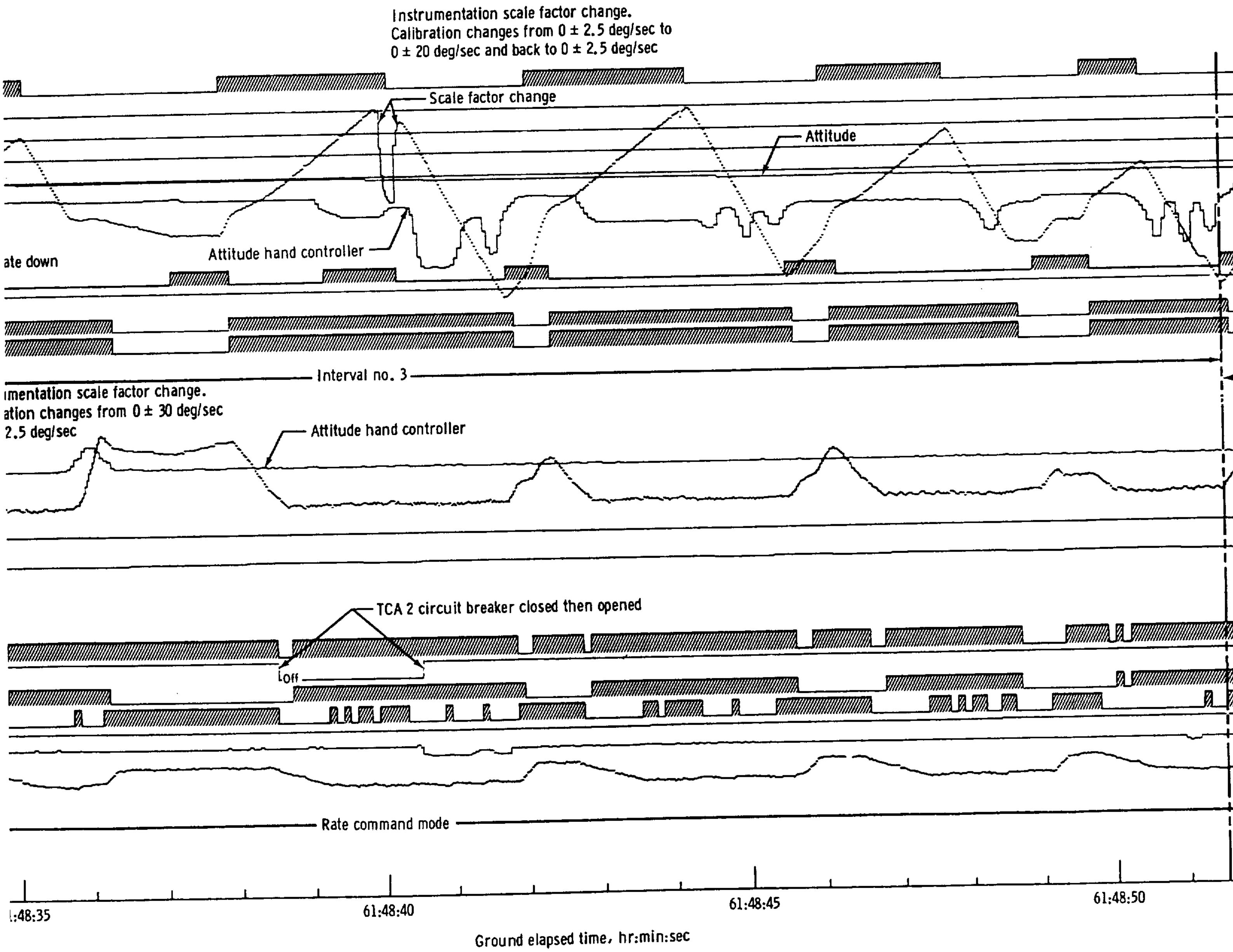
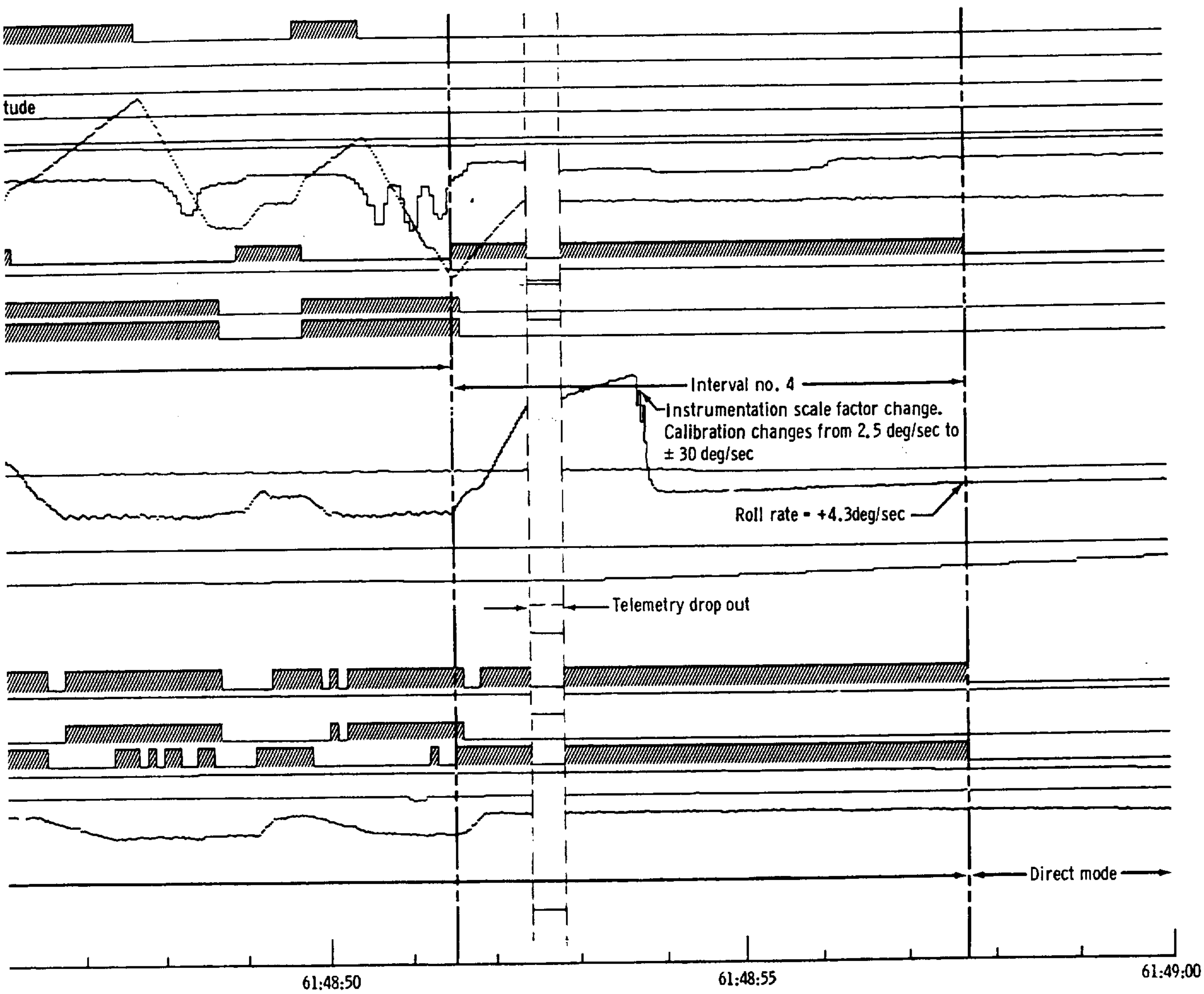



Instrumentation scale factor change.
Calibration changes from 0 ± 2.5 deg/sec to
0 ± 20 deg/sec and back to 0 ± 2.5 deg/sec
Scale factor change
Attitude
Attitude hand controller
ate down
Interval no. 3
imentation scale factor change.
ation changes from 0 ± 30 deg/sec
2.5 deg/sec
Attitude hand controller
TCA 2 circuit breaker closed then opened
Off
Rate command mode
:48:35
61:48:40
61:48:45
61:48:50
Ground elapsed time, hr:min:sec

2



(b) 61:48:25 to 61:49:00, g.e.t.

3

Figure 5.1.5-9. - Concluded.

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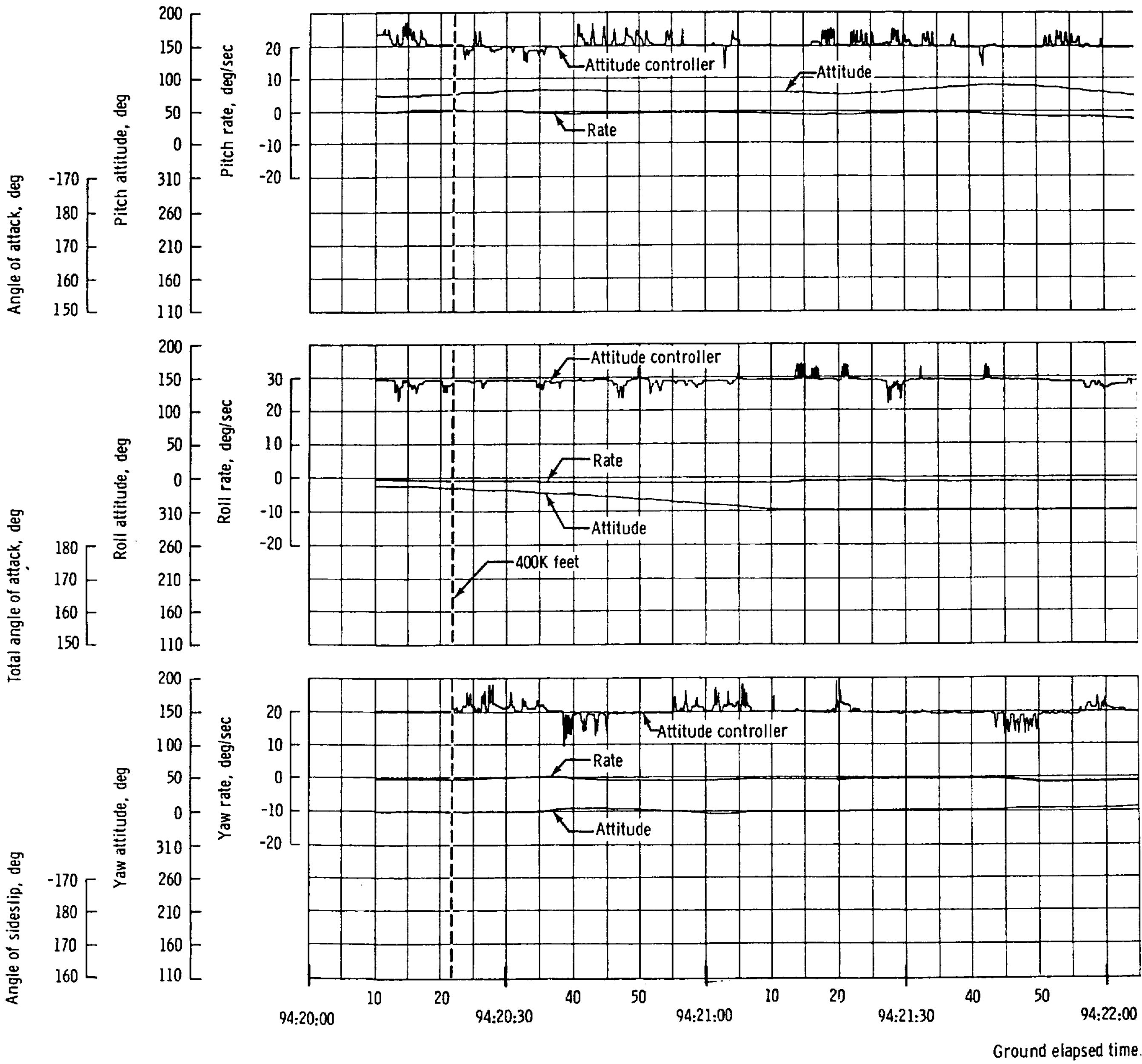
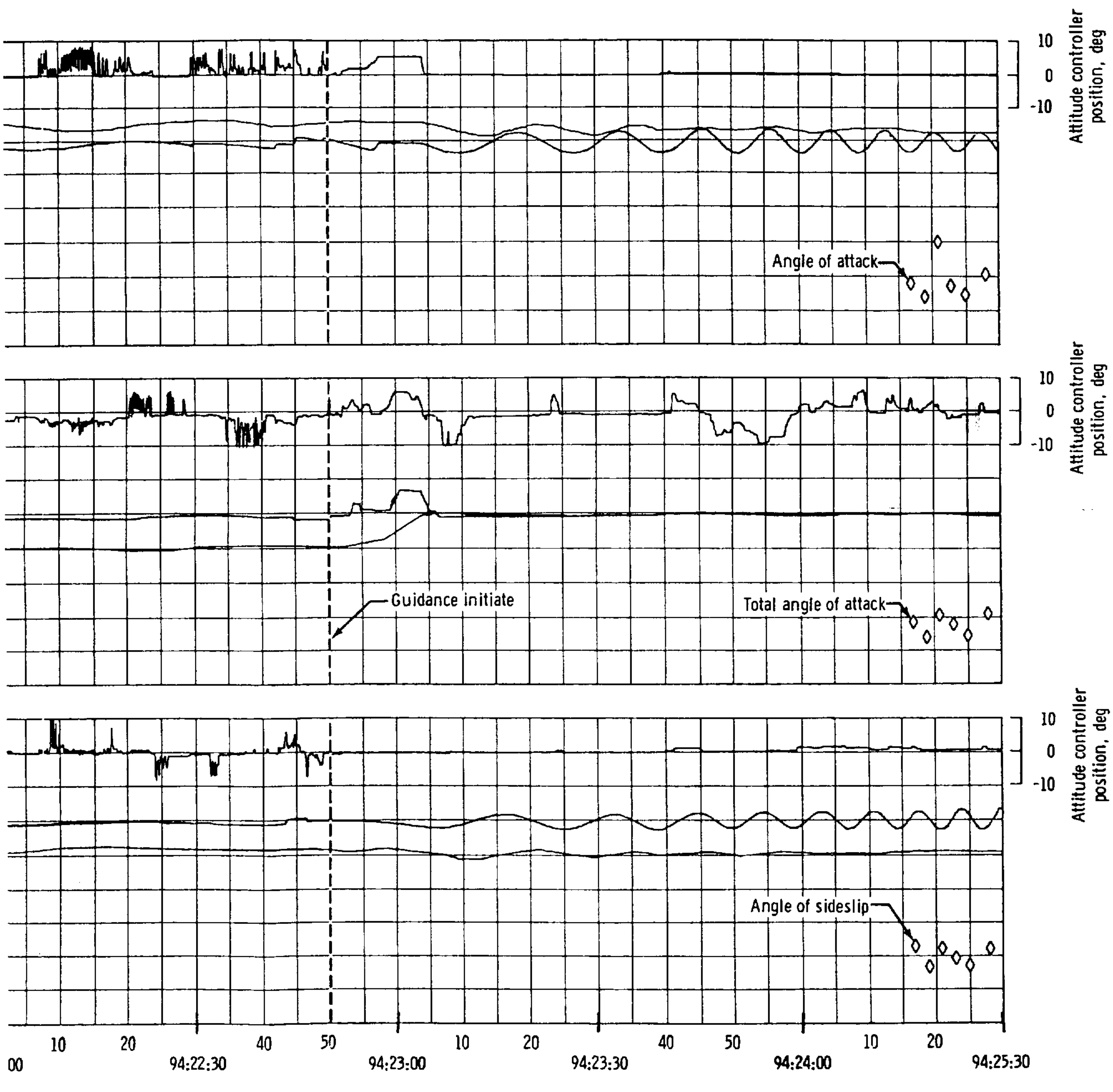


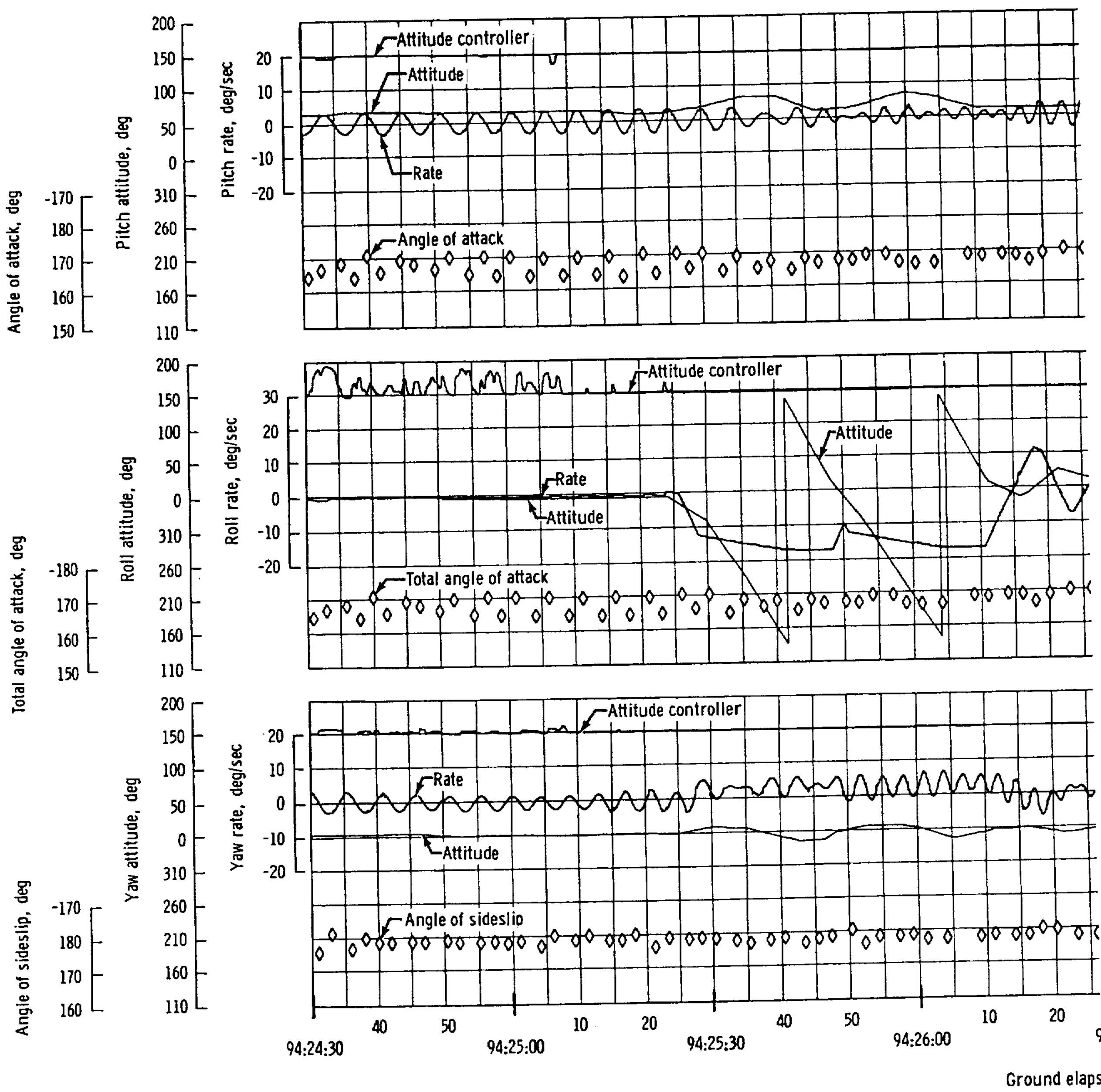
1

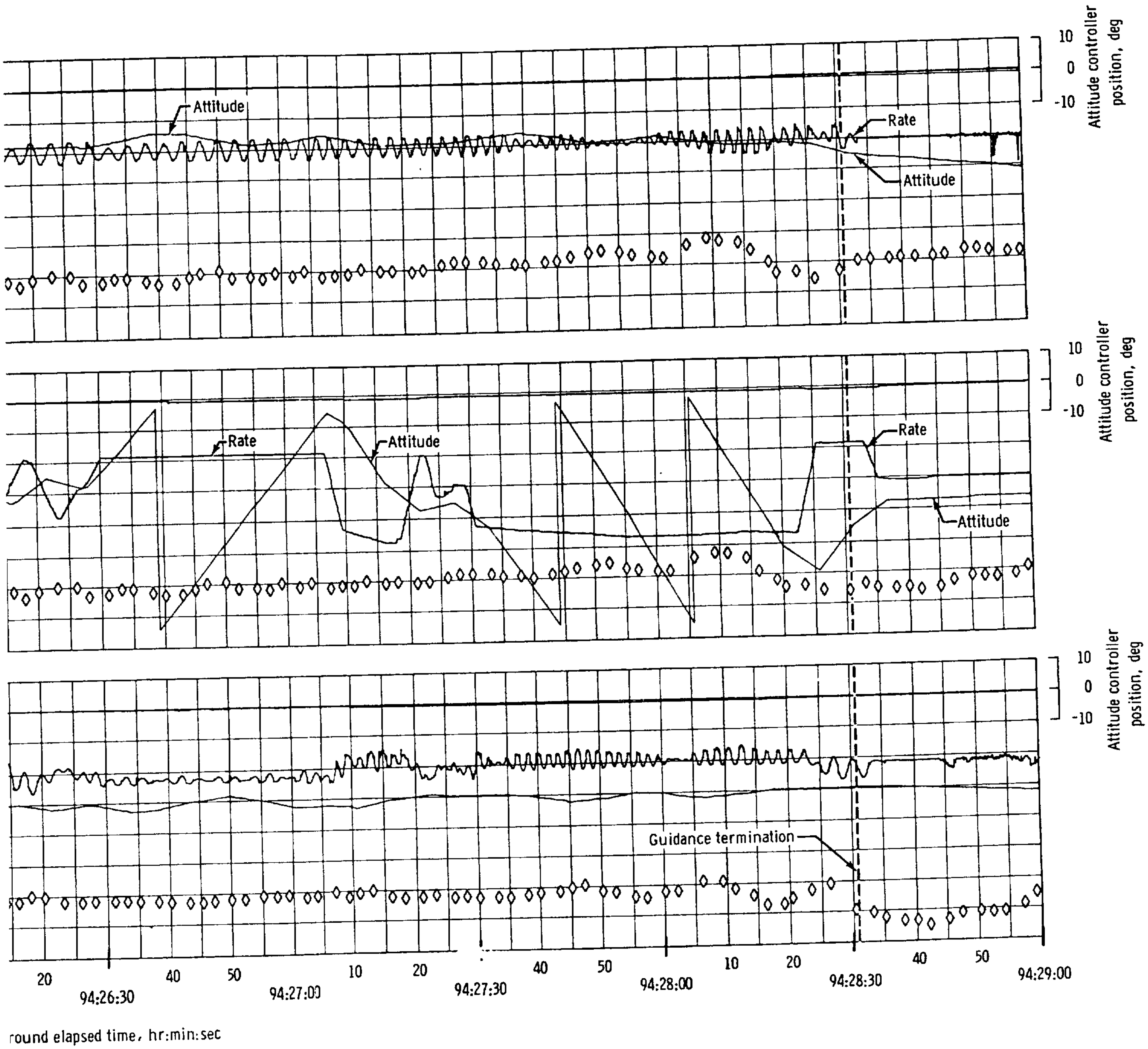


(a) 94:20:00 to 94:24:30 g.e.t.

Figure 5.1.5-10. - Reentry Control System performance.

NASA-S-66-11374 DEC 17

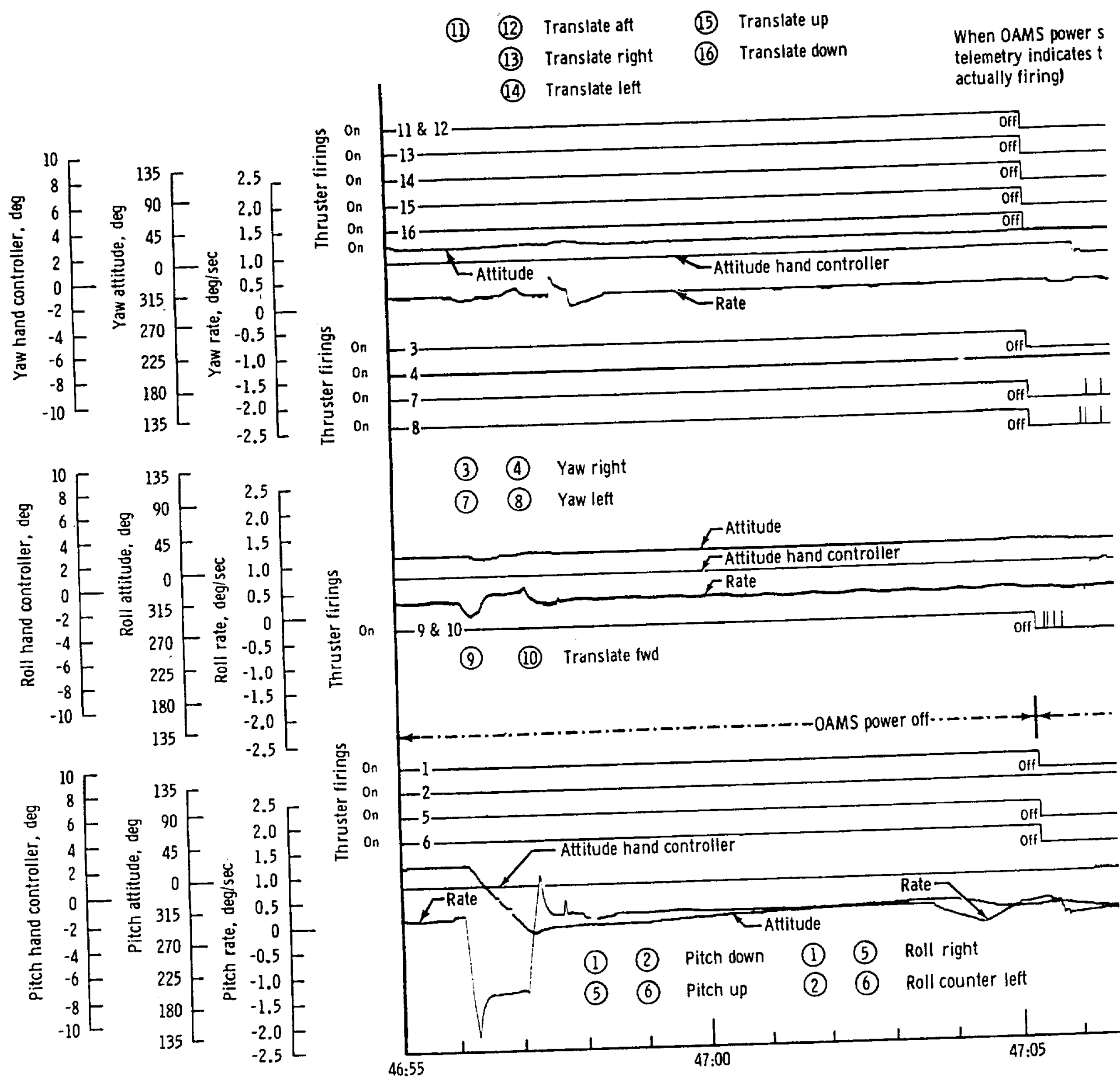


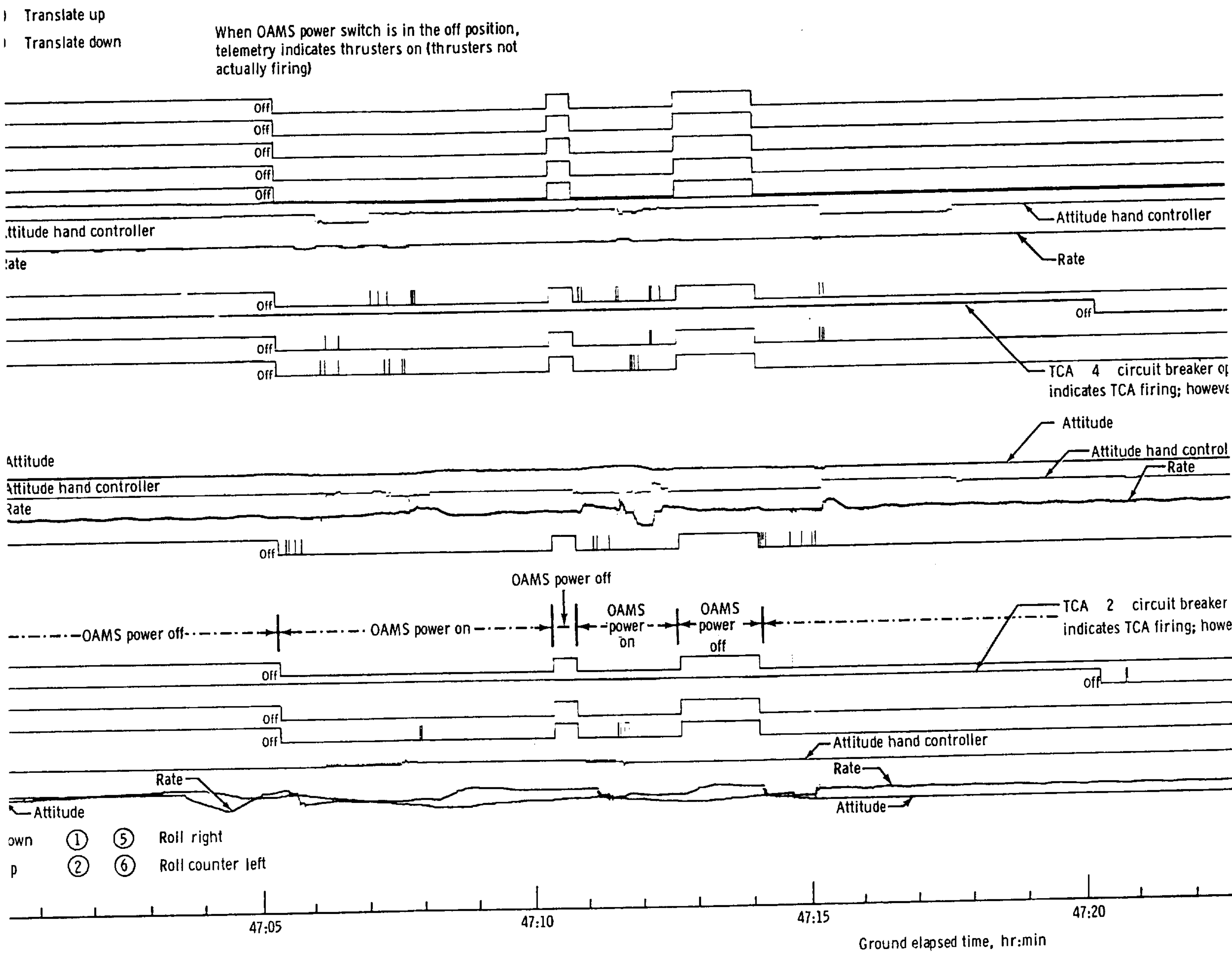


(b) 94:24:00 to 94:29:00 g.e.t.

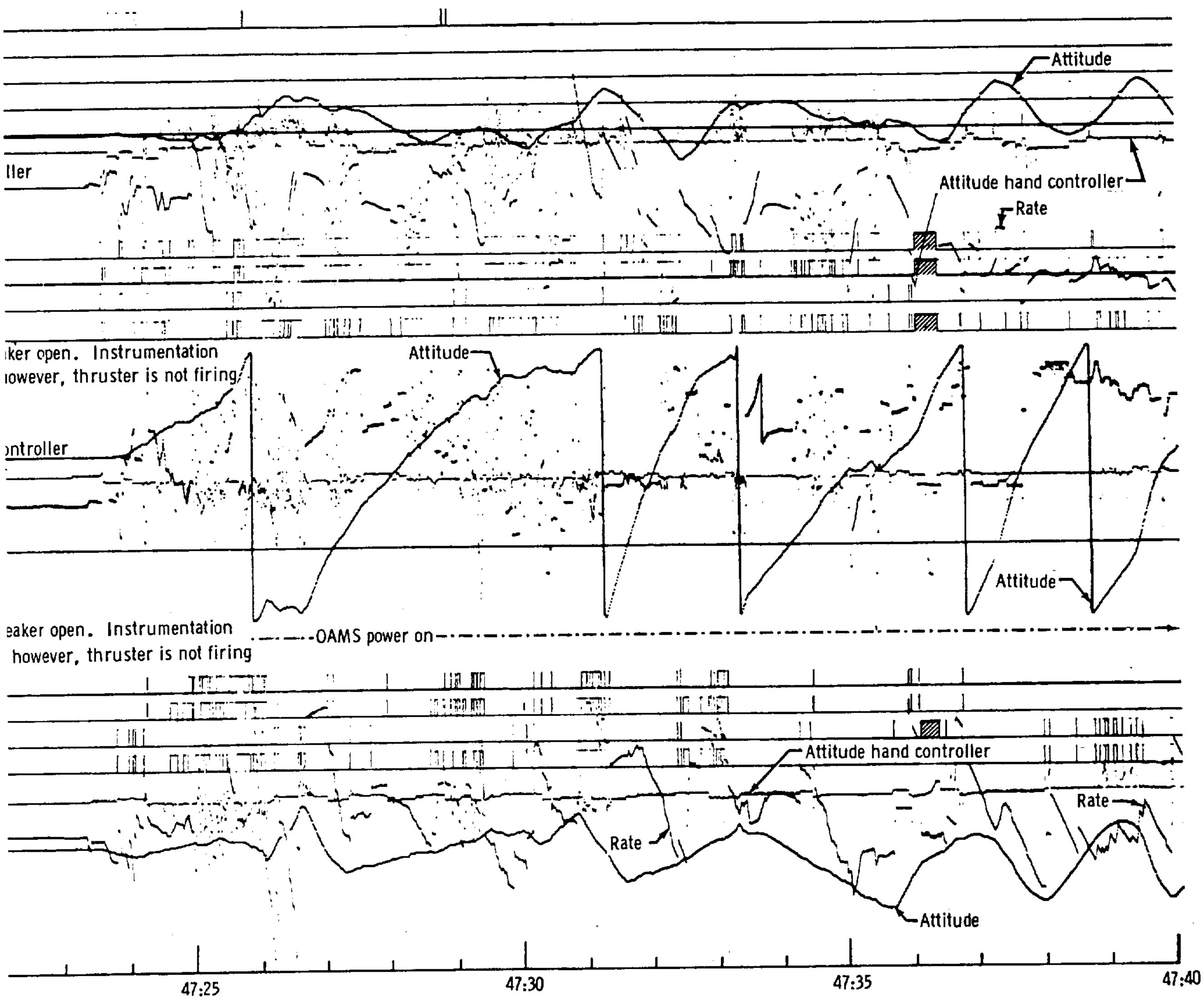
Figure 5.1.5-10. - Concluded.

NASA-S-66-11366 DEC 17





Translate up
Translate down
When OAMS power switch is in the off position, telemetry indicates thrusters on (thrusters not actually firing)
Off
Attitude hand controller
Rate
TCA 4 circuit breaker o
indicates TCA firing; howev
Attitude
Attitude hand control
OAMS power off
OAMS power on
OAMS power on
OAMS power off
TCA 2 circuit breaker
indicates TCA firing; howe
Roll right
Roll counter left
47:05
47:10
47:15
47:20
Ground elapsed time, hr:min



(a) 46 hr 55 min to 47 hr 40 min g.e.t.

Figure 5.1.5-11. - Spacecraft dynamics during the tether evaluation (gravity-gradient stabilization).

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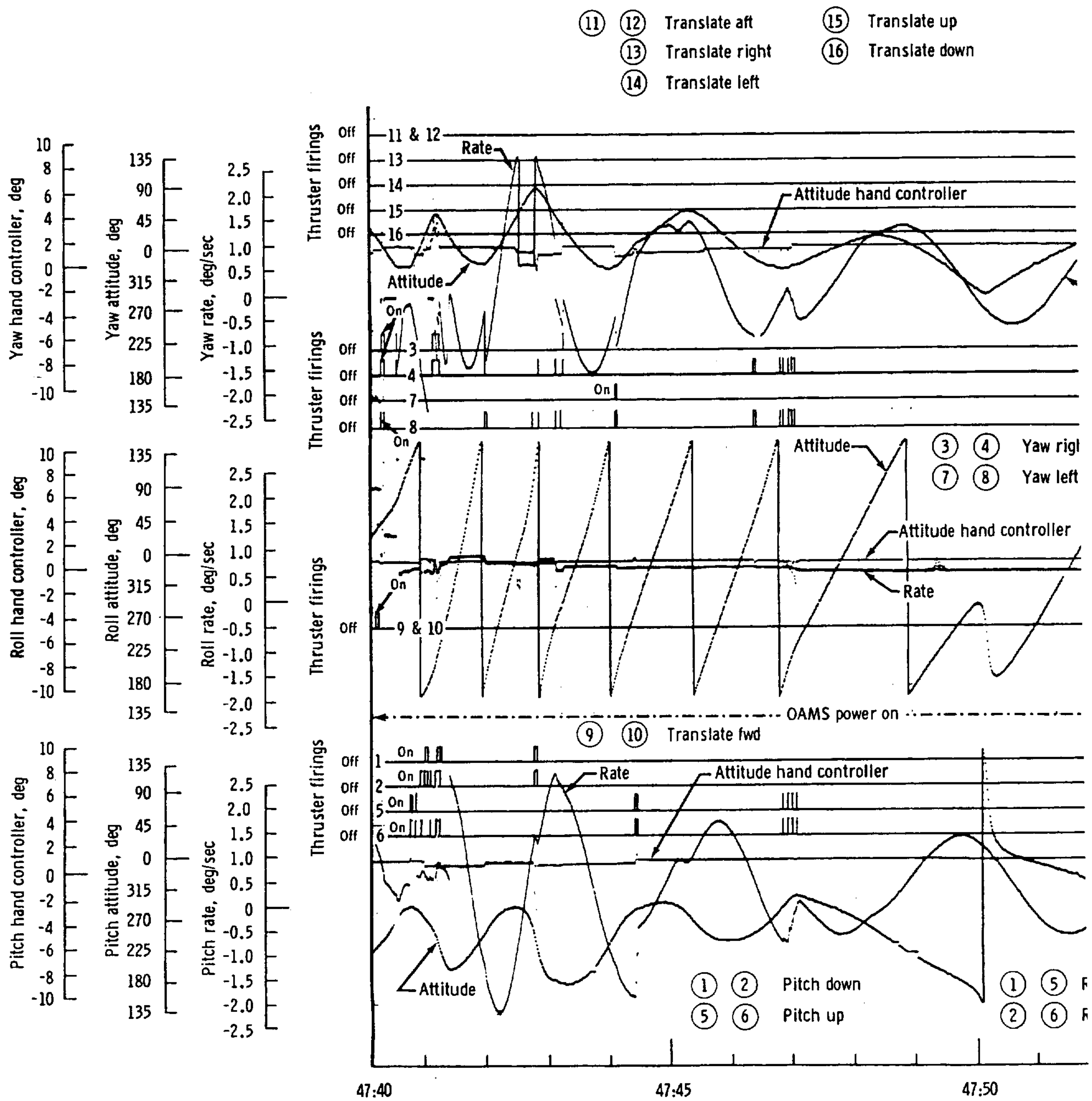
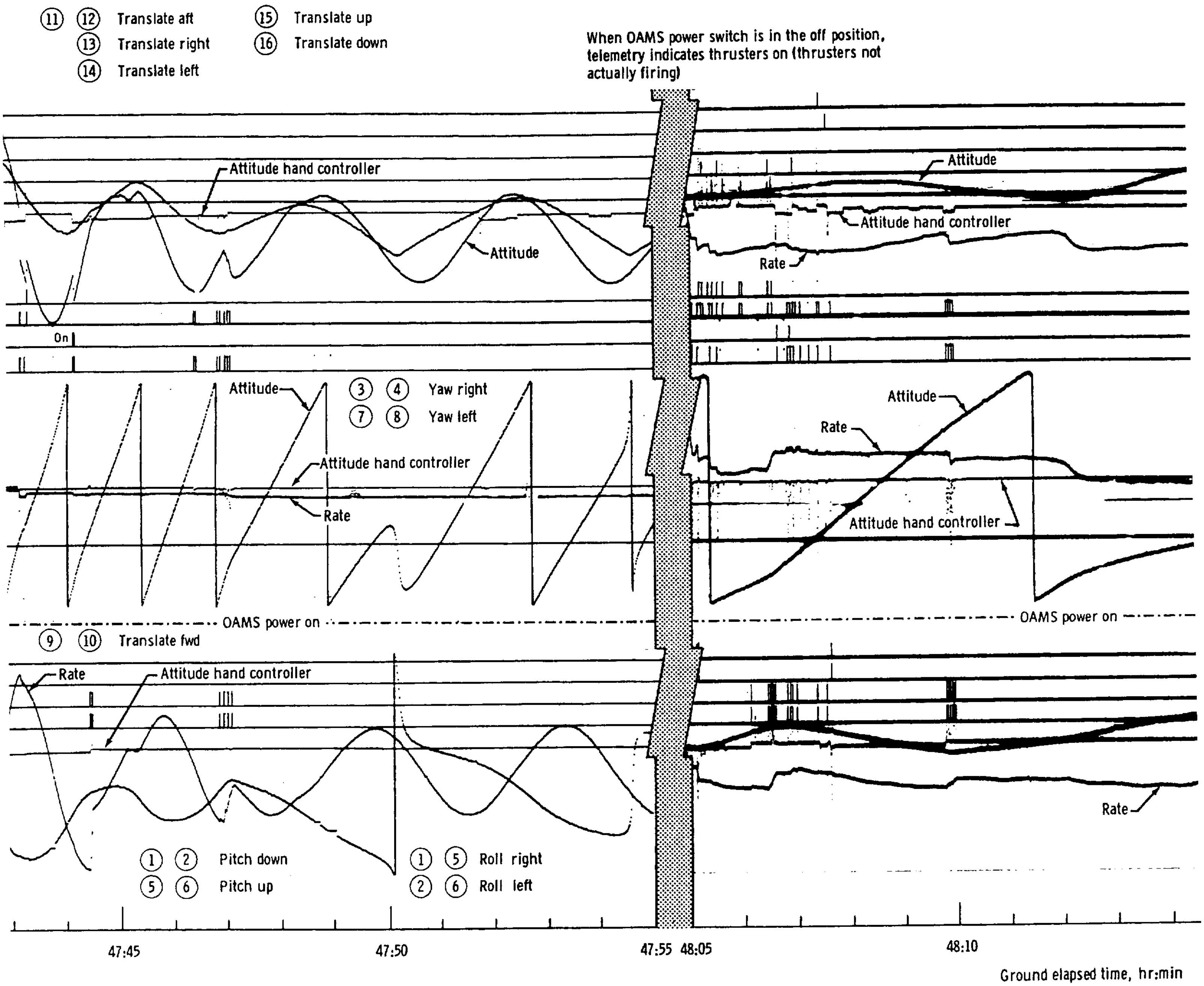
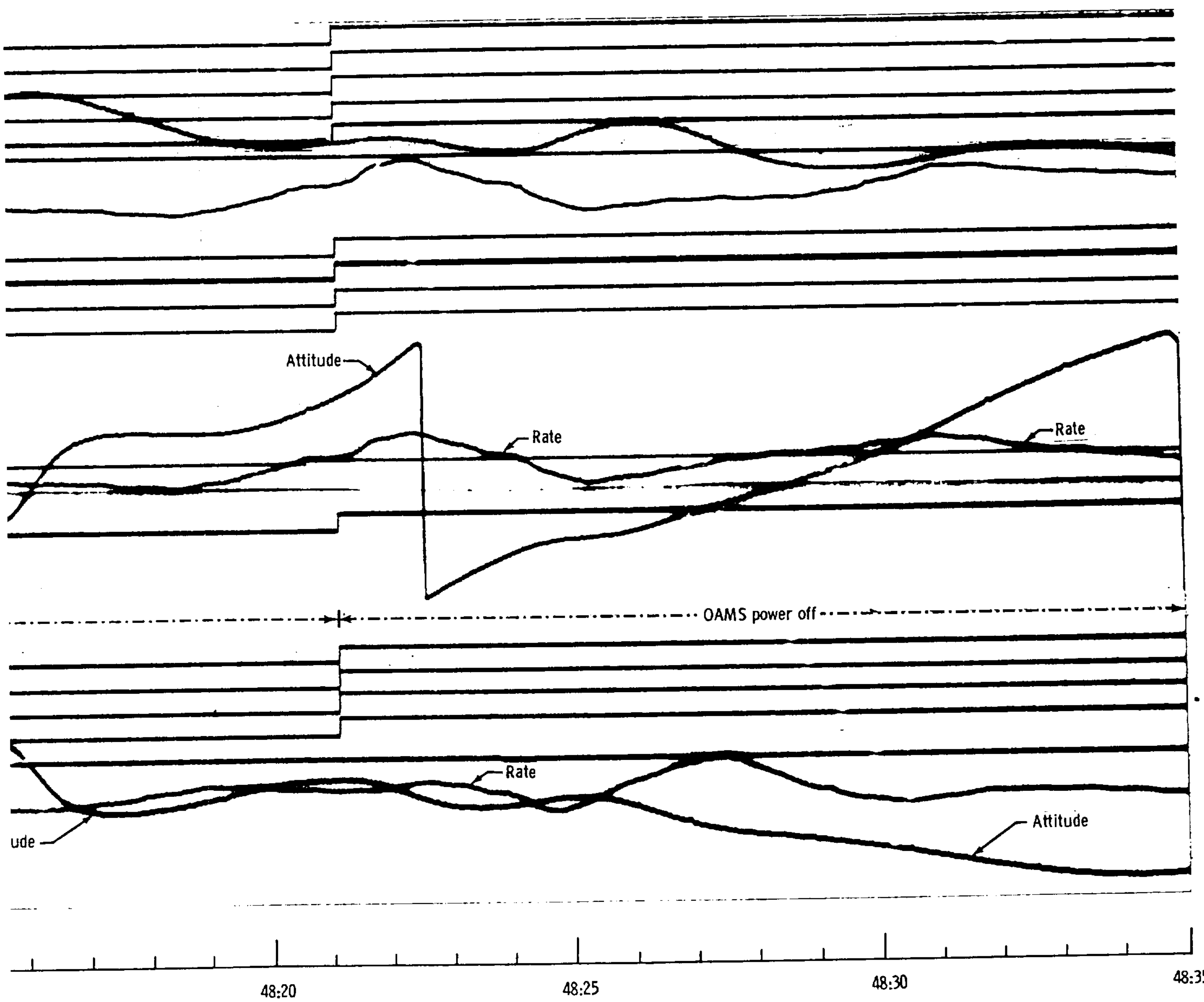




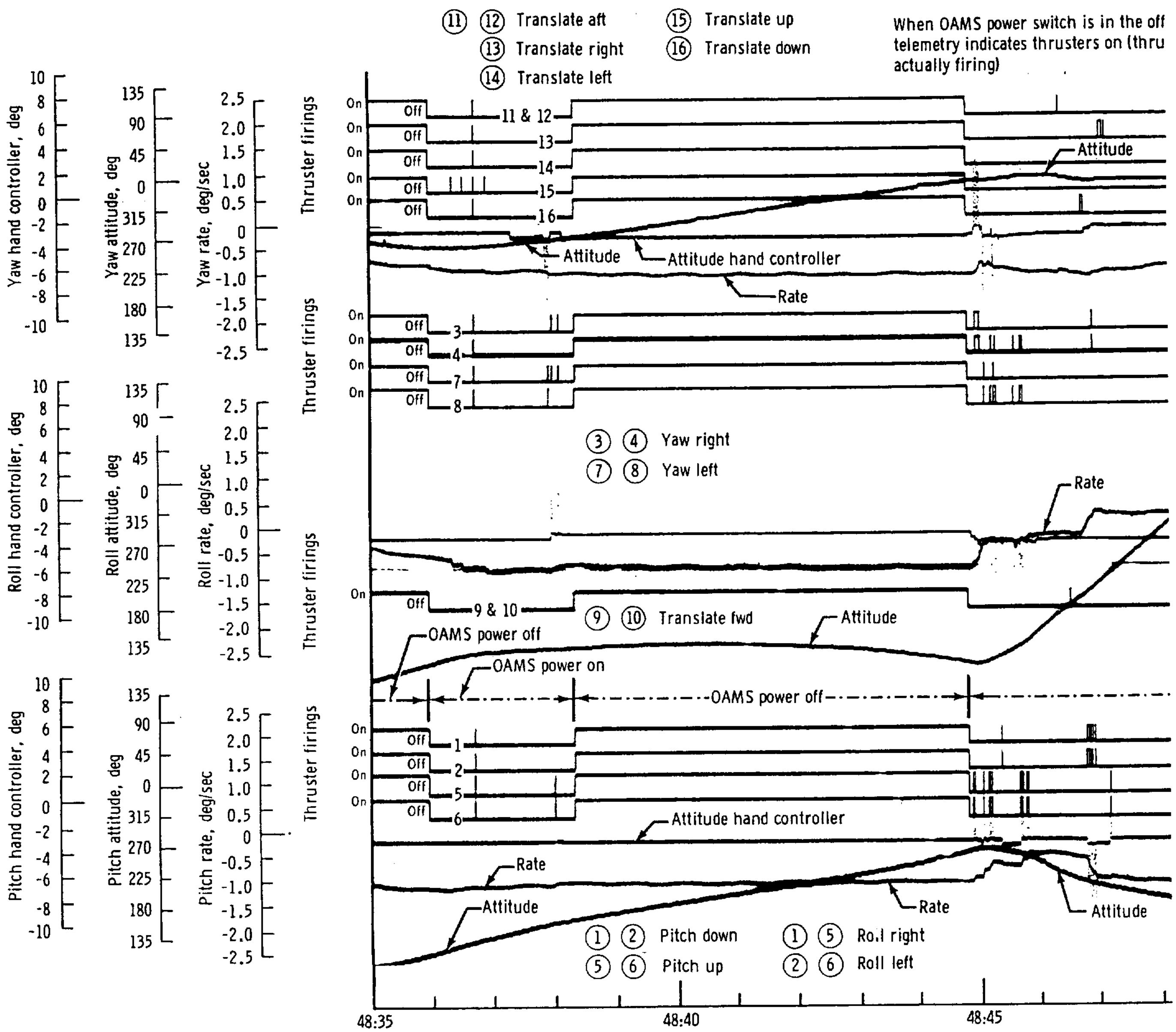
(11) (12) Translate aft
(13) Translate right
(14) Translate left
(15) Translate up
(16) Translate down
When OAMS power switch is in the off position, telemetry indicates thrusters on (thrusters not actually firing)
Attitude hand controller
Attitude
Attitude hand controller
Rate
On
Attitude
(3) (4) Yaw right
(7) (8) Yaw left
Attitude hand controller
Rate
Attitude
Rate
Attitude hand controller
OAMS power on
OAMS power on
(9) (10) Translate fwd
Rate
Attitude hand controller
Rate
(1) (2) Pitch down
(5) (6) Pitch up
(1) (5) Roll right
(2) (6) Roll left
47:45
47:50
47:55 48:05
48:10
Ground elapsed time, hr:min

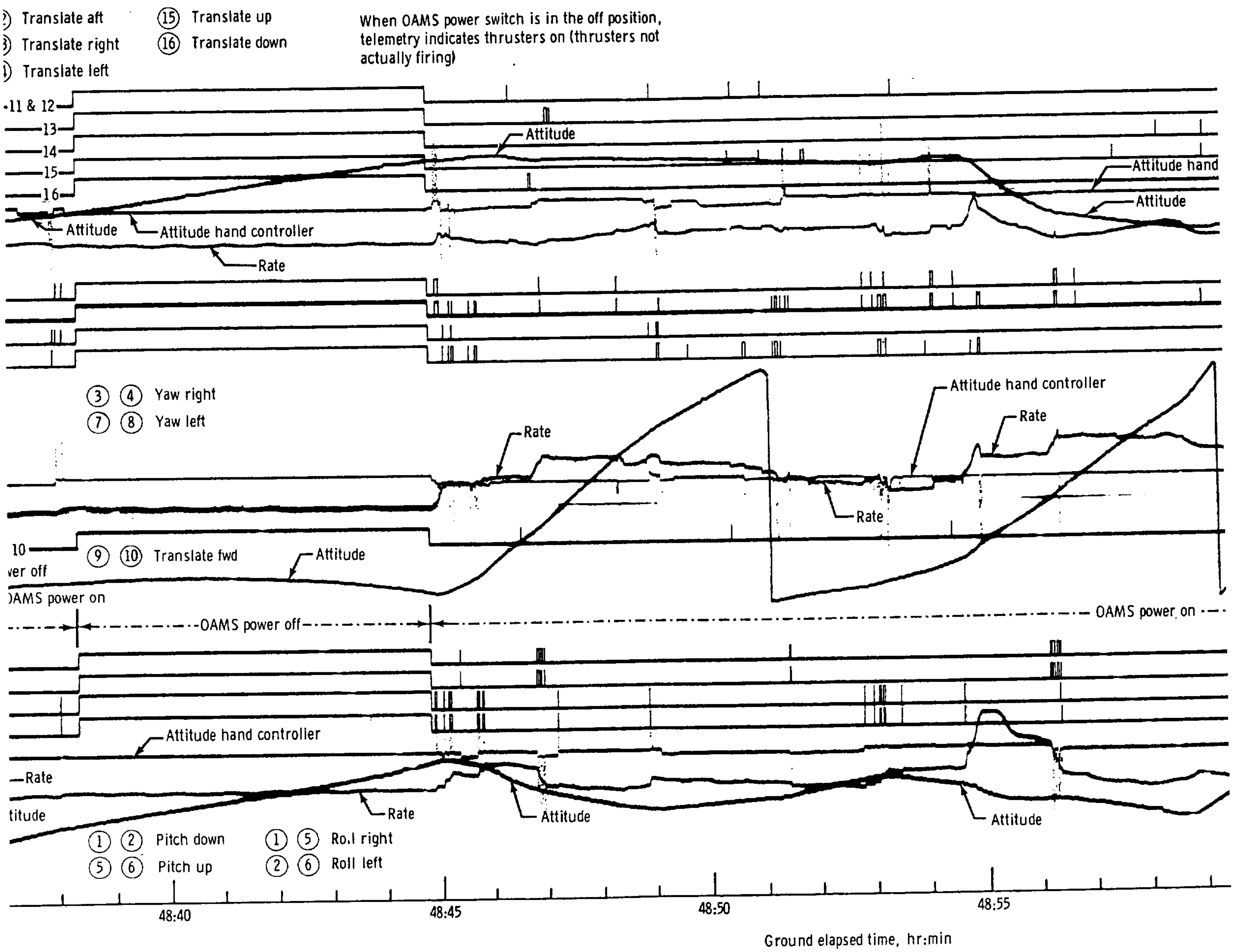


(b) 47 hr 40 min to 48 hr 35 min g.e.t.

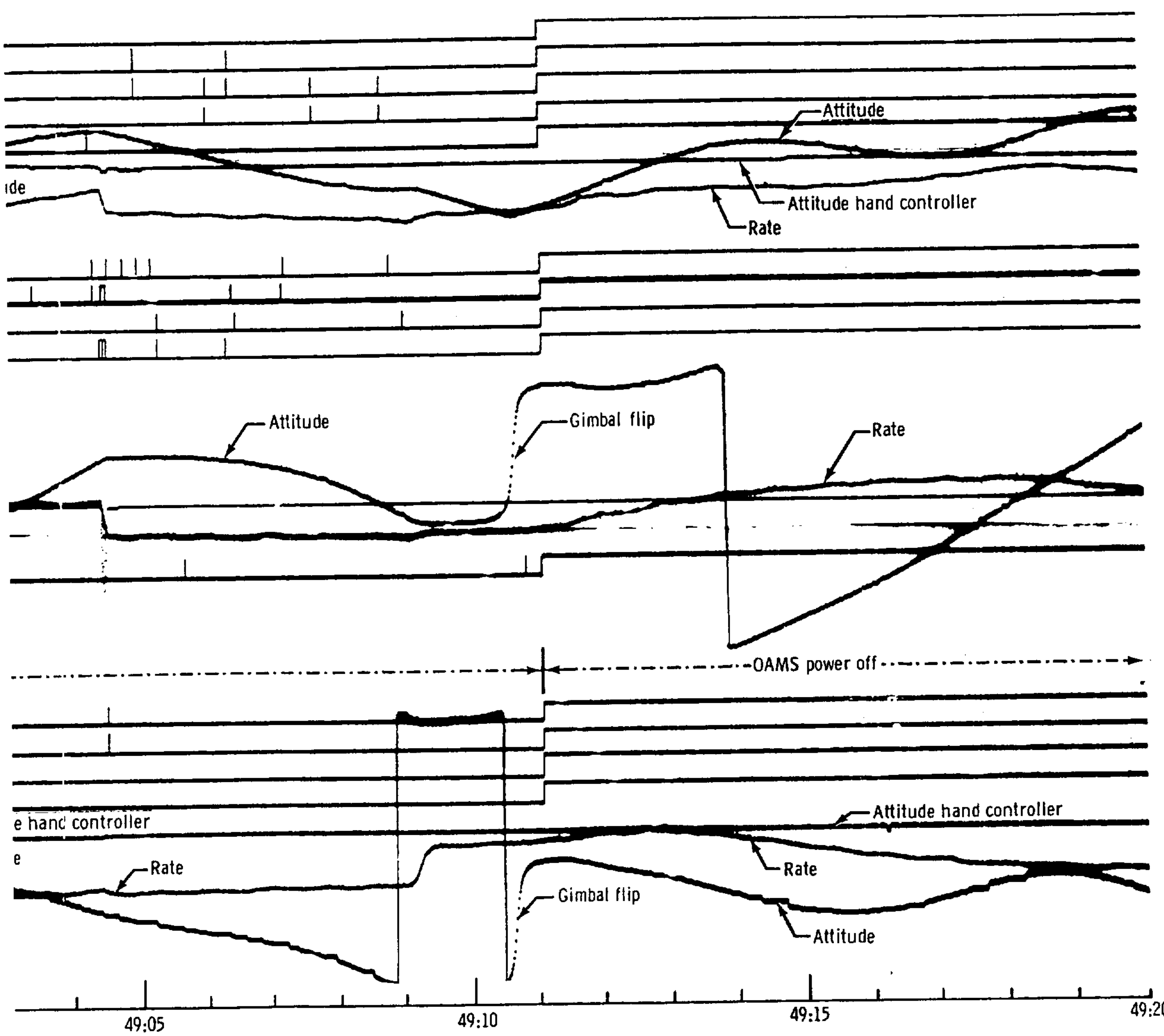
Figure 5.1.5-11. - Continued.

NASA-S-66-11369 DEC 17





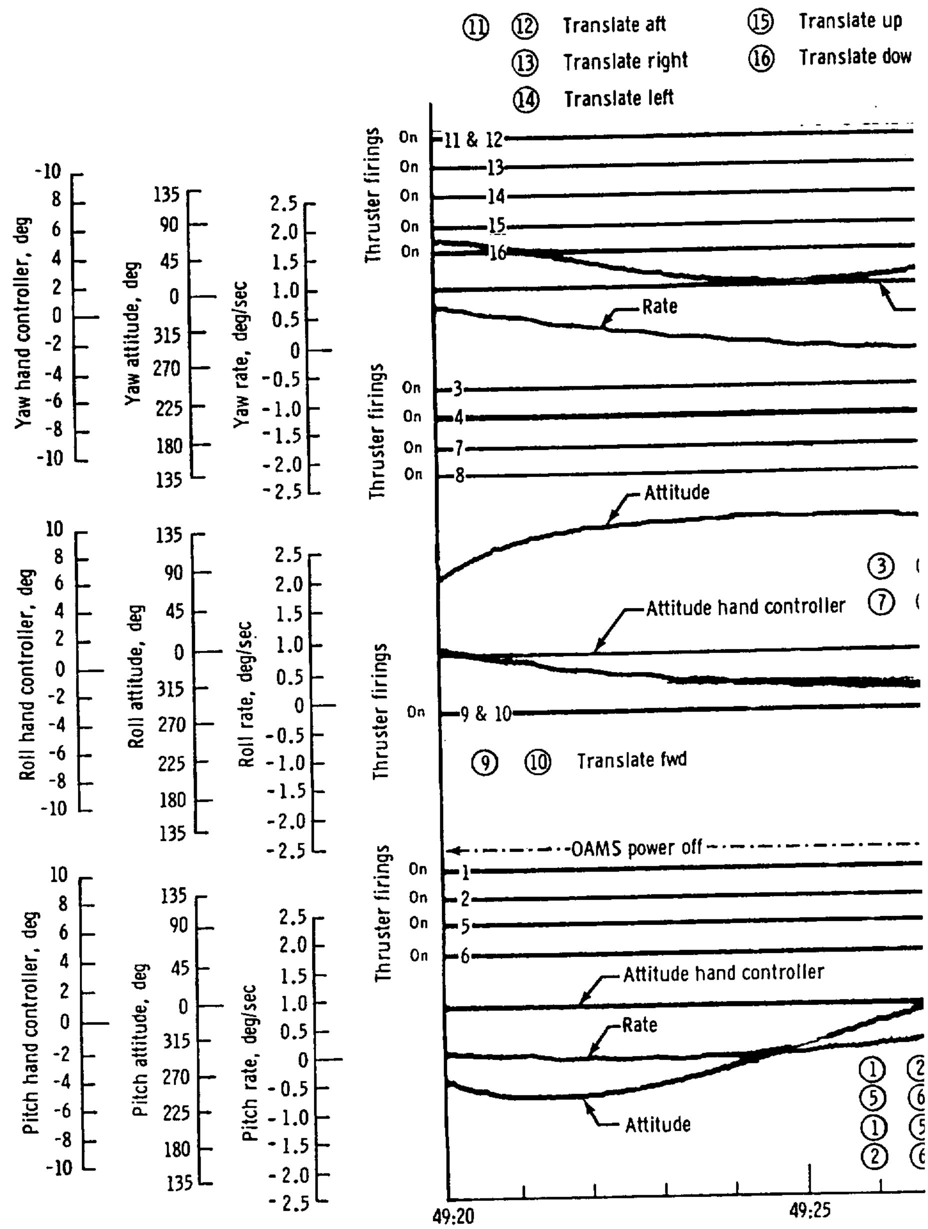
Translate aft
Translate right
Translate left
(15) Translate up
(16) Translate down
When OAMS power switch is in the off position, telemetry indicates thrusters on (thrusters not actually firing)
11 & 12
13
14
15
16
Attitude
Attitude hand controller
Rate
Attitude hand
(3) (4) Yaw right
(7) (8) Yaw left
10
(9) (10) Translate fwd
OAMS power on
OAMS power off
(1) (2) Pitch down
(5) (6) Pitch up
(1) (5) Roll right
(2) (6) Roll left
48:40
48:45
48:50
48:55
Ground elapsed time, hr:min



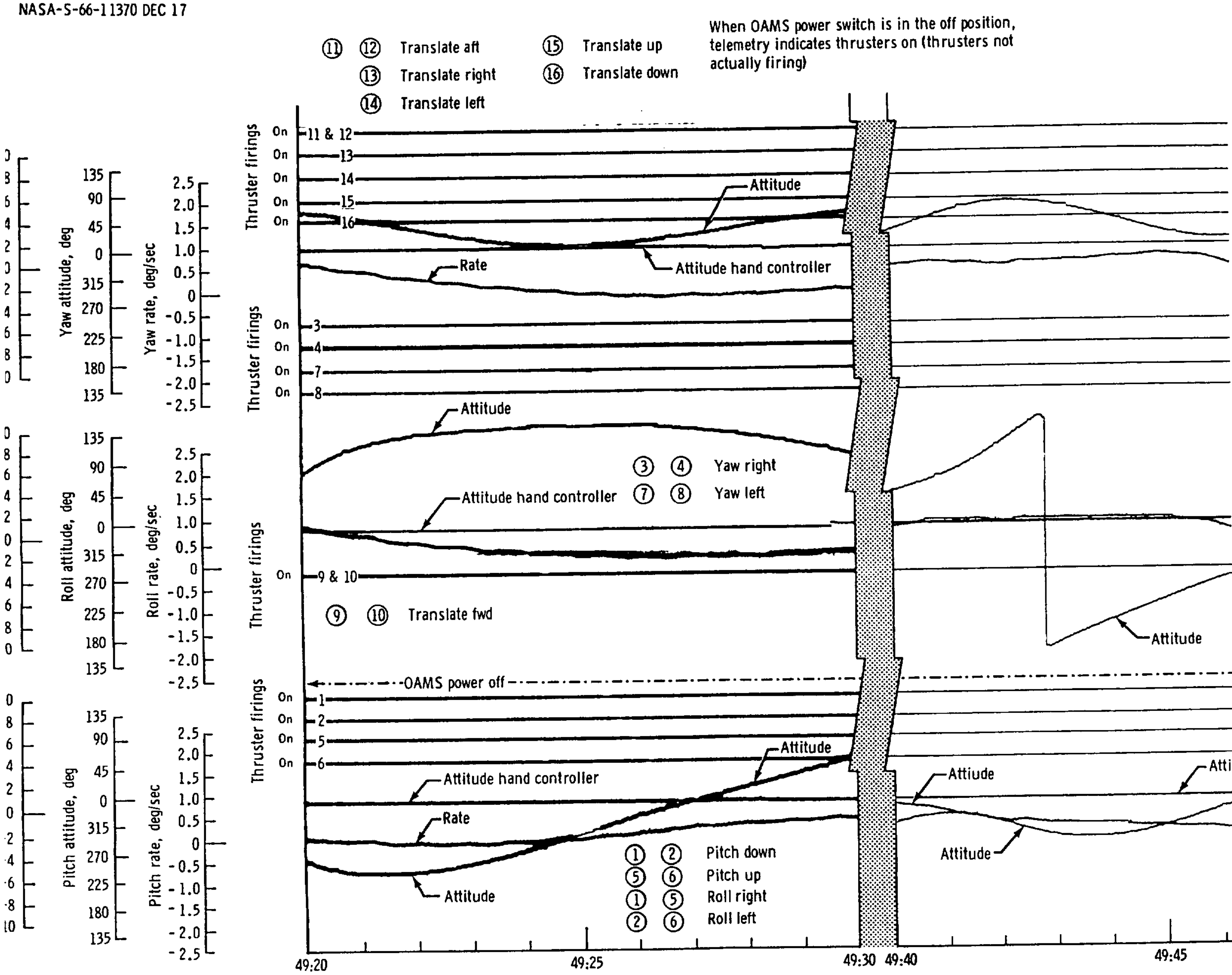
(c) 48 hr 35 min to 49 hr 20 min g.e.t.

Figure 5.1.5-11. - Continued.

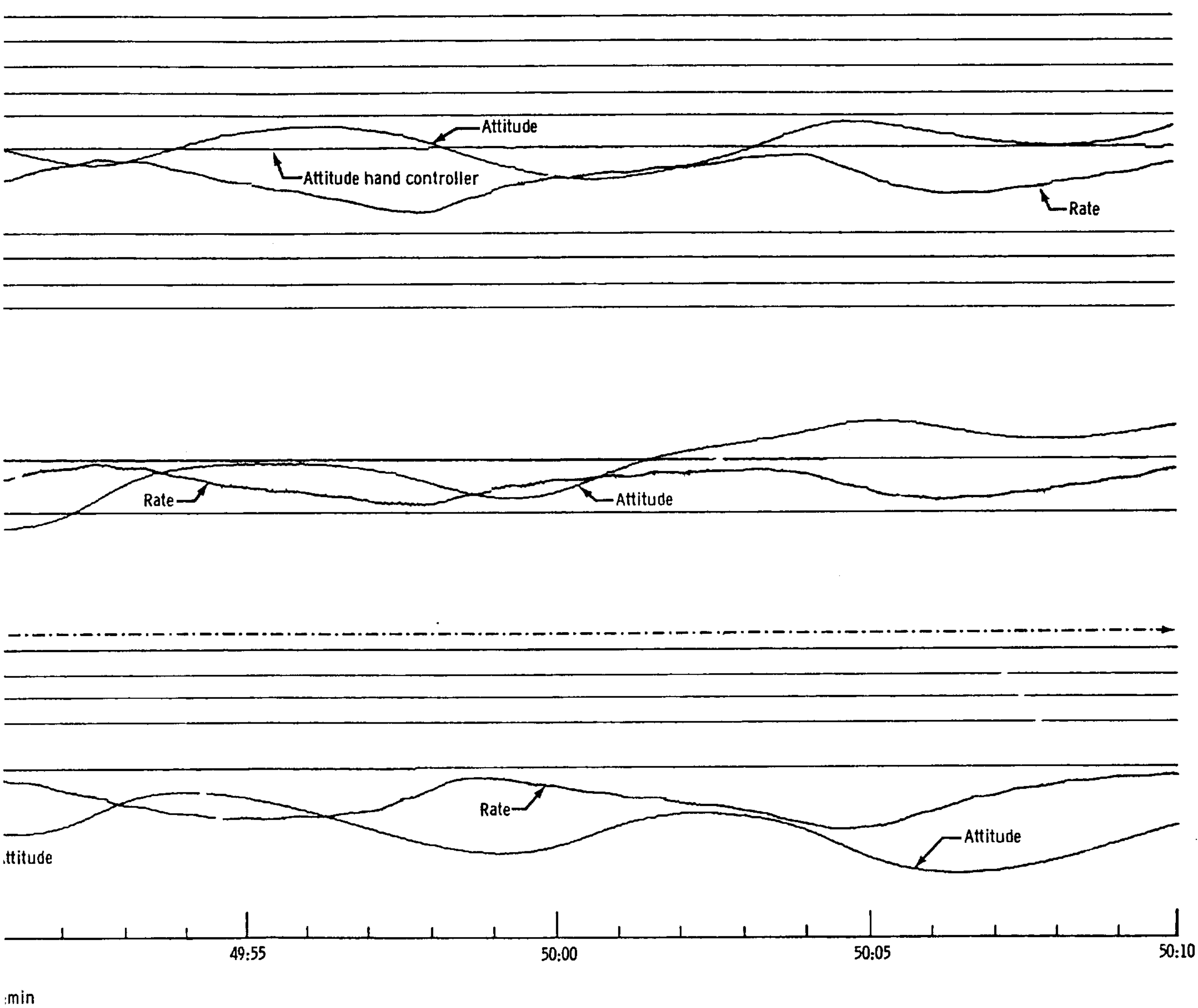
NASA-S-66-11370 DEC 17



NASA-S-66-11370 DEC 17



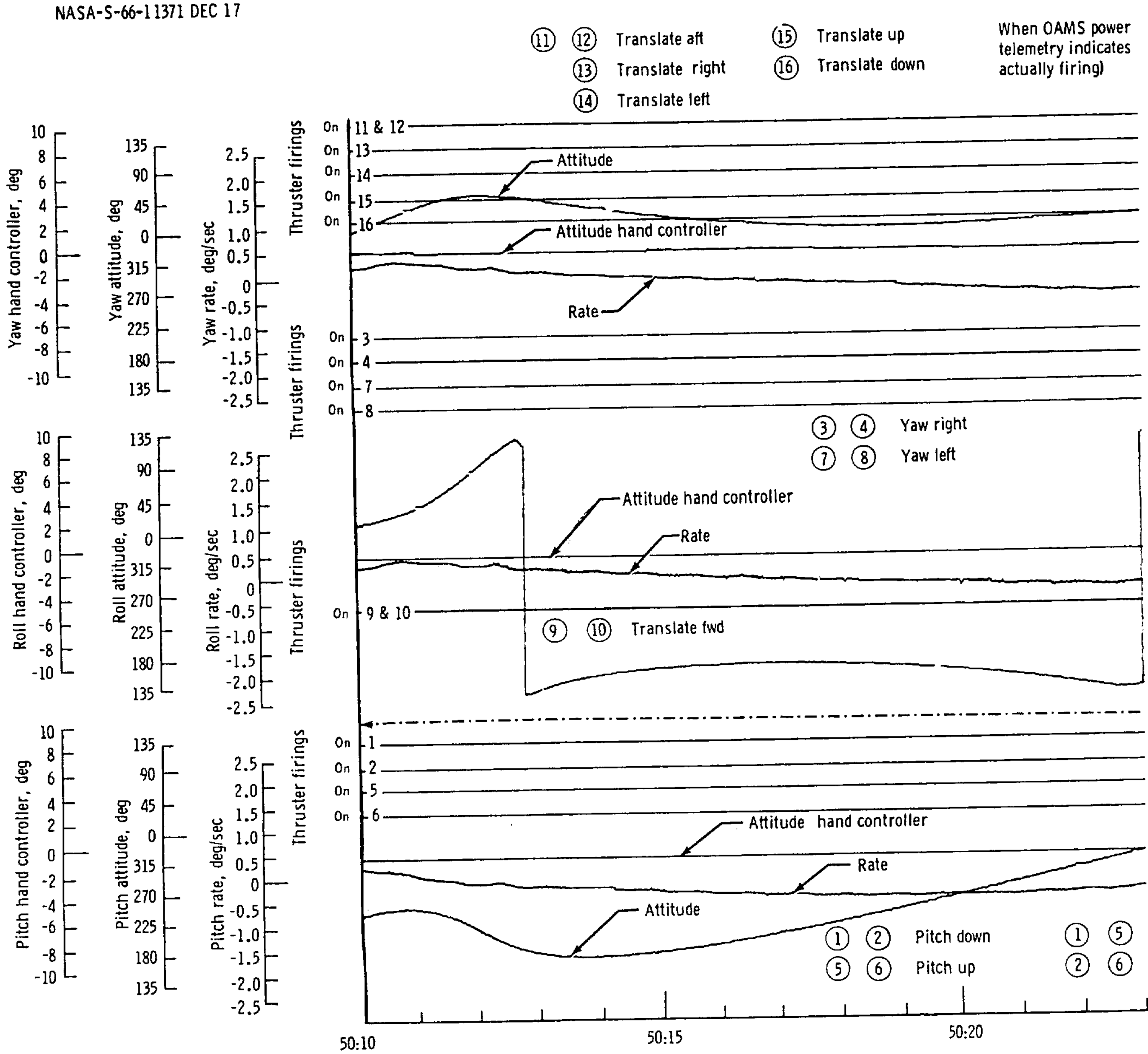
2

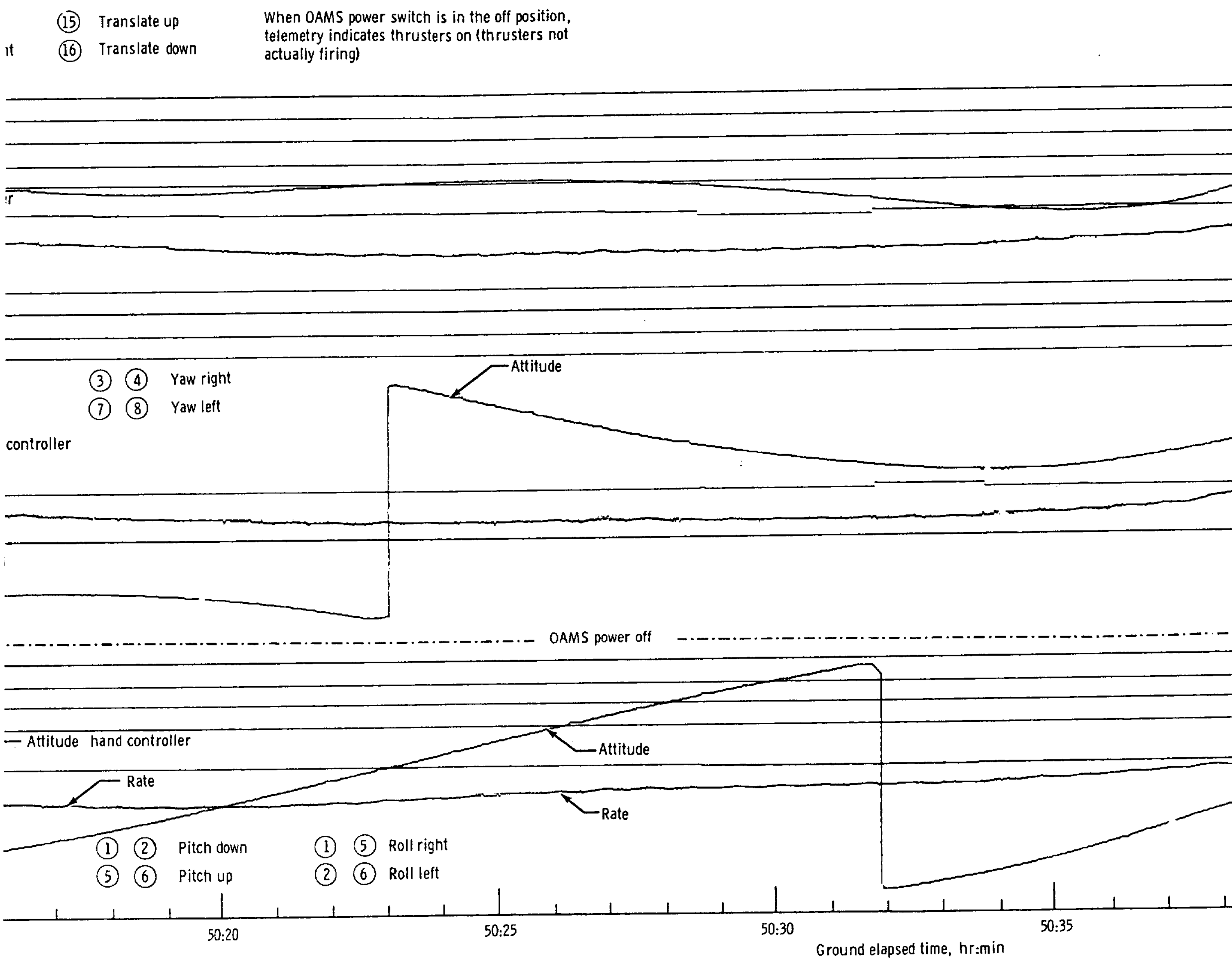


(d) 49 hr 20 min to 50 hr 10 min g.e.t.

Figure 5.1.5-11. - Continued.

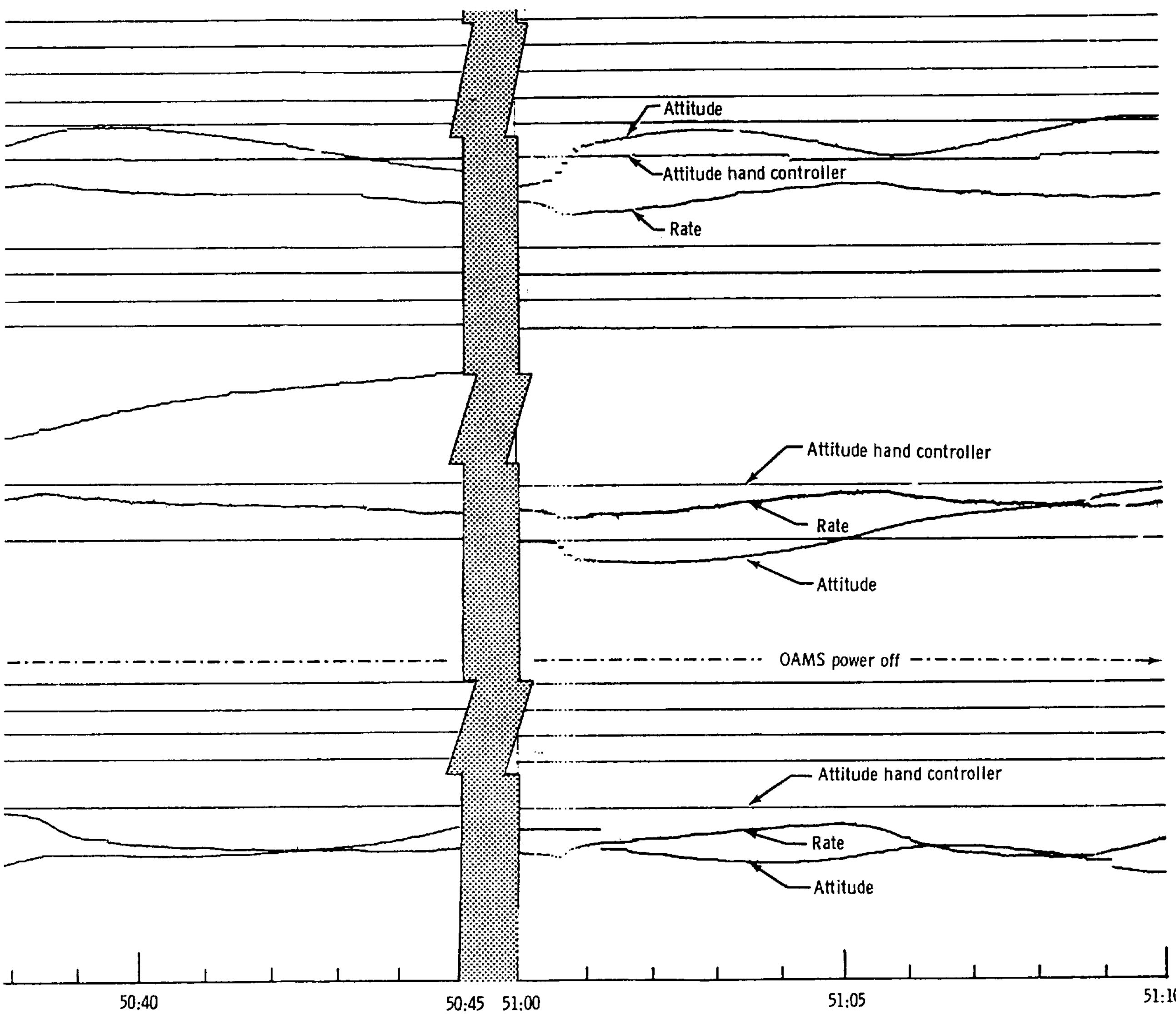
NASA-S-66-11371 DEC 17





(15) Translate up
(16) Translate down
When OAMS power switch is in the off position, telemetry indicates thrusters on (thrusters not actually firing)
(3) (4) Yaw right
(7) (8) Yaw left
controller
Attitude
OAMS power off
Attitude hand controller
Rate
Attitude
Rate
(1) (2) Pitch down
(5) (6) Pitch up
(1) (5) Roll right
(2) (6) Roll left
50:20
50:25
50:30
50:35
Ground elapsed time, hr:min

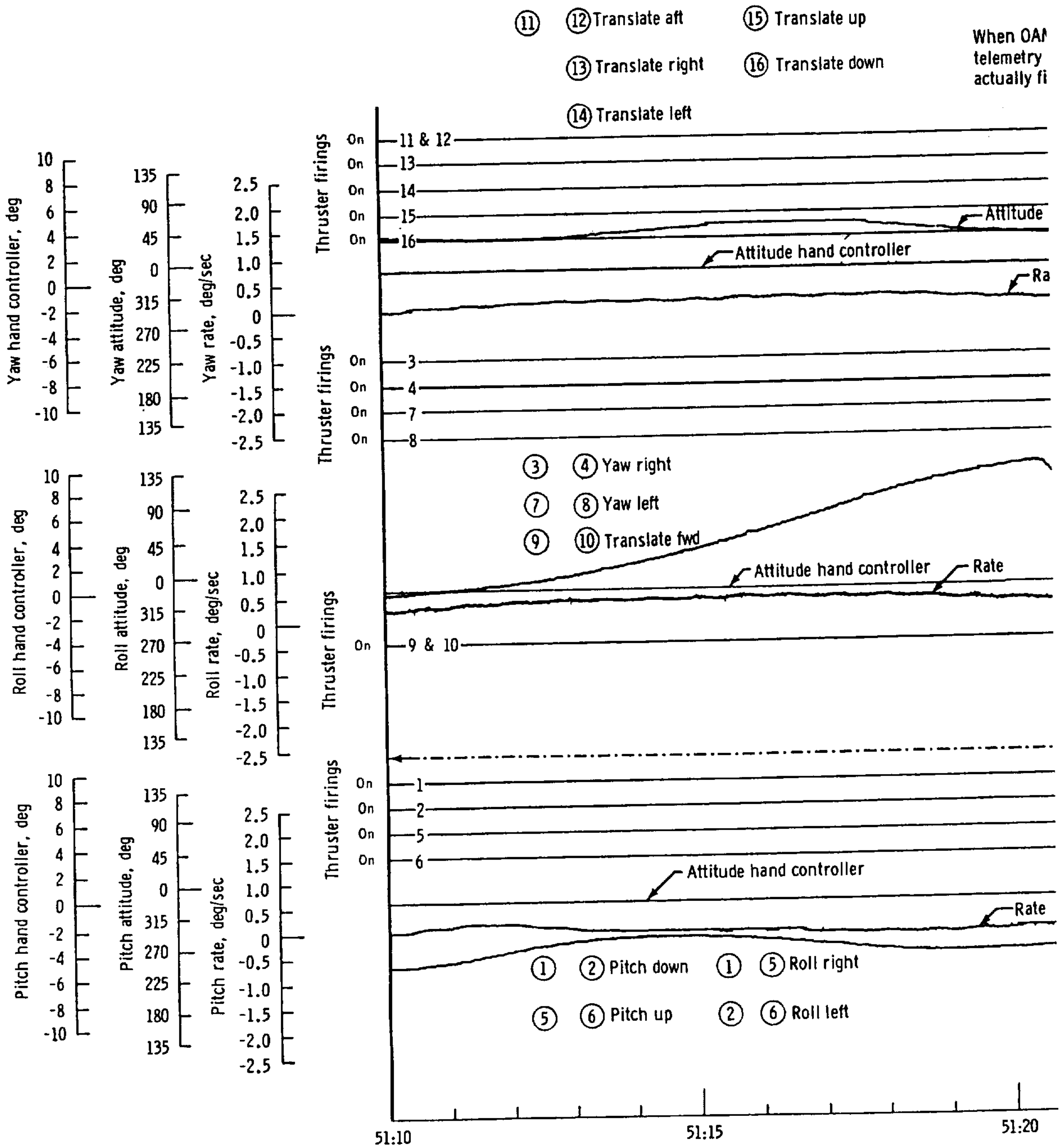
2

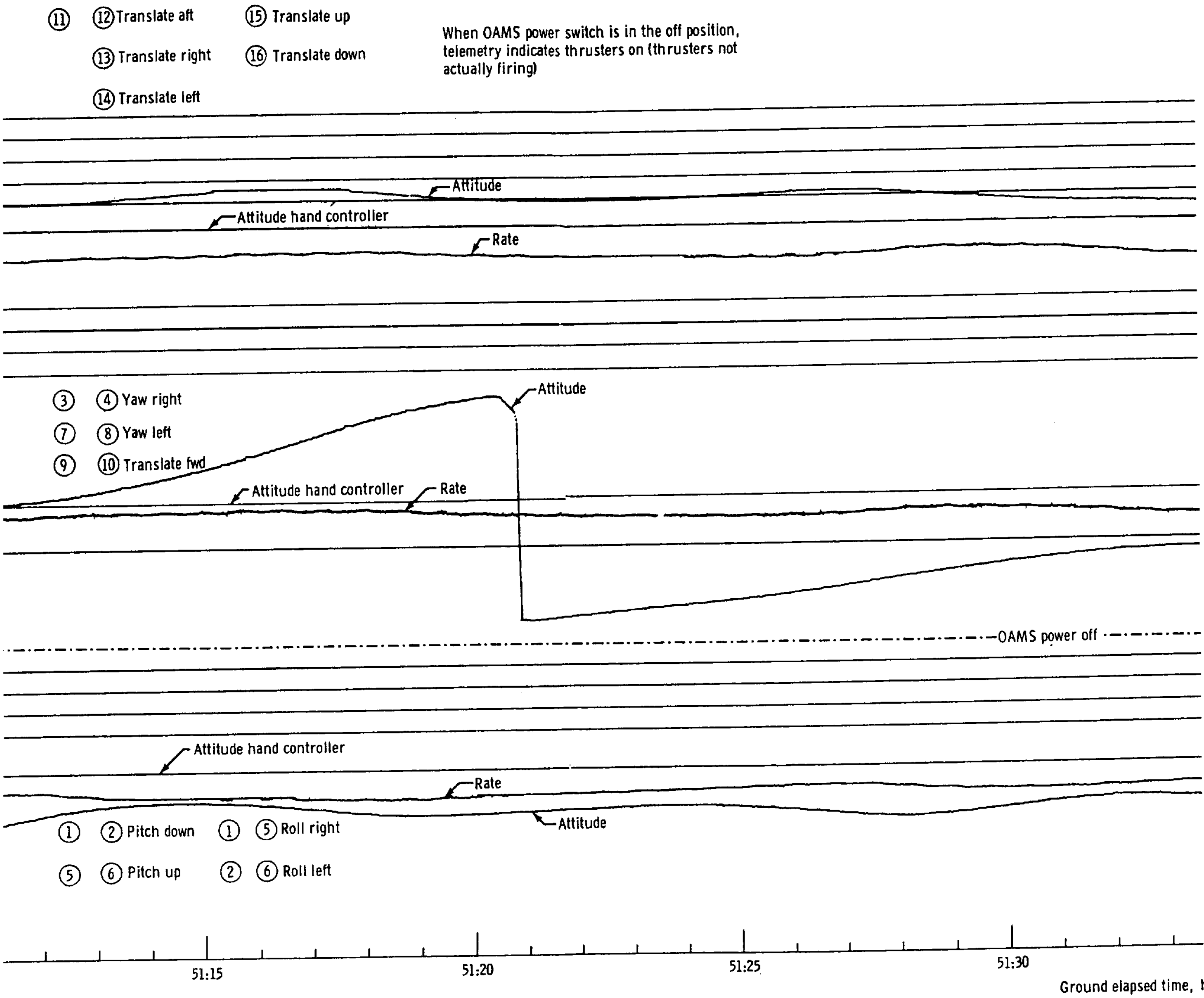


(e) 50 hr 10 min to 51 hr 10 min, g.e.t.

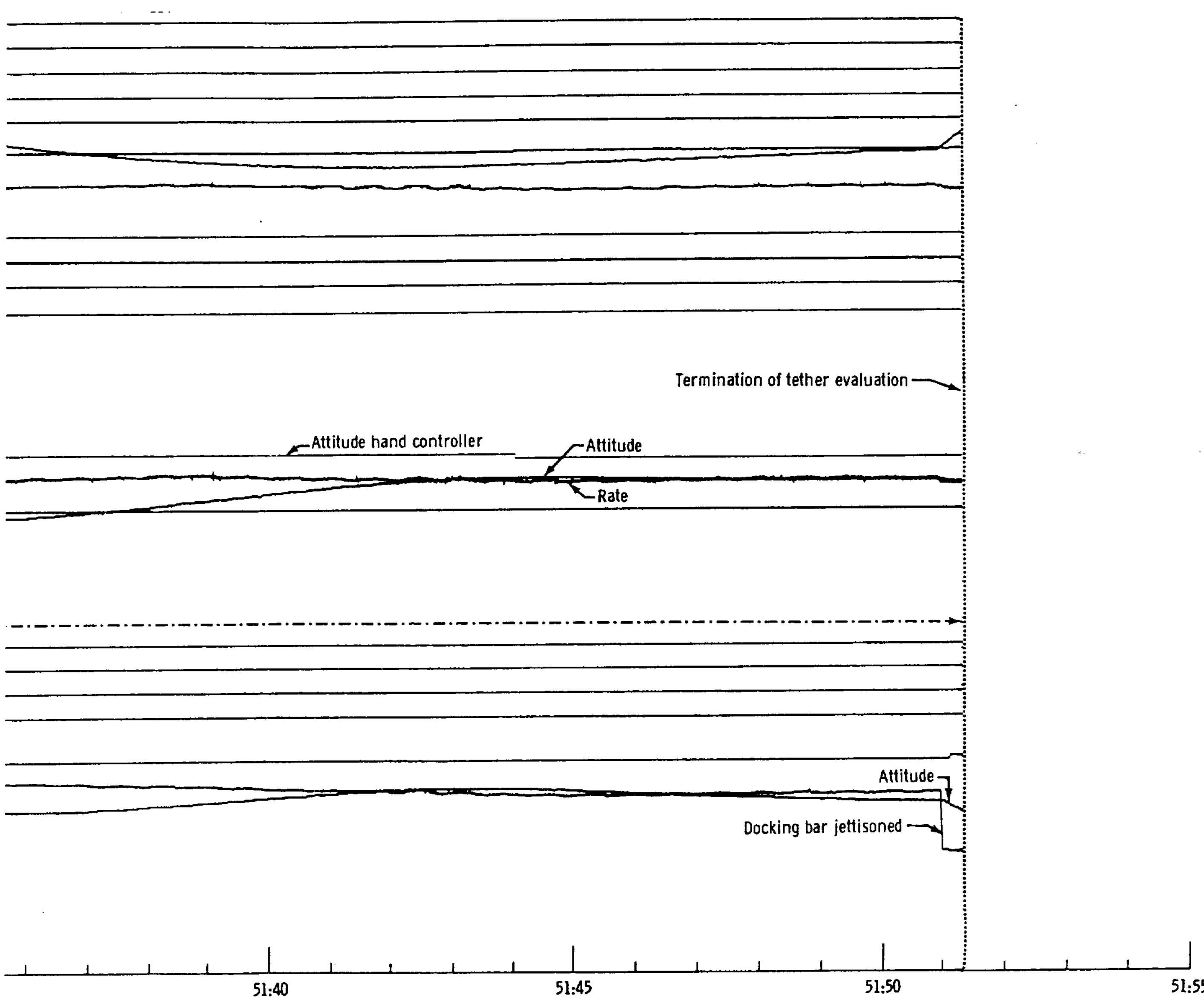
Figure 5.1.5-11. - Continued.

NASA-S-66-11372 DEC 17





⑪ ⑫ Translate aft
⑬ Translate right
⑭ Translate left
⑮ Translate up
⑯ Translate down
When OAMS power switch is in the off position, telemetry indicates thrusters on (thrusters not actually firing)
Attitude
Attitude hand controller
Rate
③ ④ Yaw right
⑦ ⑧ Yaw left
⑨ ⑩ Translate fwd
Attitude
Attitude hand controller
Rate
OAMS power off
Attitude hand controller
Rate
Attitude
① ② Pitch down
⑤ ⑥ Pitch up
① ⑤ Roll right
② ⑥ Roll left
51:15
51:20
51:25
51:30
Ground elapsed time, h



(f) 51 hr 10 min to 51 hr 55 min g.e.t.

Figure 5.1.5-11. - Concluded.

5.1.6 Time Reference System

An analysis of available data indicates that all components of the Time Reference System (TRS) performed according to specification. The electronic timer began counting elapsed time approximately 22 milliseconds after lift-off. At 334 446.003 seconds of elapsed time (92:54:06.003 g.e.t.), the electronic timer indicated 334 445.468 seconds, which represents an error of approximately 0.535 of a second or 1.6 parts per million, well within the specification requirement of 10 parts per million at 25 $\pm 10^{\circ}$ C. In addition, the electronic timer initiated the automatic retrofire sequence correctly at 93:59:58 g.e.t.

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 digital clock was accidentally turned off by the flight crew during EVA preparation. This was discovered a short time later, and the clock was started and reset without difficulty. The clock read 95:43:15 when shut off after recovery. 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. Satisfactory timing on the tapes from the onboard voice tape recorder indicated normal operation of the time correlation buffer.

5.1.7 Electrical System

The electrical power system supplied the required power for the mission, although product-water storage problems, similar to those experienced on Spacecraft 7, were a constant problem. The fuel-cell system completed the mission with four of the six stacks in operation; however, two of the four were supplying only 24 percent of the load.

5.1.7.1 Silver-zinc batteries.- The squib and common-control batteries performed normally throughout the mission. After equipment-section separation and prior to retrofire, the reentry-section batteries supplied the main bus with a satisfactory 23.8 volts at the required 35.5 amperes.

5.1.7.2 Fuel-cell power system.-

5.1.7.2.1 Prelaunch history: The first and second fuel cell activations were performed on September 28 and November 10, 1966, respectively. The performances at second activation are shown in figures 5.1.7-1. These performances were within the range achieved by other sections that had experienced similar storage periods after initial activation (43 days). All other prelaunch history of the sections appeared to be normal.

5.1.7.2.2 Inflight performance: The first indication that the fuel-cell power system was not functioning properly occurred when the oxygen-to-water differential-pressure warning lights came on briefly at 5 hours 48 minutes g.e.t. for no apparent reason. Thereafter, the lights came on for extended periods until 40 hours 48 minutes g.e.t., at which time they came on and remained on. There was no noticeable degradation in the performance of the stacks until 35:29:19 g.e.t., when the output of stack B in section 2 began to decline. This output continued to decline for 1 hour 45 minutes until the stack could no longer support any of the spacecraft load, after which it was removed from the bus. At that time, a review of the data showed that the reference water pressure remained constant instead of decreasing when drinking water was extracted. The conclusion during the mission was that the bladder in the fuel-cell water tank had bottomed-out. This meant that unless water was extracted by the crew or water was removed by purging, a zero oxygen-to-water pressure differential would exist across the water separators. Consequently, the water produced by the fuel cells would not be removed and would eventually cause flooding and system failure. The crew was instructed by the flight controllers to perform 30-second oxygen purges approximately each revolution and to extract drinking water as frequently as practical. Performance of the remaining stacks was satisfactory until 84:34:12 g.e.t. when stack 1C failed. This stack was also removed from the bus. At 88:51:57 g.e.t., the load was increased to 36.1 amperes and the bus voltage reduced to 21.9 volts. Stacks 2A and 2C were both degrading severely. Prior to a reduction in load at 89:34:03 g.e.t., the bus voltage had decreased further to 21.6 volts at 37.1 amperes. At 91:10:56 g.e.t., two reentry batteries were placed on the bus to assist the fuel cells in handling the pre-retrofire loads. The remaining reentry batteries were placed on the bus and fuel cell section 2 was removed at 92:05:29 g.e.t.

Although the system had difficulties, it supplied approximately 2450 ampere-hours during the mission. The mission load profile is shown in figure 5.1.7-2. The maximum load of 57.4 amperes was delivered at a bus potential of 24.3 volts.

Prior to the failure of stack 1C, the section 1 degradation rate was normal (fig. 5.1.7-1). However, the degradation rate of section 2 was excessive, approximately three times the normal rate.

Stack load-sharing data are shown in figure 5.1.7-3. Initial load sharing was nominal, with stack C in each section carrying the highest percentage of the load. Following the loss of stack 2B, load sharing of stacks 2A and 2C varied considerably and was very sensitive to purges.

5.1.7.2.3 Water system data analysis: All of the symptoms of the problem seem to relate directly or indirectly to the water storage system. Therefore, an analysis of performance of this system during the prelaunch period and during the mission is presented here.

Four factors indicated that the water storage system was not functioning properly.

(a) The differential pressure warning lights going on for extended periods and being sensitive to water extraction

(b) The insensitivity of the water reference pressure to water extraction

(c) The performance sensitivity of stack 2A to oxygen purges

(d) The failure of stacks 2B and 1C.

In the analysis, an attempt is made to relate the performance of the system to the above factors. To assist in the analysis, a block diagram of the water system is shown in figure 5.1.7-4. The total storage capacity of the system, including tanks A and B, was 87.4 pounds. Figure 5.1.7-5 shows three curves which summarize the performance of the water storage system. Curve A is the estimated capacity for fuel-cell product water, considering the initial servicing of 42.9 pounds of drinking water and the water consumed by the crew during the mission. Curve B represents the initial servicing of 42.9 pounds of drinking water and the water consumed by the crew during the mission. Curve B represents the initial servicing of 2.3 pounds equivalent water-plus-gas and the amount of product water generated by the fuel cells, based on a theoretical water production rate of 100 percent. The difference between curves A and B, shown as curve C, is the estimated available fuel-cell product-water storage capacity for a normal operating system at any time during the mission. Also shown in the figure are the periods when the warning lights of both sections were on. The points corresponding to the lights coming on and off are circled on curve C for reference.

The illumination of the lights at 5 hours 45 minutes g.e.t. indicated that the 32-pound net storage capacity shown by curve C was not available. Comparison of subsequent light illuminations on curve C indicates that further losses of net storage capacity occurred, summarized as shown in the following table.

Storage capacity loss, lb	Ground elapsed time, hr:min	
	From	To
1.5	5:45	7:22
0	7:22	20:04
4.0	20:04	32:10
0	32:10	40:10
2.0	40:10	40:45
7.5 (Total)	5:45	40:45

Considering the total apparent loss in storage capacity of 39.5 pounds (32 plus 7.5), curve C shows an excellent correlation between the warning lights going off and water extracted by the crew.

The light being off on two occasions—at 28 hours 30 minutes and 40 hours 12 minutes g.e.t.—indicates that step losses in capacity probably occurred between 28 hours 30 minutes and 32 hours 12 minutes g.e.t. and between 40 hours 12 minutes and 40 hours 48 minutes g.e.t.

During all periods when the light was on, the differential pressure driving water from the fuel cell sections was reduced. As water was further produced, the driving force was reduced to zero and all the water produced remained in the section. The shaded areas shown in figure 5.1.7-5 represent periods when water was being accumulated in the section. The quantity of water shown is in error by the amount of water removed by purging, since this quantity could not be estimated. The performance increase of stack 2A following purges at 45 hours 39 minutes, 51 hours 30 minutes, 52 hours 12 minutes, 60 hours 48 minutes, and 76 hours 9 minutes g.e.t. (fig. 5.1.7-3) and the estimated water accumulated in the sections at these times from curve C, offers evidence that water was being removed during the purges. The large quantity of water accumulating in the sections during the last few hours of the mission suggests an explanation for the loss of stack 1C and the seriously degraded performance of stacks 2A and 2C. The presence of water on the membrane-electrode surface decreases the effective operating area and, consequently, the current density increases. If this process continues, the membrane breaks down and a perforation is formed. Once the perforation is formed, oxygen rushes into the hydrogen cavity and mixes with the hydrogen; in the presence of the platinum catalyst, combustion takes place. The rapid failure of stack 1C (55 seconds) is indicative of a perforation-type failure. Examination of the coolant inlet temperature showed a measurable increase at the time of the failure (fig. 5.1.7-6). Examination of section-2 coolant inlet temperatures for the period in which stack 2B failed showed no evidence of a burn through. Also, the failure took 1 hour 45 minutes, which would not be expected for a perforation-type failure.

5.1.7.2.4 Problem isolation: The cause for the loss of 32 to 39 pounds of water storage capability cannot be definitely determined from the available data. Unfortunately, the equipment cannot be examined because it was in the adapter equipment section. There are three possible sources for the problem:

(a) A leak in the plumbing at the inlet to storage tank B, which could freeze the water by evaporation and cause a complete blockage

(b) A stuck bladder, which would restrict the storage volume

(c) Oxygen gas leakage into the water storage system.

The first two possibilities are less likely than the third because the loss in storage capacity increased during the mission. The third possibility appears likely because additional leakage would effect additional loss of storage capacity.

Assuming that it was an oxygen leak, the nature of the leakage was examined, leading to the following hypothesis:

Prior to 5 hours 45 minutes g.e.t., a total of 16 000 cc of gas leaked into the water system. This gas plus the water generated up to that time probably bottomed-out the bladder in tank B, which could explain the insensitivity of the water reference pressure to water extraction by the crew. Prelaunch data indicate no apparent leakage up to T minus 80 minutes; therefore, the leakage must have occurred between that time and 5 hours 45 minutes g.e.t. (a period of 7 hours 5 minutes) for a minimum leak rate of 2260 cc/hr.

Similar analysis of further losses of net storage capacity indicates the following leak rates:

Leak rate, cc/hr	Ground elapsed time, hr:min	
	From	To
450	5:45	7:22
0	7:22	28:30
675	28:30	32:10
0	32:10	40:10
1050	40:10	40:45

The only direct interfaces of oxygen and the water system where leakage was possible are (1) across the bellows of the ΔP mechanism and (2) across the water separator plates.

Only one ΔP sensor removed from a test section has been found to have a hole in its bellows, whereas excessive water separator leakages due to cracked plates or poor wetability have been experienced many times in ground tests. Unpredictable leak rates are also a characteristic of water separator leaks. These factors make the water separator plates the prime suspect.

5.1.7.3 Reactant supply system.- The performance of the reactant supply system was good throughout the mission except for a failure in the heater circuitry of the hydrogen storage container. This subject is discussed in paragraph 5.1.7.6. The problem occurred following a special test conducted to determine the improvement of the thermal performance of the hydrogen container resulting from venting the container annulus to ambient. The performance improved by approximately 25 percent on this particular container.

5.1.7.4 Power distribution system.- The only problem in the power distribution system was in the hydrogen heater circuitry. As part of a special test, the hydrogen tank pressure was to be increased to 300 psi. At approximately 76 hours 44 minutes g.e.t., after reaching a pressure of 290 psi, the heater switch was presumably turned to the AUTO position (fig. 5.1.7-7). In the AUTO position, the heater normally comes on when the tank pressure drops to 225 ±15 psi. The current drawn by the heater was 0.67 amperes at 27 volts. A current of this size is within the fluctuating loads of the spacecraft, so the heater performance could not be determined from bus current. At 78 hours 26 minutes g.e.t. the ground station noticed that the hydrogen pressure had risen. The crew was awakened and informed of the situation. Attempts to remove power from the heater by using the hydrogen-tank heater switch were unsuccessful. Therefore, the hydrogen heater circuit breaker was opened at 78:31:45 g.e.t. The heater was not used again during the mission. Evaluation of all circuits available in the reentry module has revealed no problem, and the switch operated satisfactorily during postflight testing. At the present time the switch is in the failure analysis laboratory for further investigation.

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

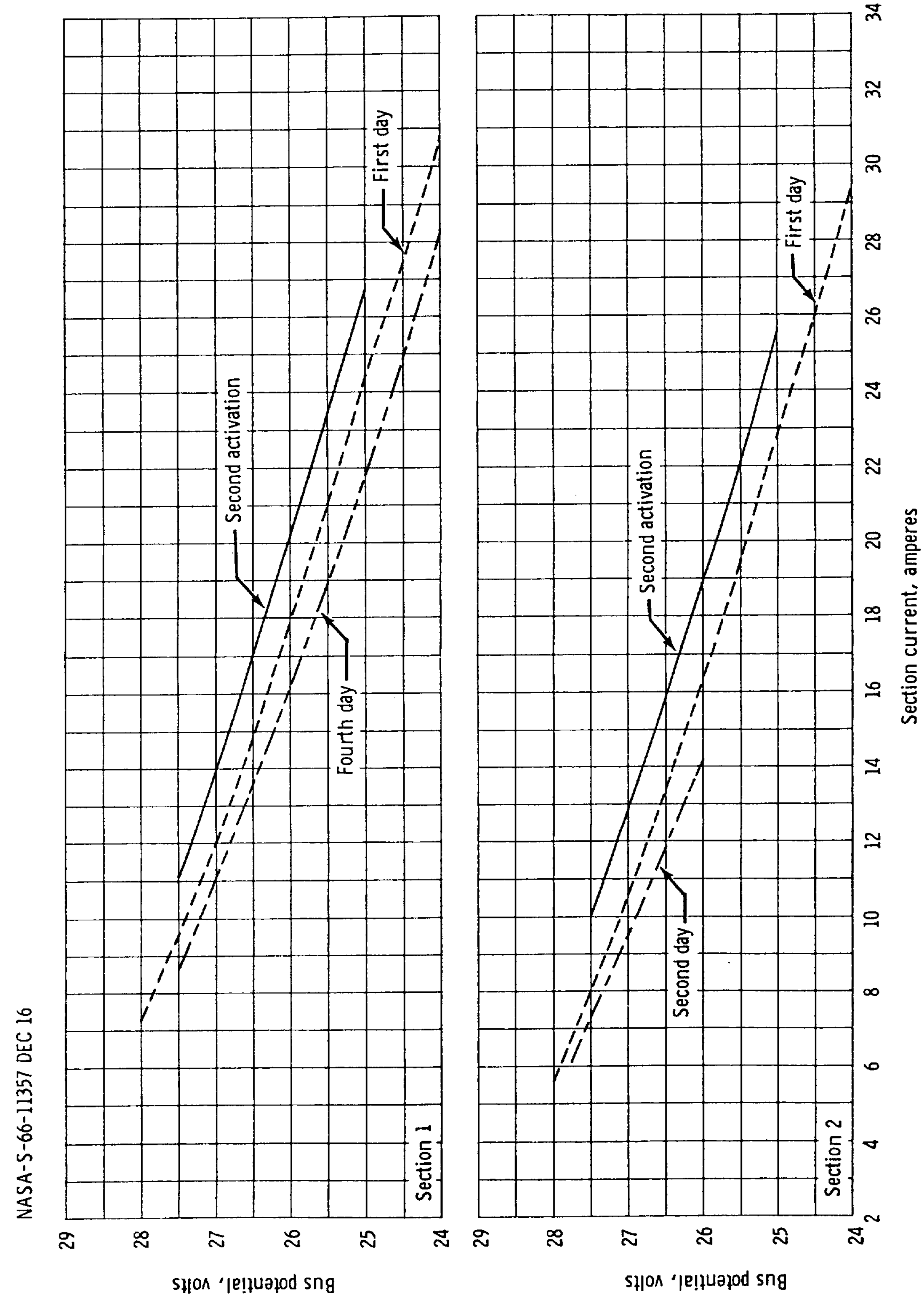


Figure 5.1.7-1. - Fuel-cell section 1 and 2 performance.

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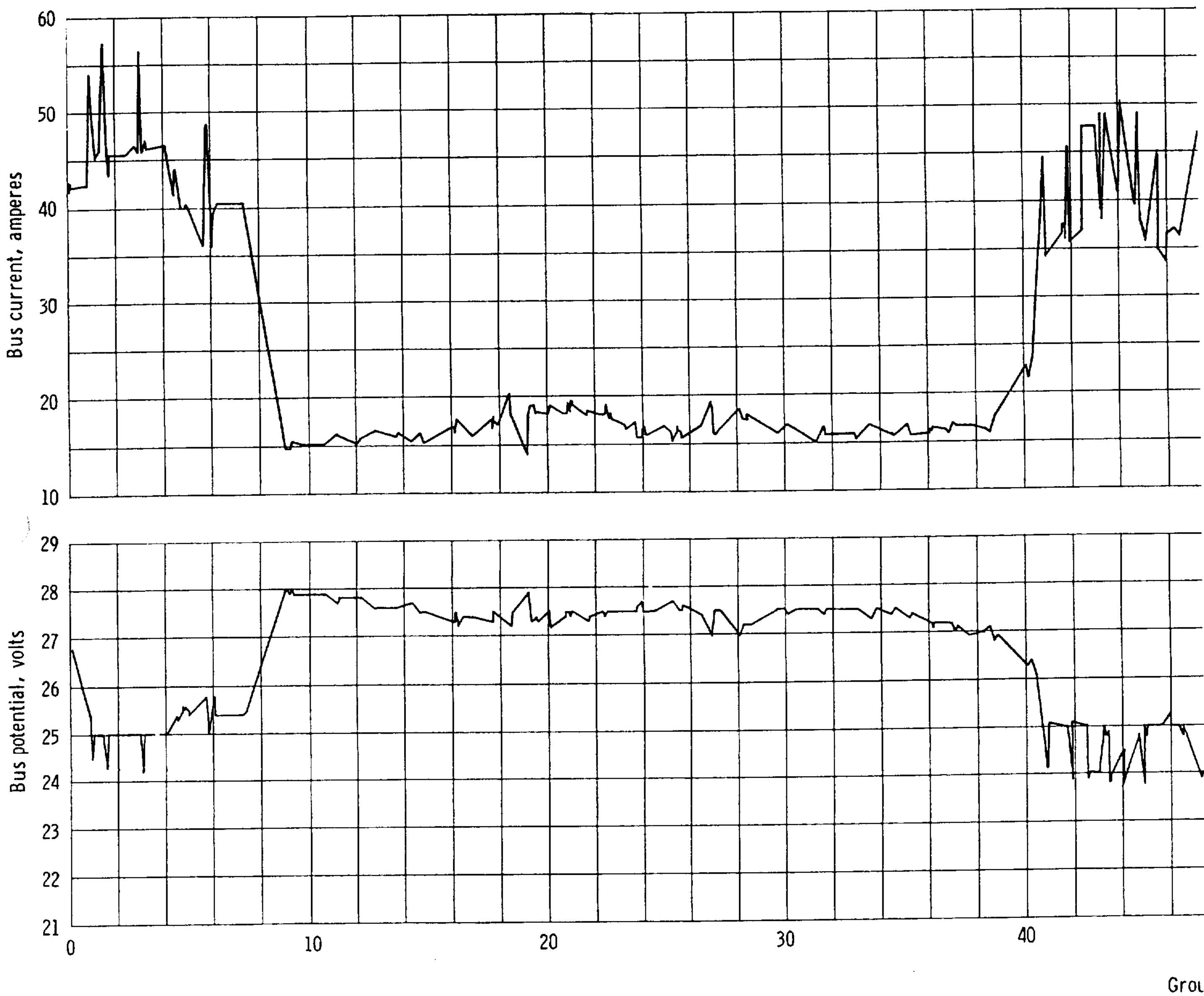
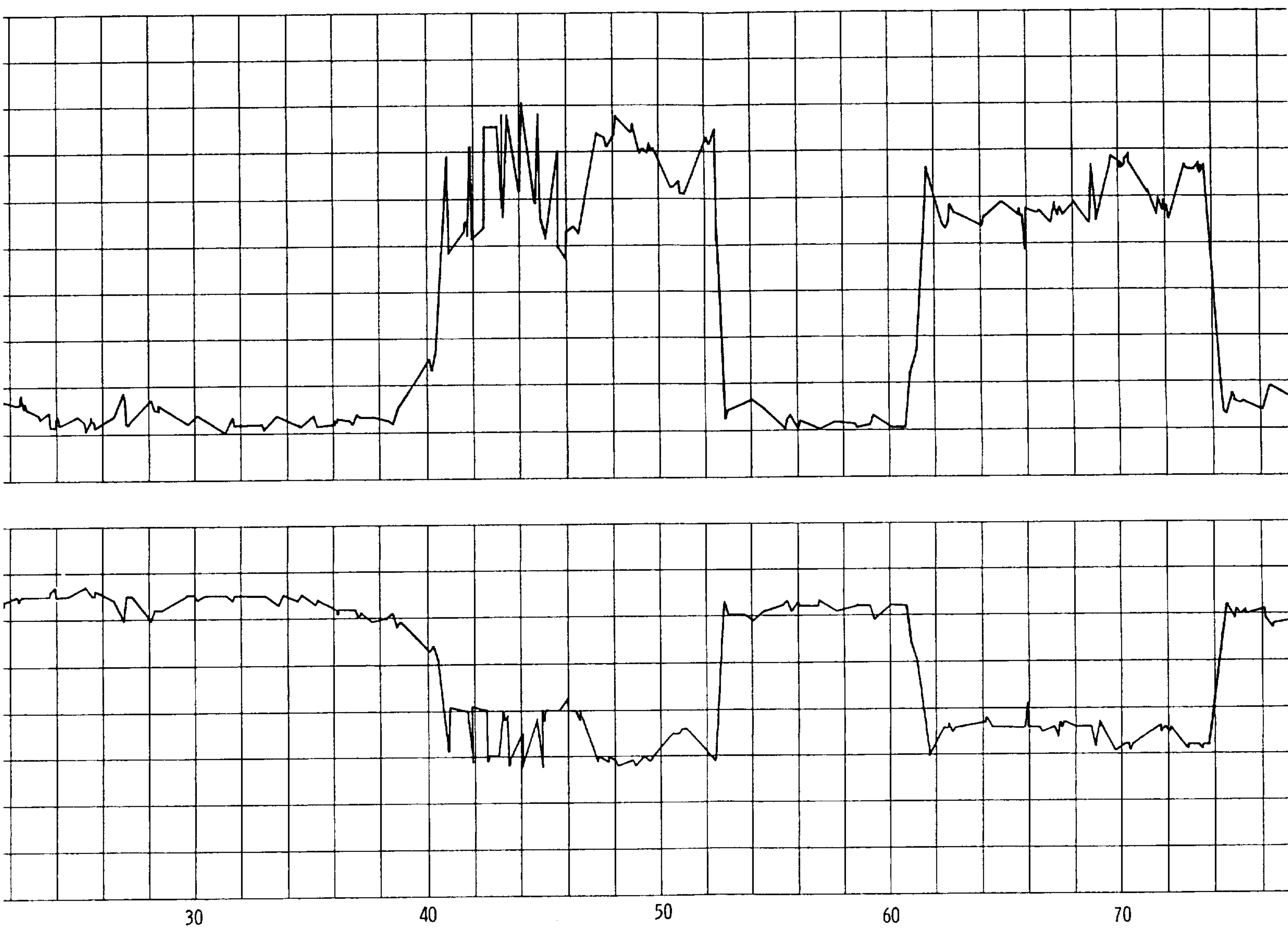
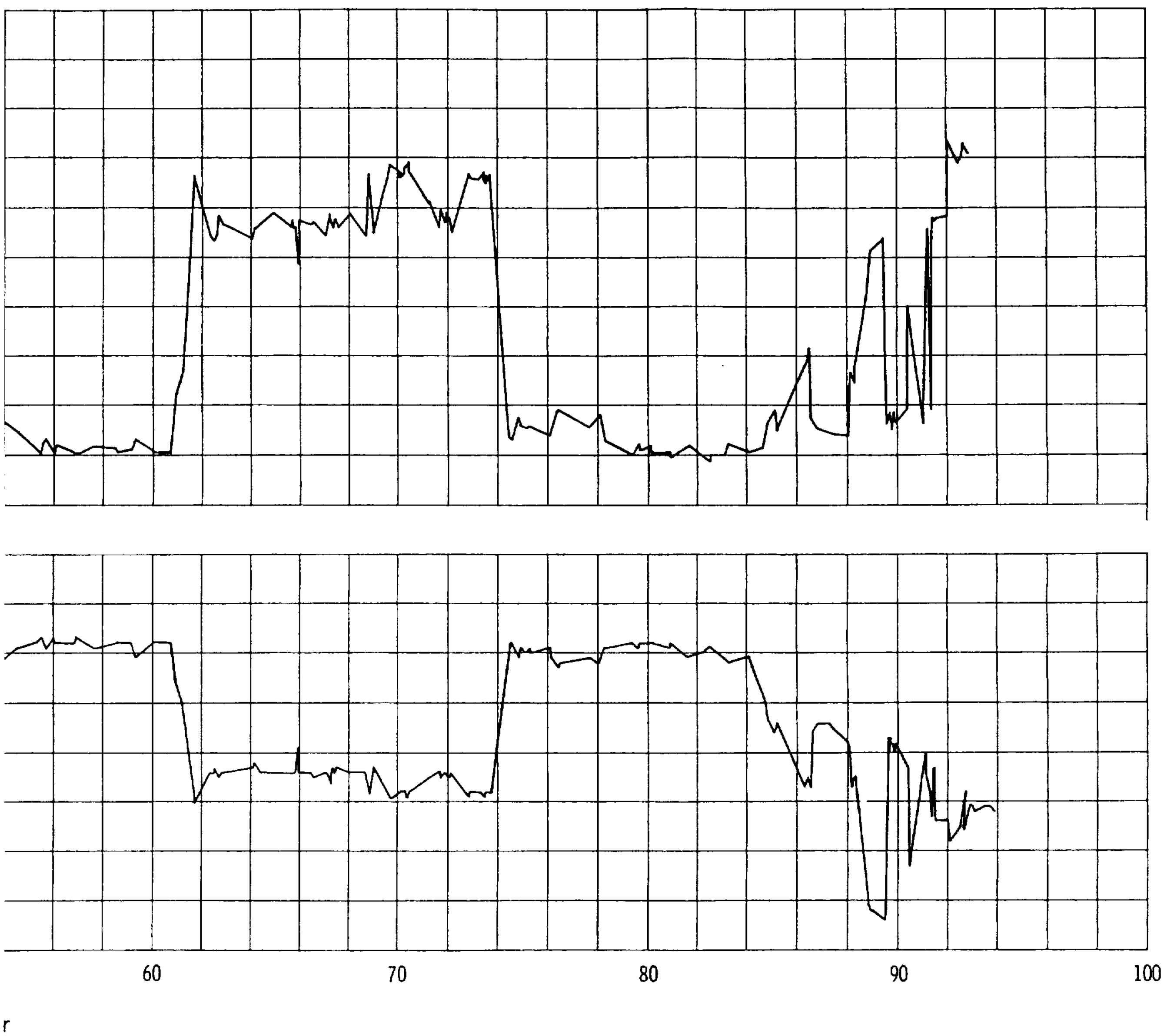


Figure 5.1.7-2. - Spacecraft 12 fuel-cell performance.

D

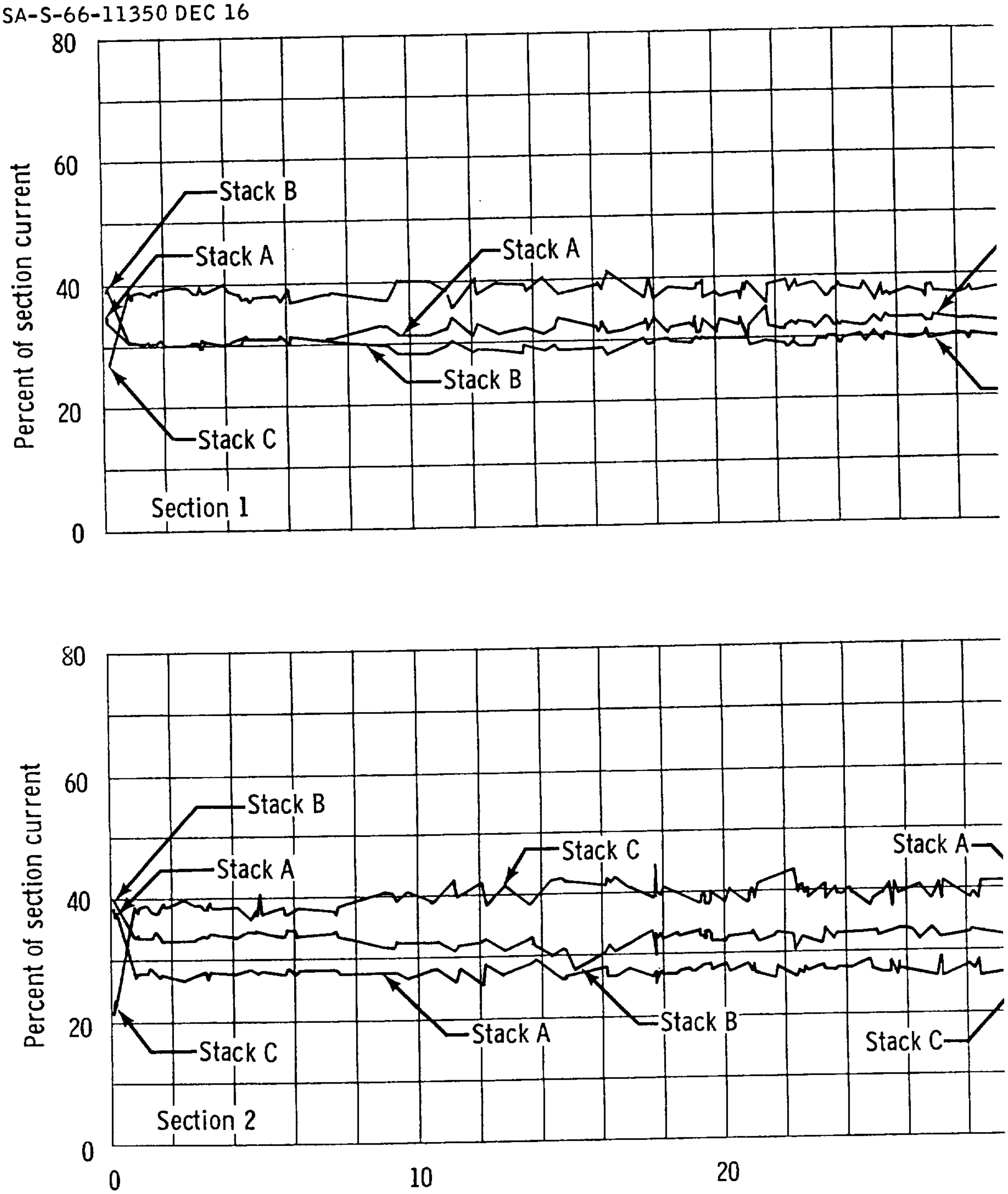


D

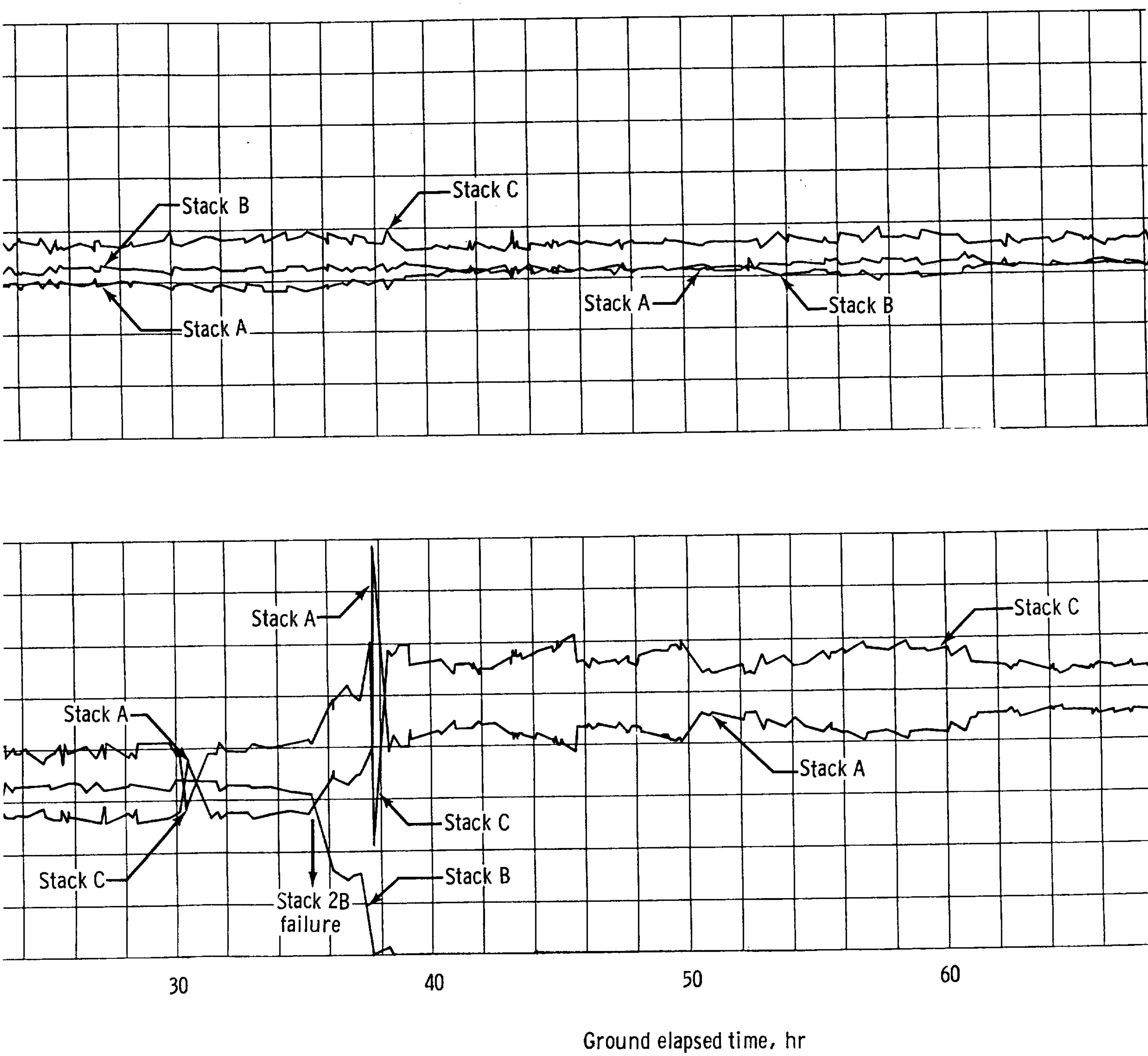


60
70
80
90
100
r

NASA-S-66-11350 DEC 16



/


Stack B
Stack C
Stack A
Stack A
Stack B
Stack A
Stack C
Stack A
Stack A
Stack C
Stack C
Stack B
Stack 2B failure
30
40
50
60
Ground elapsed time, hr

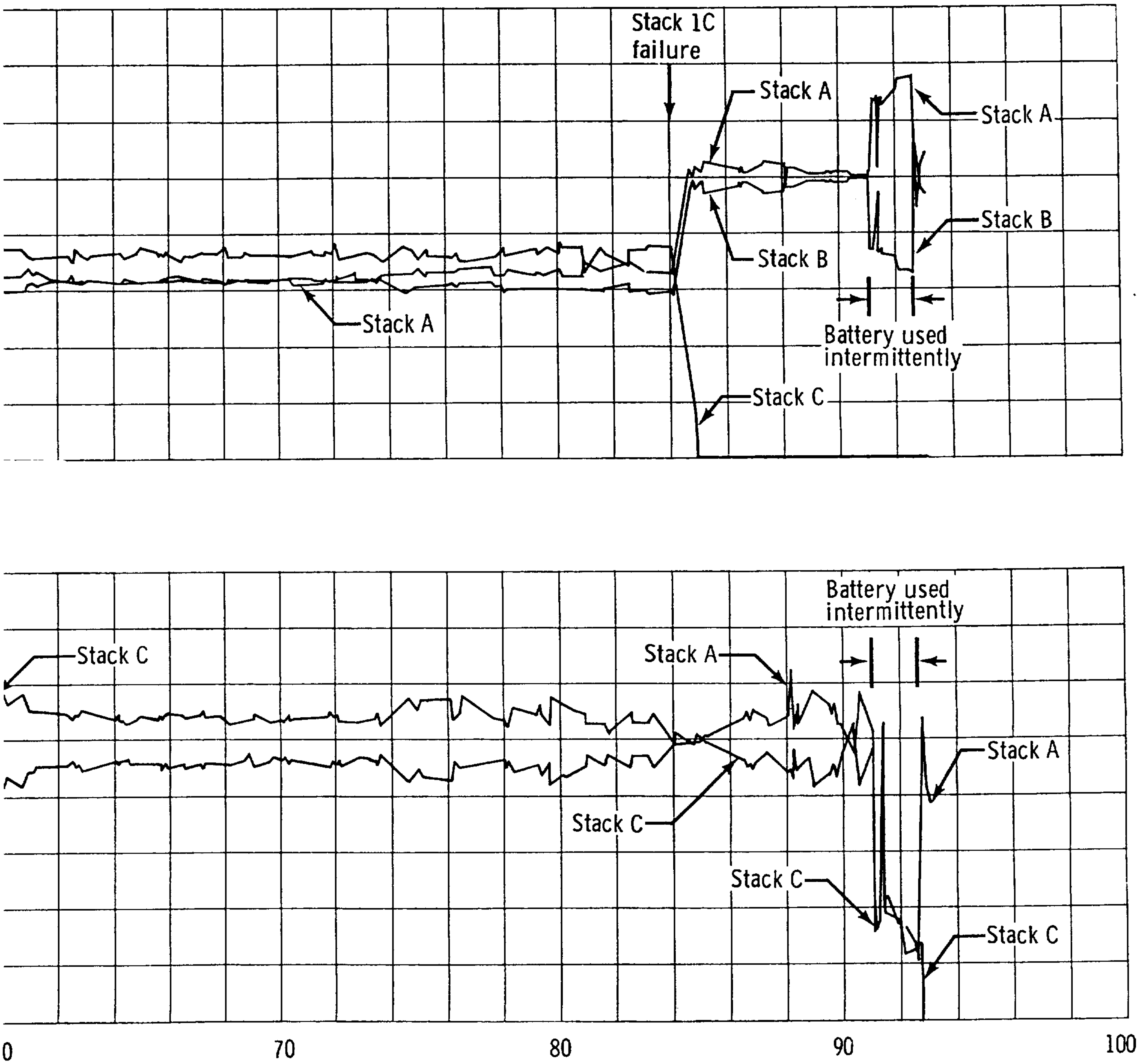


Figure 5.1.7-3.-Load sharing between fuel-cell stacks.

NASA-S-66-11321 DEC 13

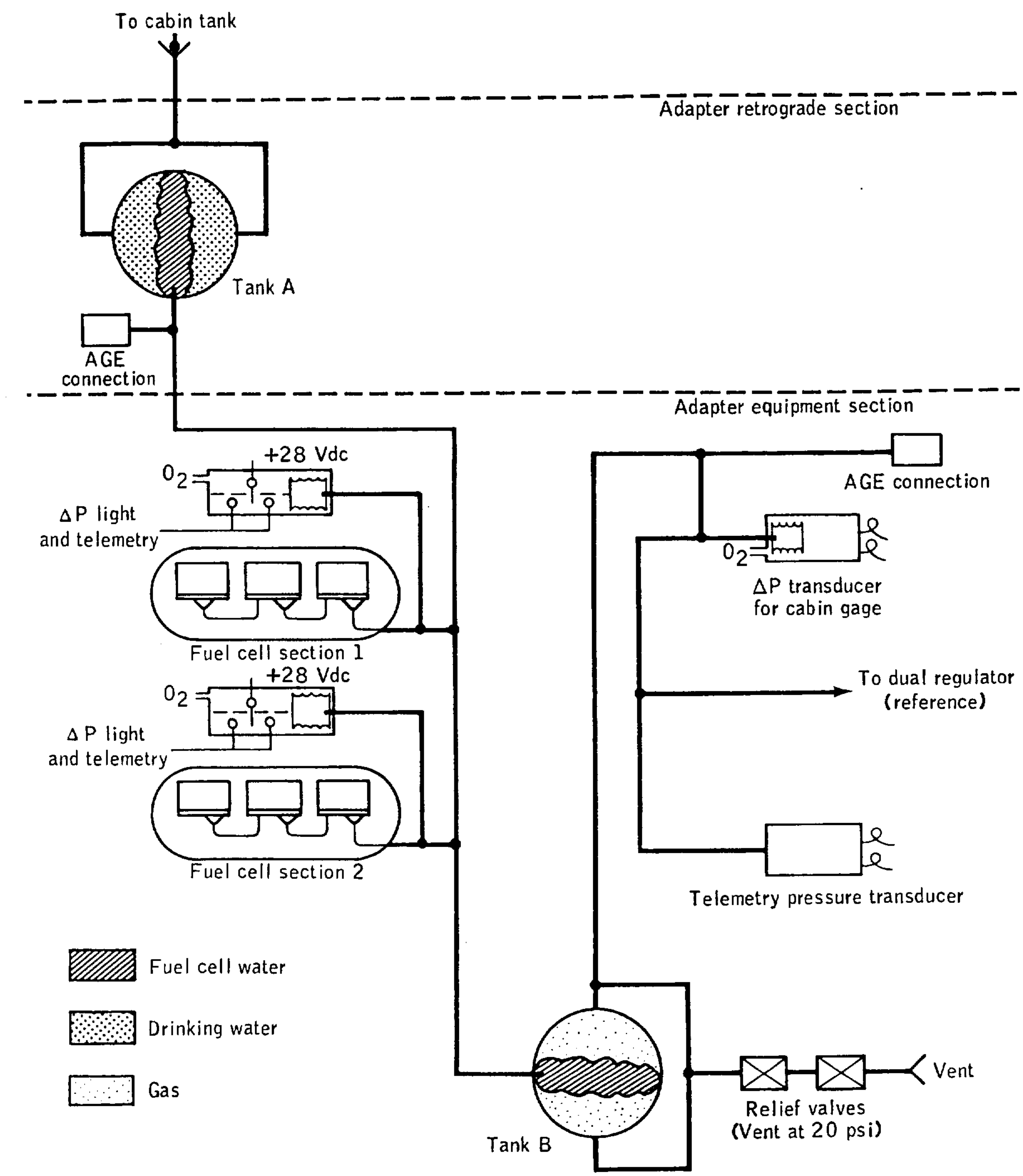
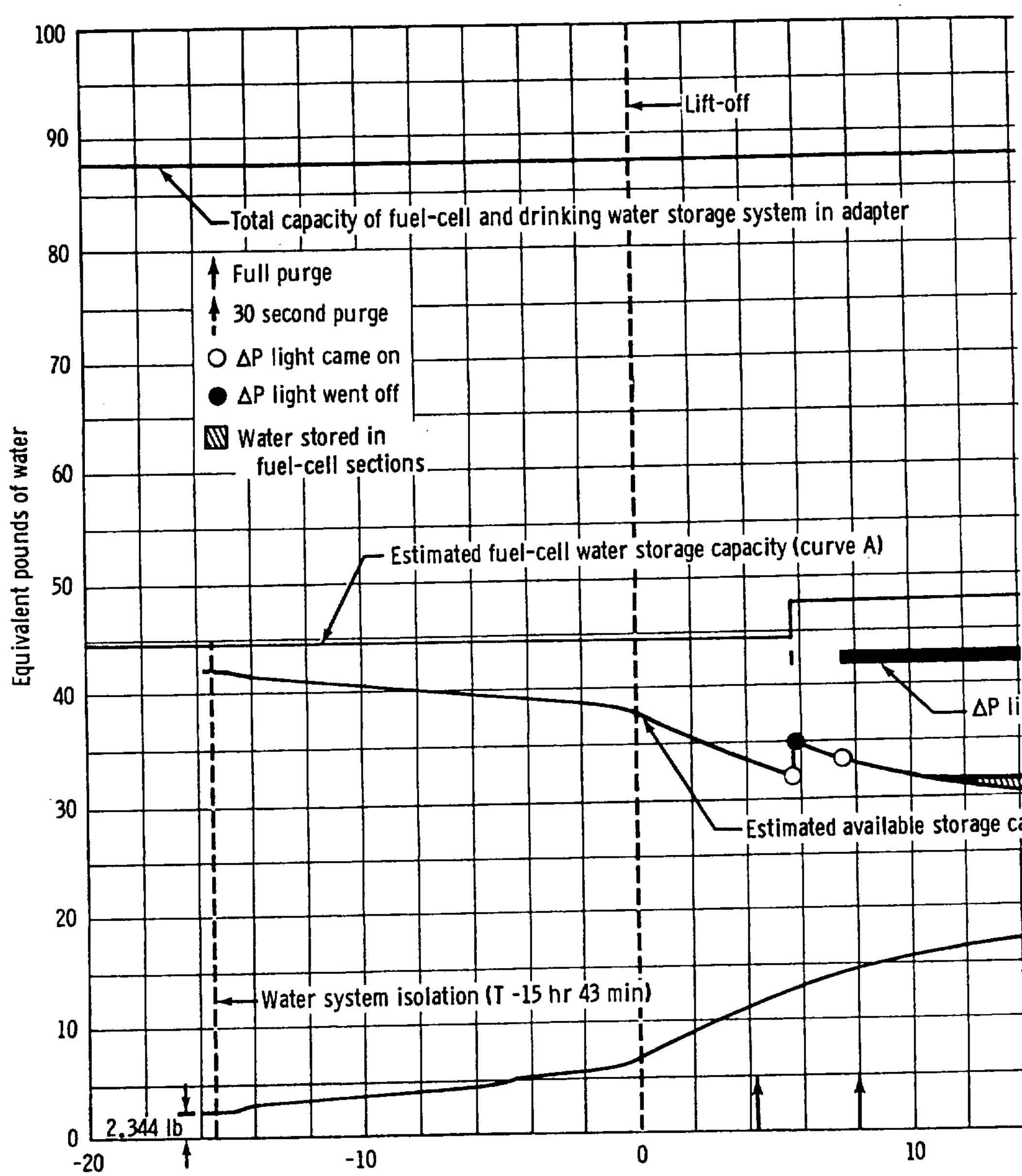
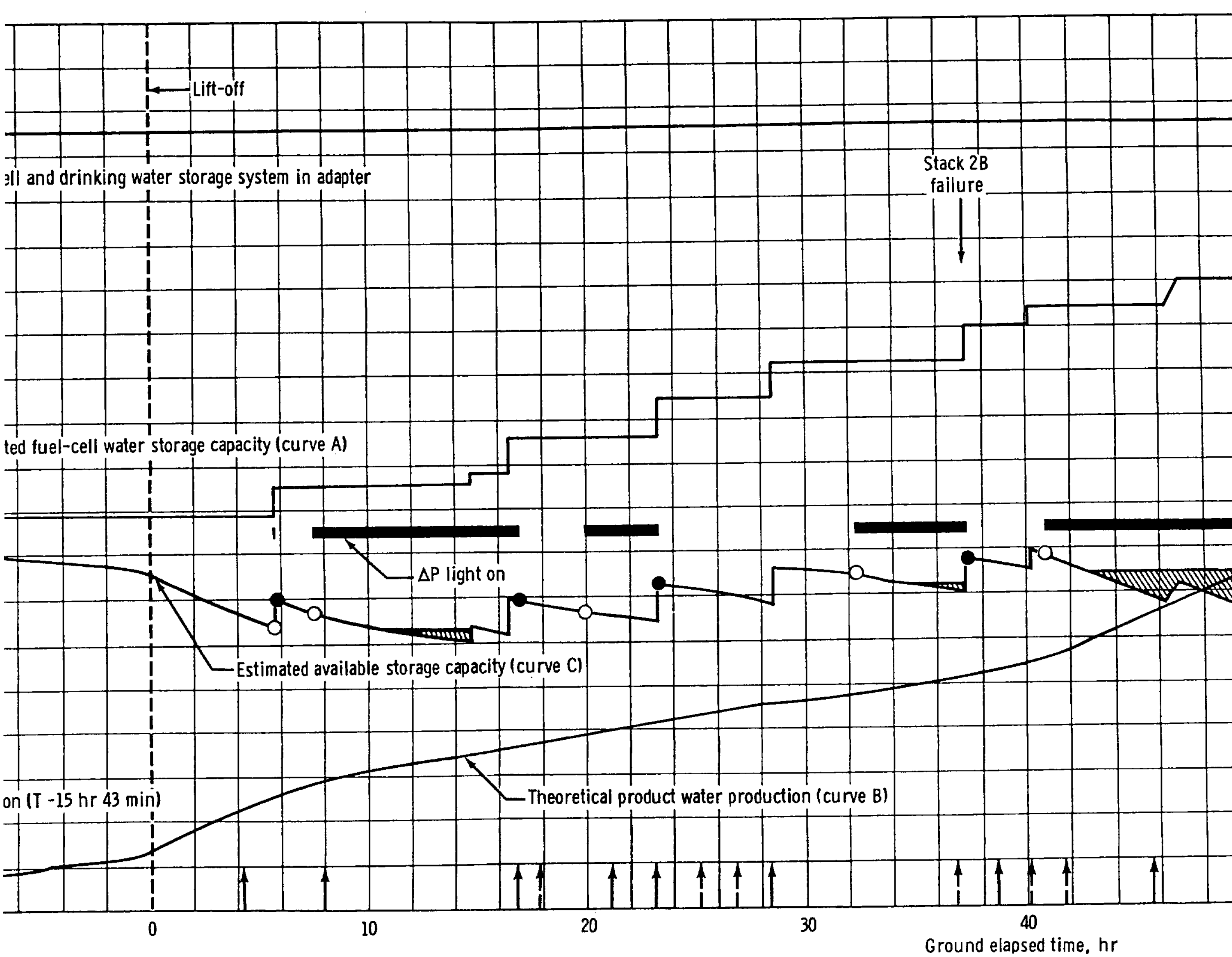


Figure 5.1.7-4. - Fuel-cell water system.

NASA-S-66-11335 DEC 14





Lift-off
ell and drinking water storage system in adapter
Stack 2B failure
ted fuel-cell water storage capacity (curve A)
ΔP light on
Estimated available storage capacity (curve C)
on (T -15 hr 43 min)
Theoretical product water production (curve B)
0
10
20
30
40
Ground elapsed time, hr

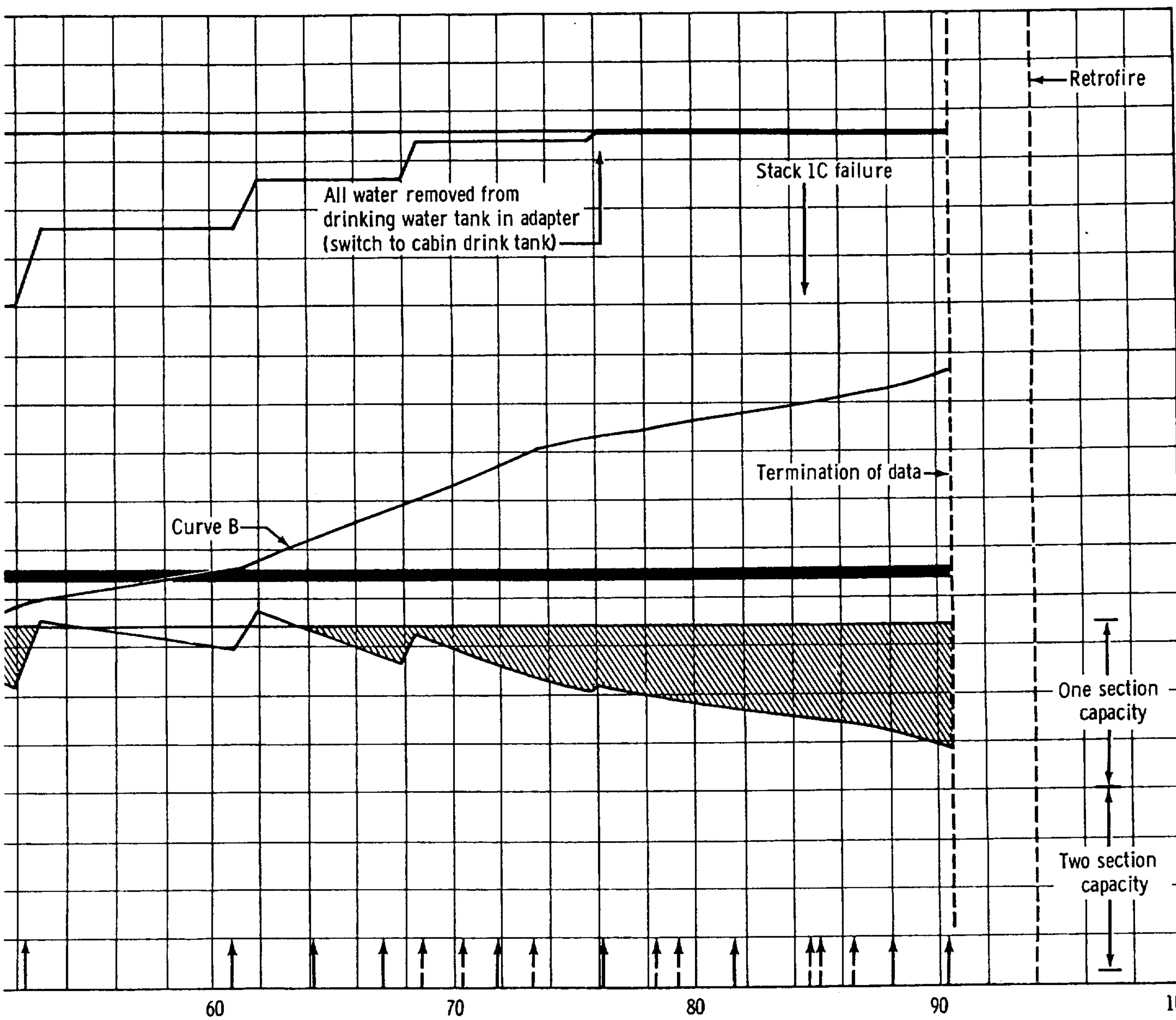


Figure 5.1.7-5. - Fuel-cell water storage.

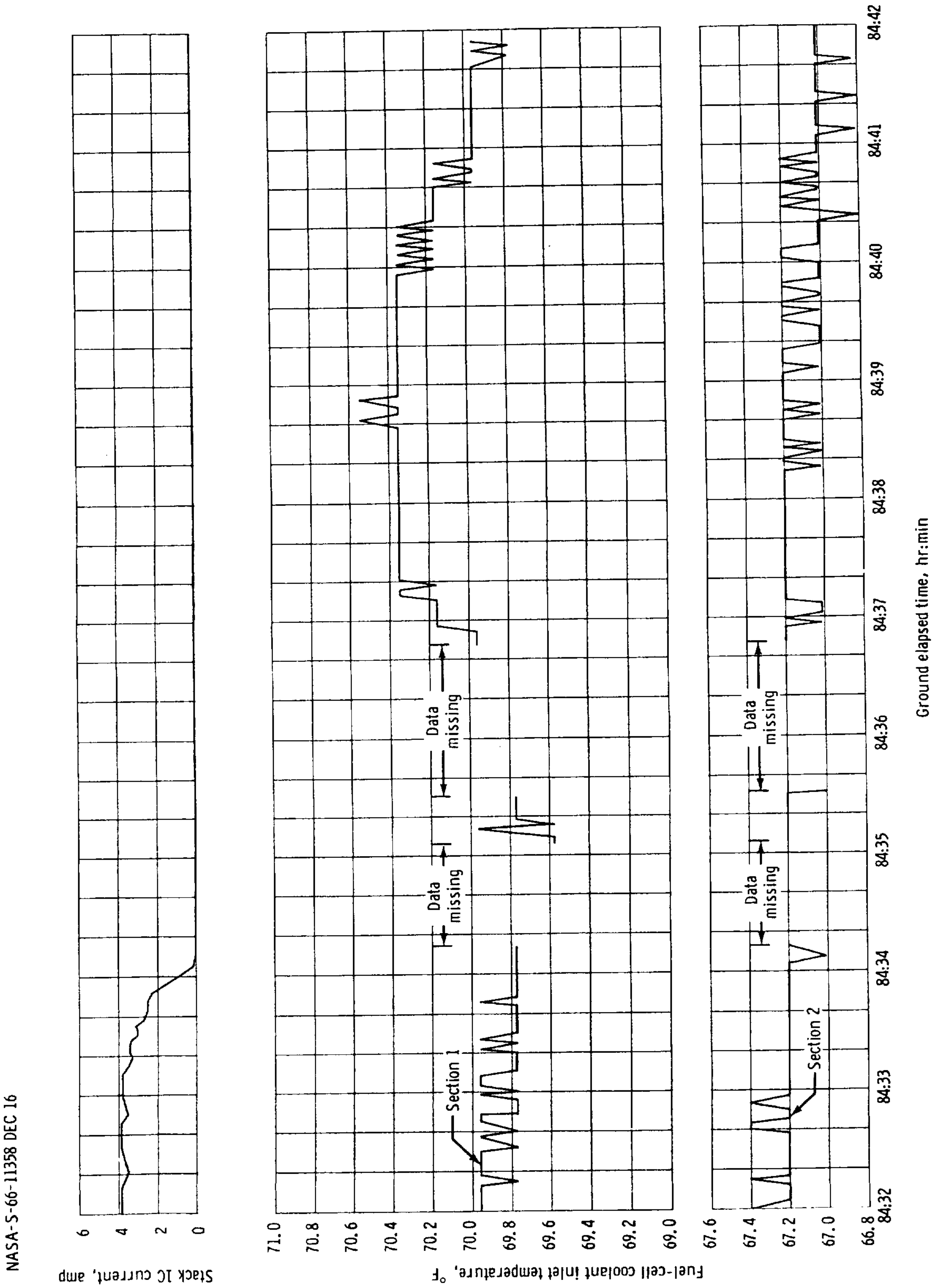


Figure 5.1.7-6. - Stack 1C failure.

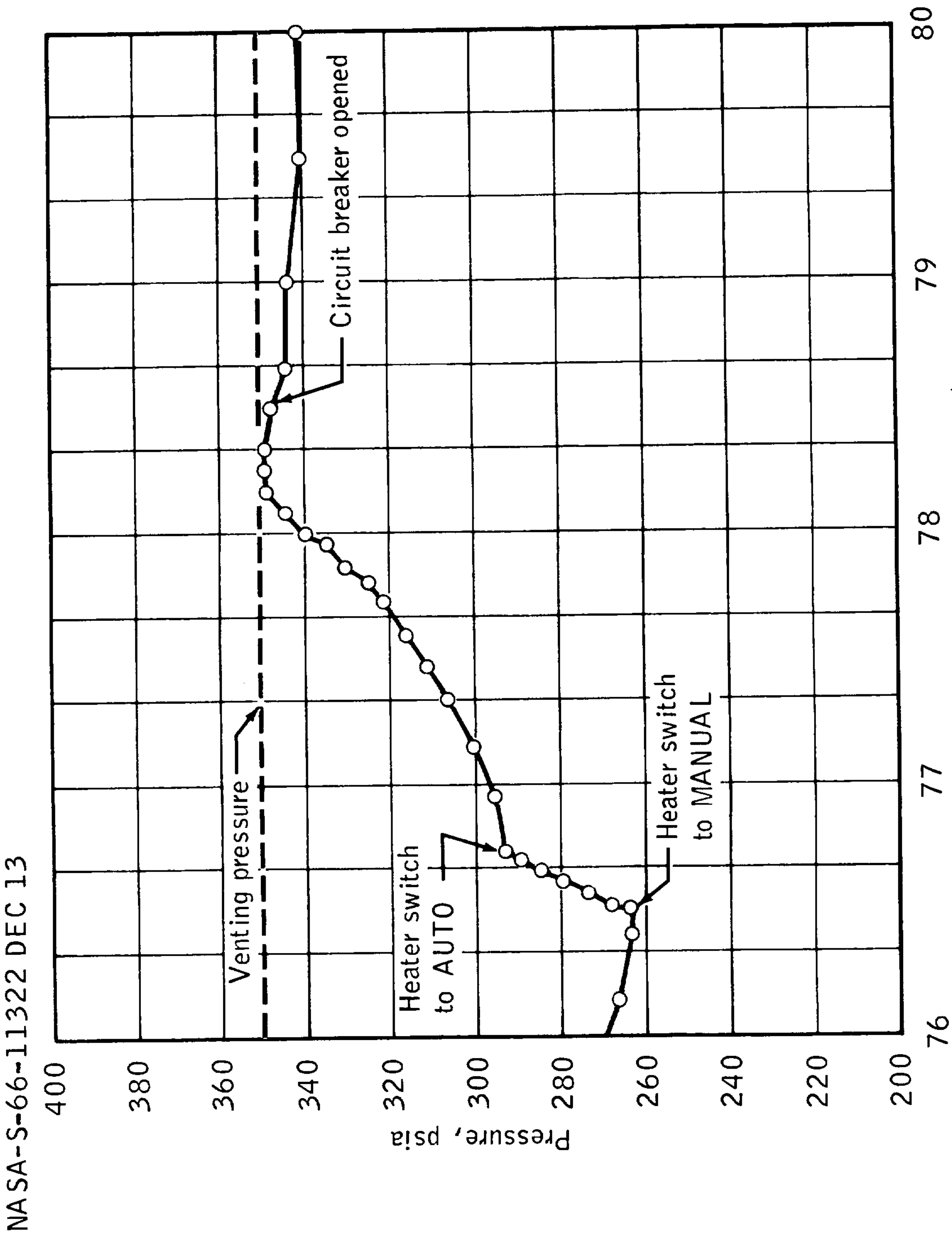


Figure 5.1.7-7.- RSS hydrogen container pressure.

5.1.8 Propulsion

Overall flight performance of the three spacecraft propulsion systems—the Orbital Attitude and Maneuver System (OAMS), the Reentry Control System (RCS), and the retrograde rocket system—was generally adequate to meet mission objectives. Severe OAMS attitude-engine thrust degradation was encountered on this flight, requiring the crew to accomplish the tether evaluation and the T002 experiment under very difficult conditions. RCS A-ring regulated pressure significantly exceeded the specification limit after activation. The crew used A-ring propellant prior to retrofire to contain the pressure rise. The performance of each system is discussed below.

5.1.8.1 Orbital Attitude and Maneuver System.-

5.1.8.1.1 Attitude engine performance: The first crew report of degraded attitude-engine thrust occurred during revolution 26 when engines 2 (pitch down and roll left) and 4 (yaw right and roll left) appeared low. In revolution 28, before closing the hatch at the end of the umbilical EVA, the pilot observed the plumes of engines 4 and 5 (pitch up and roll right) while standing in the open hatch. Engine 4 exhibited a visible exhaust plume about one half the size of that of engine 5, with an apparent lower exhaust velocity and longer tail-off. From this test it was apparent that both valves of engine 4 were operating to some extent and that combustion was occurring. Engine 8 (yaw left and roll left) was reported low during revolution 43, and by the end of the mission, a number of other engines appeared degraded. This overall degradation actually tended to alleviate the problem associated with controlling the spacecraft because of the accompanying reduction of unwanted cross coupling.

The postflight data evaluation in nearly all cases substantiated the crew's report concerning the degraded condition of the attitude engines. Engine 4, in fact, exhibited anomalous performance as early as the first revolution. This probably caused the roll control difficulty that was first noted during revolution 4 with the spacecraft and GATV in the docked configuration.

As for previous missions during which degraded thrust was encountered, angular velocity data were analyzed to determine the characteristics of the thrust decay. In the limited time available for analysis of these data, only the earliest noted time of thrust degradation can be presented. This is shown in the following table, including a comparison of how the various engines degraded during the mission.

Engine number	First observed occurrence, revolution	Percent of normal thrust
1	56	90
[a]2	13	26
3	56	34
[a]4	1	59
5	25	35
6	30	68
7	25	5
[a]8	43	10

[a]These engines continued to deteriorate further, eventually reducing to within two to five percent of nominal thrust.

Normally in past Gemini missions, the most serious effect of the loss of attitude-engine thrust has been some additional unplanned propellant consumption to correct rates induced by cross coupled components, plus a decrease in accuracy of the propellant quantity gaging system. During this mission the problem was more serious, because engines providing functions necessary in automatic modes had essentially no output. In the platform and rate-command modes, the control system appeared to be unstable (divergent) under certain special conditions. This required the crew to select manual control to restore the spacecraft to a stable attitude. Section 5.1.5 discusses these special conditions in detail.

The crew conducted two separate tests of the individual engines during revolutions 25 and 56. The first test revealed severe thrust degradation in engines 2 and 4. The angular acceleration from engine 2 was less than five percent of nominal, and thrust from engine 4 was too small to measure. The second engine test showed that all engines had degraded and that engine 1 was the only one producing near nominal thrust.

The cause of the problem of degraded thrust cannot be determined precisely with the data available and the hardware could not be analyzed after the flight. There is probably more than one cause for the reduced thrust. On the basis of available flight data, none of the following possible causes can be eliminated:

(a) Contamination, with attendant blocking of ports and lines

(b) Local freezing of the propellant feedlines

(c) Iron nitrate deposits

(d) Restricted poppet movement in the propellant valves.

The prime contractor has initiated an investigative program, including tests if necessary, in an attempt to establish the probable cause of the problem. The results of this program will be published as a supplemental report (see section 12.4).

5.1.8.1.2 Maneuver engine performance: From an analysis of velocity changes produced by the maneuver engines, the performance of these engines was within expected limits. There was a peculiar characteristic of the angular velocity data which may be indicative of some degradation of engine 12. This characteristic is a change in the angular velocity slope near the end of the firing and appears as a large tail-off. The slope change was first observed during revolution 39 and persisted until the end of the mission. Unfortunately, there are no other data that provide assistance in analysis of the pulse. Figure 5.1.8-1 compares a normal pulse of engine 11 with a typical one being described above. The engines appeared to shut down satisfactorily with no noticeable gradual rate changes subsequent to the tail-off period. Had such rates been noted, this would have been indicative of a leaking valve. This was examined and noted because engines 9 and 12 were fired for attitude control in pitch and yaw and hence exposed to a more active duty cycle than normally encountered with maneuver engines.

Data analysis also revealed some unusual angular velocity changes in all three axes prior to the second docking in the time interval between 4:49:44 and 5:05:04 g.e.t. The measured angular accelerations in pitch were as large as 6.7 deg/sec^2 and in yaw as large as 4.7 deg/sec^2, both of which are greater than the disturbance produced by either a single aft engine firing or a single forward engine firing. The accelerations are noted during times when no engines were firing and are accordingly attributed to several contacts of the spacecraft with the target vehicle. Like disturbances appeared in the GATV data. The data do not indicate that the degraded performance of engine 4 directly caused these contacts. See section 5.7 for further discussion.

5.1.8.1.3 Propellant utilization: Propellant consumption over the duration of the mission is presented in figure 5.1.8-2. The curve compares actual quantities with preflight-planned quantities. Initial consumption was higher than planned because the IVAR maneuver was not included in the preflight-planned values. The problem with completing the second docking accounted for another unplanned expenditure. Other excessive unplanned expenditures are attributed to the rate and attitude corrections required as a result of cross coupling induced by degraded attitude-engine thrust.

It should be noted that the program which computes propellant consumption assumes a nominal mixture ratio in both the maneuver and attitude engines. Undoubtedly, with the degraded attitude engine problems, the computations are not as accurate as is normally possible with this type of system.

5.1.8.2 Reentry Control System.- RCS activation occurred at 92:38:25 g.e.t. (revolution 58). The regulator in the A-ring system apparently leaked nitrogen pressurant gas internally, thus causing the feed system pressure to rise, reaching a maximum of 414 psia. (Specification is 295 ±15 psia.) Periodically throughout the remainder of the mission, the crew relieved this pressure by firing the A-ring engines. This was done to prevent rupture of the burst diaphragm (420 to 500 psia). This effort proved successful as determined by postflight test evaluation, but the pressure reduction did consume 7.3 pounds (18 percent) of the usable A-ring propellant. From the pressure data characteristics presented in figure 5.1.8-3, it is apparent that the regulator leaked until retrofire.

During the retrofire sequence, large control system demands appear to have caused the regulator to open, permitting passage of the contamination that is presumed to have caused the problem. Disturbances created by the retrorockets might have provided some assistance. The regulator from this point on maintained a lock-up value of 298 psia until it sensed atmospheric pressure to which it is referenced. This caused the regulated pressure to continually increase to a 315 psia maximum at sea level as designed.

A review of preflight ground test data revealed no prior occurrence of leakage in this regulator. Initial postflight testing has shown no tendency of the regulator to leak, but has revealed an out-of-specification lock-up pressure of 317 psia. The unit will be further tested and then disassembled for failure analysis. On all ten manned missions, utilizing a total of 30 units, this is the only significant out-of-specification condition encountered in these regulators.

RCS propellant consumption for specific periods of interest is shown in the following table. The total absolute accuracy of these data is ±4 pounds (±10 percent); for quantities determined over a short time interval, the accuracy is closer to ±2 pounds (±5 percent).

Event	Time, g.e.t.	Propellant remaining, lb	
		A-ring	B-ring
Activation	92:38:25	33.4	33.4
Begin retrofire	93:59:57.7	23.1	30.4
End retrofire	94:00:17.2	21.6	28.7
Guidance initiate	94:22:50.3	19.7	28.7
Guidance terminate	94:28:24.4	10.0	25.7
Drogue deploy	94:29:09.6	4.0	17.5
End of mission	94:32:00	0.7	7.8
Deserviced	Nov. 18	0.3	6.5

5.1.8.3 Retrograde rocket system.- The predicted velocity change to the spacecraft produced by the retrograde rocket system was 322.4 ft/sec. This compares well with the IGS measured value of 321.5 ft/sec.

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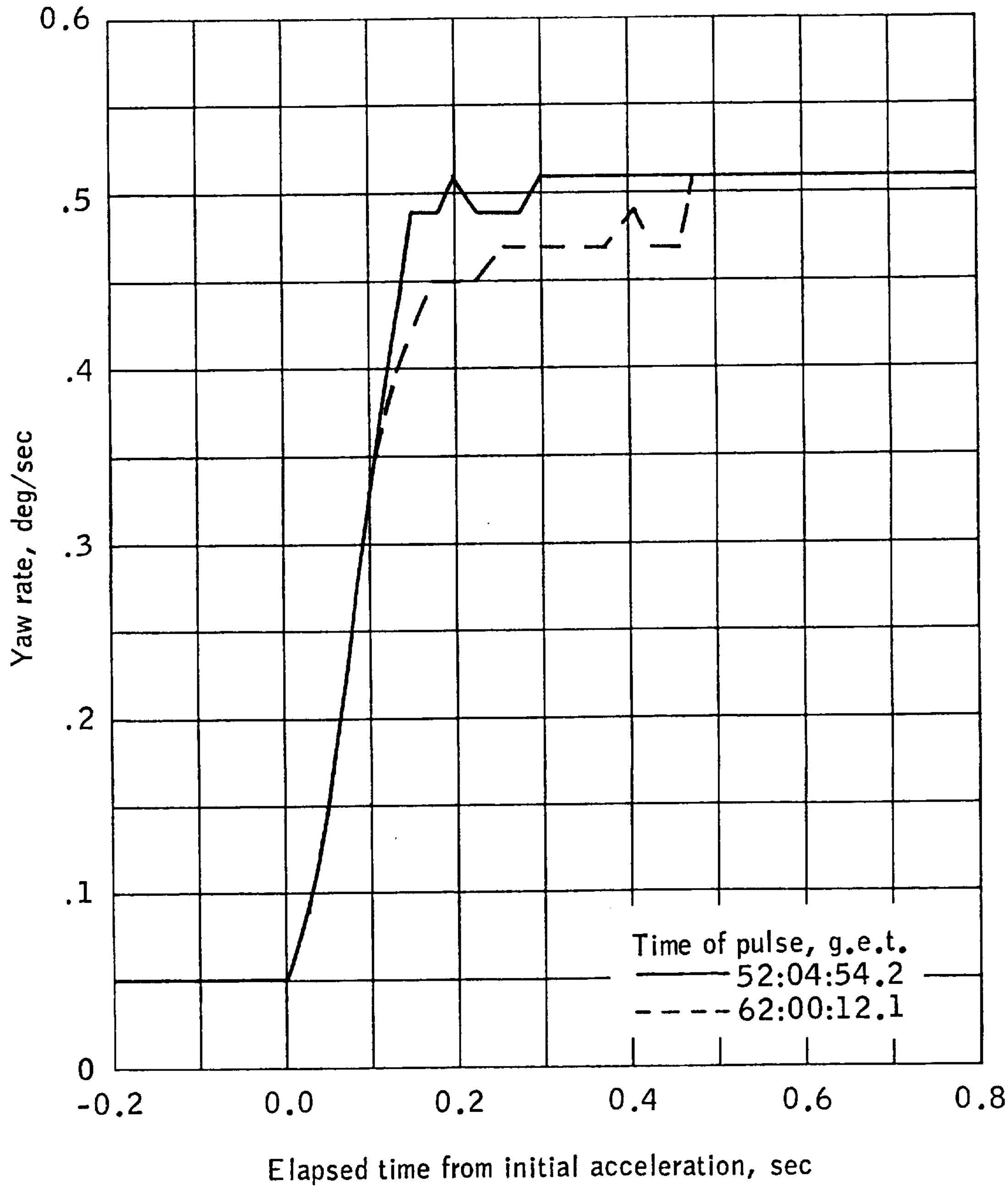

Figure 5.1.8-1. - Maneuver engine number 12 tail-off variation.

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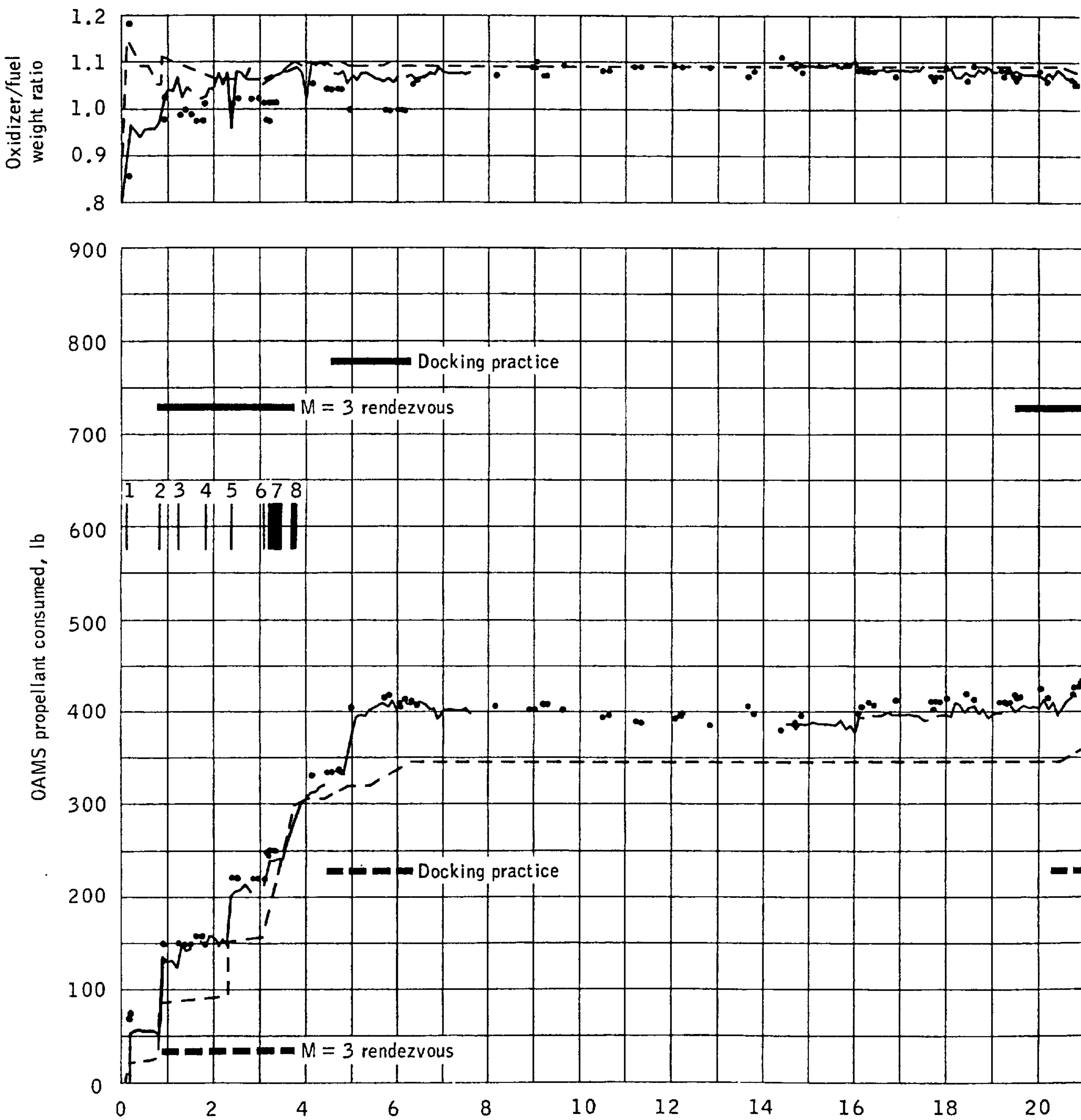
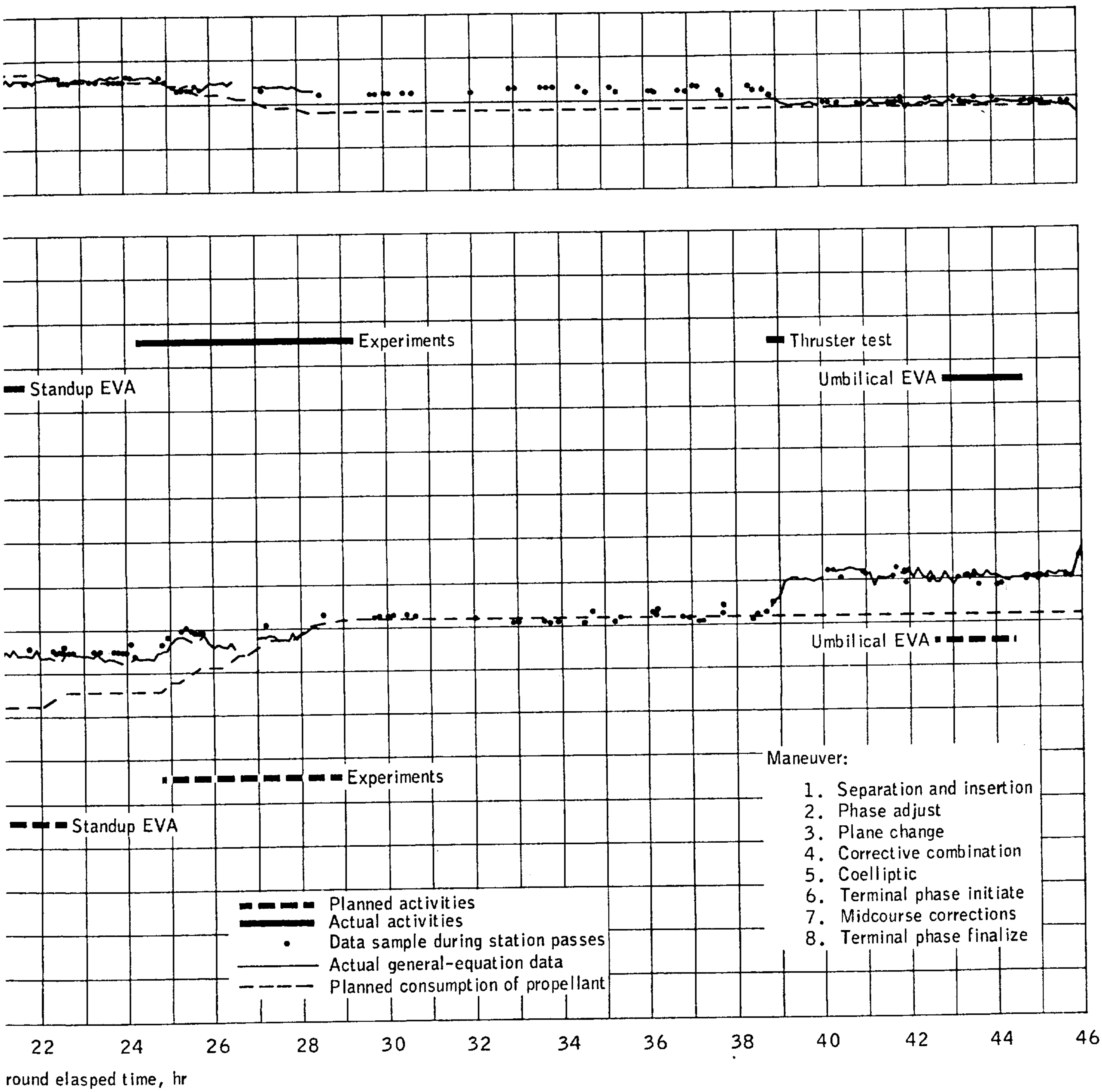
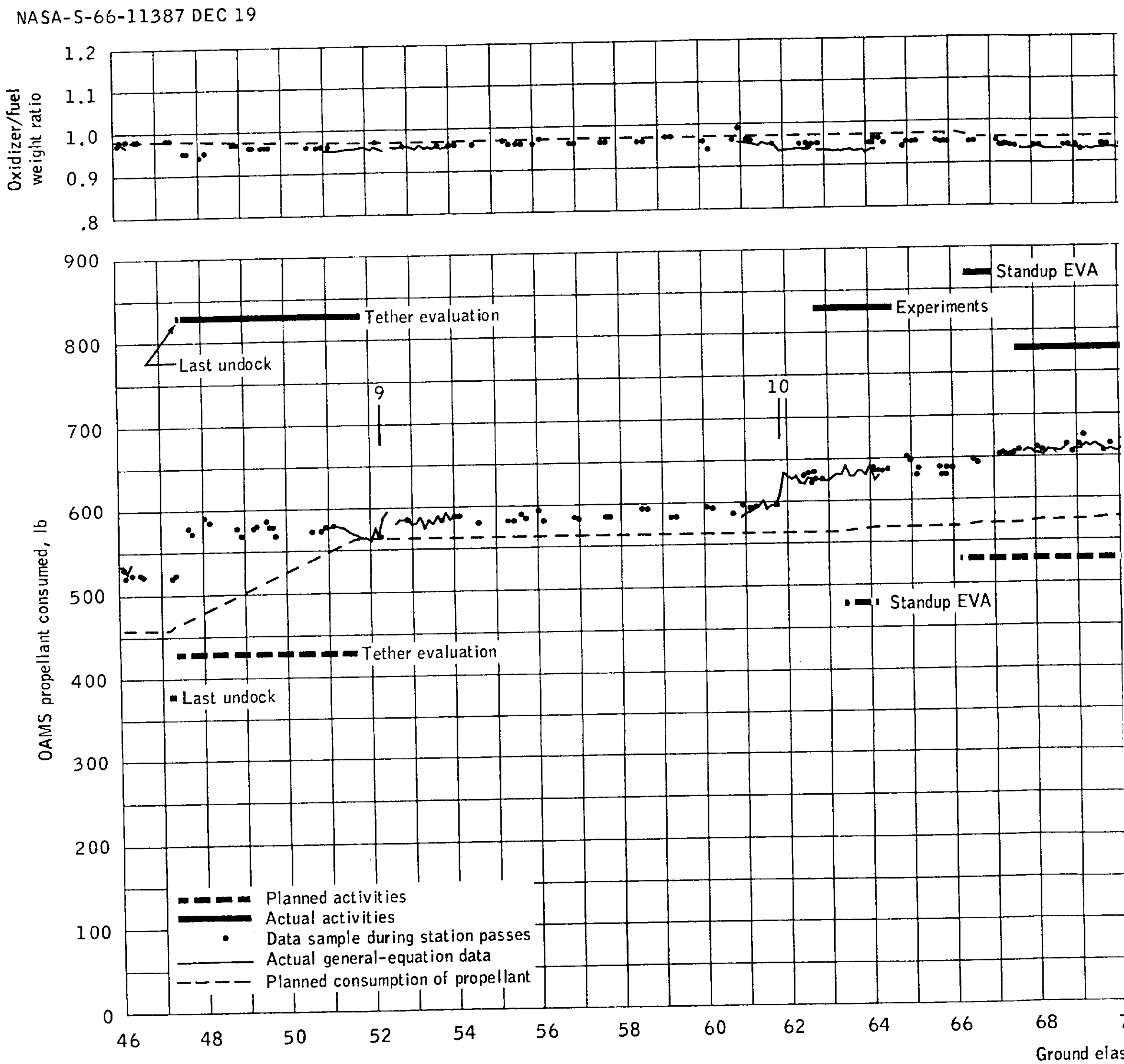


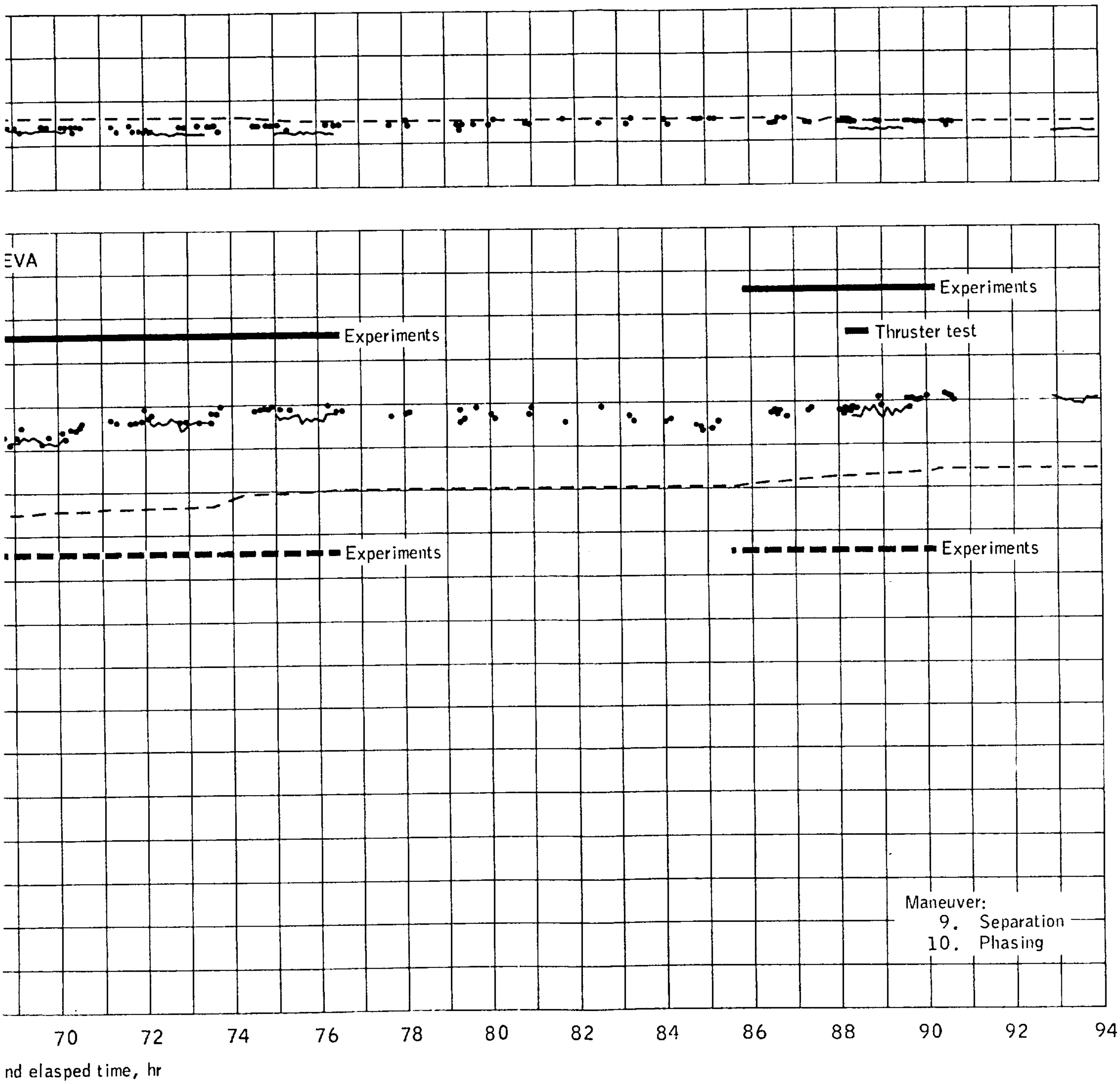
(a) 0 to 46 hours g.e.t.

Figure 5.1.8-2. - OAMS propellant consumption.


Experiments
Thruster test
Standup EVA
Umbilical EVA
Umbilical EVA
Experiments
Standup EVA
Planned activities
Actual activities
Data sample during station passes
Actual general-equation data
Planned consumption of propellant
Maneuver:
1. Separation and insertion
2. Phase adjust
3. Plane change
4. Corrective combination
5. Coelliptic
6. Terminal phase initiate
7. Midcourse corrections
8. Terminal phase finalize
22
24
26
28
30
32
34
36
38
40
42
44
46
round elapsed time, hr

2


NASA-S-66-11387 DEC 19
Oxidizer/fuel weight ratio
1.2
1.1
1.0
0.9
.8
900
800
700
600
500
400
300
200
100
0
OAMS propellant consumed, lb
Standup EVA
Experiments
Tether evaluation
Last undock
9
10
Standup EVA
Tether evaluation
Last undock
Planned activities
Actual activities
Data sample during station passes
Actual general-equation data
Planned consumption of propellant
46
48
50
52
54
56
58
60
62
64
66
68
Ground elas



(b) 46 to 94 hours g.e.t.

Figure 5.1.8-2. - Concluded.

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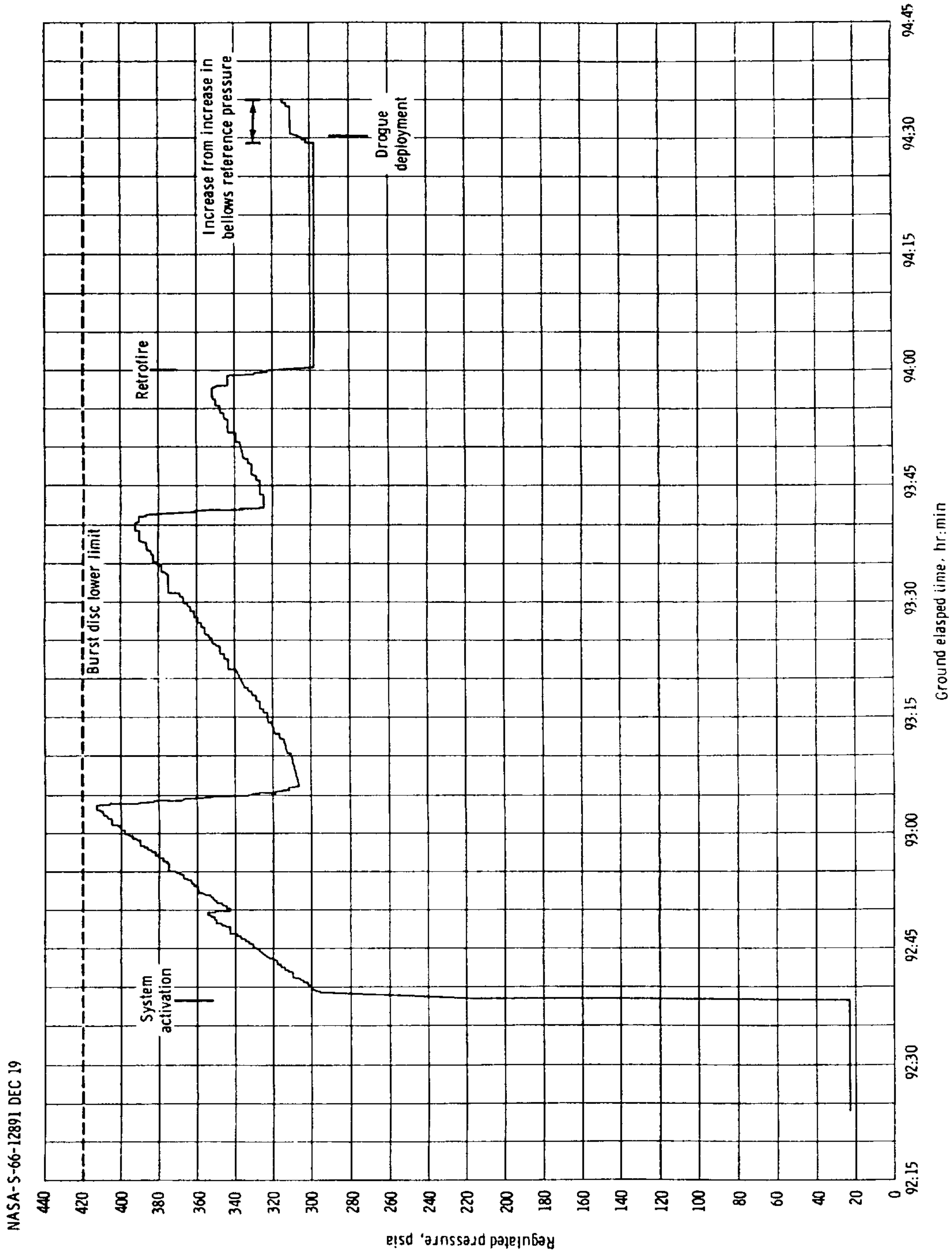


Figure 5.1.8.-3. - A-ring regulated pressure history.

5.1.9 Pyrotechnic System

All pyrotechnic functions were satisfactory. A postflight test revealed a low resistance in the firing circuit of one of the pyrotechnic switches. Examination of the switch revealed a residue which had formed after the normal detonation of the cartridge. The switch had functioned properly and the residue was the cause of the low resistance.

5.1.10 Crew Station Furnishings and Equipment

5.1.10.1 Crew station design and layout.- The overall design of the crew station was satisfactory for the Gemini XII mission. Minor discrepancies are discussed in the following paragraphs.

5.1.10.1.1 Displays and controls: The displays and controls functioned normally. The only exceptions were the secondary oxygen pressure gage (which varied during the launch phase) and the primary oxygen pressure gage (which fluctuated once during EVA). The oxygen gages returned to normal immediately and these problems did not recur during the mission.

Several circuit breakers were found in the off position at different times during the mission. Two of these circuit breakers were in the OAMS circuits. A check of the overhead circuit breaker panel when the radar problem was first noticed revealed that the OAMS motorized-shutoff-valve breaker and one of the OAMS emergency regulator bypass-valve breakers were off. The TONE/VOX circuit breaker on the left panel was inadvertently knocked off and stopped the digital clock which is also supplied through this breaker. In all cases the circuit breakers were turned back on and remained so during the rest of the mission.

5.1.10.1.2 Equipment stowage: Equipment stowage provisions for the mission were satisfactory and no significant problems were reported. Changes in the flight plan during the mission caused minor difficulties in the order of stowage and unstowage.

During reentry a pouch, attached by Velcro to the left sidewall, came loose and moved against the command pilot's legs. The pouch contained no heavy items and caused no serious problems.

Orbital stowage during the mission was less difficult than that reported for previous missions. Jettisoning of equipment during the second standup EVA relieved the congestion in the cabin area. Items such as lan yards with snap-hooks for loose equipment and pouches with Velcro flaps prevented any loss of equipment while the hatch was open.

5.1.10.1.3 Lighting: The interior lighting in the cabin was adequate for the mission. There was one discrepancy—the red filter over the left auxiliary light came off during the mission. The docking light was effective in illuminating the tether during the tether evaluation.

5.1.10.1.4 Crew furnishings: The ejection seats were not used except for restraint and support of the crew. A snap which holds the fabric cover over the D-ring housing broke early in the mission. The crew used the remaining good cover for D-ring protection on the right seat during EVA. During the first standup EVA, the command pilot's lap belt became disengaged and allowed him to float up in the cockpit. Use of the lap belt fitting after the EVA was normal and no further problems were noted. The command pilot subsequently reported that he may not have had the lap belt fastened completely.

5.1.10.2 Flight crew equipment.-

5.1.10.2.1 Still camera: The 70-mm general-purpose still camera did not function properly when used with the special lens for the S011 and the S051 experiments. The shutter stuck open during the S051 photography and no usable pictures were obtained. Postflight investigation showed that one of the iris shutter-leaf pins was bent out of position, possibly causing the lens to hang up. The investigation is continuing in an attempt to identify the cause of the damage (see section 8.9.5 for the latest information).

The shutter release cable used for remote control of the 70-mm general-purpose camera was difficult for the pilot to operate for extended periods in a pressurized space suit. Depressing the release button required finger action against the forces of the space suit gloves and induced fatigue very quickly.

5.1.10.2.2 Sequence cameras: Several problems were experienced with the three 16-mm sequence cameras supplied for this mission. The EVA camera used in the spacecraft adapter assembly failed immediately after it was mounted and turned on, and no usable photographs were obtained. Postflight investigation indicated that the shutter mechanism had jammed. After EVA the crew attempted to use the same magazine in a different camera. At this time the magazine jammed and would not run in the second camera. Inspection of the magazine after the flight indicated that a chip of metal had become trapped between the shaft and the bushing of the film metering sprocket, causing the sprocket to jam. The chip of metal had apparently been knocked off one of the rivets in the magazine.

The bracket for mounting the 16-mm camera in the adapter assembly also malfunctioned during the umbilical EVA. A small pin acting as a pivot for the thumb lever came loose. The pilot was able to install and remove the camera bracket by pressing directly on the ball detent lever with his finger.

5.1.10.2.3 Blood pressure bulb: The crew used the blood pressure bulb successfully to pressurize the cabin water tank when the water pressure dropped to a low level late in the mission. The bulb has been carried as a backup device for this purpose on all manned Gemini flights, but its use had never been required before.

5.1.10.2.4 Food: The meals consisted of rehydratable and bite-size foods similar to those provided for previous missions. There was a total of 24 meals on the spacecraft and the crew reported that this amount was adequate for the mission. The only problem reported with the food was that lumpy foods, such as shrimp creole, were difficult to extract from the food bags. The change in size from the bag to the extraction tube caused the lumpy foods to lodge at that point. This difficulty made eating these foods slightly slower and, in some cases, prevented the crew from emptying the bags completely.

5.1.10.2.5 Waste equipment: The launch-day urine collection device was worn by the pilot during each EVA period. Because of the long preparation time and length of the EVA, it had been planned to utilize the device for this operation. The pilot reported that reuse of the rubber cuff was undesirable; however, no difficulties were experienced.

5.1.10.3 Space suits and accessories.-

5.1.10.3.1 Command pilot's suit: The space suit configuration for the command pilot (G-4C-41 with a lightweight coverlayer) was basically the same as that used for the command pilots of previous Gemini missions. The command pilot's suit functioned normally throughout the mission.

5.1.10.3.2 Pilot's suit: The space suit for the pilot (G-4C-42) utilized the G-4C configuration pressure garment assembly of the same design as that provided the command pilot, but it was fitted with an extravehicular coverlayer, a coated polycarbonate pressure visor, and a single-lens sun-visor assembly. The pilot's suit functioned normally throughout both the intravehicular and extravehicular phases of the mission. At the end of the first daylight period of umbilical EVA, the pilot reported that the main entrance zipper to the suit had become noticeably warm in the posterior area. Inflight sequence photographs showed that the lower part of the pilot's zipper was exposed and susceptible to direct solar heating. The pilot assumed that this was the cause since the minor discomfort ceased after sunset.

5.1.10.3.3 Space suit accessories: The space-suit modifications noted in previous reports were included for this mission. Additional items of interest are:

(a) Ventilation gas connectors: Low-profile locking tabs and locking tab guards were incorporated on the suit gas connectors to provide increased insurance against inadvertent operation.

(b) EVA visor cover: An EVA visor cover was utilized over the gold-coated surface of the EVA visor during all intravehicular use to prevent possible surface damage. Postflight inspection showed the visor coating to be in excellent condition.

(c) Hose nozzle interconnects: The hose nozzle interconnects utilized a clip-on clamp for redundant locking of the latching tabs.

5.1.10.3.4 Space suit postflight review:

(a) Command pilot's suit: Postflight inspection of the command pilot's space suit assembly revealed the equipment to be in very good condition. The leak rate was 96 scc/min at 3.7 psig, which was well within the allowable limit of 1000 scc/min.

(b) Pilot's suit: The postflight inspection of the pilot's space suit assembly indicated the equipment to be in very good condition. The leak rate was 45 scc/min at 3.7 psig. Both wrist disconnects were very difficult to operate. Disassembly revealed salt precipitate on the latching dogs. The suit had been subjected to heavy salt spray during helicopter recovery of the pilot.

5.1.10.4 Extravehicular equipment.- All extravehicular equipment except one of the EVA cameras and its bracket operated satisfactorily during the Gemini XII mission. Three extravehicular periods were planned and conducted: the first was a standup EVA from 19 hours 29 minutes to 21 hours 58 minutes g.e.t.; the second, an umbilical EVA from 42 hours 49 minutes to 44 hours 55 minutes g.e.t.; and the third, a second standup EVA from 66 hours 6 minutes to 67 hours 1 minute g.e.t. The detailed activities are outlined in figure 5.1.10-1. The configuration of the pilot's equipment during the umbilical EVA was as shown in figure 5.1.10-2.

5.1.10.4.1 Extravehicular Life Support System: The Extravehicular Life Support System (ELSS) flown on Gemini XII consisted of a 25-foot umbilical, an electrical jumper cable, a chestpack, 18-inch and 22-inch ELSS hoses, two dual connector valves, and a restraint system. The umbilical flown on Gemini IX-A was modified slightly (tether shortened and tether breakout point on the man end repositioned) and reflown on

Gemini XII. The chestpack was similar in configuration to that flown on Gemini XI.

The ELSS performed normally during the EVA preparations. The pilot selected "medium" flow for the pre-egress period, and at hatch opening the setting was moved to "high" flow, where it remained for the duration of ELSS operation. The ELSS maintained a comfortable suit environment for the entire 126-minute EVA mission.

During the umbilical EVA the pilot commented twice that he was cool and once that his feet were cold. After the mission the pilot commented that his feet had been cold but not to the extent that there was any discomfort. This is in contrast with pilot reports on Gemini IX-A, X, and XI, after which the pilots reported that they were neither warm nor cool during EVA.

The oxygen allotment for umbilical EVA was 25 pounds, with 2.9 pounds scheduled for egress preparation and 22.1 pounds for a projected 2-hour and 10-minute timeline. In view of the experience of the Gemini XI pilot at the TDA, the use of the "medium-plus-bypass" flow mode was planned for all TDA work. This mode increases dry makeup-oxygen flow to the ELSS chestpack, hence increasing the capability of the ventilation gas to remove latent heat and to provide for helmet carbon-dioxide washout. In the event of workloads beyond the design limits, "medium-plus-bypass" flow would provide greater protection against visor fogging over that obtained in the normal "high" flow mode. However, the pilot elected to remain in the "high" flow mode for the entire hatch-open period because of the satisfactory cooling and absence of visor fogging experienced in that flow condition. He also stated that he felt that his work rate had not tasked the capability of the system in the high flow mode, and that he could have worked somewhat harder without discomfort.

Total ELSS oxygen usage for the 126-minute EVA period was approximately 18.9 pounds, which indicated a usage rate of 8.9 lb/hr, as compared with the measured value of 8.5 lb/hr during preflight testing.

The EVA pilot performed several tasks intended to evaluate any forces acting on him from either thrust or pressure force from the ELSS outflow. He reported that he was unable to detect any forces acting on him which might be attributed to the ELSS. There was no noticeable "float out" or "float up" tendency when he was standing in the cockpit with the hatch open.

After ingress, the cabin was repressurized using the ELSS self-contained emergency oxygen supply. "High-plus-bypass" flow was selected to increase the rate of cabin pressurization. Flow from this source was verified by the ELSS emergency alarm tone, which was actuated by flow through the emergency oxygen supply line.

5.1.10.4.2 Work station equipment: Work tasks of varying complexity were carried out as the means of evaluating body restraints. The equipment for these work tasks was located in the adapter work station and the TDA work station as shown in figures 3.1-6 and 3.4-2.

The following tasks were performed satisfactorily using the adapter foot restraints:

(a) Torquing boltheads

(b) Disconnecting and connecting a two-handed push-turn electrical connector

(c) Disconnecting and connecting a one-handed push-pull fluid quick-disconnect

(d) Cutting wire bundles of 6, 11, and 15 strands

(e) Cutting a high-pressure fluid hose.

The following tasks were performed satisfactorily using waist tethers only:

(a) Removing, installing, and tightening a Saturn bolt

(b) Connecting a large hook to a large ring and a small hook to a small ring

(c) Stripping nylon and steel Velcro strips up to five inches wide

(d) Disconnecting and connecting one-handed and two-handed push-turn electrical connectors

(e) Using an Apollo torque wrench

(f) Connecting the GATV/spacecraft tether

(g) Activating the S010 experiment.

All of the equipment functioned normally except as indicated in the following paragraphs.

While the pilot was using the conventional torque wrench for the Saturn bolt task, the zero shifted on the indicating dial. This shift probably resulted from exceeding the upper torque limit on the wrench. Subsequent torque readings were calculated by the pilot using the new zero reference.

The friction in the ratchet fitting on the torque wrench used in the adapter was too high for convenient use with the Saturn bolt. After the bolt was initially loosened, it turned more freely than the ratchet. When the wrench was used in an attempt to loosen or tighten the bolt, the tool would not ratchet and the bolt merely turned back and forth in the threads.

One of the penlights stowed external to the tool pouch on the adapter work station showed signs of overheating. The plexiglass lens had bulged outward, and the paint near the lens showed fingerprints which were matched with paint marks on the EVA thermal gloves. This penlight was stowed with the lens end exposed, and it is probable that direct solar heating caused the damage. The softening point of the plastic lens material is about 190° F.

The pilot reported that the rubber strap attached to the simulated Saturn bolt in the adapter section had partially melted and stuck to the bolt mounting box. Since the Saturn bolt was immediately adjacent to the overheated penlight, it is likely that the rubber strap was also affected by the direct solar heat.

In order to complete the bolt task, the pilot had to pry the rubber strap from the bolt. After removing the bolt and washer, the pilot lost his grasp on them, but he was able to recover both before they drifted beyond his reach. Removal and replacement of the Saturn bolt took a total of nine minutes, using either the foot restraints or the waist tethers. Part of this time was taken in connecting the waist tethers and removing the feet from the foot restraints.

5.1.10.4.3 Tethers: Equipment tethers were used to maintain control of loose equipment during the EVA periods. The basic purpose of the tethers was to prevent loss of the items being installed or being passed between pilots while the hatch was open. The tethers were adequate for the mission in that no equipment was inadvertently lost.

5.1.10.4.4 Body restraints: Evaluation of body restraints was one of the primary objectives of the EVA in this mission, and several types of restraints were tried. Foot restraints were used for the first time during this mission. The foot restraints were made of fiber glass and molded to fit the pilot's boots. The restraints were mounted in the adapter assembly in a position to give proper access to the work station. The pilot reported that the foot restraints were excellent and allowed him to control his body position readily, leaving both hands free. He was able to perform the tasks outlined in paragraph 5.1.10.4.2 without difficulty while using the foot restraints. He also evaluated the use of a single foot restraint and found it to be nearly as useful as two.

For the torquing task in the adapter work station, by using the foot restraints, the pilot was able to apply a torque between 200 and 250 inch-pounds in the clockwise direction. With a 9-inch handle on the wrench, this torque is equivalent to 22 to 28 pounds of force. In the counter-clockwise direction the pilot was able to apply approximately 200 inch-pounds torque; however, the exact values were not obtained because of the zero shift in the wrench.

Waist tethers were also used for the first time during this mission. The tethers were attached to the parachute harness near the waist and were adjustable in length from 21 to 32 inches. Hooks on the ends were snapped into rings in the adapter section or on the TDA. These tethers prevented the pilot from pushing himself away from the designated work area. He was able to perform the assigned tasks without difficulty while using the waist tethers. The use of body tethers eliminated the constant concern about drifting into an unknown and uncontrolled body position, and allowed the pilot to concentrate directly on the task to be performed.

The pilot found that when he was working in the adapter assembly with waist tethers, his feet tended to drift away and his head tended to drift in toward the work. When he pushed his head away from the work station, he tended to drift up (in the direction away from his feet). As a result, the optimum work location with waist tethers was 20 to 30 inches higher than the line between the tether attachment rings. The pilot also commented that a wider spread between the two tether attachment points would probably have given better lateral stability than the 28-inch width provided on the adapter work station.

In comparing waist tethers and foot restraints, the pilot indicated that the foot restraints were ideal for all tasks which were reasonably stationary. Good mobility around the area of the foot restraints was consistently possible; however, the waist tethers permitted more movement and were entirely acceptable for all of the tasks evaluated. The pilot stated that they appeared to be superior to the foot restraints when greater freedom of movement was required.

Use of the waist tethers on the TDA was found to be satisfactory for all tasks attempted. The pilot was able to shift waist tether attachments from one place to another with relative ease. When working with the Apollo wrench he was able to apply torque values in excess of 100 inch-pounds without difficulty. With a 5-inch handle on the Apollo wrench, this value corresponds to 20 pounds force on the wrench.

A standup tether was used by the extravehicular pilot during the standup EVA's and functioned satisfactorily. This tether was attached to the parachute harness and connected to the left side of the seat. The main purpose of the tether was to prevent any load from being placed on the oxygen or electrical lines.

5.1.10.4.5 Handholds: Both fixed and portable handholds were evaluated during the EVA. Most of the fixed handholds had been evaluated on previous flights but the portable ones were flown for the first time on this mission.

A portable handrail was installed between the hatch opening and the TDA cone. The handrail was telescoped to full extension and installed during the first standup EVA. The primary purpose was to provide a means of moving from the hatch area to the nose of the spacecraft. The pilot moved along the handrail with a sideways motion instead of a hand-over-hand movement. He found the handrail to be satisfactory for transit between the cockpit and the TDA.

Individual portable handholds were also evaluated on the mission. The handholds were made from rectangles of sheet metal, about three by eight inches, with handles on top. The rectangular sections were covered with Velcro pile to mate with patches of Velcro hooks on the adapter station and on the TDA. Evaluation in flight showed that the handholds had a tendency to start peeling off when a load was applied across the short dimension of the rectangle. They held well in the lengthwise direction.

Pip-pins which could be inserted into receptacles on the TDA were also used for handholds. The pip-pins were made with a T-handle on the top and a large loop on the side for use as attachments for the waist tether hooks, as well as handholds. The pilot found the pip-pins to be satisfactory for both uses when they were restrained from rotating in the receptacle. Part of the pip-pin receptacles were designed to prevent rotation of the pin and this installation provided better torquing control and ease of tether installation.

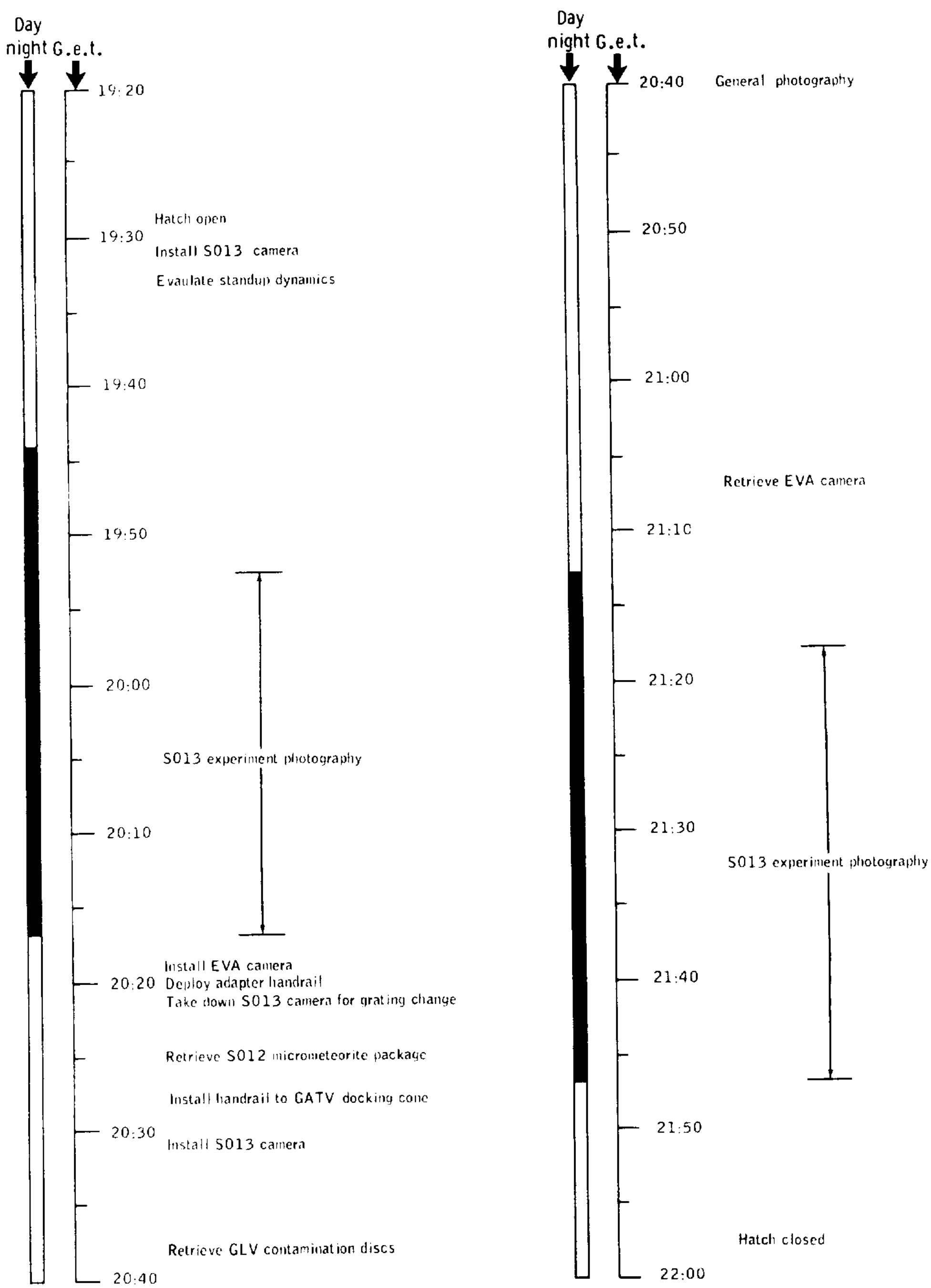
Fixed handholds were evaluated on the adapter surface, in the adapter, and on the TDA cone. All of these had been evaluated on previous missions, and they were again found to be satisfactory.

5.1.10.4.6 Validity of EVA simulations: One of the problems reported after previous EVA missions was that the ground simulations did not provide a realistic simulation of the extravehicular environment. Equipment designs which had been evaluated as satisfactory during zero-gravity aircraft simulations occasionally proved inadequate in space flight because of the limitations of the short-term zero-gravity periods in the aircraft.

The preparations for Gemini XII included two intensive periods of underwater simulation with the prime pilot and realistic underwater mock-up hardware. During the mission the pilot found that the conditions he experienced in umbilical EVA were very similar to the conditions he experienced in the underwater simulations. As a part of the postmission evaluation, the Gemini XII pilot participated in another underwater simulation using his flight space suit and flight-configuration mock-up hardware. In the postflight simulation he was able to verify that, for the Gemini XII EVA mission, all the work tasks and the use of body-positioning equipment could be simulated underwater with high fidelity. The space-suit mobility forces and the reactions to body forces were the dominant factors. Hydrodynamic damping forces and variations in buoyancy were small in comparison with the suit forces. Consequently, these small forces had a negligible effect on the overall results of equipment and procedures evaluation. The close correlation between in-flight EVA and the underwater simulation indicated that, if a task could be readily accomplished underwater, there was a high probability that it could be readily accomplished in flight.

5.1.10.5 Bioinstrumentation.- The bioinstrumentation equipment performed satisfactorily during this mission, and satisfactory biomedical data were obtained on both pilots.

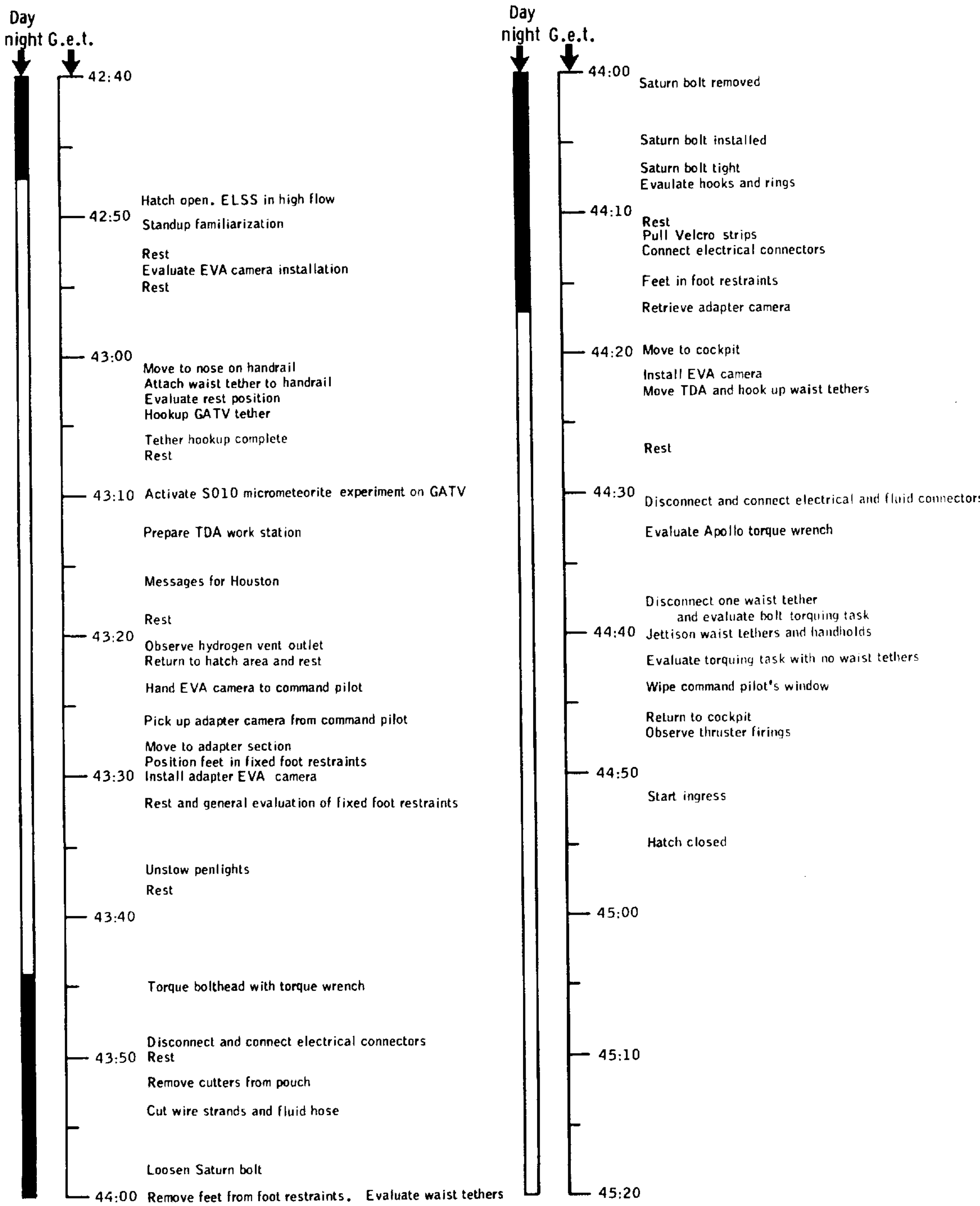
NASA-S-66-11332 DEC 14



(a) First standup.

Figure 5.1.10-1. - EVA events.

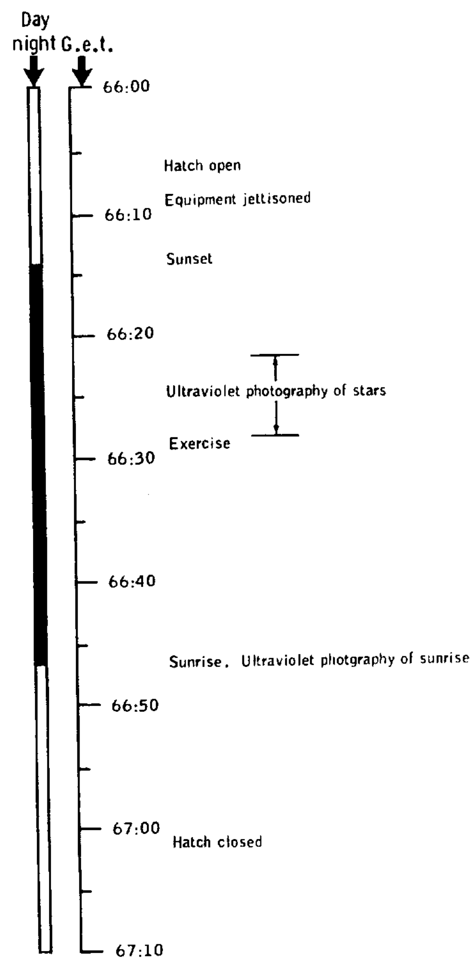
NASA-S-66-11333 DEC 14



(b) Umbilical.

Figure 5.1.10-1. - Continued.

NASA-S-66-11334 DEC 14



(c) Second standup.

Figure 5.1.10-1.- Concluded.

NASA-S-67-307 JAN 9

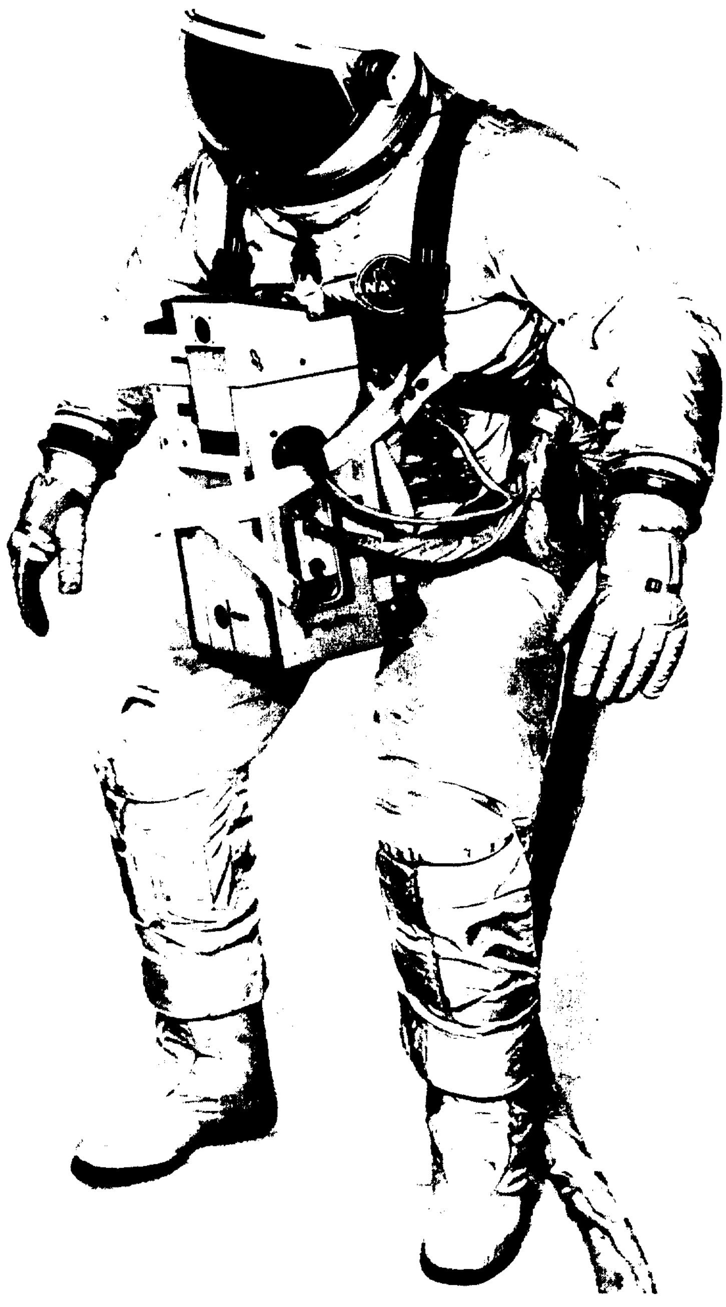


Figure 5.1.10-2. - Gemini XII extravehicular equipment.

5.1.11 Landing System

The parachute landing system provided a safe landing for the spacecraft and crew. All sequential events occurred when initiated by the crew and all took place within established tolerances. Figure 5.1.11-1 illustrates the sequence of the major landing system events with respect to ground elapsed time and pressure altitude.

The crew reported that they experienced much higher forces than expected when the spacecraft landed. Photographic coverage of the final descent and landing indicates that the parachute descent was normal and the spacecraft was in the correct attitude for landing. Recorded data indicate that the landing system events were initiated at the correct altitudes and that the total time from landing system deployment to landing was nominal. It is therefore concluded that the high landing forces resulted from a combination of wind drift, normal spacecraft oscillation on the parachute, and the angle at which the spacecraft contacted the wave. (For further information, refer to section 5.1.1.)

A five-gore tuck in the main parachute could be seen in the photographic coverage of the spacecraft landing. Similar tucking occurred during several previous missions and during the qualification program. This phenomenon is a result of excess material in the parachute and has been previously determined to have a negligible effect on the landing system performance.

The main parachute, the drogue, the pilot parachute, and the Rendezvous and Recovery (R and R) section were not recovered because they sank during the recovery operations. Therefore, no engineering inspection or analysis of these items could be made.

NASA-S-66-11279 DEC 7

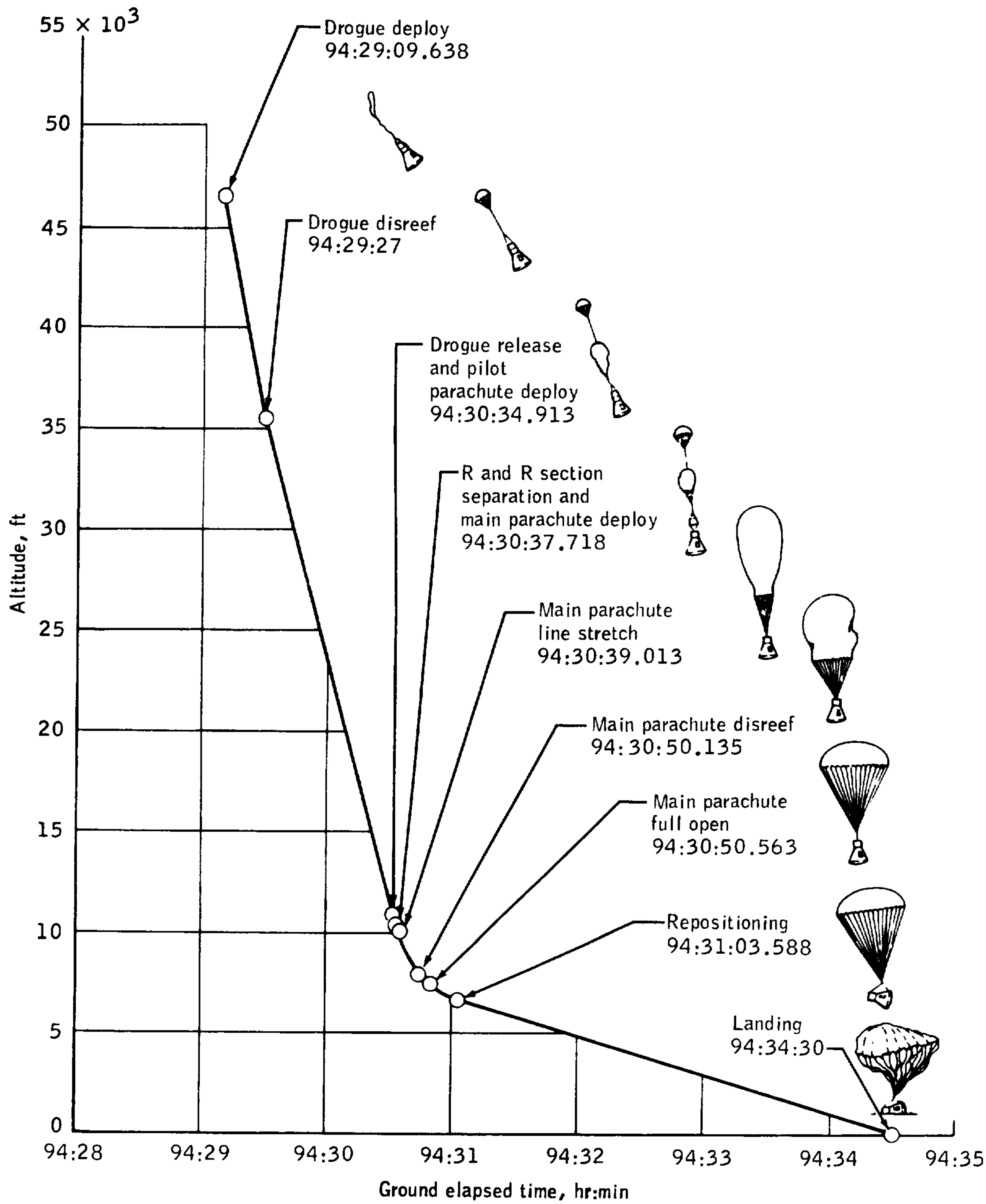


Figure 5.1.11-1. - Landing system performance.

5.1.12 Postlanding

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

5.2 GEMINI LAUNCH VEHICLE PERFORMANCE

The Gemini XII Launch Vehicle (GLV-12) was launched on time after a countdown that involved no unplanned holds. All systems performed satisfactorily and the spacecraft was inserted into a satisfactory orbit.

Tracking films indicate that both Stage I propellant tanks ruptured after the staging sequence was completed. The event had no detectable effect on the satisfactory operation of Stage II.

Calculations performed during the countdown indicated that the nominal payload capability would be 8851 pounds and the minimum payload capability (minus 3 sigma) would be 8223 pounds, providing a payload margin of minus 73 pounds. The postflight reconstructed burning-time margin was plus 2.53 seconds, indicating that the achieved vehicle performance was 9181 pounds or 330 pounds more than the predicted nominal payload capability and 885 pounds more than the actual payload weight.

5.2.1 Airframe

Flight loads and vibration environment on GLV-12 were comparable to those of previous flights and were well within the structural capability of the vehicle.

5.2.1.1 Structural loads.- During the prelaunch phase, ground winds of approximately seven miles per hour caused a peak GLV bending moment equal to two percent of the design-limit wind-induced oscillatory bending moment.

Estimated loads on the launch vehicle during the flight are shown in the following table.

Station	Maximum $q\alpha$			Pre-BECO		
	Compression load, lb	Percent of design		Compression load, lb	Percent of design	
		Limit	Ultimate		Limit	Ultimate
276	28 050	33.4	26.7	50 210	59.8	47.8
320	140 070	45.7	36.6	270 690	88.4	70.7
935	416 290	69.4	55.5	433 200	72.2	57.8

5.2.1.2 Longitudinal oscillation (POGO).- Accelerometer data indicated the same intermittent characteristic of the suppressed longitudinal oscillation that had been experienced on previous flights, but the oscillations were of shorter duration. Maximum response at the spacecraft/launch vehicle interface occurred for about two seconds at lift-off + 126.1 seconds and had an amplitude of ±0.14g at a frequency of 11.2 cps as evidenced in the filtered data.

5.2.1.3 Post-SECO disturbance.- Only one indication of a post-SECO disturbance was noted on the low-range axial accelerometer. The magnitude of this disturbance was 0.02g peak-to-peak at SECO + 4.1 seconds.

5.2.1.4 Post-staging event.- Motion-picture tracking films indicate a normal staging sequence and separation followed by an amber cloud, then a white cloud, providing evidence that both Stage I propellant tanks ruptured. All Stage I measurements are disconnected at staging so that telemetered data do not indicate a cause of the tank rupture, but the event had no detectable effect on the satisfactory operation of Stage II.

5.2.2 Propulsion

5.2.2.1 Engines.-

5.2.2.1.1 Stage I: The Stage I engine performance throughout the flight was nominal. Corrected to standard inlet conditions, the Stage I engine mixture ratio after the ignition signal was 1.00 percent lower than the acceptance test value. This value is within the 3-sigma run-to-run repeatability of ±1.38 percent, as shown in table 5.2-I. The lower than predicted mixture ratio resulted in a fuel depletion shutdown. The cause of the mixture ratio shift has not been determined at present, but further investigations are being conducted.

The start transient was normal, although the true magnitude of the chamber pressure spike was obscured by the overdamped type of transducers used on GLV-12.

The steady-state thrust and specific impulse were very close to the predicted values. The shutdown transient was normal for a fuel-exhaustion shutdown.

5.2.2.1.2 Stage II: The Stage II engine performance showed good agreement with the predicted values, as noted in table 5.2-II. The engine mixture ratio, corrected to standard inlet conditions, was 0.40 percent lower than the acceptance test value but well within the 3-sigma limits of ±2.28 percent. The start transient was within the range experienced on other GLV's and Titan II missiles, and is considered normal. The steady-state thrust and specific impulse were close to the predicted values.

The Stage II engine shutdown was initiated by guidance command. The shutdown impulse was less than the GLV-11 shutdown impulse, as shown in the following table:

Vehicle	Shutdown impulse, lb-sec	
	Predicted	Actual
GLV-11	36 100 ±7000	34 552
GLV-12	36 100 ±7000	33 971

One minor post-SECO disturbance was seen in the data at SECO plus 4.10 seconds.

5.2.2.2 Propellants.-

5.2.2.2.1 Loading: GLV-12 was loaded with propellants to within the required ±0.35 percent. The propellant loading summary for the launch is shown in the following table. The actual flight loads were calculated from the GLV-12 engine performance and level-sensor data.

Tank	Load, lb		Difference, percent
	Requested	Actual	
Stage I oxidizer	171 199	171 468	+0.16
Stage I fuel	90 140	90 042	-0.11
Stage II oxidizer	38 882	38 758	-0.32
Stage II fuel	22 126	22 048	-0.35

5.2.2.2.2 Utilization: A Stage I oxidizer outage is the amount of usable oxidizer remaining after a fuel depletion shutdown. A 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. The predicted and actual outages by both stages are shown in the following table:

Engine	Type	Predicted mean, lb	Predicted maximum, lb	Actual, lb
Stage I	Oxidizer	882	2584	133
Stage II	Oxidizer	208	621	99

The amount of propellants remaining at Stage II engine shutdown could have sustained Stage II flight an additional 2.53 seconds. This is 0.95 of a second greater than the burning-time margin of 1.58 seconds predicted at Stage I engine ignition.

5.2.2.3 Pressurization.- The predicted and actual GLV-12 tank pressures for various flight times are given in tables 5.2-III and 5.2-IV. The close agreement between predicted and actual pressures indicates nominal performance of the GLV pressurization system.

5.2.3 Flight Control System

The Flight Control System performed normally during both Stage I and Stage II flight. The primary system remained in command throughout the flight. The agreement of the secondary system with the primary system indicated proper operation, and switchover could have been accomplished successfully at any time during the powered phase.

5.2.3.1 Stage I flight.- Normal actuator transients occurred during the engine ignition. The peaks of actuator travel during the ignition transient and during the holddown null check period are listed in table 5.2-V.

The combination of thrust-vector and engine misalignments at full thrust initiated a roll transient at lift-off requiring a momentary roll rate correction of 1.0 deg/sec counterclockwise. A normal recovery from this transient followed.

The Three Axis Reference System (TARS) roll and pitch programs were performed as planned. The planned and actual rates and times are listed in table 5.2-VI. The TARS discretes were initiated at the specified times.

The primary (TARS) and secondary (Inertial Guidance System (IGS)) attitude error signals correlated well throughout Stage I flight. These attitude errors indicate the response of the control system to the first-stage guidance programs and to 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-VII. The dispersions between the primary and secondary attitude error signals were the combined result of drift and open-loop guidance programs and were of normal magnitude.

5.2.3.2 Staging sequence.- Data during the staging sequence indicated normal staging rates and attitudes. The maximum rates and attitude changes during staging are given in table 5.2-VIII.

5.2.3.3 Stage II flight.- Primary Flight Control System (TARS) pitch and yaw responses to radio guidance commands were satisfactory. The pitch and yaw steering commands transmitted to the launch vehicle during Stage II flight are discussed in section 5.2.5. The Stage II attitude biases resulted from thrust-vector misalignment, center-of-gravity offset from the longitudinal axis, and roll-thrust offset from the longitudinal axis. The primary (TARS) and secondary (IGS) attitude error signals were as shown in figure 5.1.5-1.

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-IX.

5.2.4 Hydraulic System

The GLV hydraulic systems performed satisfactorily during both Stage I and Stage II operation. No anomalous pressures were noted during ignition transients or steady-state flight. Selected hydraulic system pressures are shown in the following table.

[redacted]

Event	Hydraulic pressure, psia		
	Stage I system		Stage II system
	Primary	Secondary	
Starting transient (minimum)	2740	--	--
Starting transient (maximum)	3150	3300	3820
Steady-state operation	3020	2970	2910
BECO	2740	2710	--
SECO	--	--	2790

5.2.5 Guidance System

Performance of the Stage I and Stage II guidance system was satisfactory throughout powered flight and resulted in attaining an acceptable spacecraft velocity vector at insertion.

5.2.5.1 Programmed guidance.- Programmed guidance, as shown by actual and nominal data in section 5.2.3 (table 5.2-VI), is considered to have been within acceptable limits. The trajectory was nominal and the errors at BECO, compared with the no-wind prelaunch nominal trajectory, were 5.0 ft/sec high in velocity, 948 feet low in altitude, and 0.25 of a degree low in flight-path angle.

5.2.5.2 Radio guidance.- The Radio Guidance System (RGS) acquired the pulse beacon of the vehicle, tracked in the monopulse automatic mode, and was locked on continuously from lift-off (LO) to 34 seconds after SECO. Track was maintained to an elevation angle of 3.3 degrees above the horizon. The average strength of the signal received at the central station during Stage II operation was satisfactory. Rate lock was continuous from LO + 29.5 seconds to LO + 374.8 seconds (31.3 seconds after SECO).

Commencing at LO + 168.01 seconds, pitch steering commands were initiated, as planned, by the airborne decoder. At that time, an initial 7-percent pitch-down steering command (0.14 deg/sec) was given for 0.5 of a second, followed by the characteristic 100-percent pitch-down steering command (2.0 deg/sec) for 2.0 seconds. The steering gradually decreased during the following 10.0 seconds to continuous pitch-down commands of less than 0.16 deg/sec until LO + 321 seconds. At this time, because of

[redacted]

noisy tracking data, the rates appeared oscillatory. This particular phenomenon is a normal characteristic of tracking data when the ground guidance system is being influenced by atmospheric effects. Past experience has shown that the high frequency noise increases as the tracking elevation angle decreases. As a result, the peak amplitude of steering commands range from plus 0.16 deg/sec to minus 0.28 deg/sec until termination of guidance (SECO minus 2.5 seconds).

Yaw steering was initiated at LO + 168.01 seconds, with the first command being sent, as expected, at LO + 168.51 seconds. As a result, yaw-left commands of 100 percent (2.0 deg/sec) were sent for 2.0 seconds. The steering gradually returned within 10 seconds to yaw-right commands of less than 0.04 deg/sec, and remained within that magnitude until termination of guidance. At SECO + 20 seconds, the yaw velocity was 6.0 ft/sec and the yaw position was minus 6274 feet, as compared with the planned values of 1.9 ft/sec and minus 4627 feet (prelaunch guidance residuals due to insertion targeting accuracies).

SECO occurred at LO + 343.539 seconds at an elevation angle of 6.28 degrees above the horizon. The conditions at SECO + 20 seconds were within 3-sigma limits. Table 4.3-I shows a comparison of the actual values with the planned values. The errors at SECO + 20 seconds may be attributed primarily to the noise in the guidance data and to the lower-than-expected shutdown thrust transient (tail-off). Analysis indicates that the low shutdown transient contributed 5.6 ft/sec to the estimated total 13.0 ft/sec underspeed at SECO + 20 seconds. Vehicle attitude rates at SECO + 20 seconds were 0.45 deg/sec pitch up, 0.48 deg/sec yaw right, and 0.20 deg/sec roll clockwise.

The ground-based A-1 guidance computer, in conjunction with the MOD III Tracking and Missile-Borne Guidance System, performed satisfactorily during prelaunch and flight. No anomalies were encountered with the airborne pulse, rate, and decoder hardware. All guidance discretes were properly generated and executed as required.

The target ephemeris data were satisfactorily transmitted and verified at approximately T minus 25 minutes between the Real Time Computer Complex at Houston and the Guided Missile Computer Facility at Cape Kennedy. After lift-off, IGS updates were correctly sent by the ground-based computer and are listed in the following table.

Time from lift-off, sec		Crossrange velocity, ft/sec
Update reference	Update transmission	
100	105	+5.05
140	145	-235.09

5.2.6 Electrical

All airborne electrical power supplies performed satisfactorily throughout the flight. Both the Instrumentation and the Auxiliary Power Supplies operated nominally at 29.7 and 30.0 Vdc, respectively.

5.2.7 Instrumentation

5.2.7.1 Ground.- For the scheduled launch on November 9 and all subsequent preparations, including the launch on November 11, 1966, there were 153 recorder channels utilized on the Launch Complex 19 landline system. Data acquisition was 100 percent with no anomalies. The umbilical connector separated without incident from the vehicle in the planned sequence, and the event was complete in 0.83 of a second.

5.2.7.2 Airborne.- The airborne instrumentation system was identical to that used for the last three GLV's. The system consisted of 188 measurements programmed for use, and there were no major anomalies. Measurement 0514, oxidizer pump inlet temperature of the second-stage engine, falsely indicated a temperature shift prior to launch. This bias, approximately 10 degrees high, remained throughout the flight.

The normal telemetry data loss during staging RF blackout lasted only 280 milliseconds. This was the first GLV utilizing Telemetry Receiving Station 4 (TEL-4), and data reception was good. The final loss of telemetry signal, as monitored at TEL-4, occurred at lift-off + 633 seconds (26 seconds after spacecraft separation).

5.2.8 Malfunction Detection System

Performance of the Malfunction Detection System (MDS) during preflight checkout and flight was satisfactory. Flight data indicated that all MDS components functioned properly. MDS parameters are shown in table 5.2-X.

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

Switch	Condition	Actuation time from lift-off, sec	Pressure, psia
Stage I			
Subassembly 1 MDTCPS	Make	-2.304	590
	Break	+154.694	567
Subassembly 2 MDTCPS	Make	-2.304	590
	Break	+154.672	550
Stage II			
Subassembly 3 MDFJPS	Make	+155.468	(a)
	Break	+343.680	(a)

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

5.2.8.3 Tank pressure indications.- All tank pressure indicators performed acceptably and all paired sensors agreed within specification throughout flight.

5.2.9 Range Safety and Ordnance Systems

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

5.2.9.1 Flight termination system.- Both GLV command receivers received adequate signal for proper operation throughout powered flight and beyond spacecraft separation. The following command facilities were used:

Time from lift-off, sec	Facility
0 to 67	Cape Kennedy - 600-watt transmitter and single helix antenna
67 to 120	Cape Kennedy - 10-kilowatt transmitter and quad-helix antenna
120 to 259	Grand Bahama - 10-kilowatt transmitter and steerable antenna
259 to 434	Grand Turk - 10-kilowatt transmitter and steerable antenna
434 to 722	Antigua - 10-kilowatt transmitter and steerable antenna

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

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

5.2.10 Prelaunch Operations

5.2.10.1 Launch attempts.- On November 8, 1966, during the initial F minus 1-day precount testing, a problem was indicated by the Flight Control System Test Set (FCSTS). The secondary system Stage I and Stage II rate gyro spin motor rotation detectors (SMRD) indicated NO-GO and the 26-volt, 800-cps hold-fire indication was NO-GO. Correction of this condition resulted in the replacement of both the secondary autopilot package and the secondary Stage I rate gyro package. Subsequent testing at Baltimore isolated the cause of the malfunction to a power transistor in the 800-cps power supply in the autopilot package.

On November 9, 1966, during the second F minus 1-day precount tests, the Stage II secondary rate gyro SMRD no-go light was illuminated for 45 seconds on the FCSTS. This problem was resolved by replacing the secondary autopilot package. The problem has not been isolated to any component in the autopilot package.

5.2.10.2 Final countdown.- Propellant loading was initiated at 2:53 a.m. e.s.t on November 11, 1966, and was completed by 6:13 a.m. e.s.t. The primary propellant conditioning and loading system was used throughout the loading, and the total elapsed time was 3 hours 20 minutes. The range sequencer was initiated at 6:53 a.m. e.s.t. (T minus 530 minutes) on November 11, 1966. The POGO hardware was remotely charged at T minus 63 minutes. The GLV progressed throughout the countdown with no problems. The scheduled 6-minute hold, programmed for T minus three minutes, lasted 3 minutes 29 seconds. Lift-off was accomplished at 3:46 p.m. e.s.t., without incident.

[redacted]

TABLE 5.2-I.- STAGE I ENGINE PERFORMANCE

Parameter	Preflight prediction	Postflight reconstruction	Percent difference
Standard inlet condition performance			
Thrust, lb	433 538	433 973	+0.10
Specific impulse, lb-sec/lb	259.90	260.29	+0.15
Engine mixture ratio, oxidizer to fuel	1.9247	1.9196	-0.26
Oxidizer overboard flow rate, lb/sec	1097.40	1095.88	-0.14
Fuel overboard flow rate, lb/sec	570.71	571.40	+0.12
Flight average performance			
Thrust, lb	461 905	458 905	-0.46
Specific impulse, lb-sec/lb	276.64	277.03	+0.14
Engine mixture ratio, oxidizer to fuel	1.9166	1.9057	-0.51
Oxidizer overboard flow rate, lb/sec	1094.83	1086.09	-0.80
Fuel overboard flow rate, lb/sec	571.78	570.46	-0.23

[redacted]

[redacted]

TABLE 5.2-II.- STAGE II ENGINE PERFORMANCE

Parameter	Preflight prediction	Postflight reconstruction	Percent difference
Standard inlet condition performance			
Thrust, lb	100 689	99 369	-1.31
Specific impulse, lb-sec/lb	312.04	312.58	+0.17
Engine mixture ratio, oxidizer to fuel	1.8106	1.7810	-1.63
Oxidizer overboard flow rate, lb/sec	208.03	203.74	-2.06
Fuel overboard flow rate, lb/sec	114.64	114.16	-0.42
Flight average performance			
Thrust, lb	100 490	99 296	-1.19
Specific impulse, lb-sec/lb	312.93	313.22	+0.09
Engine mixture ratio, oxidizer to fuel	1.7685	1.7526	-0.90
Oxidizer overboard flow rate, lb/sec	205.29	202.00	-1.61
Fuel overboard flow rate, lb/sec	115.83	115.01	-0.71

[redacted]

TABLE 5.2-III.- STAGE I ULLAGE GAS PRESSURES

Tank	Lift-off		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	27.7	19.5	19.1	18.5	18.4	20.0	19.3
Fuel	29.5	26.1	22.7	22.4	22.5	22.4	23.3	23.3

TABLE 5.2-IV.- STAGE II ULLAGE GAS PRESSURES

Tank	Lift-off + 155 sec[a]		Lift-off + 205 sec		Lift-off + 255 sec		Lift-off + 335 sec	
	Predicted, psia	Actual, psia	Planned, psia	Actual, psia	Planned, psia	Actual, psia	Planned, psia	Actual, psia
Oxidizer	55.5	55.7	21.0	21.8	13.3	13.7	8.5	8.9
Fuel	50.5	49.1	48.0	47.1	47.9	47.6	47.5	48.8

[a]Staging.

TABLE 5.2-V.- TRANSIENTS DURING STAGE I HOLDDOWN PERIOD

Actuator	Maximum travel during ignition, in.	Time from lift-off, sec	Maximum travel during holddown null check, in.
Pitch, 1_1	-0.05	-2.43	0.04
Yaw-roll, 2_1	+0.07	-2.41	0.03
Yaw-roll, 3_1	+0.07	-2.42	0.02
Pitch, 4_1	-0.06	-2.43	0.03

TABLE 5.2-VI.- ROLL AND PITCH PROGRAMS

Program	Program times, LO + sec		Torquer monitor, deg/sec	Nominal rate, deg/sec
	Actual	Nominal		
Roll				
Start	7.99	8.00	-1.25	-1.25
Stop	20.43	20.48		
Pitch, Step 1				
Start	23.05	23.04	-0.69	-0.709
Pitch, Step 2				
Start	88.26	88.32	-0.50	-0.516
Pitch, Step 3				
Start	118.96	119.04	-0.25	-0.235
Stop	162.41	162.56		

TABLE 5.2-VII.- MAXIMUM STAGE I RATES AND ATTITUDE ERRORS

Axis	Maximum rate, deg/sec	Time from lift-off, sec	Maximum attitude error, deg	Time from lift-off, sec
Pitch	+0.37 -0.99	0.0 25.3, 60.2, and 80.9	+1.26 -0.21	113.5 43.3, 62.5, and 120.5
Yaw	+0.49 -0.29	73.0 76.6	+0.74 -0.47	74.3 119.9
Roll	+0.50 -1.61	154.6 8.6	+0.53 -0.26	154.7 0.3

TABLE 5.2-VIII.- MAXIMUM STAGING RATES AND ATTITUDE ERRORS

Axis	Maximum rate, deg/sec	Time from BECO[a], sec	Maximum vehicle attitude change, deg	Time from BECO, sec
Pitch	+3.92 -4.23	0.47 0.49	-0.58	1.6
Yaw	+1.34 -1.44	0.73 0.47	+0.95	2.6
Roll	+1.86 -3.62	1.44 0.74	-1.68	1.1

[a]BECO occurred 154.75 seconds after lift-off.

TABLE 5.2-IX.- VEHICLE RATES BETWEEN SECO

AND SPACECRAFT SEPARATION

Condition	Rate, deg/sec
Pitch:	
Maximum positive rate at SECO + 2.03 and 7.23 sec	+0.80
Maximum negative rate at SECO + 0.03 sec	-0.09
Rate at SECO + 20 sec	+0.50
Rate at spacecraft separation (SECO + 23.20 sec)	+0.50
Yaw:	
Maximum positive rate at SECO + 10.33 sec	+0.47
Maximum negative rate at SECO + 1.63 sec	-0.69
Rate at SECO + 20 sec	+0.47
Rate at spacecraft separation (SECO + 23.20 sec)	+0.47
Roll:	
Maximum positive rate at SECO + 0.53 sec	+0.40
Maximum negative rate at SECO + 8.18 sec	-0.38
Rate at SECO + 20 sec	+0.19
Rate at spacecraft separation (SECO + 23.20 sec)	+0.29

TABLE 5.2-X.- MALFUNCTION DETECTION SYSTEM SWITCHOVER PARAMETERS

Parameter	Switchover setting	Maximum or positive (a)	Time from lift-off, sec	Minimum or negative (b)	Time from lift-off, sec
Stage I primary hydraulics	Shuttle spring (1500 psia equivalent)	3160 psi	-2.08	2710 psi	-2.32
Stage I tandem actuators					
No. 1 subassembly 2 pitch	±4.0 deg	0.53 deg	62.5	0.35 deg	92.0
No. 2 subassembly 2 yaw/roll	±4.0 deg	0.55 deg	74.0	0.30 deg	56.5
No. 3 subassembly 1 yaw/roll	±4.0 deg	0.30 deg	56.5	0.45 deg	74.0
No. 4 subassembly 1 pitch	±4.0 deg	0.47 deg	92.0	0.50 deg	62.5
Stage I pitch rate	+2.5 deg/sec -3.0 deg/sec	0.37 deg/sec	0.0	0.92 deg/sec	81.5
Stage I yaw rate	±2.5 deg/sec	0.48 deg/sec	73.0	0.30 deg/sec	76.6
Stage I roll rate	±20 deg/sec	0.50 deg/sec	154.7	1.61 deg/sec	8.6
Stage II pitch rate	±10 deg/sec	0.15 deg/sec	157.5	1.75 deg/sec	170.5
Stage II yaw rate	±10 deg/sec	0.80 deg/sec	156.2	2.10 deg/sec	171.0
Stage II roll rate	±20 deg/sec	1.80 deg/sec	156.2	0.25 deg/sec	157.5

[a] Positive indicates pitch up, yaw right, and roll clockwise, looking in the direction of flight.

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

5.3 SPACECRAFT/GEMINI LAUNCH VEHICLE INTERFACE PERFORMANCE

The spacecraft/Gemini Launch Vehicle interface requirements, as defined in reference 18, were met without exception.

The Electrical and Malfunction Detection System circuitry performed satisfactorily. As experienced on previous launches, the electrical shorting at spacecraft/launch vehicle separation (cable cutting) was present, but no problems resulted.

5.4 GEMINI AGENA TARGET VEHICLE

Performance of the Gemini Agena Target Vehicle (GATV) was satisfactory during the launch and ascent phase, and the GATV attained an orbit of 162.7 by 156.4 nautical miles. However, at approximately 140 seconds after initiation of the primary propulsion system (PPS) ascent firing, a momentary 6-percent decay in the thrust chamber pressure was observed, but the thrust recovered to within one percent of nominal for the remainder of the firing. This anomaly is discussed in detail in section 5.4.2.

In preparation for rendezvous and docking, the GATV was gyrocompassed to a minus 90-degree attitude (engine south) using real-time commands from the Carnarvon tracking station. With the spacecraft at a range of 64 nautical miles from the GATV, a radar anomaly occurred which was similar to that experienced during the Gemini XI mission. The crew reported intermittent radar lock-on and lack of message acceptance pulses (MAP's) when commanding the GATV through the L-band system. Radar lock-on was indicated on the ground, and the GATV was receiving and responding to the commands and generating valid MAP's (see sections 5.1.5 and 5.7).

The first docking occurred at 4:13:30 g.e.t. and was normal. Some difficulty was encountered by the crew during the second docking attempt at 4:49:44 g.e.t. (in darkness), apparently due to a spacecraft misalignment with the GATV docking cone and a low closing velocity, with the result that vehicle rates were imparted to the GATV. A successful second docking was achieved at 5:07:14 g.e.t. (in darkness). The third and final docking sequence was completed satisfactorily at 6:08:03 g.e.t and the two vehicles remained docked until initiation of the tether evaluation at 47 hours 23 minutes g.e.t.

A scheduled docked posigrade PPS firing was canceled as a result of the 6-percent PPS thrust chamber pressure decay noted during the ascent firing. Two GATV secondary propulsion system (SPS) Unit II firings were then scheduled into the flight plan to provide proper vehicle phasing with the solar eclipse and were initiated at 7:05:06 g.e.t. and 15:16:18 g.e.t. The first firing was shut down by the crew, rather than the velocity meter, when the planned firing time was reached. The second firing was shut down by the velocity meter.

During the docked portion of the mission, excessive attitude control gas was used, and the gas was depleted shortly after completion of the tether evaluation. Since the GATV attitude control system (ACS) was functioning normally, the excessive gas usage appears to be the result of considerable control activity required during the EVA periods, in addition to last-minute flight plan changes and accompanying procedural problems.

The tether evaluation was initiated during spacecraft revolution 30 when the GATV attitude control system was utilized to pitch the docked combination into a GATV-engine-down position.

A velocity meter problem was encountered during spacecraft revolution 34 when the velocity meter was loaded for a planned SPS Unit II firing. The telemetry read-out of the velocity meter word indicated all "1's" instead of the word loaded. Further testing of the velocity meter resulted in the same indication and no cause could be determined.

After spacecraft landing and recovery, a PPS maneuver was attempted to evaluate the velocity-meter anomaly and the PPS. Start sequence A was used and the SPS Unit I ullage firing was normal; however, the pilot-operated solenoid valve (POSV) was slow in opening, preventing the main fuel valve from opening. Failure of the main fuel valve to open resulted in a turbine overspeed and subsequent shutdown of the PPS.

Final tracking indicated that the orbit of the GATV was 156.0 by 138.2 nautical miles. No further attempts were made to operate the PPS.

5.4.1 Airframe

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

5.4.1.1 Launch phase.- Temperature measurements on the shroud indicated that the maximum temperature of 239° F was reached at about lift-off (LO) plus 170 seconds. The maximum temperature measured on the Target Docking Adapter (TDA) was 120° F at LO + 150 seconds. The horizon sensor fairing temperature reached a maximum of 496° F at LO + 136 seconds. All of these maximums are similar to those of Gemini X and XI, both in temperature and time. The acceleration at booster engine cutoff (BECO) was 6.3g and the acceleration at sustainer engine cutoff (SECO) was 3.0g.

5.4.1.2 Separation.- The GATV separated from the Target Launch Vehicle (TLV) with an average relative velocity of 43 inches per second, calculated using data from the separation monitor. This value compares favorably with values from previous flights and with the nominal value.

5.4.1.3 Ascent maneuver.- During the ascent maneuver, there were no abnormal vibrations or accelerations with the exception of a short 2.8g spike on both lateral accelerometers at the time of the PPS anomaly (LO + 511.7 seconds). This period included main engine ignition, horizon sensor cover jettison, and shroud separation. All measured temperatures were close to predicted values and to those measured on previous flights. The aft-section temperatures started increasing at separation, with peaks ranging to 255° F for the aft bulkhead temperatures. These peaks occurred at PPS cutoff, as had been noted on previous flights, and then decreased to orbital values.

5.4.1.4 Docking phase.- Docking the spacecraft with the GATV was attempted four times, with three successful dockings performed. During the second docking attempt, probable misalignment of the spacecraft with the GATV permitted only one or two of the three latches to engage and the automatic rigidizing sequence was not initiated. Considerable maneuvering was required to effect disengagement of the spacecraft from the GATV. No damage was sustained by either vehicle. The docking phase is covered in detail in section 5.7.

5.4.1.5 Orbital phase.- During the mission the spacecraft was docked with the GATV for approximately 42 hours 30 minutes. During the docked GATV maneuvers (two SPS firings) and the EVA periods, no problems caused by vibration or noise were noted by the crew or in the telemetry data. During the SPS maneuvers, accelerations averaged 0.025g as measured by the spacecraft accelerometers.

As had been experienced on previous flights, temperatures varied within predicted limits. Temperature sensors on the TDA indicated a temperature range of 12° to 138° F. The highest variation (about 120° F) was again on the top of the TDA. Aft rack temperatures also showed a wider range of variations then Gemini XI, with SPS-module-radiation-shield temperatures varying between minus 9° and plus 223° F. Paint blistering was again noted on the GATV forward rack. No deleterious effects were noted on the GATV airframe during the tether evaluation.

5.4.2 Propulsion

Performance of the SPS was satisfactory for the ascent and orbital operations. Two SPS Unit I and two SPS Unit II firings were accomplished. The PPS ascent firing successfully placed the GATV into orbit; however, an anomaly during the ascent firing indicated a possible turbopump problem and therefore the docked PPS firings planned were not attmpted. Following spacecraft reentry, a PPS start sequence A was commanded over Hawaii but was terminated by the overspeed electronic gate as a result of failure of the main fuel valve to open. The ascent PPS problem is similar in most respects to four previous Agena first-firing anomalies. Complete evaluation and resolution of these previous occurrences has not been possible because of the nature of installed instrumentation and the lack of subsequent inflight indications of a failure. However, the modes of failure previously considered probable were much the same as those discussed in the following paragraphs.

5.4.2.1 Primary propulsion system.- PPS start and shutdown transients (fig. 5.4-1) during the ascent firing appeared to be normal. Steady-state performance of the main engine was also nominal until LO plus 511 seconds. Between LO + 511 seconds and 512 seconds an anomaly occurred (fig. 5.4-2). This anomaly was characterized by a drop in all pump-affected parameters, and on certain of these parameters a slow decay was seen for approximately 300 milliseconds prior to the rapid change. The total drop in pump speed (estimated at 1500 to 2000 rpm) was followed by a normal recovery period to near rated conditions. Only engine chamber pressure indicated a slight change (about a 1-percent drop) from previous operating conditions. During this period, turbine-speed telemetry data became erratic and remained so for the duration of the firing. However, there are ample data which indicate that no overspeed occurred and that, after the brief anomaly, pump speed was very close to normal. The duration of the first firing was close to the desired value and the planned velocity-meter shutdown took place.

Because there was doubt about the nature of the problem, and since adequate data did not exist to fully evaluate the engine problem at that time, a decision was made not to utilize the PPS for docked vehicle maneuvers. After spacecraft reentry, a PPS start was attempted over Hawaii during GATV revolution 63. The attempt (fig. 5.4-3) was aborted because of a failure of the main fuel valve to open, resulting in a pump-overspeed shutdown. Failure of the main fuel valve to open was possibly the result of a problem within the pilot-operated solenoid valve which controls the fuel valve.

All available evidence indicates that the first-firing problem occurred because of an excessive load within the turbo-pump assembly and that the restart attempt failure occurred in the pilot-operated solenoid valve. These anomalies have a significant number of possible causes which cannot be clearly resolved on the basis of telemetered data. These possibilities, followed by a discussion of the most probable sequence of events, are as follows:

Possible causes of pump slowdown

(a) Bearing failure caused by:

(1) Excessive drive-shaft hot-gas-seal blowby and overheating or drying of the shaft bearings

(2) Excessive pump ball-bearing side-play resulting from abnormal wear

(3) Bearing contamination due to parts failure, improper cleaning, or hot gas effects on the lubricating oil

(4) Excessive bearing loads due to uneven pump case thermal expansion as a result of its 4-piece construction.

(b) Rotating and static parts contact caused by:

(1) Improper parts dimensional stackup

(2) Thermal distortion or growth of parts or of pump case. (Note: Hot-gas-seal leakage or high bearing load could also have caused thermal distortion.)

Possible causes of POSV failure

(a) Foreign particles lodged between shuttle poppet and seat or between the poppet and its guide pin (see figure 5.4-6)

(b) Propellant residue formation between the shuttle poppet and guide pin due to fuel evaporation after the first firing.

Any of these conditions singularly or in combination could have caused the two flight problems observed. The following paragraphs describe what is believed to be a possible sequence of events, with the assumption that two related inflight failures did occur.

At LO + 511.00 seconds, thermal distortion, slippage of pump parts, or bearing wear caused a slight but increasing contact between rotating and static parts within the fuel pump, resulting in a minor slowing of the pump. This was observed as slight pressure drops in venturi inlet pressures, chamber pressure, and turbine manifold pressure.

At LO + 511.65 seconds, increasing pressure and heating caused a surface material failure in the contact area and gouging or momentary seizing of the parts. This resulted in a more pronounced decrease in pump speed and pressure during a 0.1-second period. During this interval, accelerometer data indicate possible high vibration in the vehicle aft rack area but for a very short period.

Gouging, chipping, or melting allowed relief of the contact area and, in the process, generated significant quantities of contaminant particles. It is also likely that some small pump unbalance was created at this time. This was followed by normal acceleration of the pump back to rated speed and near normal engine operation for the remainder of the firing. Erratic telemetered pump-speed data after the anomaly period are believed to be the result of vibration-induced sensor damage (loosening) or oxidizer gear axial motion and a subsequent noisy or reduced sensor output. The nature of the sensor signal required for telemetry operation is such that a noisy or low-level signal could have created erratic telemetry outputs without causing a trip of the gate (fig. 5.4-4). Proper gate sensing of the actual speed signal was demonstrated during the restart attempt.

The 1-percent drop in chamber pressure seen after the problem may have been due to a transducer shift or to contaminant particles clogging some of the fuel injector orifices.

Following the pump recovery, all pressure indications except chamber pressure appeared normal and a velocity meter shutdown occurred as planned. Engine shutdown transients indicated a normal shutdown and did not reveal excessive pump drag.

During the attempted restart (fig. 5.4-3) in GATV revolution 63, it is possible that the previously generated contaminants had progressed through the fuel valve actuation pressure line (fig. 5.4-5) and entered the POSV (fig. 5.4-6), blocking its closure and preventing operation of the main fuel valve. The telemetry data reflect a proper increase in electrical current, indicating that the solenoid had received actuation current.

Because of the relationship between the size of the orifice which controls flow into the fuel valve/POSV chamber and the POSV shuttle poppet seat, very small particles (<0.002 of an inch) on the shuttle poppet seat can effectively prevent main valve operation by inhibiting the actuation port pressure buildup. This area is protected by a wire screen filter which can readily pass 0.007-inch-diameter particles and slivers having a much greater effective cross section. Flow through the actuation line and this area occurs during pump shutdown after POSV closure and during start-up before POSV actuation. Backflow through the valve occurs during pump filling after propellant isolation valve opening, and the backflow is not filtered.

Failure of the fuel valve to open led to a predictable pump overspeed and electronic gate shutdown 2.23 seconds after the fire signal. (The start transient was normal in all other respects.) An evaluation of pressure data indicates that an actual pump speed of 29,600 to 31,000 rpm may have been attained, the lower value being more likely, as compared with the normal speed of about 24,000 rpm. This is consistent with calculated fuel-valve failure conditions.

There is evidence that the POSV shuttle poppet did close during the shutdown transient, that a momentary pressure buildup occurred in the fuel valve actuation port area, and that the main fuel valve started to open and allowed some fuel to flow into the injector area. With high pump outlet pressures, as existed at the time of the overspeed shutdown, the POSV solenoid poppet can be held open even though its electrical signal has been removed by the shutdown function. This activity could have been caused by dynamics induced by pump deceleration or by a coincidental attainment of sufficient pressure to force dislodging of a trapped particle. Minor reactions in the thrust chamber appear to have occurred after the shutdown sequence, also implying that some fuel flow occurred.

Analysis of the telemetered data is still in progress at the time of publication of this report and any new results will be published at a later date.

5.4.2.2 Secondary propulsion system.- The SPS Unit I was operated for a total of 42.16 seconds and was normal except for minus Y chamber pressure readings, which were approximately 2 psi below those expected for the observed feed pressures. Unit II was operated for a total of 73.89 seconds and was normal except for chamber pressure readings which were also approximately 1 to 2 psi below those expected. The first Unit II operation was for 54.21 seconds. The slight operation beyond specification limits (10 seconds for the plus Y system and 4.21 seconds for the minus Y system due to propellant temperature differences) did not create any apparent detrimental effects.

All operating characteristics of the SPS were nominal except for the slightly low chamber pressures. This did not appreciably affect the limited operation undertaken (two firings on each chamber). Run-to-run repeatability was satisfactory.

5.4.3 Command and Communications System

The Command and Communications System performed normally in all aspects throughout the entire mission. During GATV revolution 64 the command system was effectively disabled, thereby eliminating further exercising of the vehicle, other than monitoring existing status. Data obtained from GATV revolution 75 indicated that the Command and Communications System was still functioning normally within the limitation imposed by the command system being disabled. The following paragraphs briefly summarize the performance of the functional systems.

5.4.3.1 Command system.- The command system satisfactorily accomplished the proper receipt, processing, and execution of all real-time and stored-program commands. All real-time commands were verified by MAP's through the PCM telemetry; however, the spacecraft did not receive RF-link MAP's due to a failure in the spacecraft/TDA RF link (see section 5.1.5).

During this mission a minimum of 1041 commands were processed properly. They were as follows:

Real time	339
Stored program	545
Spacecraft	157
Total	1041

5.4.3.2 Tracking system.- The tracking system functioned normally throughout the entire mission, providing excellent tracking coverage at all stations, including several low-elevation passes at ranges in excess of 1500 nautical miles. The temperatures of both C-band and S-band transponders did not deviate beyond expected levels which were within a few degrees of nominal.

5.4.3.3 PCM telemeter system.- The PCM telemeter system provided all stations with excellent flight data throughout the mission, with synchronization losses occurring only during low-elevation, long-range passes.

Two problem areas were reported via the telemeter system—the turbine speed indication (discussed further in sections 5.4.2 and 5.4.7) and the velocity meter anomalous behavior (discussed further in paragraph 5.4.5.2.5).

The PCM tape recorder operated normally for the entire mission, providing stored data of excellent quality.

5.4.4 Hydraulic and Pneumatic Systems

5.4.4.1 Hydraulic system.- The hydraulic system operated properly during the ascent PPS maneuver. During hydraulic system operation, the pump discharge pressure increased normally from zero to 2836 psia and occasionally went as high as 2900 psia during the maneuver. After the maneuver, the pump discharge pressure decreased to zero within two seconds after engine cutoff. Hydraulic reservoir pressure varied between 52 and 92 psig, which is the normal range.

For the second PPS maneuver the hydraulic pressure increased normally from zero to 2800 psia but then kept increasing until it reached 3280 psia, at which time the turbine pump was cut off. The pump discharge pressure again went to zero within two seconds.

5.4.4.2 Pneumatic system.- The propellant-tank pressurization system functioned normally throughout the mission. Prior to lift-off, the oxidizer and fuel tanks were pressurized to 30.1 and 38.6 psig, respectively, and the helium pressurization tank was charged to 2415 psia. The pyrotechnically operated helium control valve operated properly for pressurization of the propellant tanks. The propellant tank pressures during orbit decreased from 28.9 to 21.9 psia in the oxidizer tank and from 46.2 to 36.5 in the fuel tank, all of which were within the expected values.

5.4.4.3 Attitude control system.- The attitude control system was activated a few seconds after separation of the GATV from the TLV. The system functioned normally throughout the mission. Due to considerable GATV maneuvering, the control gas was depleted after 54 hours g.e.t.

5.4.5 Guidance and Control System

The Guidance and Control System performed satisfactorily throughout the mission. Evaluation of the flight data indicates that the system performed the following required functions:

(a) Performed all inflight switching requirements and programming

(b) Responded properly to all commands (except as noted in paragraph 5.4.5.2.5)

(c) Sensed and maintained vehicle attitude properly

(d) Reacted to attitude errors with control forces of the proper magnitude and polarity

(e) Provided PPS engine cutoff through the velocity meter

(f) Provided shutdown of one SPS maneuver by command.

An anomalous condition occurred in loading the velocity meter system following final spacecraft/GATV separation. Prior to this, all velocity meter functions appeared normal. Guidance and control flight parameters are shown in tables 5.4-I through 5.4-III.

5.4.5.1 Ascent guidance sequence.- All guidance and control parameters appeared normal throughout the ascent portion of the flight. The ascent sequence timer was started by a booster discrete command at LO + 277.58 seconds (nominal is LO + 277.6 seconds). Functions that occurred during the ascent phase are listed in table 5.4-I. Sequence timer performance was normal throughout its period of operation.

The TLV/GATV separation sequence started at LO + 300.3 seconds and was complete with the opening of separation switches S3 and S4 at LO + 302.7 seconds. Rates imparted to the GATV at separation were minus 0.29 deg/sec in yaw and minus 0.21 deg/sec in roll. These are well within the allowable maximum rates of 0.6 deg/sec in all axes.

The programmed pitch-down maneuver following separation occurred at LO + 338.65 seconds at a rate of minus 1.57 deg/sec, compared with a nominal of minus 1.5 deg/sec ±15 percent. Horizon sensors gains were within specification (table 5.4-II).

The PPS insertion firing commenced at LO + 372.705 seconds and lasted for 182.98 seconds. The initial transients in pitch and yaw were greatly reduced from those seen in the flight of the Gemini XI GATV. The maximum gyro deflections were minus 3.7 degrees in pitch and plus 4.2 degrees in yaw. These transients were essentially damped out in six seconds. Roll-axis characteristics were normal and the PPS firing was terminated by a velocity meter cutoff. A summary of this firing and the subsequent SPS firings is shown in table 5.4-III.

The ascent flight data indicate that the hydraulic return pressures increased to 100 psig during the PPS firing (100 psig being the limit of the transducer). This is not an abnormal indication and has been noted on previous flights.

5.4.5.2 Orbit guidance sequence.-

5.4.5.2.1 Docking: Three dockings were completed during the flight. All docking data have been reviewed and the first docking, occurring at 4:13:30 g.e.t., was considered normal when compared with dockings on previous flights. The first docking resulted in disturbances of 1.2 degrees in pitch, 3.0 degrees in yaw, and 1.6 degrees in roll.

Another docking was attempted between the first and second completed dockings. The docking cone did not rigidize and, in separating from the GATV, the spacecraft apparently imparted large forces to the GATV. This resulted in the GATV attitude being such that the horizon sensors were inhibited throughout the second completed docking. Spacecraft attitude data at 5 hours 7 minutes g.e.t. indicate that the GATV was rolled on its side such that one horizon sensor saw all space and one saw all earth, which would inhibit both pitch and roll outputs. The disturbances to which the GATV would have to have been subjected to reach this attitude explain a significant portion of the control gas usage.

5.4.5.2.2 PPS firing: The PPS firing during revolution 63 was attempted with the attitude control gas depleted and no attitude control capability. During the turbine run-up, the following maximum rates were reached:

Roll, deg/sec . . . . +5.56

Pitch, deg/sec . . . . +0.80

Yaw, deg/sec . . . . +1.89

5.4.5.2.3 SPS Unit II firings: Two SPS Unit II firings were performed while docked and the pneumatic system provided adequate control during both SPS firings. The control gas usage was as follows:

SPS Unit II firing	Firing time, sec	Control gas usage, lb		Control gas usage rate, lb/sec
		Predicted	Actual	
1	54.0	3.0	5.0	0.092
2	19.68	1.3	1.3	0.066

5.4.5.2.4 Heading changes: Heading changes, docked and undocked, were made by two methods—programmed rates and gyrocompassing. As noted in the section on control gas usage, the docked heading changes by both methods were beset with procedural problems. This makes the determination of control gas usage for most docked maneuvers very difficult. It appears that gyrocompassing heading changes used 50 to 100 percent more gas than predicted.

5.4.5.2.5 Velocity meter operation: Velocity meter operation appeared normal until spacecraft revolution 35 when a velocity meter word was transmitted for an SPS Unit II firing. The ground station received a "no compare" on the velocity-meter word as the telemetry readout indicated an all "1's" load. Subsequent attempts to load the velocity meter resulted in the same indication. One test conducted was to load a "1" in the least-significant-bit position with all other positions being "0's." The velocity meter was then enabled for 3 hours 20 minutes to determine whether the counter was being loaded properly and whether a velocity meter cutoff would occur as a result of the positive null torque. The cutoff did not occur and the velocity meter word continued to indicate all "1's." At that point it was not possible to isolate the problem to the velocity meter loading circuits, the velocity meter counter, or the telemetry. Several tests were conducted during the remainder of the mission and the data from these tests are still being analyzed at this time. A final test was to be conducted during the GATV revolution 63 PPS firing in which the velocity meter was to be loaded with a small number and enabled just prior to PPS shutdown by stored-program command. This would have determined whether the counter was being loaded properly and would have provided a PPS shutdown signal. The PPS engine was shut down by a turbine overspeed and the test was not completed.

The data indicate that the problem was probably in the velocity meter counter rather than the loading circuits or the telemetry register; however, continued effort will be required to isolate the problem.

Two SPS Unit II firings were performed in the docked configuration. The first was cut off by the crew when it appeared to be too long. Predicted firing time was 51.6 seconds with the firing cut off by the crew after 54.21 seconds. The velocity meter had 1.04 ft/sec remaining in the counter when the firing was shut down manually. The second firing had a predicted duration of 18.0 seconds with an actual of 19.68 seconds. The actual velocity gained as derived from final data is not yet available; therefore, this information will have to be included in a supplemental report. These data may allow a more accurate determination of the location of this problem. At this time three areas of uncertainty exist for an exact determination of the problem: the spacecraft/GATV weight, SPS thrust levels, and the velocity meter.

5.4.5.2.6 Attitude control gas usage: Attitude control gas usage was normal for the ascent phase of the mission, with approximately six pounds used prior to docking.

The control gas consumption was excessive for the docked portion of the mission and the gas was depleted by approximately 55 hours g.e.t. Factors contributing to this excessive usage were procedural problems, docking practice, and spacecraft thruster malfunctions.

The undockings and the attempted docking resulted in high rates in the GATV and resultant high gas usage. The effect of spacecraft thruster malfunctions is impossible to evaluate; however, prior to the tether evaluation, tests run on the spacecraft thrusters resulted in gas usage. Procedural problems appeared primarily during heading changes and were partially the result of inflight changes to the flight plan.

5.4.6 Electrical System

The Electrical System performance was satisfactory. Any unusual or anomalous performance of the Electrical System is discussed in the following paragraphs.

5.4.6.1 Bus potential levels.- The main bus and the pyro bus voltage did not exhibit the normal high peroxide potential or tusk voltage. This change was noted during battery activation tests prior to flight and was caused by one or both of the following:

(a) Long shelf life wherein the plates had more oxidization— all GATV batteries for the Gemini Program were delivered at the same time

(b) Variation in manufacturing processes.

As a result of the change noted during activation, the worst battery from this lot was discharged at a rate of 7 amperes to a minimum of 22 volts. This test battery had a measured capacity of 429 ampere hours and nominal batteries are rated at 430 ampere hours with a maximum of 450, verifying the stated causes. The main bus was supporting 14.5 amperes at 24.3 volts at lift-off and 15.01 amperes at 24.42 volts during GATV revolution 75.

5.4.6.2 Ascent squibs.- Apparently one of the pyrotechnically operated helium-valve squibs failed to clear after firing. The fusistor cleared this circuit 2.5 seconds later without an increase in structure current.

5.4.6.3 Structure current monitors.- The structure current monitor exceeded 3 amperes many times during the mission, with fluctuations as high as 17.63 amperes on GATV revolution 75. During other missions, the structure current monitor has indicated approximately 1.8 amperes. This structure current had no correlation with main bus voltage or current excursions. Therefore, the indicated readings were generated by transients or by a malfunction in the shunt differential instrumentation amplifier. This problem is still under investigation.

5.4.6.4 Inverter temperature.- On previous missions the inverter temperature did not exceed 104° F. During this mission, the inverter reached 201.2° F without a change in output voltage, which indicates that the temperature sensor malfunctioned as discussed in section 5.4.7.

5.4.7 Instrumentation System

The Instrumentation System provided for the monitoring of 156 analog and 22 step-function (tell-tale) parameters. All instrumentation parameters were operative at lift-off and only one parameter (C-21, 400-cps three-phase inverter temperature) failed to provide satisfactory data throughout the flight. The indicated inverter temperature was higher than anticipated during short periods of time after GATV revolution 36. Data indicated that the output of the inverter did not degrade as would be expected with the indicated temperature increase. The failure was therefore isolated to the instrumentation system. Data analysis shows that the temperature sensor was probably not the cause, and that a loose connection in the transducer signal-conditioner circuitry would produce the results evident in the data.

Another minor anomaly was that of a false residual pressure indication or zero shift in measurement B1 (fuel pump inlet pressure). This anomaly has occurred on each of the other flights and is attributed to pressure transients at the opening of the fuel isolation valve.

5.4.8 Range Safety System

Performance of the Range Safety System was satisfactory.

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

5.4.8.2 Tracking system.- The C-band transponder was used by various radars to provide input position data for the instantaneous impact predictor (IIP) computer. System performance was satisfactory.

TABLE 5.4-I.- ASCENT SEQUENCE OF EVENTS

Event	Time from lift-off, sec	
	Planned	Actual
Lift-off	0.0	0.0
Start sequence timer	277.6	277.58
Gyros uncaged and horizon sensor doors jettisoned	299.0	298.0
TLV/GATV separation Retrorockets fired	301.5	300.6
Enable attitude control system	304.0	302.7
Programmed pitch-down maneuver start (-1.5 deg/sec)	338.6	338.65
Programmed pitch-down maneuver stop Geocentric rate on (-3.99 deg/sec) Enable velocity meter	351.6	351.83
Disable pitch and yaw pneumatics PPS thrust initiate	371.6	372.705
PPS thrust cutoff Enable pitch and yaw pneumatics	554.45	555.689
Extend L-band boom antenna	561.6	561.8
Attitude-control-system deadband wide	596.6	596.6
Disable velocity meter Gyrocompassing on, low gain	589.6	589.58
Attitude-control-system gain low Attitude-control-system pressure low	696.6	696.83
Unrigidize TDA	702.6	702.8
Fire horizon-sensor zero-degree position squib Stop sequence timer	703.6	703.6

TABLE 5.4-II.- HORIZON SENSOR TO INERTIAL REFERENCE PACKAGE GAINS

Axis	Very high gain		High gain		Low gain	
	Nominal	Actual	Nominal	Actual	Nominal	Actual
Pitch	3.0 ±0.6	3.1	1.00 ±0.20	0.95	1.00 ±0.20	0.95
Roll	9.0 ±1.8	8.6	1.00 ±0.20	0.91	0.33 ±0.07	0.36
Yaw (gyrocompassing)	0.0	0.0	8.00 ±1.60	8.50	0.67 ±0.13	0.56

NOTE: All gains measured in deg/min/deg horizon sensor.

TABLE 5.4-III.- GUIDANCE AND CONTROL PERFORMANCE DURING PPS AND SPS UNIT II THRUSTING

Maneuver	Heading error, deg				Velocity-meter load, ft/sec	Velocity-meter cutoff	Velocity tail-off, ft/sec	Total velocity gains, ft/sec		
	Pitch		Yaw							
	Predicted	Actual	Predicted	Actual				Desired	Indicated	Computed
PPS ascent	+0.31	-0.89	0.0	+1.1	8235.2	LO + 555.7 sec	12.5	8248.2	8247.7	--
SPS no. 1 (docked)		+0.2		+0.2	43.0	Manual[a]	0.0	43.0	41.9	44.7
SPS no. 2 (docked)		+0.1		+0.2	15.0	15:16:38.4 g.e.t.	0.0	15.0	15.0	16.3

[a]Cut off by crew before velocity-meter zero coincidence.

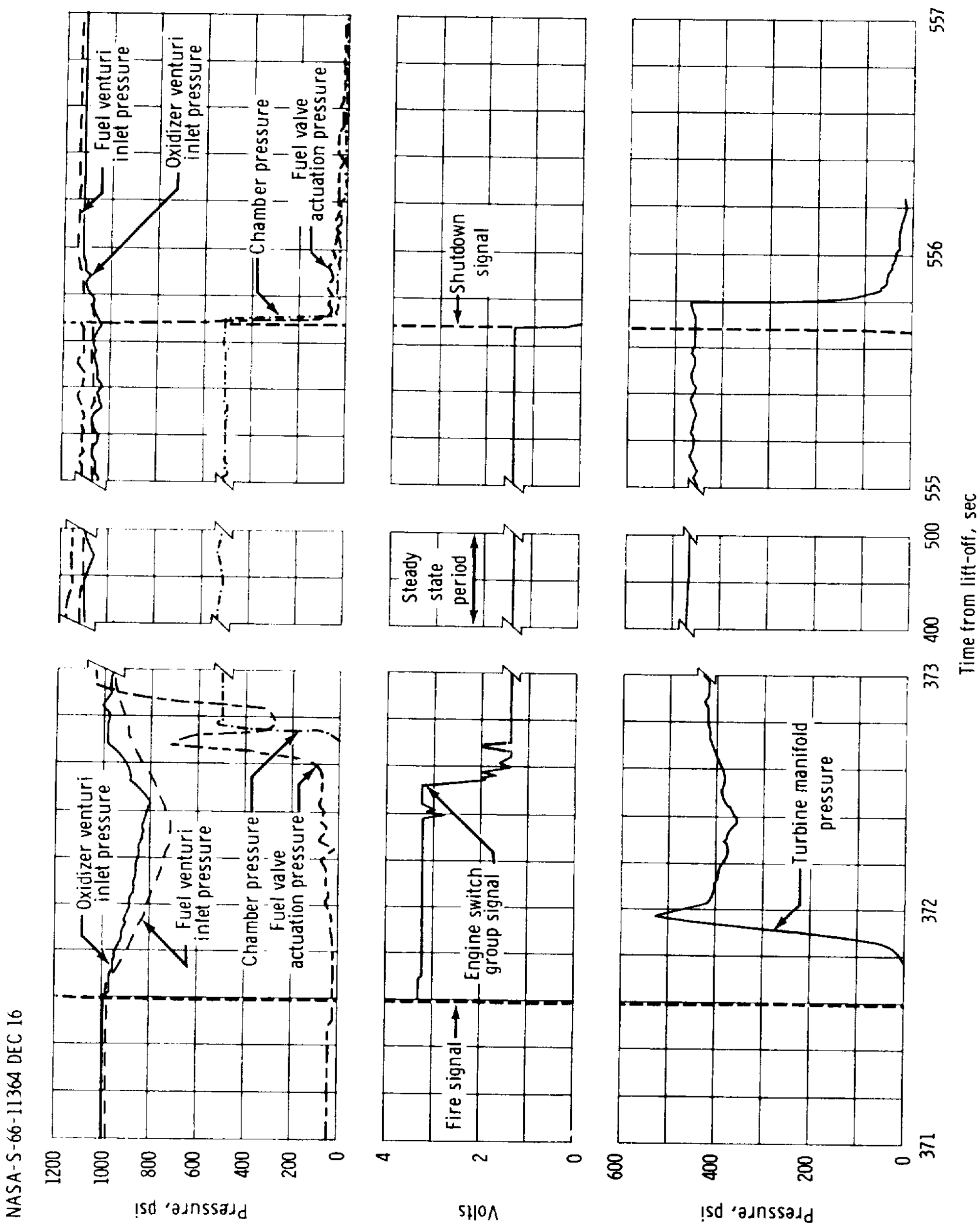


Figure 5.4-1. - PPS ascent firing.

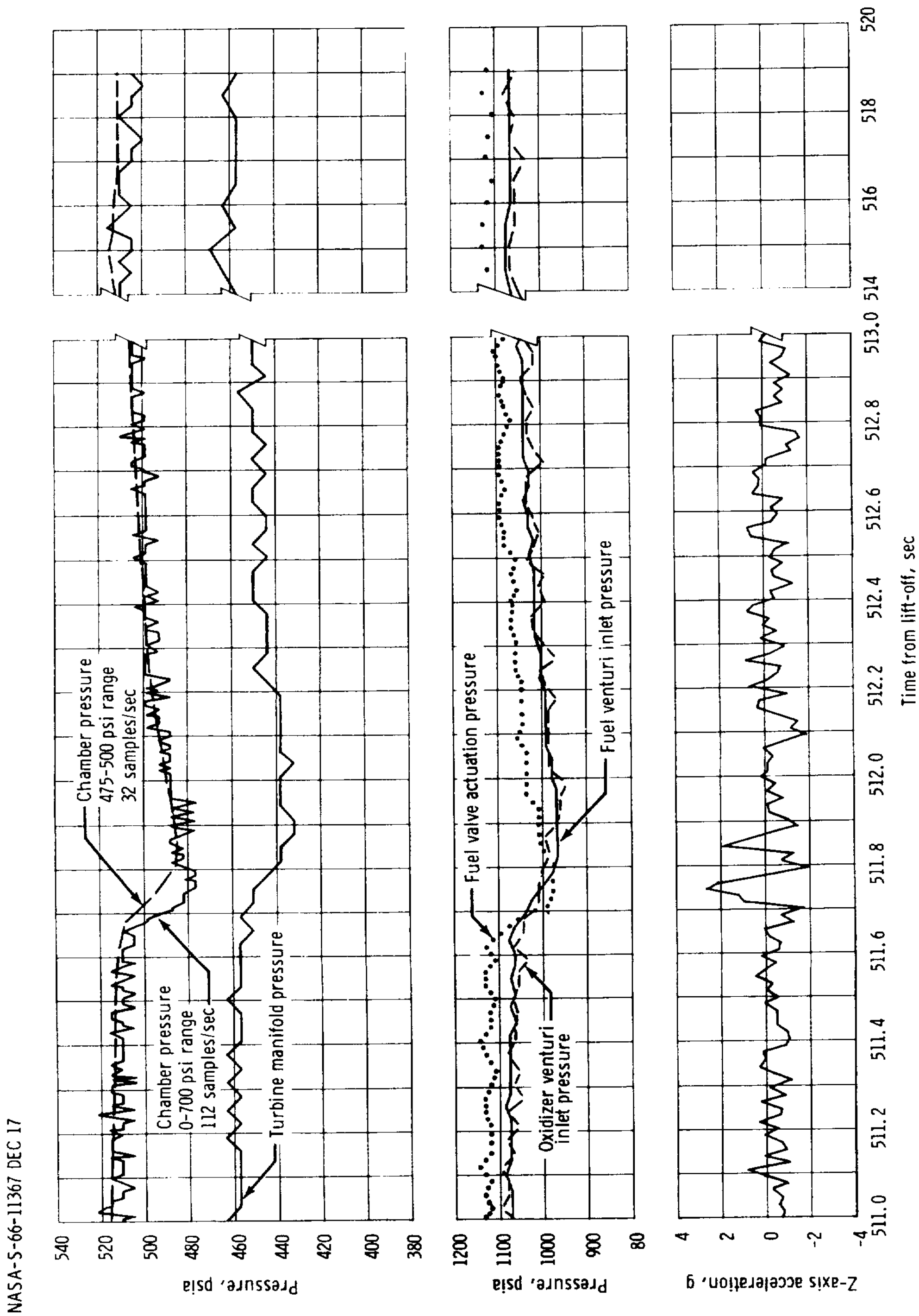


Figure 5.4-2. - PPS transients during ascent anomaly.

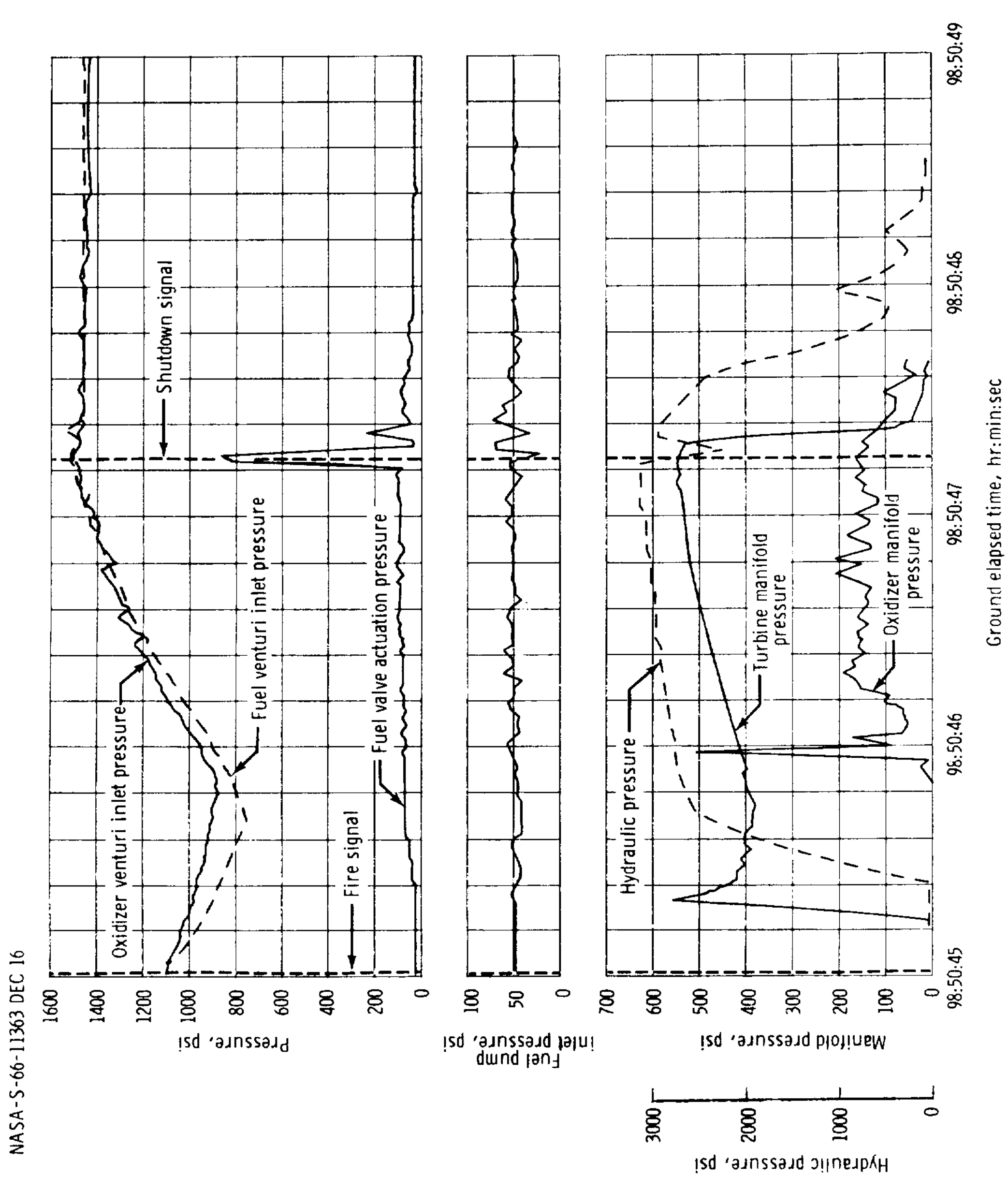


Figure 5.4-3. - Pressure transients during PPS restart attempt.

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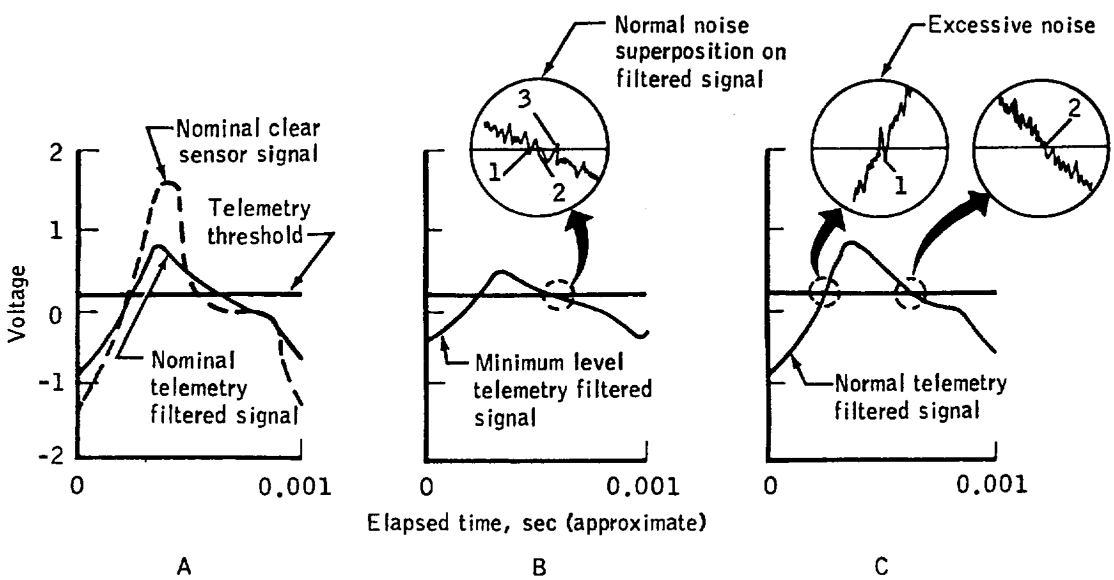

A - Nominal conditions showing sensor output and signal to telemetry counter.

B - Minimum signal to counter due to increased sensor-to-gear air gap. Circle shows how noise can trigger telemetry.

C - Same as B, but based on nominal signal to counter. This requires a higher noise level to cause a false count.

Note:

Operation of the turbine speed telemetry system is such that once the input signal has crossed above the threshold any drop below the threshold value will register as a speed pulse. The above circles show how an excessive noise-to-signal value can trigger false counts. (Numbers in circles refer to counts registered.)

Figure 5.4-4. - Engine turbine speed to telemetry counter signals.

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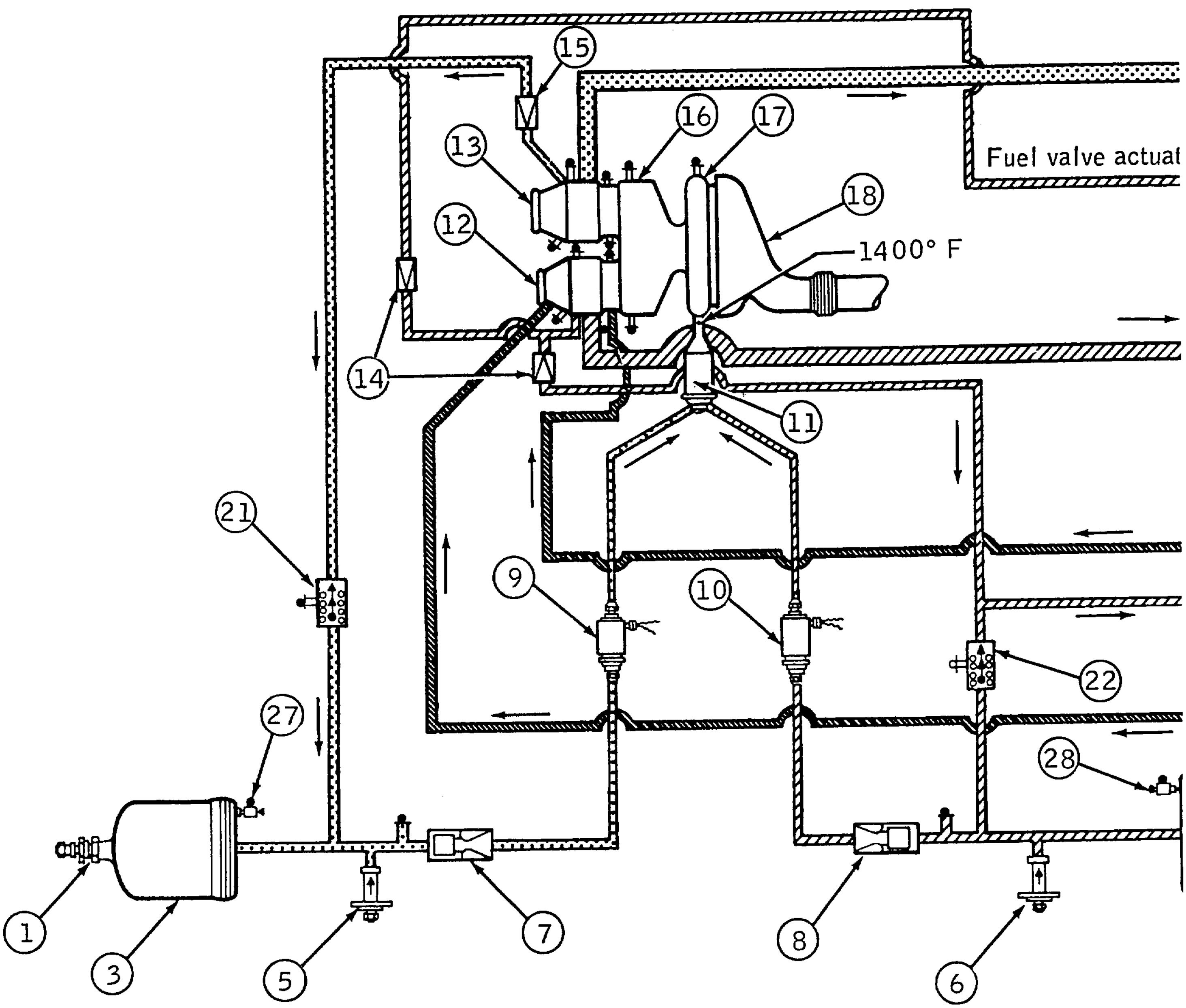
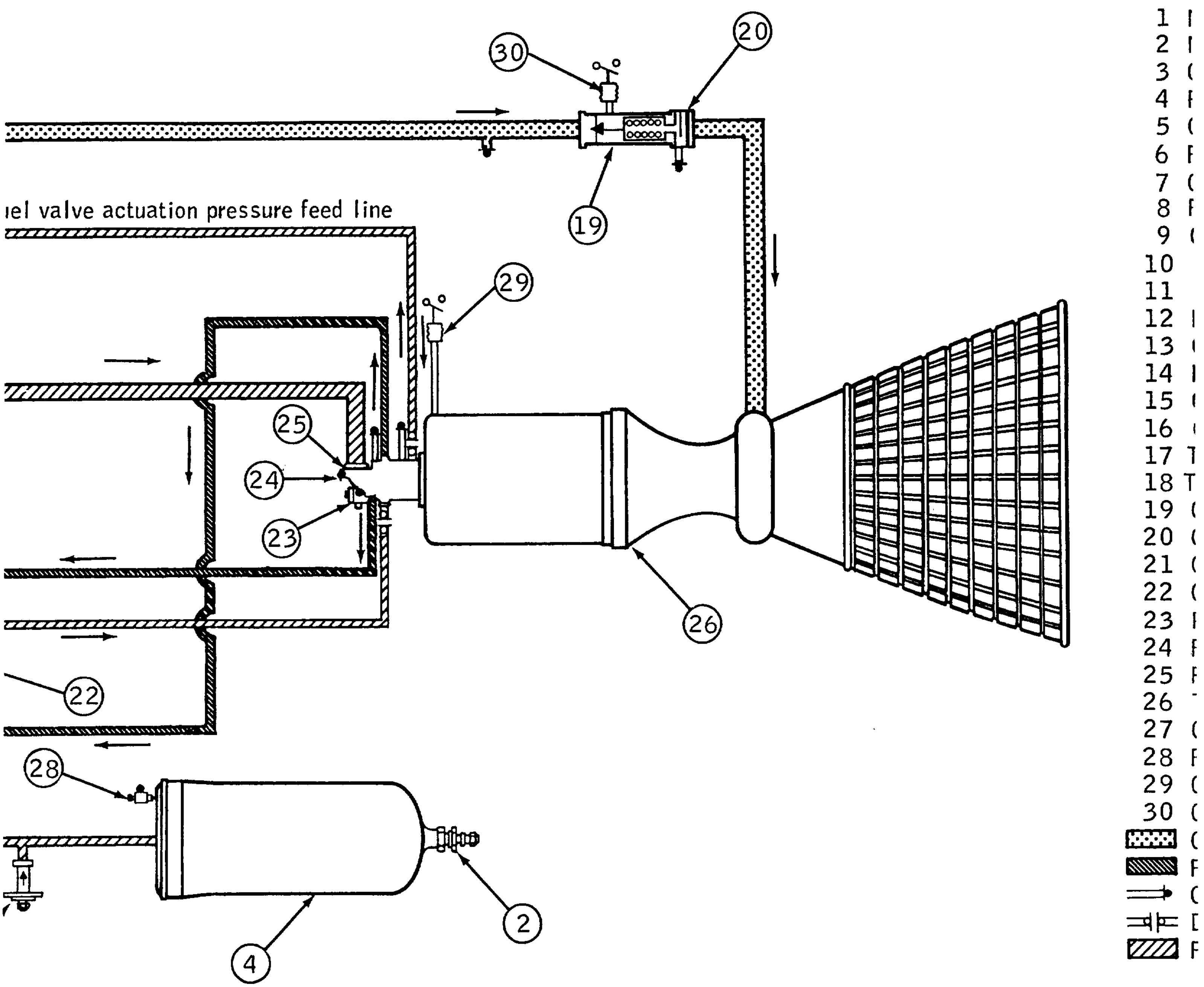
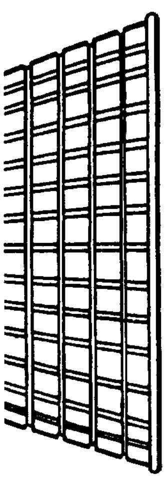


Figure 5.4-5. - Engine propellant flow diagram.

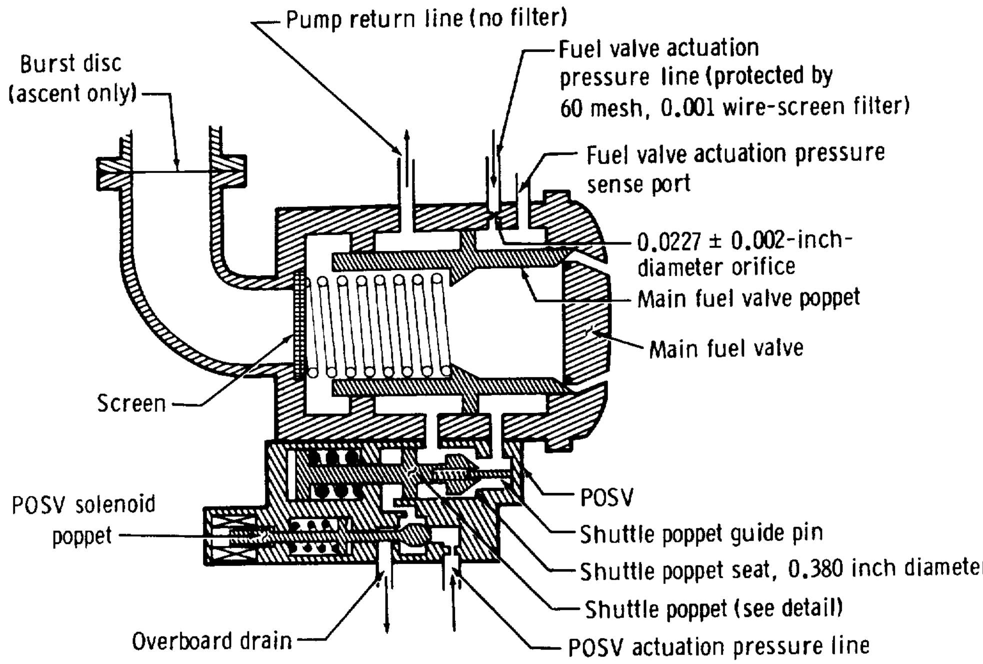

el valve actuation pressure feed line
30
20
19
29
25
24
23
26
22
28
2
4



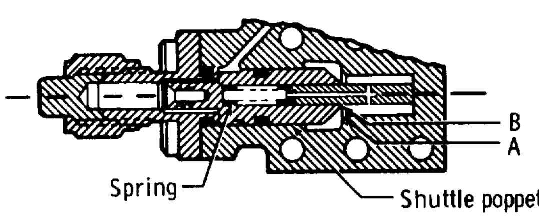
1 Nitrogen fill valve - oxidizer
2 Nitrogen fill valve - fuel
3 Oxidizer start tank
4 Fuel start tank
5 Oxidizer fill and drain valve
6 Fuel fill and drain valve
7 Oxidizer venturi
8 Fuel venturi
9 Gas generator oxidizer solenoid valve
10 Gas generator fuel solenoid valve
11 Gas generator
12 Fuel pump
13 Oxidizer pump
14 Fuel filters
15 Oxidizer filter
16 Gear case
17 Turbine manifold
18 Turbine exhaust duct
19 Oxidizer valve
20 Oxidizer frangible disc (rupture-180 psi)
21 Check valve and bleed port (oxidizer)
22 Check valve and bleed port (fuel)
23 Pilot-operated solenoid valve
24 Fuel valve
25 Fuel frangible disc (rupture-525 psi)
26 Thrust chamber
27 Oxidizer bleed valve
28 Fuel bleed valve
29 Oxidizer manifold pressure switch
30 Oxidizer feed pressure switch

Oxidizer
Fuel return
Orifice and filter
Drain, bleed, or test point
Fuel

NASA-S-66-11266 DEC 6



Main fuel valve/POSV schematic



Areas where particles can prevent valve operation:
A. Between poppet and seat
B. Between poppet and guide pin

POSV shuttle poppet detail

Figure 5.4-6. - Engine main fuel valve and pilot-operated solenoid valve (POSV).

5.5 TARGET LAUNCH VEHICLE PERFORMANCE

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

The Gemini Atlas-Agena Target Vehicle (GAATV) was launched from Complex 14, Air Force Eastern Test Range, at 19:07:58.688 G.m.t. on November 11, 1966. There were no holds or difficulties encountered during the countdown which were attributed to the GAATV.

All of the discrete times in this section, unless otherwise noted, are referenced to 2-inch motion of the TLV as zero time.

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 peak-to-peak amplitude of 0.19g at lift-off (LO) + 1.2 seconds. This oscillation is normally excited during release of the launcher hold-down arms.

Telemetered axial acceleration data indicated the following peak accelerations:

Reference	Axial accelerations, g	
	Predicted	Actual
Booster engine cutoff (BECO)	6.27	6.31
Sustainer engine cutoff (SECO)	3.05	3.00

Booster section jettison at LO + 134.243 seconds and GATV separation at LO + 300.275 seconds were normal. TLV gyro and acceleration data indicated normal transients and vehicle disturbances at these times.

Starting at approximately LO + 40 seconds, the measurement of am-

...ure on the jettison-rail ...port in Quadrant IV of the

... a cond... indicative of a cryogenic leak.

... from 8... at lift-off to 49° F at LO + 82 sec-

... temperature change starting at approximately

Booster section jettison at LO + 134.243 seconds and GATV separation at LO + 300.275 seconds were normal. TLV gyro and acceleration data indicated normal transients and vehicle disturbances at these times.

Starting at approximately LO + 40 seconds, the measurement of ambient temperature on the jettison-rail support in Quadrant IV of the engine compartment reflected a condition indicative of a cryogenic leak. The temperature decreased from 82° F at lift-off to 49° F at LO + 82 seconds, with a more rapid temperature change starting at approximately LO + 40 seconds; after LO + 82 seconds, the temperature decayed at a faster rate (43° F/sec) and reached the lower instrumentation band limit of minus 50° F at LO + 106.5 seconds. The temperature increased gradually after booster-section jettison and reached 28° F at SECO (LO + 279.941 seconds). This was the fifth TLV of the seven launched that recorded evidence of cryogenic leakage.

Slightly decreasing levels were also indicated on three other thrust-section temperature parameters; however, there were no indications of cryogenic leakage reflected by these measurements.

The maximum boost-phase temperature, recorded at BECO, was 140° F in the area of the sustainer fuel pump. Ambient pressure and temperature conditions within the interstage adapter were satisfactory. The pressure exhibited a normal exponential decay during the flight. The ambient temperature increased from minus 12° F at lift-off to plus 64° F at TLV/GATV separation.

5.5.2 Propulsion System

5.5.2.1 Propulsion System.- The engine system, utilizing MA-5 booster, sustainer, and vernier components, was satisfactory in performance and operational characteristics. A comparison of actual computed thrust with the predicted thrust levels is shown in the following table:

Engine	Condition	Thrust, lb			
		Lift-off	BECO	SECO	VECO[a]
Booster	Predicted	330 236	379 890	NA	NA
	Actual	328 990	378 455	NA	NA
Sustainer	Predicted	56 940	80 445	79 675	NA
	Actual	57 161	80 785	79 574	NA
Vernier	Predicted	1 151	1 407	1 149	1 155
	Actual	1 188	1 457	1 217	1 063

[a]Vernier engine cutoff.
NA - Not applicable.

The engines started at LO minus 2.78 seconds, and ignition, thrust rise, and thrust levels were normal prior to launch. The booster, sustainer, and vernier engines were cut off by guidance system commands, and the shutdown characteristics were as expected. The vernier system transitioned to tank-fed operation satisfactorily. A summary of the relay activations and start-of-thrust-decay times for all engines is shown in the following table:

Event	Engine relay box activation, LO + sec	Start of thrust decay, LO + sec
BECO	131.263	131.332
SECO	279.880	279.941
VECO	297.932	298.041

As noted in section 5.5.1, engine compartment ambient temperature data indicated a low temperature environment in Quadrant IV of the thrust section. Engine system data did not indicate any operating condition that might isolate the source of the leak; however, the measurement of sustainer fuel-pump discharge pressure exhibited characteristics of a frozen sensing line to the transducer. The indicated discharge pressure began a decay at LO + 91 seconds, dropping from 915 psia to 120 psia by LO + 140 seconds. The data indicated that the pressure remained below 120 psia after that time. This reduced pressure as seen by the transducer was the result of a blocked instrumentation sense line and did not reflect the true operating pressure. The effect of another frozen instrumentation sensing line was reflected in the hydraulic system data (see section 5.5.4).

Because of a prior history of cryogenic leakage, several design changes and precautionary measures were accomplished on the Gemini X and XI TLV's (SLV-3 5305 and 5306) and were also effective on this TLV.

5.5.2.2 Propellant utilization.- The propellant utilization system operated satisfactorily. The system sensed levels in the liquid-oxygen and fuel tanks at six discrete points during flight and commanded the propellant utilization valve so as to end the flight with a nearly optimum ratio of propellants remaining.

Propellant residuals at SECO were calculated by use of the uncover 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 propellant utilization valve angle changes 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	1037	599	5.47	159
Actual	902	723	4.87	320

5.5.2.3 Propellant loading.- The normal propellant loading procedure was used for this vehicle. Fuel was tanked to 13 gallons over the 100-percent-probe level on November 8, 1966. Liquid-oxygen was tanked during the countdown to the 100-percent-probe level and maintained there until the vent system was closed. Total fuel and liquid-oxygen weights at ignition were 77 139 pounds and 173 541 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 flight, and the proper sequence of events was initiated by the autopilot programmer.

Higher-than-usual roll transients occurred at lift-off but were rapidly damped following autopilot activation at TLV 42-inch motion. The lift-off roll transient reached 1.6 degrees in the clockwise direction at a peak rate of 5.6 deg/sec recorded at LO + 0.75 of a second. Vehicle first-mode bending, excited at lift-off, was predominant in pitch until LO + 3.0 seconds. Maximum oscillations at a frequency of 2.4 cps reached 0.8 deg/sec peak-to-peak. Second-mode bending was excited by the 5-cps lift-off longitudinal oscillations. Maximum oscillations in yaw at a frequency of 5.0 cps reached 0.8 deg/sec peak-to-peak but were damped by LO + 28 seconds. Very low second-mode bending was evident in pitch beginning at LO + 40 seconds and was intermittent until BECO. Maximum oscillations reached 0.4 deg/sec peak-to-peak. Third-mode bending at a very low amplitude was indicated in yaw beginning at LO + 27 seconds and was intermittent until LO + 110 seconds. Maximum oscillations reached 0.4 deg/sec peak-to-peak.

Gyro data provided indications that the roll and pitch 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 + 83 seconds, with an average deflection of 0.6 of a degree.

The programmer enabled guidance steering at LO + 80 seconds; however, no booster phase steering was required. Rigid-body oscillations began at approximately LO + 65 seconds but were completely damped by LO + 86 seconds. Negligible amplitudes were indicated.

The guidance-initiated staging discrete signal was indicated at the programmer input at LO + 131.119 seconds, and the resultant switching sequence was successfully executed. Vehicle transients associated with

BECO and booster-section jettison were not excessive and were damped by the autopilot system. The vehicle first-mode bending which occurs between BECO and booster-section jettison was evident in the pitch and yaw planes. These oscillations, comparable in both frequency and magnitude to those obtained on the previous TLV's, were damped by LO + 135.243 seconds. Rigid-body oscillations at a frequency of approximately 0.25 cps in pitch and yaw were excited by booster jettison but did not exceed 0.6 deg/sec peak-to-peak in pitch and 1.8 deg/sec peak-to-peak in yaw. These oscillations were slightly reinforced by guidance steering commands at LO + 155 seconds. The oscillations were damped to negligible values in yaw by LO + 220 seconds, although they continued intermittently at low amplitude in pitch until SECO. There was no evidence of propellant slosh or vehicle bending during the sustainer phase.

Proper system response was exhibited to all sustainer steering commands, including a small spurious booster steering command from LO + 120.4 seconds to LO + 122.1 seconds. The TLV response to this spurious command, however, was negligible (see section 5.5.5).

The SECO signal was received by the programmer at LO + 279.880 seconds. Vernier phase steering consisted of a very small pitch-up command and a slight yaw-right command. TLV rate and displacement gyro signals indicated a high degree of vehicle stability throughout the vernier phase. The VECO signal was received at LO + 297.928 seconds. GATV separation occurred at LO + 300.275 seconds, and a normal TLV retrorocket firing sequence followed.

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 liquid-oxygen and the fuel ullage pressures in the main tanks during the booster phase of flight, and the control system provided adequate pressurization for sustainer and vernier propulsion system control.

The liquid-oxygen and the fuel ullage pressures were 29.3 and 66.3 psig at lift-off, and 29.5 and 66.0 psig at BECO, respectively. The differential pressure across the intermediate bulkhead (fuel tank ullage pressure minus the sum of liquid-oxygen ullage and head pressures) was positive throughout flight. The minimum differential pressure of 8.1 psid across the bulkhead was recorded at LO + 3.8 seconds.

During the boost phase, 85.4 pounds of the 152.4 pounds of helium aboard the vehicle were used for pressurization of the propellant tanks.

5.5.4.2 Hydraulic System.- The booster and sustainer/vernier hydraulic subsystems supplied adequate pressure to support the demands of user systems throughout the countdown and flight.

Booster and sustainer hydraulic evacuations were successfully accomplished at LO minus 31.4 and LO minus 31.8 seconds, respectively. Normal hydraulic pressure transients were indicated at engine start, followed by stabilization of system pressure at 3215 psi in the booster subsystem and 3115 psi in the sustainer subsystem. The pressures in both systems were satisfactorily maintained until BECO and SECO, respectively. After SECO and cessation of sustainer hydraulic pump operation, hydraulic pressure was supplied to the vernier subsystem by the dual vernier-solo accumulators for 65.0 seconds, before the pistons bottomed out at 840 psia.

Data from the sustainer hydraulic pump indicated that, at LO plus 190 seconds, the discharge pressure from the pump began to decay from 3010 psia and that the pressure reached a minimum of 490 psia by LO plus 268 seconds. The data then indicated a gradual increase in pressure, which reached a maximum of 1260 psia and then gradually decayed to reservoir pressure by LO + 330 seconds. These indications are not valid, as evidenced by satisfactory system operation shown in other hydraulic data and data from user systems. Freezing of the hydraulic fluid within the pressure transducer sensing line is suspected as the cause for these invalid pressure variations. An indication of a cryogenic leak in the general area is also reflected by invalid pressure readings for the sustainer fuel-pump discharge and by low temperature readings in the Quadrant IV engine compartment during the booster phase of flight.

5.5.5 Guidance System

The TLV was guided by the MOD III-G Radio Guidance System, which operated satisfactorily throughout the countdown and flight. The five planned discrete commands and the required steering commands were received and properly decoded by the TLV equipment.

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, were properly accomplished. The preset roll and pitch programs of the Flight Control System successfully guided the vehicle into the planned trajectory (see section 5.5.3).

5.5.5.2 Radio Guidance.-

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 + 57.9 seconds and LO + 62.1 seconds, respectively. Track lock-on was intermittent between LO + 118 seconds and LO + 123 seconds, when antenna look angles were unfavorable. As a result, spurious pitch and yaw steering commands were evident during this period. Because booster steering was enabled at this time, the spurious commands were acted upon by the Flight Control System. These commands, however, were minor, reaching maximum values of less than four percent, and had a negligible effect on the vehicle attitude. Spurious steering commands can be expected during periods of intermittent track lock-on and have been noted on many earlier Atlas flights. Following the period of intermittent track lock-on and the expected dropout during the BECO/staging sequence, both rate and track lock-on were satisfactorily maintained until well beyond GATV/TLV separation, when tracking was intentionally terminated.

Booster steering, implemented to correct open-loop dispersions, was enabled by the TLV Flight Control System at LO + 80 seconds, as planned. No corrections were required and, therefore, no steering commands were generated. BECO, as indicated at the autopilot programmer input, occurred at LO + 131.119 seconds. The errors at BECO were 37 ft/sec low in velocity and 2801 feet low in altitude (see table 4.3-V).

5.5.5.2.2 Sustainer steering: Sustainer steering was initiated at LO + 146.1 seconds, with initial peak commands of 35 percent pitch-up and 80 percent yaw-right. The commands were reduced to below 10 percent by LO + 150 seconds and remained below that level for the remainder of the sustainer phase.

5.5.5.2.3 Vernier steering: Vernier steering was initiated at LO + 280.1 seconds and consisted of approximately 0.3 of a degree pitch-up and 0.2 of a degree yaw-right commands. VECO, as indicated at the autopilot programmer input, occurred at LO + 297.928 seconds.

5.5.5.2.4 VECO conditions: The VECO conditions were very close to the planned values. The space-fixed velocity was less than 1 ft/sec low, the vertical velocity was approximately 1 ft/sec low, and the lateral velocity was about 1 ft/sec left.

The following table is a comparison of the filtered inflight actual insertion values with the filtered inflight desired values.

VECO conditions	Filtered inflight	
	Desired	Actual
Time from lift-off, sec	299.03	297.93
Space-fixed velocity, ft/sec	17 571	17 571
Vertical velocity, ft/sec	2 846	2 845
Lateral velocity, ft/sec	0	-1

5.5.6 Electrical System

Operation of the Electrical System was satisfactory during the countdown and throughout flight. All electrical parameters were at normal levels and remained within tolerance.

A low-level ripple voltage at a frequency of 12 cps was apparent between LO + 383.9 and LO + 444.7 seconds. The maximum amplitude of the ripple was 0.3 of a volt. The same condition has occurred on previous vehicles and has been determined to be a non-detrimental operating characteristic of the inverter.

At LO + 393.1 seconds, a step increase of plus 0.15 Vdc was evidenced. This increase was reflected in the ac system as a drop of minus 0.1 Vac. The system remained stable at the new levels through the end of recorded data (LO + 570 seconds). The dc and ac fluctuations were well within Electrical System performance specifications and had no detrimental effect on the performance of vehicle systems.

5.5.7 Instrumentation System

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

Measurement P330P (sustainer fuel-pump discharge pressure) indicated a slight intermittently open condition from lift-off to LO + 12 seconds. The condition is attributed to lifting of the transducer wiper arm.

Measurements P330P and H130P (sustainer hydraulic pump discharge pressure) began to exhibit data characteristic of frozen transducers or sensing lines at LO + 91 seconds and LO + 190 seconds, respectively. The frozen sensing lines have been attributed to a cryogenic leak in the thrust section. This condition is further discussed in sections 5.5.1, 5.5.2, and 5.5.4.

5.5.7.2 Landline.- The landline instrumentation system provided a total of 44 analog and 56 discrete vehicle measurements. Two analog measurements failed—both the sustainer turbine inlet temperature measurement and the B2 turbine inlet temperature measurement opened during the start sequence.

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 ground-based transmitter was turned off at LO + 312.1 seconds.

The RF signal strength, measured at command receiver no. 1, indicated that sufficient signal margins were available for proper operation of the RF command link at all times during the flight.

5.6 GATV/TLV INTERFACE PERFORMANCE

The Gemini Agena Target Vehicle (GATV)/Target Launch Vehicle (TLV) interface was satisfactory during the ascent and separation phase. Accelerometer data indicated a normal separation sequence between the GATV and TLV. The pressure and temperature conditions within the interstage adapter were satisfactory.

5.7 GEMINI SPACECRAFT/GATV INTERFACE PERFORMANCE

The performance of the spacecraft/Target Docking Adapter (TDA)/Gemini Agena Target Vehicle (GATV) interface was satisfactory throughout the flight with the exception of the L-band system malfunction which is discussed in section 5.1.5. All other systems functioned within the specification requirements of reference 19. The performance of the electrical, mechanical, and command system interface was derived from instrumentation of the various systems, crew observations, and onboard cameras.

The GATV status display panel and the acquisition and approach lights functioned normally throughout the flight. Aerodynamic shroud jettison at 383 seconds after lift-off was normal. The TDA skin temperatures are discussed in section 5.4.1.

The GATV was initially acquired by L-band radar at a range of approximately 235 miles. Visual acquisition of the GATV acquisition lights, using the sextant telescope, occurred at a range of about 85 miles.

Three dockings were accomplished, the first by the command pilot and the other two by the pilot. Following the first docking and undocking in daylight, a night attempt at docking by the command pilot resulted in incomplete docking. This is attributed to a low closing velocity with slight vehicle misalignment which resulted in engagement of only the lower TDA latch. Failure to engage all latches thus precluded completion of the automatic rigidizing sequence. Attempts to complete the engagement by forward thrusting were not successful and disengagement from the TDA was accomplished by translating up, followed by a right and left translation. This period of partial engagement was 39 seconds. After disengagement, a second night docking attempt, with greater closing velocity, was successful. The subsequent undocking, and the final docking and undocking in daylight were successful.

6.0 MISSION SUPPORT PERFORMANCE

6.1 FLIGHT CONTROL

The Gemini XII 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 analyses and evaluations in other sections of the 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 Joint Combined Systems Test on October 10, 1966; the Final Systems Test on October 18, 1966; the Simultaneous Launch Demonstration on November 1, 1966; the Simulated Flight on November 2, 1966; the Precount on November 9, 1966; the final Midcount on November 10, 1966; and the Terminal Count on November 11, 1966.

In addition to the normal in-house simulations, flight controller training, confidence testing, network simulations, and crew launch and reentry simulations, supplemental targeting tests with the launch guidance complex were conducted by the Flight Dynamics Officers. These tests included the manual setting of octal constants for the contingency procedures.

The fuel cells were activated after Midcount with no reported anomalies and were placed on a 1-ampere dummy load per stack until T minus 43 minutes, when they were brought on the line to support the spacecraft. Main bus voltage was 26 volts at 40 amperes.

6.1.1.2 Documentation.- Documentation for Gemini XII was the best for any mission to date. Only minor changes were required after remote-site flight controller deployment.

6.1.1.3 MCC/Network flight control operations.- The remote-site flight controller teams began deployment to the remote sites on October 25, 1966, and the Manned Space Flight Network (MSFN) went on mission status on October 29, 1966. The command and telemetry data-flow tests were concluded successfully, after which all sites were ready to support the launch.

6.1.2 Launch Operations

6.1.2.1 Gemini Atlas-Agena Target Vehicle countdown.- The first countdown of the Gemini Atlas-Agena Target Vehicle (GAATV) was canceled on November 8, 1966, because of a suspected Gemini Launch Vehicle (GLV) secondary autopilot problem. The second countdown was canceled during the Midcount of the Gemini Space Vehicle, November 9, 1966, because of a problem in the replacement GLV secondary autopilot system. The third Midcount of the GLV was picked up on November 10, 1966, and the final GAATV countdown began on November 11, 1966. The final launch countdown was nominal and had no unscheduled holds.

6.1.2.2 GAATV powered flight.- The Gemini Atlas-Agena Target Vehicle lift-off occurred at 19:07:58.688 G.m.t. on November 11, 1966. The GAATV powered flight was very near nominal. Ground track and crossrange plotboard traces were normal and only a slight depression in maximum flight-path angle was noted. At the start of the Gemini Agena Target Vehicle (GATV) primary propulsion system (PPS) insertion firing, there was a small negative pitch of 3.8 degrees and a positive yaw of 4.5 degrees. After five seconds, this transient had been corrected to +0.4 of a degree in yaw and +0.2 of a degree in pitch.

During the GATV insertion firing, the PPS exhibited an anomaly at approximately 8 minutes 31 seconds after GAATV lift-off. This anomaly occurred 2 minutes 20 seconds into the insertion firing, and consisted of a 10-percent decrease in thrust chamber pressure lasting less than a second, and an indicated change in turbo-pump turbine speed which continued until PPS cutoff (see section 5.4.2). (During the ascent firing, the overspeed shutdown circuitry is disabled by the D-timer to preclude PPS shutdown during this critical phase of the mission.) Subsequent investigation of available data revealed approximately 10-percent decrease in turbo-pump oxidizer and fuel outlet pressures at the time of the anomaly. However, inlet pressures remained constant throughout the firing. Based on this information, an internal malfunction of the turbo-pump was suspected. Five pounds of attitude control gas were used for orbital insertion, leaving 141 pounds aboard the vehicle.

The GATV was inserted into orbit at an inclination angle of 28.85 degrees. The insertion conditions, as indicated by the high-speed tracking data from the Impact Predictor (IP) 3600 (raw) and Bermuda, are shown in the following table:

Source	Inertial velocity, ft/sec	Inertial flight-path angle, deg	Altitude, n. mi.
IP (raw)	25 359.0	+0.05	161.0
Bermuda	25 366.0	+0.05	162.0
Best estimate	25 364.9	+0.04	161.5

6.1.2.3 Period between GAATV lift-off and Gemini Space Vehicle lift-off.- The high-speed solution from Bermuda was transferred to the orbit phase and predicted a GATV orbit of 159.0 by 164.6 nautical miles. Low-speed C-band radar data from Bermuda and Antigua predicted 159.0 by 163.6 nautical miles. The Canary Islands correction showed 158.4 by 163.8 nautical miles with a recommended Gemini Space Vehicle lift-off time of 20:46:33 G.m.t. and a biased launch azimuth of 100.6 degrees. Tracking data from Carnarvon and Woomera constituted the final GATV ephemeris update and predicted the following:

M=3 targeting

Recommended lift-off time, G.m.t. 20:46:33

Launch azimuth, deg 100.6

M=4 latest targeting

Lift-off time, G.m.t. 20:48:56

Launch azimuth, deg 101.8

The phase adjust maneuver updated to the crew prior to launch was:

Ground elapsed time (g.e.t.) of maneuver, min:sec 49:45

ΔV, ft/sec 60.2

Thrusters . . . . Aft-firing

6.1.2.4 Gemini Launch Vehicle (GLV) countdown.- The Mission Control Center-Houston (MCC-H) began final countdown support at T minus 615 minutes on November 11, 1966. At T minus 90 minutes, the Environmental Control System (ECS) functions were complete. The suit decay rates were checked and were 0.05 psi in 30 seconds for both crewmembers. At T minus 20 minutes, GE/Burroughs reported a successful Agena Ephemeris Data (AED) request and a verification. The launch window for a rendezvous in the third spacecraft revolution (M=3) was computed to be 40 seconds in duration; however, the usable window within MCC-H backup targeting capability was 30 seconds. At T minus 15 minutes, the MCC-H targeting parameters were transmitted to the spacecraft via the Master Digital Command System (MDCS), and these parameters were then verified by the Blockhouse Computer Operator. At T minus 10 minutes, an oxygen heater cycle was initiated with a nominal increase of 12 amperes noted. At T minus 8 minutes, the following roll program information was passed to the crew:

Start roll program:	8 seconds
Ball reading on pad:	78 degrees (94 degrees after roll)
Roll gimbal angle:	101.31 degrees
Launch azimuth:	100.6 degrees
Steering azimuth:	96.5 degrees

At T minus 3 minutes, the proper targeting load was transmitted by GE/Burroughs to the spacecraft and to MCC-H.

6.1.2.5 Gemini Space Vehicle powered flight.- The launch phase was essentially as planned. The recommended lift-off time was 20:46:33.0 G.m.t. and the actual lift-off occurred at 20:46:33.4 G.m.t. IP (smooth) data were selected at lift-off and showed a very low noise level. The maximum flight-path angle was depressed approximately 0.7 of a degree (corresponding to a velocity ratio of 0.095) because of tailwinds. At staging, the flight-path angle was 0.02 of a degree below normal. The altitude was approximately one nautical mile low until 400 nautical miles downrange. The necessary steering to correct this altitude dispersion caused the flight-path angle of Stage II to be 0.02 of a degree high until about a velocity range (V/V_R) of 0.99 was attained. The average cutoff conditions before Insertion Velocity Adjust Routine (IVAR) thrusting are shown in the following table:

Source	Velocity, ft/sec	Flight-path angle, deg
GE/Burroughs	25 700.7	+0.083
IP (smooth)	25 713.7	+0.042
Bermuda	25 715.0	+0.060
IP (raw)	25 673.0	+0.080

The evaluation of the separation and IVAR maneuvers showed that a ΔV of 33.8 ft/sec was applied, as shown in table 6.1-I. The components were a ΔV_X of 31.4 ft/sec, ΔV_Y of 7.0 ft/sec, and ΔV_Z of 9.4 ft/sec. A sample of the accelerometer biases was taken and the Z-axis was found to be in error by 0.015 pulse/sec. The insertion conditions after IVAR thrusting are shown in the following table:

Source	Velocity, ft/sec	Flight-path angle, deg
IP (raw)	25 741.0	+0.1
Bermuda	26 001.0	+0.77
Grand Turk Island	25 739.3	+0.078

Two anomalies were noted during the launch phase. At staging, a telemetry dropout of 10 to 15 seconds was experienced at MCC-H. After staging, the telemetry came back and was solid until nominal loss of signal (LOS). During second stage flight, the pilot reported that the right secondary oxygen pressure had gone off-scale high but settled down to the proper reading after second stage engine cutoff (SECO).

6.1.3 Spacecraft Orbital Flight

The IP (raw) solution was transferred to the orbit phase and predicted an insertion orbit of 87.3 by 151.5 nautical miles. Subsequent low-speed radar updates through Antigua showed the orbit to be 87 by 146 nautical miles. It had been planned to update the phase adjust maneuver through Ascension but, due to an error in the interrange vector and a downrange procedural error, tracking data from Ascension and Pretoria were lost, as well as voice through Ascension. The phase adjust maneuver was updated to the crew through Tananarive during revolution 1 as follows:

Time of maneuver, g.e.t., ...	0:49:40
ΔV, ft/sec	66.6
Pitch, deg	0.0
Yaw, deg	0.0
Thrusters	Aft-firing

This maneuver was completed as scheduled and the crew reported 85 percent propellant remaining in the Orbital Attitude and Maneuver System (OAMS). The ground computation also showed 85 percent remaining. Over Carnarvon during revolution 1, the crew was given a GO to use the onboard-calculated solution for the plane change maneuver since it was within the proper tolerances of the ground-computed solution. The plane change maneuver was performed and the quantities were as follows:

Time of maneuver, g.e.t.	1:14:22
ΔV, ft/sec	8.5
Direction	South

During the first few remote-site passes after launch, the coolant loop temperature tended to be a little warm with the high power loads. Over Hawaii during revolution 1, the crew reported that the spacecraft had a tendency to yaw left, which indicated that the water boiler was still cooling down. Also over Hawaii, an accelerometer bias check showed the Z axis to be in error by only a very small amount, so it was decided not to update the computer accelerometer constants. At California during revolution 1, the crew reported solid radar lock at 235 nautical miles. Over Texas during revolution 1, a propellant cutoff of 16 percent for rendezvous was passed to the crew. It was noted at this time that the Experiment S003 (Frog Egg Growth) package temperature was beginning to rise, and the crew was advised to put on the S003 thermal cover. Updates for the corrective combination maneuver and the coelliptic maneuver were transmitted to the crew. The differential altitude associated with this update was the nominal 10.0 nautical miles and the time between the maneuvers was 35 minutes 2 seconds. The crew reported over Ascension during revolution 2 that they had performed the corrective combination maneuver as planned. The maneuver was planned as follows:

Time of maneuver, g.e.t.	1:47:52
ΔV, ft/sec	7.6
Pitch, deg	+27.3
Yaw, deg	+9.6
ΔV_X, ft/sec	+6.7
ΔV_Y, ft/sec	-3.5
ΔV_Z, ft/sec	-1.1

The crew onboard calculation of the coelliptic maneuver was within tolerence of the ground solution, so the onboard solution was used and the maneuver performed as follows:

Time of maneuver, g.e.t.	2:22:55
$\Delta V_{X_{S/C}}$, ft/sec	+49.5
$\Delta V_{Y_{S/C}}$, ft/sec	-6.4
$\Delta V_{Z_{S/C}}$, ft/sec	-0.2

At Carnarvon during revolution 2, the crew reported they had lost radar lock at a range of 64 nautical miles. The telemetry signal representing transponder output power indicated a change from 3.96 to 3.09 volts from the stateside pass to Carnarvon during revolution 2. Telemetry also showed encoder lock. The radar power circuit breaker was verified to be closed at this time.

At Hawaii during revolution 2, the crew commanded Status Display Panel bright and dim, but they did not receive a message acceptance pulse (MAP). However, the MAP's were received on the ground via GATV telemetry and the command functions were verified. The crew also went to the spiral antenna and reported that the radar lock appeared a little more steady. The Hawaii onboard computer summaries showed that the computer was not being updated with radar information at the correct rate (due to intermittent radar lock-on). At this time, the crew agreed to leave the computer in the catch-up mode rather than switch to the rendezvous mode for the terminal phase initiate (TPI) maneuver because it was believed that, due to intermittent radar data, the TPI solution could not be obtained in the limited time remaining. The ground backup TPI solution was transmitted to the crew as follows:

Time of maneuver, g.e.t.	3:05:51
ΔV, ft/sec	23.1
$\Delta V_{X_{S/C}}$, ft/sec	+22.8
$\Delta V_{Y_{S/C}}$, ft/sec	-2.7
$\Delta V_{Z_{S/C}}$, ft/sec	-3.2
Range, n. mi.	24.7

The TPI maneuver, as computed by the crew, was to be performed at 3:05:48 g.e.t. and consisted of 22 ft/sec forward and 3 ft/sec up. At 3:46:00 g.e.t. over Tananarive during revolution 3, the crew reported that they were station keeping with the GATV. The first docking was completed at 4:13:53 g.e.t., over the Coastal Sentry Quebec (CSQ) during revolution 3, and the crew reported 67 percent propellant remaining. At this time, the cryogenic readout indicated that the oxygen quantity was four pounds below the predicted value. Calculated values for lift-off, based on ampere-hours used prelaunch, indicated that the oxygen quantity at lift-off was 112 pounds, but the CSQ cryogenic readout corresponded to a lift-off quantity of 108 pounds. Further conversation with Cape Kennedy technicians confirmed that more oxygen had been used during prelaunch activities than had been anticipated. This loss was later made up through flight plan changes and oxygen flow selection during the umbilical EVA.

Over Hawaii during revolution 3, the crew performed the first undocking and reported that the GATV tether loop had deployed and looked good. Over the Rose Knot Victor (RKV) during revolution 4, the crew reported that they were having some minor control problems with the GATV and had undocked. Over Tananarive during revolution 4, the crew reported that they had attempted docking and might have upset the GATV in doing so. They were advised to go to GATV flight control mode 2 and let the GATV gyrocompass to the correct heading. This was accomplished and the crew redocked. Spacecraft propellant quantity at this time was 56 percent. Over the CSQ during revolution 4 at 5:44:30 g.e.t., both fuel-cell differential-pressure warning lights came on. These indications extinguished approximately two minutes later. An analog readout indicated that differential pressure was normal; however, the analogs were referenced to the nitrogen side and the bilevels were referenced to water. This indicated a pressure differential across the adapter water tank bladder. No degradation was noted in fuel cell performance when the differential pressure indications were present. At this time, the crew was advised that the high-altitude PPS firing would not be attempted due to the anomaly noted during the insertion firing, and that a phasing maneuver would be made in order to photograph the solar eclipse. At 7:05:06 g.e.t. an eclipse phasing maneuver was performed as follows:

ΔV, ft/sec	43 (retrograde)
Pitch, deg	0
Yaw, deg	180
Thrusters	GATV secondary propulsion system (SPS) Unit II

The orbit following this maneuver was 155.3 by 139.6 nautical miles. Over Hawaii during revolution 5, the crew started their first sleep period. The differential pressure indications came back on after spacecraft power down and remained on through the sleep period. Fuel-cell water pressure remained steady through the sleep period, which indicated that no water was being transferred to the adapter tank. This could have been due to a stack water separator leaking gaseous oxygen into the tank, thus depleting the nitrogen ullage and collapsing the tank bladder.

Over the RKV during revolution 6, the command was sent to open the Experiment S012 (Micrometeorite Collection) door, and over the RKV during revolution 10, the command was sent to close and lock the S012 door.

The crew were awakened over the Canary Islands during revolution 10, and a second SPS eclipse phasing maneuver was transmitted to the crew. This maneuver was accomplished as follows:

Time of maneuver, g.e.t.	15:16:18
ΔV, ft/sec	15.0 (posigrade)

Pitch, deg 0

Yaw, deg 0

Thrusters . . . . SPS Unit II

The crew reported seeing the eclipse "right on the money at 16:01:44 g.e.t." At 16 hours 54 minutes g.e.t. just prior to reaching Carnarvon during revolution 11, the differential-pressure lights extinguished. The crew reported that they had drunk water just prior to the lights going out, which indicated that the fuel-cell product water was probably being transferred to the adapter water tank. The lights came back on over Carnarvon at 20 hours 2 minutes g.e.t. and remained on for most of the mission, extinguishing only when the crew withdrew drinking water from the adapter tank.

Over Cape Kennedy during revolution 13, the crew reported that the right microphone in the pilot's helmet was inoperative. At 19 hours 15 minutes g.e.t., the crew performed their suit integrity checks for the standup extravehicular activity (EVA). The suit-pressure decay rates were 0.26 and 0.23 psi in 30 seconds for the pilot and the command pilot, respectively. The spacecraft hatch was opened for the standup EVA at 19:29:01 g.e.t. After termination of the EVA, the hatch was closed at 21:58:35 g.e.t.

Over Tananarive during revolution 16, the crew reported that they were still having some GATV control problems, in that the GATV was overshooting the desired attitude and was slow in settling down. Over Hawaii during revolution 16, 30-second oxygen purges were initiated on each fuel cell section to be performed once every revolution in order to force more water from the fuel cells but with the amount of oxygen originally allotted for purges. Over Ascension during revolution 17, the crew reported difficulty with the 70-mm general-purpose camera shutter opening and closing at the proper time, but over Tananarive during revolution 17, they reported that they believed they had found a way to make it work properly.

Over Carnarvon during revolution 17, at 26 hours 30 minutes g.e.t., the Environmental Control System (ECS) control valve in the primary loop was fluctuating from 26 to 54 degrees in 20-second cycles, indicating that the loop had inadequate flow using the B-pump to maintain control. The A-pump was selected and the temperature stabilized at 40 degrees. Over Texas 45 minutes later, the B-pump was again selected; however, there was no recurrence of the problem.

Over Tananarive during revolution 19, the crew entered the second sleep period. The propellant quantity remaining at this time was 49 percent.

Over the CSQ during revolution 23 at 36 hours 5 minutes g.e.t., fuel cell stack 2B was noted to be reading 1.06 amperes. Over the RKV during revolution 24 at 36 hours 48 minutes g.e.t., the crew was awakened to check the open-circuit voltage. The voltage was 30.7 volts, so the stack was placed back on the line and an oxygen purge was performed. A data replay indicated that stack 2B had started degrading at 35:29:40 g.e.t. The current held steady at 1.08 amperes until Kano during revolution 24 at which time the crew reported that the stack was producing no current. Stack 2B was turned off at that time.

Over Antigua during revolution 26, the crew reported that OAMS thrusters no. 2 (pitch) and no. 4 (yaw) were not operating. These thrusters reportedly gave no response at all when fired singly. Over Cape Kennedy during revolution 27, the crew reported that they believed that there was a definite correlation between extraction of drinking water and the differential pressure lights going off.

The spacecraft was prepared for the umbilical EVA over the Canary Island station during revolution 27, and a GO for cabin depressurization was given at Carnarvon, even though oxygen pressure was dropping with the manual heater on. This was due to the high flow through the Extravehicular Life Support System (ELSS) with the visor open. Cabin depressurization was initiated at 42 hours 26 minutes g.e.t. between the Canton and Texas stations.

Over Texas, the oxygen pressure was normal, with a 30 psi/min rate of rise with the heater on. During the umbilical EVA, the pilot remained on high flow and bypass closed instead of medium flow and bypass open, which helped the critical oxygen situation considerably. After exit from the hatch, the pilot hooked the tether from the GATV Target Docking Adapter (TDA) to the spacecraft docking bar. He reported that the EVA was going well but that his feet were cold. Shortly before starting for the adapter area, the pilot reported seeing a large icicle on the hydrogen vent. The pilot was in the adapter area and had his feet in the adapter foot restraints at 42 minutes 20 seconds after hatch opening. Twenty seconds later, the pilot reported difficulty in installing the adapter EVA camera and that the linkage was broken. The pilot completed the adapter work tasks, returned to the TDA, completed all tasks at that station, and returned to the hatch area, where he cleaned the command pilot's window. The pilot examined thruster 4 and reported that he saw pieces of something white similar to a urine dump when the thruster was fired. The oxidizer feed temperature had been as low as 27° F during the previous sleep period and it was believed that oxidizer had frozen in the OAMS plumbing. The hatch was locked at 44:55:25 g.e.t. over Cape Kennedy during revolution 29, and cabin repressurization was initiated with the ELSS. The cryogenic oxygen quantity was reported to be six percent above nominal at this time. The command pilot reported that while the cabin was being repressurized, his eyes had started burning but had cleared up as soon as he was able to open his face plate and rub his eyes.

Over Canton during revolution 29, the crew reported difficulty in attitude control with the OAMS in direct or rate command because of thrusters no. 2 and no. 4 being inoperative. Over Tananarive during revolution 30, the crew started setting up the spacecraft for the tether evaluation. At Hawaii during revolution 30, the crew reported little success in holding position because of poor attitude control. At Hawaii during the next revolution, the crew reported that the spacecraft control systems were off and everything looked stable. Over the United States during revolution 31, the GATV attitude control system was turned off after the control gas had been depleted to five pounds (49 hours 30 minutes g.e.t.). Over Tananarive during revolution 33, the tether was released. Following the tether evaluation, a separation sequence was planned such that, if the GATV could be restarted, it would still be feasible to perform a re-rendezvous. This proposal involved two spacecraft maneuvers of 6 ft/sec ΔV. The first maneuver was to have been posigrade and the second retrograde, separated by one orbit in time; however, this plan was modified because of a desire to allow the flight crew to go to sleep as early as possible. An alternate plan was derived to do the second maneuver posigrade with the GATV, but, due to the attitude control gas shortage on the GATV, this second maneuver was never performed. The net result of this planning was a 6 ft/sec posigrade separation maneuver with the spacecraft at 52:14:27 g.e.t. The separation maneuver was accomplished by using the forward-firing maneuver thrusters for yaw control. The crew reported that the rate command mode had worked for attitude control for the maneuver. The loss of thrusters no. 2 and no. 4 caused a loss of pure roll-left capability. However, roll left could be obtained through the use of the remaining thruster (no. 8) and the use of pitch and yaw to remove cross-coupling.

The spacecraft was powered down over the RKV during revolution 34 in preparation for the third sleep period. The crew were awakened over the RKV during revolution 39 at 60 hours 46 minutes g.e.t. A retrograde maneuver of 5.5 ft/sec was performed at 61:47:48 g.e.t. to stop the opening rate between the spacecraft and the GATV. The separation rate was essentially stopped by this maneuver, and the separation distance was approximately 120 nautical miles at retrofire.

Over Antigua during revolution 40, the crew was given the pointing data for Experiment S051 (Sodium Vapor Cloud), and they attempted to take the required photographs. Two French rockets were launched at the required times on successive orbits and the sodium was deployed. The crew reported no visual contact on either launch. Launch times were:

First rocket launched	62:41:48 g.e.t.	Revolution 40
Second rocket launched	64:16:41 g.e.t.	Revolution 41

Over Kano during revolution 41 at 66:05:55 g.e.t., the cabin was depressurized for the second standup EVA. The suit integrity check had been performed over the Canary Islands, with a leak rate of 0.27 psi in 30 seconds for the command pilot and 0.10 psi in 30 seconds for the pilot. Suit pressures during the EVA were 3.62 psi for the command pilot and 3.60 psi for the pilot, which indicated that the pressure reference had shifted lower by 0.1 psia for the suit demand regulators from the first EVA. However, this was still greater than the minimum acceptable pressure of 3.5 psia. The EVA was completed over the United States and repressurization initiated at 67:03:03 g.e.t.

Over Carnarvon during revolution 43, the crew reported that thruster no. 8 had failed. This completely canceled the roll-left capability, so the crew was advised to go to PITCH on the ROLL JETS logic switch.

Over Texas during revolution 43, the crew was informed that the adapter water tank was almost depleted. The water gun counter at that time was reading 2024, and 2060 was the predicted empty point. The crew was advised that after the tank was empty they would have to use the blood pressure bulb to pressurize the reentry water tank.

Over Texas during revolution 46, the crew reported that all thrusters had degraded to the point that rotational imbalances were even. A procedure for calibrating OAMS thrust was transmitted to the crew, and an OAMS test was scheduled to be performed over Kano during revolution 56, after the final sleep period.

The crew entered the final sleep period over Hawaii during revolution 48. The propellant quantity remaining at this time was 24 percent and the water gun read 2074.

Over Hawaii during revolution 49 at 78 hours g.e.t., the cryogenic hydrogen pressure was noted to be rising. The hydrogen pressure was reading 344 psia, which was 44 psia greater than desired. Over the RKV during revolution 50 at 78 hours 24 minutes g.e.t., the hydrogen pressure was still rising, so the crew were awakened and asked to check the position of the hydrogen heater switch. They reported the heater switch to be in AUTO, thus indicating a pressure switch failure. Based on the decay rate being small with the heater off, the cryogenic oxygen and hydrogen heater circuit breaker was opened to disable the heater. Later, troubleshooting by closing the circuit breaker with the switch in the OFF position isolated the problem to the MANUAL/OFF/AUTO switch or associated circuitry.

Over the RKV during revolution 54, the crew was again awakened to turn off fuel cell stack 1C, which was indicating zero current.

Over Cape Kennedy during revolution 55, the crew reported that they had run out of water in the adapter tank and were having little success pressurizing the reentry tank with the blood pressure bulb.

Over Kano during revolution 56, the OAMS thrust calibration test was completed. Thrusters no. 4 and no. 8 appeared to be completely inoperative and thrusters no. 2 and no. 7 were delivering very little thrust.

The spacecraft was powered up over the United States during revolution 57. During the next pass, main batteries 1 and 4 were placed on the line to aid the fuel cells with the additional load of the platform. The computer was brought up over Carnarvon during revolution 58, and an accelerometer bias check was made. The Z axis was in error by 0.022 pulse/sec, so it was updated over the United States at the end of revolution 58.

The retrofire time (T_R) for area 1A in revolution 60 and the pre-retrofire update were transmitted to the crew over the United States at the end of revolution 58. The load and the T_R were based on a Woomera revolution 58 solution. The spacecraft T_R was within 0.125 seconds of the ground T_R. The computed retrofire time of 93:59:58 remained unchanged as subsequent tracking data were accepted. The retrofire update quantities input into the Real Time Computer Complex (RTCC) on November 15, 1966, were as follows:

Time of retrofire, G.m.t. 18:46:31

Landing

Revolution 60

Geodetic latitude . . . . 24 deg 35 min north

Longitude . . . . 70 deg west

Area 60-1A

The Reentry Control System (RCS) was armed over Texas during revolution 58. The B-ring was normal, but the A-ring regulated pressure did not hold within the regulator tolerance. At arming, A-ring regulated pressure was 310 psia, but began increasing at a rate of 5 psia/min. The crew was requested to fire the A-ring to ascertain whether propellant usage would slow down the rate of increase. At the Canary Islands acquisition of signal (AOS), the A-ring regulated pressure was 413 psia, so the crew was advised to use the A-ring in order to prevent breaking the burst diaphragm (burst point is 420 to 500 psia). The crew continued their platform alignment using the A-ring. At Carnarvon AOS, the regulated pressure was 385 psia, but the crew were instructed to use enough control to drop the pressure to 310 psia at Carnarvon LOS. Carnarvon data showed that 10 pounds of A-ring propellant had been used for platform alignment and control of the regulated pressure.

The last two main batteries (2 and 3) were placed on the line over Cape Kennedy during revolution 59, and fuel cell section 2 was turned off. This was done to prevent a failure in the section going unnoticed due to stack currents displaying main battery currents. The main batteries continued to carry the full current loads with a bus voltage of 23 to 24 volts.

6.1.4 Retrofire and Reentry

The countdown to retrofire occurred over Canton Island during revolution 59. The crew reported an on-time, automatic retrofire, with incremental velocity indications of 301 ft/sec aft, 115 ft/sec down, and 4 ft/sec left. These compared well with the desired values of 302 ft/sec aft, 113 ft/sec down, and zero right/left. Hawaii telemetry showed the retrofire velocities to be 302.4 ft/sec aft, 115.3 ft/sec down, and 3.4 ft/sec left. The Auxiliary Computer Room (ACR) calculated an initial downrange deflection of 61 nautical miles uprange compared with the nominal value of 64 nautical miles which had been transmitted to the crew prior to retrofire. Tracking data following retrofire showed a backup bank angle of 51 degrees with a reverse bank time of 26 minutes 3 seconds after retrofire. The backup guidance quantities transmitted to the crew were as follows:

Roll left, deg	46
Roll right, deg	56
Time after retrofire to reverse bank angle, min:sec	26:03

The Hawaii data after retrofire showed the A-ring regulated pressure to be 298 psia, which was normal. The A-ring regulator appeared to function satisfactorily after that. The regulator "creep" was apparently arrested once the pressure dropped below regulator opening pressure (280 psia), and a new seat was established.

The following event times were updated prior to blackout: 400K feet, drogue deployment, and main parachute deployment. Begin and end blackout times were within four seconds of the predicted times, so they were not updated. The onboard computer indication of 400K feet occurred at 20 minutes 14 seconds after retrofire, which was six seconds later than predicted but well within the allowable tolerance of 40 seconds late. The final RTCC predicted landing point was 24 degrees 33 minutes north and 69 degrees 55 minutes west. This indicated a total miss distance of 5.4 nautical miles. An automatic reentry was flown, and at drogue deploy, 3 pounds of propellant remained in the A-ring and 20 pounds in the B-ring. At the final loss of telemetry signal, the errors were 2.9 nautical miles downrange and 1.6 nautical miles crossrange to the left (north).

6.1.5 GATV Orbital Flight

The GATV was gyrocompassed to minus 90 degrees over Carnarvon during revolution 1. At Hawaii during revolution 1, the L-band transponder and the lights were turned on. Over Texas during revolution 1, the GATV was verified to be in the proper rendezvous configuration.

Over the United States during revolution 2/3, the crew reported solid lock-on at a range of 235.5 nautical miles. Over Carnarvon during revolution 3, the crew reported they had lost radar lock-on at a range of 64 nautical miles. The GATV telemetry showed a good encoder lock-on with an L-band transponder output of 3.09 to 3.96 volts. Over Hawaii during revolution 3, the crew sent several commands to the GATV. The Hawaii controllers confirmed the commands by MAP's and events, although the crew did not receive MAP's or range data. The crew abandoned the RF closed-loop rendezvous technique and used the visual rendezvous technique. As a result of the command test at Hawaii, it was evident that the GATV command system was functioning properly and that the problem was probably in the radar signal-return system. After docking was accomplished, electrical continuity was verified between the GATV and the spacecraft.

The initial docking occurred over the CSQ during revolution 4, after which a docked gyrocompass 90-degree yaw was made to a 0, 0, 0 heading. The first undocking was made between Texas at the end of revolution 4 and RKV at the beginning of revolution 5. Over the RKV, the crew reported that they were having some minor control problems with the GATV and that the GATV was very slow in recovering. The undocking had been made using the spacecraft forward-firing thrusters, thus probably inducing some rates into the GATV. Since the GATV was in flight control mode 1, it took about 24 minutes and nine pounds of control gas to damp the rates.

The second docking took place over Tananarive during revolution 5. It was later stated that the spacecraft was flying at a 5-degree yaw heading, and everytime the crew pulsed their control system to 0, 0, 0, the GATV would bring them back to a 5-degree yaw heading. During these pulse maneuvers, the GATV was in flight control mode 2 and 12 pounds of attitude control gas were used.

During revolution 6, a GATV SPS Unit II retrograde maneuver of 43.0 ft/sec was performed. It was then noted by the crew that their platform was misaligned, resulting in the 5-degree yaw error. A realignment was made before the sleep period and the control problems appeared to be solved. Approximately 38 percent of the GATV attitude control gas was used up to the beginning of the sleep period.

No GATV problems were noted during revolutions 11 through 16. The activities included two yaws, an SPS maneuver, and a standup EVA. The SPS maneuver was a phasing maneuver and was accomplished during revolution 11. The firing was an SPS Unit II 15 ft/sec posigrade maneuver at 15:16:18 g.e.t. Over the Canary Islands during revolution 12, it was noted that the spacecraft thrusters were being used during a yaw maneuver. This worked against the GATV system and caused seven pounds of attitude control gas to be used. During the pass over the United States during revolution 17/18, the crew was configuring for an Experiment S029 (Libration Regions Photography) sequence and inadvertently left the horizon sensors on. This resulted in the GATV trying to move to its initial heading. The horizon sensors were turned off and the control systems worked correctly. During revolution 18, the crew reported that they were having difficulties controlling the vehicle in the gyrocompassing mode. Over the CSQ during revolution 19, a gyrocompassing test was started and the GATV systems checked out properly. In the period between the first and second sleep periods, 44 pounds of attitude control gas were used.

During revolution 27, the GATV was gyrocompassed from 180 degrees to 90 degrees in preparation for the umbilical EVA. The EVA functions were accomplished during revolutions 28 and 29 with the GATV in inertial attitude hold and operating correctly. During revolution 31, the spacecraft was undocked from the GATV and the tether evaluation was started. The tether evaluation continued through revolution 34, when the tether was jettisoned over Tananarive.

At Hawaii during revolution 34, the GATV was placed in flight control mode 1 and gyrocompassed to a 0, 0, 0 heading, and at RKV during revolution 35, the GATV memory was flushed.

Over the CSQ during revolution 35, a stored program command and a velocity meter load were transmitted. The program command load was verified, but a velocity-meter "no compare" was received on the velocity meter load. Over Hawaii during revolution 35, the GATV ran out of attitude control gas, so the attitude control system gains were placed to low and the velocity meter and the stored program commands were disabled. Over the RKV during revolution 36, the GATV memory was flushed. Over the CSQ, the velocity meter load was again transmitted and MAP's were received, but the velocity meter word did not change. It could not be determined whether the velocity meter telemetry was bad or whether the hardware had malfunctioned. The velocity meter was considered inoperative after similar tests were performed over subsequent sites.

After spacecraft reentry, a 1550 ft/sec PPS firing was attempted over Hawaii during GATV revolution 63. A 20-second type-A start was programmed. The SPS ullage orientation maneuver was successful; however, the PPS engine start sequence was terminated by a turbine overspeed shutdown. No thrust chamber pressure was observed prior to the shutdown. The turbine overspeed indication verified turbine speed data obtained during the insertion firing. No further PPS operation was attempted.

At the RKV during revolution 64, the MCC-H ended support of the GATV by disabling the UHF command system and leaving the telemetry, C-band transponder, and S-band transponder on for range calibration exercises. The vehicle weight and consumables remaining were as follows:

PPS ΔV, ft/sec	5810
SPS ΔV, ft/sec	270
PPS firing time, sec	58.23
SPS (Unit II) firing time, sec	138.89

Vehicle weight, lb	6886
Control gas, lb	0.0
Electrical power, A-h	880

TABLE 6.1-I.- SPACECRAFT MANEUVER SUMMARY

Maneuver	ΔV, ft/sec	Time of maneuver, g.e.t.	Thrusters used	Propellant used, lb	Remarks
Separation	3.2	0:06:07	Aft	3	
IVAR	30.6	0:06:44	Aft	42	
IVAR		0:06:51	Right		
N_{C1}	66.7	0:49:40	Aft	70	
N_{PC}	8.5	1:14:22	Left	13	Onboard solution
N_{CC}	8.4	1:47:52	Aft	11	
N_{SR}	49.9	2:22:54	Aft	47	Onboard solution
TPI	21.8	3:05:47		26	
Separation	7.1	52:14:27	Aft	14	
Phasing	5.2	61:47:48	Aft	35	Thruster problems

6.2 NETWORK PERFORMANCE

The network was placed on mission status for the Gemini XII mission on October 29, 1966, and supported the mission satisfactorily. Lift-off of the GAATV occurred at 19:07:59 G.m.t. on November 11, 1966, and lift-off of the Gemini Space Vehicle occurred at 20:46:33 G.m.t. on November 11, 1966. The spacecraft landed at 19:21:03 G.m.t. on November 15, 1966.

6.2.1 MCC and Remote Facilities

The network configuration and the general support required of each station are indicated in table 6.2-I. To permit the further installation of Apollo modifications, the station at Guaymas, Mexico, was released from all support other than air-to-ground remoting, S-band radar tracking, telemetry receive and record, and teletype and voice communciation. Figure 4.3-1 shows the location of the stations which make up the world-wide network. In addition, approximately 15 aircraft provided supplementary photographic, weather, telemetry recording, and voice-relay support in the launch and recovery areas. Certain North American Air Defense Command (NORAD) radars tracked the Gemini Launch Vehicle (GLV), Target Launch Vehicle (TLV), Gemini Agena Target Vehicle (GATV), and the 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 Telemetry.- No major problems were encountered in the telemetry area. Only about one minute of data was lost during this mission. During spacecraft revolution 13, Bermuda PCM ground station no. 1 failed and no. 2 was selected as prime. Approximately 53 seconds of 2-kilobit data and 76 seconds of biomedical data were lost while the monitor patchboard was changed.

6.2.2.2 Radar.- Radar tracking during the mission was excellent. Only one problem of importance occurred during the mission. The interrange vectors for the Ascension and Pretoria stations during spacecraft revolution 1 had a 3-minute time bias. This error was due to an invalid computer input in the Real Time Computer Facility (RTCF) at Cape Kennedy. Ascension and Pretoria radars did not acquire track. This constituted the only avoidable data loss during the entire mission. Unfavorable vehicle attitudes precluded horizon-to-horizon tracking from some stations during several revolutions.

6.2.2.3 Acquisition aids and timing.- All acquisition aid systems and timing systems operated satisfactorily during the mission.

6.2.2.4 Command.- Minor hardware problems were experienced at the Bermuda, Hawaii, Carnarvon, and Texas sites. Due to equipment redundancy and rapid repairs, these problems did not adversely affect the mission.

6.2.2.5 Computers.-

6.2.2.5.1 Real Time Computer Complex (RTCC-Houston): Two system failures occurred during the mission. The first failure occurred at 58 hours 25 minutes g.e.t., when the Mission Operations Computer (C machine) stopped with a power loss on the B channel. The power loss was caused by a frozen blower fan, but the fan was replaced and the computer was operational in a matter of minutes. Because the Real Time Computer Complex (RTCC) was supporting the mission with one computer only, a second computer (B machine) was initialized from a Type B restart tape, taken only a few minutes prior to the failure, and no data were lost due to the failure.

The second failure occurred at 60 hours 26 minutes g.e.t. while attempting to write a Type A restart out of the Mission Operations Computer (B machine) to bring up a Dynamic Standby Computer (C machine). The restart was written on a bad reel of tape and the resulting continuous output statements of "Redundancy occurred while writing restart tape, will attempt recovery" were stacked in the buffer pool. The output statements could not be printed because of the bad restart tape, and caused a buffer pool overflow. This problem will be reviewed, and a recommendation will be made for future operations.

6.2.2.5.2 Real Time Computer Facility: The only significant problem involving the Real Time Computer Facility at Cape Kennedy was the erroneous loading of the Gemini lift-off time, causing a 3-minute bias in the interrange vector for the Ascension and Pretoria stations. This caused these stations not to acquire radar track after insertion.

6.2.2.5.3 Goddard Real Time System: No significant problems involving the Goddard Real Time System occurred during the mission.

6.2.2.5.4 Remote Site Data Processors: Both hardware and software performance of the Remote Site Data Processors (RSDP) was satisfactory throughout the mission. The following problems occurred during the mission:

(a) The Carnarvon RSDP faulted during revolution 11. Reinitiation cleared the fault and no data were lost.

(b) Spacecraft 40.8-kilobit data were bad at Antigua during revolution 27; however, data from Grand Turk Island were valid and no data were lost. The problem was evidently caused by a tape playback just prior to the Antigua pass during revolution 27. The computer faulted when brought up to support revolution 27. A reloading of the program and playback of the revolution 27 data produced good results and the problem did not recur.

(c) The RSDP on the Coastal Sentry Quebec faulted during spacecraft revolution 34. Reloading cleared the fault and it did not recur. No data were lost.

(d) The Hawaii RSDP produced the wrong time tag on an onboard computer summary during spacecraft revolution 49. Reinitializing cleared the fault and a replay of the data resulted in no data loss.

6.2.2.6 Communications.-

6.2.2.6.1 Ground communications: The only significant ground communications problems were associated with the Communications Processors (CP). On launch day, CP-A went down six times with a memory fault. CP-B was always available as a backup and no data were lost. The problem was finally traced to an intermittent memory address card in the B-2 section of CP-A. On F+3 day CP-B was on line and had a vacuum problem. CP-B was patched to the CP-A servos with no data loss. CP-B servos were back on line two and a half hours later.

6.2.2.6.2 Spacecraft communications: The only significant spacecraft communications problem was the nonavailability of air-to-ground voice through Ascension just after insertion. This was caused by a procedural error.

Successful two-way remoting with the spacecraft was accomplished during spacecraft revolutions 15 and 44 using an Apollo Range Instrumentation Aircraft (ARIA) when the spacecraft was near the Texas station, rather than remoting through that station.

TABLE 6.2-I.- GEMINI XII 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	Riometer	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	0	X	X
A/C		X																		X			
ANT	X	X	X			X						X	X	X			X		X	X	0		
ASC		X	X																X	X			
BDA	X	X	X			X					X	X	X	X		X	X			X	0	X	X
CAL	X	X	X													X				X		X	X
CNV			X[b]			X				X				X		X				X	X		
CRO	X		X	X			X	X					X	X	X	X	X	X		X	X	X	X
CSQ	X		X	X			X	X					$\boxed{X}$	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	X
EGL	X		X																			X	X
GBI	X	X	X			X				X		X	X	X		X	X		X	X	0		
GTI	X	X	X			X				X		X		X		X	X			X	0		
GYM	X	X					X	X					X			X	X			X	X	X	X
HAW	X		X	X			X	X					X	X	X	X	X			X	X	X	X
KNO	X	X																		X		X	X
MLA			X																				
PAT			X																				
PRE			X																				
RKV	X			X			X	X					$\boxed{X}$	X	X		X			X	X	X	X
RTK/WHE	X	X	X																	X		X	X
TAN	X	X																		X		X	X
TEX	X	X		X		X						X	X	X		X				X	(*)	X	X
LIMA																		X					
WHS	X		X																			X	X
WOM	X		X																			X	X

[a]Location of stations is shown in figure 4.3-1.

[b]Wind profile measurements in support of recovery operations.

(X) Master DCS

0 Remoting

$\boxed{X}$ Post-pass biomed remoting

(*) Real time and remoting

6.3 RECOVERY OPERATIONS

6.3.1 Recovery Force Deployment

Recovery plans and procedures were established for the Gemini XII mission to assure the rapid location and safe retrieval of the flight crew and the spacecraft, following any conceivable landing situation. Planned and contingency landing areas were defined in accordance with the termination-of-mission probabilities. Planned landing areas included the launch-site, launch-abort, and periodic areas. All landing areas other than these were considered to be contingency landing areas.

Department of Defense (DOD) forces provided recovery support in each of the various landing areas. The level of support provided was commensurate with the probability of a landing occurring within a particular area and with any special problems associated with such a landing. Table 6.3-I contains a summary of those forces committed for Gemini XII recovery support. The planned landing areas, in which support forces were positioned for search, on-scene assistance, and retrieval, were located and defined as follows:

(a) The launch-site landing area was that area in which a spacecraft landing would have occurred following an abort prior to launch or during the early part of powered flight. It included the area in the vicinity of Launch Complex 19 and extended seaward along the ground track for a distance of 41 nautical miles. Recovery force deployment in this area is shown in figure 6.3-1.

(b) The launch-abort landing area was the area in which a spacecraft landing would have occurred following an abort after approximately 100 seconds of flight but before insertion into orbit. This area originated at the seaward extremity of the launch-site area and was bounded by the most northern and southern planned launch azimuths. A map of the area showing the recovery support provided is presented in figure 6.3-2.

(c) Once the spacecraft was inserted into orbit, the ground track passed periodically through or near one of four selected landing zones. These zones (480 nautical miles in diameter) was located as follows: West Atlantic, East Atlantic, West Pacific, and Mid-Pacific. The landing areas (200 by 40 nautical mile ellipses with major axes along the ground track) were located within or near these zones. These landing areas were defined as "periodic landing areas."

Periodic landing areas were further subdivided into primary and secondary landing areas for convenience in identifying areas with the higher landing probability and level of recovery support. Primary landing areas included the region within or near the West Atlantic zone (centered at 25 degrees north latitude and 65 degrees west longitude) and were supported by the primary recovery ship. The end-of-mission landing area for the beginning of revolution 60 was located just to the west of this zone. Because areas within the West Atlantic zone were designated go/no-go areas and a high probability existed that the mission would be terminated with a landing in this zone, an aircraft carrier with airplane and helicopter detachments was assigned for recovery support. In addition, fixed-wing tracking and search/rescue aircraft were staged in the vicinity to assist in the recovery operation.

The secondary landing areas were located within, or near, three recovery zones spaced such that a rapid recovery capability existed at frequent intervals throughout the flight. These zones were located in the East Atlantic, West Pacific, and Mid-Pacific. Fixed-wing search/rescue aircraft were stationed at the nearest feasible airfield to locate the spacecraft and provide on-scene assistance in the event of a landing in one of these zones. Figure 6.3-3 illustrates the recovery zone concept and the support provided for the primary and secondary landing zones.

Provisions for recovery support in the event of a contingency landing consisted of fixed-wing search/rescue aircraft on alert at staging bases. These bases, as follows and as shown in figure 6.3-4, were located such that any point on the Gemini XII ground track could be reached within 18 hours after notification of spacecraft landing:

Bermuda
Lajes, Azores
Mauritius Island
Perth, Australia
Hickam AFB, Hawaii
Tachikawa, Japan
Dakar, Senegal
Okinawa
Pago Pago, American Samoa
Lima, Peru
San Diego, California

In addition, aircraft on normal alert at other bases were available to assist in a recovery operation if needed.

Two spacecraft target points per revolution, one designated the planned and the other the contingency landing area, were preselected. The planned landing area was selected in, or near, one of the landing zones whenever possible. The contingency point was usually selected close to recovery forces at staging bases and, when possible, in a landing zone.

6.3.2 Location and Retrieval

Based upon the precomputed ground track for a nominal mission, an end-of-mission landing area was preselected in the West Atlantic landing zone at the beginning of revolution 59. Failure to perform the high-orbit GATV PPS maneuver at the end of revolution 5 resulted in an orbital period of less duration than the nominal for several revolutions. Due to the changed ground track, a decision was made to land near the beginning of revolution 60.

Retrofire was initiated so that landing would occur in the West Atlantic recovery zone just after the beginning of revolution 60. The U.S.S. Wasp (CVS 18) was positioned at 24 degrees 33.9 minutes north geodetic latitude and 69 degrees 56.0 minutes west longitude, near the selected target point. Fixed-wing search/rescue aircraft and aircraft from the U.S.S. Wasp were positioned in an array as shown in figure 6.3-5.

Spacecraft landing occurred at 19:21:05 G.m.t. on November 15, 1966, at 24 degrees 36.4 minutes north latitude and 69 degrees 56.2 minutes west longitude, 3.8 nautical miles from the target point. Position data were determined by LORAN fixes taken at the time of recovery and were checked against celestial fixes taken during the morning and evening of the day of recovery. The position of the spacecraft at the time of retrieval was 24 degrees 35.2 minutes north latitude and 69 degrees 56.2 minutes west longitude. Figure 6.3-6 shows the Gemini XII spacecraft at landing, and figure 6.3-7 shows the relative target, landing, and pickup positions.

The following is a sequence of events as they occurred prior to and during the recovery operation on November 15, 1966:

Greenwich mean time, hr:min	Ground elapsed time, hr:min	Event
18:47	94:00	Retrofire
19:12	94:25	Radar contact by U.S.S. Wasp
19:18	94:31	Visual sighting from helicopter (Search 3) and U.S.S. Wasp
19:21	94:34	Spacecraft landing
19:24	94:37	First swimmers in water
19:25	94:38	Flight crew reported in good condition
19:27	94:40	Flotation collar attached

Greenwich mean time, hr:min	Ground elapsed time, hr:min	Event
19:28	94:41	Flotation collar and raft inflated
19:38	94:51	Flight crew in raft
19:40	94:53	Command pilot getting back inside spacecraft to retract HF antenna
19:41	94:54	HF antenna retracted
19:45	94:58	Both crewmembers aboard helicopter
19:49	95:02	Rescue helicopter with flight crew aboard U.S.S. Wasp
20:18	95:31	U.S.S. Wasp 1000 yards from spacecraft
20:24	95:37	Line attached to spacecraft
20:26	95:39	Spacecraft lifted from water
20:28	95:41	Spacecraft aboard U.S.S. Wasp, secured in dolly

Both the Rendezvous and Recovery (R and R) section (fig. 6.3-8) and the main parachute were sighted during descent but sank before they could be retrieved. The R and R section did not contain flotation equipment and, as expected, sank immediately upon landing. The PARA JETT switch was activated by the flight crew upon landing, and choppy seas caused the parachute to sink almost immediately. Retrieval of the R and R section and the main parachute was not required.

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	Aircraft altitude, ft	Inital reception range, n. mi.	Receiver
Search 1 (SH-3A)	19:18	8 000	20	SPP
Search 2 (SH-3A)	19:18	8 000	30	SPP
Search 3 (SH-3A)	19:17	4 000	2	SPP
Kindley Rescue 1 (HC-130H)	19:18	26 000	200	AN/ARD-17
Kindley Rescue 2 (HC-130H)	19:18	24 000	200	AN/ARD-17
Kindley Rescue 3 (HC-130H)	19:18	25 000	125	AN/ARD-17

In each case the reception range listed is not the maximum obtainable range but represents the distance between the spacecraft and the recovery aircraft at beacon activation time.

6.3.3.2 HF transceiver (15.016 mc).- The HF antenna was extended and retracted prior to spacecraft retrieval. There were no reports of HF voice or beacon reception received from the recovery forces. However, 13 HF-DF stations reported beacon reception and the spacecraft position calculated from these data was within five nautical miles of the actual location. This location was made known to the recovery force within four minutes after spacecraft landing.

6.3.3.3 UHF voice transceiver (296.8 mc).- One of the helicopters, Search 2, reported an unreadable voice reception at 19:06 G.m.t. and readable receptions starting at 19:16 G.m.t. The earlier reception consisted of just a few words of about 1-second duration and was assumed to originate from the spacecraft. The primary recovery ship and the remaining helicopters reported UHF voice reception starting at 19:16 G.m.t.

Neither recovery aircraft nor the recovery ship reported receiving landing position information (addresses 86 and 87) or "miss distance" information from the spacecraft.

6.3.3.4 UHF survival radio (243.0 mc).- The UHF survival radio was not activated.

6.3.3.5 Flashing light.- The flashing light erected properly but was not activated by the flight crew.

6.3.3.6 Fluorescein sea dye marker.- The sea dye marker diffused normally and was clearly visible to the primary recovery ship and aircraft in the landing area.

6.3.3.7 Swimmer interphone.- At 19:25 G.m.t. the swimmers established voice contact with the crew over the swimmer interphone. The swimmers reported that the interphone did not function with the volume control switch in "High" but operated normally in the "Low" position.

6.3.4 Postlanding Procedure

The spacecraft was observed during descent on the parachute and landed less than three miles from the recovery ship. After spacecraft landing, the recovery swimmers deployed immediately, established communications, and began installation of the flotation collar.

After collar inflation, the crew egressed the spacecraft, but the command pilot returned to the spacecraft and retracted the HF antenna. The crew were then transported to the U.S.S. Wasp by helicopter.

Spacecraft retrieval was normal, with no difficulties encountered, and observations were as follows:

(a) The HF antenna had been extended and retracted.

(b) The flashing light was erected.

(c) The dye marker was released.

(d) Both UHF antennas were erected.

(e) The center equipment-bay door had been indented.

(f) The edge of one shingle on the lower left-hand side was curled.

(g) The right window was about 80 percent fogged, and both windows were covered with residue.

(h) The parachute releases had been activated.

(i) The heating effects appeared normal and were similar to previous spacecraft.

(j) The right pitch-down B-ring thruster of the Reentry Control System (RCS) was leaking very slightly.

(k) The pyrotechnic for the fresh air door had detonated.

(l) The interior of the spacecraft was in good condition. Moisture was found in the left footwell. All gear was properly stowed; however, the Environmental Control System hoses were not interconnected.

(m) Both ejection seat D-rings and drogue mortars were pinned.

(n) The hatch seals were in good condition.

(o) No abnormal cabin odors were detected.

Approximately two and one-half hours after spacecraft retrieval, crew blood samples and news film were flown from the recovery ship by U.S.S. Wasp aircraft to Grand Turk Island for a connecting flight with an Eastern Test Range JC-130 telemetry aircraft to Patrick Air Force Base. A data delivery flight (onboard spacecraft equipment and film) departed the U.S.S. Wasp at 11:00 G.m.t. on November 16, 1966, for Patrick Air Force Base. The crew and other NASA and DOD officials departed the U.S.S. Wasp by aircraft at 14:00 G.m.t. the same day for the Cape Kennedy skid strip. The U.S.S. Wasp docked at the Boston Naval Shipyard on Friday, November 18, 1966. The spacecraft was off-loaded, and deactivation procedures were started immediately.

6.3.5 Spacecraft Reentry Control System Deactivation

The Landing Safing Team (LST), consisting of NASA and spacecraft contractor engineers and technicians, was responsible for deactivating the RCS according to the procedures of reference 20. The deactivation was accomplished at the Boston Naval Shipyard, Boston, Massachusetts, in order to safe the system prior to air-transporting the spacecraft to the contractor facility in St. Louis, Missouri.

At 2:00 p.m. e.s.t., November 18, 1966, following the arrival of the U.S.S. Wasp at Boston, the spacecraft was off-loaded from the hangar deck of the ship. The RCS shingles had previously been removed on the ship, and, because no visual damage to the system was apparent, the deactivation procedures were immediately initiated by the LST. Prior to system flushing, raw propellant samples were taken for analysis which indicated that the propellant in both rings met the required cleanliness specifications. The nitrogen source pressurant remaining in the A-ring and B-ring systems was 1030 psig and 1340 psig, respectively, and the regulated pressure for the rings was 290 psig and 285 psig, respectively. The weights of the propellants remaining in the spacecraft before deactivation were as follows:

Ring	Fuel, lb	Oxidizer, lb
A	0.06	0.25
B	2.00	4.5

Personnel on the recovery ship had reported to the team leader of the LST that the no. 1 thruster of the A-ring had been venting fuel while enroute from the recovery area to the Boston Naval Shipyard. Venting of the thruster was observed during spacecraft off-loading and prior to RCS deactivation. No visible toxic vapors were observed coming from any of the remaining thrusters.

Deactivation procedures were completed at 9:30 p.m. e.s.t., November 19, 1966, and the spacecraft was taken by truck to South Weymouth Naval Air Station for airlift by C-130 aircraft to the contractor facility in St. Louis, Missouri, on the same day. Following delivery, the RCS was vacuum dried in an altitude chamber, and the postflight analysis was started.

TABLE 6.3-I.- RECOVERY SUPPORT

Landing area	Maximum access time, hr:min		Support
	Aircraft	Ship	
Launch-site area:			
Pad	0:05		4 LARC (amphibious vehicles) 1 LCU (large landing craft with spacecraft retrieval capabilities)
Land	0:10		1 50-ft MRV (Missile Retrieval Vessel) 2 LVTR (amphibious vehicles with spacecraft retrieval capabilities) 3 M-113 (tracked land vehicles)
Water (if flight crew eject)	0:02		4 CH-3C (helicopters) (3 with rescue teams, 1 command)
Water (if flight crew are in spacecraft)	0:15		
Launch-abort area:			
A-1	4:00	[a]11:00	1 CVS (aircraft carrier) 2 DD (destroyers) 1 AO (oiler) 4 aircraft on station (HC-130H)

[a]Maximum access time with ships positioned along the actual launch ground track.

TABLE 6.3-I.- RECOVERY SUPPORT - Continued

Landing area	Maximum access time, hr:min		Support
	Aircraft	Ship	
Launch-abort area:			
A-2	4:00	[a]50:00	
B	4:00	[a]15:00	
C	4:00	[a]15:00	
D	4:00	[a]50:00	
Primary:			
West Atlantic	1:00	4:00	1 CVS from area A, station 3
Secondary:			
East Atlantic (Zone 2)	(On alert)	6:00	1 AO
West Pacific (Zone 3)	(On alert)	6:00	2 DD, rotating on station
Mid-Pacific (Zone 4)	(On alert)	6:00	1 DD, 1 AO[b]

[a]Maximum access time with ships positioned along the actual launch ground track.

[b]Deployed in this area for logistic purposes; however, it also provided recovery support as required.

TABLE 6.3-I.- RECOVERY SUPPORT - Concluded

Landing area	Maximum access time, hr:min		Support
	Aircraft	Ship	
End-of-mission (60-1A)	1:00	4:00	1 CVS (from West Atlantic zone) 2 P-3A (Air Boss 1 and 2) 4 SH-3A (3 with search equipment and swimmers aboard, and 1 photographic) 3 HC-130 (rescue aircraft)
Contingency	18:00		21 aircraft on alert at staging bases throughout the world (includes 3 shown at end-of-mission)
Total			8 ships, 8 helicopters, 23 aircraft

NASA-S-66-11280 DEC 7

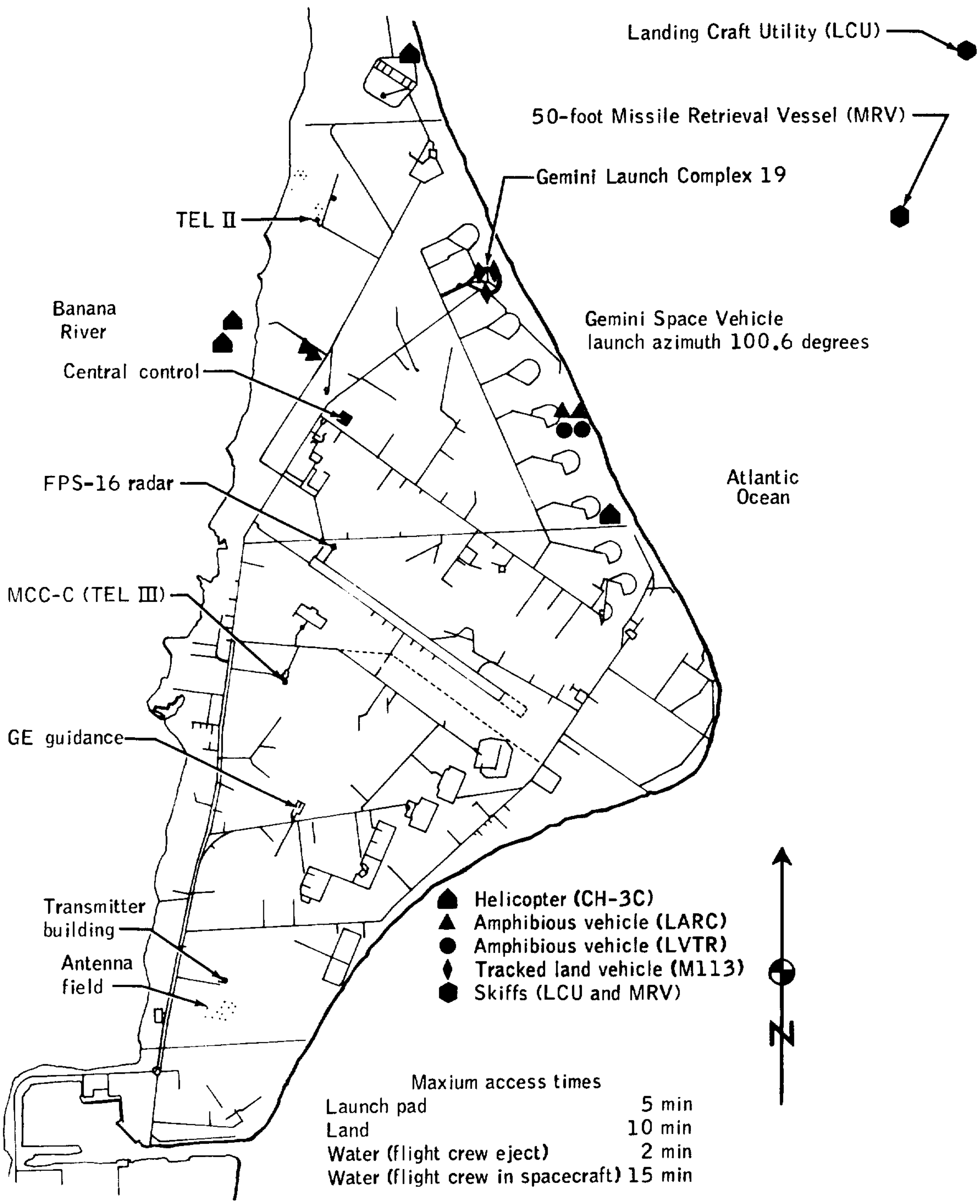


Figure 6.3-1. - Launch site landing area recovery force deployment.

NASA-S-66-11286 DEC 7

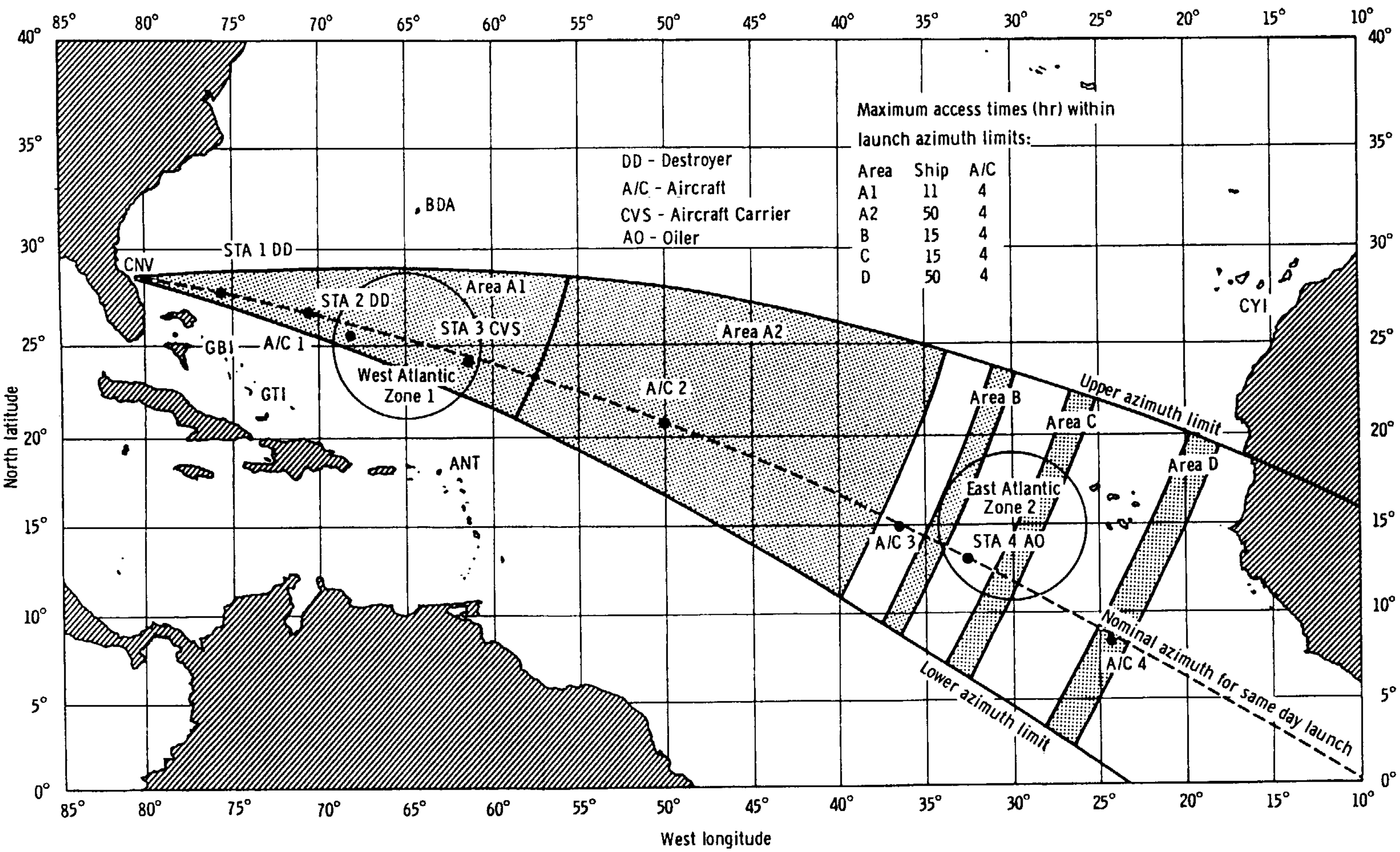


Figure 6.3-2. - Gemini XII launch-abort area with recovery ship and aircraft deployment.

NASA-S-66-11283 DEC 7

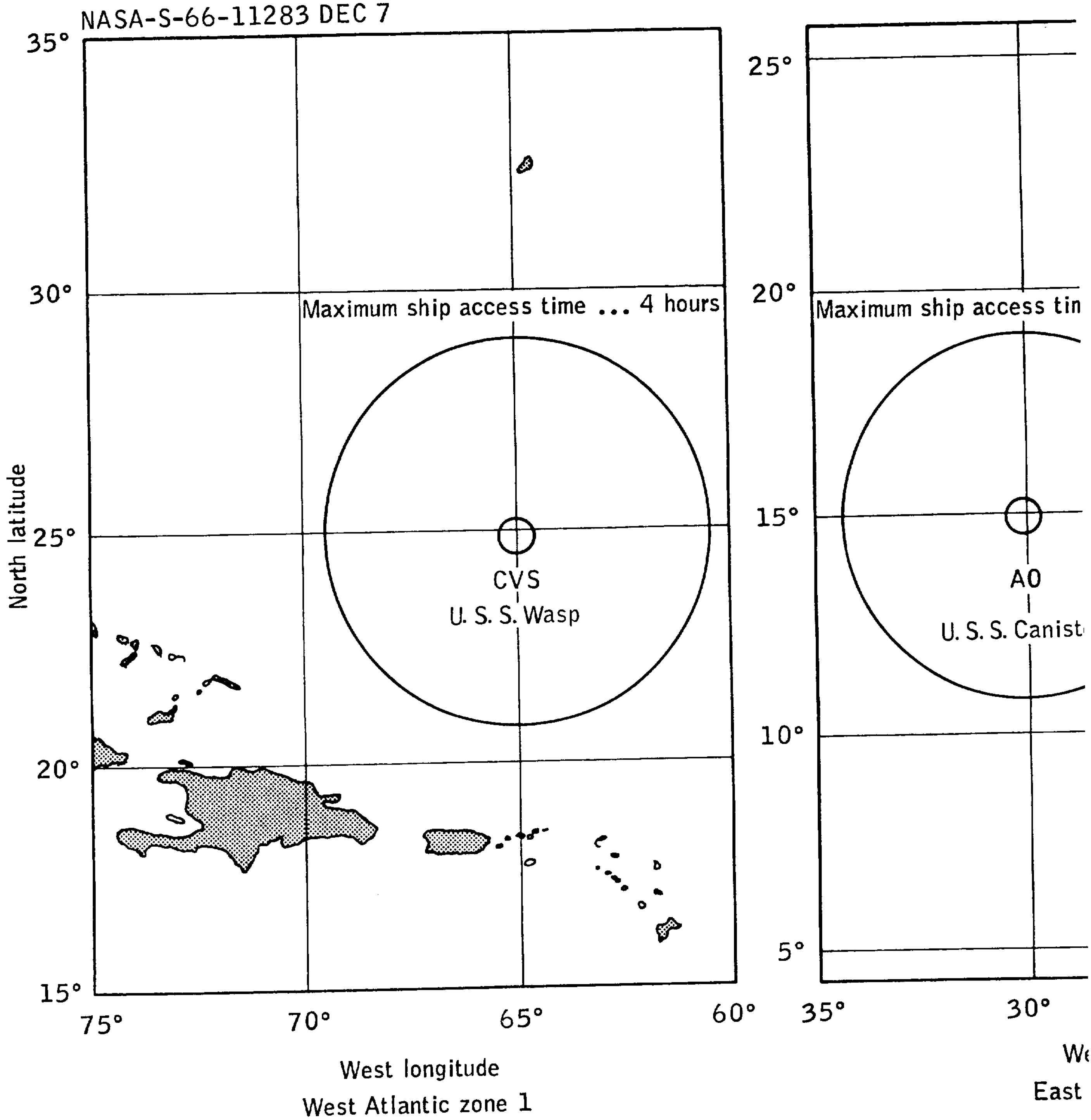
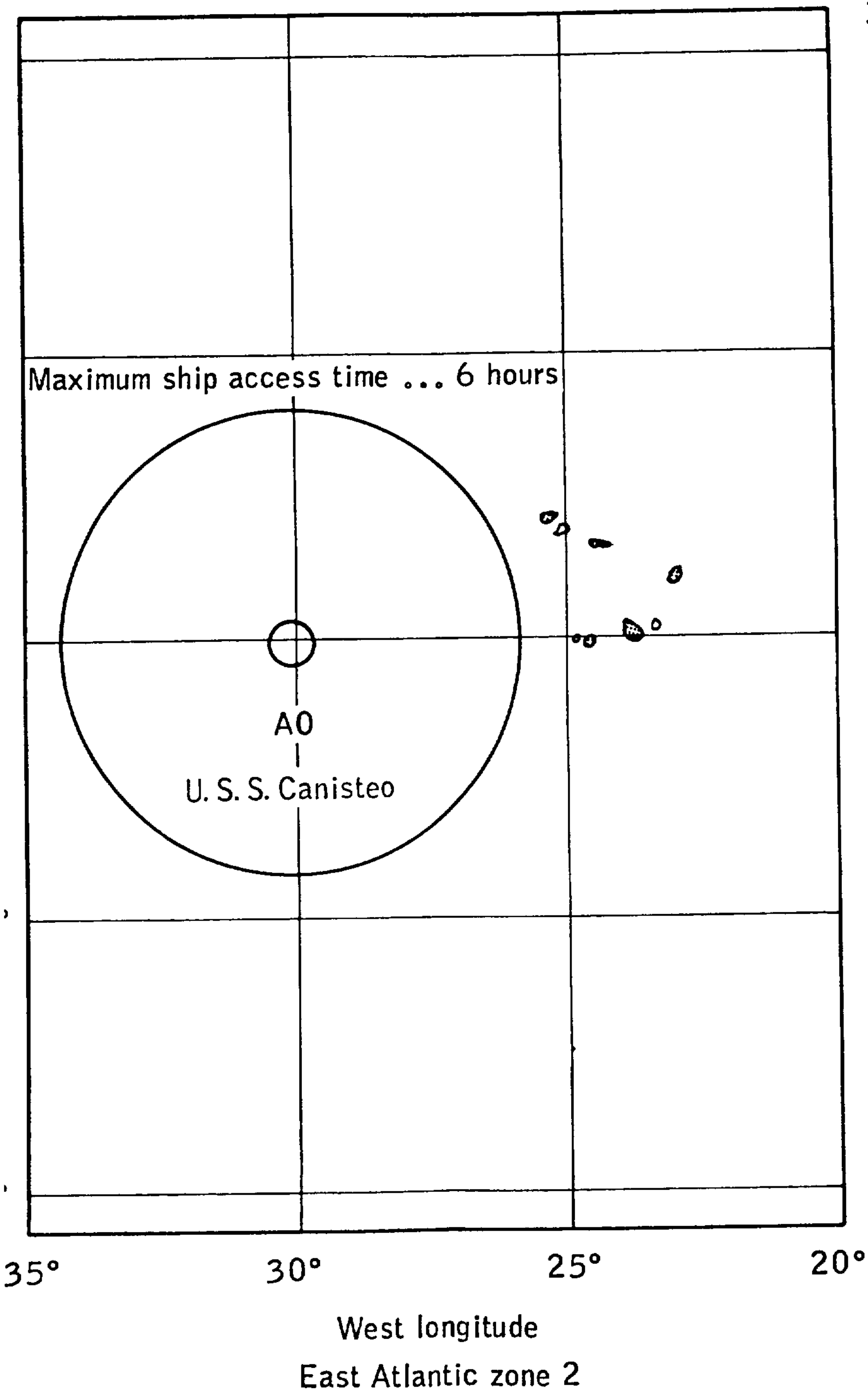
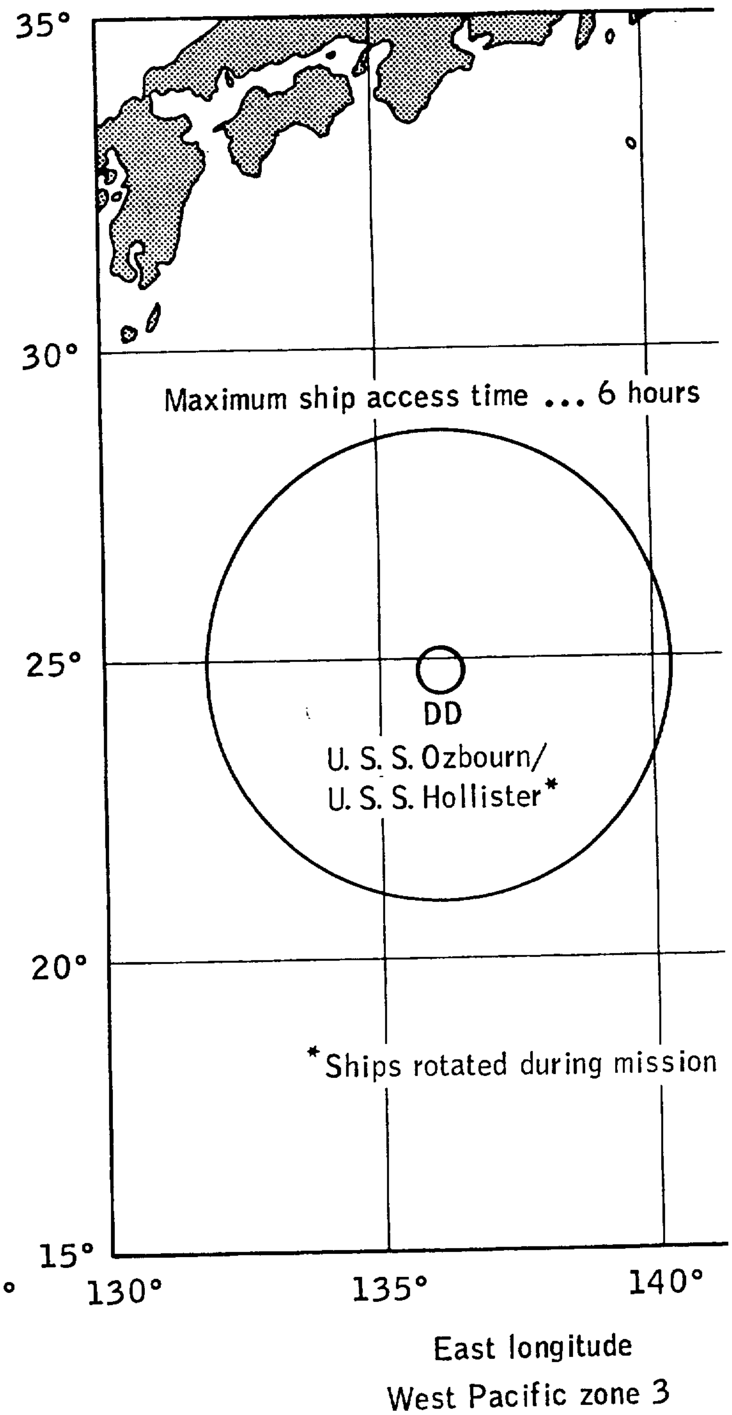
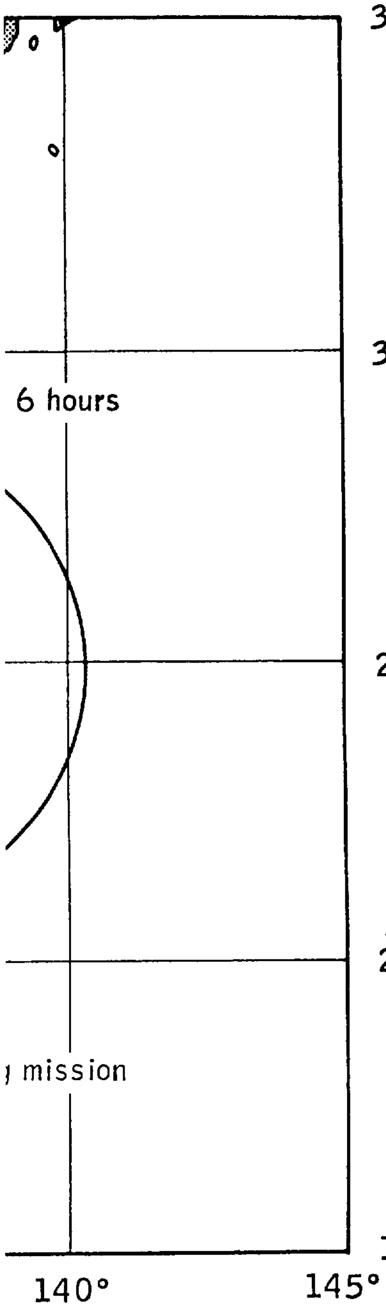


Figure 6.3-3. - Gemini XII landing zone location and force deployment.

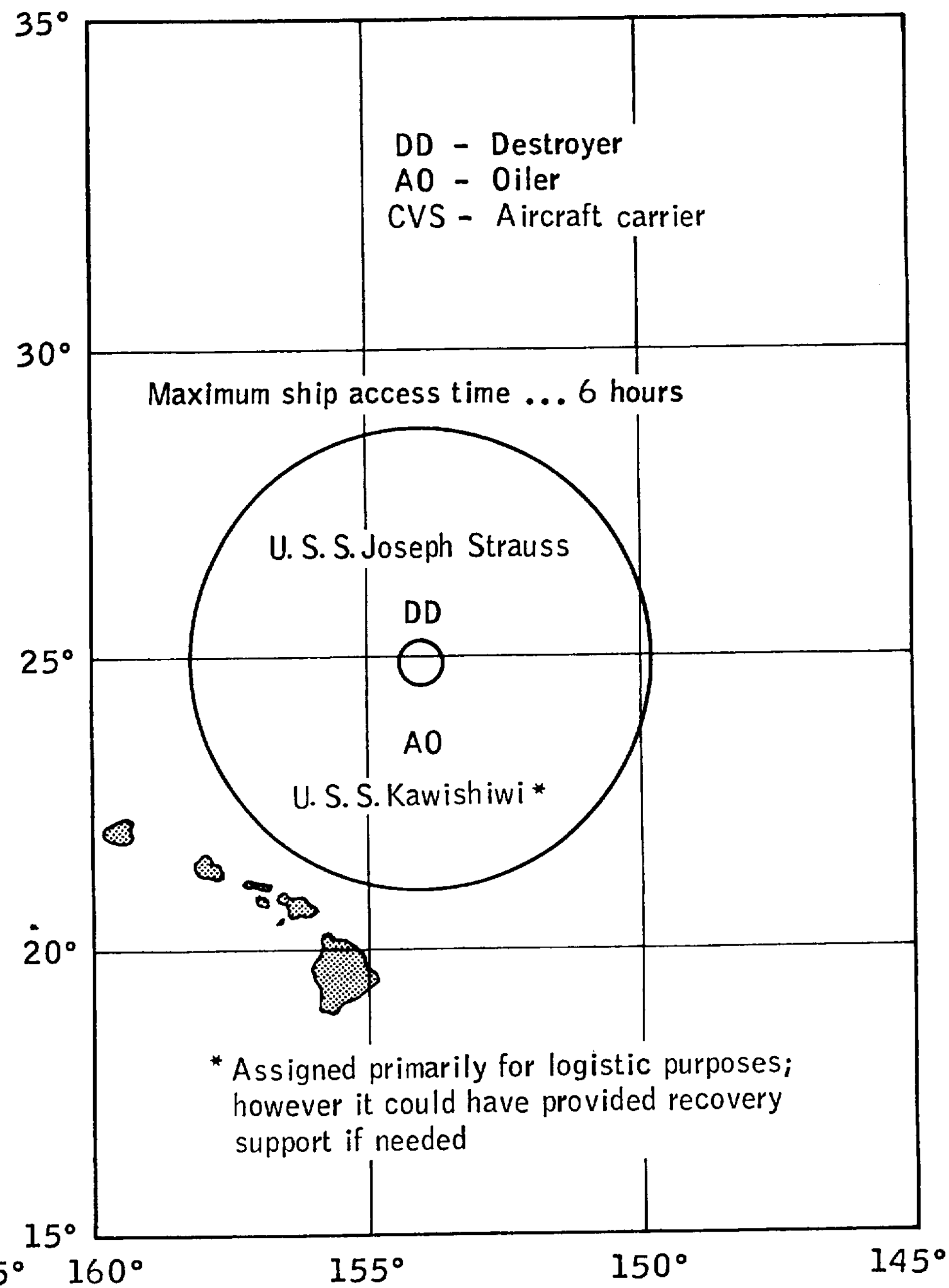

Maximum ship access time ... 6 hours
AO
U. S. S. Canisteo
35°
30°
25°
20°
West longitude
East Atlantic zone 2


35°
30°
25°
20°
15°
Maximum ship access time ... 6 hours
DD
U. S. S. Ozbourn/
U. S. S. Hollister*
*Ships rotated during mission
130°
135°
140°
East longitude
West Pacific zone 3

ment.

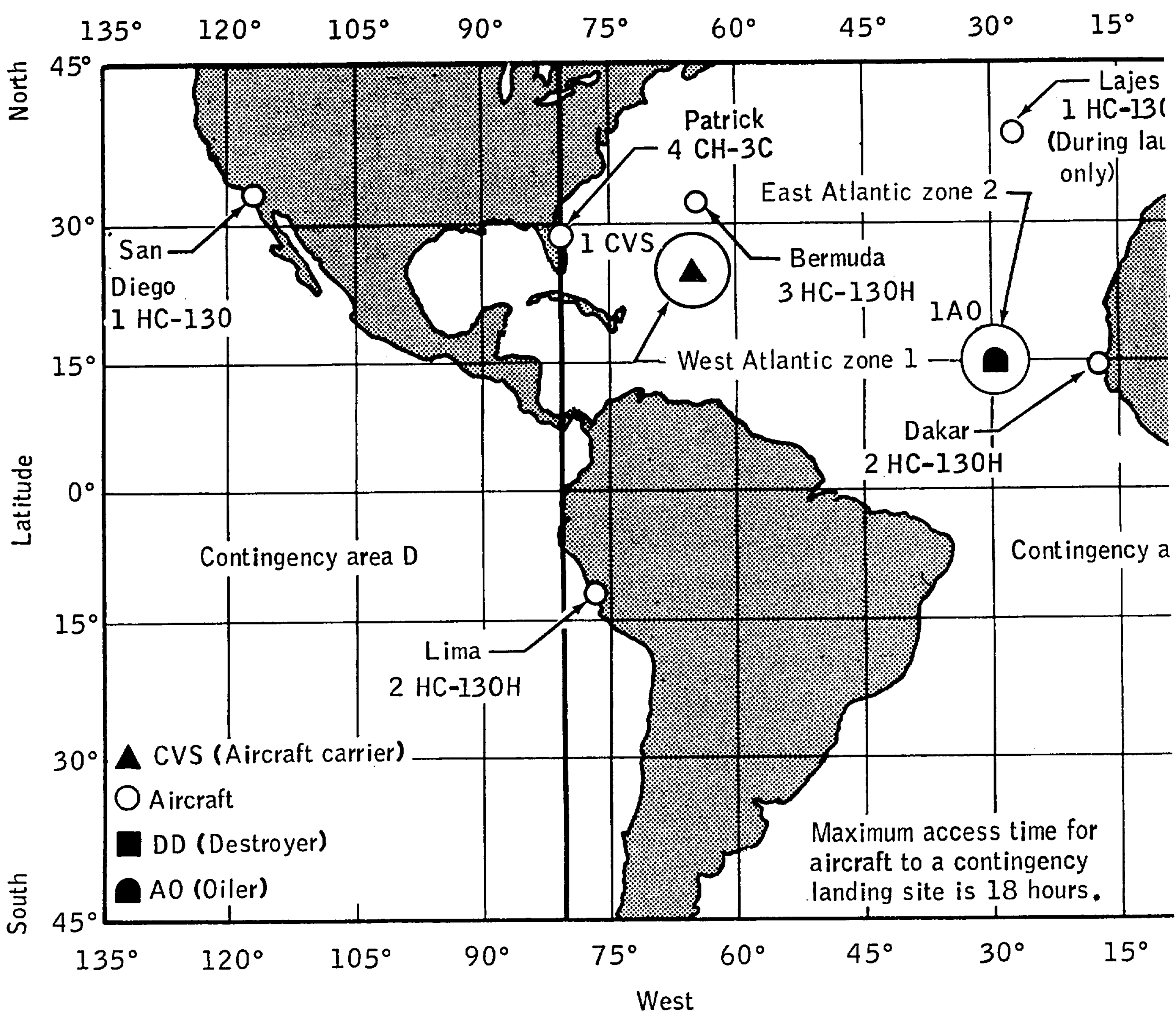


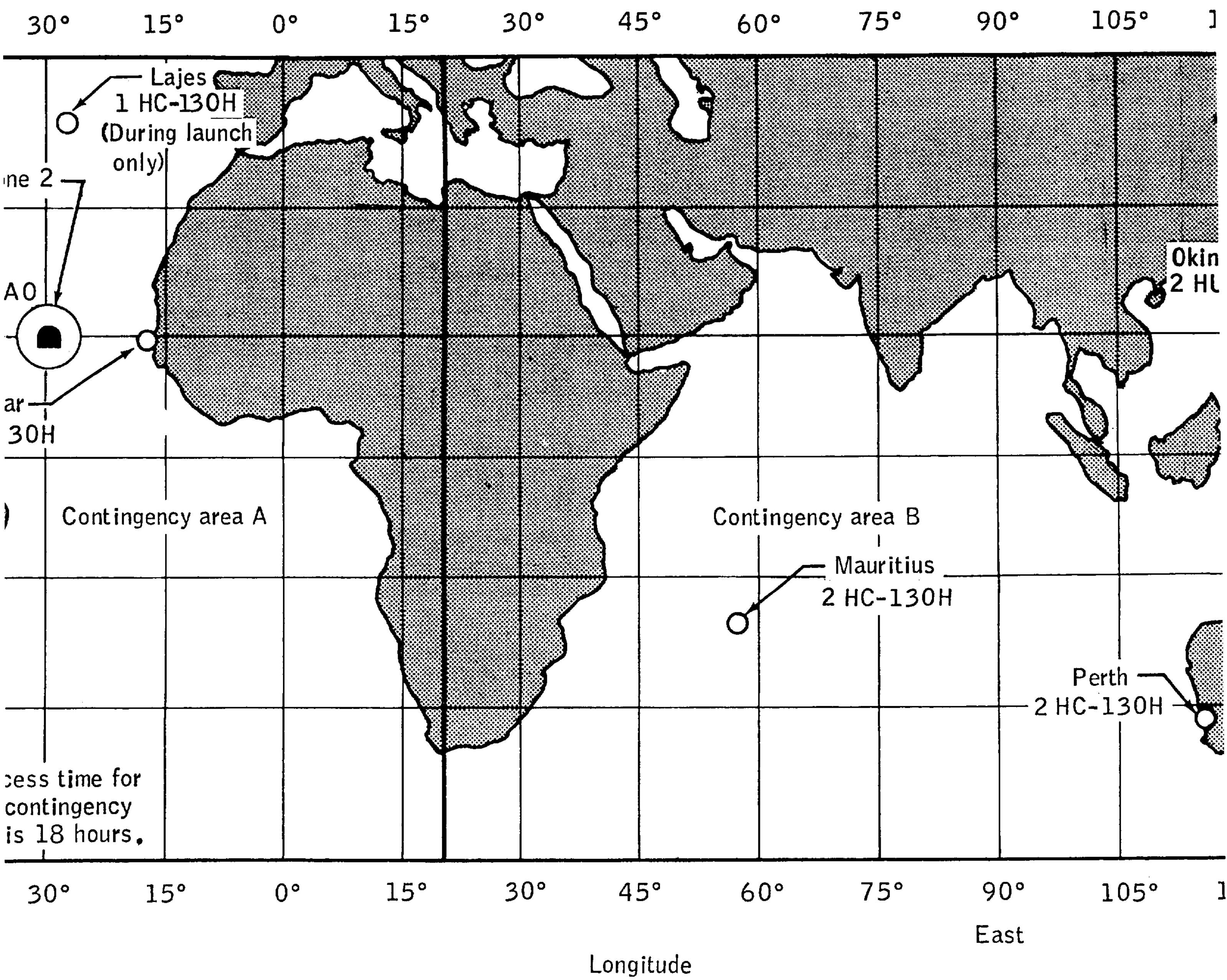
de
ne 3



West longitude
Mid-Pacific zone 4

NASA-S-66-11278 DEC 7





30°
15°
0°
15°
30°
45°
60°
75°
90°
105°
Lajes
1 HC-130H
(During launch
only)
ne 2
A O
ar
30H
Okin
2 HL
Contingency area A
Contingency area B
Mauritius
2 HC-130H
Perth
2 HC-130H
cess time for
contingency
is 18 hours.
East
Longitude

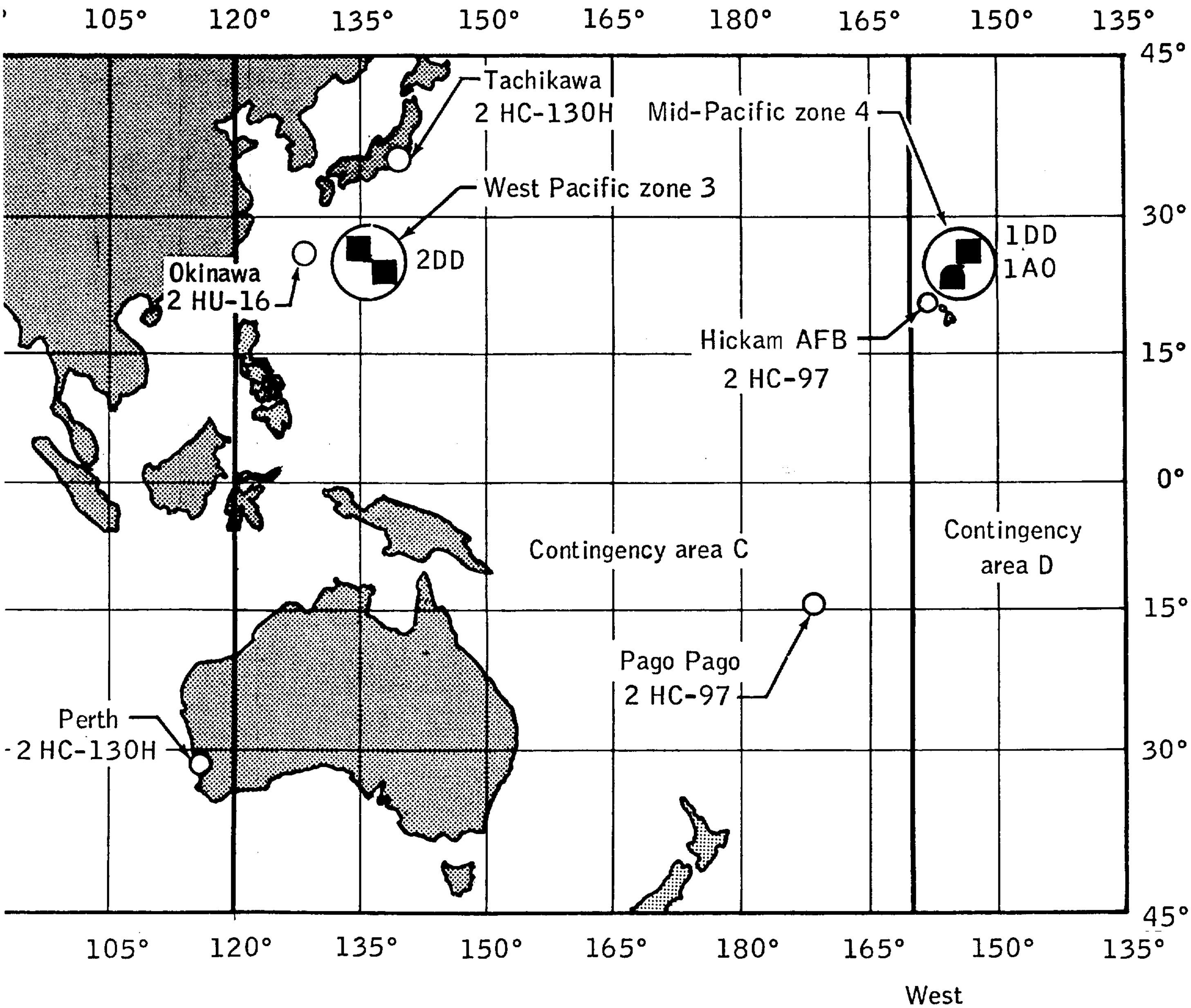


Figure 6.3-4. - Contingency recovery force deployment.

NASA-S-66-11290 DEC 8

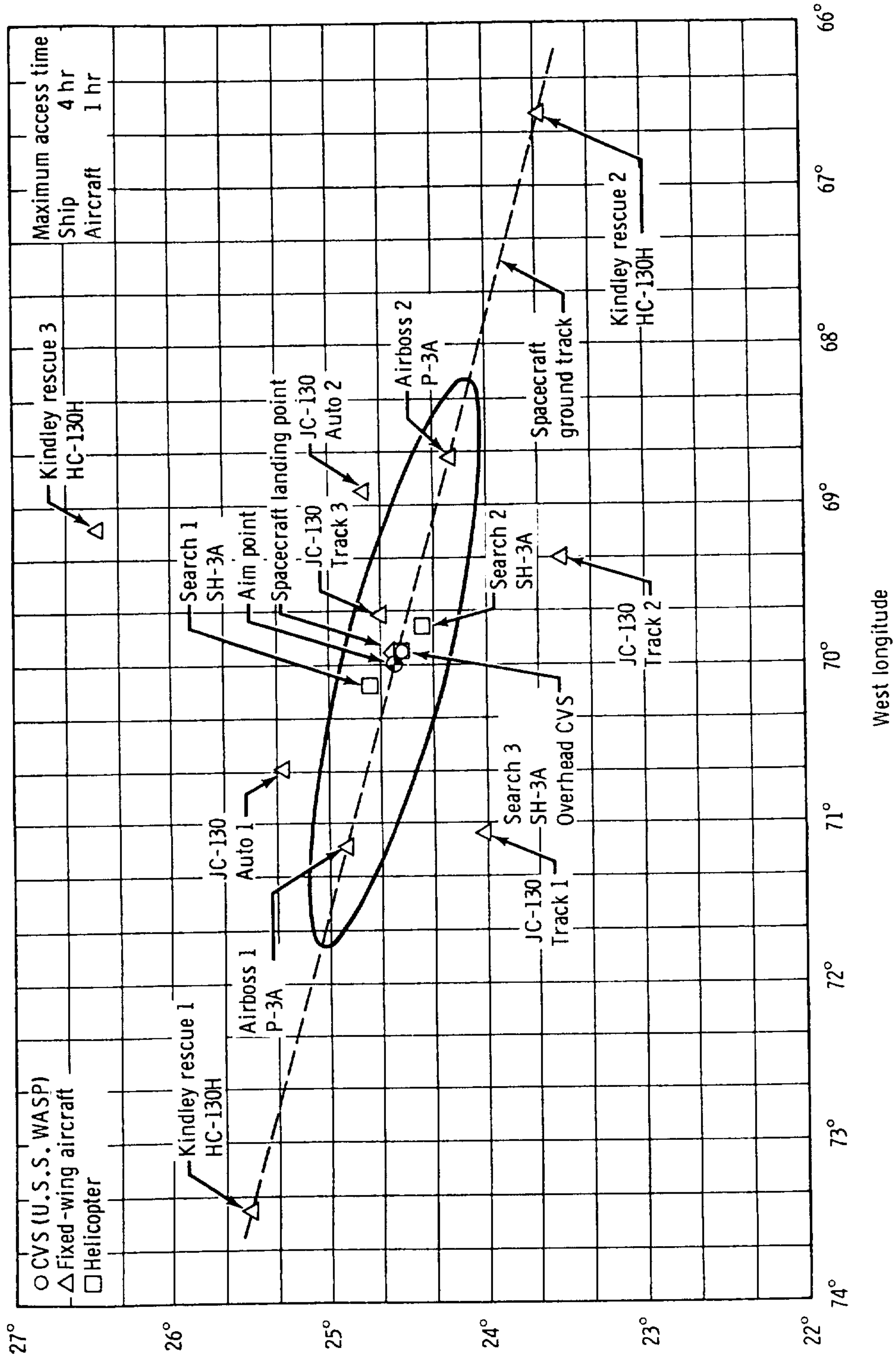


Figure 6.3-5. - Recovery force and network aircraft deployment in primary landing area.

NASA-S-66-11389 DEC 19

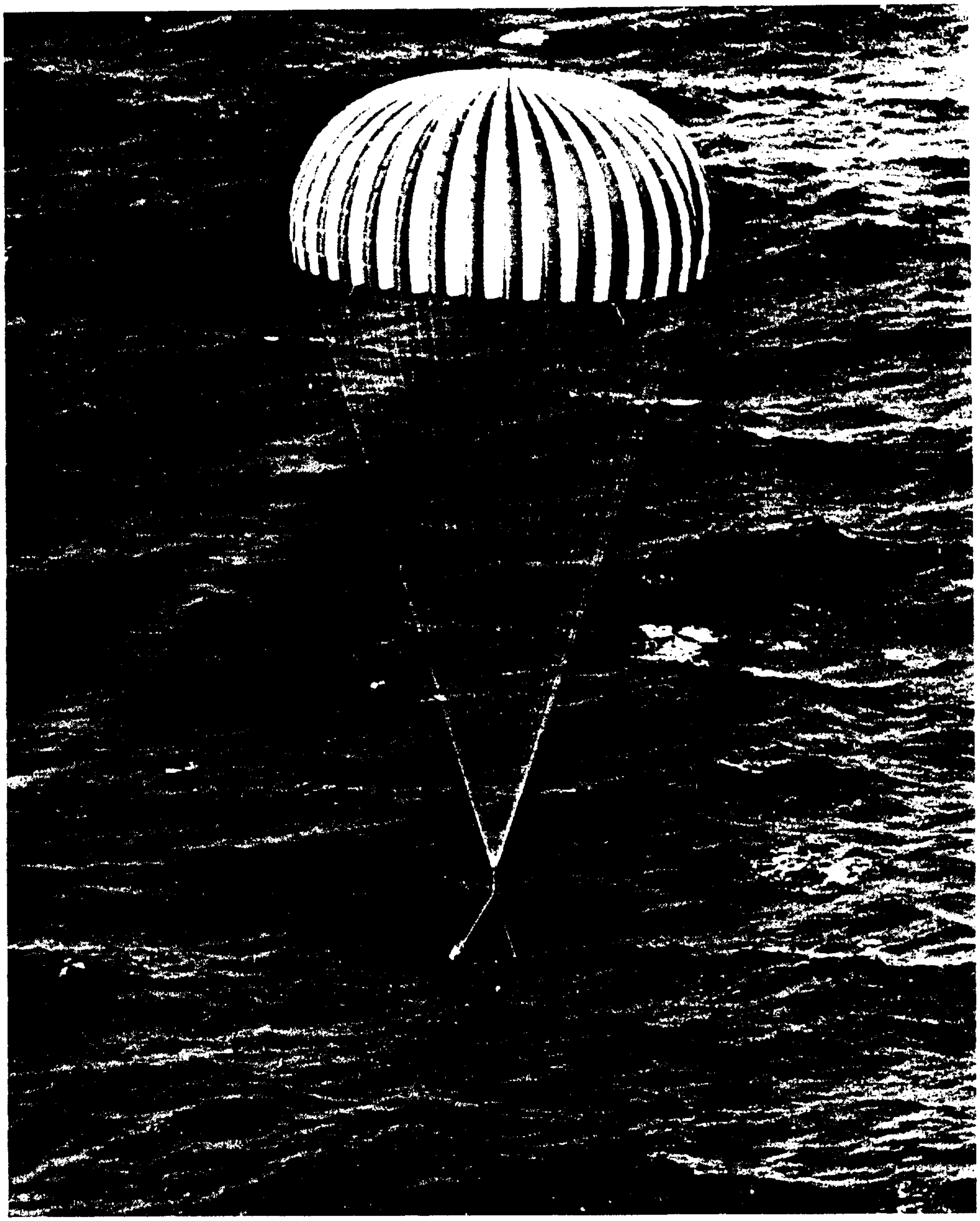


Figure 6.3-6. - Spacecraft just prior to landing.

NASA-S-66-11287 DEC 7

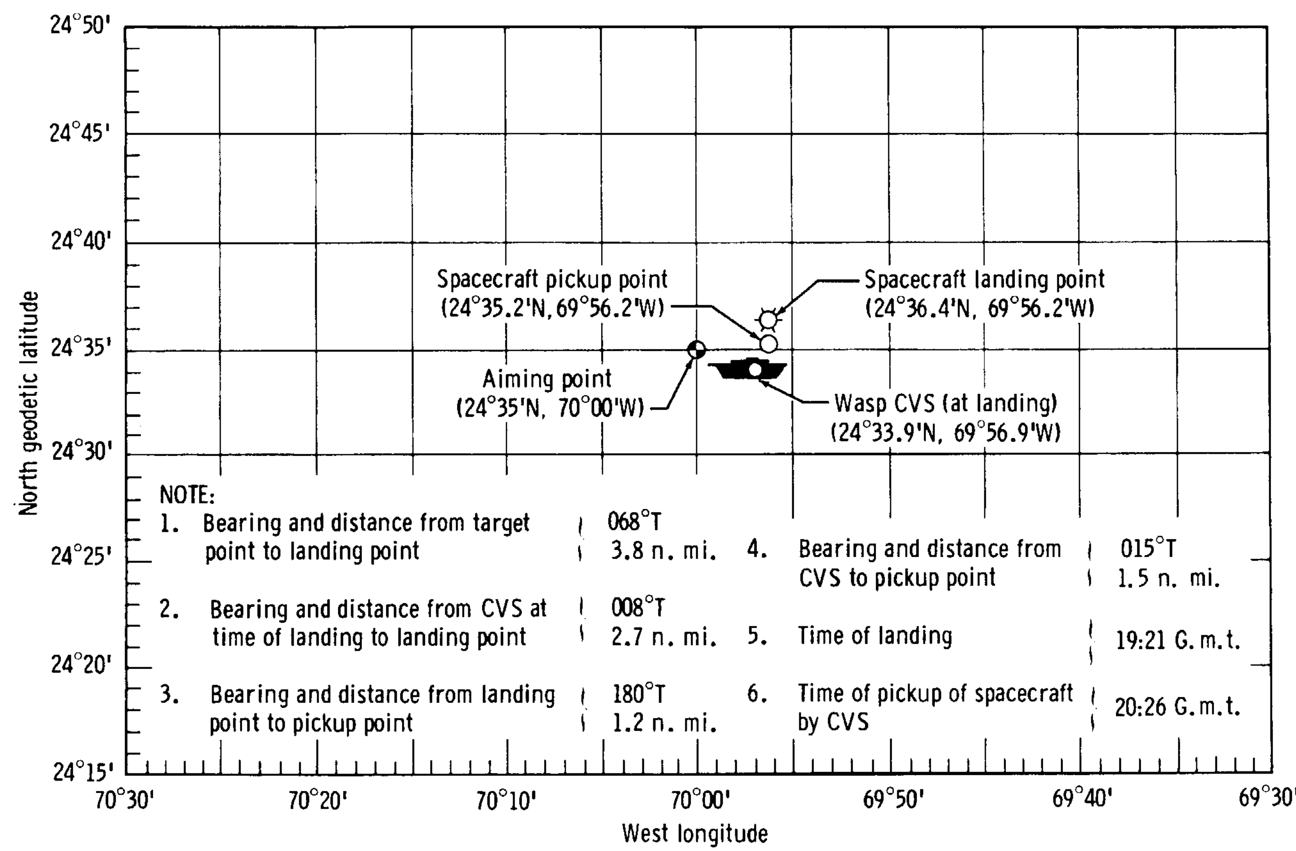


Figure 6.3-7. - Spacecraft landing information, as determined on the prime recovery ship.

NASA-S-66-11388 DEC 19

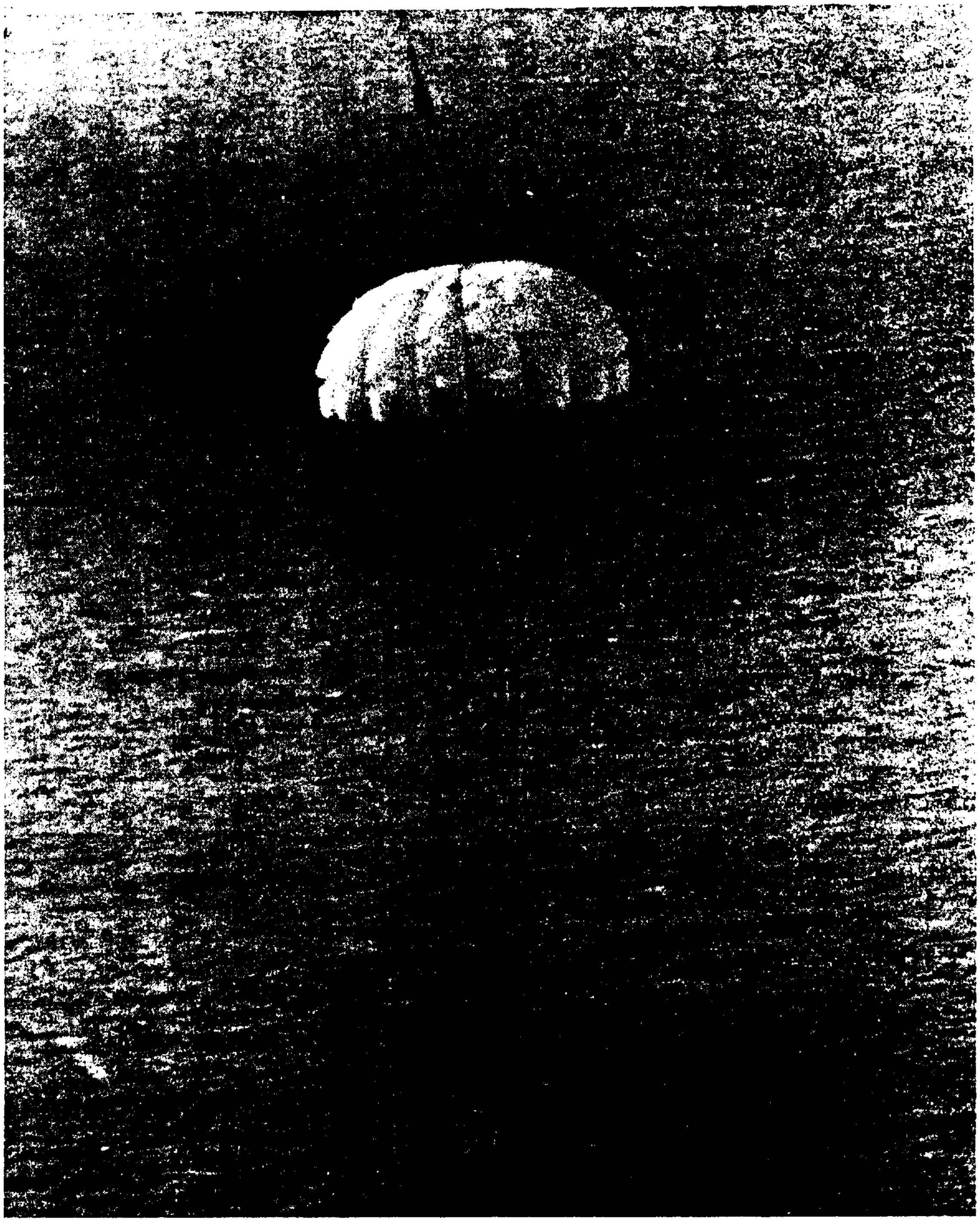


Figure 6.3-8. - Rendezvous and Recovery section just prior to landing.

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7.0 FLIGHT CREW

7.1 FLIGHT CREW PERFORMANCE

7.1.1 Crew Activities

During the Gemini XII mission, the flight crew performed rendezvous, docking, two sequences of standup extravehicular activity (EVA), one umbilical EVA, tether evaluation, docked maneuvers using the secondary propulsion system (SPS) of the Gemini Agena Target Vehicle (GATV), separation from the GATV, eclipse photography, and assigned experiments. The flight was accomplished essentially in accordance with the premission flight plan with the following exceptions:

On the first day of the mission a docking practice was deleted by the crew after they had difficulty during the second docking with the GATV. At the end of the first day, the decision not to fire the GATV primary propulsion system (PPS) eliminated the high-orbit portion of the flight and made it possible to reschedule eclipse observation and photography. The deletion of the high orbit also caused a time slip of the activities for the rest of the mission because of the lower trajectory.

During the second sleep period, failure of a fuel-cell stack resulted in rearranging the flight plan so that the experiments which required electrical power were moved up in the timeline to assure their accomplishment in the event of further deterioration. Figure 7.1.1-1 summarizes the flight plan as accomplished.

7.1.1.1 Prelaunch through rendezvous.- The crew countdown and prelaunch spacecraft checkouts proceeded normally. Crew reports and confirmation of events were received throughout powered flight, and the separation maneuver was successfully accomplished, resulting in a satisfactory orbital insertion. The crew completed the insertion checklist and then performed all required spacecraft maneuvers to effect a rendezvous with the GATV in the third spacecraft revolution (M=3). The nominal Gemini XII rendezvous plan was designed to simulate certain phases of Lunar Module (LM) lunar ascent, with onboard calculation of selected maneuvers occurring prior to terminal phase initiate (TPI). These consisted of an out-of-plane correction and a radar-determined coelliptic maneuver which, in conjunction with other maneuvers, would result in rendezvous near the third apogee. Tables 7.1.1-I and 7.1.1-II summarize the maneuvers as calculated by the ground and/or onboard the spacecraft to effect the rendezvous.

7.1.1.2 Docking.- The first docking was accomplished without difficulty; however, during the second scheduled docking the vehicles did not rigidize, probably because of too low a closing velocity. The crew attempted to initiate the rigidizing sequence by applying a burst of forward thrust while engaging the docking cone, but to no avail. Upon attempting to withdraw the spacecraft it was found that the latches had partially engaged and would not release. The spacecraft had to be maneuvered to accomplish complete unlatching. A second attempt to dock, but with higher closing velocity, resulted in successful rigidizing. A third docking was accomplished during the following daylight period. A scheduled fourth docking was deleted after more fuel than expected was used because of the attempted docking and separation difficulty.

7.1.1.3 Orbital maneuvers.- Two SPS maneuvers were made to accomplish phasing with the eclipse and were performed without any problems. The spacecraft was used to make two maneuvers other than the rendezvous maneuvers. The first was a 6-ft/sec separation maneuver to separate from the GATV after the tether evaluation. The postgrade maneuver was made at 52:14:27 ground elapsed time (g.e.t.) with the crew having some control problems because of thruster degradation. A phasing maneuver was also accomplished after separation from the GATV. Platform alignment for this maneuver required more time than anticipated because of degraded thruster performance that resulted in cross coupling. The phasing maneuver was made with the platform not fully aligned and resulted in a noticeable out-of-plane velocity.

7.1.1.4 Extravehicular activity.- Three separate periods of EVA were successfully completed—two standup EVA's and an umbilical EVA. The first standup EVA, the umbilical EVA, and the second standup EVA occurred on the second, third, and fourth days of the flight, respectively.

Preparation for the first standup EVA proceeded as planned, with the suit integrity checks being completed at 19 hours 19 minutes g.e.t. At 19 hours 20 minutes g.e.t., the pilot reported that he suspected that his right microphone was inoperative but was unable to confirm a failure. The cabin was depressurized and the hatch was opened at 19 hours 29 minutes g.e.t. Following sunrise at 20 hours 14 minutes g.e.t., the pilot successfully completed all activities planned for that day.

The pilot reported that considerable effort was required to operate the shutter release for the S013 experiment for long time exposures. The pilot found that it took much longer than he had expected for his eyes to become dark adapted in order to acquire the star patterns for the S013 experiment. The command pilot, inside the spacecraft and shielded from the sunlight glare off the white external surfaces, was able to dark adapt his eyes much sooner. After successful completion of the

S013 experiment, the hatch was closed following sunrise and the crew began repressurizing the cabin at 22 hours 2 minutes g.e.t.

The crew reported that preparation for umbilical EVA was in progress at 41 hours 30 minutes g.e.t. The EVA preparation went as planned, and, following sunrise at 42 hours 47 minutes g.e.t., the hatch was opened. At 42 hours 49 minutes g.e.t., the pilot reported that the umbilical tether hook had come loose from the egress bar. The hook was the large flange type and the pilot was able to reconnect the hook to the egress bar. Following the EVA camera installation, the pilot moved across the telescopic handrail to the TDA area, where he attached the GATV tether to the spacecraft docking bar and successfully completed all other planned tasks. The pilot then had time for a waist-tether-dynamics evaluation during which he was able to deploy the S010 experiment. At 43 hours 22 minutes g.e.t., he returned to the spacecraft hatch area and, after a 2-minute rest, he obtained the adapter work station camera and proceeded to the adapter section. At 43 hours 29 minutes g.e.t., his feet were positioned in the foot restraints and he began the adapter work station camera installation. During the camera installation, he reported a failure in the camera bracket handle ball-detent linkage. However, the pilot was able to insert his gloved finger into the linkage below the bracket handle and operate the detent ball and complete the camera installation. (On investigation of the failure after ingress, the crew reported a shear-sheared or missing pin in the linkage as the cause.) After an evaluation of the foot restraints, the pilot unstowed the penlights and reported that one of the lights appeared to have been overheated and was disfigured. One end of the light had expanded to almost twice its normal diameter, but the penlight would operate. Sunset occurred at 43 hours 44 minutes g.e.t. and the pilot proceeded with the work station tasks. During the torquing operation with the adapter torque wrench, the pilot reported that the torque needle display was erroneous. Being familiar with this failure from training, he was able to apply a zero-shift correction and complete the task. Following completion of several cutting operations with cutters ters, stowed in the work-station pouch, the pilot reported that the rubber strap on the Saturn bolt appeared to have been overheated and was stuck to the Saturn bolt housing. He was able to remove the strap with a pip-pin stowed on the adapter work station. In the process of removing the rubber strap, the Saturn bolt and its washer floated free. The pilot was able to maneuver them into an area where he could catch the bolt in one hand and the washer in the other. This took place while the pilot was operating on the waist tethers with his feet free of the foot restraints. He went on to insert the bolt into the washer and then into the bolt housing. He then proceeded to torque the bolt until it was tight, successfully completing the task. The pilot completed the other work station tasks successfully and, following sunrise at 44 hours 17 minutes g.e.t., he retrieved the adapter work station camera and proceeded to the spacecraft hatch area. He installed the EVA camera and moved to the Target Docking Adapter (TDA) area where he hooked up his waist tether and proceeded with the TDA work station task. While operating in the TDA area, he found that the portable handholds tended to move when grasped for body control. He later reported the problem as related to the low adhesive force of the Velcro used. He also found that considerable effort was required in grasping the rotating pip-pins for body control or waist tether attach points. Other handholds and pip-pins were used successfully. The pilot completed all tasks at the TDA as planned, and afterwards, was able to pass the Apollo torque wrench through a small loop in his tether to the spacecraft and bring the wrench back to the cabin, which was not a required task. At 44 hours 44 minutes g.e.t., he proceeded to clean the command pilot's window with a cloth stowed in a pocket located in the suit knee area. Following successful completion of all planned tasks for the EVA and operating on a timeline that never varied more than a few minutes from one determined during training prior to the flight, the pilot took part in a thruster firing evaluation requested by the ground controllers. After observing the thruster firings, the pilot began ingress and the hatch was closed at 44 hours 55 minutes g.e.t. The crew began repressurizing the cabin at 44 hours 57 minutes g.e.t. During the cabin repressurization, the command pilot reported that his eyes were burning. The problem disappeared after the cabin was repressurized and the command pilot later reported that he thought sweat drops had been the cause. The pilot successfully accomplished all his assigned EVA activities without excessive tiring or overheating.

Preparation for the second standup EVA proceeded as planned. Following completion of the suit integrity checks, the cabin was depressurized and the hatch opened at 66 hours 8 minutes g.e.t. The pilot jettisoned all items as planned and began taking pictures following sunset at 66 hours 19 minutes g.e.t. Prior to sunrise, the command pilot began orienting the spacecraft for the sunrise photography. Following successful completion of the sunrise photography, the pilot was inside the spacecraft at 67 hours 1 minute g.e.t.

7.1.1.5 Experiments and scientific observations.- Fifteen experiments were assigned to this mission. One of the experiments, M407 Lunar Ultraviolet Spectral Reflectance, was deleted prior to the mission because of a poor lunar phase. Two additional scientific observations were performed by the crew—the Sunrise Ultraviolet Photography experiment (formerly Experiment S064) was added just prior to the mission, and eclipse photography was added during the mission. All the experiments were performed, with some data being obtained for every experiment. Twelve of the experiments had been performed on other Gemini missions and three were new experiments.

7.1.1.5.1 Experiment D010, Ion-Sensing Attitude Control: For the D010 experiment, 30 hours of data were obtained, ten hours of which were with the platform. Six modes of operation were planned and all were completed. In addition to the normal data acquisition, some data were also obtained during the preretrofire preparation with the spacecraft blunt end forward (BEF). This experiment could have been seriously degraded by the control problem; however, the command pilot was able to overcome the problem and successfully complete the experiment.

7.1.1.5.2 Experiment M405, Tri-Axis Magnetometer: The M405 experiment was turned on at approximately 4 hours 30 minutes g.e.t. and turned off at approximately 16 hours g.e.t. The experiment was turned on for two more periods—from 27 hours 30 minutes to 38 hours 30 minutes g.e.t. and from 75 hours 20 minutes to 84 hours 45 minutes g.e.t. Good data were obtained throughout the mission.

7.1.1.5.3 Experiment M408, Beta Spectrometer: The M408 experiment was turned on and off at the same times during the mission as the M405 experiment. A controlled roll maneuver was made through the South Atlantic Anomaly region at approximately 5 hours g.e.t. Good data were obtained for a period of about five minutes during this maneuver. The experiment equipment failed after that and no more data were obtained throughout the remainder of the mission.

7.1.1.5.4 Experiment M409, Bremsstrahlung Spectrometer: The M409 experiment was turned on and off at the same times during the mission as the M405 experiment. Good data were obtained throughout the mission.

7.1.1.5.5 Experiment S003, Frog Egg Growth: Crew activities in this experiment consisted of turning on a heater switch at 17 hours 41 minutes g.e.t., turning a handle to fix Unit I at 41 hours 43 minutes g.e.t., and turning a second handle to fix Unit II at 85 hours 10 minutes g.e.t. The experiment was performed as planned. In addition to the eggs, five live tadpoles were recovered in the experiment package.

7.1.1.5.6 Experiment S005, Synoptic Terrain Photography: The crew took a total of 397 photographs with the 70-mm cameras. Of these, 180 photographs are usable for the S005 experiment. Some S005 updates were provided to the crew during the mission, but most of the photographs were made on crew initiative.

7.1.1.5.7 Experiment S006, Synoptic Weather Photography: Of the 397 photographs taken, 291 were usable for the S006 experiment. Most of the S006 photographs were made on crew initiative rather than from ground updates.

7.1.1.5.8 Experiment S010, Agena Micrometeorite Collection: The S010 experiment package was located on the GATV. At 43 hours 12 minutes g.e.t., during the umbilical EVA, the pilot reported that he had deployed the experiment. The S010 package was left on the GATV for possible retrieval during a later mission.

7.1.1.5.9 Experiment S011, Airglow Horizon Photography: The S011 experiment was planned for three night passes, with 12 photographs to be made during each night pass. During preparation for the first night pass, the crew reported control difficulty while docked. They also reported a problem with the camera shutter during the first night pass. In addition, there was a spacecraft control problem during the undocked portion of the experiment. In spite of these problems, the crew obtained nine good photographs during the first night pass, eleven during the second night pass, and three during the third night pass.

7.1.1.5.10 Experiment S012, Micrometeorite Collection: The S012 package was opened, closed, and locked by Digital Command System (DCS) commands from the ground. The only crew participation consisted of retrieving and stowing the package. This activity was performed during the first standup EVA at 20 hours 26 minutes g.e.t.

7.1.1.5.11 Experiment S013, Ultraviolet Astronomical Camera: This experiment was performed during the two night passes of the first standup EVA. Spectrographs of three starfields were to be made during each night pass. Because of the difficulty in maneuvering the spacecraft/GATV combination, only two starfields per night pass were photographed. The crew made a total of 27 spectrographs of which 25 are usable for spectrographic analysis of stars.

7.1.1.5.12 Experiment S029, Libration Regions Photographs: The S029 experiment was performed at approximately 27 hours 12 minutes g.e.t. and at 74 hours 10 minutes g.e.t. The crew had trained to photograph the L_5 libration region, but the change in launch date made it necessary to change to the L_4 libration region. In spite of the lack of preparation, the experiment was properly performed.

7.1.1.5.13 Experiment S051, Daytime Sodium Cloud: The sodium cloud photography experiment was performed at 62 hours 47 minutes g.e.t. and 64 hours 18 minutes g.e.t. The experiment was performed as planned; however, the 70-mm general-purpose camera shutter was stuck open, ruining all the S051 exposures. The crew did not see the cloud on either pass. Twelve photographs were attempted on the first pass and eight on the second.

7.1.1.5.14 Experiment T002, Manual Navigation Sightings: Five night passes were allotted in the flight plan for the T002 experiment.

The experiment was performed as planned in spite of the spacecraft control problem.

7.1.1.5.15 Sunrise ultraviolet photography: At 67 hours 37 minutes g.e.t., the crew reported that the experiment was performed as planned, although the sunrise photography was difficult. Exposures were made of night-time starfields and of daytime starfields. Two of the sunrise photographs may be usable, but because of static electricity the rest were definitely fogged beyond use.

7.1.1.5.16 Eclipse photography: This observation was planned and executed after the PPS firing to high-apogee orbit was deleted. Four exposures were made during totality with the 70-mm general-purpose camera, two exposures were made with the 70-mm superwide-angle camera, and two series of exposures were made with the two 16-mm sequence cameras. The photographs made with the general-purpose camera were too overexposed to provide any useful information. The sequence camera photography gave useful data as to time of totality. The experiment was performed at approximately 16 hours 1 minute g.e.t.

7.1.1.6 Retrofire and reentry.- Platform alignment prior to retrofire was accomplished primarily with the Orbital Attitude and Maneuver System (OAMS), with final corrections and attitude hold being accomplished on the A-ring of the Reentry Control System (RCS). The regulated pressure in the RCS A-ring was high because of an apparent regulator malfunction, and the ground controllers advised the crew to use the system to keep the pressure within limits.

Retrofire and reentry were performed using automatic control, with the crew monitoring systems performance, being prepared to take over in the event of a malfunction. The spacecraft was initially flown heads down during reentry, followed by some roll to reduce lift.

The backup bank angle of 46 degrees left was rolled in and maintained until guidance initiate, which appeared to be a little late. After guidance initiate, the automatic reentry mode was selected, and the spacecraft started its automatic reentry. The second RCS ring was switched on just prior to the maximum deceleration period, and the spacecraft flew nominally to 80K feet, at which time the rate command mode of control was selected. Some small oscillations were damped out, and at the 50K-foot altitude the drogue was deployed and the control system shut down. The descent to 10.6K feet was normal, and at that point the main parachute was deployed. The descent on the main parachute was also normal with no noticeable oscillations.

7.1.1.7 Landing and recovery.- The landing was very hard. The spacecraft touched down on the side of a swell with the nose pointed up the swell. The sea state at the time was reported to be 4-foot swells with 2-foot waves. The spacecraft sank below the surface and pitched slightly forward. The parachute was released and collapsed downwind and clear of the spacecraft. Some water was taken into the spacecraft, probably through the vent valve. There were also light fumes of burned paint and metal in the cockpit. The cockpit was sealed and the repressurization valve opened to build up a positive pressure against any possible leaks.

Communication was good. The HF antenna was deployed and contact was made with Cape Kennedy. The swimmers used the interphone to notify the crew that the collar installation would take longer because of the rough seas.

Egress was normal except that the HF antenna had not been retracted. One crewman had to return to the spacecraft to retract the HF antenna so that the helicopter could pick up the crew.

7.1.1.8 Mission training and training evaluation.- Flight crew training was accomplished as shown in the Gemini XII Mission Training Plan. In addition to this, the command pilot had trained as backup pilot for the Gemini IV mission and as backup command pilot for the Gemini IX mission. He trained and flew as prime pilot on the 14-day Gemini VII mission. The pilot had trained as backup pilot for the Gemini IX mission. Table 7.1.1-III contains a summary of crew training for the Gemini XII mission.

The Gemini Mission Simulator and the Rendezvous Simulator were used to train for the M=3 rendezvous. The Translation and Docking Trainer and the Gemini Mission Simulator were used to practice docking and station-keeping maneuvers. The Dynamic Crew Procedures Trainer was used to practice launch-abort techniques and the tether evaluation. The crew made extensive use of the Gemini mockup, the zero-g aircraft, and the underwater zero-g facilities to practice and develop EVA procedures. The 100-percent completion of the three EVA's during the flight clearly shows the value and worthiness of this training, especially that accomplished in the underwater zero-g facility. This was the first mission for which the complete EVA timeline was accomplished in a simulated zero-g field.

Crew performance during the mission showed that they were well trained to accomplish the very strenuous flight plan. Rendezvous was accomplished using an unusually small amount of fuel even though a failure in the radar system caused the crew to use backup procedures for the terminal phase maneuvers. The gravity-gradient tether evaluation was successful. The crew completed all major objectives of the mission and, in addition, successfully took the first pictures from space of a total solar eclipse. They completed a large portion of the planned experiments. After four days in space, they landed in the Atlantic Ocean after a very accurate automatic reentry.

The crew utilized the facilities of the Moorehead Planetarium to train for experiments, meeting there with the astronomical experimenters to train in star aiming to achieve the desired results.

TABLE 7.1.1-I.- PRE-TRANSFER RENDEZVOUS MANEUVER COMPUTATIONS

[All values in ft/sec]

Maneuver	Ground-computed	Onboard-computed	Applied
Insertion Velocity Adjust Routine		28 fwd, 9 lt	Onboard
Phase adjust	66.6 fwd (0,0,0 ref)		Ground
Plane change[a]	7.4 rt (0,0,0 ref)	8.5 rt (0,0,0 ref)	Onboard
Corrective combination	6.7 fwd 3.5 up 1.1 rt (0,0,0 ref)		Ground
Coelliptic	49.8 fwd 3.5 up 0.7 rt (Boresighted)	49.5 fwd 6.5 up 0.1 rt (Boresighted)	Onboard

[a]Performed with a yaw-right of 26 degrees, which eliminated the retrograde correction normally required after a maneuver using the lateral thrusters.

TABLE 7.1.1-II.- TERMINAL PHASE MANEUVERS

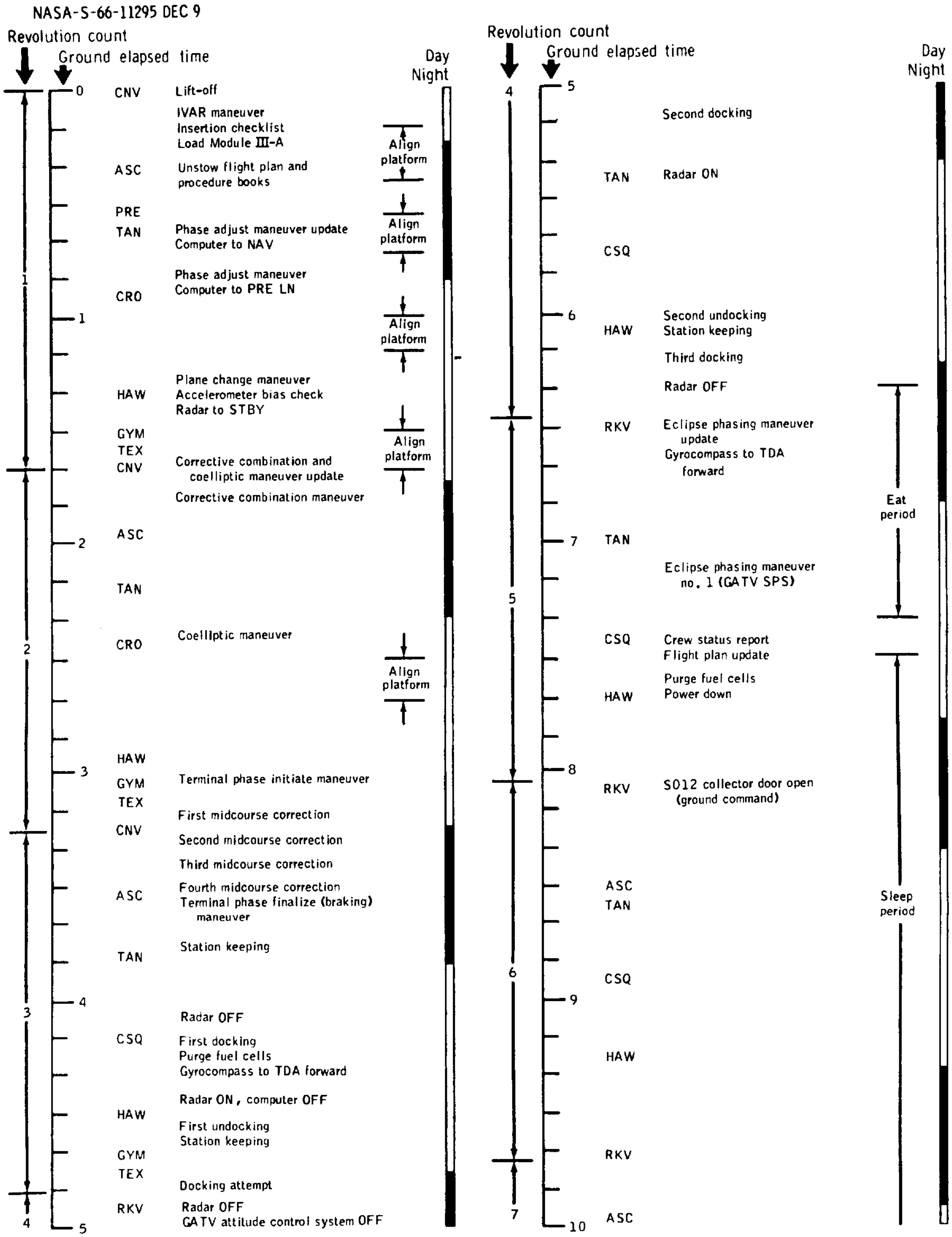
[All values in ft/sec, based on a boresighted spacecraft]

Maneuver	Onboard backup solution	Intended to apply
TPI[a]	22 fwd, 3 up	22 fwd
First midcourse correction	1.5 up	0
Second midcourse correction	2 up	2 up
Third midcourse correction	0	1 lt
Fourth midcourse correction	5 aft, 1 dn	5 aft, 1 dn

[a]Ground-computed solution was 22.8 ft/sec forward, 3.2 ft/sec up, and 2.7 ft/sec right.

TABLE 7.1.1-III.- CREW TRAINING SUMMARY

Activity	Training time, hr:min	
	Command pilot	Pilot
System briefings	40.45	40:45
Spacecraft tests	68:15	75:15
Gemini Mission Simulator	153:20	135:00
Rendezvous Simulator	89:00	89:00
Dynamic Crew Procedures Simulator	15:40	3:30
Translation and Docking Trainer	7:00	1:45
EVA	70:00	79:00
Zero-g (aircraft)	6:00	14:45
Zero-g (underwater)	4:00	32:00
Planetarium	28:00	28:00
Experiments	24:30	32:30



(a) 0 to 10 hours g.e.t.
Figure 7.1.1-1.- Summary flight plan.

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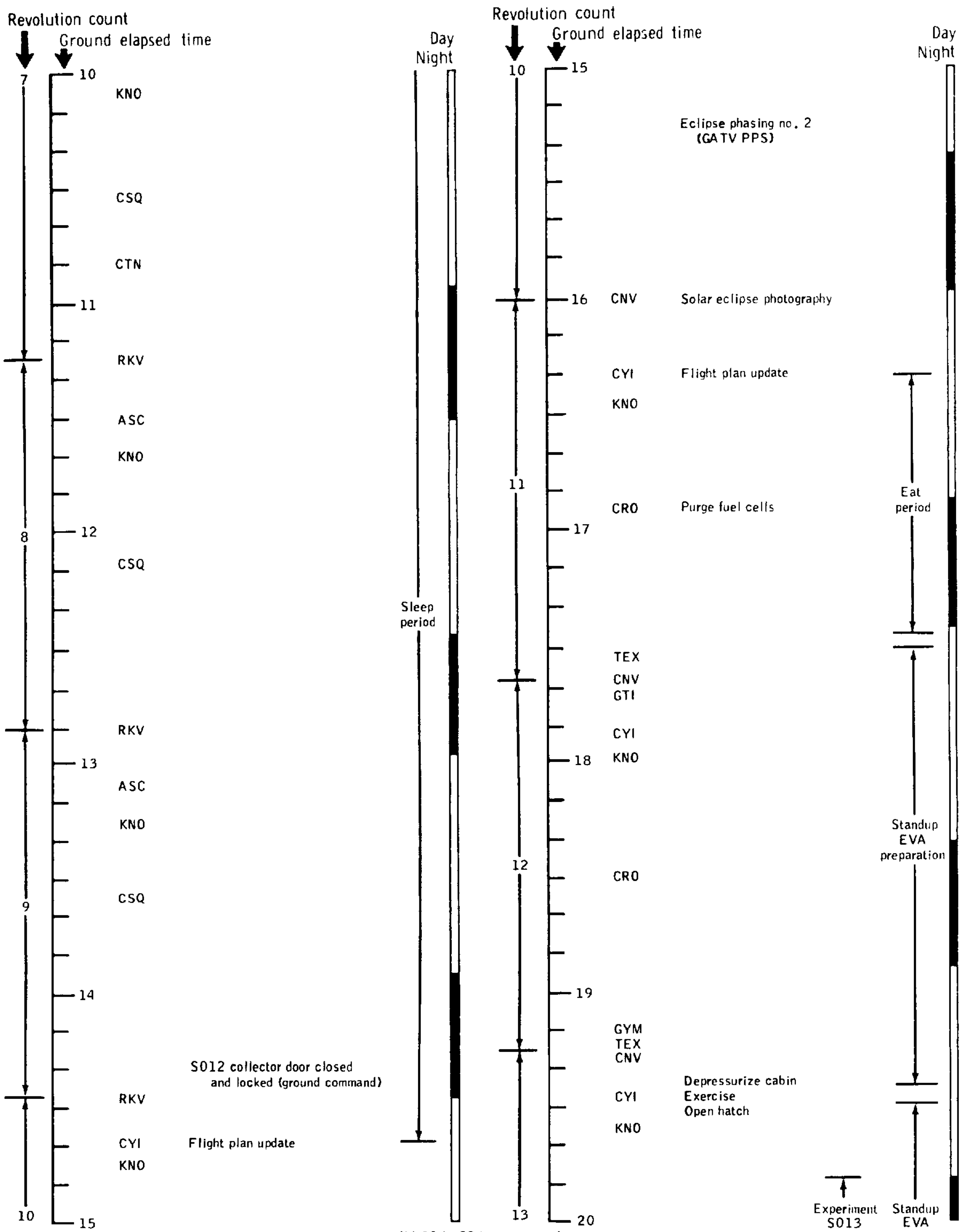
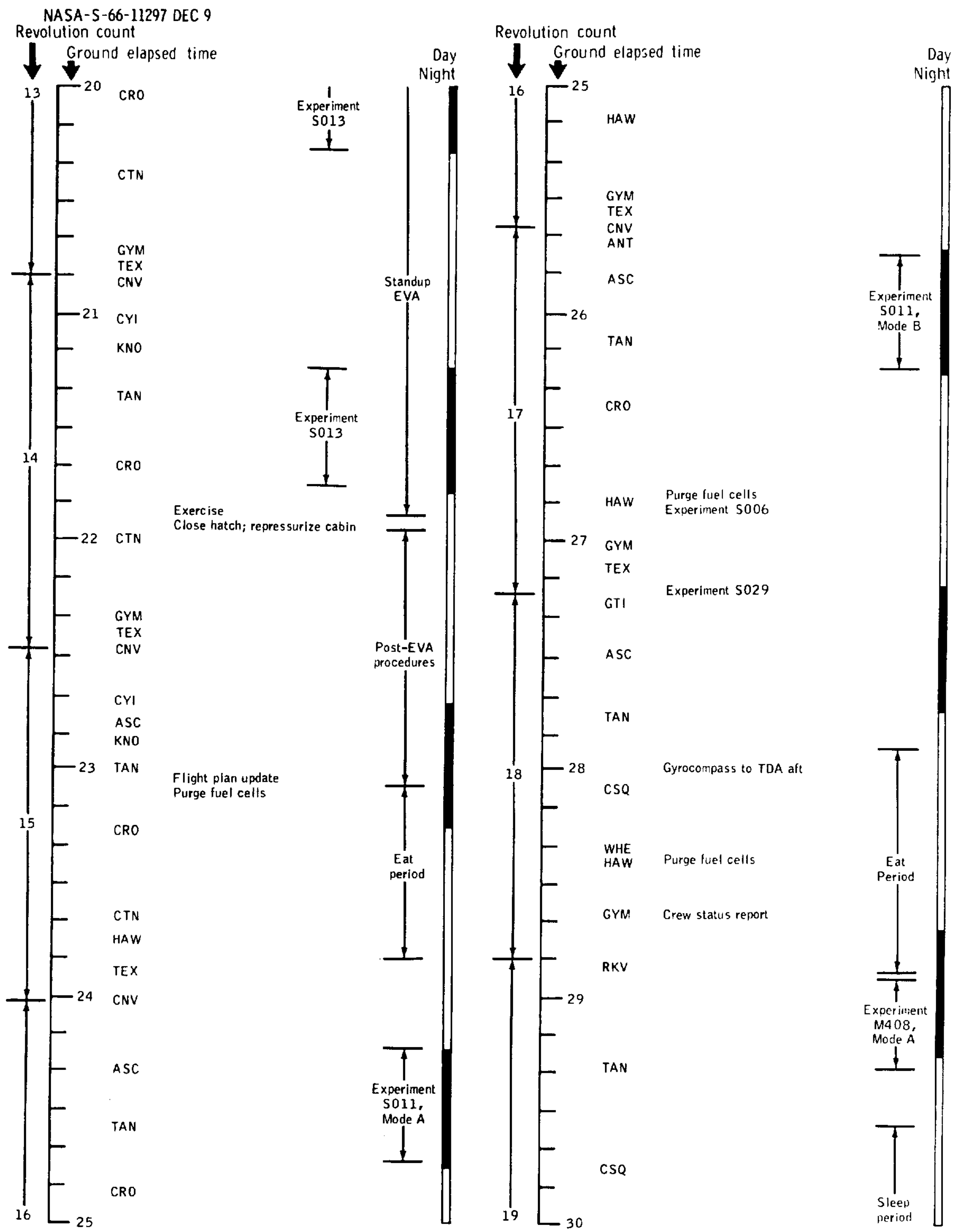
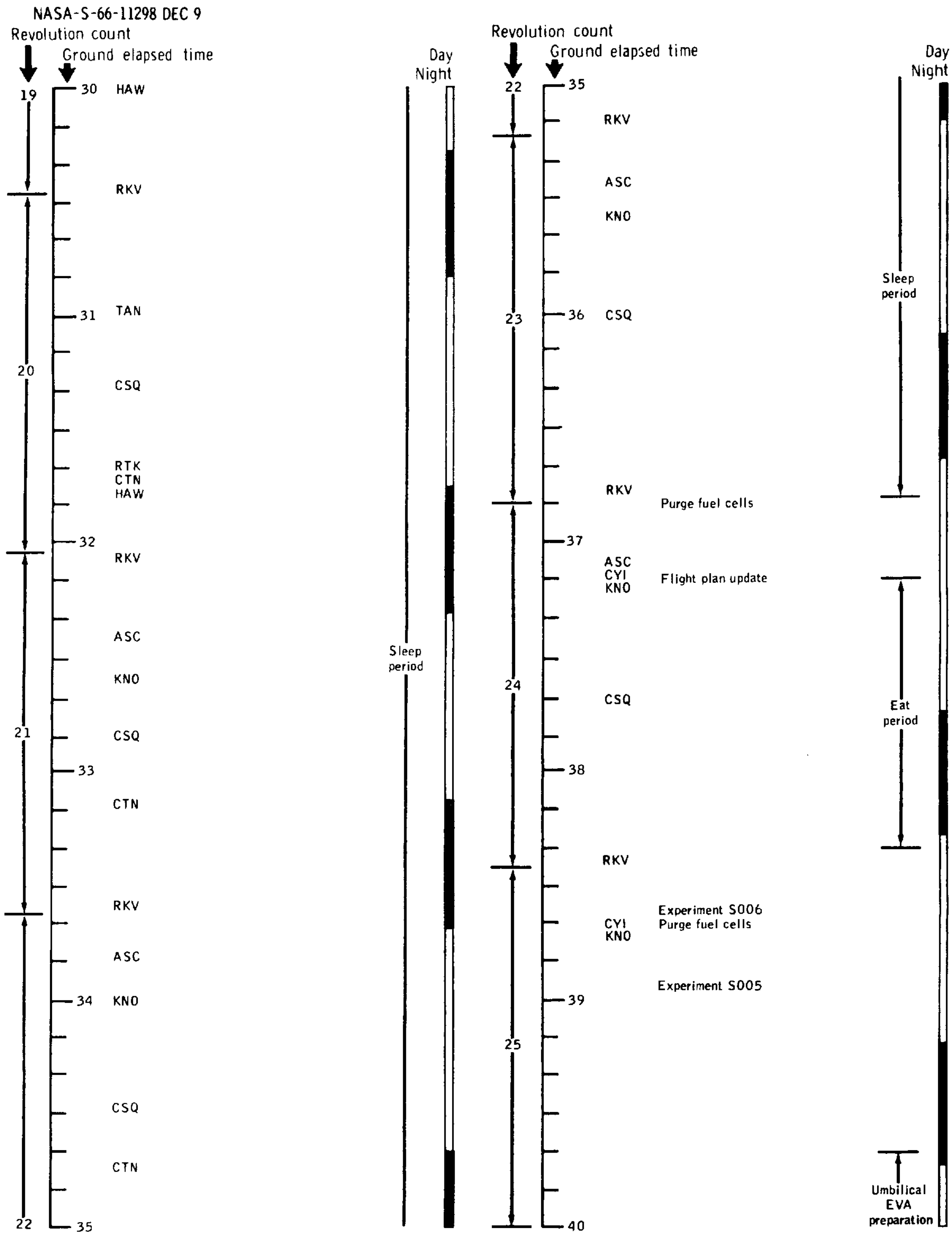


(b) 10 to 20 hours g.e.t.
Figure 7.1.1-1.- Continued.

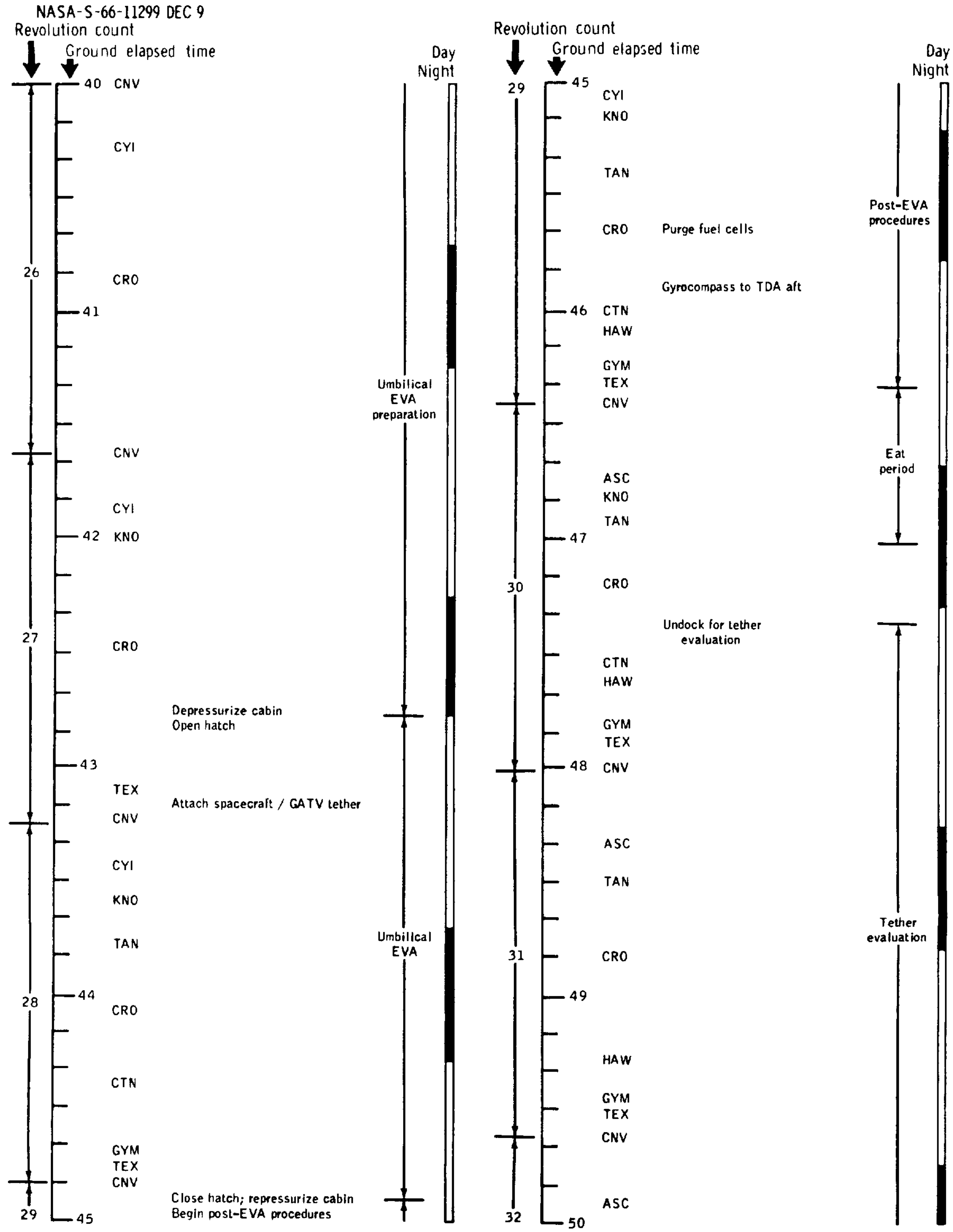


(c) 20 to 30 hours g.e.t.
Figure 7.1.1-1.- Continued.



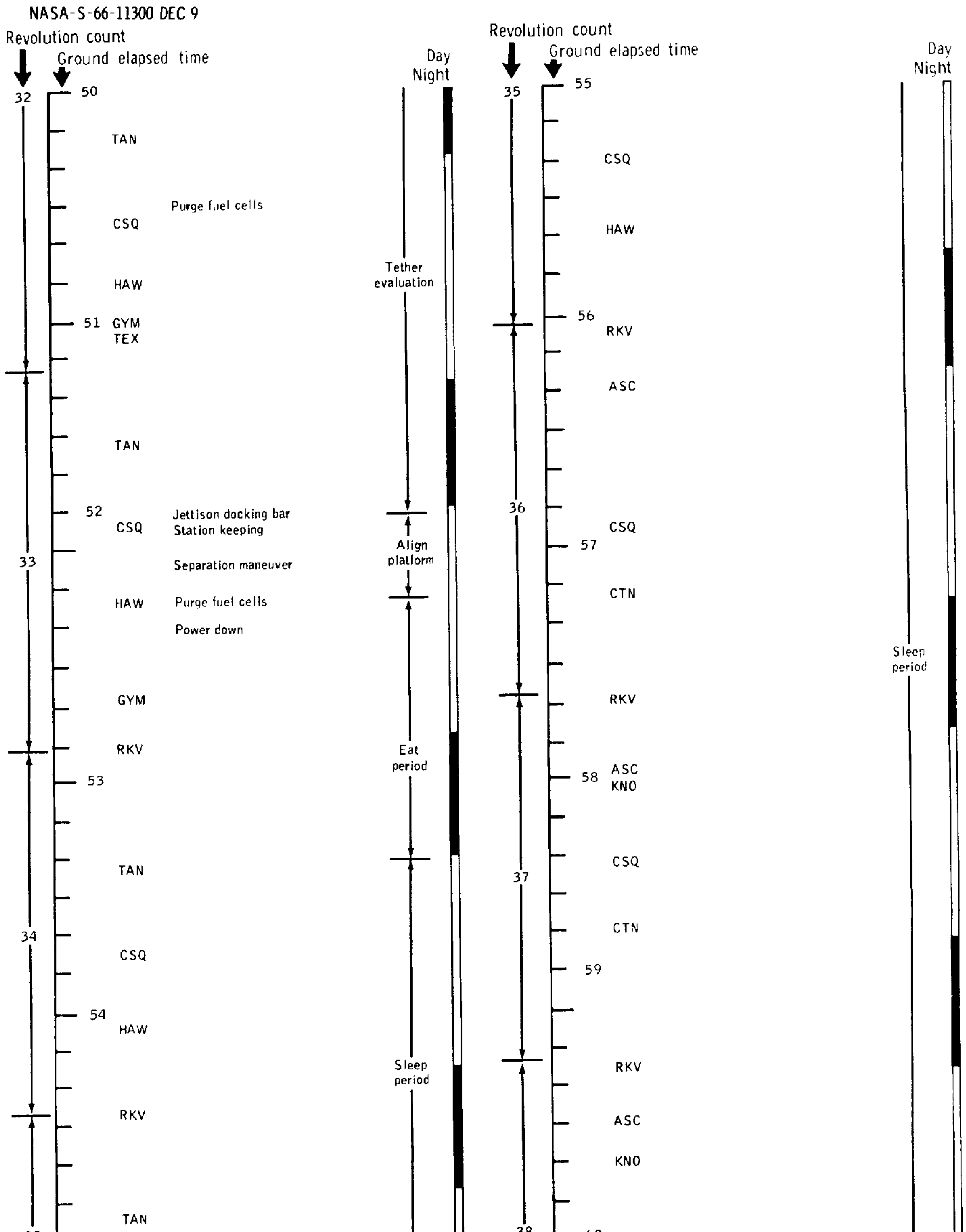
(d) 30 to 40 hours g.e.t.

Figure 7.1.1-1. - Continued.

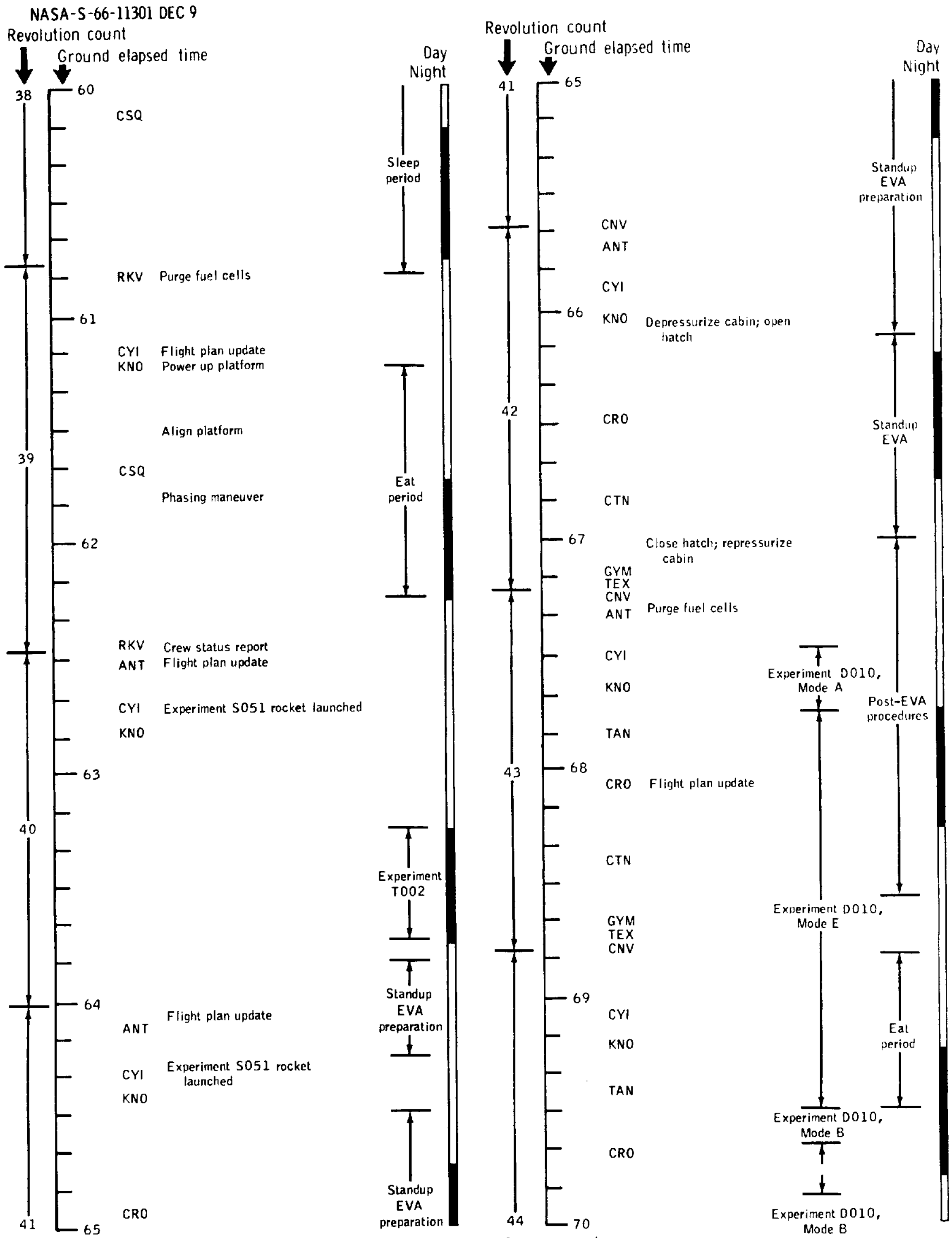


(e) 40 to 50 hours g.e.t.

Figure 7.1.1-1.- Continued.

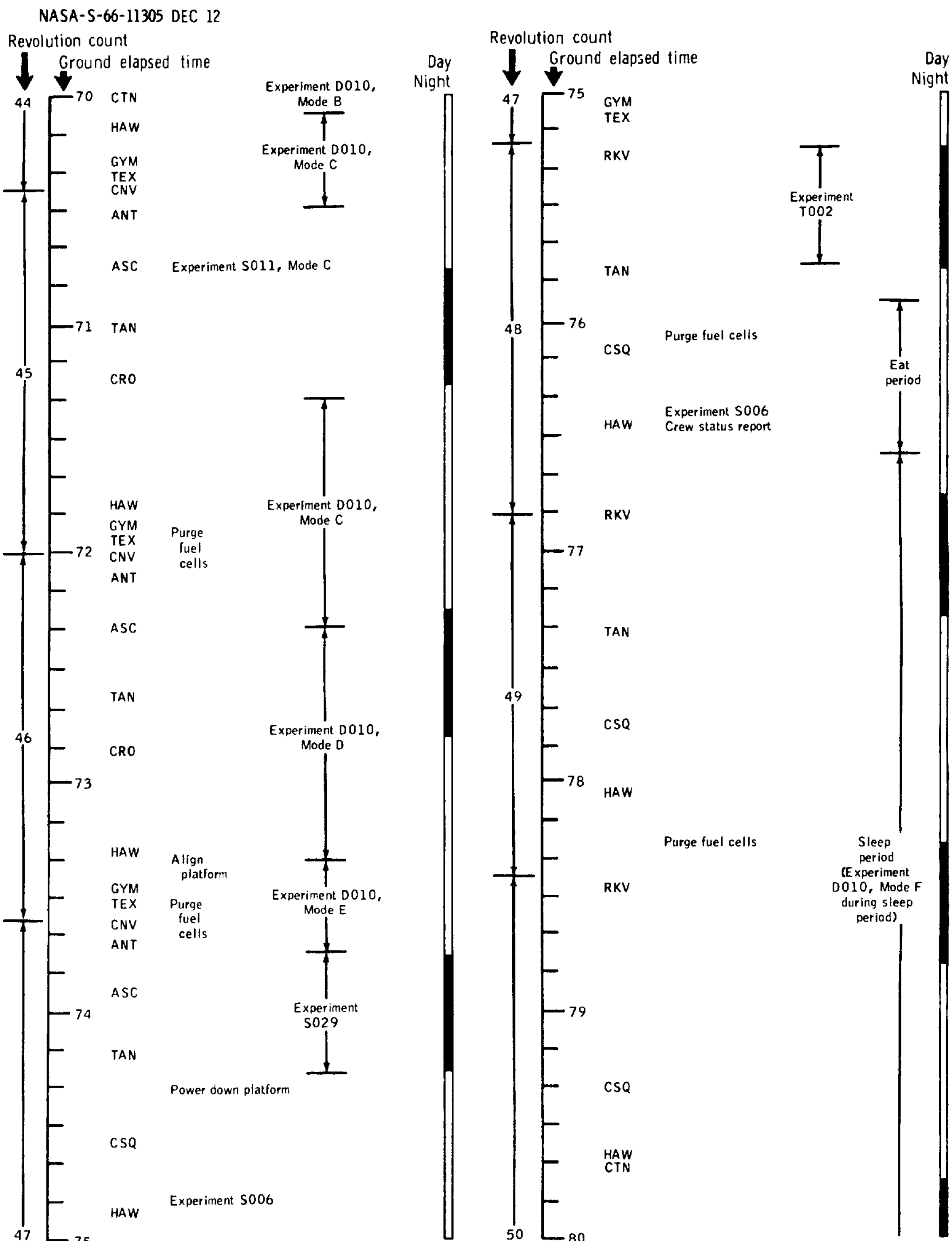


(f) 50 to 60 hours g.e.t.
Figure 7.1.1-1.- Continued.

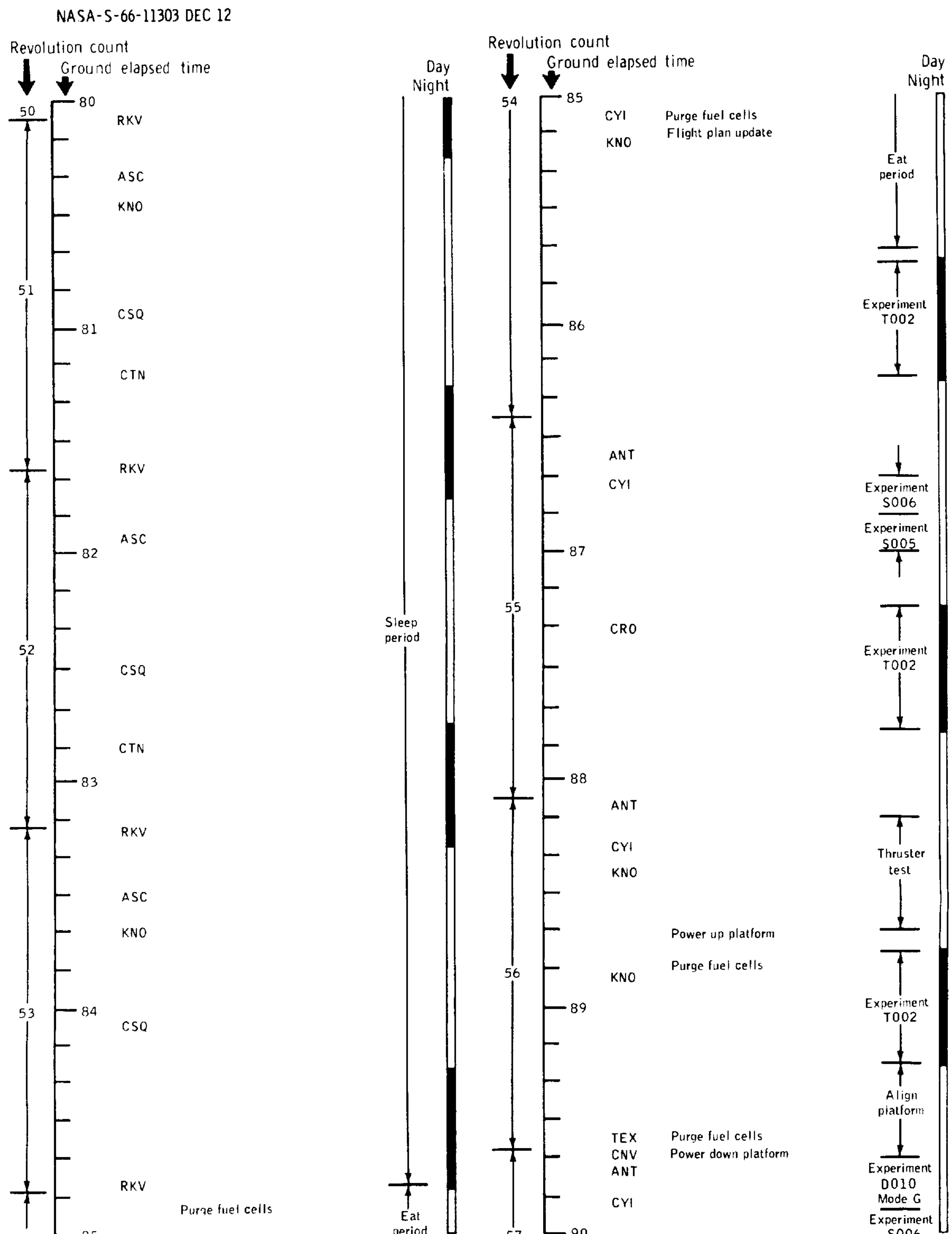


(g) 60 to 70 hours g.e.t.

Figure 7.1.1-1.-Continued.



(h) 70 to 80 hours g.e.t.
Figure 7.1.1-1.-Continued.



(i) 80 to 90 hours g.e.t.

Figure 7.1.1-1.-Continued.

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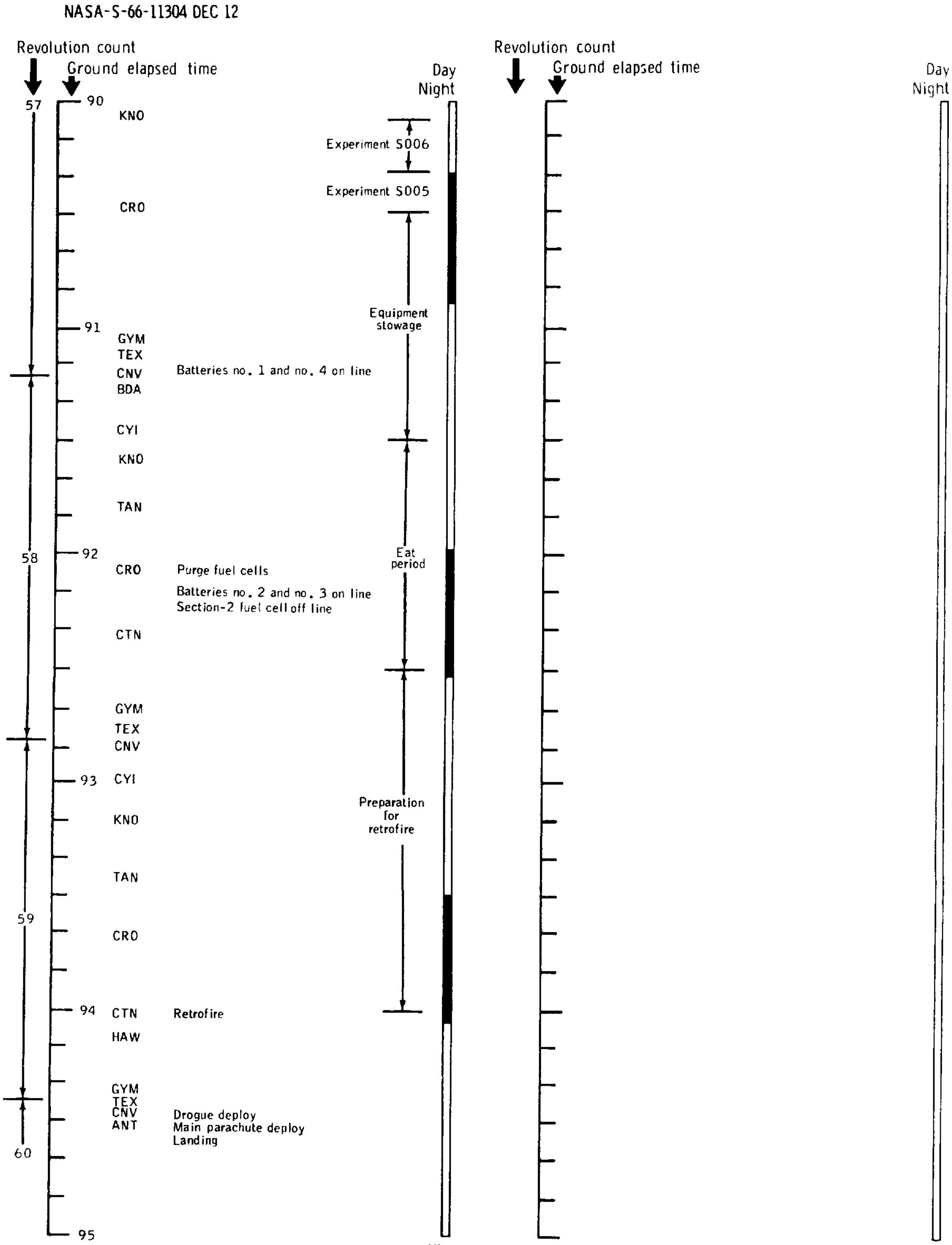

(j) 90 to 95 hours g.e.t.
Figure 7.1.1-1.-Concluded.

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7.1.2 Gemini XII Pilots' Report

7.1.2.1 Crew ingress.- The final count was smooth and on time up to the suiting procedure when the pilot ran into difficulty with the venting system in the arms of his suit. This required reconfiguration, and additional time was consumed in redonning the suit. The crew went through this process in a minimum of time and managed to leave the suiting trailer at very near to the correct time. The remainder of the final count and crew ingress into the spacecraft was very smooth. The crew believed that the simulator training in the pressure suits and the dress rehearsal with the simultaneous launch demonstration made the entire launch-day operation very easy and familiar to them. In the ingress procedure the suit technicians unstowed and gave the crew the ejection seat D-rings. Communications between all ground personnel and the crew were excellent. The push-to-talk mode was selected on the voice control center to eliminate background noise, and all prelaunch updates were received on time. To save oxygen the crew did not use the oxygen high-rate position during the cabin purge. The negative suit pressure that resulted during the cabin purge was not too uncomfortable to the crew.

7.1.2.2 Powered flight.- Lift-off was nominal and on time. Movement off the pad could definitely be felt by both crewmembers. All the onboard cues—starting of the clocks, computer light operation, the cabin pressure indication, and roll and pitch programs—were nominal. Two unexpected items happened during the powered flight. The right secondary oxygen indicator reading increased to 6000 psi during both maximum acceleration periods of powered flight. After second-stage engine cutoff (SECO), this pressure indication returned to 5000 psi. The left secondary oxygen pressure indication was normal and remained at 5400 psi during the entire powered flight.

During maximum Stage I acceleration, the oxygen-to-water differential pressure (ΔP) warning light for fuel cell section 1 came on but went out after staging. During Stage II operation, both ΔP lights came on in the maximum acceleration region but went out at SECO. Staging was noticed by both crewmembers. There was a definite flash in the window and the window appeared to fog over at this time. Communications with the ground during the boost phase were excellent. Guidance steering during second stage flight was nominal. At radio guidance initiate there was a slight Stage I yaw deviation and then both rate and attitude error indicators remained at zero during the remainder of the boost phase. SECO occurred at 5 minutes 44 seconds on the event timer, which was about four seconds later than had been anticipated by the crew. Preflight briefings indicated that SECO would occur at 5 minutes and 40 seconds. (Editor's note: SECO is based on velocity and varies in time with engine performance.)

7.1.2.3 Insertion.- At SECO plus 3.5 seconds, the Incremental Velocity Indicators (IVI's) read 25 ft/sec forward, 13 ft/sec left, and 3 ft/sec down. Spacecraft separation was completed at 6 minutes ground elapsed time (g.e.t.) and the horizon sensor fairings were jettisoned at the same time. The spacecraft was maneuvered to zero-roll and zero-yaw indications on the Flight Director Attitude Indicator (FDAI) and then pitched up to zero the pitch-error indicator. The IVI's then indicated an insertion maneuver of 28 ft/sec forward and 9 ft/sec left. This maneuver was completed at approximately 7 minutes 10 seconds g.e.t. While still using the ascent mode of the computer, the pilot determined the plane change maneuver for the rendezvous to be 8.5 ft/sec right, to be applied at 1:14:22 g.e.t.

7.1.2.4 Rendezvous platform alignment.- Immediately after the insertion maneuver, the guidance platform alignment sequence was initiated. The command pilot used the horizon for pitch reference. The platform mode of the attitude control system was utilized for initial platform alignment and was found to work satisfactorily. Fine align was later accomplished using the pulse mode of the attitude control system. Initial alignment included the use of both primary and secondary horizon scanners to check their operation. No difference in performance could be detected by the crew. However, most of the spacecraft alignment was conducted using the secondary narrow-band horizon scanner. The spacecraft yaw-left tendency which has been caused by water boiler operation in the early part of all Gemini flights was noted during the initial platform alignment. A summary of spacecraft maneuvers performed during the rendezvous is shown in tables 7.1.1-I and 7.1.1-II.

Performing the phase adjust maneuver in the platform mode proved to be quite satisfactory. The onboard solution for the plane change maneuver was accepted by ground control, and was made with the spacecraft yawed 26 degrees right to compensate for the forward component of the lateral thruster. The ground-computed values of the corrective combination maneuver and the coelliptic maneuver were received from the ground. While the pilot computed the start time for the coelliptic maneuver, utilizing ground-computed maneuver values, the command pilot completed the corrective combination maneuver. Prior to this period, the radar was turned on and a solid lock-on was indicated at a range of approximately 235 nautical miles from the GATV. The 34-minute period between the corrective combination and coelliptic maneuvers was sufficient for the pilot to complete an onboard coelliptic maneuver determination. The rendezvous mode of the computer was selected to give the pilot range marks at 1-minute intervals. In addition, this permitted the command pilot to monitor and plot the plane change maneuver solution (address 27), for an input of ωt (angle of orbit travel to rendezvous) of 270 degrees. During the coelliptic maneuver computation the command pilot used the radar mode of the FDAI to maintain boresight. Although optical track was not used during this period, the pilot was able to visually sight the target, using the T002 sextant, at a range of approximately 84 nautical miles. Onboard values of the coelliptic maneuver were within mission-rule requirements and the onboard solution was applied for all three axes.

After the last range data point was taken at approximately 2 hours 13 minutes g.e.t., the platform was aligned while the onboard solution of the coelliptic maneuver was being computed. In the period between the time when the pilot completed the coelliptic maneuver determination and the time the maneuver was completed, radar lock with the GATV was lost. Routine troubleshooting onboard the spacecraft did not reveal the source of the problem. The crew proceeded with the rest of the rendezvous using preplanned procedures for use in the case of a failed radar. A platform alignment was conducted during this period of radar troubleshooting. The sextant was used to determine the end of this alignment period by measuring the angle between the target and the local horizontal of the spacecraft. Visual acquisition occurred at a range of approximately 50 nautical miles. The command pilot found that day or night tracking of the target was very easy under these conditions of sun angle, and alignment between the reticle and the sextant was within ±4 degrees in yaw and ±1-1/2 degrees in pitch. During this period of optical tracking, intermittent lock-on was noted, as indicated by the signal light. In coordination with the ground, the GATV antenna select was switched from dipole to spiral. However, it was noted that the intermittent lock-on was more frequent with dipole than spiral and dipole was reselected.

7.1.2.5 Terminal phase initiation.- The terminal phase initiate (TPI) maneuver was computed using backup procedures for use in case of a failed radar system. The onboard solution was initiated at 3:05:47 g.e.t. and agreed closely with the ground-computed value (see table 7.1.1-II). A misunderstanding between the command pilot and the pilot resulted in omitting the up component of the maneuver.

All midcourse corrections were small (table 7.1.1-II). Since the first midcourse correction indicated an up correction of only 1.5 ft/sec, no action was taken. The polar plot made by the pilot (fig. 7.1.2-1) showed that the rendezvous trajectory was close to nominal. About the time of the fourth correction, the pilot believed that the range and range-rate information from the computer was fairly reliable, and this information was used in monitoring the braking phase of the rendezvous. The sextant was used to assure that a closing rate was maintained, but no closing rates were computed from the sextant readings.

Braking was accomplished in increments, starting with a 3 ft/sec decrease to a closing rate of 45 ft/sec at the fourth midcourse correction. At a range of about 1.5 nautical miles, the closing rate had been decreased to about 17 ft/sec. The inertial indicators were used for an inertial reference during this period. The line-of-sight rates were very small throughout this portion of the braking sequence, requiring only one input correction, according to the inertial indicators. At a range of approximately one nautical mile, inertial reference was shifted to the stars.

From approximately one mile range, braking was accomplished in small increments. The analog range and range-rate indicators did not function during the braking phase. Position and closing rates were maintained by monitoring computer information and by visual observations of the target. At the completion of the rendezvous, the propellant-quantity-remaining indicator read 69 percent.

7.1.2.6 GATV operations.- In the period from 6 hours to 47 hours 30 minutes g.e.t., flight plan activities were conducted with the spacecraft and the GATV docked. The initial docking occurred at 4:13:52 g.e.t. No problems were encountered with the docking, the gyrocompassing to Target Docking Adapter (TDA) forward, or the subsequent undocking and GATV fly-around.

At 4:49:44 g.e.t., a night docking attempt was made, but the rigidize sequence did not occur. When attempting to back away, the spacecraft hung on the lower docking latch of the TDA. Several forward and aft thruster firings were made to disengage before separation was actually accomplished. Spacecraft thruster firing appeared to have disturbed the GATV attitude. This disturbance, combined with what appeared to be a spacecraft attitude control problem, delayed the second docking until 5:07:14 g.e.t. A third undocking and redocking occurred at approximately 6 hours 5 minutes g.e.t., mainly to check the spacecraft attitude control system in the undocked mode.

Maneuvering by gyrocompassing the GATV proved to be a more tedious task than expected. The large amount of fuel aboard the GATV caused the GATV to overshoot the desired heading. A long time was required for the vehicle to settle down in both flight control modes 1 and 2, and it would often oscillate 30 to 60 degrees to either side of the desired heading. In general, the flight plan did not allow sufficient time for maneuvering with the GATV. Spacecraft control was required to maneuver the combination. Even using this technique, a slight error in yaw would cause excessive hunting by the GATV.

The GATV proved to be an excellent platform for holding inertial angles. The small deadband in flight control mode 2 contributed greatly to the success of star photography experiments. Two secondary propulsion system (SPS) maneuvers were made for eclipse phasing. Normal procedures were used and no problems were encountered.

The GATV display panel was adequate and readable. Message acceptance pulse (MAP) lights were received for all hardline commands but none were received for radar commands after the radar problem. The ground did confirm that the radar commands were being received and executed by the GATV.

7.1.2.7 Eclipse.- Two ground-computed SPS maneuver (43 ft/sec retrograde and 15 ft/sec posigrade) were made to intercept the umbra of the eclipse on November 12, 1966. Totality was observed at 16:01:44 g.e.t. and movement of the moon with respect to the sun agreed closely with the computer information received before the flight. The eclipse filter was adequate for tracking; however, the docked GATV complicated the photography. It was not possible, due to the short time period and low sun angle, to photograph the shadow of the moon on the earth.

7.1.2.8 Extravehicular activity.- New EVA procedures had to be developed because of a late change from evaluation of the Astronaut Maneuvering Unit to general EVA work tasks. These procedures had to be developed in conjunction with rendezvous training, which had been scheduled late in the training cycle because of conflicts with training of previous crews. An extensive amount of crew activity was required to develop and refine both the equipment and the procedures for this revised EVA flight plan.

7.1.2.8.1 Objectives and general plans: The general plans for the EVA were (1) to evaluate task complexity as a function of body restraint, (2) to demonstrate ability to maintain a reasonable workload with an open-ended approach, allowing for periods of rest and subjective evaluation, (3) to evaluate short-time learning and acclimation, (4) to evaluate the effects of the Extravehicular Life Support System (ELSS) due to the exhaust-flow pressure forces and the incumbrance of the ELSS and space suit combination, and (5) to evaluate the work level indications.

The first standup EVA was scheduled on the second day for crew familiarization prior to the umbilical EVA. It included a rest and calibrated exercise task and Experiments S013, S012, S005, and S006. Simple work tasks without the ELSS were also to be evaluated and compared with tasks during the umbilical EVA. The second standup EVA was scheduled on the fourth day and included ultraviolet photography and equipment jettison.

In comparison with previous umbilical EVA's, the Gemini XII plan was simplified to eliminate end-to-end complexity and time-critical tasks. Incorporated in this plan was an evaluation of various body restraint systems including fixed foot restraints, adjustable waist tethers, and portable handholds. Frequent rest periods were scheduled throughout the EVA. An effort was made to evaluate work tasks which would have future EVA applications.

7.1.2.8.2 Sequence of events: Preparation for the first standup EVA went according to plan, except that additional time had to be set aside for conducting the eclipse photography. This tended to disrupt the continuity of the preparation sequence. Following a suit integrity check, a calibrated 1-minute exercise period was conducted by raising the arms to the helmet at a rate of one cycle per second. Hatch opening, which occurred 20 minutes prior to sunset, was extremely simple. Body motions due to light hand forces against the open hatch were evaluated and the ultraviolet camera was mounted with little difficulty. It was noted that approximately eight minutes were required after sunset until the extravehicular pilot could see a sufficient number of stars to give spacecraft pointing commands to the command pilot. Spacecraft thruster problems resulted in a slow rate of motion from one star field to the next. The crew was able to photograph only two of the three star patterns scheduled for the first night pass. During the daylight period, the 16-mm movie camera was installed on the adapter retrograde section, in both tightly and loosely constrained configurations. Two glass contamination strips were exchanged, the S012 Micrometeorite Collection experiment package was recovered and stowed, and the telescoping handrail was extended and mounted with little difficulty. During the second night pass, ultraviolet photographs were taken of two separate star fields. After sunrise, the ultraviolet camera was stowed and another exercise session was conducted in the open hatch. Body position during this exercise period was maintained by an outward pressure of the legs against the inside of the spacecraft. The pilot found that positioning his body down in the seat for hatch closing was quite easy and required only a moderate expenditure of energy. The hatch closing was smooth and required very low forces.

Preparation for the umbilical EVA was conducted according to the flight checklists; however, medium flow was maintained on the ELSS until just prior to hatch opening. The spacecraft attitude was controlled by the GATV to a TDA south attitude. The spacecraft was apparently not completely depressurized because of the ELSS flow, as the hatch sprang open 10 to 12 inches when the latches were unlocked. Body motions in the open hatch were similar to those experienced during the standup EVA. The pilot observed no significant tendency to float upward or out of the spacecraft. The EVA camera was installed twice in the immediate hatch area, and the task compared very favorably to the previous camera installations. This camera was installed a third time, with the pilot entirely free of the hatch, using the handrail to maintain body position. Again, no difficulties were encountered. The pilot then proceeded along the handrail to the GATV. A slow controlled turn was executed, with small hand and finger forces, to face the docking cone. The adjustable waist tethers were connected to the handrail and the docking cone. The modified GATV-to-spacecraft tether was looped over the docking bar, and the clamp was installed with very little effort. The waist-tether restraint system permitted the pilot to concentrate his energy usefully. Rest periods with waist tethers attached were extremely relaxing.

Following the deployment of the S010 experiment package on the GATV, the pilot deployed the portable handholds on the Velcro panels in preparation for a return to the TDA area. During these operations at the TDA, three different pairs of attach points were utilized for the waist tethers. The only difficulty encountered in attaching the waist tethers was that the nearest handhold to the tether attach point was about two feet away.

Upon returning to the hatch area, the pilot recovered the remaining glass contamination strips. The EVA camera was handed to the command pilot in exchange for the adapter-work-station camera. Movement to the adapter was made along the deployed handrails. After turning the corner and entering the adapter, the pilot routed the umbilical through the guide. Using a cartwheel-like body motion and assisted by the hand bars, he placed his feet above the foot restraints. Engagement into the foot restraints proceeded without difficulty once the exact foot position had been visually observed.

Following a slight bracket linkage problem, the adapter camera was installed, and its operation was verified. Resting in the foot restraints enabled complete body relaxation. The mobility afforded by the foot restraints was outstanding, ranging from 90 degrees backward to 45 degrees to either side.

When the penlights were deployed, the pilot observed that one had been considerably deformed by what appeared to be overheating. He also felt evidence of heating on the portions of his space suit exposed to the sun, particularly at the metallic entrance zipper in the rear.

Bolt torquing operations were conducted with a rachet-type hand wrench. The forces required to exert given torque values were very similar to those experienced in underwater training. A comparison was then conducted by performing similar torquing operations using only the waist tethers. The pilot found that the bolt location point was too close to the tether attach point. Tasks involving the use of hooks and rings located higher in the panel were found to be considerably easier. In evaluating electrical and fluid connectors, the pilot found that a two-handed operation was preferable. In evaluating operations using just one foot restraint, he observed an increase of mobility but at the expense of higher forces required. The adapter camera was removed and returned to the hatch area. In remounting the EVA camera on the adapter, considerable difficulty was encountered in obtaining a proper handhold on the camera and bracket lever. It was difficult to install the camera in the mount with only one hand.

In moving along the handrail to the nose of the spacecraft, the pilot had no difficulty in backing into position above the TDA work station. Portable handholds had been brought from the spacecraft adapter to the TDA. They were hooked to the waist tethers and attached with Velcro to the chest pack. In turn, the waist tethers were transferred from the handholds to pip-pins, which were then engaged in the aft retaining holes on the TDA. The two waist tethers, together with the feet contacting the skin of the GATV, afforded a stable and comfortable body position at the work station. In this configuration, tasks were performed involving fluid and electrical connectors and the Apollo torque wrench. These tasks were repeated using only one tether and then without any tethers, with increasing attention required to maintain body positions. The waist tethers were jettisoned by shortening the adjustment, unfastening the snaps at the parachute harness, and pushing against the GATV to pull the tether free. Return to the hatch was accomplished using the handrail. The camera was removed and the handrail was jettisoned with little difficulty. Zero-g handling of the umbilical and the ELSS during ingress was considerably easier than had been anticipated. Again the hatch closing forces were light and the cabin was repressurized using the emergency oxygen supply in the ELSS.

The second standup EVA, conducted for one hour on the fourth day, was accomplished without using oxygen or electrical extensions. In this configuration there was adequate mobility to jettison the ELSS, the umbilical and various other EVA gear, and the waste food containers. Ultraviolet photographs were taken of two star constellations during the day and night periods. The photographs of the sunrise necessitated an awkward one-handed operation. Because of attitude thruster problems, the spacecraft was maintained small end forward (SEF) and this placed the sun behind the open hatch. Two exercise periods were accomplished. The exertion and tiring effects noted were quite comparable to the preflight calibrations.

7.1.2.9 Tether evaluation.- After the GATV was placed in a local-vertical position, the spacecraft was undocked and the GATV tether deployed at the beginning of a day period (47 hours 23 minutes g.e.t.). Small thrusts of the forward-firing thrusters were required to overcome the friction of the packed tether.

Initial opening velocities were kept small since the degree of spacecraft attitude control was questionable. The opening rate was decreased to zero with about 80 feet separation between the spacecraft and the GATV. At the beginning of the tether evaluation, normal procedures were utilized; attitude control was maintained by the pilot using local vertical on the Flight Director Indicator while translation control was handled by the command pilot. It soon became apparent that spacecraft attitude could not be controlled to maintain a local vertical. Small translations resulted in attitude excursions which could not be nulled because of the thruster problems. Attitude control was abandoned and the crew attempted to position the spacecraft directly above the GATV with the relative motion nulled. When it appeared to the crew that this condition was achieved, translational control was stopped and the spacecraft/GATV system was allowed to stabilize. During the initial phase, the tether was loose and the spacecraft experienced large attitude excursions. As time progressed the tether became taut and it first appeared that the system was gravity-gradient stabilized.

During the following night pass, the system slowly rotated until the GATV rose above the horizon. At this time the spacecraft attitude control system was again activated to reposition the spacecraft for another attempt at capture. This time the initial system amplitude was smaller than on the first attempt. A small correction thrust was made during the first-period oscillation to stop the relative motion when the spacecraft was directly above the GATV. Both control systems were shut down and the spacecraft/GATV system was allowed to stabilize. From approximately 47:32:15 to 49:51:10 g.e.t., the system oscillated about two to three times. The amplitude appeared to be decreasing and never got above 50 degrees as measured from the vertical. The tether was jettisoned at 51:50:57 g.e.t.

7.1.2.10 Reentry.- Reentry preparations were started approximately three hours prior to retrofire time. Final stowage of equipment was made with emphasis on stowing as much film as possible in the centerline camera container. Because of the degraded OAMS thrusters, spacecraft alignment blunt end forward (BEF) was started early to assure adequate time for the alignment. Loading Module IV into the computer was completed normally.

Digital Command System (DCS) updates were received from the ground and verified by onboard readout. Pre-retrofire checklists were adequate and completed ahead of time. The crew activated the Reentry Control System (RCS) early to ensure adequate control in the event the OAMS system could no longer be used for alignment. After activation, telemetry indicated to the ground that the regulated pressure in the RCS A-ring was high. To reduce this pressure prior to retrofire, the A-ring was used for spacecraft attitude control for the remainder of the flight. Retrofire occurred on time and was automatic. The retrorocket alignment was very good, with retrorocket no. 4 being slightly misaligned, resulting in an IVI reading of 4 ft/sec left. The other IVI's read 301 ft/sec aft and 115 ft/sec down, as compared with the values of 302 and 113 which had been transmitted to the crew prior to retrofire.

Post-retrofire procedures were completed without incident. The spacecraft was oriented heads down with a 10-degree-left bank angle. Single ring (A) operation in the pulse mode was selected to control spacecraft attitude. The indication of 400K feet was normal and on time. The spacecraft was then positioned to the backup angle of 46 degrees left. It seemed to the crew that the time between the 400K indication and guidance initiate was quite long, occurring just two to three minutes prior to the reverse-bank-angle time. The first error-indicator deflection at guidance initiate was approximately 60 miles uprange, agreeing with the ground update. The crossrange error was almost zero. The command pilot selected the reentry rate command mode and manually controlled the spacecraft to the computer-commanded attitude. When the crew was satisfied that the position error indications were converging, the attitude control was switched to the reentry position. The automatic reentry system kept the roll error indicator stationary at about 1/2 degree right of center and the pitch and yaw indicators at zero. The maximum acceleration reached during the reentry was approximately 6g.

A close monitor on the source pressure of the RCS A-ring was maintained, and just prior to maximum acceleration, the B-ring was turned on. At the time of maximum acceleration, the left-hand sidewall pouch broke loose from the wall and slammed into the seat between the command pilot's legs. The D-ring was deployed just after retrofire and was lying in the same vicinity. There was some concern that the pouch could have pulled the D-ring. The crew felt this presented a potentially dangerous situation.

At 80K feet the attitude mode was switched to rate command to damp any residual oscillations. Drogue and main-parachute deployment and two-point suspension actuation were normal. The prelanding checklists were completed prior to landing.

The landing impact was quite severe, causing water to enter the spacecraft. The main parachute towed the spacecraft briefly through the water before the parachute jettison switch was actuated by the crew.

The smell of burned metal was noticed immediately after landing but was not objectionable. The cabin repressurization valve was opened to clear out the cabin. Wave action was more severe than anticipated, but checklist completion and recovery operations proceeded smoothly. The UHF frequency of 296.8 mc was used by recovery personnel to broadcast a running commentary of recovery operations which would have made communication on that frequency difficult had it been necessary. The crew elected to be recovered by helicopter.

7.1.2.11 Experiments.- Gemini XII was assigned fifteen experiments. Quantitative data on the results of these experiments are covered in section 8.0 of this report and crew comments are confined to operations problems and procedures that might be beneficial to future space-flight operations.

The thruster problems were a major factor that influenced the results and degree of completion of the experiments. The thruster malfunctions increased the time required to acquire particular pointing angles to star fields or geographic locations. Maintaining a steady inertial position for long-exposure-time photography was more difficult. Tracking was sporadic mainly because of the right roll that always resulted in pitch or yaw attitude control.

The spacecraft was mated to a heavily fueled GATV during many of the experiments. The GATV attitude control system was excellent for maintaining inertial position but, since many pointing angles were at odd pitch and yaw angles, the spacecraft was required to provide inertial attitude control. Attitude changes while mated to a heavily fueled GATV were very difficult.

Many of the operational procedures and equipment verification necessary for the successful completion of the experiments were worked out by the crew in their preflight training program. The crew believes that the experiments personnel should have a more active participation in working out these procedures, taking into account operational considerations and spacecraft limitations.

In general, the operation of the T002 sextant in zero-g was much simpler and easier to manage than had been anticipated from preflight training and simulations. It is believed that the use of filters would reduce the blurring effects of both star images. Even slight reduction in blurring would greatly increase the accuracy of the zero-bias measurements. Acquisition of the star patterns was found to be marginal with the restricted field of view and the hand-held mode of operation.

7.1.2.12 System malfunctions.-

7.1.2.12.1 Orbital Attitude and Maneuver System: The first indication of thruster problems in the Orbital Attitude and Maneuver System (OAMS) occurred during the night docking on the first day. In attempting to stop spacecraft rates, the rate command mode of the attitude control system was selected. Immediately thereafter the spacecraft began to roll right. At the start of the unexplained right roll, the roll logic was switched from yaw thrusters to pitch thrusters, and attitude control was switched to the direct mode. A successful night docking was made in this configuration. A short thruster check in the direct mode was made while undocked during the next day pass but revealed no discrepancies. It was noted, however, that attitude changes of the combined spacecraft/GATV system did not appear to be consistent with the corresponding spacecraft thruster inputs.

On the second day of operation, when the rate gyros were powered up, it was noted that pitch-down and yaw-right control inputs resulted in an inadvertent right roll. A more detailed attitude thruster check indicated that thrusters no. 2 and no. 4 were inoperative. In addition to the right-roll tendency, the imbalance of thrusters also made it difficult to command pitch-down and yaw-right motions.

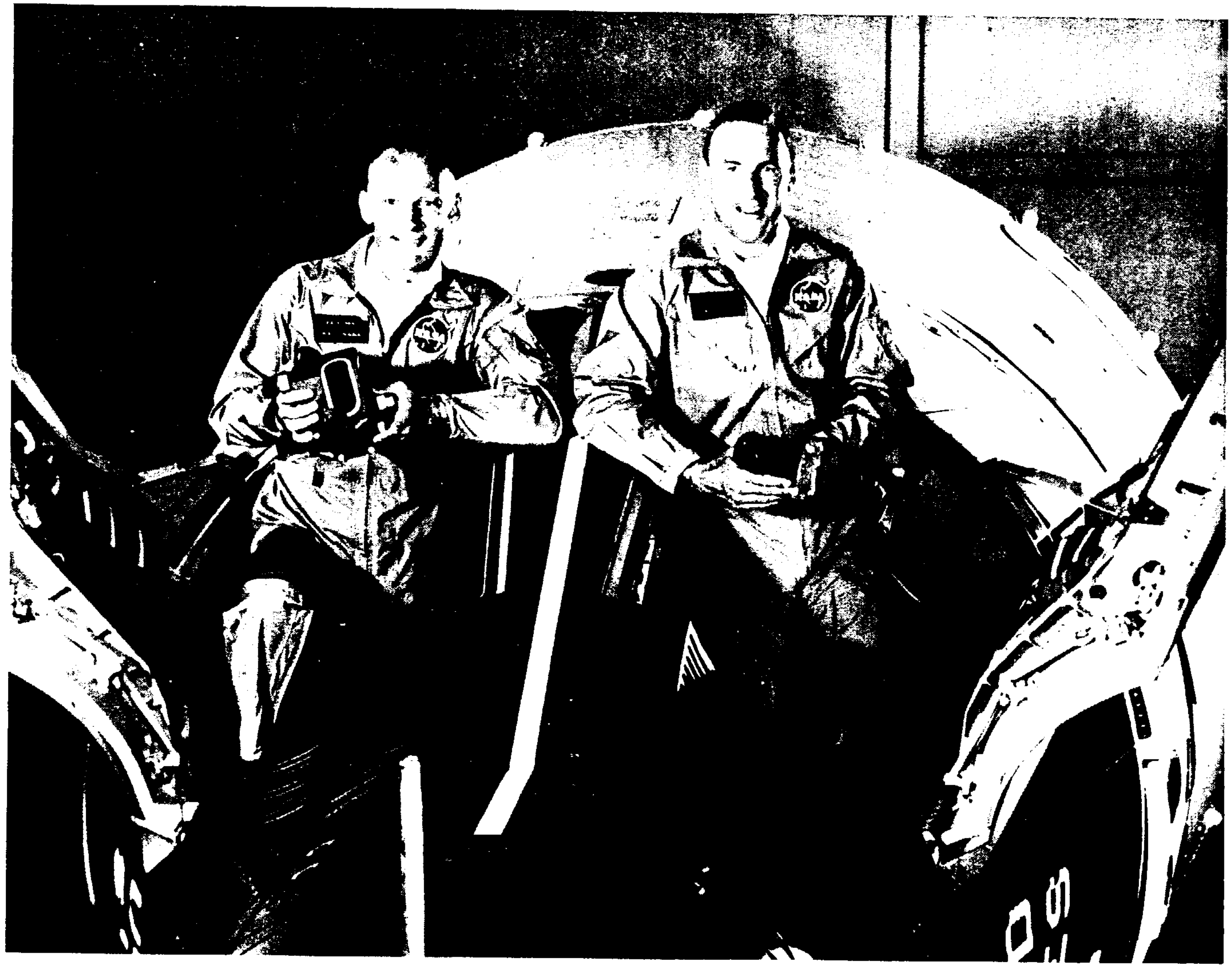
To help with the attitude control, maneuver thrusters no. 9 and no. 12 were used. Although the minimum impulse obtainable from a maneuver thruster was greater than that from the attitude thrusters, this procedure permitted adequate spacecraft control.

Additional characteristics of the thruster problem were evident. Selecting rate command mode resulted in right roll going divergent if the spacecraft had any rates at the time of selection. It was possible to use the platform mode to align the platform only if rates were practically zero at the time of selection.

As the mission progressed, it was noted that thruster no. 8 was not giving adequate response, and, during a thruster check at 88 hours g.e.t., thruster no. 7 was also observed to be deficient in thrust.

In general, the thruster problem complicated the entire mission. First, it required considerably more time to obtain the proper attitude for experiments requiring definite pointing angles. Second, smooth tracking was impossible. Third, maintaining low rates for long-exposure photography was difficult. It was interesting to note, however, that as the mission progressed and more thrusters became degraded, it became easier to control the spacecraft. It was still possible, up to the time of retrofire, to align the platform.

7.1.2.12.2 Radar: Shortly after completion of the coelliptic maneuver, the pilot noted a consistent computer range reading of approximately 64 nautical miles. No angle track indication was noted on the radar indicators and the lock-on light was out. The radar power circuit breaker



Astronauts Edwin E. Aldrin, pilot, and James A. Lovell, command pilot.

had not opened and all other circuit breakers were in their proper position except the OAMS propellant and one of the OAMS regulator circuit breakers. These were quickly closed but had no effect on the radar problem.

During the remainder of the rendezvous, the lock-on light came on intermittently. The frequency of illumination was greater when the dipole antenna was selected than when the spiral antenna was used. During periods of intermittent operation, the radar did update range and range-rate information in the computer. At no time did the analog range-rate work; however, the analog range operated intermittently, but only during the final phase of the rendezvous.

7.1.2.12.3 Water management: The adapter water supply was depleted on the morning of the fourth day. Procedures needed to obtain the water from the cabin tank were adequate, except that the crew believe the slider valve in the blood pressure bulb should be eliminated to prevent inadvertent pressure bleed off.

7.1.2.12.4 Fuel cells: Fuel cell difficulties are well documented in other portions of this report and will not be detailed here. The only crew comment is that the steady illumination of the ΔP lights interfered with the photographic experiments. They were also annoying during sleep periods. The crew believe that they did not receive adequate information on the troubleshooting procedures.

7.1.2.13 Crew training.- The crew believes that a firm flight plan several months prior to the launch date is a necessity for proper training and utilization of time. Late flight plan changes not only cause difficulties in getting properly trained, but also take the time of the crew to assist in the development of operational procedures and any new associated hardware required.

The crew training facilities were excellent and contributed greatly to the success of the mission. The Gemini Mission Simulator (GMS) again proved to be the most valuable tool in the overall procedure training of the crew. Suited operations in the GMS made real-time operations easy and more familiar. The abort trainer and the docking trainer were also valuable in preparing for the flight.

The crew station mock-up was used extensively in developing EVA preparations procedures and was invaluable in integrating the EVA training. Some minor problems developed in keeping the mock-up in an up-to-date functioning status.

The underwater zero-g facility afforded the crew an outstanding means of exercising and validating the various restraint systems to be used during the flight and pacing the entire work-rest cycle for the complete EVA flight plan. In retrospect the underwater training was the most important single factor leading to the success of the EVA.

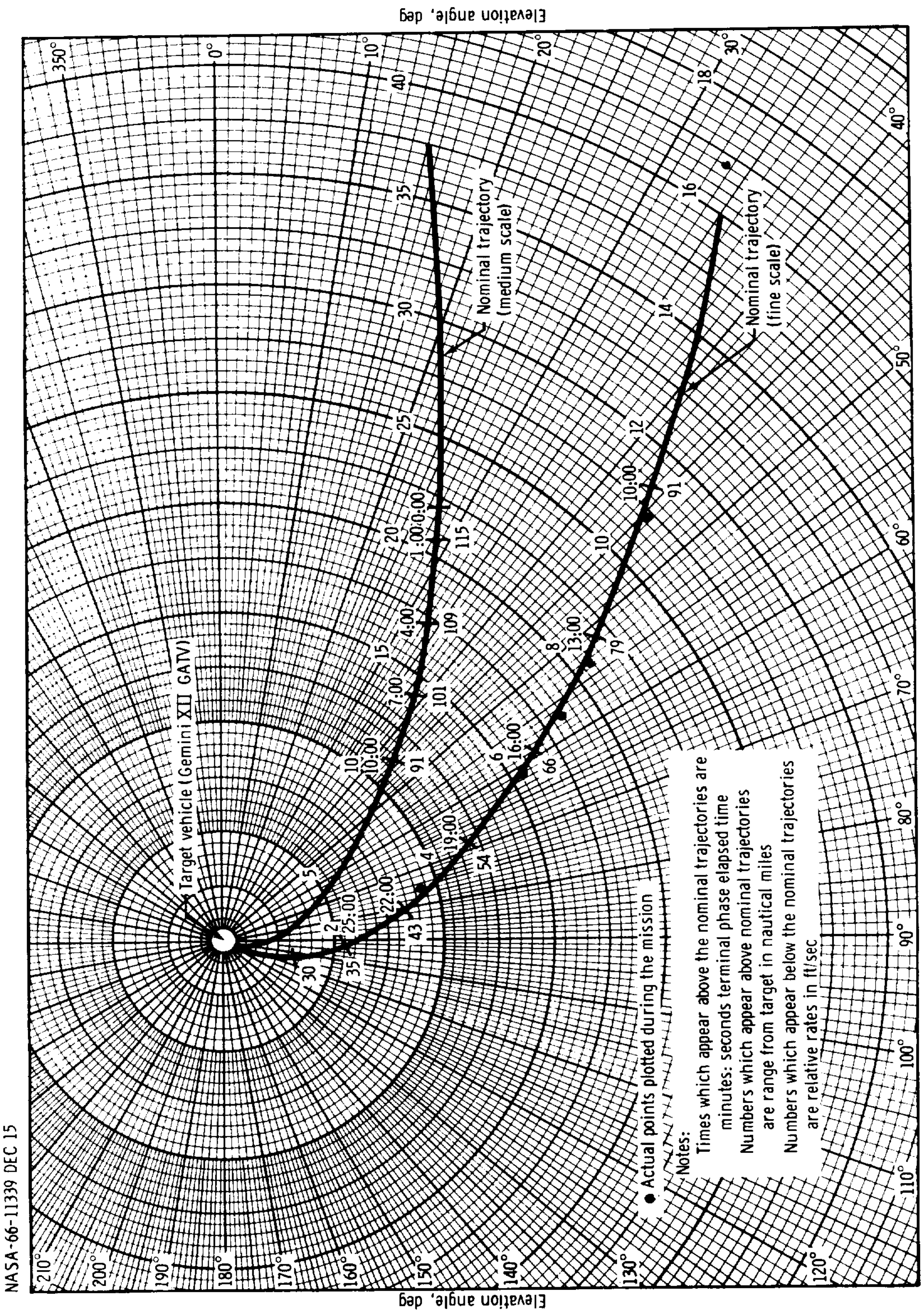


Figure 7.1.2-1. - Onboard target-centered coordinate plot of rendezvous.

7.2 AEROMEDICAL

Gemini XII was a short-duration mission which included a rendezvous, docking, docked maneuvers, a spacecraft/GATV tether evaluation, and three periods of extravehicular activities. There were no significant medical problems during this flight.

7.2.1 Preflight

7.2.1.1 General preparations.- Review of the medical records for the Gemini XII crew revealed nothing that could be considered a constraint to their participation in this flight. There were two interesting facts in the command pilot's record which deserve mention. During the first postflight tilt table study after Gemini VII, he experienced a brief period of syncope and a drop in blood pressure. Recovery was immediate on termination of the tilting procedure. The command pilot also has a well documented history of idiopathic hyperbilirubinemia which was thoroughly studied during his selection for the astronaut program. The pilot also has a history of hyperbilirubinemia which is considered to be secondary to infectious hepatitis. There has been no clinical evidence of this disease since November 1959.

There were no sensitivities to onboard medications or biosensoring materials found in the course of sensitivity testing.

7.2.1.2 Specific preflight preparations.- Due to late changes in the flight plan and changes in the EVA objectives, the pace of preflight activities was greater than on previous flights. In order to accomplish the necessary planning, training, and other preparation in time to meet the schedule for flight, the crew was forced to extend their work day, forego periods of rest and relaxation, and even compromise their physical exercise program prior to this flight. There was no attempt to recycle their daily activities to conform to the flight plan. The crew began a modified low-residue diet on November 5, 1966, and remained on this diet for the entire preflight period. On November 7, 1966, a throat culture taken on the command pilot was positive for beta-hemolytic streptococci. He was given 1.2 million units of a long-acting penicillin and 250 mg of an oral penicillin four times a day for the remainder of the preflight period. The pilot was also started on prophylactic oral penicillin. On the following day, the pilot developed symptoms of diarrhea. Although this was not considered related to the antibiotic, the penicillin was discontinued. The diarrhea subsided by the flight date and was attributed to a low-grade entero-virus present in other residents of the crew quarters at this time. Follow-up throat cultures on both crewmen on the day prior to launch were negative for pathogens.

In order to decrease the likelihood of inflight defecation, two bisacodyl tablets were given to each crewman on the morning prior to launch. This produced the desired effect; however, as the launch was delayed 48 hours, bisacodyl was repeated on the morning of November 10. The results again were satisfactory. On the morning of the flight, the pilot requested a bisacodyl suppository. This medication was given and was also effective.

7.2.1.3 Medical examinations.- On October 31, 1966, a medical examination was performed on the prime and backup crews by a specialist in internal medicine. On November 5, 1966, a comprehensive examination was performed on the prime crew by the crew flight surgeons and specialists in otolaryngology, neuropsychiatry, and ophthalmology. There were no unusual or disqualifying findings. At this time, however, the command pilot was noted to have an upper respiratory infection which is discussed in section 7.2.1.2. Laboratory determinations (table 7.2-I) were considered to be within normal limits.

7.2.1.4 Special data collection.- Two tilt-table studies were performed on each crewman prior to the flight. The data from these studies are shown in figure 7.2-1. A bicycle ergometry test of the pilot's exercise capacity was performed on November 5, 1966. The results of these studies are seen in figures 7.2-2 and 7.2-3. On November 5, 1966, in conjunction with the ophthalmological examination, retinal photographs of both crewmen were obtained. Since a flight plan revision included photographic studies of the solar eclipse, it was considered advisable to obtain these photographs for comparison with postflight retinal studies.

7.2.1.5 Precount medical activities.- Both crewmen received approximately eight hours of sleep on the night before launch. There was no medical difficulty with biomedical sensoring or suiting, and the crew countdown was accomplished as programmed. Both crewmen were considered to be well prepared for flight.

7.2 Inflight

7.2.2.1 Physiological monitoring.- The bioinstrumentation system was similar to that used on previous Gemini flights. During the latter portion of the flight, there was some degradation of the signal received from the command pilot's axillary electrocardiographic leads and the pilot's sternal leads. This was determined to be caused by loose sensors and did not significantly compromise the data collection.

7.2.2.1.1 Electrocardiogram: Heart rates plotted from real-time records and biomedical tape recorders may be found in figure 7.2-4. The pilot's heart rates during extravehicular portions of the flight are shown in figures 7.2-5 and 7.2-6. Figure 7.2-3 is a plot of the pilot's heart rate compared with his Btu output during the preflight and post-flight ergometry studies. This figure is presented only to document heart rates which were produced at given levels of energy expended during these studies. Direct correlation of EVA heart rates with workload is inaccurate since various factors other than workload influence heart rate. Heart rate data indicate, however, that the workloads experienced during the EVA portion of this flight were lower than those experienced during previous flights.

7.2.2.1.2 Respiration: Respiration rates are included in figures 7.2-4 through 7.2-6. These rates were well within the normal expected limits.

7.2.2.2 Medical observations.-

7.2.2.2.1 Launch and powered flight: No disorientation or unusual sensations were experienced during powered flight or upon transition into the weightless state.

7.2.2.2.2 Orbital phase: Medical observations and physiological responses during the orbital phase of this mission were influenced by system failures in the thrusters and the fuel cells and by other less significant problems which arose during the flight. During this flight, as in previous manned space efforts, man showed no evidence of failure and may still be considered the most reliable system aboard the spacecraft.

(a) Environment - Because of the thermal layer in the pilot's EVA suit, he was subjectively warmer than the command pilot during the entire flight. This was accentuated by the fact that, while in platform mode, the sun always rose on the pilot's side and caused his side of the spacecraft to be somewhat warmer than the command pilot's side. Both crewmen, however, were comfortable during the entire flight, and each crewman found it necessary to turn his suit coolant completely off during some portions of the sleep periods.

Immediately after the umbilical extravehicular activities, while the spacecraft was being pressurized from the ELSS emergency oxygen supply, the command pilot noted a burning in his right eye. There was no nose or throat irritation and no odor which could be associated with this occurrence, but his eyes began to water at that time. As he blinked, both eyes began to burn. After the cabin was fully pressurized and his face plate was opened, the command pilot wiped his eyes and the problem subsided. He deduced that salt from perspiration which had accumulated on his eyelids was washed back into his eyes by tearing. There are no indications that contaminants in the Environmental Control System (ECS) were the cause of this problem.

(b) Food, water, and sleep - Three meals of Gemini flight food per crewman per day were stored aboard the spacecraft. Approximately 42 pounds of potable water were stored in the adapter tank, with an additional 13 pounds stored in a tank in the reentry vehicle. The food and water intakes were reported during crew status reports at scheduled intervals. While there is no method to ascertain the amount of uneaten food which was jettisoned during extravehicular activities, a close approximation of the food and water intake based on crew status reports and both mission debriefings is found in table 7.2-II. On the third day, the crew was asked to increase water consumption because of problems with the fuel cells. Prior to retrofire, the water supply from the adapter tank was exhausted while the crew was preparing the morning meal. The crew pressurized the spacecraft water tank with the blood pressure bulb and found this method satisfactory.

The crew found it difficult to sleep during the first night. The approximate sleep times are shown in figure 7.2-4.

(c) Medications - No medications except aspirin were taken during this flight. The command pilot took 10 grains before retiring each night, and the pilot took 10 grains the first and third nights. It was deduced by the crew that aspirin was ineffective for inducing sleep.

(d) Vision - Because of a failure of their radar system, the crew accomplished the rendezvous using a backup method. This method relied heavily on visual contact and range-rate information obtained using an optical sextant. At a range of approximately 85 miles, the pilot visually acquired the target using the sextant, which had a magnification of 8×. The command pilot acquired the target visually at approximately 40 to 50 miles and lighting conditions were good. The target was a bright point of light once it was acquired, and tracking was no problem for the command pilot. The crew was unable to acquire star targets during daylight conditions; however, if a star was acquired at night and visually tracked, it could be seen for a short period into daylight. During extravehicular activities, the pilot was impressed by the brightness of the lighted objects and stated that after sunset the command pilot was able to acquire star targets four to five minutes before he could. During the second day of the flight, the spacecraft was maneuvered into position to observe a total solar eclipse without discomfort or danger of damaging the retina.

The pilot was not required to look directly at the sun and refrained from doing so; however, he was able to take pictures of the eclipse with a hand-held camera. Preflight and postflight examinations, including retinal photographs, revealed no evidence of retinal damage.

(e) Orientation - No abnormalities of orientation were experienced. During EVA, the spacecraft was in platform mode and was therefore stabilized inertially with no attitude reference to the earth. The horizon and the position of the ground were continually changing, but the pilot did not find this disturbing. His orientation was always in relation to the task at hand. Standing in the cockpit, he was oriented to the work station in the adapter; while moving along the handrails, he was oriented to the handrails and to the spacecraft with no reference to the earth, the sun, or any other object. He stated that he was much more comfortable in this condition than when he was a passenger on a balloon flight. There was no feeling at any time that he might fall. There was no confusion as to his position, and no evidence that his coordination was in any way affected by his position.

(f) Radiation - A high apogee maneuver over the northern hemisphere and into possible higher radiation levels had been planned for this mission. This maneuver was not performed because of a problem with the GATV primary propulsion system. Radiation levels during this flight were insignificant and are presented in the following table:

Location of measurement	Radiation dose, millirads	
	Command pilot	Pilot
Helmet	15 ±1	12 ±1
Thigh	15 ±1	14 ±1
Left chest	15 ±1	15 ±1
Right chest	19 ±1	15 ±1

7.2.2.2.3 Extravehicular activities: Extravehicular activities, conducted during three periods, totaled nearly 5-1/2 hours. Over two hours were spent entirely outside the hatch on an umbilical. All three periods of extravehicular activity were considered completely satisfactory. One of the prime reasons for the success of this mission is the fact that this was the fifth Gemini flight which included extravehicular activities. During each training cycle and subsequent flight, a considerable amount of valuable information was accumulated. This information was used in planning the Gemini XII EVA. From a medical standpoint, there was considerable concern over the high workload which was apparently encountered during previous extravehicular activities. Accurately measuring the workload during these flights was impossible; however, when correlated with ground studies, heart rates give a reasonable indication of work performed. The uncertainties associated with this correlation (thermal, environmental, or psychological factors) would lead to the assumption that a great deal of energy was expended during EVA to perform small amounts of work. It was important during the umbilical EVA to avoid workloads which would overload the ELSS. In the medium-plus-bypass position, the ELSS is capable of dissipating approximately 2000 Btu/hour while maintaining a carbon dioxide level of no greater than 6 mm of mercury. During the umbilical EVA, the pilot elected to keep his ELSS oxygen flow set on the high position, which will give a sustained heat rejection of approximately 1600 Btu/hour. A total heat dissipation higher than these figures is possible for short periods of time. During EVA, heart rates corresponded well with the rates obtained during underwater zero-g simulation studies and were considerably lower than expected, based on previous flight experience. There was no indication that the capabilities of the ELSS were exceeded at any time during the flight.

Two periods of programmed exercise were performed during each of the standup EVA's. The exercise consisted of raising the hands to the helmet against the neutral position of the hard suit once each second for 60 seconds. An attempt was made to correlate these exercise periods with the same exercises performed preflight using heart rates for correlation (fig. 7.2-7). There appears to be no significant difference in exercise performed preflight and inflight when compared in this manner. The pilot followed the concept of "minimum exertion" during EVA. He was able to subjectively relax muscle groups which were not required for the specific task at hand. He used the various restraints devised to optimum advantage. He used slow deliberate motions and positioned himself to operate within the neutral position of the suit. It is felt that underwater zero-g training, good preflight planning, and a thorough understanding of previous EVA experiences gave the pilot the advantage he needed to successfully complete this EVA.

7.2.2.2.4 Reentry: Retrofire and reentry were normal. Landing was significantly harder than the command pilot had expected, based on his experience as the pilot of Gemini VII, but the impact was easily tolerated.

7.2.3 Postflight

This portion of the report includes aeromedical observations beginning with spacecraft landing through medical evaluations completed at the Kennedy Space Center. The medical information presented was derived from postflight medical examinations, tilt-table studies on each crewmember, routine laboratory studies, and postflight ergometry studies on the pilot. Variations from normal included the following:

(a) Slight to moderate crew fatigue

(b) Subjective dehydration

(c) Marked diaphoresis

(d) Weight loss

(e) Labile pulse pressure and elevated heart rate during initial postflight tilt-table study

(f) Bilateral conjunctivitis (command pilot only)

(g) Scleral icterus and elevated serum bilirubin (command pilot only).

7.2.3.1 Recovery medical activities.- The recovery medical activities for Gemini XII were essentially unchanged from other Gemini rendezvous missions.

7.2.3.1.1 Planned procedures: Because of the late time of day for recovery, the usual plan for medical procedures was slightly modified. Only one tilt-table study was scheduled for recovery day. The second postflight tilt was scheduled after return of the crewmen to the Kennedy Space Center approximately 20 hours after recovery. Subsequent tilt-table studies were planned daily thereafter until crewmember responses returned to preflight values. Laboratory procedures were limited to routine chest roentgenograms and the collection of blood and urine specimens. The roentgenograms and the prepared laboratory specimens were to be airlifted to Cape Kennedy for interpretation. The postflight medical examination was to be less comprehensive than those performed following long-duration Gemini flights; therefore, only the internist-cardiologist member of the medical evaluation team was deployed on the primary recovery ship. Additional medical evaluations were to be performed as indicated by the NASA physicians and the Department of Defense members of the recovery team.

7.2.3.1.2 Recovery procedures: The U.S.S. Wasp was stationed in the prime recovery area. Personnel aboard the ship were able to observe the spacecraft on the main parachute and during landing. The spacecraft was easily visible throughout the remainder of the recovery operations, since the ship was positioned less than three miles away. The seas were moderate, with 2-foot waves and 4-foot swells. Because of the immediate availability of the recovery helicopters and the expeditious retrieval of the crew, no postlanding motion sickness was experienced by either crewman. As a result of a problem in the spacecraft water system, both crewmembers were thirsty during the postlanding period because of reduced water intake on the last day of flight. After spacecraft landing, the crew elected to egress the spacecraft and board the raft as soon as the swimmers had secured the flotation collar. Egress was performed without difficulty, and both crewmembers remained in their pressure suits. They were hoisted aboard the helicopter within 25 minutes after spacecraft landing. Portable space-suit cooling and ventilation units were available in the helicopter and were used by both crewmembers while in flight. Shortly thereafter, when the helicopter landed on the deck of the prime recovery ship, the crew walked without difficulty from the helicopter to the ship's medical area. They gave no indication of ill effects from their space flight and reported no symptoms suggestive of orthostatic hypertension either on the water or on the deck of the recovery ship. The command pilot experienced none of the heaviness in his legs present during the Gemini VII postrecovery period.

7.2.3.2 Examinations.- Postflight medical examinations were completed approximately two hours after the crew arrived on the deck of the recovery ship. Both crewmembers showed evidence of slight to moderate fatigue and slight dehydration. During the de-suiting process, it was noted that the undergarments of both crewmembers were saturated with perspiration. One sternal electrocardiogram sensor on the pilot was completely detached. All sensors on the command pilot appeared intact. The skin of both crewmembers was normal except for minimal reaction at the sensor sites. Skin cultures were taken from several sites for comparison with preflight cultures. The results of these cultures are reported in table 7.2-III. The command pilot showed a few slight pressure points over the proximal interphalangeal joints of the right hand and on his chin. He had a slight area of swelling tenderness in the right infraclavicular and pectoral region, and a slight nummular abrasion was noted on the dorsal aspect of the right forearm. He had a mild to moderate bilateral conjunctivitis and blepharitis. The pilot showed slight erythema of the infraclavicular notch and the right lateral thorax. On the dorsal surface of his left forearm, there was a ring-shaped pressure area which was caused by the suit pressure gage. During the flight, the command pilot lost 6-1/2 pounds and the pilot lost 7-1/4 pounds. These weights were determined by subtracting the weights determined aboard the recovery ship from the weights obtained during the launch-day physical examinations. Neither crewman was found to have any significant postflight change in visual acuity. As noted in section 7.2.1.1, both crewmen have a history of intermittent indirect hyperbilirubinemia. Neither crewmember had clinical evidence of this condition preflight; however, the recovery internist found slight icterus on examining the command pilot's sclera immediately postflight. An elevated postflight serum bilirubin is seen in table 7.2-I. The pilot showed no postflight clinical evidence of hyperbilirubinemia. The remainder of the examinations and laboratory studies showed no significant changes from the preflight evaluations except for slightly high protein on the command pilot. The laboratory findings are listed in table 7.2-I.

7.2.3.3 Tilt-table studies.- Three postflight tilt-table studies were performed on each crewmember. The results and times are presented in figures 7.2-1 and 7.2-2. During the second preflight tilt on November 5, the command pilot showed marked lability. This was attributed to the fact that he was in the prodromal period of acute upper respiratory infection for which he was subsequently treated. The response to the tilt-table studies of both crewmembers was considered to have returned to within the normal envelop by the third postflight tilt. In contrast to his Gemini VII experience, the command pilot demonstrated no pre-syncopal tendency during any of these tilt-table studies.

7.2.3.4 Bicycle ergometry studies.- A bicycle ergometry study was performed on the pilot approximately 21 hours after spacecraft landing. The result of these studies is shown in figure 7.2-3.

TABLE 7.2-I.- HEMATOLOGY

(a) Command pilot

Determination	Preflight		Postflight	
	October 31, 1966	November 5, 1966	November 15, 1966 4:25 p.m. e.s.t.	November 16, 1966 Recovery + 24 hours
Hematocrit, percent	46.8	49.3	52	48
Hemoglobin, gm percent	15.6	15.7	17	16
Red blood cells/mm^3	4.65	5.48	5.72	5.33
Indices:				
Mean corpuscular volume, μ^3	101	90	91	90
Mean corpuscular hemoglobin, $\gamma\gamma$	33.5	28.6	29.7	30
Mean corpuscular hematocrit, percent	33.4	31.8	32.7	33.3
White blood cells/mm^3	5900	7800		
Neutrophiles, percent	54	56		
Lymphocytes, percent	40	40		
Monocytes, percent	4	2		
Eosinophiles, percent	2	2		
Basophiles, percent	-	-		
Reticulocytes, percent	1.4	1.3		
Volume received, cc	1.5	2.1	1.8	3.6
Color	Slight icteric	Slight icteric	Slight icteric	Slight icteric
Appearance	No precipitate	No precipitate	Some precipitate	No precipitate
Occult blood	Moderate amount	Small amount	Moderate amount	Small amount
Sodium, mEq/l	153	143	140	
Potassium, mEq/l	4.9	4.6	4.3	
Chlorine, mEq/l	109	103	99	
Protein, total, gm percent	8.3	7.7	8.9	8.5
Urea nitrogen, mg percent	18	16	21	24

TABLE 7.2-I.- HEMATOLOGY - Continued

(a) Command pilot

Determination	Preflight		Postflight	
	October 31, 1966	November 5, 1966	November 15, 1966 4:25 p.m. e.s.t.	November 16, 1966 Recovery + 24 hours
Total bilirubin, mg percent	QNS	1.1	2.0	1.9
Direct bilirubin, mg percent	QNS	0.2	0.3	0.3
Osmolality, mOs/kg	312	QNS	291	293
Calcium, mEq/l	5.1	QNS	QNS	QNS
Calcium, mg percent	10.0	QNS	QNS	QNS
Cholesterol, mg percent	184	QNS	QNS	201
Uric acid, mg percent	QNS	QNS	QNS	6.0
Alkaline phosphatase B and L units				1.4
Albumin, percent total protein	68	68	67	66
α1, percent total protein	4.5	3.3	3.1	2.8
α2, percent total protein	7.4	7.4	8.5	8.3
β, percent total protein	8.0	7.8	8.7	9.5
γ, percent total protein	13.2	14.2	13.2	13.5
Albumin, gm percent protein	5.6	5.2	6.0	5.7
α1, gm percent protein	0.38	0.25	0.28	0.24
α2, gm percent protein	0.61	0.57	0.76	0.71
β, gm percent protein	0.66	0.60	0.78	0.81
γ, gm percent protein	1.1	1.1	1.2	1.2

QNS = quantity not sufficient.

TABLE 7.2-I.- HEMATOLOGY - Continued

(b) Pilot

Determination	Preflight		Postflight	
	October 31, 1966	November 5, 1966	November 15, 1966 4:25 p.m. e.s.t.	November 16, 1966 Recovery + 24 hours
Hematocrit, percent	45	47.3	46.5	43.6
Hemoglobin, grams percent	15.6	15.1	15	14.6
Red blood cells/mm^3	4.79	5.33	4.75	4.53
Indices:				
Mean corpuscular volume, μ^3	94	89	97	96
Mean corpuscular hemoglobin, $\gamma\gamma$	32.688	28.388	31.688	32.288
Mean corpuscular hematocrit, percent	34.7	31.9	32.3	33.5
White blood cells/mm^3	5700	7150		
Neutrophiles, percent	58	62		
Lymphocytes, percent	37	31		
Monocytes, percent	1	3		
Eosinophiles, percent	4	3		
Basophiles, percent		1		
Reticulocytes, percent	0.6	0.9		
Volume received, cc	1.6	2.1	1.1	3.7
Color	Normal	Normal	Yellow	Normal
Appearance	No precipitate	No precipitate	Some precipitate	No precipitate
Occult blood	Small amount	Small amount	Negative	Trace
Sodium, mEq/l	140	146	138	144
Potassium, mEq/l	4.8	4.4	4.8	4.5
Chlorine, mEq/l	102	104	102	105
Protein, total, gm percent	7.1	7.3	7.0	7.6
Urea nitrogen, mg percent	19	19	20	21

TABLE 7.2-I.- HEMATOLOGY - Concluded

(b) Pilot

Determination	Preflight		Postflight	
	October 31, 1966	November 5, 1966	November 15, 1966 4:25 p.m. e.s.t.	November 16, 1966 Recovery + 24 hours
Total bilirubin, mg percent	QNS	1.2	QNS	1.3
Direct bilirubin, mg percent	QNS	0.2	QNS	0.3
Osmolality, mOs/kg	298	QNS	295	303
Calcium, mEq/l	4.7	QNS	QNS	4.9
Calcium, mg percent	9.4	QNS	QNS	9.8
Cholesterol, mg percent	188	QNS	238	202
Uric acid, mg percent	6.8	QNS		
Alkaline phosphatase B and L units				
Albumin, percent total protein	70	71	70	71
α1, percent total protein	4.8	5.2	3.3	3.6
α2, percent total protein	7.3	7.1	7.3	7.5
β, percent total protein	8.4	7.4	8.7	8.1
γ, percent total protein	10.3	10.6	11.0	10.7
Albumin, gm percent protein	5.0	5.2	4.9	5.4
α1, gm percent protein	0.34	0.38	0.24	0.28
α2, gm percent protein	0.52	0.52	0.51	0.57
β, gm percent protein	0.60	0.54	0.61	0.62
γ, gm percent protein				

QNS = quantity not sufficient.

TABLE 7.2-II.- FOOD AND WATER

Day	Time, g.e.t., hr	Food, calories		Water, lb[a]	
		Command pilot	Pilot	Command pilot	Pilot
1	0 to 7	657	657	1.54	1.54
2	7 to 29	2514	2514	6.59	6.59
3	29 to 52	2533	2533	8.00	8.00
4	52 to 76	1621	2513	5.00	5.00
5	76 to 86	236	1622	3.00	3.00

[a]The water consumption is estimated assuming that each crewman drank half of the total water consumed during the day.

TABLE 7.2-III.- MICROBIOLOGICAL EVALUATION

(a) Command pilot

Area of body sampled	Preflight (November 11, 1966)		Postflight (November 15, 1966)	
	Microorganisms isolated and identified	Organisms per 25 cm^2	Microorganisms isolated and identified	Organisms per 25 cm^2
1. Dorsal area right foot	Staphylococcus epidermidis	184	Bacillus badius	29
2. Dorsal area left foot	Staphylococcus epidermidis Corynebacterium Alternaria	33	Staphylococcus epidermidis	55
3. Right popliteal fossa	Staphylococcus epidermidis	39	Staphylococcus epidermidis Corynebacterium	100
4. Left popliteal fossa	Staphylococcus epidermidis	113	Staphylococcus aureus	(a)
5. Right inguinal area	Staphylococcus epidermidis	263	Staphylococcus epidermidis E. coli	(a)
6. Left inguinal area	Staphylococcus epidermidis	291	Staphylococcus aureus	(a)
7. Right axillary region	Staphylococcus epidermidis	(a)	Staphylococcus epidermidis	(a)
8. Left axillary region	Staphylococcus epidermidis	(a)	Staphylococcus epidermidis Staphylococcus aureus	(a)
9. Top of right hand	Staphylococcus epidermidis	33	Staphylococcus epidermidis	410
10. Top of left hand	Micrococcus Mycelia sterila	26	Staphylococcus epidermidis	450
11. Hair line behind right ear	Staphylococcus epidermidis	182	Staphylococcus epidermidis	(a)
12. Hair line behind left ear	Staphylococcus epidermidis	(a)	Staphylococcus epidermidis Staphylococcus aureus	(a)

[a]Too numerous to count.

TABLE 7.2-III.- MICROBIOLOGICAL EVALUATION - Continued

(a) Command pilot

Area of body sampled	Preflight (November 11, 1966)		Postflight (November 15, 1966)	
	Microorganisms isolated and identified	Organisms per 25 cm^2	Microorganisms isolated and identified	Organisms per 25 cm^2
13. External auditory canal	Staphylococcus epidermidis		Staphylococcus epidermidis	
14. Nose	Staphylococcus epidermidis		Staphylococcus epidermidis	
15. Throat	Alpha-hemolytic streptococci		Staphylococcus aureus Neisseria catarrhalis Alpha-hemolytic streptococci	

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TABLE 7.2-III.- MICROBIOLOGICAL EVALUATION - Continued

(b) Pilot

Area of body sampled	Preflight (November 11, 1966)		Postflight (November 15, 1966)	
	Microorganisms isolated and identified	Organisms per 25 cm^2	Microorganisms isolated and identified	Organisms per 25 cm^2
1. Dorsal area right foot	Corynebacterium Staphylococcus epidermidis	125	Staphylococcus epidermidis Micrococcus varians Micrococcus	79
2. Dorsal area left foot	Staphylococcus epidermidis Mycelia sterila	66	Staphylococcus epidermidis	495
3. Right popliteal fossa	Mycelia sterila	76	Staphylococcus epidermidis Corynebacterium	(a)
4. Left popliteal fossa	Staphylococcus epidermidis Sarcina flava Pseudomonas	27	Staphylococcus epidermidis Corynebacterium	(a)
5. Right inguinal region	Staphylococcus epidermidis	205	Staphylococcus epidermidis Corynebacterium	238
6. Left inguinal region	Staphylococcus epidermidis	114	Staphylococcus epidermidis	(a)
7. Right axillary region	Staphylococcus epidermidis	9	Staphylococcus epidermidis	375
8. Left axillary region	Staphylococcus epidermidis	2	Staphylococcus epidermidis Micrococcus Corynebacterium	84
9. Top of right hand	Staphylococcus epidermidis	15	Micrococcus Staphylococcus epidermidis Pencillium	136
10. Top of left hand	Staphylococcus epidermidis Corynebacterium	117	Staphylococcus aureus Corynebacterium	39 39

[a]Too numerous to count.

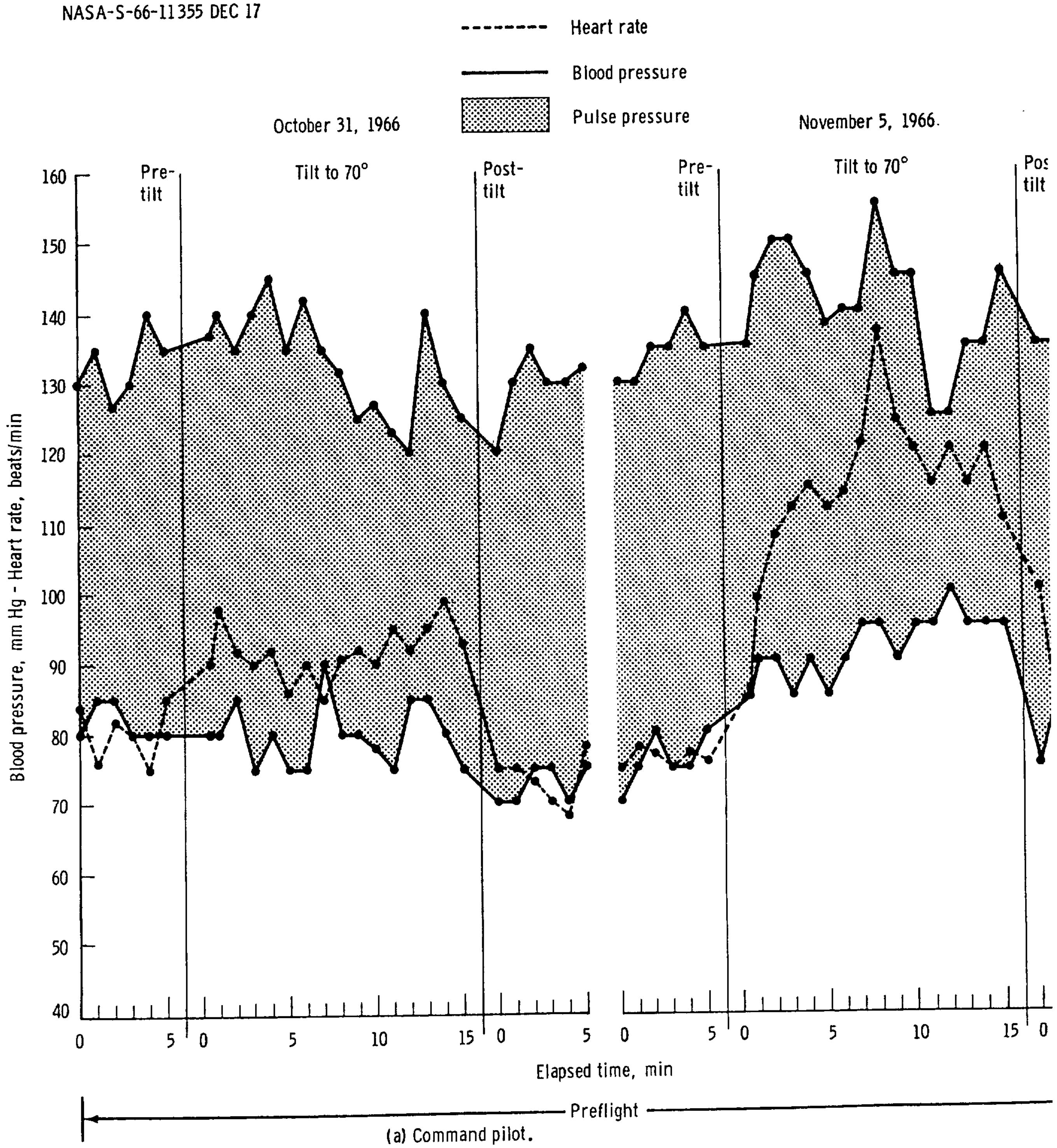
UNCLASSIFIED

TABLE 7.2-III.- MICROBIOLOGICAL EVALUATION - Concluded

(b) Pilot

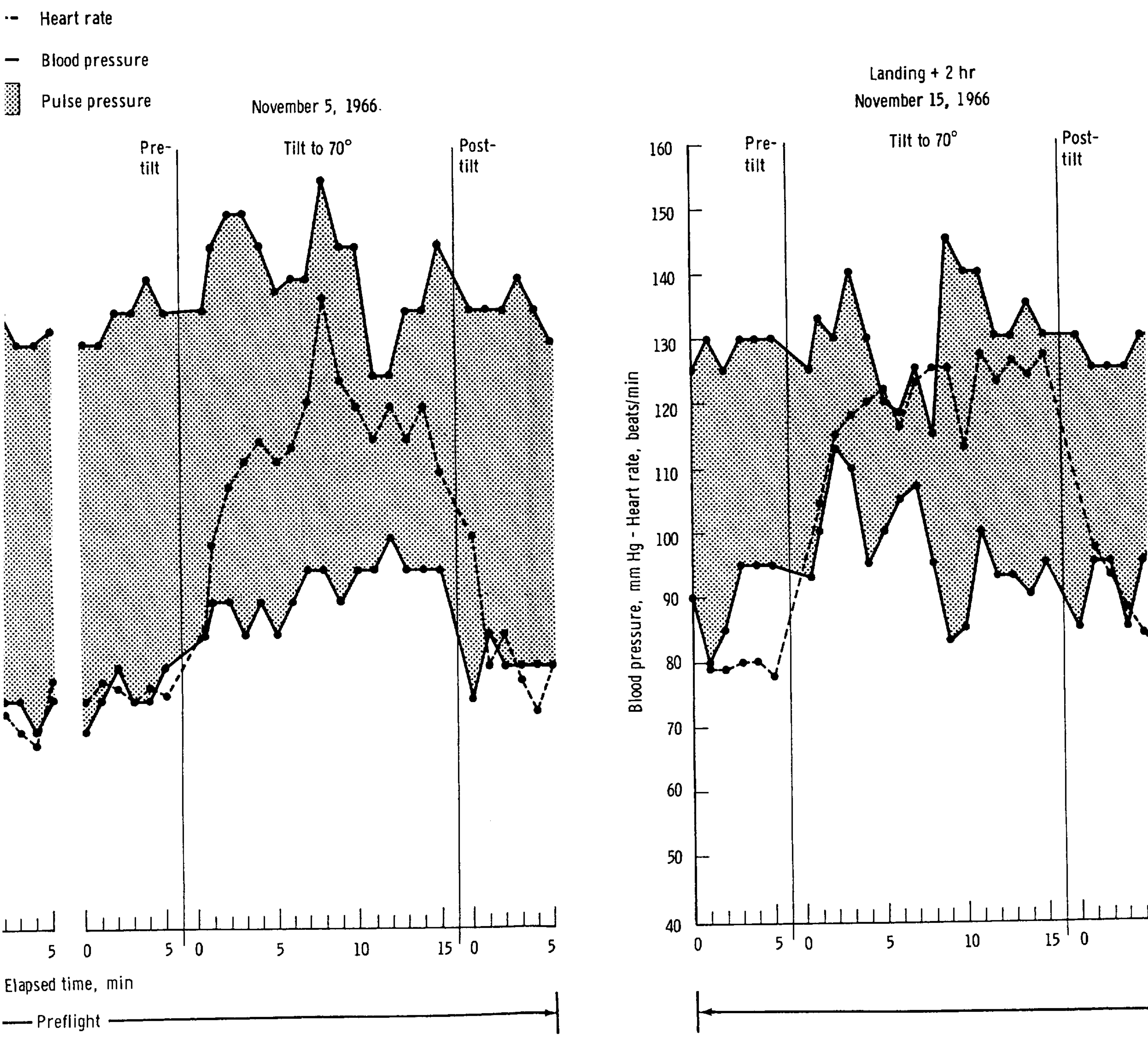
Area of body sampled	Preflight (November 11, 1966)		Postflight (November 15, 1966)	
	Microorganisms isolated and identified	Organisms per 25 cm^2	Microorganisms isolated and identified	Organisms per 25 cm^2
11. Hair line behind right ear	Staphylococcus epidermidis Corynebacterium	124	Staphylococcus epidermidis Corynebacterium	210
12. Hair line behind right ear	Staphylococcus epidermidis	143	Corynebacterium Staphylococcus epidermidis Bacillus firmus	410
13. External auditory canal	Staphylococcus epidermidis		Staphylococcus epidermidis	
14. Nose	Staphylococcus epidermidis		Staphylococcus epidermidis	
15. Throat	Alpha-hemolytic streptococci		Neisseria Alpha-hemolytic streptococci	

NASA-S-66-11355 DEC 17



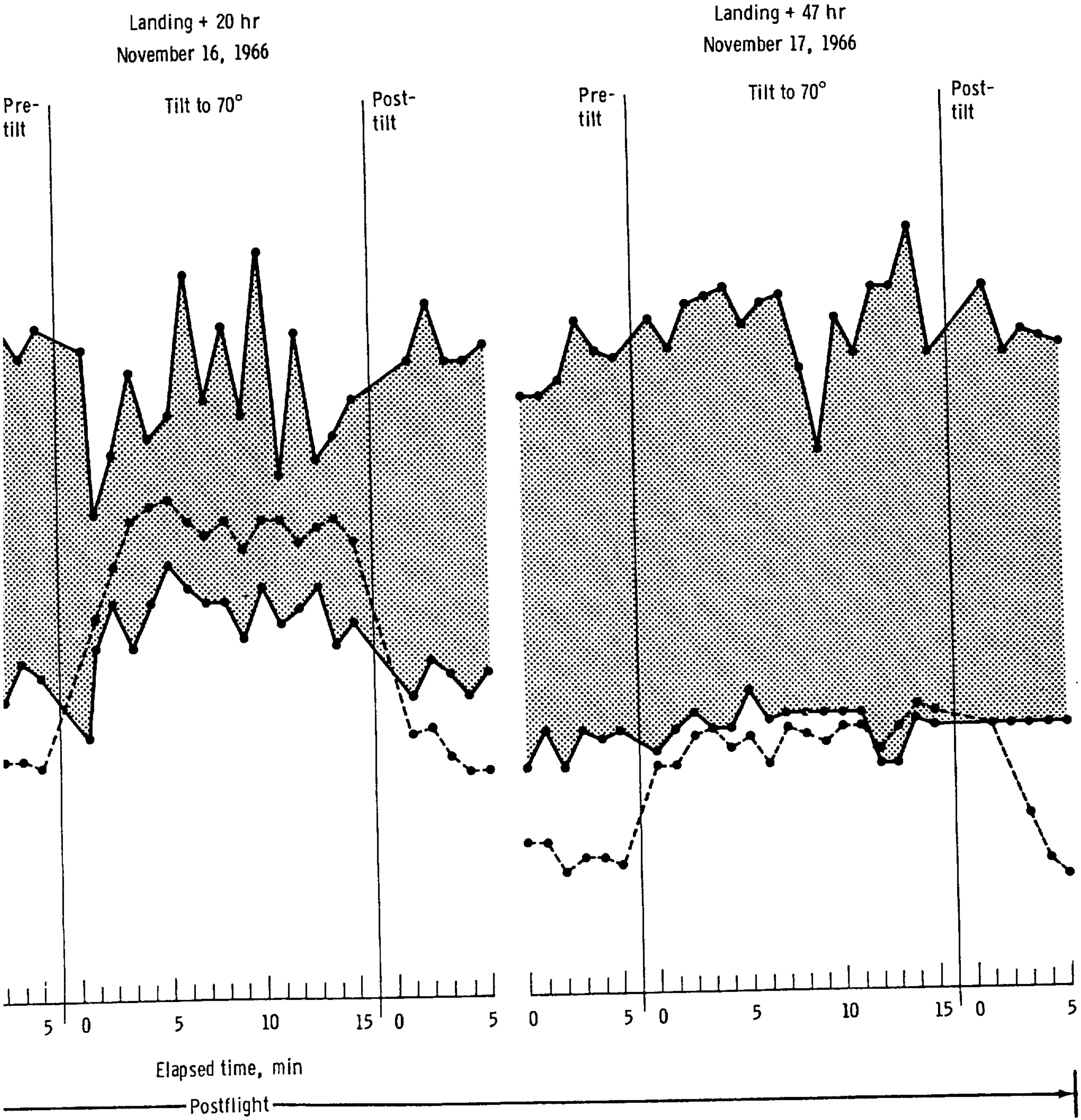
(a) Command pilot.

Figure 7.2-1. - Tilt table studies.



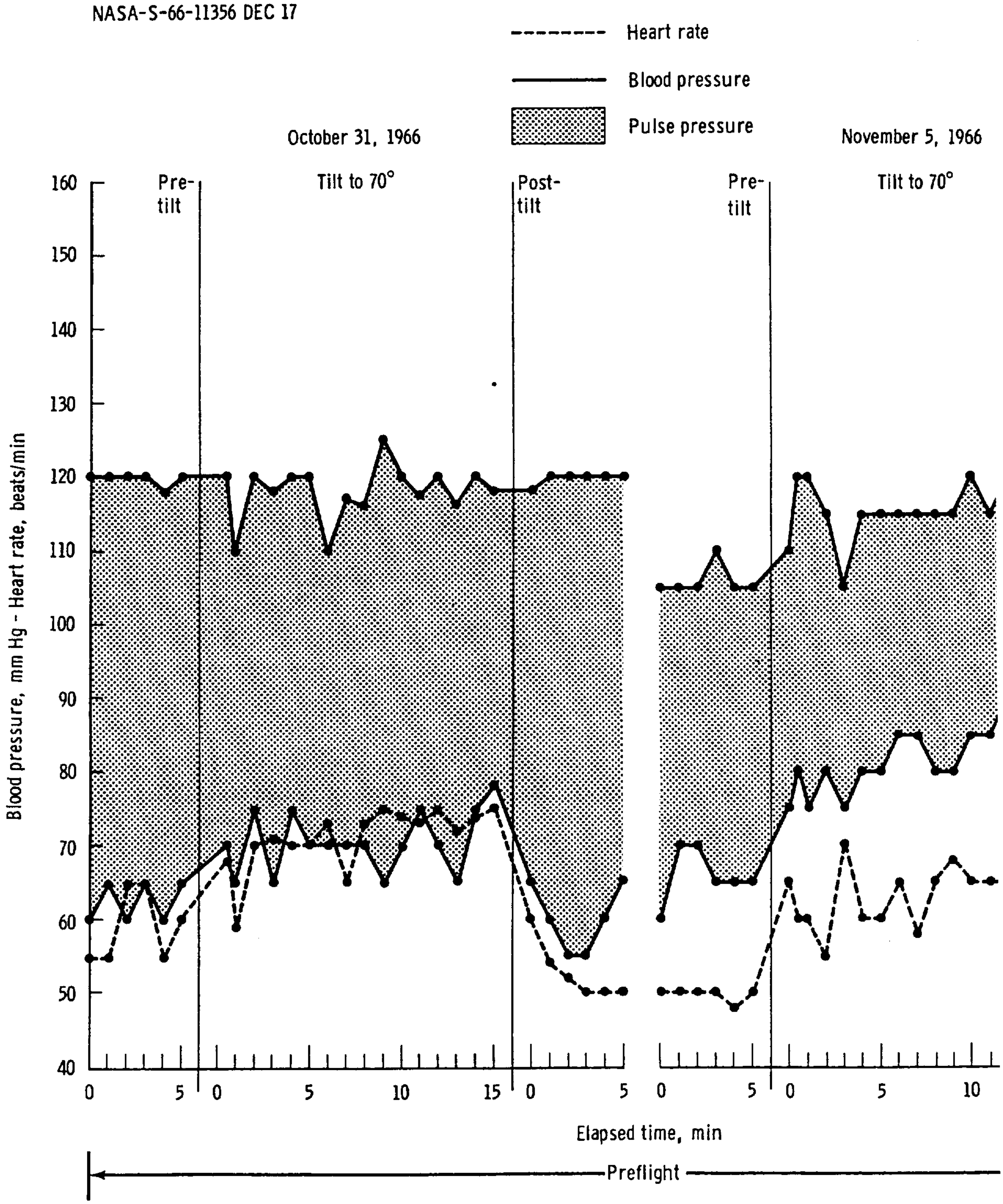
Heart rate
Blood pressure
Pulse pressure
November 5, 1966
Landing + 2 hr
November 15, 1966
Pre-tilt
Tilt to 70°
Post-tilt
Blood pressure, mm Hg - Heart rate, beats/min
160
150
140
130
120
110
100
90
80
70
60
50
40
0
5
10
15
Elapsed time, min
Preflight

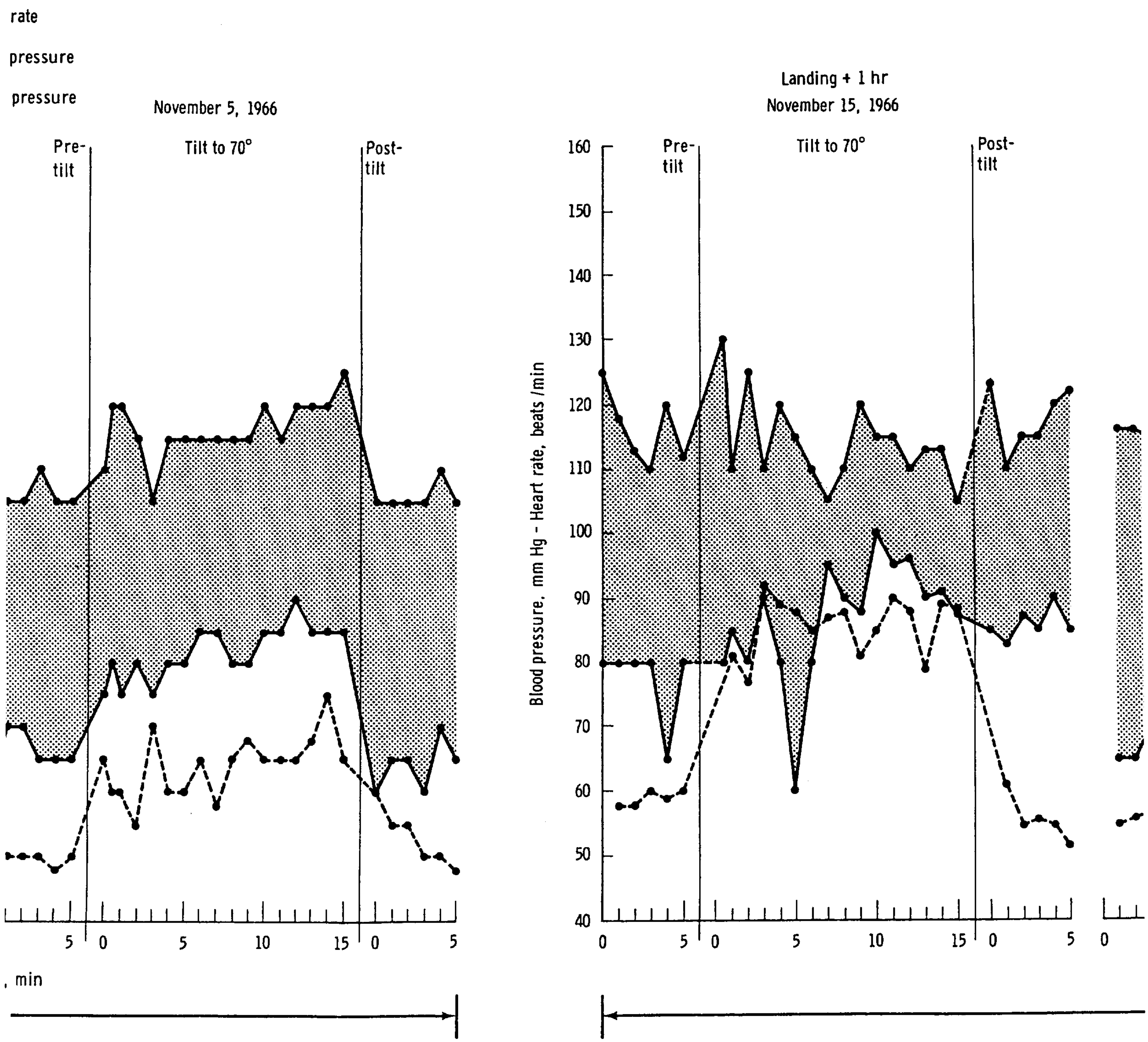
studies.

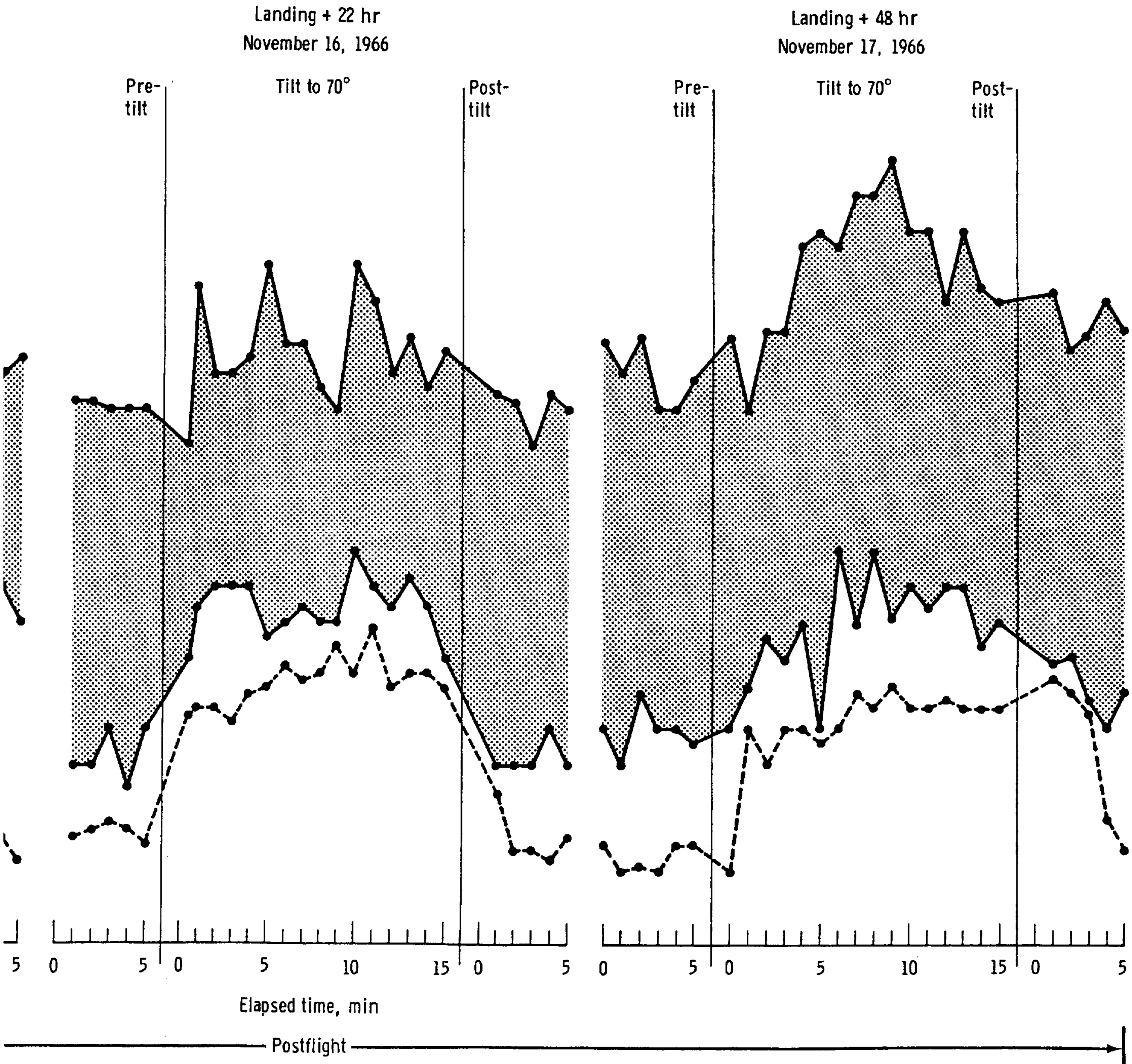


Landing + 20 hr
November 16, 1966
Landing + 47 hr
November 17, 1966
Pre-tilt
Tilt to 70°
Post-tilt
Pre-tilt
Tilt to 70°
Post-tilt
5 0 5 10 15 0 5
0 5 0 5 10 15 0 5
Elapsed time, min
Postflight

NASA-S-66-11356 DEC 17



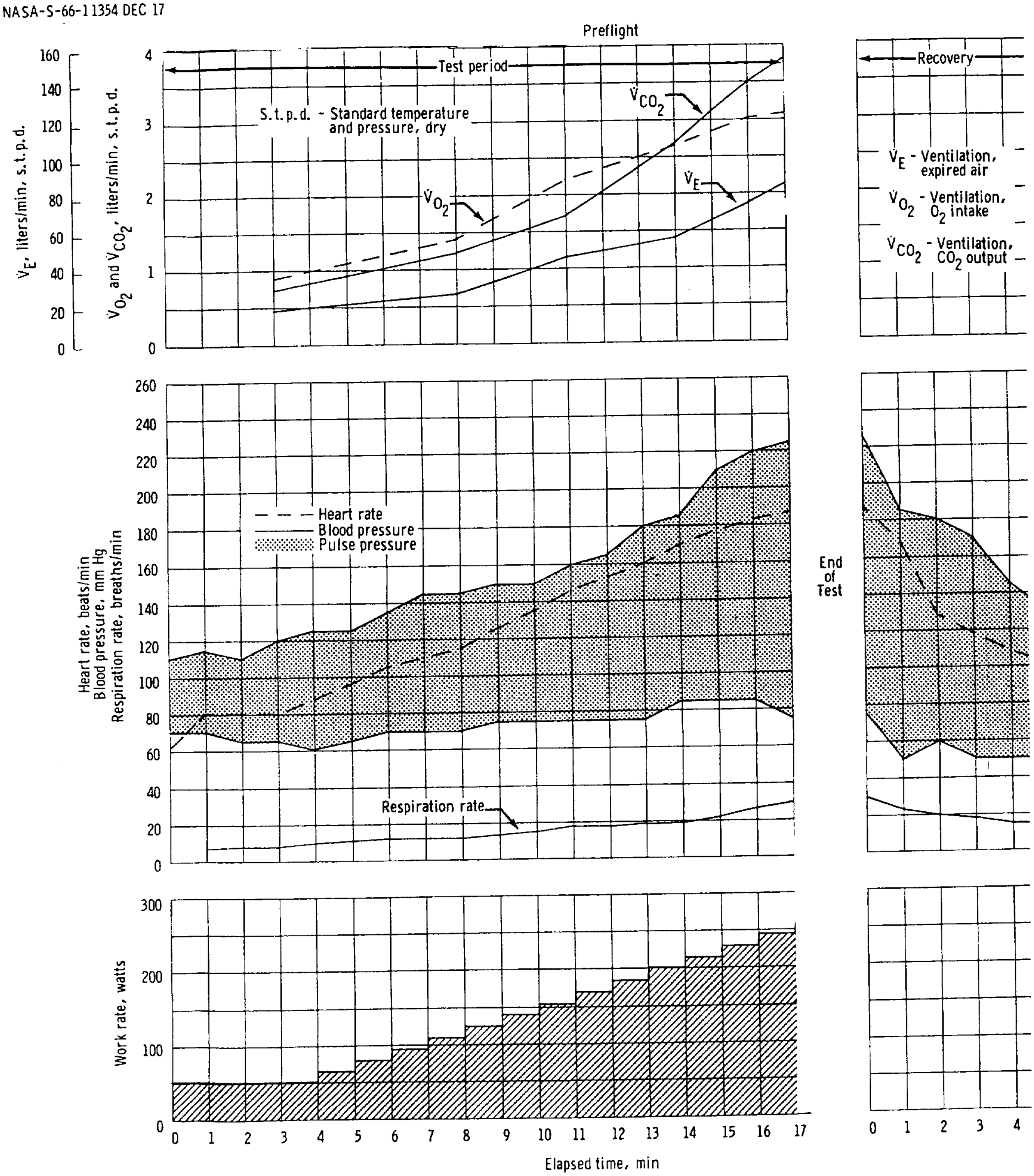

rate
pressure
pressure
November 5, 1966
Pre-
tilt
Tilt to 70°
Post-
tilt
5
0
5
10
15
0
5
, min
Landing + 1 hr
November 15, 1966
Blood pressure, mm Hg - Heart rate, beats /min
160
150
140
130
120
110
100
90
80
70
60
50
40
Pre-
tilt
Tilt to 70°
Post-
tilt
0
5
0
5
10
15
0
5
0



(b) Pilot.

Figure 7.2-1. - Concluded.

NASA-S-66-11354 DEC 17



/

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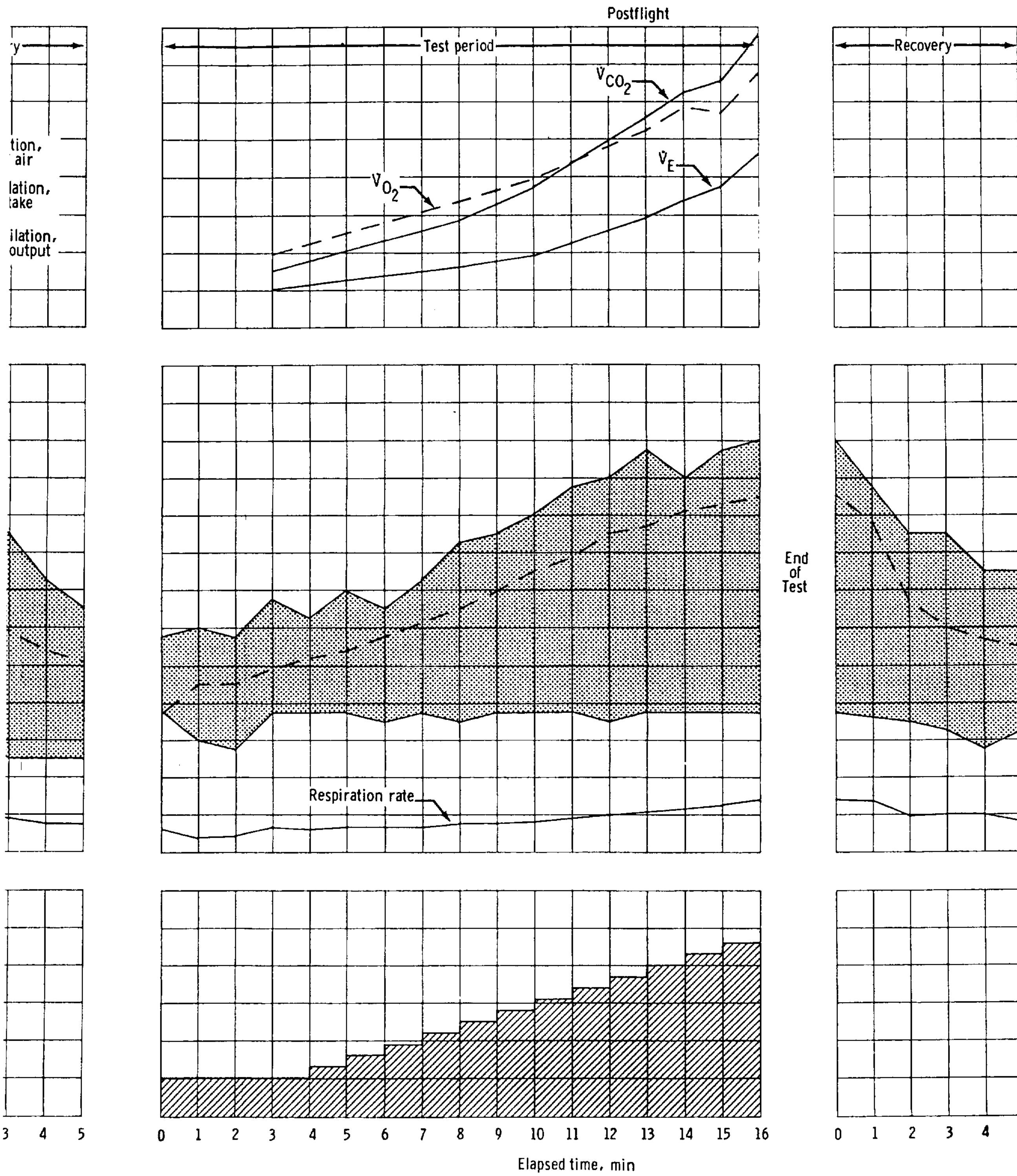


Figure 7.2-2. - Exercise capacity test result, pilot.

NASA-S-66-11346 DEC 16

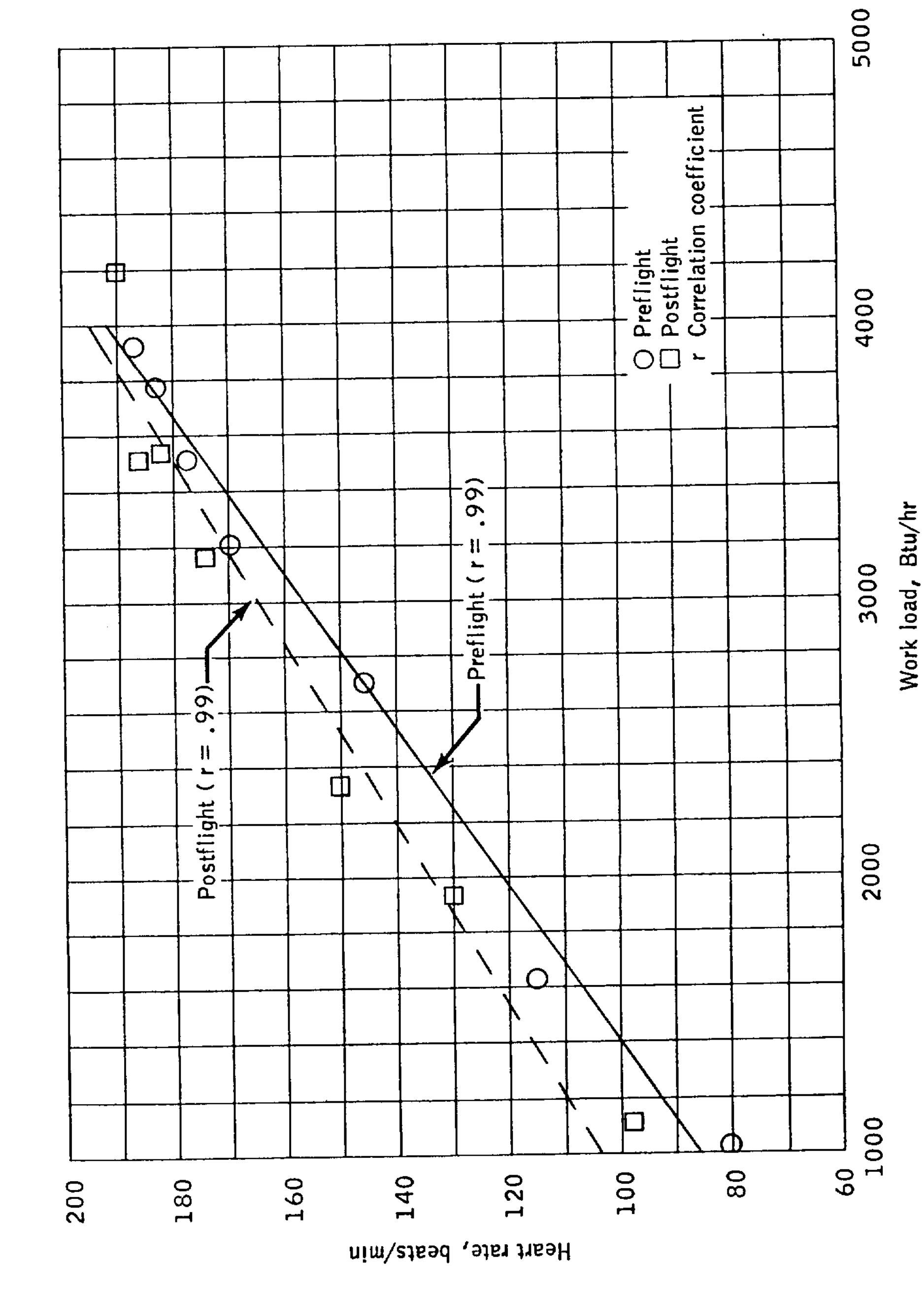
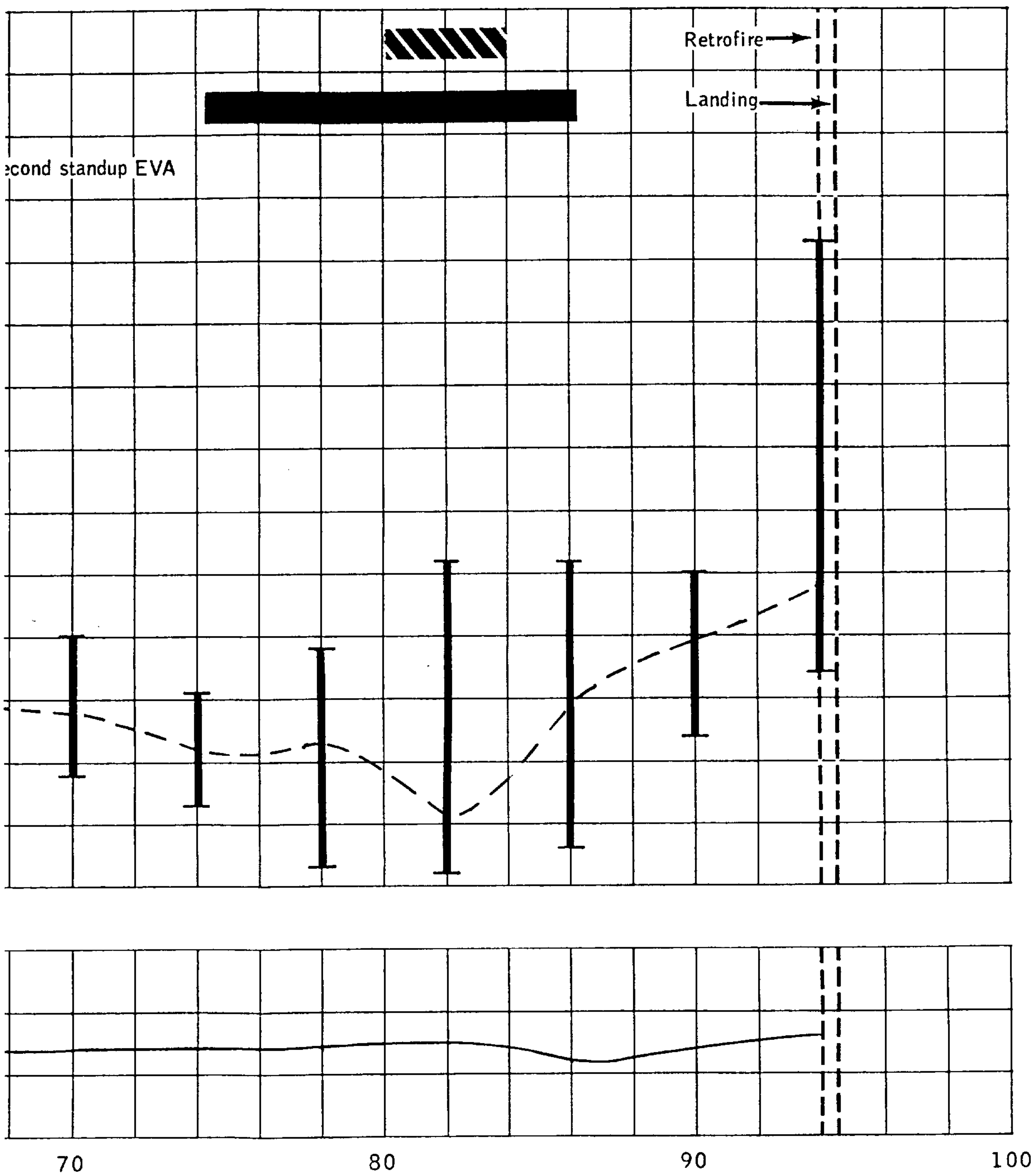
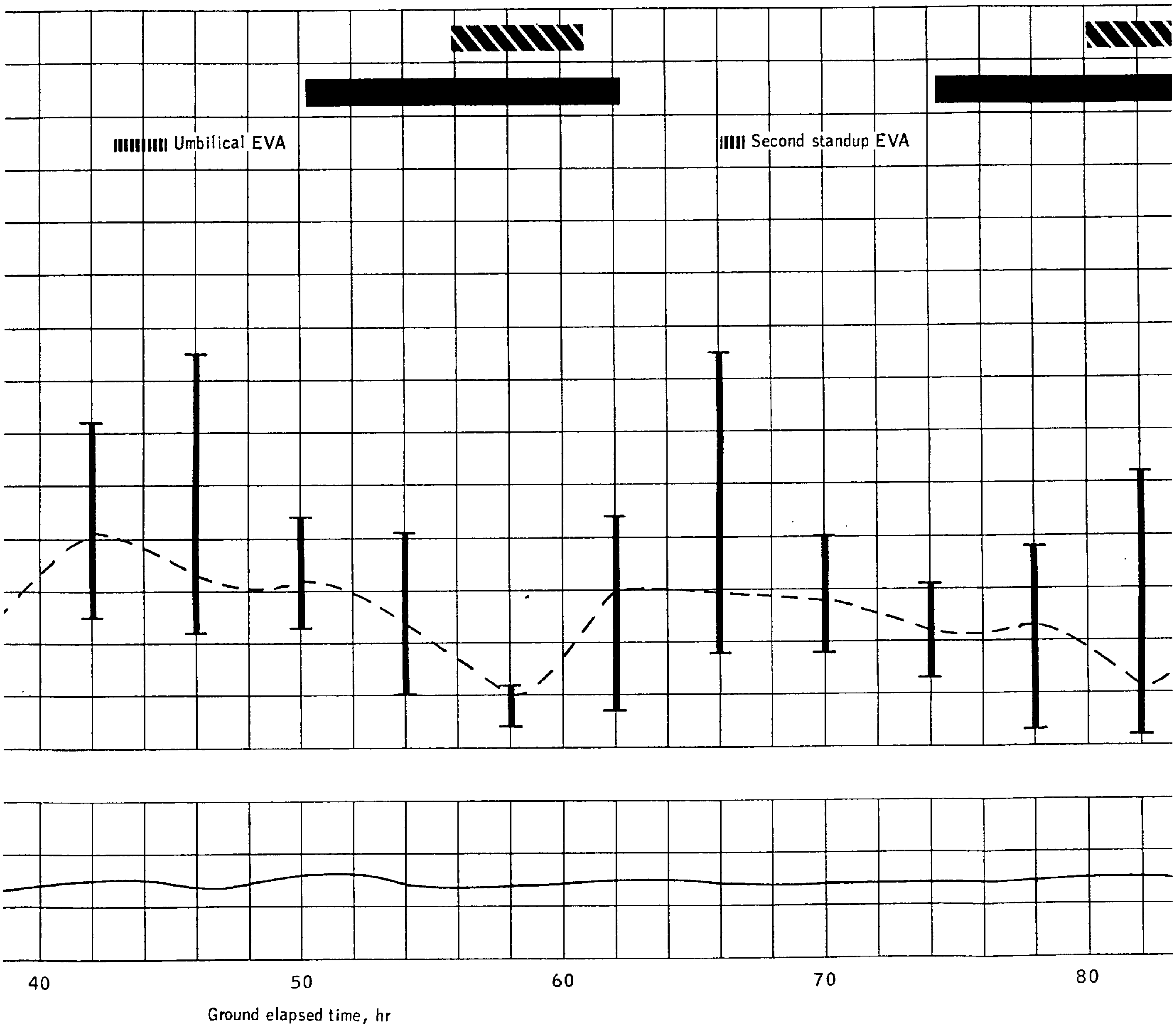
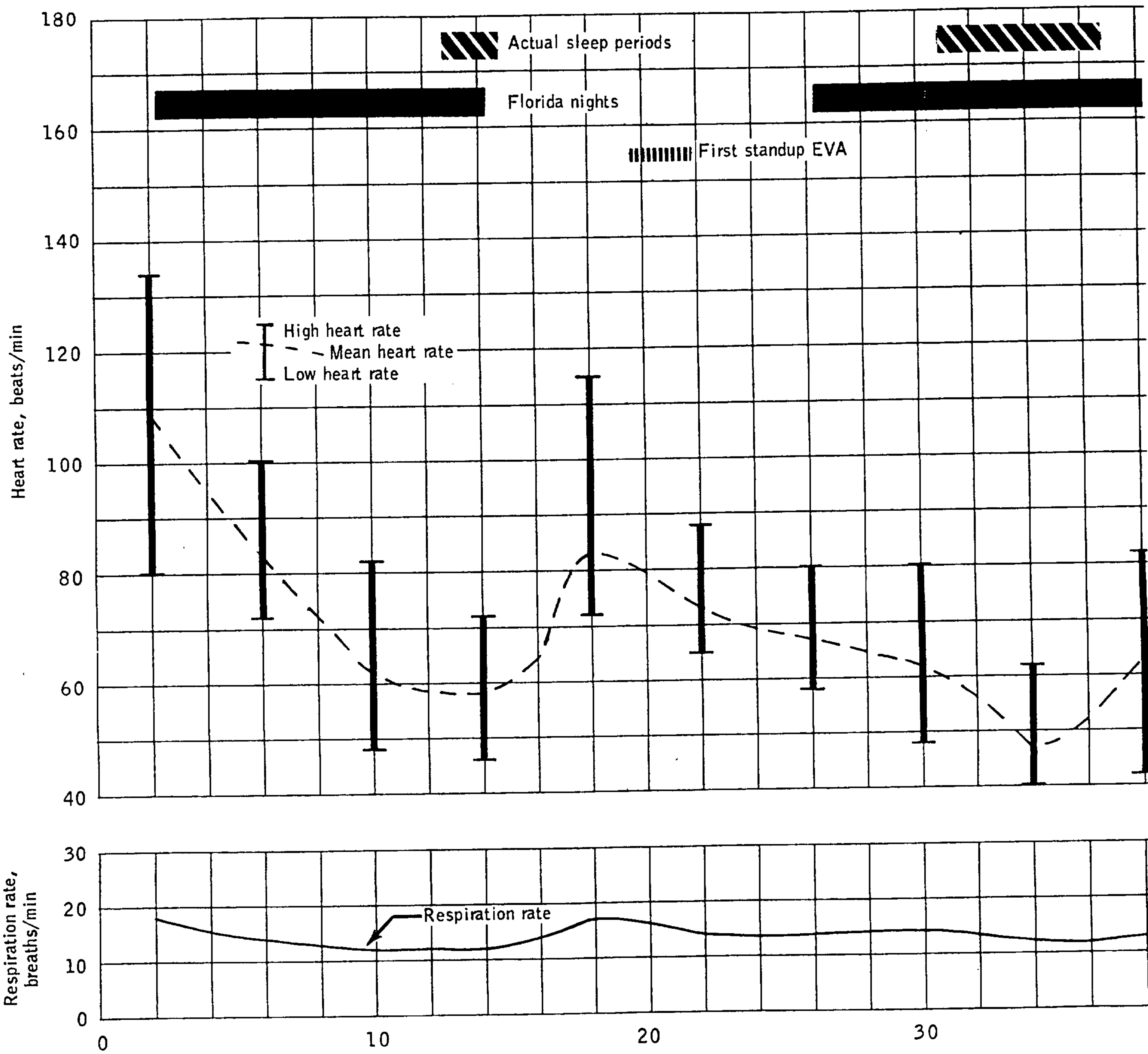


Figure 7.2-3.- Exercise studies on the Gemini XII pilot.


Retrofire
Landing
econd standup EVA
70
80
90
100


Umbilical EVA
Second standup EVA
40
50
60
70
80
Ground elapsed time, hr

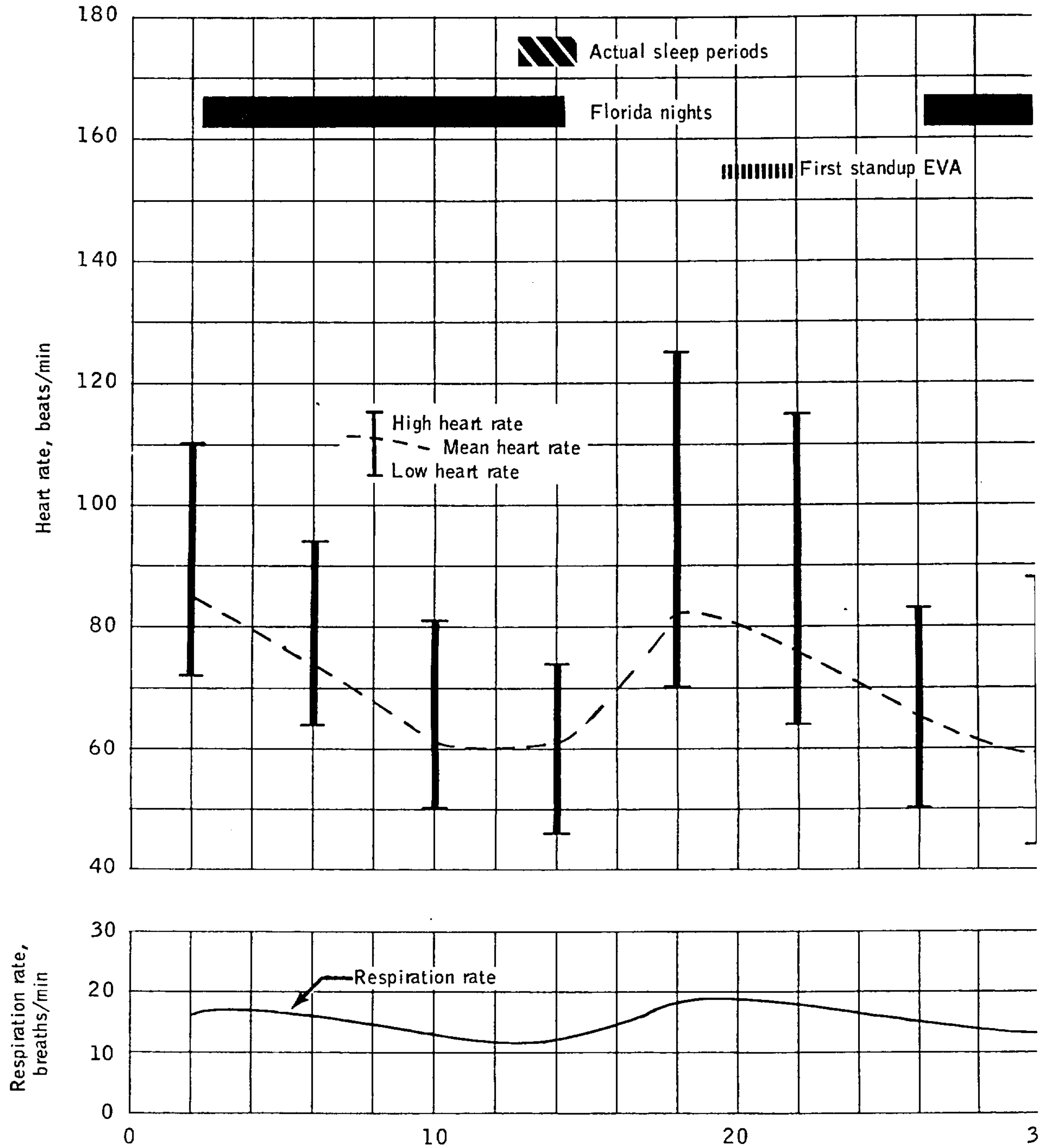
NASA-S-66-11293 DEC 9

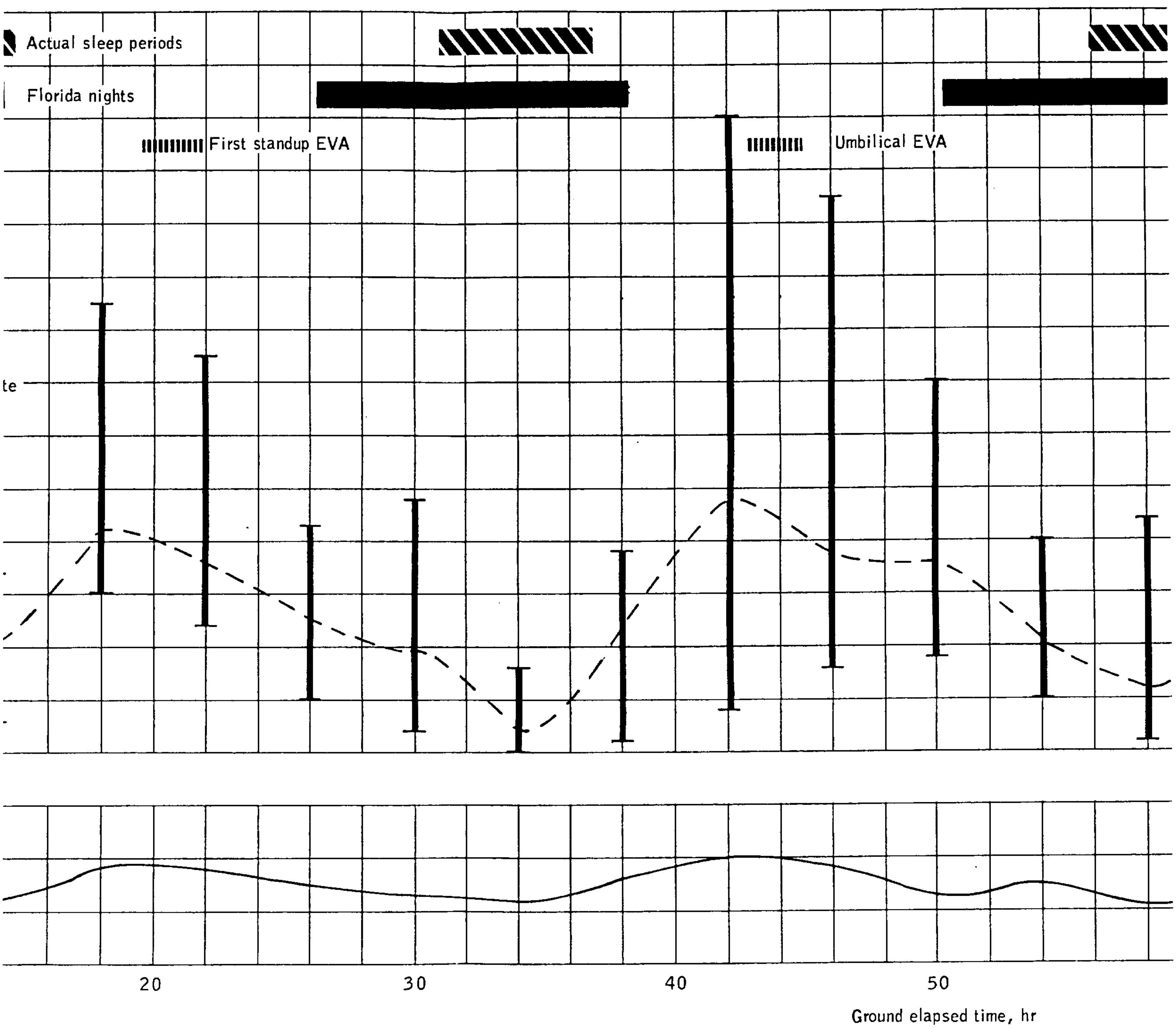


(a) Command pilot.

Figure 7. 2-4.- Physiological measurements.

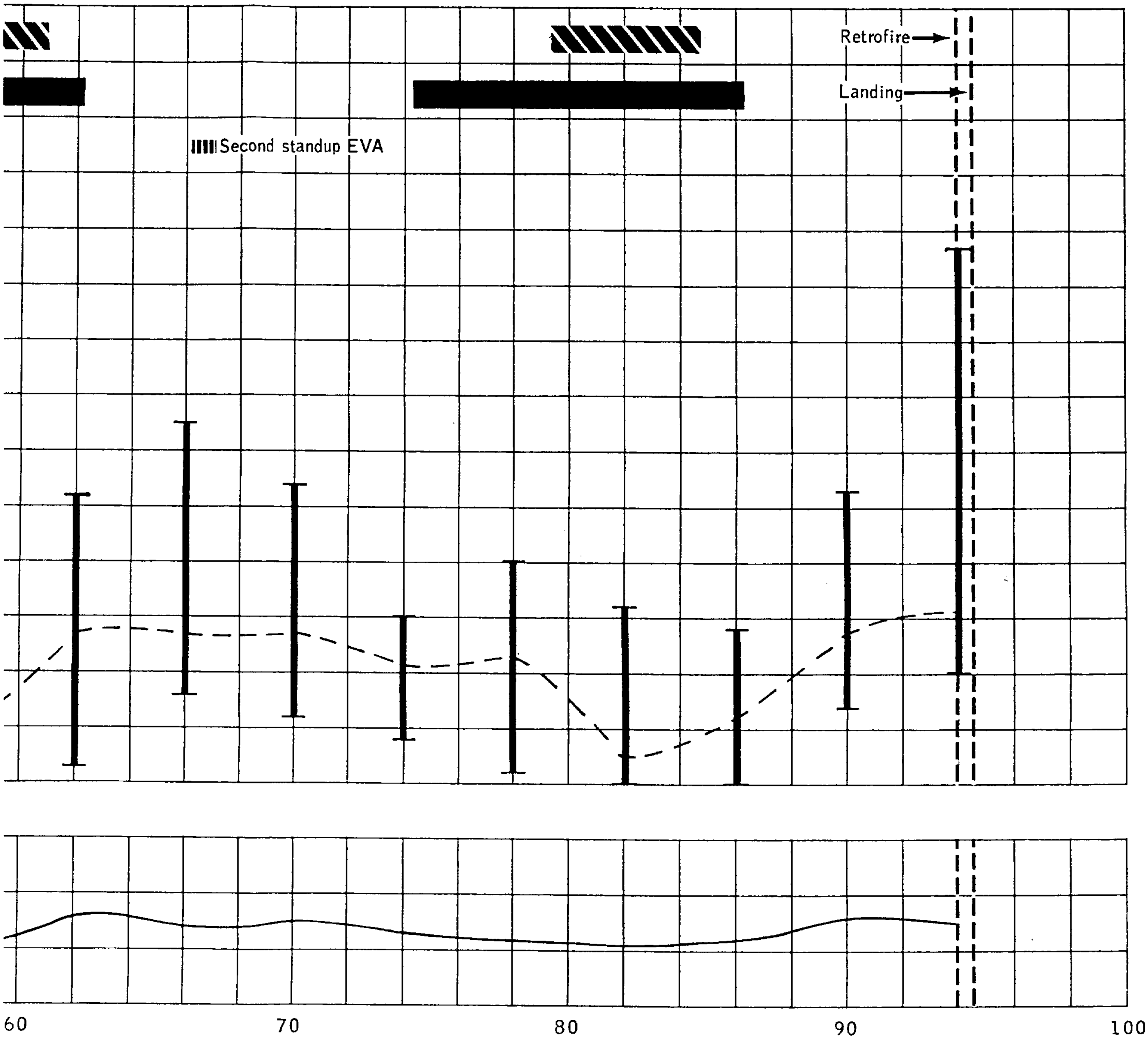
NASA-S-66-11294 DEC 9





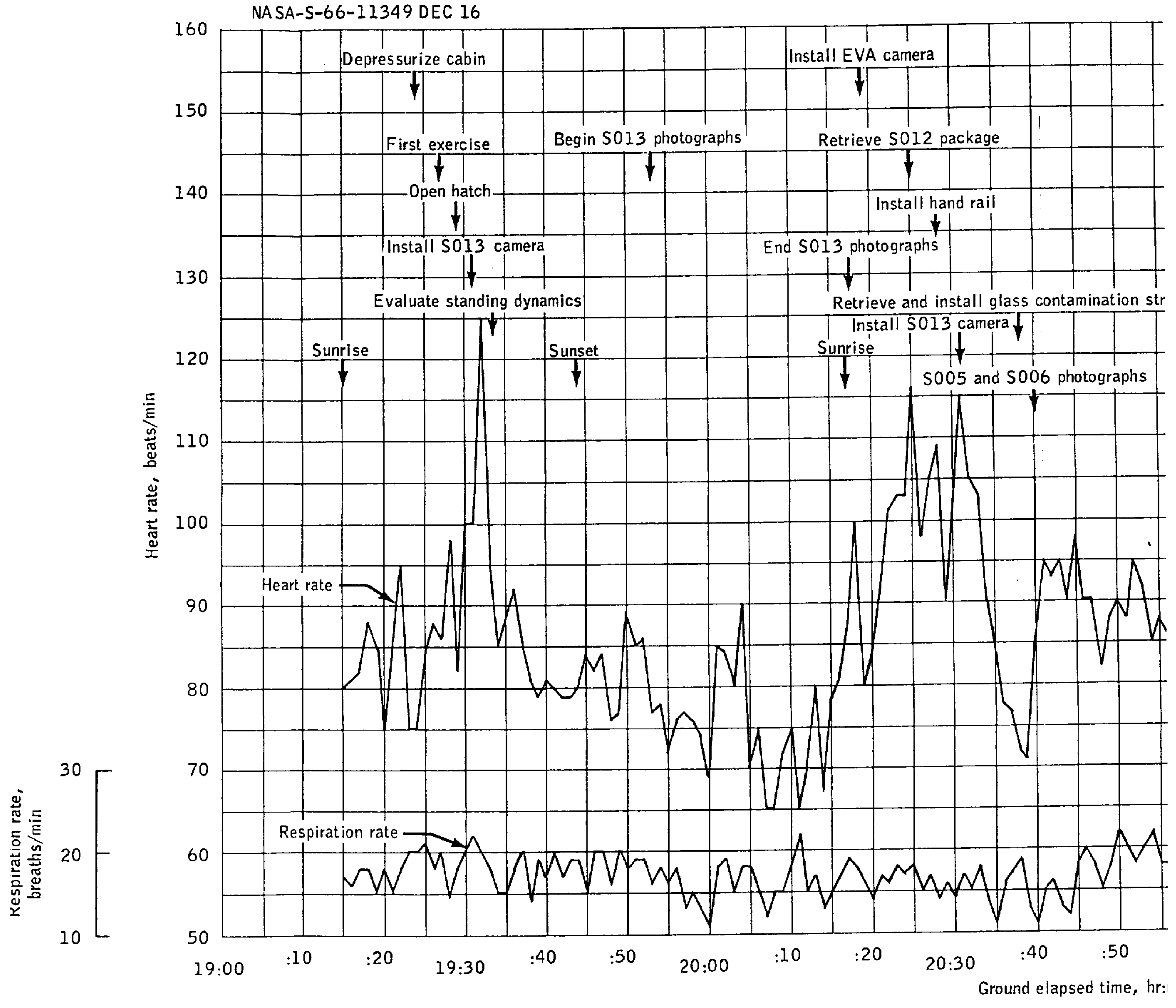
Actual sleep periods
Florida nights
First standup EVA
Umbilical EVA
te
20
30
40
50
Ground elapsed time, hr

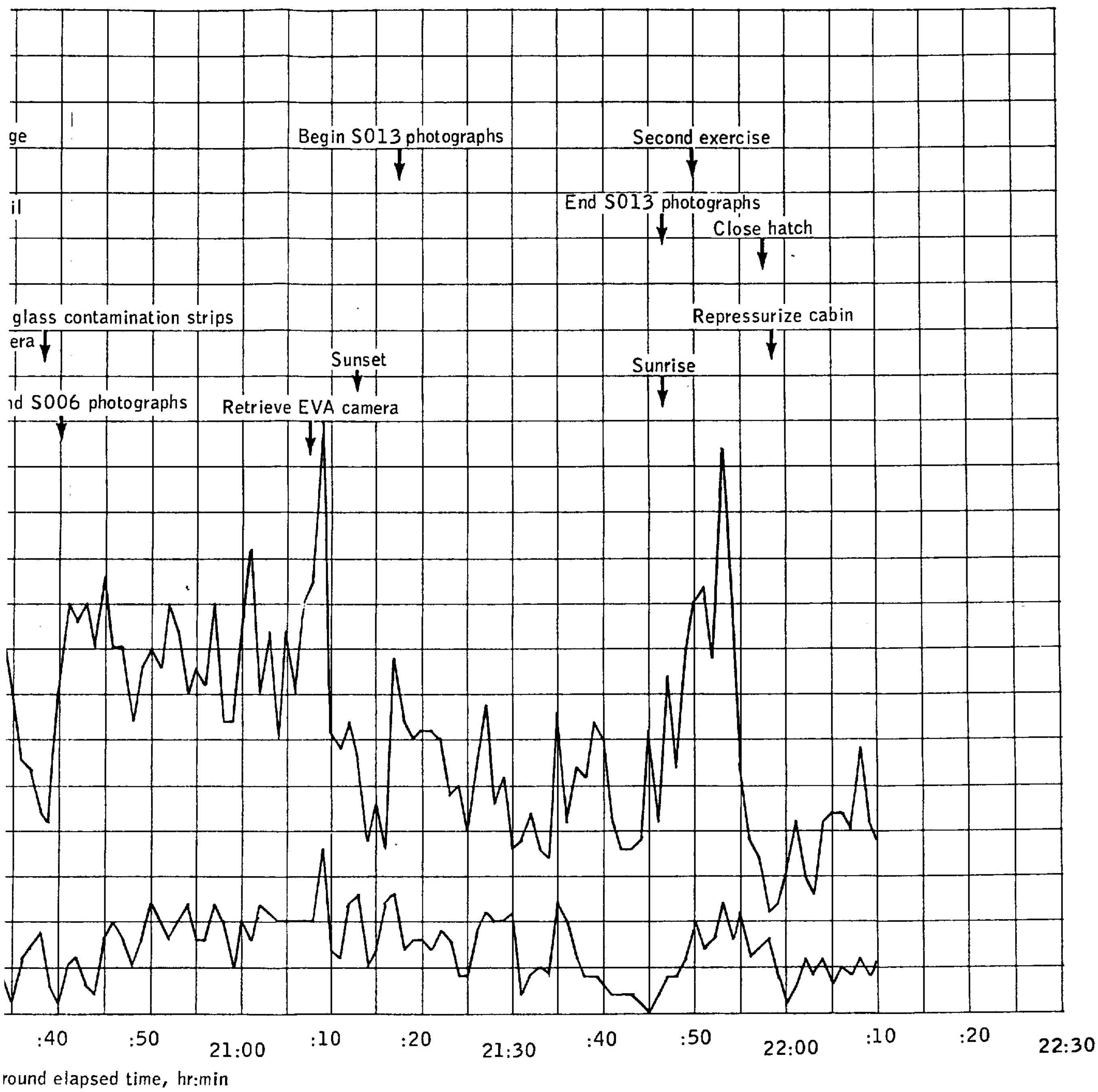
2



(b) Pilot.

Figure 7. 2-4.- Concluded.

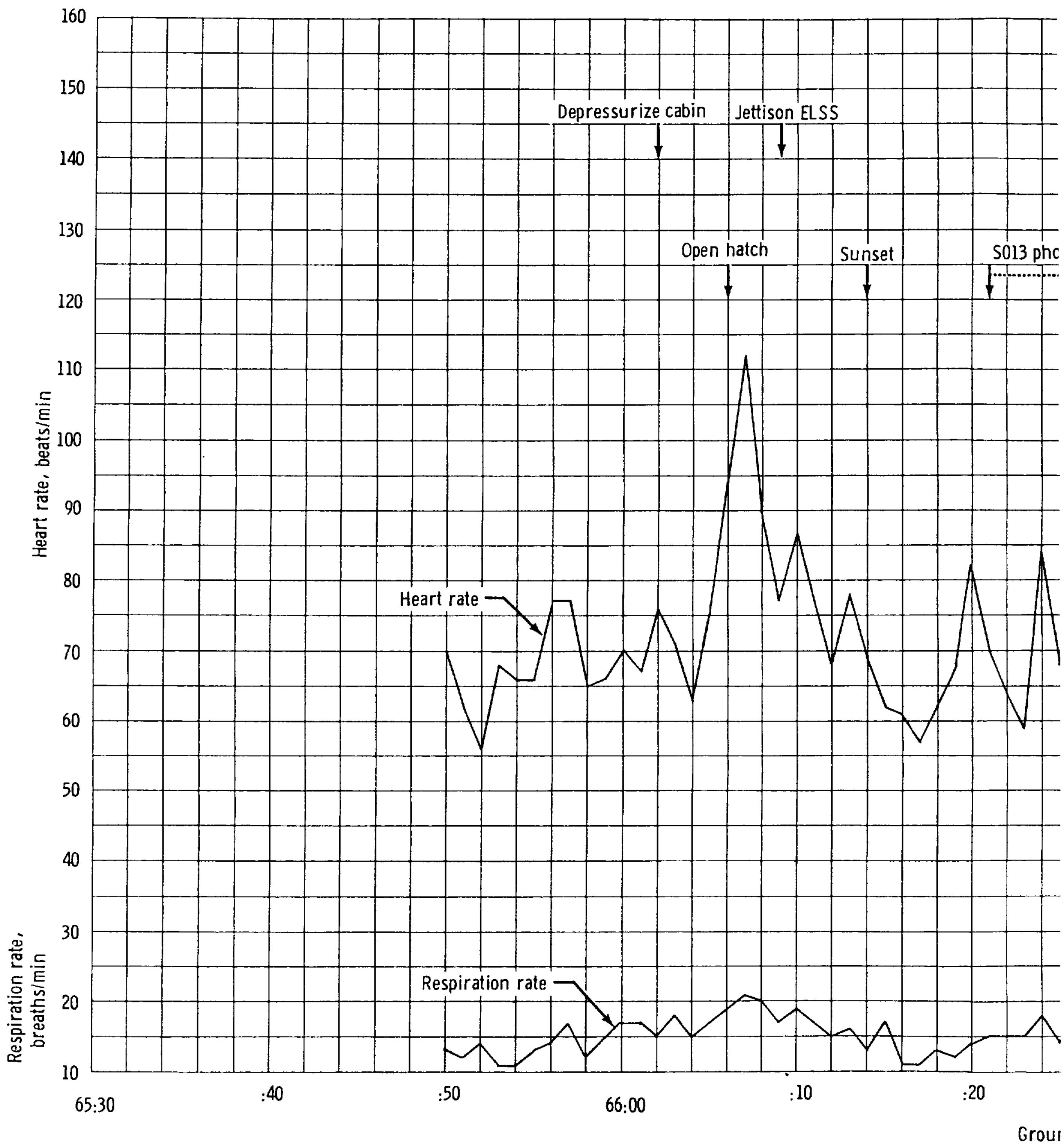

NASA-S-66-11349 DEC 16
Depressurize cabin
First exercise
Open hatch
Install S013 camera
Evaluate standing dynamics
Sunrise
Begin S013 photographs
Sunset
Install EVA camera
Retrieve S012 package
Install hand rail
End S013 photographs
Retrieve and install glass contamination str
Install S013 camera
Sunrise
S005 and S006 photographs
Heart rate
Respiration rate
Heart rate, beats/min
Respiration rate, breaths/min
160
150
140
130
120
110
100
90
80
70
60
50
30
20
10
19:00
:10
:20
19:30
:40
:50
20:00
:10
:20
20:30
:40
:50
Ground elapsed time, hr:

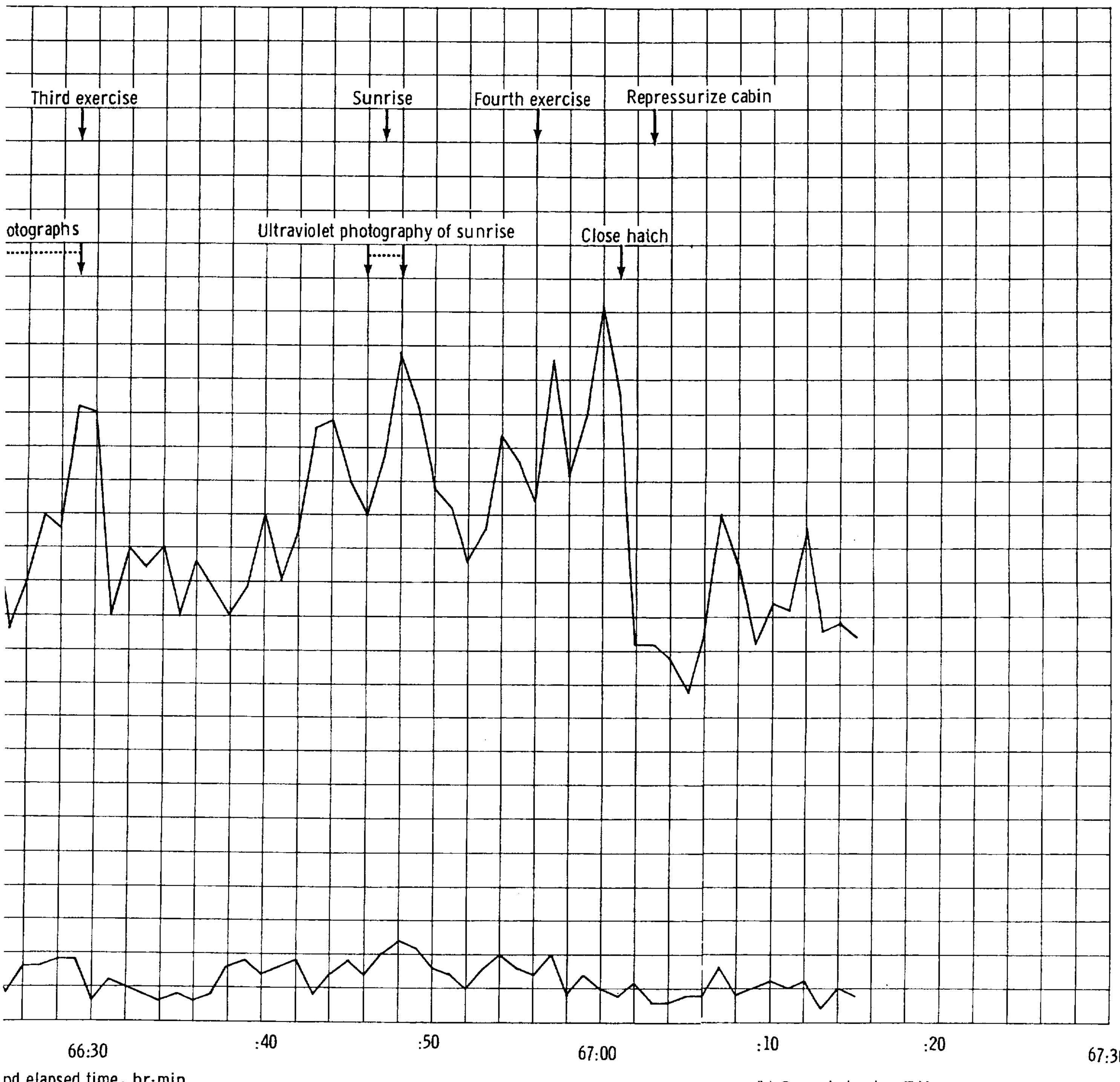


(a) First standup EVA.

Figure 7.2-5.- Physiological data during standup EVA, pilot.

NASA-S-66-11352 DEC 16

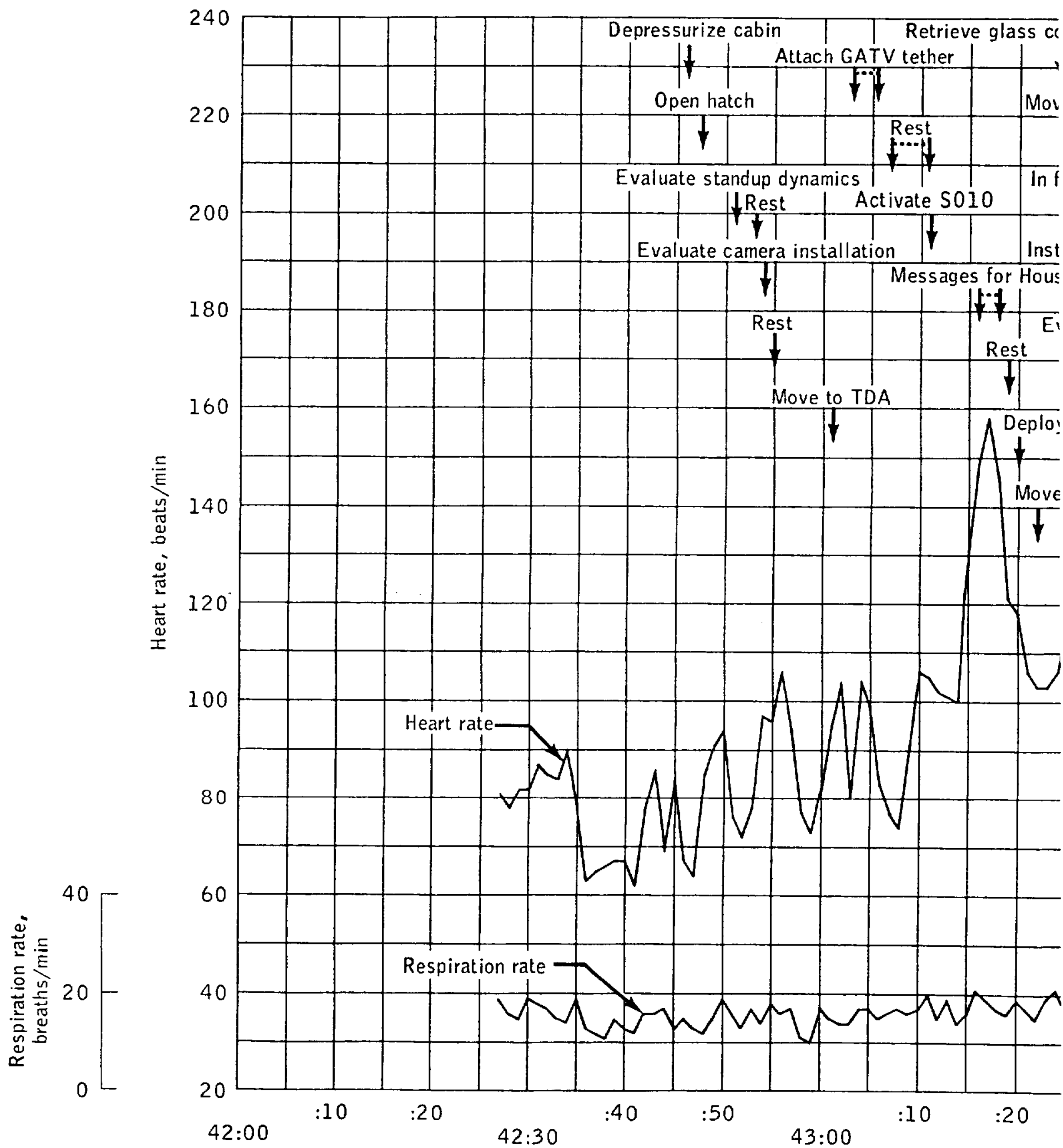




(b) Second standup EVA.

Figure 7.2-5. - Concluded.

NASA-S-66-11361 DEC 16



/

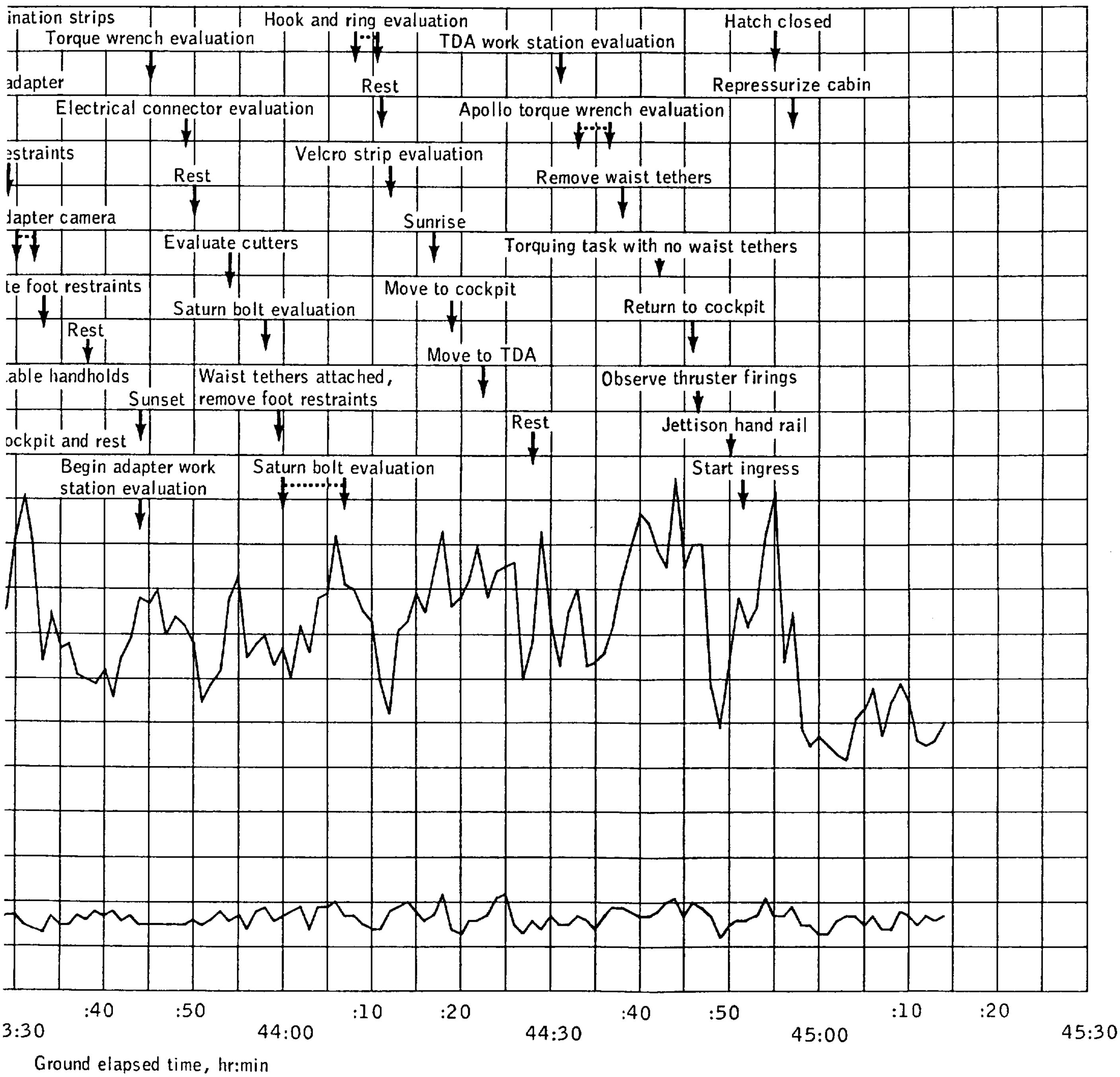


Figure 7.2-6.- Physiological data during umbilical EVA, pilot.

NASA-S-66-11302 DEC 9

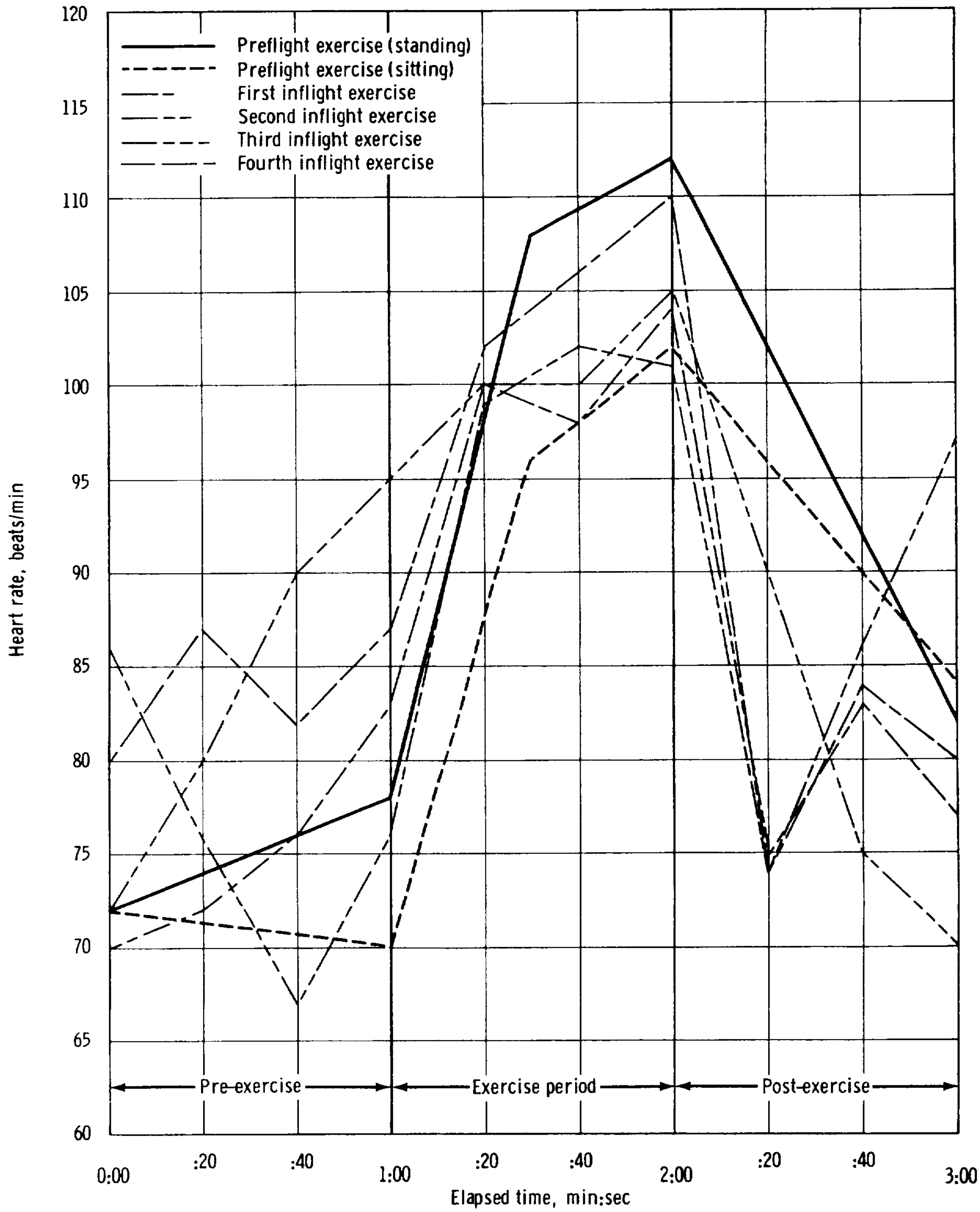


Figure 7.2-7. - Preflight and inflight exercise studies.

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8.0 EXPERIMENTS

Fourteen scientific or technological experiments were planned for the Gemini XII mission. During the second standup extravehicular activity (EVA), the pilot performed an additional non-scheduled activity—taking ultraviolet photography of predicted dust clouds within the earth's upper atmosphere. Table 8.0-I lists, in alphanumeric order, the 14 experiments performed and shows the experiment title, sponsoring agency, principal investigator, and qualitative success met during this mission. The actual schedule of experiment operations is shown in table 8.0-II and was reconstructed from the preflight plans, real-time flight plan updates, onboard voice tapes, mission notes, crew flight logs, and technical and scientific debriefings.

Preliminary analyses of available photographic and telemetry data indicate that the fundamental objectives were obtained for 11 of the 14 scheduled experiments. The S010 Agena Micrometeorite Collection experiment was opened by the pilot during umbilical EVA; however, it will probably not be retrieved because reentry of the target vehicle is estimated to occur prior to Apollo earth orbital missions. The S029 Libration Regions Photography and S051 Daytime Sodium Cloud experiments were successfully completed operationally. Because of camera malfunctions, the exposures obtained were not of usable quality for scientific analysis.

Each experiment is described in the sections that follow, with the success or incompleteness so indicated. For most experiments, detailed evaluation requires several months of data analyses and correlations. Only preliminary results are, therefore, reported for those experiments. Specific scientific or technological reports are published at 90-day, 6-month, and other appropriate intervals as the analyses are completed.

TABLE 8.0-I.- EXPERIMENTS ON GEMINI XII

Experiment number	Experiment title	Principal investigator	Sponsor	Data obtained	Completion of planned objectives
D010	Ion-Sensing Attitude Control	Air Force Cambridge Research Laboratory, Hanscom AFB, Massachusetts	Department of the Air Force, Detachment 2, Space Systems Division (AFSC)	Seven periods of useful data	Completed
M405	Tri-Axis Magnetometer	NASA Manned Spacecraft Center, Space Sciences Division	NASA/MSC	Data available during all revolutions	Completed
M408	Beta Spectrometer	NASA Manned Spacecraft Center, Space Sciences Division	NASA/MSC	One 5-minute period of useful data	Partial
M409	Bremsstrahlung Spectrometer	NASA Manned Spacecraft Center, Space Sciences Division	NASA/MSC	Data available during all revolutions	Completed
S003	Frog Egg Growth	NASA Ames Research Center, Moffett Field, California	NASA/OSSA	Ten frog eggs, five tadpoles	Completed
S005	Synoptic Terrain Photography	NASA Goddard Space Flight Center	NASA/OSSA	160 frames of photographic data	Completed
S006	Synoptic Weather Photography	U.S. Weather Bureau, National Weather Satellite Center	NASA/OSSA	200 frames of photographic data	Completed
S010	Agena Micrometeorite Collection	Dudley Observatory Albany, New York	NASA/OSSA	None	Substantial[a]

[a]Package deployed but probably will not be recovered.

TABLE 8.0-I.- EXPERIMENTS ON GEMINI XII - Concluded

Experiment number	Experiment title	Principal investigator	Sponsor	Data obtained	Completion of planned objectives
S011	Airglow Horizon Photography	U.S. Naval Research Laboratory, Washington, D.C.	NASA/OSSA	23 good photographs	Substantial
S012	Micrometeorite Collection	Dudley Observatory, Albany, New York	NASA/OSSA	Data package recovered	Completed
S013	Ultraviolet Astronomical Camera	Dearborn Observatory, Northwestern University	NASA/OSSA	25 frames of film	Substantial
S029	Libration Regions Photography	U.S. Geological Center, Flagstaff, Arizona	NASA/OSSA	Two photographs only (camera malfunction)	Flight operations completed
S051	Daytime Sodium Clouds	CNES-France	NASA/OSSA	None (camera malfunction)	Flight operations completed
T002	Manual Navigation Sightings	NASA Ames Research Center, Moffett Field, California	NASA/OSSA	5 sightings, 13 measurements	Completed
Objects of opportunity (formerly S064)	Ultraviolet Atmospheric Dust Photography	Dudley Observatory, Albany, New York	NASA/OSSA	None (camera malfunction)	Flight operations completed

TABLE 8.0-II.- FINAL EXPERIMENT FLIGHT PLAN FOR GEMINI XII

Experiment title	Priority	Activation time, g.e.t., hr:min	Revolution	Condition	Remarks
D010 Ion-Sensing Attitude Control	1	67:30	42	Mode A	Mode A is equipment extension and activation
		67:48	42	Mode E, photo emission effects	All four temperature parameters stabilized during the mission
		69:30	43	Mode B, roll attitude study	Sensors were used for platform alignment; good results were obtained
		69:55	43	Mode B	No degrading effects caused by thruster firings
		71:20	44	Mode C, pitch attitude study	
		72:20	45	Mode D, yaw attitude study	
		73:20	46	Mode E	
		76:30	47	Mode F, random data accumulation	
		89:40	55	Mode G, translation thruster effects	
		92:49	57/58	Equipment off	
M405 Tri-Axis Magnetometer	12	4:57	3/4	Boom extended; equipment turned on	Orbits of particular interest were those passing through the South Atlantic Anomaly region
		14:44	9	Equipment off	Sensors performed satisfactorily throughout the mission
		27:05 to 30:15	17 to 19	Equipment on	
		51:07 to 60:30	32 to 37	Equipment on	
		75:00 to 84:45	47 to 52	Equipment on	

TABLE 8.0-II.- FINAL EXPERIMENT FLIGHT PLAN FOR GEMINI XII - Continued

Experiment title	Priority	Activation time, g.e.t., hr:min	Revolution	Condition	Remarks
M408 Beta Spectrometer	13	Same as M405			Attitude control was provided; data for revolution 4 showed good correlation with M405 data
		5:00 to 5:20	4	South Atlantic anomaly passes	Approximately five minutes of useful data were obtained during revolution 4. Equipment failed during remainder of mission.
		28:50 to 29:10	18		
		54:00 to 54:20	33		
M409 Bremsstrahlung Spectrometer	14	Same as M405		Sensors on during same periods as M405	Data from revolutions 4, 17, 18, 19, 32, and 33 showed sensors functioned normally Quick-look data showed useful results throughout mission
S003 Frog Egg Growth	2	41:44	24	Unit no. 1 fixed	Frog egg compartment no. 1 was actuated (fixed) by flight crew
		85:10	53	Unit no. 2 fixed	Temperature stabilized at 72° F during mission
		96:00	End	Recovery	Five live tadpoles retrieved
S005 Synoptic Terrain photography	9	Varied		Targets of opportunity	Approximately 160 useful 70-mm pictures taken; most are of Texas and Mexico
S006 Synoptic Weather Photography	10	Varied		Targets of opportunity	The crew confirmed that weather photographs were taken during stateside passes Jet stream clouds photographed over Red Sea on successive passes Approximately 200 pictures were taken

TABLE 8.0-II.- FINAL EXPERIMENT FLIGHT PLAN FOR GEMINI XII - Continued

Experiment title	Priority	Activation time, g.e.t., hr:min	Revolution	Condition	Remarks
S010 Agena Micrometeorite Collection	4	43:11	27	Pilot opened S010 hardware during umbilical EVA	S010 package not recoverable due to limited Agena lifetime
S011 Airglow Horizon Photography	6	24:13	15	Red filter used for sunset exposures; red, green and no filter used during night pass	23 of 36 planned photographs were obtained
		25:13	16	Yellow filter used	
		70:45	44	Yellow filter used	
S012 Micrometeorite Collection	3	8:05	5	Opened during crew sleep period by ground commands	Total exposure time was 6 hours 20 minutes. Retrieval was accomplished during the first standup EVA
		14:29	9	Closed from MCC-H	
		20:26	13	First standup EVA	
S013 Ultraviolet Astronomical Camera	5	19:38 to 20:10	12/13	Standup EVA	25 exposures were taken with useful information
		21:13 to 22:00	13/14		Gamma Velorum star field sequence not accomplished
					Sirius ultraviolet spectrum excellent
S029 Libration Regions Photography	7	27:13	18	Anticipated L_5 regions photographed	Six exposures taken during each sequence
		73:45	47	L_5 region photographed	10 of 12 photographs unusable (overexposed)
					Two photographs obtained both of poor quality

TABLE 8.0-II.- FINAL EXPERIMENT FLIGHT PLAN FOR GEMINI XII - Concluded

Experiment title	Priority	Activation time, g.e.t., hr:min	Revolution	Condition	Remarks
S051 Daytime Sodium Cloud	8	62:41:48	40	Controlled attitude over Hammaguir, Algeria	A total of 20 frames exposed
		64:16:49	41		All photographs overexposed due to camera malfunction
T002 Manual Navigation Sightings	11	63:06 to 63:45	40	Betelgeuse/Rigel	Approximately 14 measurements obtained on each sequence
		75:15 to 75:43	48	Betelgeuse/Rigel	Both deviation and bias errors below 10 arc seconds
		85:45	54	Betelgeuse/Bellatrix	
		87:15 to 87:40	55	Betelgeuse/Bellatrix	
		88:45 to 89:10	56	Betelgeuse/Rigel	
Objects of Opportunity	None	67:00	42	Photographs of atmospheric dust cloud taken prior to sunrise	42 frames exposed All photographs fogged by static electricity

8.1 EXPERIMENT D010, ION-SENSING ATTITUDE CONTROL

8.1.1 Objectives

The principal objective of the D010 Ion-Sensing Attitude Control experiment was to investigate the feasibility of an attitude control system using environmental positive ions and an electrostatic detection system to measure spacecraft pitch and yaw. A secondary objective was to measure the spatial and temporal variations of ambient positively charged particles along the orbital path of the Gemini spacecraft.

8.1.2 Equipment

The onboard spacecraft equipment consisted of two independent systems for the measurement of pitch and yaw attitudes. Dimensionally and electrically, each system was identical, except for placement of the sensor about the pitch and yaw axis. Each sensor was mounted on a boom approximately three feet in length. The boom was extended by crew command after spacecraft orbital insertion. The locations of the booms and sensors are shown in figure 8.1-1. The sensor locations and boom lengths used were selected to minimize vehicle shadowing and space charge effects.

To illustrate the principle of operation of the sensor systems, the measurement of pitch is analyzed. Except for the alignment change, the analysis of the yaw measurement is identical. By aligning two sensors along the pitch axis as shown in figure 8.1.2, the current to the collector of each sensor is given by

$$i_1 = N\ e\ v\ a\ A \cos\ (45 - \theta) \qquad (1)$$

where i_1 is the current to sensor 1, and by

$$i_2 = N\ e\ v\ a\ A \cos\ (45 - \theta) \qquad (2)$$

where i_2 is the current to sensor 2, and when

N = ambient positive ion density

e = electron charge

v = spacecraft velocity

a = experimentally determined grid transmission factor

A = aperture area of sensors 1 and 2 (identical)

θ = pitch-angle deviation from zero

Solving equations 1 and 2 for θ,

$$\tan \theta = \frac{i_1 - i_2}{i_1 + i_2}$$

For θ less than or equal to 20 degrees, tan θ is approximately equal to θ, in radians. The output of the sensors may, therefore, be displayed on a meter calibrated in degrees.

The output of each sensor is amplified by two electrometer amplifiers. To obtain desired accuracy over the current range of 10^{-6} to 10^{-10} amperes, linear amplifiers with range switching are employed. The outputs of the electrometers, designated V_1 and V_2, are then electronically added, subtracted, integrated, and compared. The final output, tan θ, referred to as the compared output, is indicated on a meter in the crew station and transmitted by telemetry to the ground stations. To fully evaluate the experiment, the direct outputs of the electrometers, the range analog indication, and the calibrate monitor signal are transmitted by the spacecraft telemetry. These outputs would not be required in an operational attitude control system. The experiment was designed for precise pitch and yaw angular measurements over the range of ± 20 degrees; however, there is no basic limitation to the magnitude of the angle which can be measured.

Sensor system characteristics are as follows for each of the two systems:

Weight (including electronics and sensors), lb	7
Power (at 28 V), W	3.5
Electronics response time, msec	<1

Dimensions, in.	11 by 6.5 by 6
Angular measurement range, deg	±20

8.1.3 Procedures

Six principal modes of operation excluding Mode A (Equipment Activation) were requested for the mission and four were accomplished. These were as follows:

(a) Mode B, Roll Attitude Study: This procedure consisted of rolling the spacecraft through 720 degrees at a rate of approximately 3 deg/sec while holding the spacecraft pitch and yaw constant at zero.

(b) Mode C, Pitch Attitude Study: This procedure consisted of maintaining a fixed yaw and roll attitude, then varying the pitch angle through a specified angular range at a rate of approximately 0.1 deg/sec. This rate was specified to ensure good comparison of the experiment results with the Inertial Guidance System. The rate of 0.1 deg/sec was determined by the telemetry bandwidth available for the experiment.

(c) Mode D, Yaw Attitude Study: This procedure consisted of maintaining a fixed pitch and roll position, then varying the yaw angle through a specified angular range of zero to 360 degrees at a rate of approximately 0.1 deg/sec.

(d) Mode E, Photo Emission Effects and Ambient Data Accumulation: The equipment measured effects of the sun on the ion environment. The measurements were scheduled before, during, and after the sunrise phase of the vehicle orbit.

(e) Mode F, Random Data Accumulation: The ion-sensor switch was left in this mode while the spacecraft was in drifting flight.

(f) Mode G, Translation Thruster Effects: The spacecraft OAMS thrusters were fired to observe the degrading effects on the sensors accuracy.

8.1.4 Results

A quick look at the experiment signals on real-time telemetry records shortly after power was turned on indicated that all parameters were within the ranges expected. Because of the volume of data required from postflight reduction, most data were not scheduled for delivery prior to the publication of this report.

Discussions with the flight crew at the experiment debriefing provided information on the flight operation of the experiment. These discussions resulted in the following conclusions:

(a) In both Mode D and Mode E, the crew were able to compare the two flight-direction meters. One meter showed the output of the Inertial Guidance System and one showed the experiment sensor output of pitch plus yaw. The results showed that the experiment sensors agreed very well with the spacecraft Flight Director Indicator (FDI).

(b) The response of the experiment sensors to variations in pitch and yaw was extremely rapid.

(c) When the spacecraft thrusters were firing, the experiment sensor indications went off-scale due to the varying charge on the vehicle and/or the contamination in the immediate vicinity of the spacecraft. Readings returned to normal rapidly after the thrusters ceased firing.

(d) No appreciable damage to the sensors was caused by close thruster activities.

(e) The experiment was initially turned on at 67 hours 20 minutes g.e.t. and operated for approximately 15 hours—2 hours in Mode A, 1 hour in Mode B, approximately 30 minutes in Mode C, 30 minutes in Mode D, 2 hours in Mode E, 8 hours in Mode F, and 1 hour in Mode G.

An example of the simultaneous measurements of the ion yaw sensor output and the inertial yaw data during a controlled maneuver is shown in figure 8.1-3. The magnitudes of the angles at given times agree within the errors of the systems. The inertial yaw measurement accuracy for the Gemini spacecraft is approximately ± 2 degrees and the ion yaw measurement accuracy is $\pm 1/4$ of a degree. The inertial data shown illustrate characteristics which introduce difficulties in the manual control of the spacecraft; when the yaw angle is varied, a lag in the response time of approximately eight seconds occurs and the step-like variations cause jumps of 1-1/2 degrees. The addition of an ion yaw sensor would then be a significant improvement over existing attitude measurement systems. Also, the fast response of the ion attitude sensing system will be of importance in conserving thruster fuel when precise angular positioning is desired.

During controlled maneuvers on this mission, the ion attitude sensors operated both in the forward and in the reverse direction. Ion attitude measurements could therefore be obtained from 0 to 360 degrees.

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It was also demonstrated during the flight that firing the spacecraft thrusters did not affect operation of the ion sensor.

Accurate measurements of the charged particle distribution in the shadow or wake of the vehicle were obtained. This is an important measurement for the utilization of charged particle systems for spacecraft docking and maneuvering. The results of the ambient data under controlled spacecraft conditions have also provided much information on the variation in the E and F regions of the earth's ionosphere and of the charged particle density distribution around the earth during a period of rising solar activity.

Transients in the ion sensor outputs were seen to occur for a fraction of a second when spacecraft thrusters were turned on. Ground tests will be conducted to determine the exact source of this transient and to properly filter it out in future ion attitude systems.

Because only preliminary data were available for analysis prior to submission of this report, complete detailed results will be published in subsequent documents.

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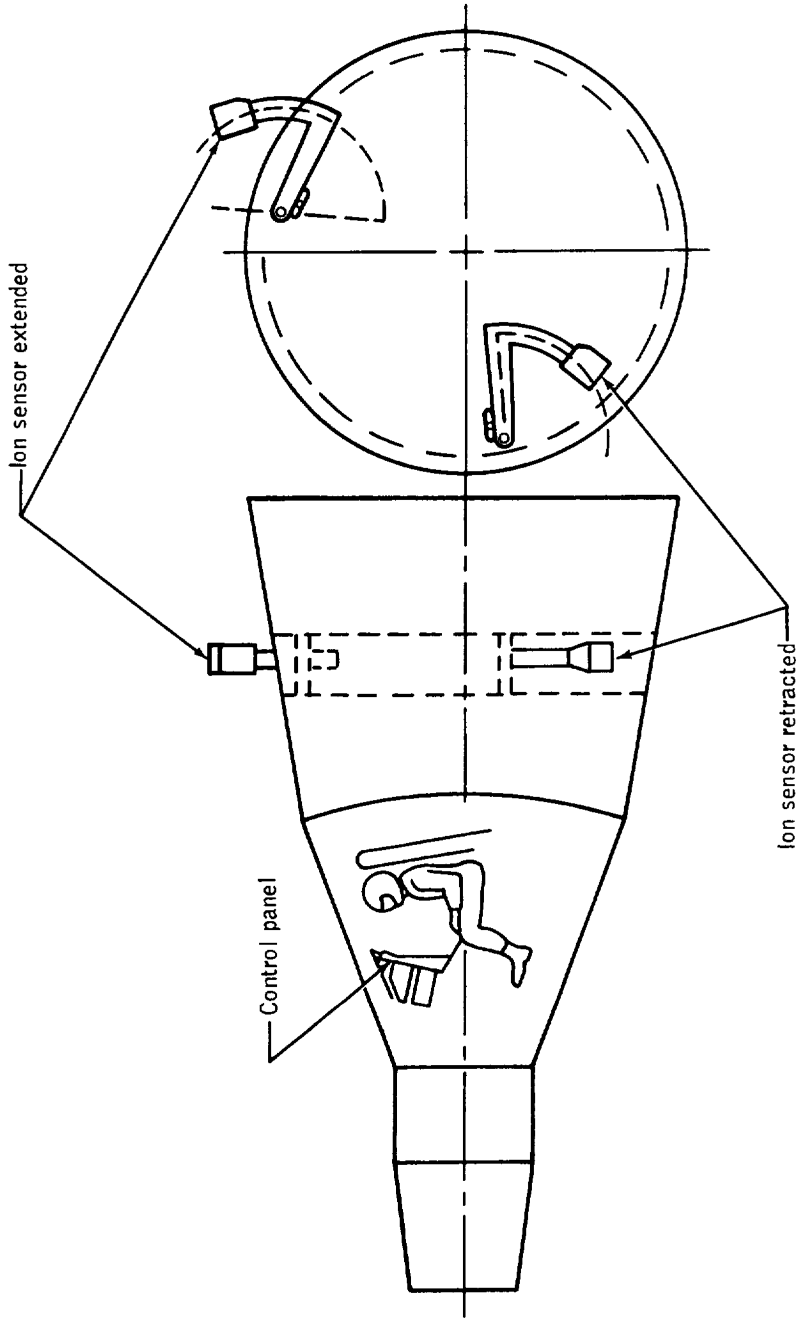

Figure 8.1-1. - Experiment D010, location of equipment.

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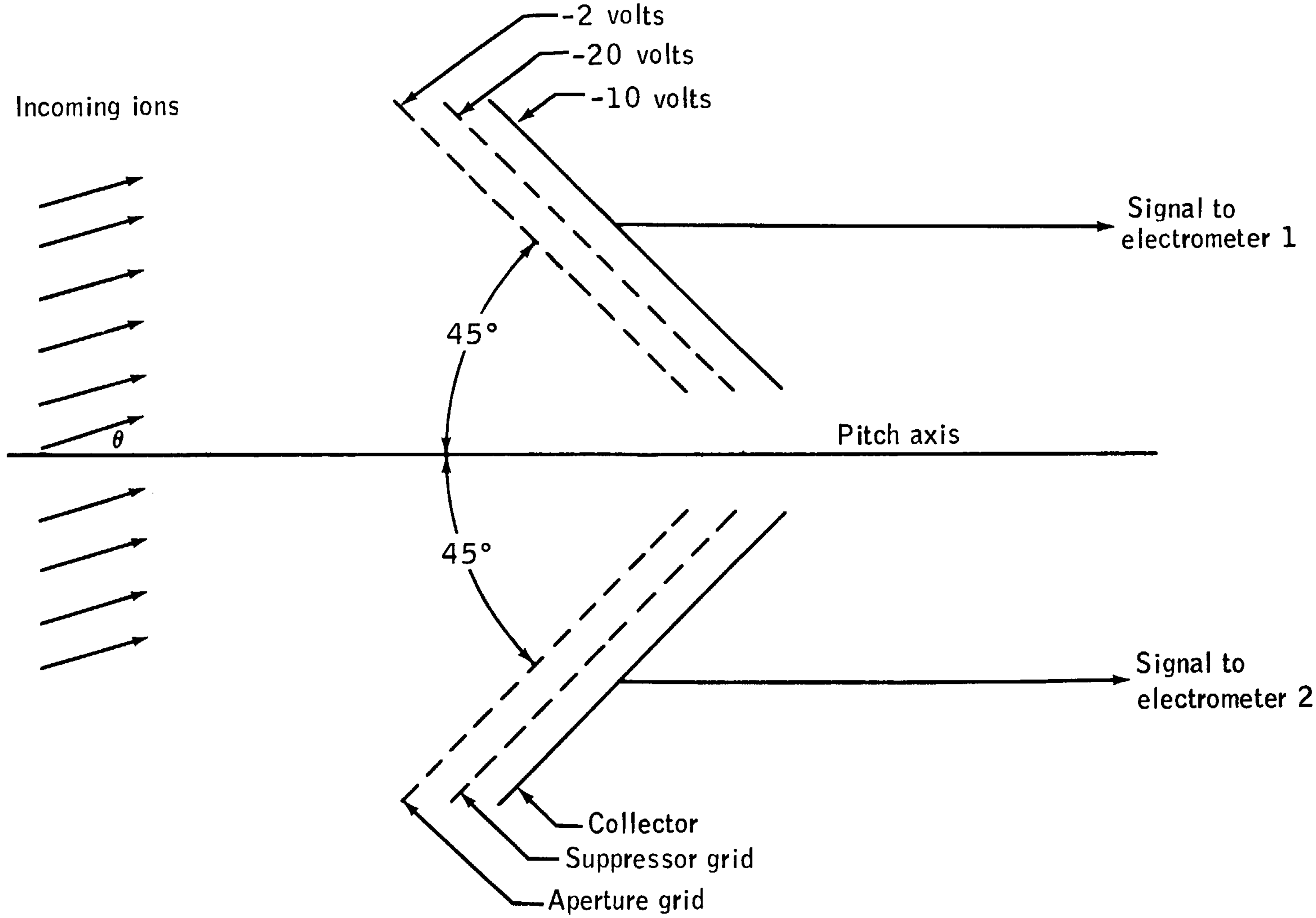

Figure 8.1-2. - Experiment D010, ion-sensing configuration.

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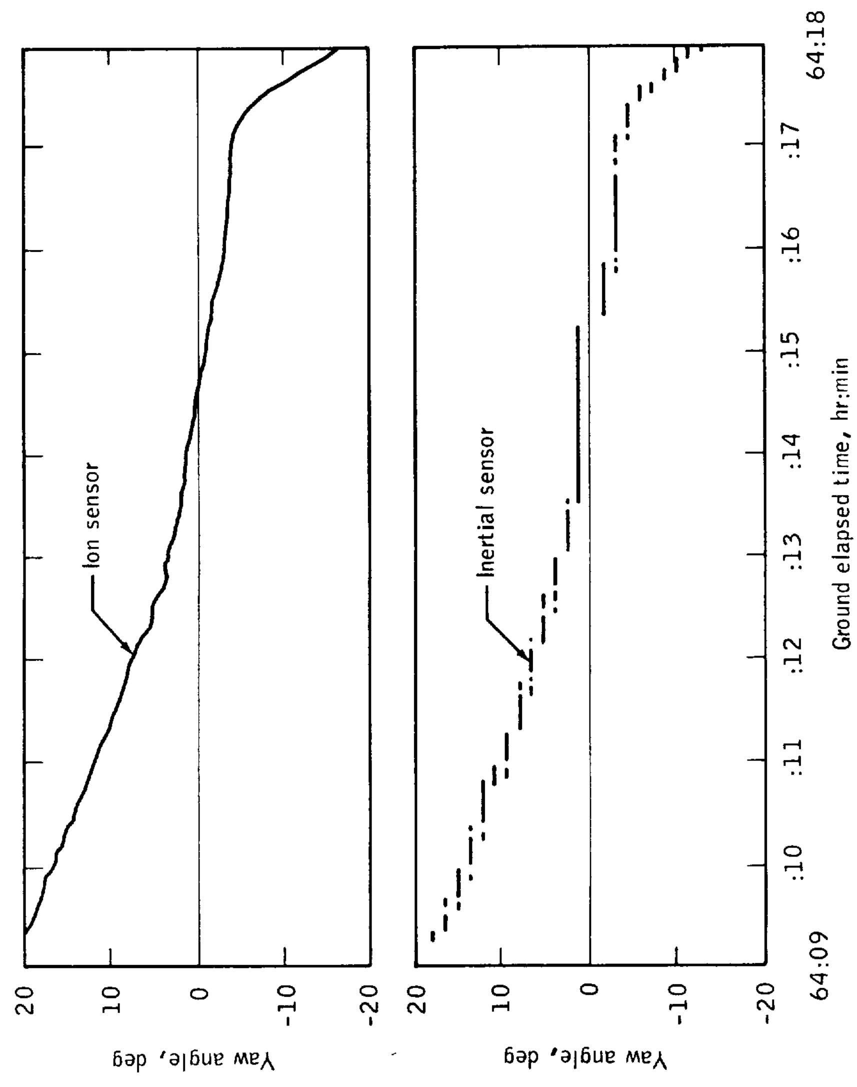

Figure 8.1-3. - Comparison of spacecraft inertial sensor and D010 ion sensor measurements.

8.2 EXPERIMENT M405, TRI-AXIS MAGNETOMETER

8.2.1 Objective

The objective of the M405 Tri-Axis Magnetometer experiment was to determine the magnitude and direction of the earth's geomagnetic field in the South Atlantic Anomaly regions to support the M408 Beta Spectrometer experiment.

8.2.2 Equipment

The tri-axis magnetometer equipment is described in reference 11.

8.2.3 Procedures

The equipment was turned on by the flight crew at 4:57:33 g.e.t. and was turned off at 14:44:20 g.e.t. It was turned on again at 27 hours 15 minutes, 51 hours 6 minutes, and 75 hours 5 minutes g.e.t. The total ON time was approximately 32 hours.

The magnetometer and the beta spectrometer were scheduled to operate for at least ten revolutions while the spacecraft passed over the region bounded approximately by 30 degrees east longitude and 60 degrees west longitude, and by 15 degrees and 55 degrees south latitude. In addition, the equipment was to be operated for a period of at least 15 minutes while the spacecraft was not within this region.

8.2.4 Results

Data obtained from the experiment hardware while passing through the South Atlantic Anomaly were played back by telemetry at the Hawaii tracking station for on-site evaluation in support of the M408 Beta Spectrometer experiment requirements. The data indicated that the equipment functioned as designed and provided information throughout the mission.

An example of data obtained through an anomaly pass is presented in figure 8.2-1. For a typical pass, these data illustrate relative values of the total beta spectrometer count rate and the angle made with respect to the center line of the spectrometer detector. The figure shows the earth's magnetic field angle alpha in degrees as a function of ground elapsed time. The data shown in the figure were measured during a crew sleep period. Analysis is continuing as computer-determined computations become available.

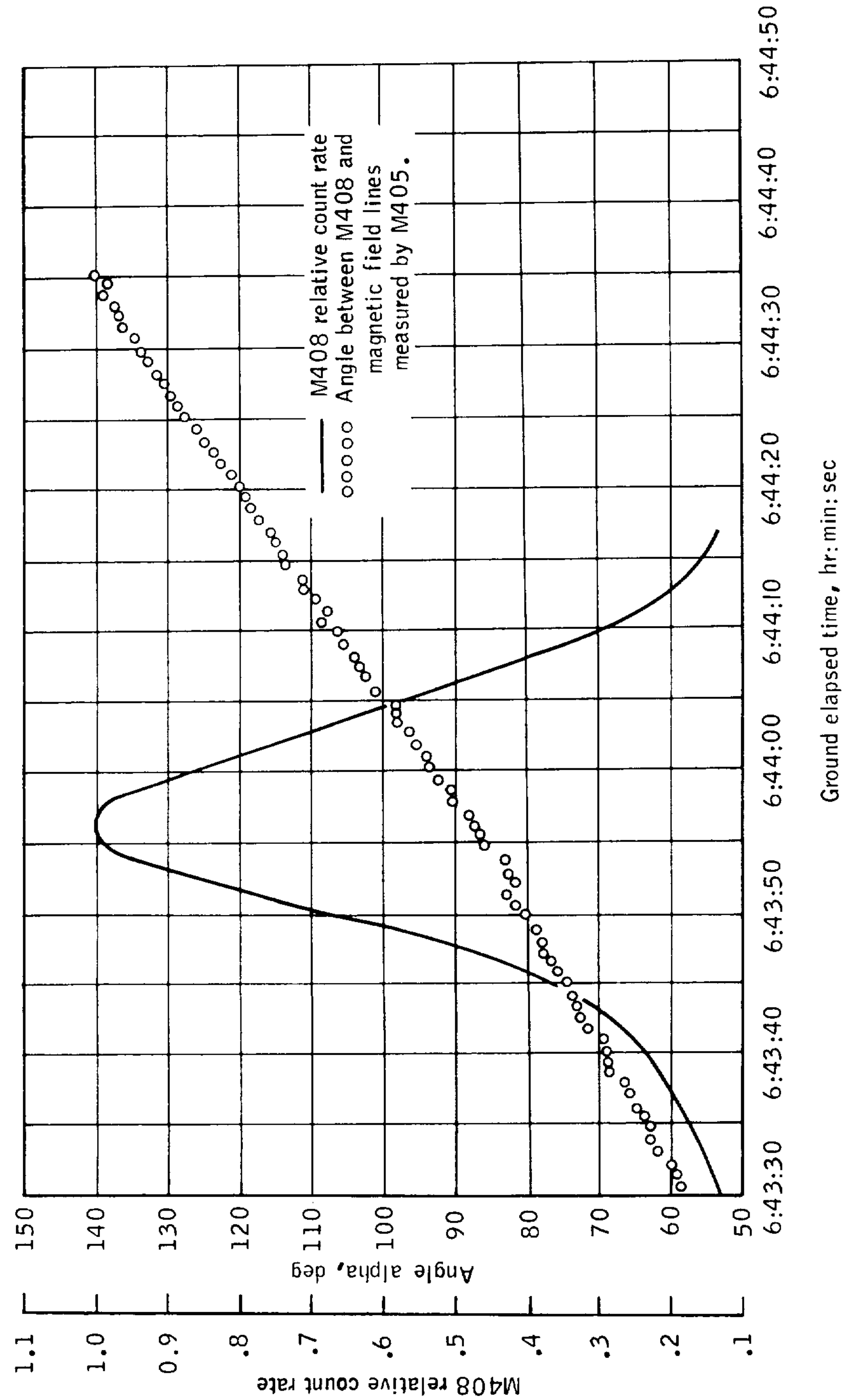

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Figure 8.2-1. - Experiment M408, tri-axis magnetometer response.

8.3 EXPERIMENT M408, BETA SPECTROMETER

8.3.1 Objective

The objective of the M408 Beta Spectrometer experiment was to determine the radiation environment external to the spacecraft. The data would provide information for calculational techniques being developed whereby radiation hazards to flight crews can be estimated prior to a mission.

The radiation dose was estimated for the particular mission and the estimate was compared with measured values obtained on all manned space flights. A check on the mathematical approach is thereby realized. The data obtained were also used to update and complete voids in knowledge of the radiation environment of manned earth orbital missions.

8.3.2 Equipment

The Beta Spectrometer experiment utilizes a stack of four lithium-drifted silicon semiconductors as the detector and provides seven channels of electron-energy density information in a digital format. The spectrometer is constructed to be highly directional having the advantage that the sensors can provide information on the highly directional nature of trapped beta radiation encountered in the Van Allen radiation belts. The equipment was located in the adapter retrograde section of the Gemini spacecraft and used the spacecraft PCM telemetry system for data recording.

8.3.3 Procedures

The only operation required by the flight crew was to turn on a toggle switch early in the mission, then turn it off prior to retrofire. Because of other electrical power requirements, it was decided that the equipment would be turned on and off four times during the mission. A total ON time of approximately 32 hours was obtained, five hours of which were within the desired earth magnetic anomaly regions. Because the spectrometer is directional, as is the measured radiation, the success of the experiment depends on the attitude of the spacecraft when passing through the radiation belt. The trapped radiation lies very nearly in a plane normal to the direction of the earth's magnetic field. Ideally, the instrument should detect radiation normal to this field whenever data are desired in the radiation belt regions. A slow traversal of the instrument through the normal is desirable to obtain a map of the directional distribution of the radiation and useful data statistics.

When operations permitted during the mission, the flight crew was to conduct a controlled roll maneuver through the South Atlantic Anomaly, where radiation is prevalent. This maneuver sweeps the experiment sensors through the normal to the field twice for every 360-degree roll of the spacecraft.

The principal investigator was stationed at the Kokee tracking site in Hawaii, where dumped telemetry data were evaluated following each of the anomaly passes. The requirements for additional spacecraft attitude control could then be augmented or reduced during the mission. The M405 Tri-Axis Magnetometer experiment provided instantaneous "magnetic attitude" of the beta spectrometer so that the quick-look data received could be continuously related to spacecraft attitude.

8.3.4 Results

Preliminary examination of data received during the flight has indicated that the instrument functioned normally during approximately the first three hours after it was initially turned on. At 7 hours 40 minutes g.e.t., a sudden change in the raw data indicated an apparent instrument failure. This condition persisted throughout the rest of the mission.

The data available for a failure analysis at the time of publication of this report indicate that the instrument suddenly became sensitive to noise on the telemetry line that provides synchronization pulses to the spectrometer equipment. An unusually high noise level and an unusual sensitivity to the noise resulted in the odd data printouts that occurred after 7 hours 40 minutes g.e.t. This and other possible causes of failure are presently being investigated.

The data recorded prior to 7 hours 40 minutes g.e.t. appear at first analysis to be normal. Good spacecraft attitudes during revolution 5 resulted in about four minutes of excellent data. These data will be thoroughly analyzed and reported as soon as the failure analysis reveals what occurred and assures that the data are reliable.

8.3.5 Conclusion

Although the beta spectrometer apparently failed during the first day of the mission, sufficient data were collected prior to the failure to deem the experiment a partial success. A complete failure analysis is being conducted to determine what occurred and what effect it had on the accuracy of the data obtained prior to the failure.

8.4 EXPERIMENT M409, BREMSSTRAHLUNG SPECTROMETER

8.4.1 Objective

The objective of the M409 Bremsstrahlung Spectrometer experiment was to determine the bremsstrahlung flux-energy spectra inside the Gemini spacecraft while passing through the South Atlantic Magnetic Anomaly regions. The spectra will be compared with computer-predicted bremsstrahlung spectra using data from the M408 Beta Spectrometer experiment.

Secondary gamma rays produced in the spacecraft material by externally trapped electrons did not reach biologically significant levels during any Gemini mission. On long-duration missions which may be flown in high trapped-electron flux environments, however, the problem will attain considerably more importance. The calculations of bremsstrahlung radiation involve uncertainties due to the small amount of information available on cross-section interaction and the complex, heterogeneous makeup of the spacecraft. The bremsstrahlung detector was designed to give a time-differentiated measurement of the electron-induced gamma rays over a large section of the vehicle.

8.4.2 Equipment

The equipment (fig. 8.4-1) is described in reference 11.

8.4.3 Procedures

The requirements for the flight crew were to turn the equipment on and off four times during the mission for a total ON time of approximately 32 hours. About five hours of this time was within the geographic anomaly areas of interest.

8.4.4 Results

A few spectra were observed during post-pass telemetry playbacks at the Hawaii ground station. These spectra indicated that the spectrometer functioned as expected. The computer data processing will consist of reconstruction of spectra as a function of spacecraft time and location. The reconstruction will involve decompressing transmitted numbers; adding sensor efficiency, dead time, and calibration factors; and correlating spacecraft attitude and position. The final results of the experiment will be determined after data from several complete revolutions are processed, analyzed, and compared with associated experiment activities.

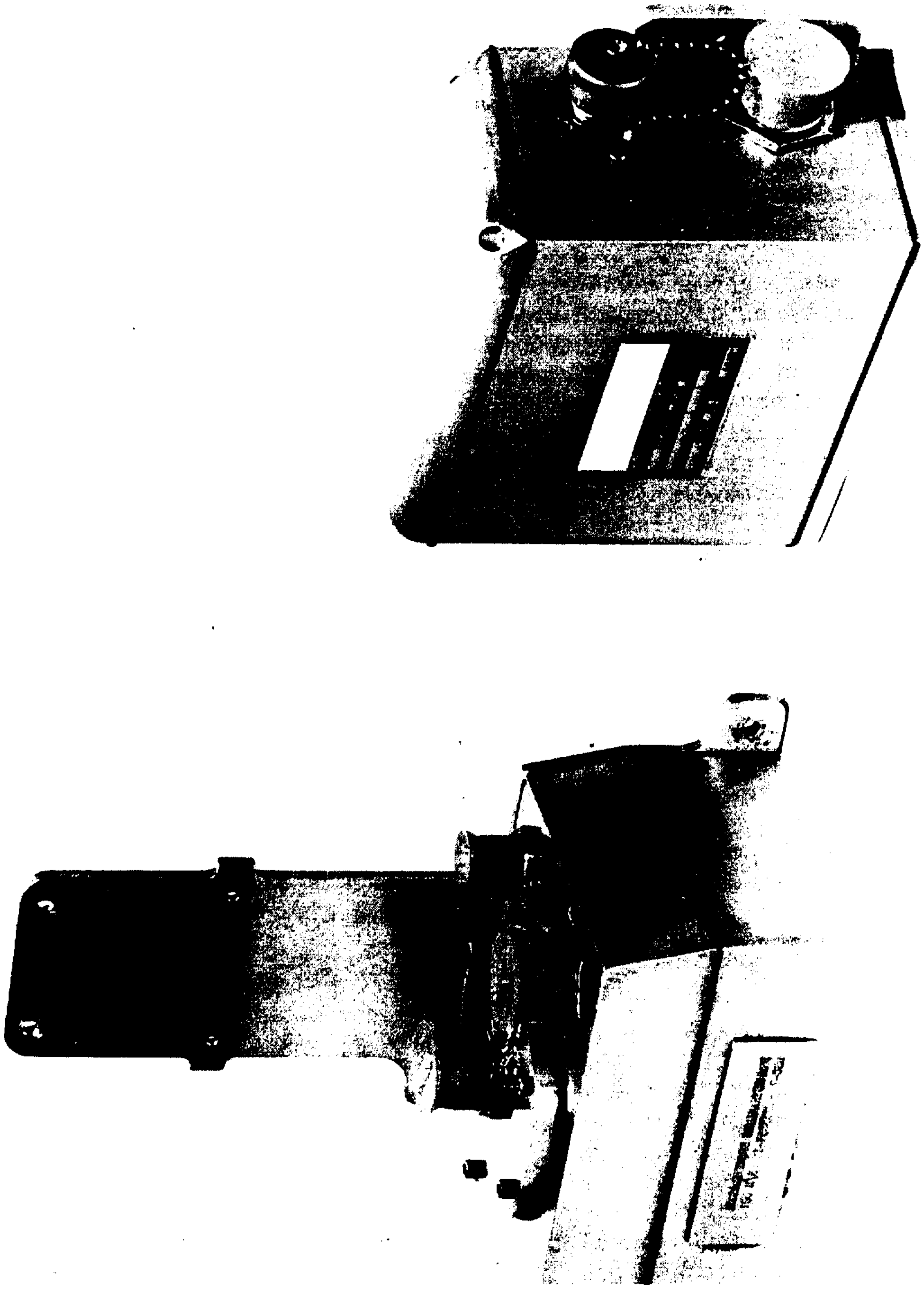

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Figure 8.4-1. - Experiment M409, bremsstrahlung spectrometer.

8.5 EXPERIMENT S003, FROG EGG GROWTH

8.5.1 Objective

The objective of the S003 Frog Egg Growth experiment was to determine the effect of weightlessness on the ability of a fertilized frog egg to divide normally and to differentiate and form a normal embryo. This experiment was attempted on the Gemini VIII mission, but was only partially completed because of the early termination of that mission.

8.5.2 Equipment

The experiment was contained in one package mounted on the right-hand hatch of the spacecraft. The package had four experimental chambers containing frog eggs in water, with a partitioned section containing a fixative (5-percent formalin). The package was insulated and contained a temperature control system for both heating and cooling in order to maintain an experiment temperature of approximately 70° F. Electrical power was obtained from the spacecraft electrical system. The experiment was actuated by two handles on the outside of the package. These two handles and a toggle switch for the heating element were manipulated by the pilot, either on ground command or according to a predetermined schedule. Identical equipment was used as a control on the ground. Figure 8.5-1 shows the experiment equipment installed on the right-hand hatch of the spacecraft.

8.5.3 Flight Procedures

Eggs were obtained from several dozen female frogs (Rana pipiens) by injection of frog pituitary glands approximately 48 hours prior to launch, in order to induce ovulation at the desired time. The best of these eggs (from two females) were selected for flight and fertilized by immersion in a sperm suspension made by macerating frog testes in pond water (fig. 8.5-2). The fertilized eggs were then removed to a cold room (43° F) and placed in approximately 10 cc of pond water in each of the four experimental chambers. The formalin fixative was placed behind leak-proof partitions in three of the four chambers. The fourth chamber had water instead of formalin. Each chamber received 5 eggs, so that a total of 20 eggs was flown. Two sets of control experiments were set up in identical equipment on the ground. The first control was to be conducted simultaneously with the flight experiment. The second control was delayed approximately two hours so that changes in temperature experienced by the flight experiment could be duplicated more precisely than in the simultaneous control. Since telemetered temperatures were not received continuously, such a delayed control was necessary to duplicate the actual flight environment.

The flight experiment was installed in the spacecraft approximately 2-1/2 hours prior to launch. The fertilized eggs were kept at approximately 43° F until spacecraft installation to retard the first cell division of the eggs. This pre-cooling of the eggs was sufficient to retard first cleavage until the zero-g phase of the flight. At 41 hours g.e.t., the pilot was scheduled to turn the first handle on the experimental package to inject the formalin fixative into two of the four egg chambers. This would kill the eggs in these two chambers and preserve them for microscopic study after recovery. A second handle was scheduled to be actuated at 85 hours g.e.t. to fix the eggs in one of the remaining two chambers. The last chamber was to remain unfixed and those embryos recovered alive. All eggs and embryos were to be studied after recovery for gross morphological abnormalities in cleavage planes and differentiation. Histological examination and electron microscopy were also anticipated.

8.5.4 Results

During the Gemini VIII mission early cleavage stages were successfully attained. However, because of the short duration of this flight, the later cleavage and developmental stages were not obtained and were the main reason for conducting this experiment during the Gemini XII mission. Postflight analysis of the results of this mission indicate that all phases of the experiment were performed as scheduled with good results. The desired later cleavage and embryonic stages were obtained to complete the experiment successfully.

The experiment package maintained temperatures between 66° F and 74° F throughout the mission, stabilizing at approximately 72° F. Although this temperature was 4° F above the expected average, it was well below the maximum allowable of 80° F. The temperature history is shown in figure 8.5-3. The experiment package toggle switch to turn on the internal heater was actuated at 17:41:55 g.e.t., to assure proper experiment temperatures during extravehicular activities. The first handle actuation was accomplished at 41:43:40 g.e.t. to release the formalin and fix the eggs in two of the egg chambers. The second handle actuation, performed at 85:10:22 g.e.t. to fix the eggs in the third chamber, completed the flight crew's participation in this experiment.

The ten embryos in the 40-hour fixation chambers appeared to be morphologically normal when compared with the ground-control experiments.

No abnormalities were detected by gross observation in either the flight or ground-control embryos. The five embryos fixed at 85 hours g.e.t. were well developed and morphologically normal tadpoles. The five embryos which were not fixed were found to be well developed, live, swimming tadpoles when the experiment package was opened onboard the prime recovery ship. Three of these embryos were morphologically normal and two were abnormal. However, the abnormalities were not inconsistent with the ground-control embryos and they cannot be ascribed to development under a zero-g environment at this time. The five live tadpoles died several hours after their recovery and were fixed for histological sectioning. The reasons for their deaths have not yet been determined. All embryo specimens are being sectioned for histological study, after which a more conclusive report can be prepared.

8.5.5 Conclusions

In spite of the fact that the frog egg is known to orient itself with respect to gravity during its very early development, a gravitational field apparently is not necessary for the egg to divide normally. This was a preliminary conclusion reached after the Gemini VIII mission. To this can now be added the conclusion that gravity is also not necessary for differentiation and morphological changes in the later stages of embryonic development. Whether the frog egg will divide and develop normally if it is fertilized under zero-g conditions so that it never has a chance to become oriented with respect to gravity is still an unanswered question. It is hoped that this question can be answered on later space flights.

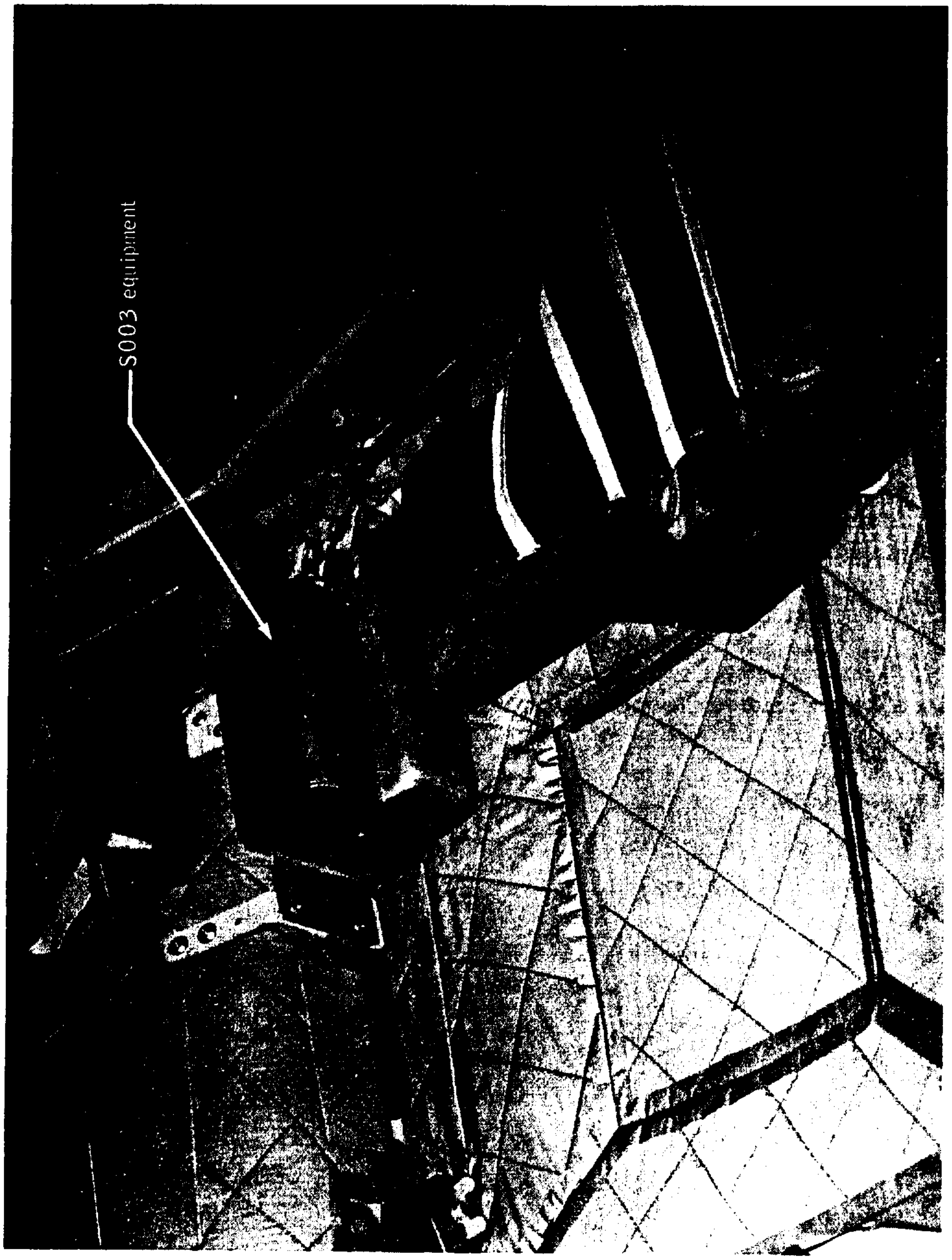

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Figure 8.5-1. - Experiment S003, frog egg package installed on the right-hand hatch.

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Figure 8.5-2. - Experiment S003, frog eggs being fertilized prior to loading in the experiment flight package.

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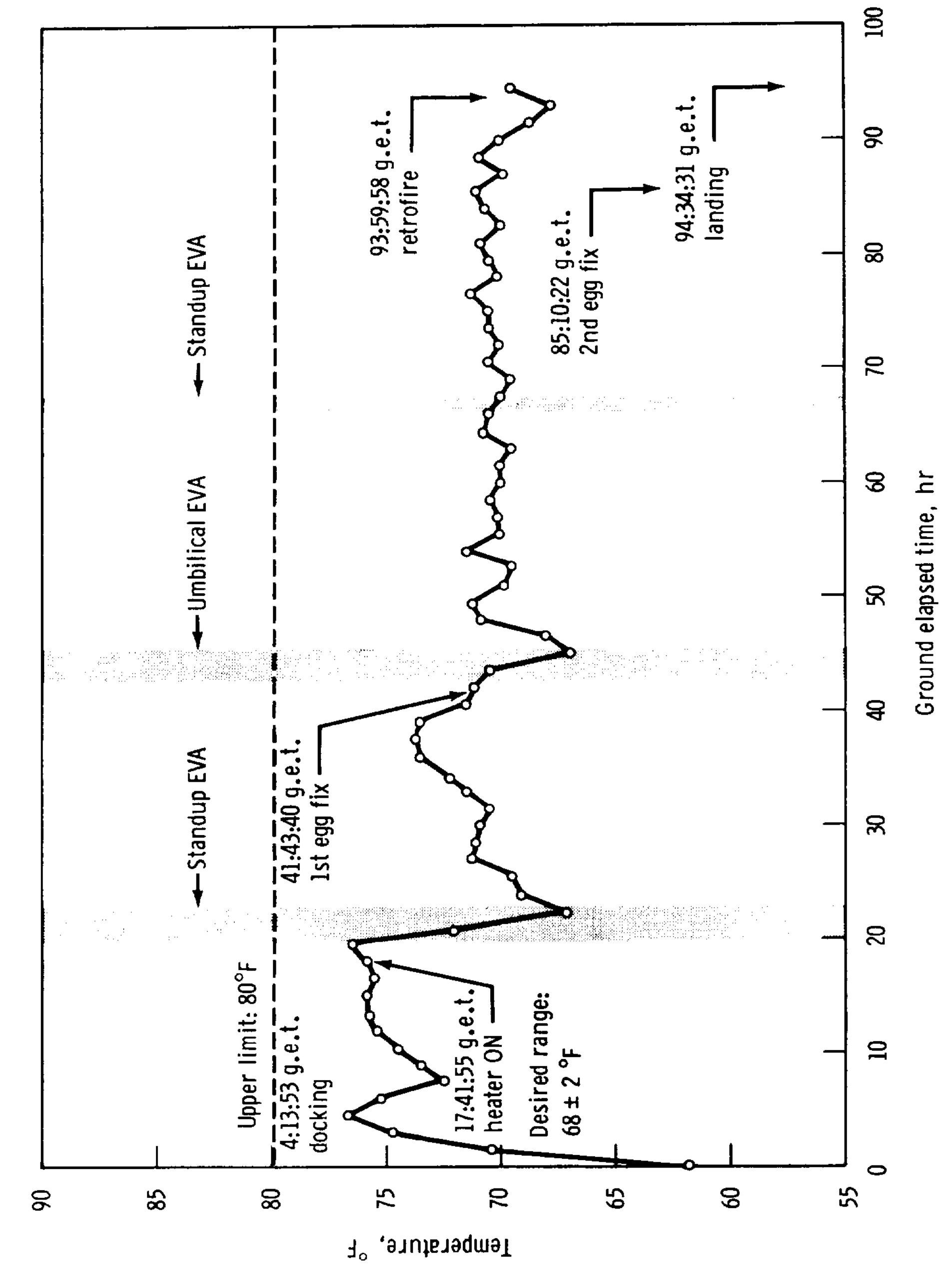

Figure 8.5-3. - Experiment S003, frog egg package internal temperatures.

8.6 EXPERIMENT S005, SYNOPTIC TERRAIN PHOTOGRAPHY

8.6.1 Objectives

The objective of the S005 Synoptic Terrain Photography experiment was to obtain high-quality, small-scale photographs, using color film, of terrain and ocean areas for geological, geographic, and oceanographic research. Pictures were desired of southern Mexico, southern India, West Pakistan, lower Baja California, the Rift Valley (Africa), and northwest South America. In addition, certain ocean areas and river mouths were listed for photography.

8.6.2 Equipment

Two operational cameras were used for terrain photography. Most of the pictures were taken with a modified wide-angle camera, with a 38-mm focal-length lens. One magazine was exposed with the 70-mm general-purpose camera using an 80-mm focal-length lens.

8.6.3 Procedure

The crew was instructed to take vertically oriented, systematic, overlapping photographs of the terrain areas listed above, or of any other cloud-free land areas. Pairs of photographs were desired of the various oceanic sites selected. Procedures were essentially the same as on previous Gemini missions.

8.6.4 Results

About 160 photographs usable for the purpose of the experiment were taken, nearly all between good and excellent quality. The majority of pictures were taken over North America, chiefly Texas and Mexico. In addition, a substantial number of good pictures were obtained of the Rift Valley (northern Red Sea), southwest Asia, and the Irrawaddy River/Andaman Sea area. Several high oblique pictures facing to the north show parts of the United States never before photographed, at least during the Gemini missions.

Most of the pictures appear to be of great potential value for study of regional tectonics. The juncture between different tectonic regions in southern Iran and West Pakistan is clearly shown on photographs such as figure 8.6-1(a). In figure 8.6-1(b), major geologic structures of the Texas coastal plain, such as the Balcones escarpment, are visible. Figure 8.6-1(c) shows much detail of structure near the Rift Valley around the branches occupied by the Gulf of Suez and the Gulf of Aqaba.

Considerable oceanographic information is contained in photographs such as figure 8.6-1(d), which shows the distribution of muddy effluent from the mouth of the Irrawaddy River. Other pictures of potential oceanographic value show details of the Gulf of Mexico.

The success of the S005 experiment was aided by good weather over large areas and because there was little apparent obscuration of the spacecraft windows. These factors have hampered photography on previous missions.

8.6.5 Conclusions

The S004 experiment can be classified as highly successful in terms of quality of the pictures, coverage of desired areas, and number of photographs.

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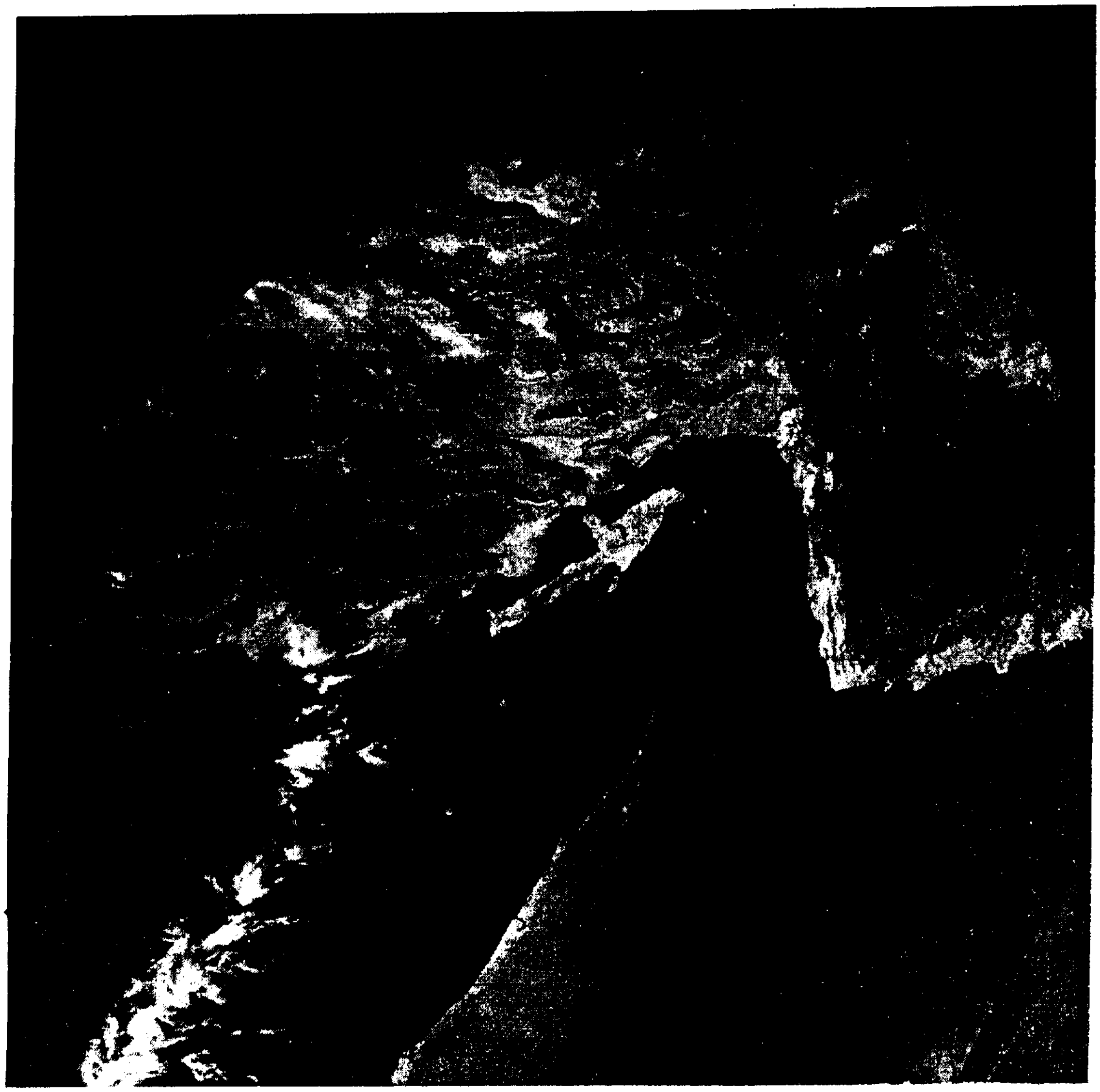

(a) Junction between different tectonic regions in southern Iran and West Pakistan. Spacecraft pointing down toward the Persian Gulf and Gulf of Oman with east at the top. Taken at 85 hours 24 minutes g.e.t. on November 15.

Figure 8.6-1.- Experiment S005, typical synoptic terrain photography.

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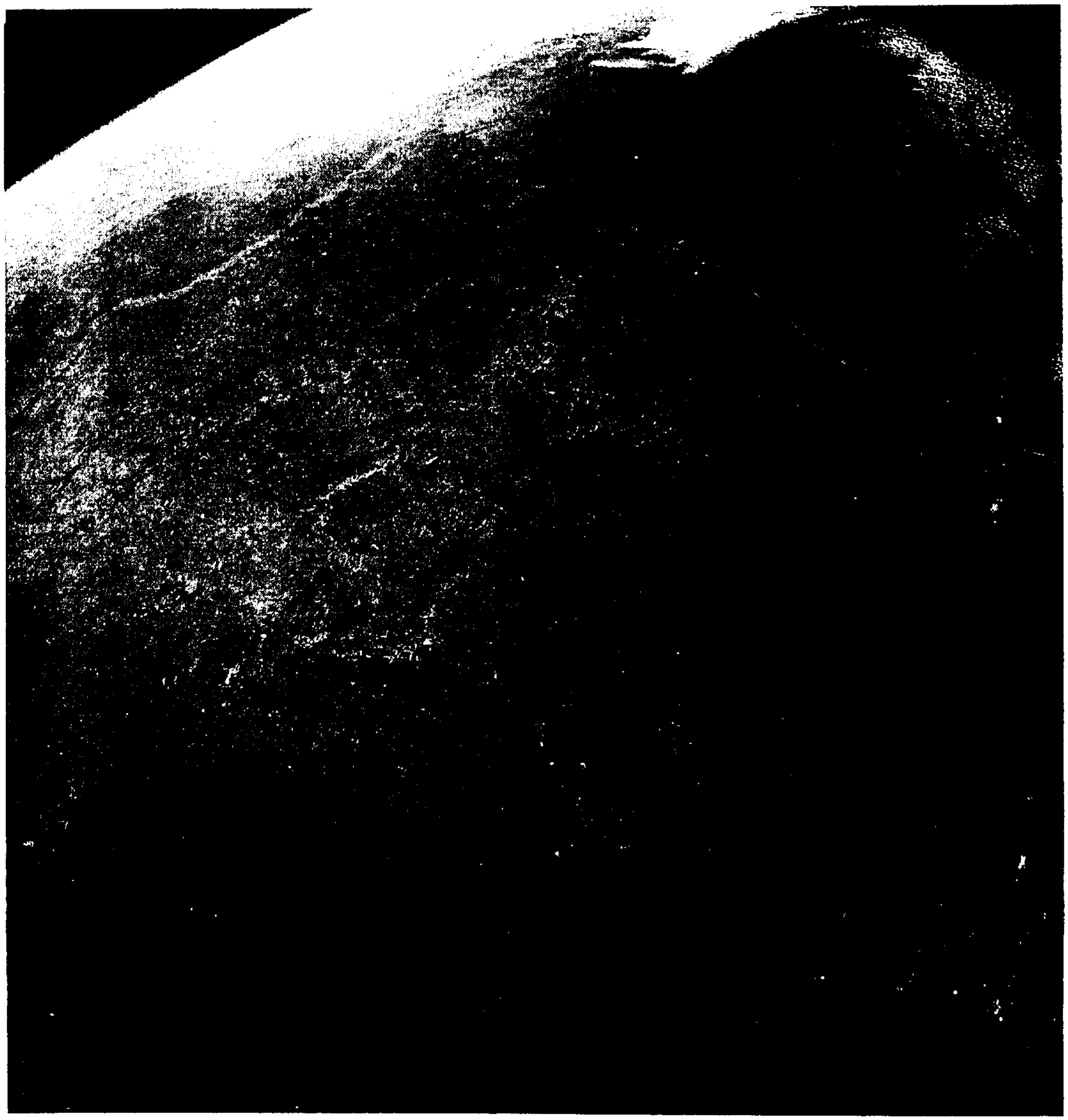

(b) Major geologic structures of the Texas coastal plain. Spacecraft pointing toward San Antonio and Austin showing Houston toward the east at the upper center. Taken at 70 hours 21 minutes g.e.t. on November 13.

Figure 8.6-1. - Continued.

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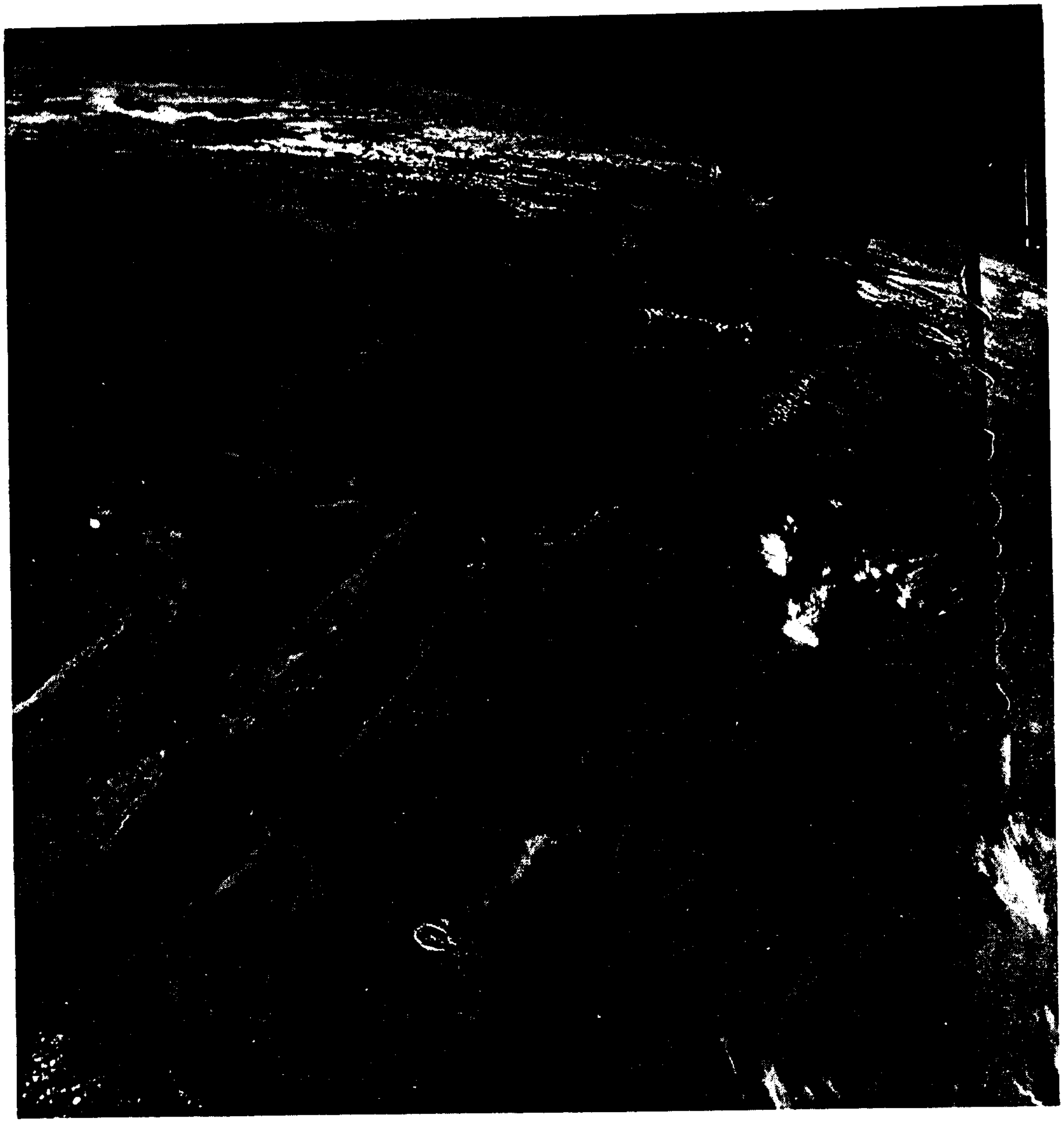

(c) Detailed structure shows near the Rift Valley around the branches occupied by the Gulf of Suez and the Gulf of Aqaba. Looking southeast with the Nile valley in right foreground. Taken at 38 hours 56 minutes g.e.t. on November 13.

Figure 8.6-1. - Continued.

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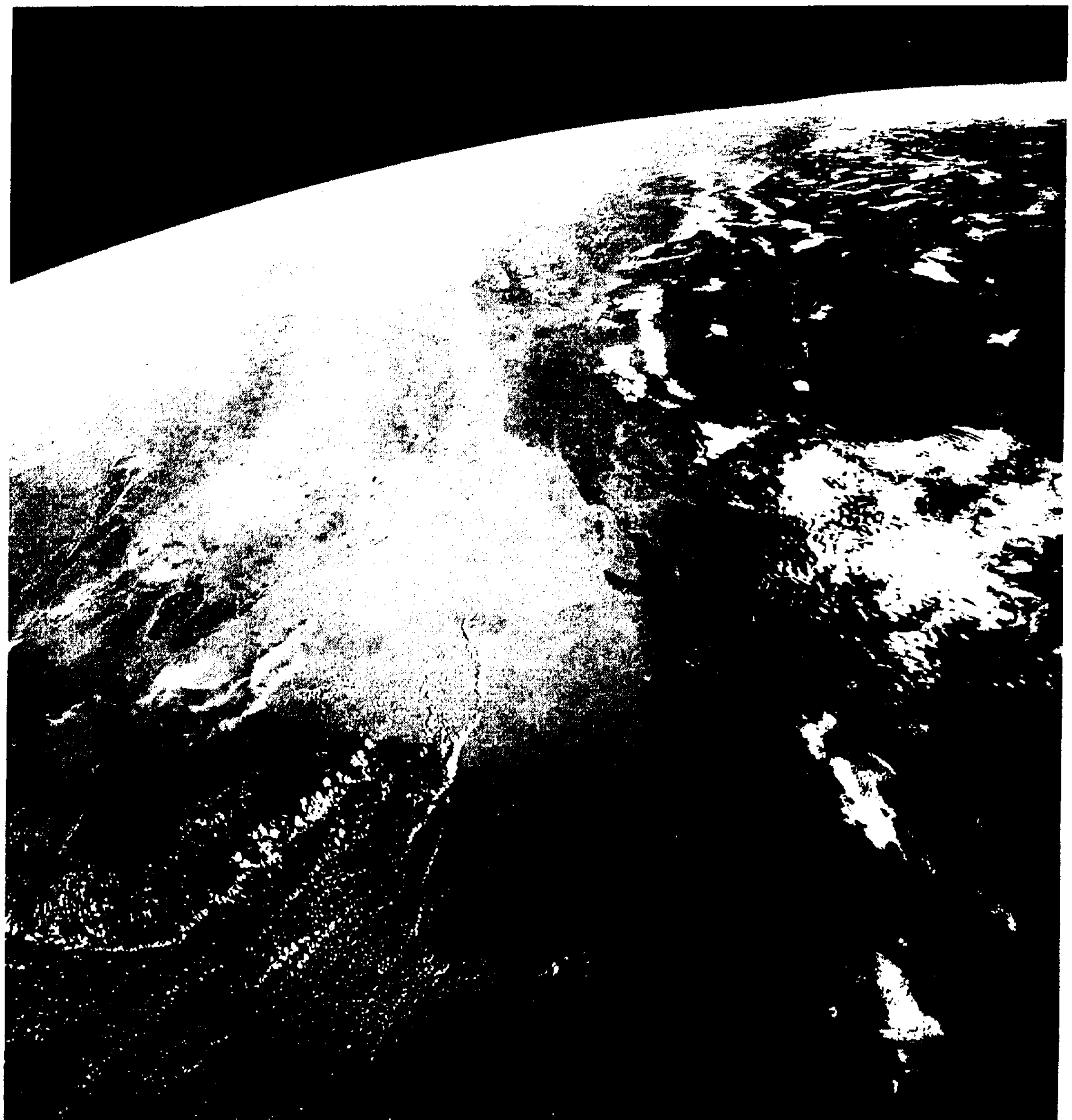

(d) Distribution of muddy effluent from the mouths of the Irrawaddy River looking north with Burma and Rangoon in the right foreground. Taken at 53 hours 36 minutes g.e.t. on November 13.

Figure 8.6-1.- Concluded.

8.7 EXPERIMENT S006, SYNPOTIC WEATHER PHOTOGRAPHY

8.7.1 Objective

The objective of the S006 Synoptic Weather Photography experiment was to obtain photographs of clouds for augmenting those obtained on previous flights, for use in studies of the earth's weather systems, and to aid in interpreting weather satellite photography. An objective stressed in this mission was to obtain views of the same areas on at least two passes during the same day, thus providing data for the study of cloud change and movement.

8.7.2 Equipment

Photographs of meteorological interest were taken with the same camera as used for the S005 Synoptic Terrain Photography experiment and general documentary photography. Most views were obtained with the superwide-angle 70-mm camera, using a 38-mm focal length and f/4.5 lens. Some pictures of interest were taken with the 70-mm general-purpose camera, using an 80-mm focal length and f/2.8 lens. A haze filter was attached to both cameras.

8.7.3 Flight Procedure

The crew was briefed in advance of the mission as to clouds of particular concern. During the mission, weather maps and weather satellite pictures were evaluated as a means of selecting areas of potential photographic interest. Constraints of time and fuel for orienting the spacecraft permitted a limited opportunity during the mission for these areas to be communicated to the crew for observation of specific meteorological activities. Some of the specific cloud formations suggested to the crew were photographed, however.

8.7.4 Results

Approximately 200 photographs obtained during the mission show cloud patterns and are of excellent quality. Several categories of photographed cloud systems are of particular interest for study.

8.7.4.1 Cirrus bands.- Observations made on the ground and in aircraft have indicated that there is often a band of cirrus clouds on the equatorial side of the core of the upper westerly winds. These "jet stream" cirrus clouds also appear frequently on weather satellite pictures and are used to approximate the location and orientation of the upper-wind maximums. The flight crew obtained several excellent views of this phenomenon in response to real-time requests. Figure 8.7-1(a) shows a narrow cirrus band of this type above the Red Sea. A photograph taken on the previous revolution showed the band location to be about the same, but with certain changes in the cirrus elements comprising the band.

Other views of "jet stream" cirrus were obtained over western North Africa, over western Mexico, and above lower frontal cloudiness across the southeastern United States as shown in figure 8.7-1(b). The wind maximum at the cirrus level regions (approximately 30 000 feet) was near 100 knots in velocity.

8.7.4.2 Vortices in the lee of islands.- Vortices occurring in stratocumulus clouds in the lee of islands have been the subject of continuous studies. Figure 8.7-1(c) shows several such vortices in the lee of Guadalupe Island, off the west coast of Mexico.

8.7.4.3 Cellular patterns.- Also shown in figure 8.7-1(c) are cellular patterns which are the result of organized convection in areas of little wind shear. In this photograph, both the "open" and "closed" types occur in proximity, the former with ascending motion around the edges of the cell and descending motion in the center, and the latter with a circulation in the opposite sense.

8.7.4.4 Sunglint.- Sunglint patterns often appearing on weather satellite pictures are related to sea conditions and, hence, to wind speed. The flight crew obtained a number of sunglint photographs. Figure 8.7-1(d) shows a very large area of reflected sunlight. Winds in the area were less than ten knots and in some parts of the area less than five knots. Ship observations in the same area reported sea waves of three feet or less in height.

8.7.4.5 Views of areas on several passes.- Of interest are views of the same cloud systems on successive revolutions, notably over the southern United States. These will be studied relative to change and cloud movement.

8.7.4.6 Other features.- Other features of interest photographed during this mission are the smoke spread from fires, clouds near island weather stations which can be related to concurrent atmospheric sounding, cloud streets, and clouds associated with a typical frontal low pressure system in the North Pacific.

8.7.4.7 Comparison with weather satellite photography.- Daily coverage of most of the world by the ESSA III meteorological satellite provides photographic data for comparison with the photography from this experiment.

8.7.5 Conclusion

The flight crew obtained a variety of interesting and significant views of cloud formations. These will be studied relative to weather satellite pictures and to other meteorological data. The photographs have provided many contributions to the knowledge of the earth's cloud systems.

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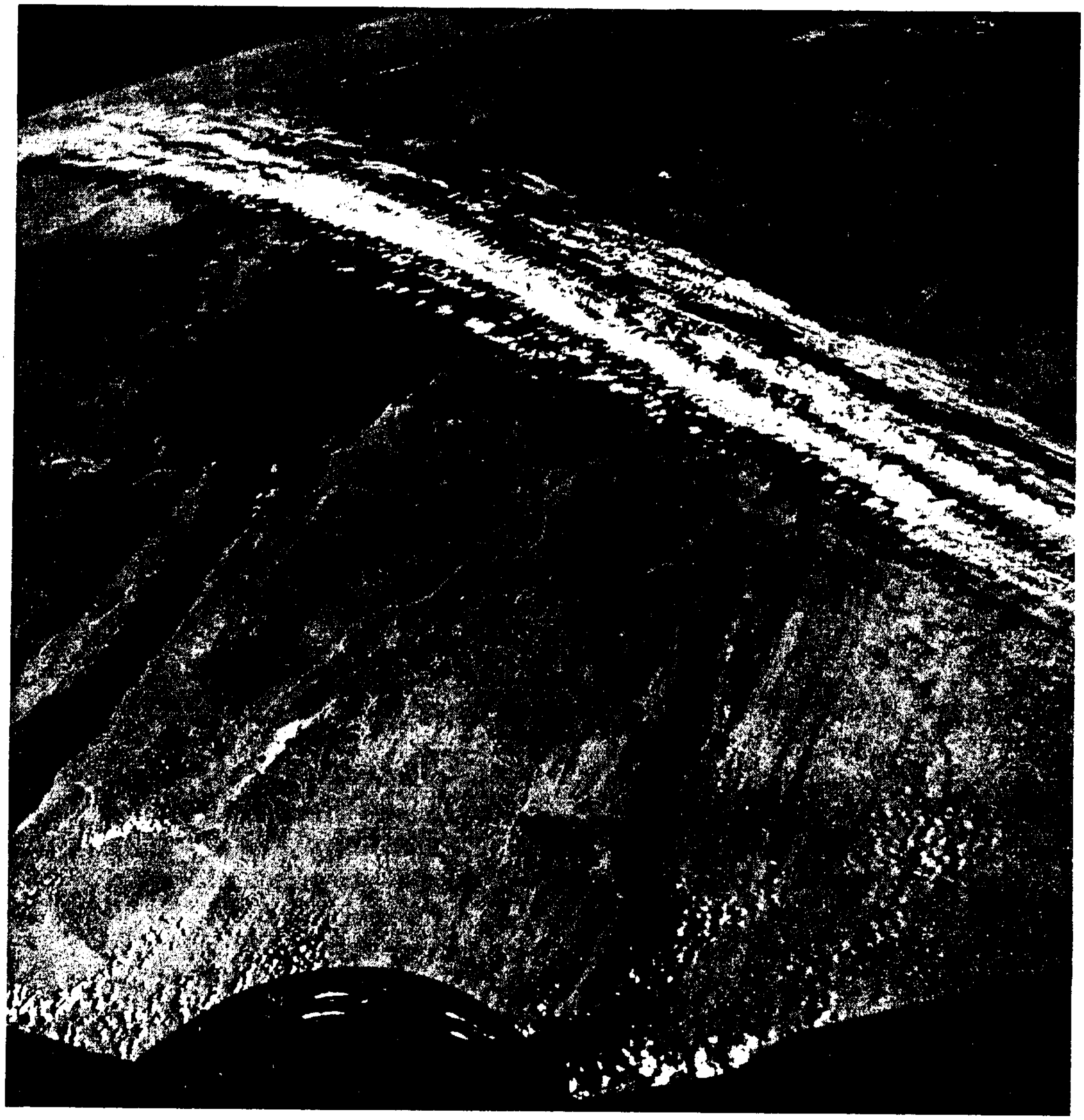

(a) A band of cirrus clouds showing strong upper winds above the Red Sea area. Looking down with southeast at the top of the page. Taken at 61 hours 18 minutes g.e.t. on November 14, 1966.

Figure 8.7-1. - Experiment S006, typical synoptic weather photography.

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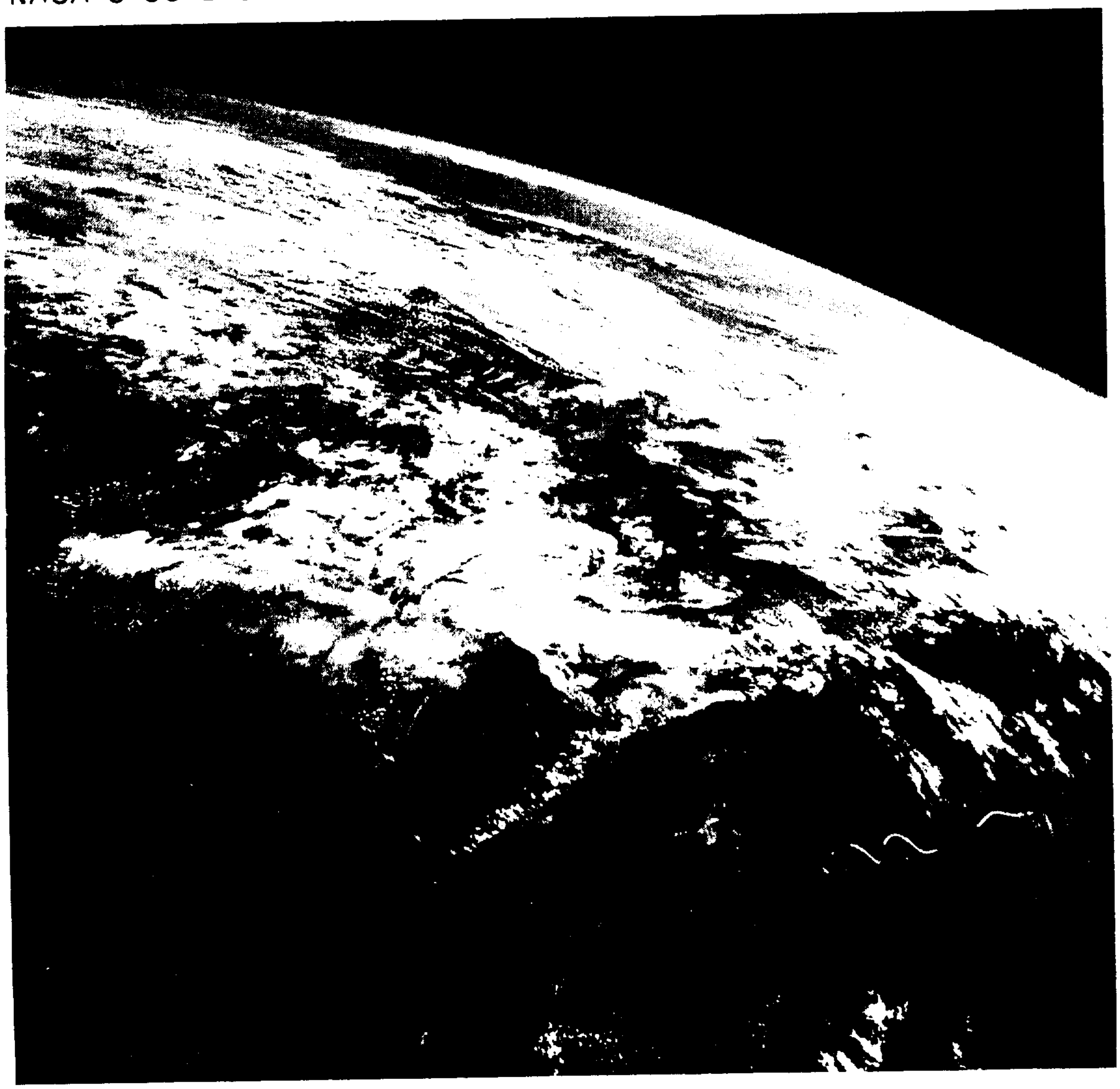

(b) A narrow band of cirrus shown above lower frontal clouds over the southeastern United States and adjacent portion of the Atlantic Ocean. Spacecraft is pointing down looking northward with Florida in the foreground. Taken at 19 hours 55 minutes g.e.t. on November 12, 1966.

Figure 8.7-1. -Continued.

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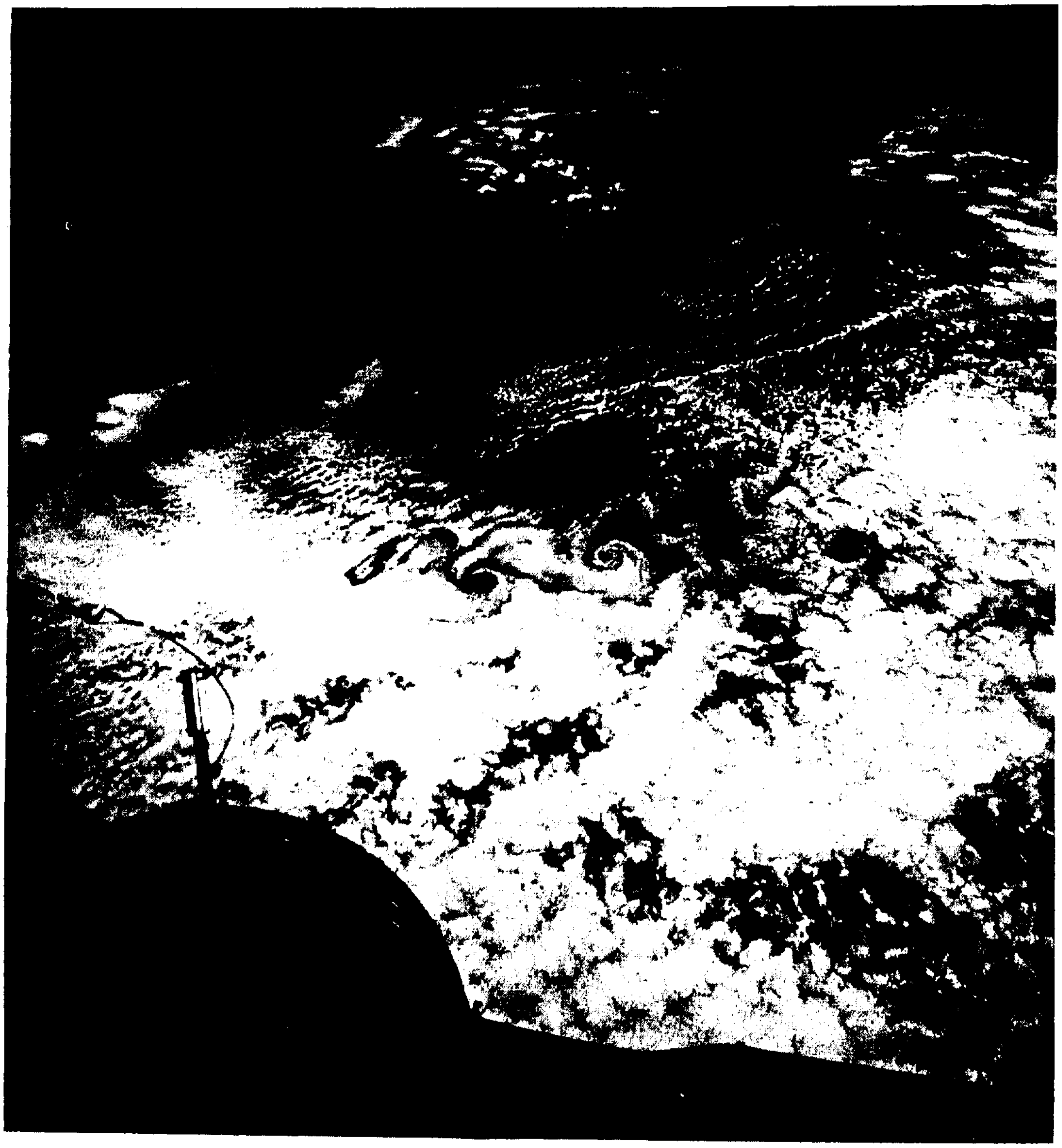

(c) Vortices and cellular cloud patterns in stratocumulus clouds near the Guadalupe Islands. Spacecraft is pointing toward Baja California, looking eastward. Taken at 46 hours 13 minutes g.e.t. on November 13, 1966.

Figure 8.7-1. - Continued.

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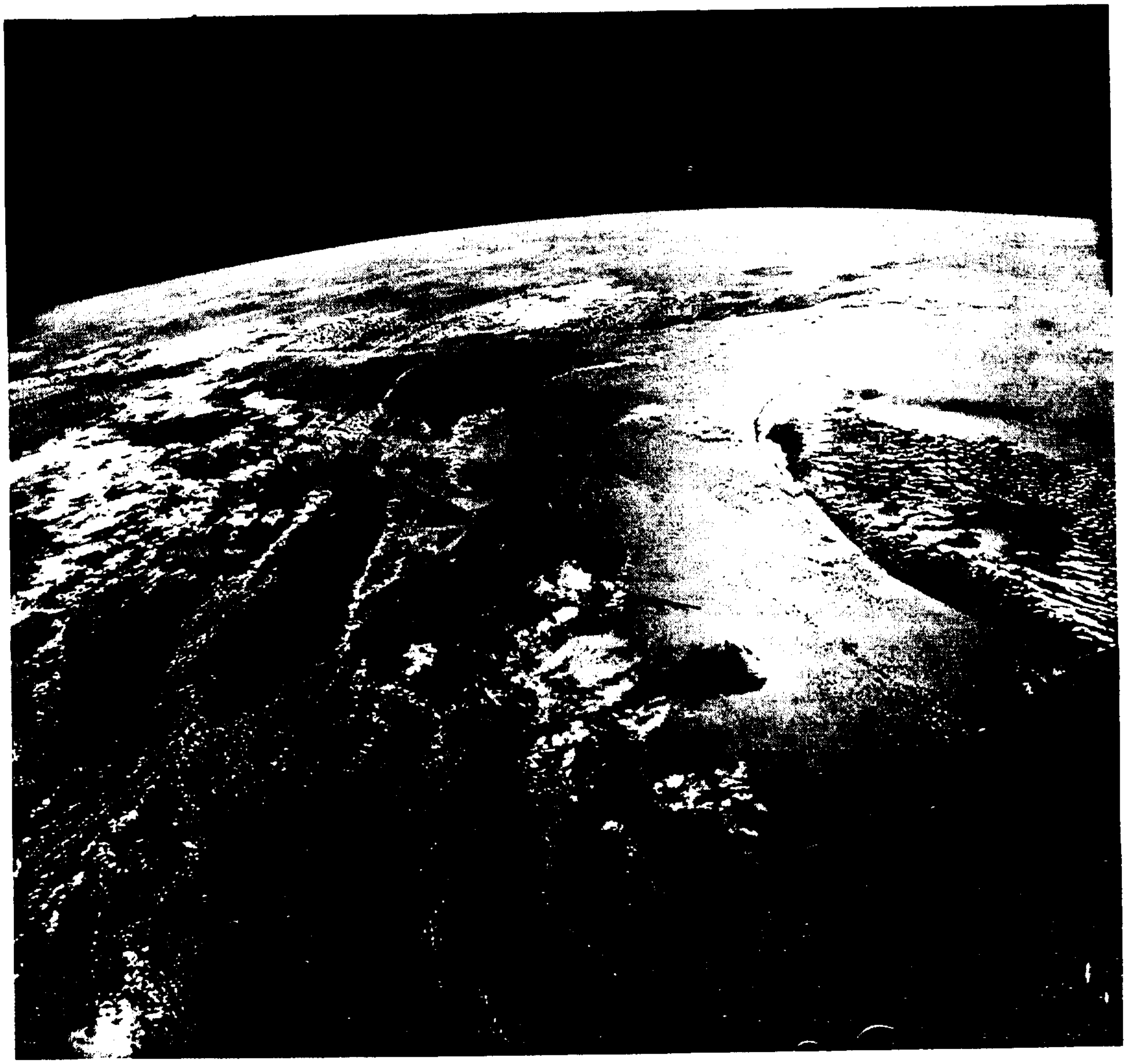

(d) Sunglint from the ocean surrounding the southern part of Florida on the right and nearby Bahama Islands in the foreground. Spacecraft is pointing south with Cuba in the background. Taken at 22 hours 32 minutes g.e.t. on November 11, 1966.

Figure 8.7-1. - Concluded.

8.8 EXPERIMENT S010, AGENA MICROMETEORITE COLLECTION

8.8.1 Objective

The basic scientific objective of the S010 Agena Micrometeorite Collection experiment 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.

8.8.2 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 GATV 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 removed this fairing cover. Figures 8.8-1 and 8.8-2 show the S010 hardware in both the closed and open positions attached to the GATV. Figure 8.8-2 includes the actual placement of specimens within the hardware package.

8.8.3 Procedures

During EVA and while the spacecraft was docked with the GATV, the extravehicular pilot was to have activated the S010 micrometeorite experiment hardware, thereby exposing the inner collection surfaces to the outside environment. This occurred at 43 hours 11 minutes g.e.t.

8.8.4 Results and Conclusions

The S010 hardware was left on the GATV for possible recovery during a later mission. After Gemini XII recovery operations were completed, an attempt was made to put the GATV into a higher, longer-life orbit. The attempt failed because of primary propulsion system malfunctions. The calculated GATV lifetime in the present orbit is 84 days, which is insufficient for experiment hardware retrieval during any later mission.

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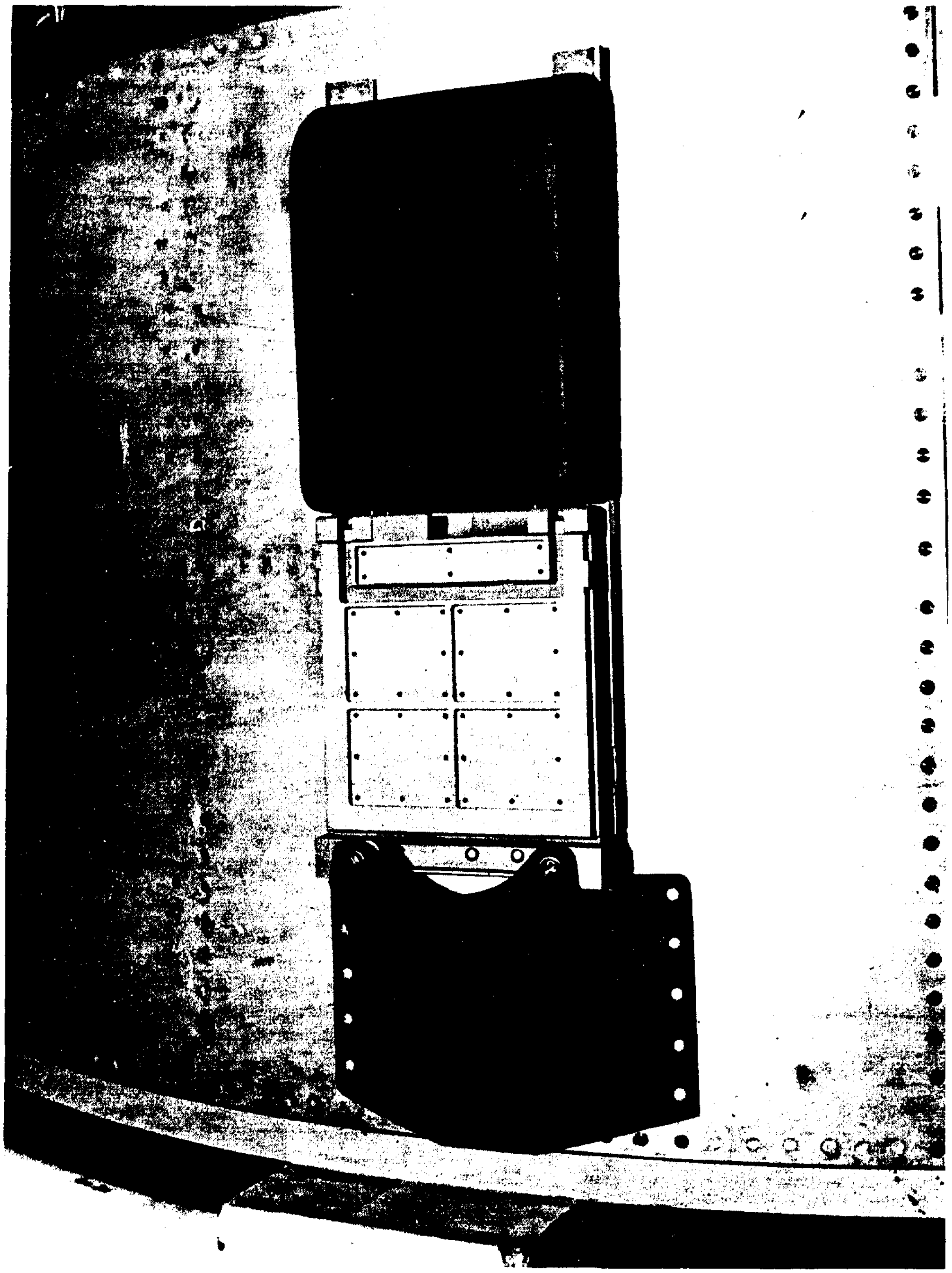

Figure 8.8-1. - Experiment S010, micrometeorite collection device installed in launch configuration.

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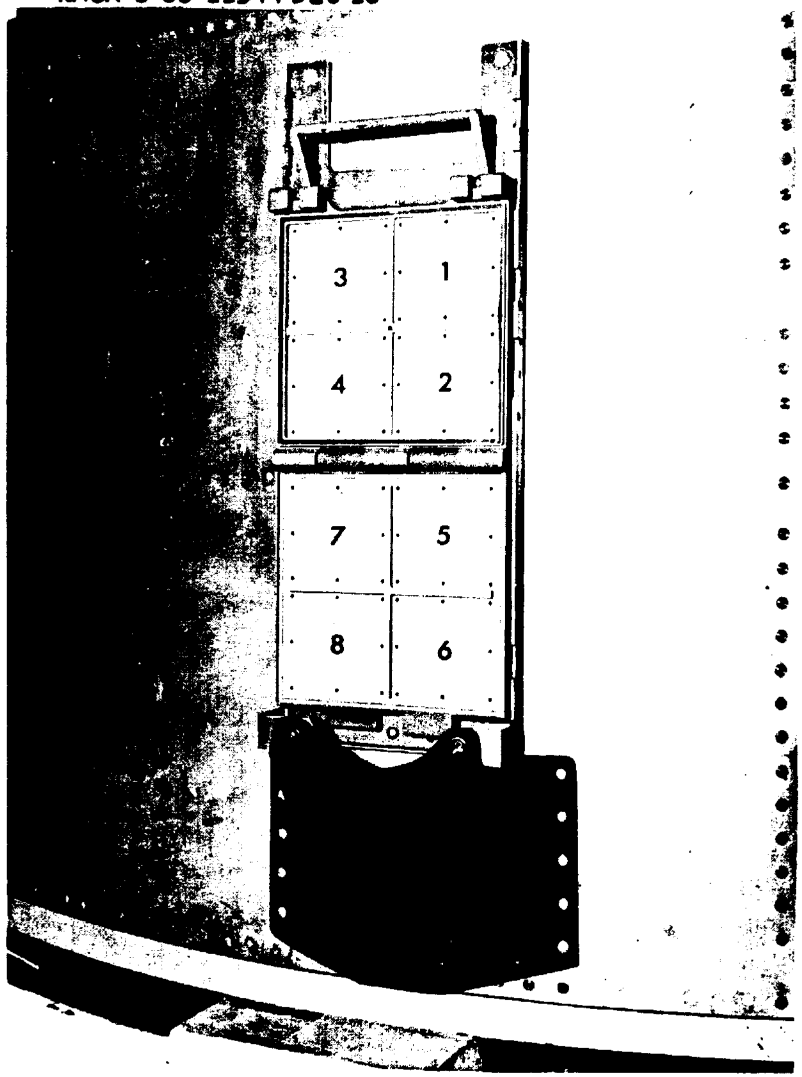

1. Shielded screening
2. Chronium-coated glass
3. Biological exposure with filters
4. Glass
5. Polished glass
6. Gold foil two μ thick; two layers plus thick plastic film below
7. Stereoscan samples, copper and stainless steel
8. Stainless steel

Figure 8.8-2. - **Experiment** S010, specimen loading.

8.9 EXPERIMENT S011, AIRGLOW HORIZON PHOTOGRAPHY

8.9.1 Objectives

The primary objective of the S011 Airglow Horizon Photography experiment for this mission was to obtain photographs of the twilight and nighttime airglow using narrow-band objective filters. Another objective was to take several photographs without the use of optical filters.

Three optical emissions bands were chosen for study—the green line at 5577Å caused by atomic oxygen, the sodium D lines at 5893Å, and the oxygen red line at 6300Å. The first two emission bands lie in layers centered at an altitude of approximately 90 kilometers and can be photographed edge-on from nominal orbital altitudes. The red line occurs at a higher altitude and must be photographed from an altitude 300 kilometers or greater.

8.9.2 Equipment

The basic components are shown in figure 8.9-1. These include the 70-mm general-purpose still camera with an f/0.95 lens and two film magazines loaded with black and white film. These magazines were shared with the S029 and S051 experiments. The split-field focal plane filter arrangement used on previous missions was not employed on this mission. Three narrow-band objective filters that passed more selected color radiation were used instead of the split field arrangement. These filters have peak transmittances occurring at wavelengths of 5577Å (green), 5893Å (yellow), and 6330Å (red). The green and yellow filters had half widths of 45Å and the red filter had a half width of 150Å in order to photograph the 6300Å to 6364Å doublet emission bands caused by atomic oxygen. An illuminated sight mounted on top of the camera was used with an adjustable window-mounted bracket so that camera motion in the pitch plane could be minimized.

8.9.3 Procedures

To obtain as many twilight exposures as possible, the experiment activities were accomplished during three revolutions. The first two sequences were scheduled for the two planned high-apogee orbits.

During the first sequence, beginning at 24 hours 13 minutes g.e.t., five exposures using the red filter were made of the western sunlit airglow after sunset. Exposure times ranged from 4 to 40 seconds, increasing with time from sunset. Seven exposures were made during the night portion of this same revolution. These included three photographs without any filter and two each with the red and green filters. For the second sequence, beginning at 25 hours 43 minutes g.e.t., the procedures were the same except that the yellow filter replaced the red filter. The third and last sequence commenced at 70 hours 45 minutes g.e.t., and consisted of three twilight exposures with the yellow filter, followed by eight 3-second exposures of the night airglow without any filter (fig. 8.9-2). For these latter pictures, the spacecraft was yawed 50 degrees between exposures.

8.9.4 Results

The flight crew obtained 23 good pictures of the sunlit and night airglow in the wavelength bands described. In the oxygen-green and sodium-yellow photographs there is clear evidence that the primary desired emission line is recorded on the film, with negligible contaminating radiation. Figure 8.9-3 shows this radiation band.

No high-apogee orbits were attained during the mission; however, the twilight photographs with the 6300Å filter do show a low-layer emission band, presumably caused by the OH radical which emits in the red wavelengths (fig. 8.9-4). This is the first photograph of an OH layer from a spacecraft and represents an interesting and useful addition to airglow observations.

The shutter in the f/0.95 lens assembly stuck in the open position many times during the experiment and caused several overexposed frames in addition to the 23 good ones obtained. The flight crew repeated several frames when they were aware of the malfunction and much of the experiment success can be attributed to the crew's inflight efforts. Since the shutter malfunction did cause deviations from the timing of the scheduled experiment sequences, the geographical location of the spacecraft during some of the exposures will have to be redetermined.

8.9.5 Camera Failure Analysis

The primary camera failure was an open shutter in the red lens assembly, thereby overexposing the photographic film. The probable cause for the open-shutter condition was the use of excessive force when mounting the red lens assembly to the camera body and misalignment of the shutter actuating coupling due to partial camera shutter cock. The coupling misalignment can cause dowel pin deformation, resulting in a chain of misalignments and possible shutter override, thereby preventing the return movement of the shutter closure mechanism.

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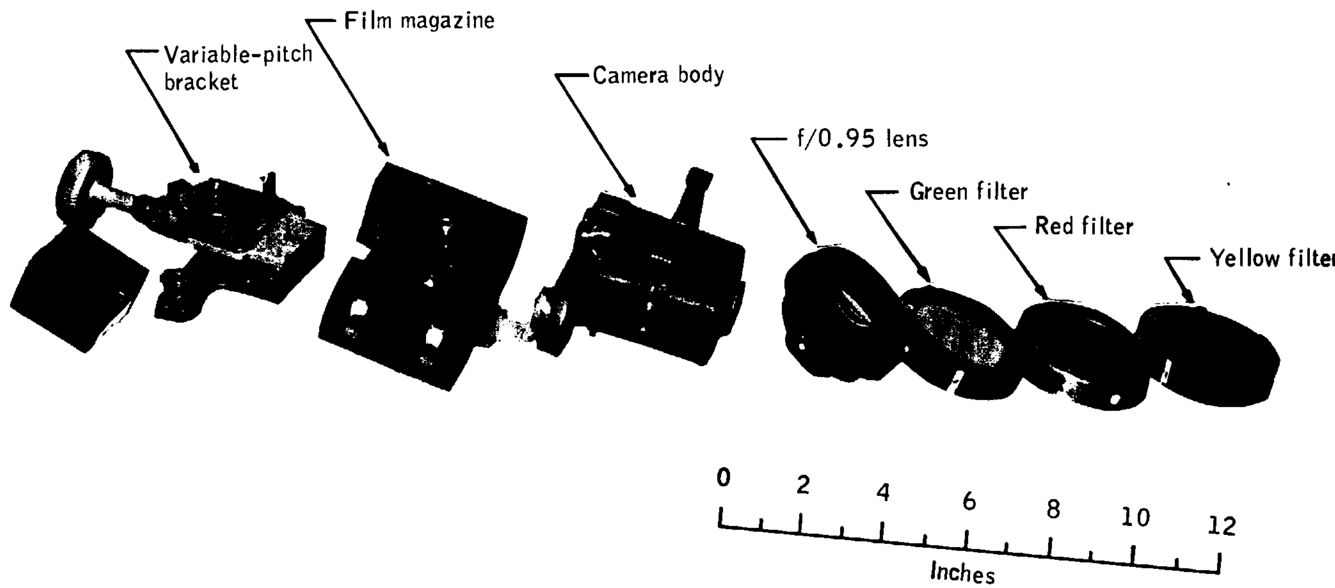

Figure 8.9-1. - Experiment S011, camera system.

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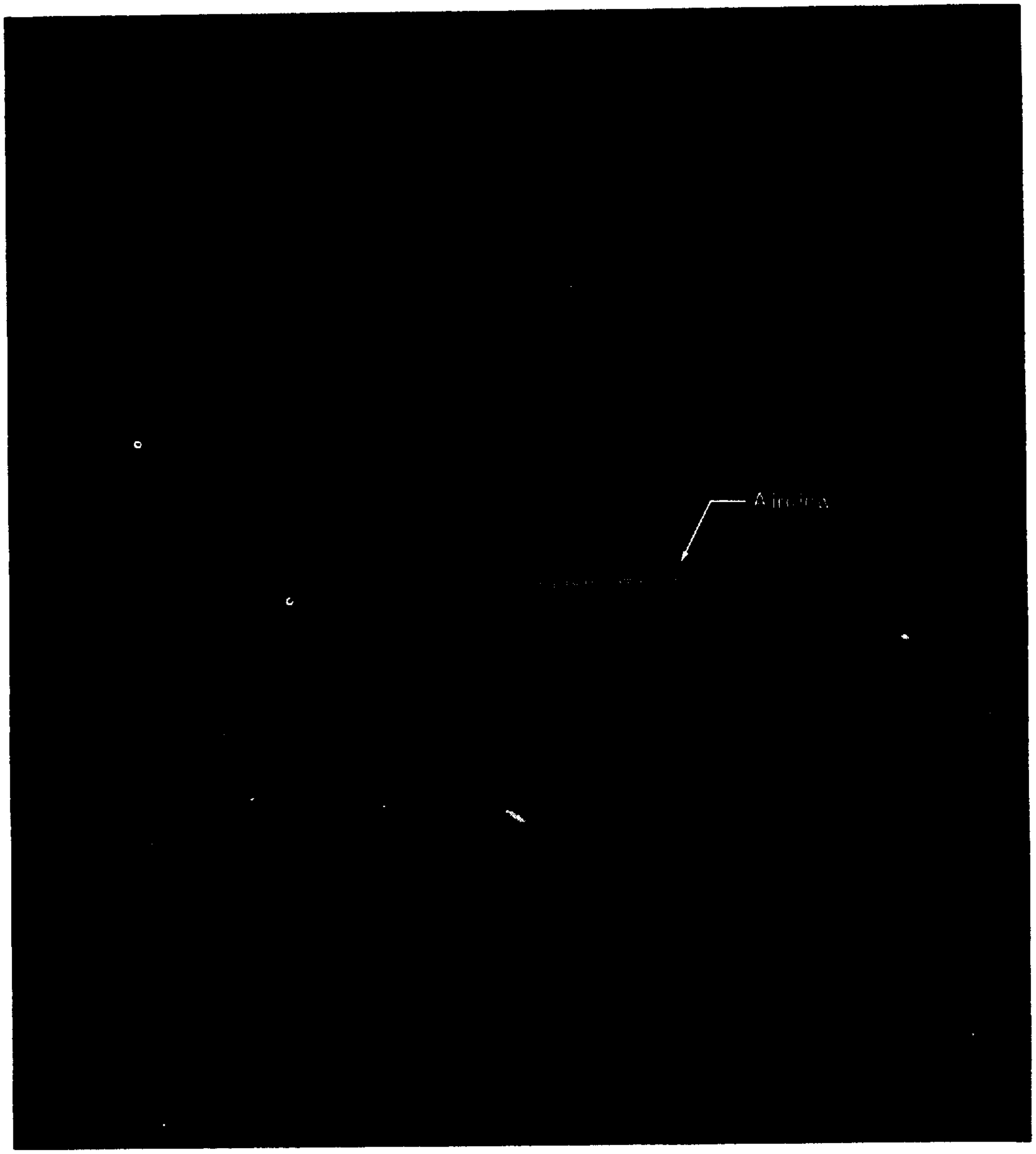

Figure 8.9-2. - A 3-second exposure of the night airglow without optical filters. Stars and lights from cities are easily observable in the upper and lower areas, respectively.

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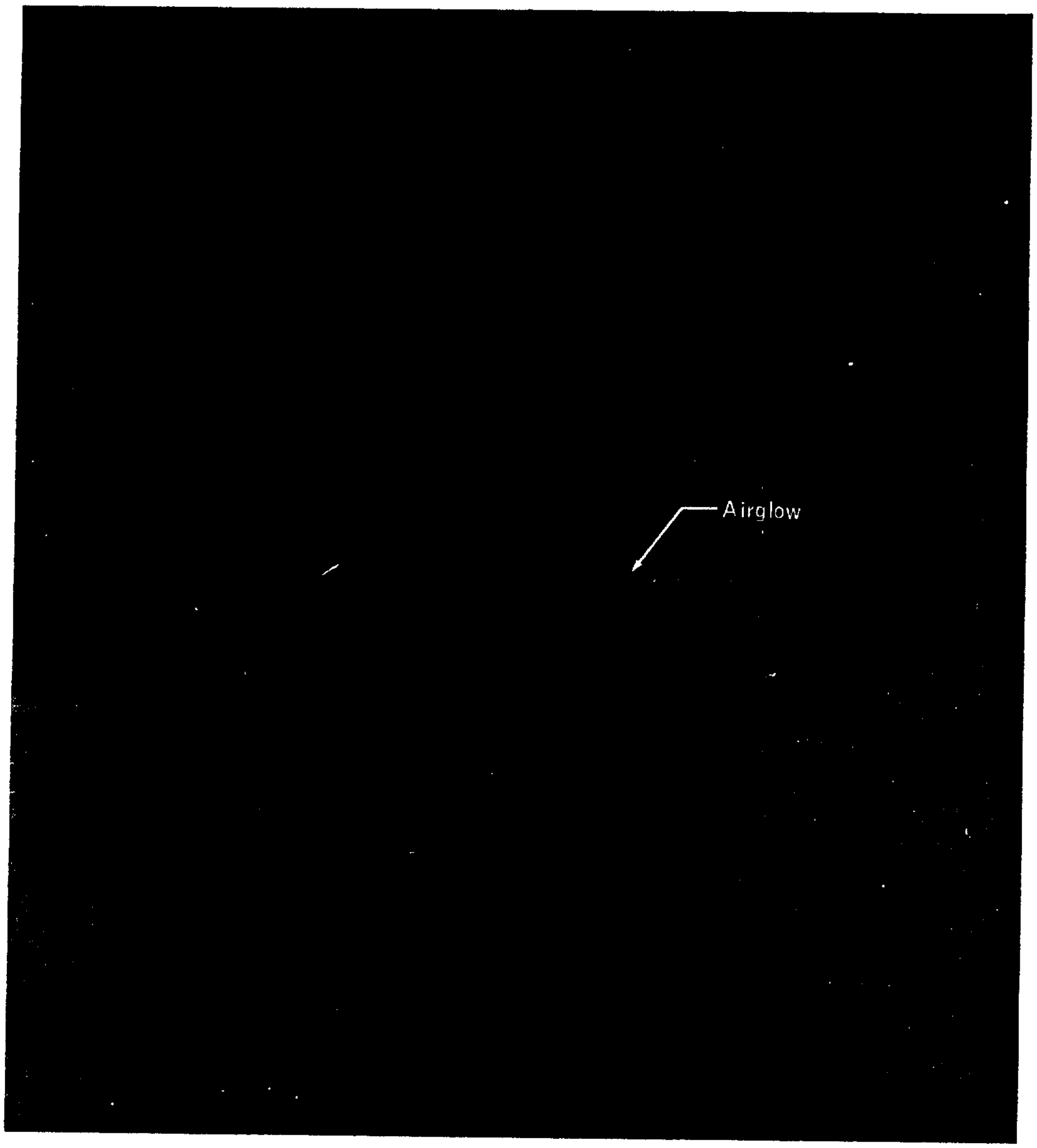


Figure 8.9-3. - A 20-second exposure using a green interference filter. The airglow does not extend the entire width of the picture as was anticipated.

NASA-S-66-11320 DEC 13

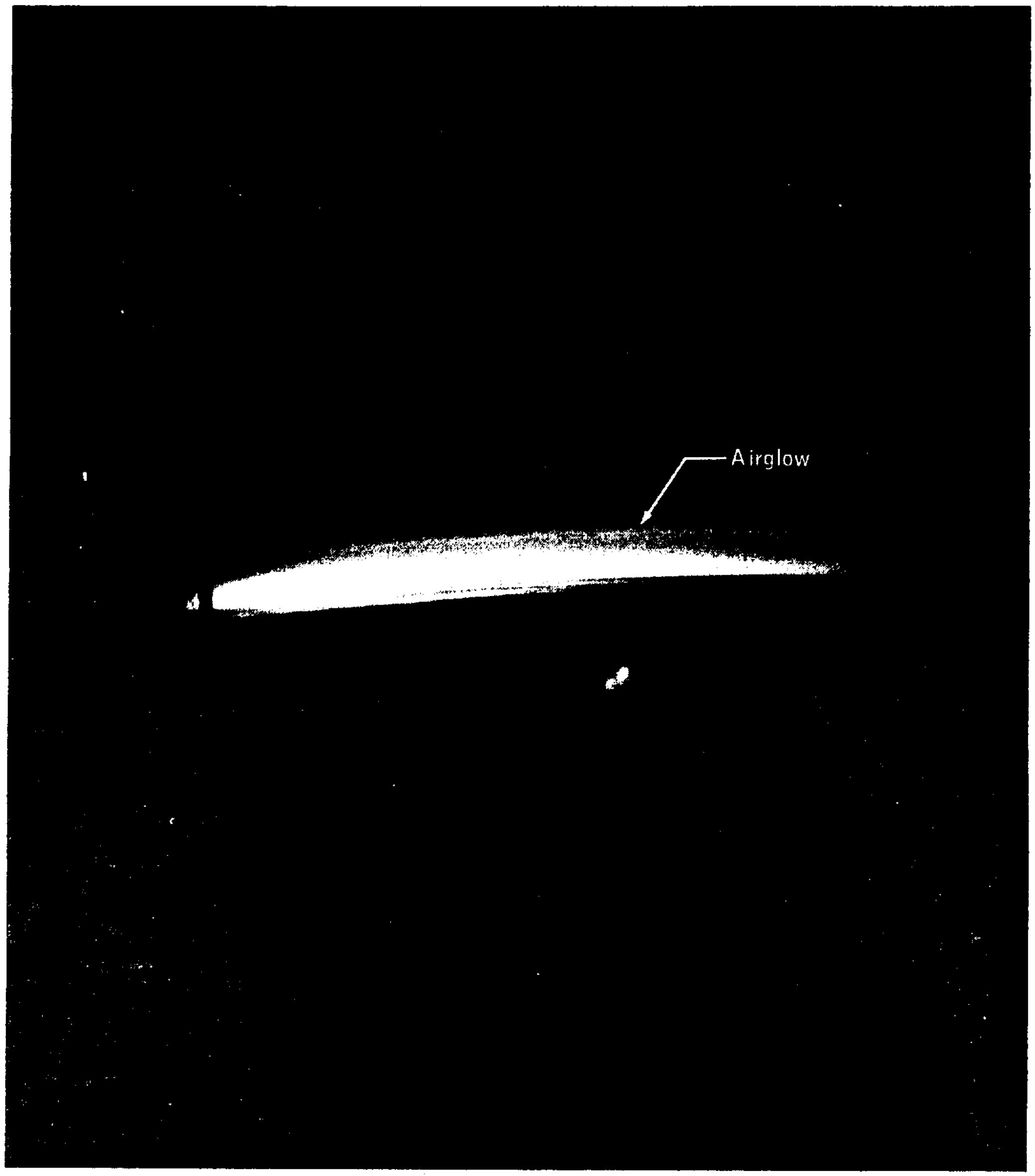


Figure 8.9-4. - A 20-second exposure using a red filter — taken four minutes after sunset. The emission layers observed are probably due to the OH radical.

8.10 EXPERIMENT S012, MICROMETEORITE COLLECTION

8.10.1 Objectives

The objectives of the S012 Micrometeorite Collection experiment were to determine the micrometeorite activity in a near-earth environment and to study the effect of the environment on biological microorganisms.

8.10.2 Equipment

The basic objectives were to be accomplished by exposing polished metal and plastic surfaces to the environment outside the Gemini spacecraft. Environmental data to be acquired included 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. Laboratory tests have shown these organisms to be resistant to adverse conditions, hence their selection for space studies. All material specimens were to be returned to earth by stowage in the Gemini reentry assembly for postflight examination and analysis at special laboratories.

The micrometeorite collection hardware consisted of an aluminum structure mounted on the spacecraft adapter retrograde section. Mounting spaces were designed for 24 surfaces, materials, or specimens. Figure 8.10-1 shows the hardware configuration on the spacecraft. The location and the type of specimens used within the experiment are listed in table 8.10-I. The sponsoring agency for each test object is also shown. Photographs of these specimens and their placements are shown in figure 8.10-2. The collector cover door was remotely controlled by ground command, thereby allowing the cover to be opened or closed, as required, to expose the experiment samples.

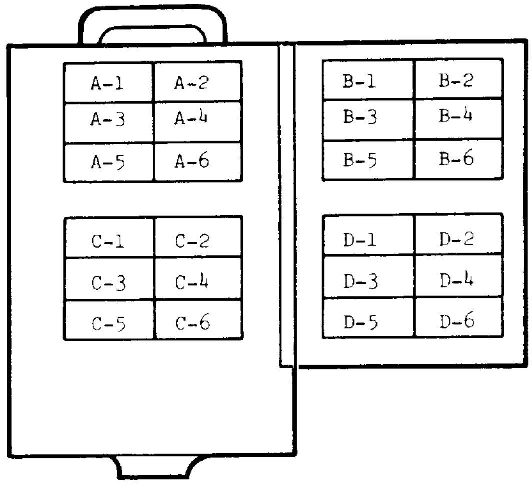
8.10.3 Procedures

The cover door of the micrometeorite collection device remained in the closed position until the first crew sleep period. This activation time was required to prevent exposing the sample surfaces to particles caused by thruster firing, fuel-cell purging, or dumping of liquids overboard. The collector door was left open for one period of 6 hours 20 minutes.

8.10.4 Results

The experiment equipment was opened by ground commands at 8 hours 5 minutes g.e.t., during the crew sleep period. It was closed and locked by ground command at 14 hours 29 minutes g.e.t. The equipment package was retrieved by the pilot during standup EVA at 20 hours 26 minutes g.e.t. and stowed in the spacecraft. Recovery for postflight analysis was satisfactory. Preliminary analyses of the exposed plates have not been completed at this time.

TABLE 8.10-I.- S012 LOADING FOR GEMINI XII



Location	Sponsor	Specimen type
A-1	Dudley Observatory	Nitrocellulose film over glass
A-2	Air Force Cambridge Research Laboratory	Nitrocellulose film
A-3	Dudley Observatory	Biological exposure
A-4	Dudley Observatory	Biological exposure
A-5	Dudley Observatory	Stereoscan sample, gold and indium-coated glass
A-6	Dudley Observatory	Nitrocellulose film over glass
B-1	Air Force Cambridge Research Laboratory	Layers of nitrocellulose film
B-2	Dudley Observatory	Stereoscan sample - copper
B-3	U. S. Geological Survey	Nitrocellulose on gold mesh
B-4	Tel Aviv University	Penetration through film
B-5	Dudley Observatory	Layers of silicon oxide film
B-6	Max Planck Institute	Layers of nitrocellulose film
C-1	Dudley Observatory	Nitrocellulose film over glass
C-2	Max Planck Institute	Stereoscan plates
C-3	Dudley Observatory	Sterile collection plates
C-4	Dudley Observatory	Sterile collection plates
C-5	Ames Research Laboratory	Gold-coated plastic
C-6	University of Washington	Polished copper
D-1	Dudley Observatory	Stereoscan samples - stainless steel
D-2	Manned Spacecraft Center	Aluminum on stainless steel
D-3	Birkbeck College	Aluminum on stainless steel
D-4	Smithsonian Observatory	Gold on plastic
D-5	Goddard Space Flight Center	Chromium on glass
D-6	Ames Research Laboratory	Metal-coated plastic

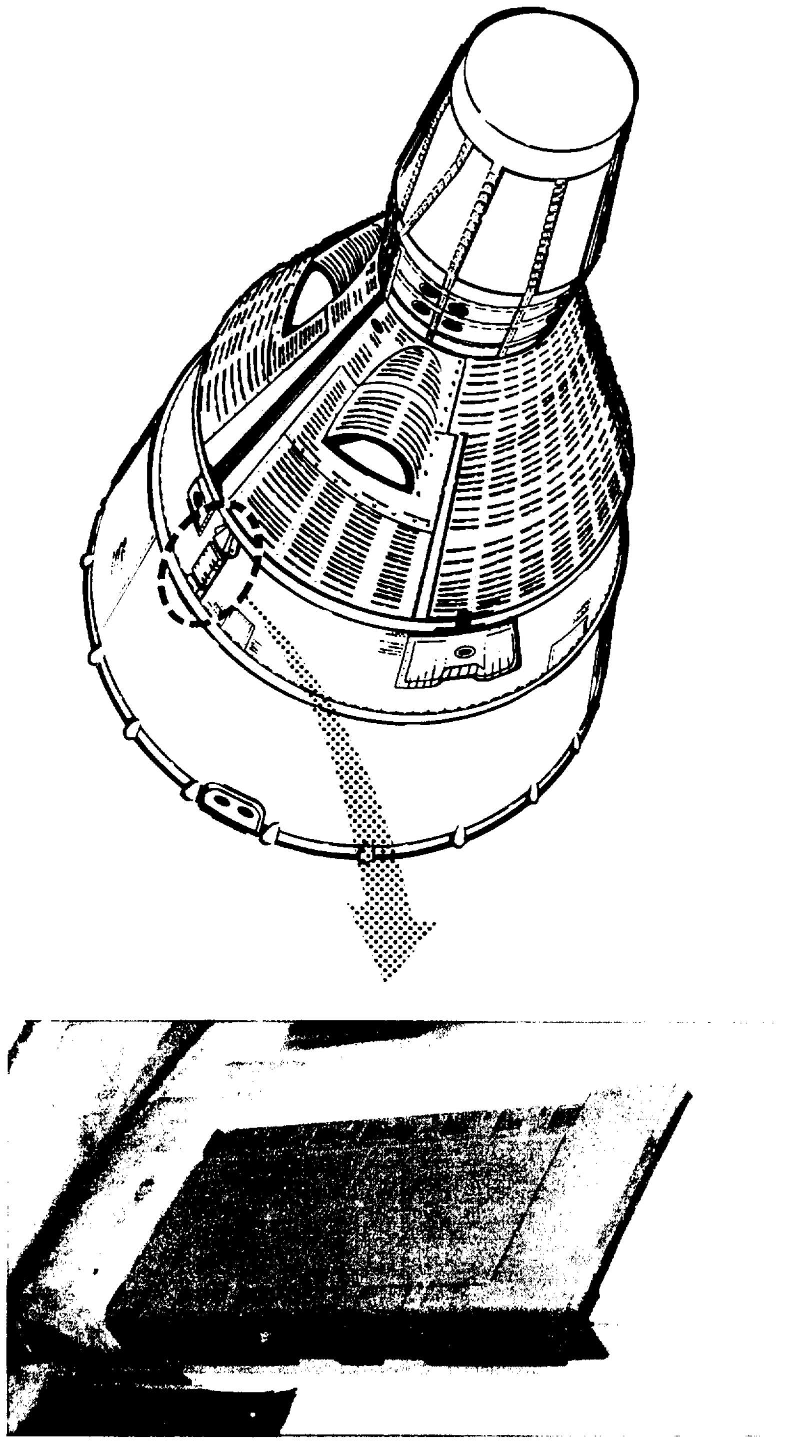


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Figure 8.10-1. - Experiment S012, hardware location.

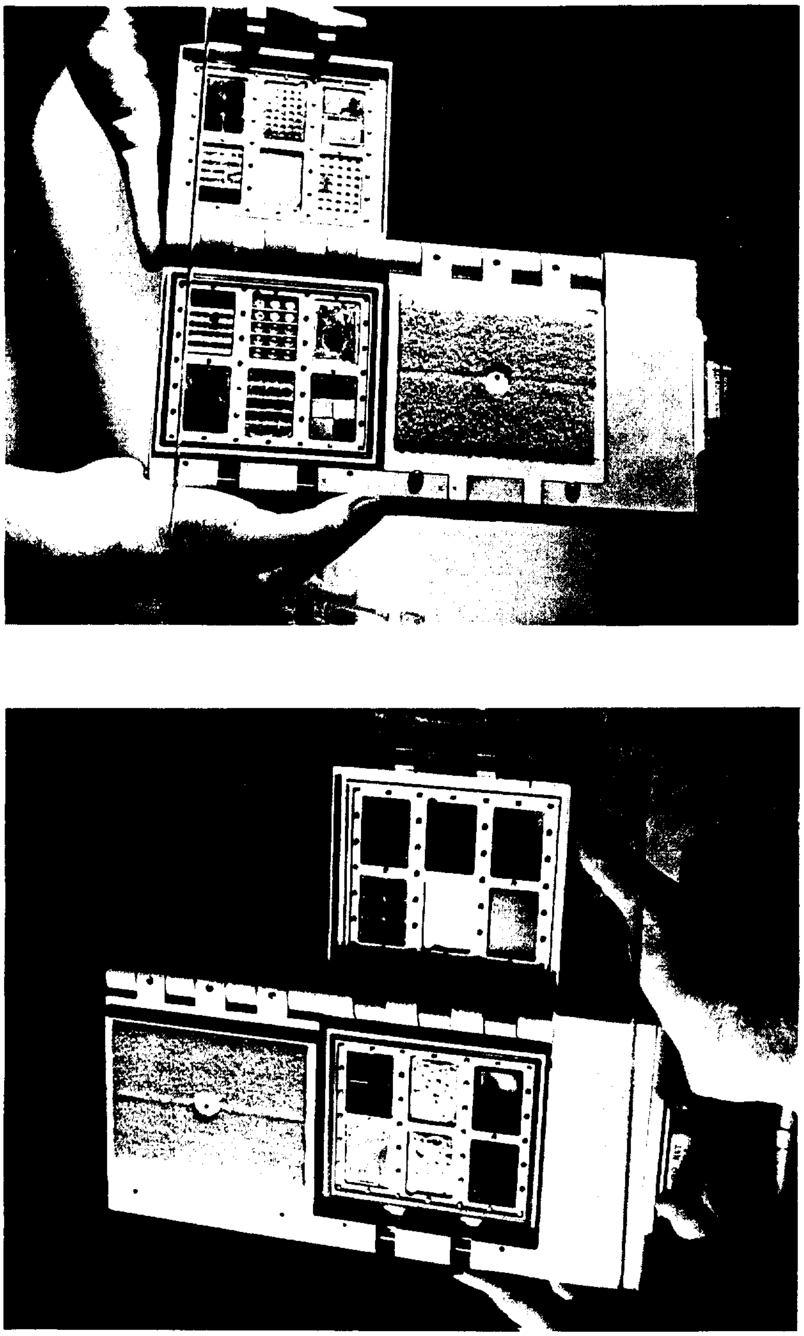

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Figure 8.10-2. - Experiment S012, specimens (see table 8.10-1 for identification).

8.11 EXPERIMENT S013, ULTRAVIOLET ASTRONOMICAL CAMERA

8.11.1 Objective

The fundamental objective of the S013 Ultraviolet Astronomical Camera experiment was to record the ultraviolet radiation of stars in the wavelength regions from 2000 to 4000Å. The objective was to be accomplished by recording radiation spectra, using the 70-mm general-purpose camera and an objective prism or an objective grating. An analysis of the surface temperatures of these stars, of the absorption effects taking place in their atmospheres, and of the absorption effects of the interstellar dust will be made of the photographic data obtained. The high resolution photographs are expected to show the absorption and emission lines, making possible the study of atomic excitation and ionization processes in these wavelength regions.

In addition to the acquisition of basic astronomical data, techniques by which objective-prism spectra may be best obtained were determined. The practical experience gained will be useful in planning similar astronomical observations with larger telescopes on future missions.

8.11.2 Equipment

The experiment equipment consisted of the 70-mm general-purpose camera equipped with a 73-mm ultraviolet lens, a 10-degree objective prism in a cell which attached to the ultraviolet lens, and a reflection grating in a cell which attached to the ultraviolet lens.

8.11.3 Procedures

Prior to the standup EVA, the pilot unstowed the camera and the prism or grating, then locked them to the bayonet joint of the lens. The camera was then attached to the bracket located near the pilot's seat.

After hatch opening, the spacecraft was pointed toward the first star target, using a reticle located on the command pilot's window. Because the camera axis was parallel to the roll axis of the spacecraft, the roll rate was the least critical of the three spacecraft motions. Roll rates up to 0.5 deg/sec could have been tolerated with little loss of image definition. Both pitch and yaw rates were to be decreased to 0.1 deg/sec or less. Pitch motion was the most critical because the pitch axis was parallel to the direction of dispersion and motion would degrade the wavelength resolution of the spectra.

Six time exposures were made on each star field and the film advanced between each exposure. Two exposures of one minute and two of two minutes were made during periods when the stabilizing thrusters were operated to hold the spacecraft attitude constant. The additional two exposures were of 30 seconds duration.

The experiment was performed while the spacecraft was docked with the GATV in order to use the GATV control system for stabilization. During each set of exposures, the GATV was stabilized using flight control mode 2, with the geo-rate and horizon stabilization switched OFF.

8.11.4 Results

Four star fields were photographed: two with the grating and two with the prism. The grating fields were centered on γ Cassiopeiae and Sirius. The prism fields were centered on Deneb and γ Velorum. Grating spectra in the γ Velorum field and prism spectra in the Algol field were not obtained because of crew workloads. The decision not to use platform orientation required the crew to use planned observing time at the beginning of each night in order to acquire the first star field.

There were apparently no major problems in the assembly and operation of the camera. Problems do remain, however, concerning focus, static marks, and light leaks.

The image quality produced by the camera (as judged by zero-order grating images) was considerably better than that obtained during the Gemini XI flight. In particular, the central area of very poor focus has been eliminated, indicating that the increased tension of the film-retaining spring used in the Gemini XII camera prevented warping of the film. The worst image diameter on the Gemini XII photographs is about 100 microns whereas the worst image diameter on the Gemini XI photographs is about 200 microns. The superb resolution in the spectrum of Sirius (as shown in figure 8.11-1) is a good illustration of the improved image quality with the grating.

The best prism spectra do not appear as sharp as those of the Gemini XI flight. The reason for this discrepancy between the performance with the grating and that with the prism is not immediately apparent.

All frames show effects of static electricity. The carbon dioxide cartridge was less effective in eliminating this effect than it had been during the Gemini XI mission. Possibly the difference is due to gas loss by a greater-than-expected venting rate through the film back. The static marks do not interfere with the study of lines and bands in the spectra but they will reduce the accuracy of photometric measures on the film.

Although light leaks are not severe, some fogging of frames adjacent to the central one are noted during camera exposure to full sunlight during the interval between night orbits.

The GATV guidance was generally good during the grating pass but only fair during the prism pass. Two holds on Cassiopeiae (as seen in figure 8.11-2) were stable enough to give essentially point images on 2-minute exposures, while several holds on Canis Major showed very little motion in roll and pitch, with a smooth motion in yaw. During holds on Cygnus and Vela, several instances of excessive motion occurred in pitch and uneven motion in yaw.

8.11.5 Conclusions

A frame-by-frame log of the flight film is given in table 8.11-I. An excellently widened spectrum of γ Cassiopeiae shows no lines whatever. This star is of particular interest for having ejected a shell which subsequently dissipated. The spectrum in recent years has been that of an ordinary, rapidly rotating B star, but within the past year there have been indications of hydrogen emission lines. These observations provide negative evidence for the return of a shell spectrum at this time.

The line spectra of ϵ Cassiopeiae, δ Cassiopeiae, and β Cassiopeiae are all resolved on two frames, as shown in figure 8.11-2. The variations in ultraviolet energy distribution with spectral type are particularly striking in this photograph. Lines below 3000Å are seen in the spectra of β Cassiopeiae and δ Cassiopeiae, while the middle-ultraviolet spectrum of ϵ Cassiopeiae is devoid of strong features.

The middle-ultraviolet spectrum of Sirius (in figure 8.11-1) shows six or more fine absorption lines. The 2795.5 and 2802.7Å lines of MgII are here resolved for the first time in a photograph of a stellar spectrum. Lines are also seen in the spectra of α Leporis and, possibly, ν Orionis.

The stronger grating spectra are listed in table 8.11-II. In both fields, the presence of very many weak spectra is suspected on the long exposures. The spectra listed in the table are for the most part well-enough exposed to yield measures of energy curves.

The failure to obtain grating spectra of γ Velorum and ζ Puppis star fields is unfortunate, since they were well placed for observation during the mission. Also, the grating spectra from this flight were of high quality.

The prism spectra are of lower quality than the best Gemini XI results. On most frames, little detail is apparent in the spectra. The helium discontinuity is apparently present in the spectra of HR 7767 and 40 Cygni; metal multiplets are apparently present in the spectrum of γ Cygni; and some detail is suspected in the spectra of HR 3817, γ Velorum, and ζ Puppis.

TABLE 8.11-I.- S013 INFLIGHT EXPOSURES WITH GRATING ATTACHED

Frame[a]	Field	Vehicle attitude hold	Remarks
S66-63557	Wasted frame	--	Lightstruck
58	Cassiopeia	Fair	Images triple; spectra streaked; no lines
59	Cassiopeia	Excellent	Spectra smoothly widened; no lines
60	Cassiopeia	Good	Spectra unevenly widened; lines in β Cassiopeia, δ Cassiopeia
61	Cassiopeia	Fair	Spectra unwidened, slightly trailed in wavelength direction; lines in β Cassiopeia, δ Cassiopeia, ζ Cassiopeia
62	Cassiopeia	Good	Spectra rather narrow; lines in β Cassiopeia, ϵ Cassiopeia, ζ Cassiopeia, λ Cassiopeia
63	Cassiopeia	Good	Spectra rather narrow; lines in β Cassiopeia, ϵ Cassiopeia, ζ Cassiopeia, λ Cassiopeia
64	Canis Major	Excellent	Spectrum rather wide; many fine lines below 3000Å
65	Canis Major	Fair	Spectrum rather narrow; Balmer lines in Sirius
66	Canis Major	Fair	Spectra unevenly widened; lines of MgII, FeII
67	Canis Major	Good	Spectra smoothly widened; many lines below 3000Å
68	Canis Major	Excellent	Spectra smoothly widened; lines below 3000Å
69	Canis Major	Good	Spectra smoothly widened; lines around 2400Å; lines in α Lep, ν Ori
S66-63570	Canis Major	Good	Sirius ultraviolet superposed on sunlit GATV

[a]All frames are marred by static marks or light leaks.

TABLE 8.11-II.- S013 INFLIGHT EXPOSURES WITH PRISM ATTACHED

Frame[a]	Field	Vehicle attitude hold	Remarks
S66-63571	Wasted frame	------	Grossly overexposed; positive image of GATV and sunlit earth
72	Cygnus	Fair	Spectra streaked; no detail
73	Cygnus	Poor	No detail
74	Cygnus	Poor	Some detail
75	Cygnus	Fair	Some detail
76	Cygnus	Poor	No detail in spectra
77	Puppis-Vela	Poor	Spectra very wide; possible detail
78	Puppis-Vela	Poor	No detail
79	Puppis-Vela	Poor	No detail
80	Puppis-Vela	Fair	Spectra streaked; apparent line in 3817
81	Puppis-Vela	Poor	No detail
82	Puppis-Vela	Good	Spectra smoothly widened; some detail
83	Puppis-Vela	Poor	Spectra streaked; no detail
84	Puppis-Vela	Fair	Spectra streaked; apparent detail in γ Vel, ζ Puppis
85	Puppis-Vela	Fair	Spectra streaked; no detail
S66-63586	Wasted Frame	------	Lightstruck

[a] All frames are marred by static marks or light leaks.

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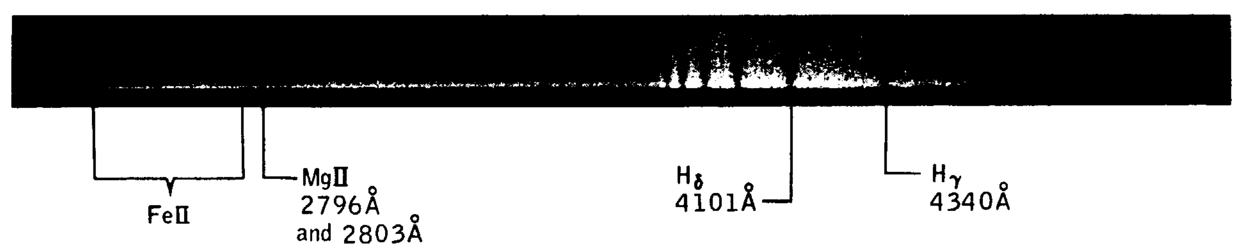

Figure 8.11-1. - Experiment S013, 20-second exposure of the ultraviolet spectrum of Sirius. The Balmer series of hydrogen appears at the right. The Mg II doublet at 2800Å and several sharp, but weak, lines of Fe II appear at the left.

NASA-S-66-11337 DEC 15

Obtained during the standup EVA, November 13, 1966. The docked GATV and the R and R section of the spacecraft are superposed on the starfield. The spectrum is produced by a diffraction grating which gives both a direct image (above) and a spectrum (below) of each star.

Figure 8.11-2. - Experiment S013, grating spectra taken of the region around Cassiopeia, exposure time of two minutes.

8.12 EXPERIMENT S029, EARTH-MOON LIBRATION REGIONS PHOTOGRAPHY

8.12.1 Objective

The objective of the S029 Earth-Moon Libration Regions Photography experiment was to investigate by photographic technique the possible existence of clouds of particles or dust orbiting the earth in these regions. The L_4 and L_5 libration points lie in the orbital path of the moon, 60 degrees ahead of and 60 degrees behind the moon. Dust clouds in these regions would be visible by reflected sunlight.

8.12.2 Equipment

The following equipment was used for experiment S029:

(a) The 70-mm general-purpose still camera with f/0.95 lens.

(b) Film magazine with black and white, high-sensitivity film.

(c) Experiment T002 and S011 camera mounting brackets.

8.12.3 Procedure

During the night pass on the 17th orbit, the camera was mounted to the pilot's window with the experiment T002 bracket, which aligned the camera to the spacecraft axis. In accordance with procedures, the spacecraft was oriented towards the constellation Capricornus, the starfield area in which the L_4 libration region would be present 45 hours later. Five photographs were taken of this area with a series of exposures approximately 30 seconds, 1 minute, and 2 minutes, and were taken with the spacecraft docked with the Gemini Agena Target Vehicle (GATV).

During the night pass of the 46th orbit, six exposures were taken of the L_4 libration point in the constellation Capricornus. The photographs were taken with the camera mounted on the Experiment S011 mounting bracket which aligned the camera perpendicular to the pilot's window but not parallel to the spacecraft axis. The spacecraft was then oriented so that the camera would be pointed at the libration point. The spacecraft was not docked with the GATV during the second sequence of pictures. Exposure times for the second set of photographs were 30 seconds, 1 minute, and 2 minutes.

8.12.4 Results

Eleven photographs were taken of the L_4 libration point, of which only three were exposed properly. A mechanical failure of the shutter mechanism caused overexposure of many of the photographs, and failure of the film advance at the end of the first roll caused an unknown number of double exposures. None of the first sequence of five photographs could be identified, one of which was totally unrecognizable as to image content. Star fields could be recognized in the two remaining photographs as shown in figures 8.12-1(a) and (b); however, because of difficulties in stabilizing the spacecraft, the stars were badly smeared. A light flare, the source unknown but possibly caused by light reflecting off the window of the spacecraft, was present in both pictures. A reconstructed star field area as seen by the camera is shown in figure 8.12-2.

8.12.5 Conclusions

Isodensitraces of the two recognizable star field (figs. 8.12-1 and 8.12-2) were made; because of image smear and the light flare, no conclusive results can be obtained.

8.12.6 Camera Failure Analysis

A concise failure analysis is discussed in section 8.9.5 of this report.

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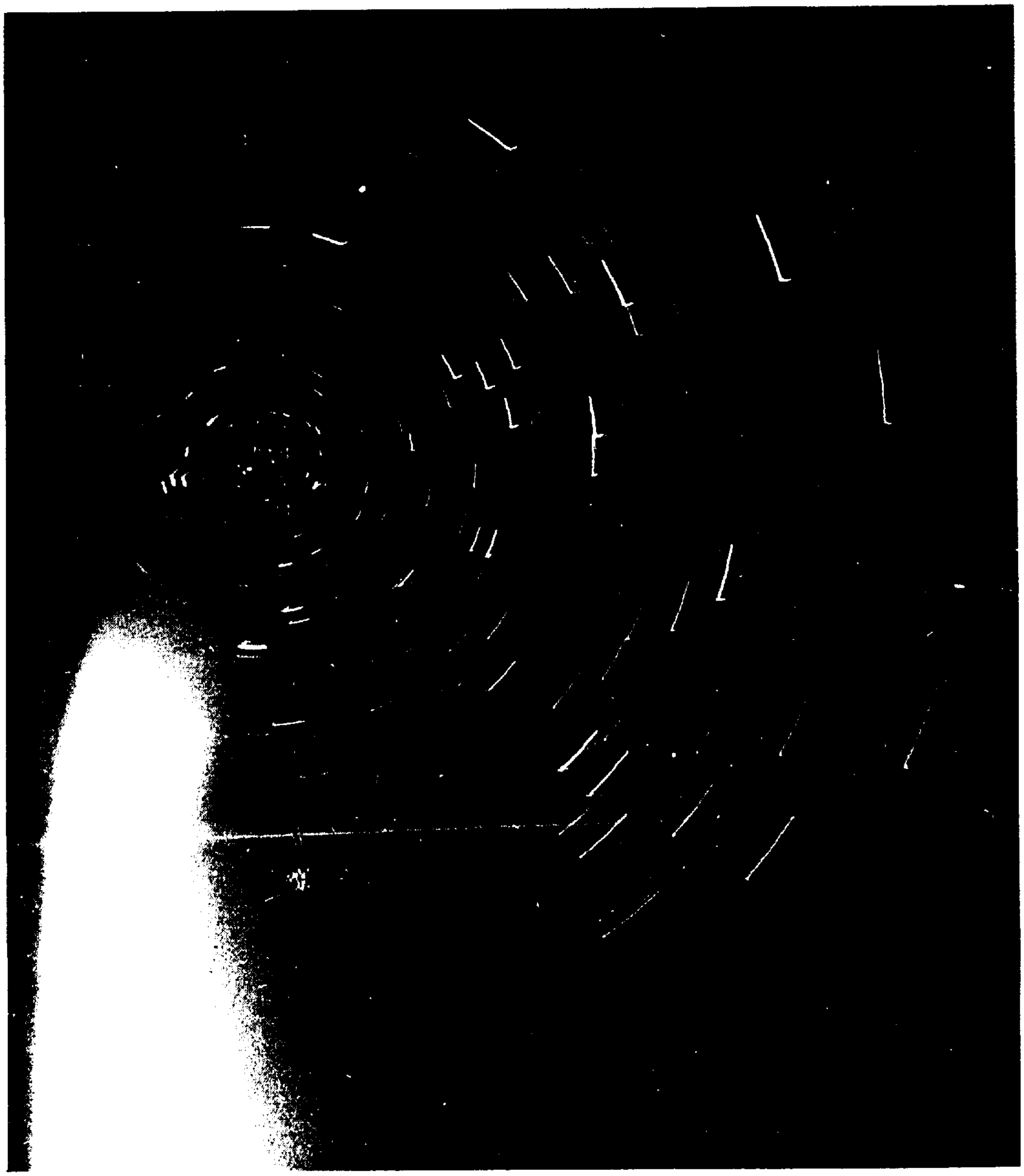
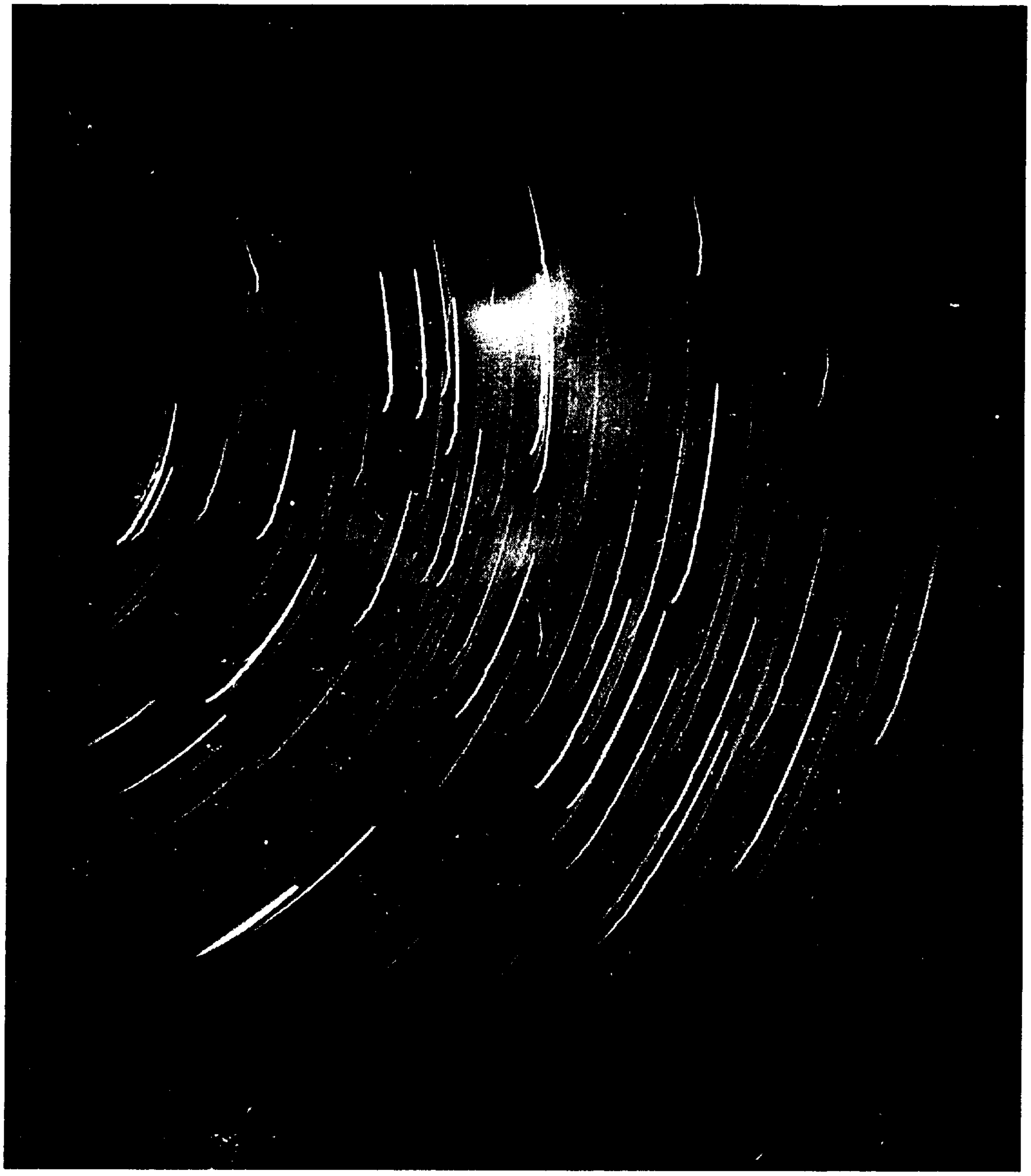


(a) Photograph of the L_4 libration region taken at approximately 73 hours 45 minutes g.e.t. with an exposure time of 30 seconds.

Figure 8.12-1.- Experiment S029, earth-moon libration region photography.

NASA-S-66-11314 DEC 13



(b) Photograph of the L_4 libration region taken at approximately 73 hours 45 minutes g.e.t. with an exposure time of 60 seconds.

Figure 8.12-1. - Concluded.

NASA-S-66-11315 DEC 13

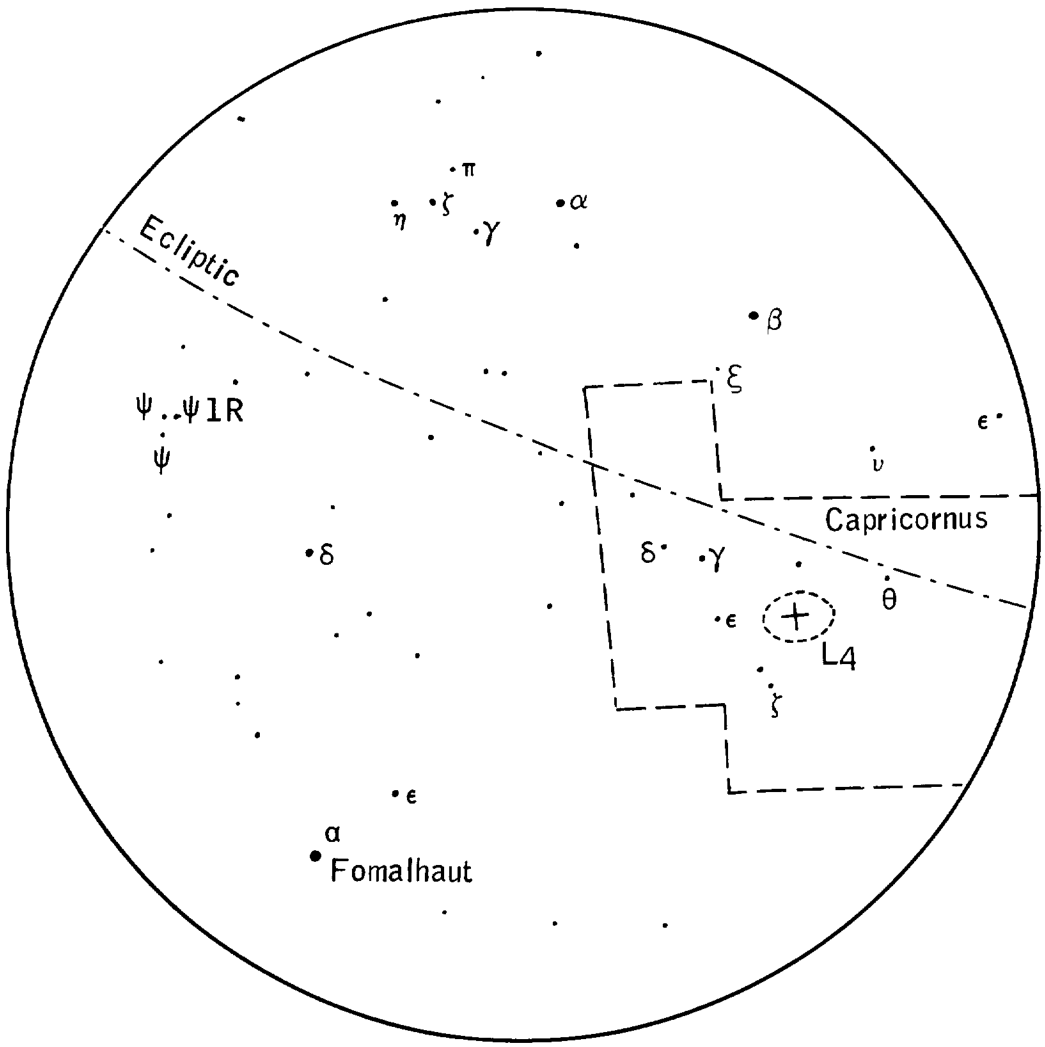


Figure 8.12-2.- Identification of stars shown in figure 8.12-1, showing the position of the L_4 libration point on November 14, 1966, at 73 hours 45 minutes g.e.t.

8.13 EXPERIMENT S051, DAYTIME SODIUM CLOUD

8.13.1 Objective

The objective of the S051 Daytime Sodium Cloud experiment was to measure the daytime wind velocity of the earth's high atmosphere as a function of altitude between 90 and 135 nautical miles.

The measurements were to be obtained from the deformations of a rocket-made vertical sodium cloud. Rockets launched in front of the spacecraft continuously ejected sodium vapor from approximately 35 to 100 nautical miles in altitude during ascent and descent. The vapor should have been visible from the spacecraft as a faint yellow cloud above the horizon.

8.13.2 Equipment

The clouds were to be sequentially photographed using the 70-mm general-purpose camera used on other experiments. At least 20 frames from the S011 experiment film pack were required for adequate photographic coverage.

An interference filter with wavelength response dependent on incidence angle was mounted on the camera. This filter allowed sodium yellow light to enter a camera cone angle of 10 degrees. For larger entrance angles, the wavelength response was displaced toward the green. It was essential that the camera be aimed precisely at the cloud direction within a tolerance of one degree.

8.13.3 Procedures

The sodium release was made by a Centaure rocket from the Hammaguir, Algeria, launch site, which is under the responsibility of the C.N.E.S. French Space Agency. This was a two-stage solid-propellant rocket which ejected sodium continuously from 70 kilometers to an apogee of 180 kilometers and down to the ground.

Two rockets were launched from Hammaguir in the southeast direction, with approximately 1-1/2 hours between the two launches. The firing conditions were such that the rockets descended from apogee when the Gemini spacecraft was at a distance of 1000 kilometers. Assuming a nominal spacecraft altitude of 161 nautical miles, the position of the cloud relative to the horizon could be determined.

From pictures obtained of airglow (Experiment S011), it is apparent that the horizon line does not correspond to the real geometrical limit of the solid earth but to the sunlit atmosphere up to an altitude of 30 kilometers. This has been taken as the horizon line and the horizon angle had been taken as 16.50 degrees below the spacecraft horizontal plane. Figure 8.13-1 shows the evolution of the sodium cloud sightings as probably observed by the flight crew. When the cloud is below the horizon, it is not visible because of poor contrast with sunlit background. The background may be 250 times more intense than the cloud. Therefore, this experiment can succeed only when the cloud is above the horizon.

From a study of this figure it is apparent that wind measurement can be obtained from slant distances between 1000 and 800 kilometers in the altitudes of interest. This was the basic objective of the experiment.

When the distance between the spacecraft and the cloud is less than 800 kilometers, the lowest part of the cloud begins to sink below the horizon. The possibility of obtaining useful pictures of the cloud on the top of the earth background was another objective to be obtained. It necessitated obtaining a series of exposures during the time the spacecraft went from a distance of 800 to a distance of 500 kilometers.

The average wind velocity measured is of the order of magnitude of 50 to 100 meters per second. To measure this velocity, the motion of the cloud has to be a few times larger than the film resolution. This corresponds to a difference in time between successive exposures of approximately thirty seconds at a distance of 1000 kilometers.

Two cloud pictures taken about 60 kilometers apart would provide tridimensional cloud shapes by stereogrammetric recombination. From a second identical pair of pictures taken some time later, the tridimensional configuration of the cloud could again be obtained. A comparison of the state of the cloud at different times would give the wind velocity in the atmosphere at all relevant altitudes, assuming that only horizontal winds exist. It has been found that a series of successive photographic pairs are necessary to reduce wind velocity errors to an acceptable level.

8.13.4 Results

This experiment was performed following launch of the sodium rocket from the French launch site at 62:41:48 g.e.t. Visual acquisition was not established by the crew; however, the spacecraft was pointed in the direction of the anticipated sodium cloud location and 12 exposures were taken. Another sodium rocket was launched from the launch site near Hammaguir, Algeria, at 64:16:49 g.e.t. The crew again did not visually observe the sodium cloud. Eight additional photographs were taken of the geographical areas specified in premission planning. The launch crew at Algeria confirmed the successful firing, activation, and observation of sodium clouds from the ground during both of the spacecraft passes. Preliminary evaluation of the film indicates that all twenty photographs were overexposed during the photographic sequences. The overexposures were caused by a camera shutter locked in the open position. Useful photographic data were therefore not obtained for analysis.

8.13.5 Camera Failure Analysis

A concise failure analysis is discussed in section 8.9.5.

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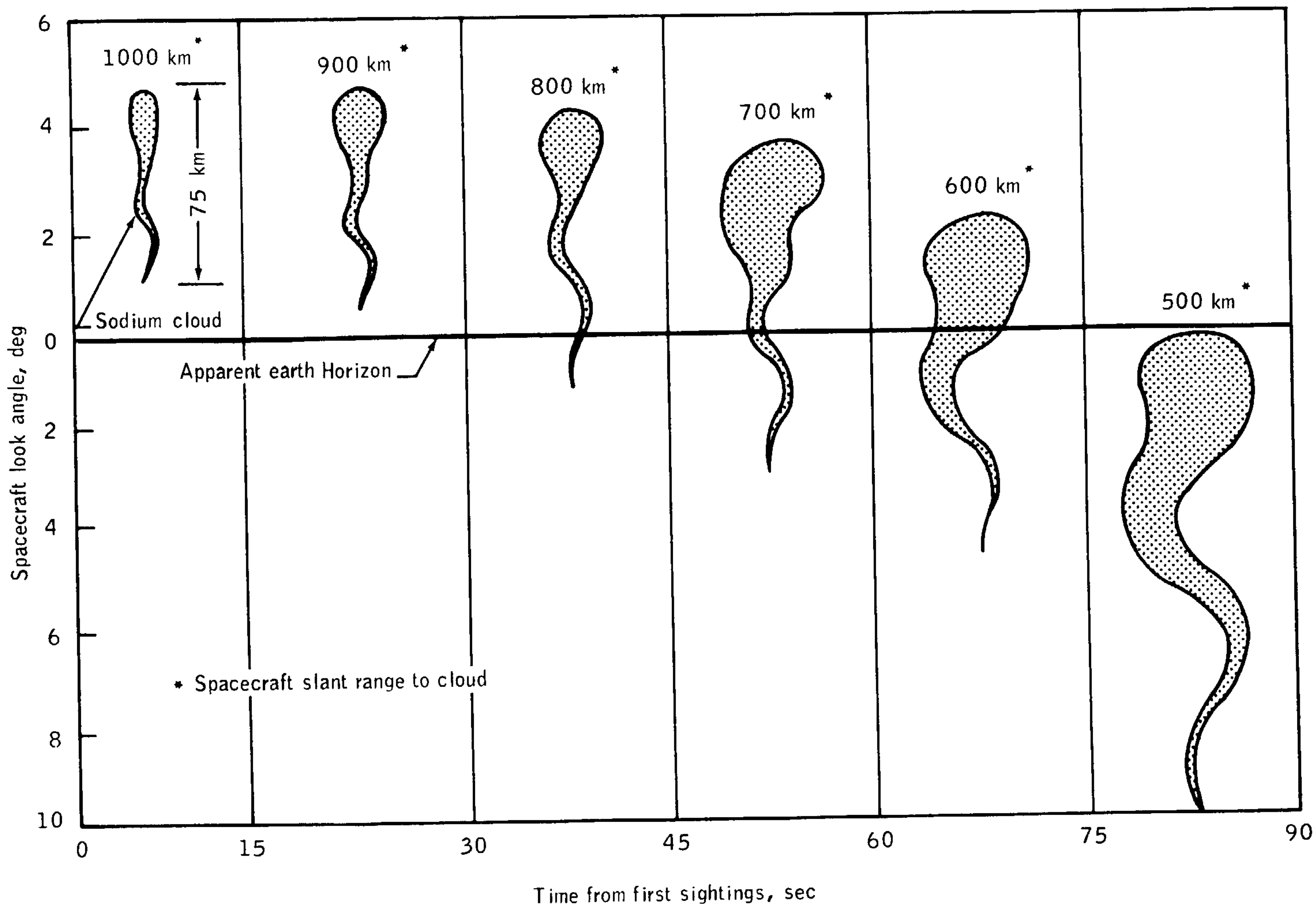

Figure 8.13-1. - Experiment S051, predicted sodium-cloud visual characteristics.

8.14 EXPERIMENT T002, MANUAL NAVIGATION SIGHTINGS

8.14.1 Objectives

The general objective of the T002 Manual Navigation Sightings experiment was to make navigation-type measurements through the window of the stabilized Gemini spacecraft using a hand-held sextant. The major objectives were to:

(a) Evaluate the flight crew's ability to make accurate navigation measurements using simple instruments in an authentic space-flight environment

(b) Examine the operational feasibility of the measurement techniques using the pressure suit helmet off and also with the helmet on and the visor closed

(c) Evaluate operational problems associated with the spacecraft environment

(d) Validate ground simulation techniques by comparison of the inflight results with baseline data obtained by the pilot using simulators and actual celestial targets from ground observatories.

8.14.2 Equipment

The equipment used in this experiment was a two-line-of-sight optical sextant, shown in figure 8.14-1. It was designed to measure (within 10 arc seconds) the angle between various types of celestial targets.

The view through the fixed line-of-sight of the sextant was imaged in the focal plane through a plate beamsplitter and an objective lens and prism-mirror erecting system, shown in figure 8.14-2. The view through the scanning field was reflected from an articulated scanning mirror; it was then combined with the fixed field in the beamsplitter and imaged in the focal plane by the same objective lens and erecting system. The operator, by observing the focal plane through the eyepiece and adjusting the scanning fields of view, could superimpose the selected targets in the fixed and scanning fields of view and thus establish the angular separation of the targets. The angular rotation of the scanning mirror was controlled by the two-speed scanning control knobs which provided target optical motions of one degree and five degrees per revolution of the knobs.

The sextant was equipped with two removable eyepieces, one providing normal eye relief, and the other providing long eye relief. The normal-eye-relief eyepiece was used when the sextant could be brought directly to the eye for viewing, while the long-eye-relief eyepiece allowed the sextant to be used with the pressure suit helmet on and the helmet visor down.

Data readout was accomplished by direct reading of a mechanical counter located below the instrument eyepiece. The measured angle between the fixed and scanning lines of sight was indicated on the counter in degrees and thousandths of a degree, the smallest count being 0.001 of a degree or 3.6 arc seconds. A dual-cell rechargeable nickel-cadmium 2.5-volt battery was contained within the sextant and was used for illuminating both the data readout and the reticle.

An event timer button and switch were located on the right side of the instrument. The event timer switch was connected to the spacecraft telemetry recorder through the utility cord. Depression of the event timer button placed a time-correlated signal on the onboard PCM recorder tape for use in the data analysis.

Two filters of different density were provided in each line of sight to reduce the amount of light transmitted through them. The purpose of filters was to permit viewing images of widely varying intensities such as a star and a lunar landmark.

The general characteristics of the sextant are as follows:

Size, in.	7 by 7-1/4 by 6-1/16
Weight, lb	6.25
Magnification	
Normal-eye-relief eyepiece	8X
Long-eye-relief eyepiece	4.6X
Field of view, deg	7
Exit pupil	
Normal-eye-relief eyepiece, mm	4
Long-eye-relief eyepiece, mm	60
Diopter adjustment	-3 to +3
Resolution, arc sec	7

Image . . . . Erect

Range, deg 76

The preflight calibration of the instrument is presented in figure 8.14-3.

8.14.3 Procedures

The T002 experiment was performed during the dark periods of revolutions 40, 48, 54, 55, and 56.

The sextant was taken from its stowed location and the pilot hooked up the timing-event system connector to the spacecraft utility cord. The command pilot installed his reticle on the left-hand window and started his elapsed time clock on an even minute, noting the time; then the spacecraft lights were extinguished. The command pilot established spacecraft orientation with respect to the selected stars using his reticle and stabilized the spacecraft about all three axes within ±2 degrees in pitch and yaw and ±10 degrees in roll, with very low limit cycle rates of less than 0.25 deg/sec.

After the spacecraft was stabilized, the pilot focused the sextant, set the reticle illumination to a comfortable level, and acquired the star Aldebaran in both lines of sight. The pilot then superimposed the two images and marked the time of superposition by depressing the event time button. An oral time "Mark" was called out by the pilot and the command pilot read his elapsed timer, noting the time in the experiment log, along with the measured angle read off the sextant by the pilot. This procedure was repeated five times for the same star to provide an indication of the zero bias of the sextant-operator combination.

The spacecraft was then reoriented and stabilized by the command pilot, and the pilot acquired the prescribed target pair for the sighting period. A procedure similar to that described above was then followed for at least ten consecutive measurements of the angle between the target pair. A selected number of the sextant measurements were transmitted to the ground for real-time evaluation of the pilot's performance in making the prescribed measurements.

8.14.4 Results

The results of the T002 experiment consist of learning-curve data obtained during the initial period of familiarization and training with the sextant, baseline data for comparison with flight results, and in-flight data obtained during the Gemini XII fight.

8.14.4.1 Initial training and familiarization.- The initial training of the pilot was performed in the Docking Simulator at the Manned Spacecraft Center during the period of August 5, 1966, through August 10, 1966. Two simulated star targets were installed in the simulator room. These targets consisted of 12-inch parabolic mirrors which projected toward the sighting station the collimated light of a small source placed at the focal point of the mirror. A second magnitude star was simulated. Using the hand-held sextant in the darkened Docking Simulator, the pilot performed 15 consecutive measurements of the angle between the simulated stars and 10 consecutive measurements of the sextant angle when sighting on one star with both sextant lines of sight. These measurements were repeated in 15 sighting sessions over a period of four days. The standard deviation of the 15 measurements of the angle between the two stars from their mean value was used as a measure of the pilot's proficiency. The standard deviation varied throughout the training period from a maximum value of about 13 arc seconds near the beginning of the training period to a minimum of about 4 arc seconds near the end. These data indicate that the pilot had achieved a high degree of proficiency during the four days of training.

8.14.4.2 Baseline data.- All baseline data were obtained at Ames Research Center, Moffett Field, California, during the period from September 7, 1966, through September 9, 1966. The majority of these data were obtained in the Ames Midcourse Navigation and Guidance Simulator. The basic components of the simulator are a visual scene that simulates a moon-star field and a movable cab which simulates a manned space vehicle. The two simulated stars used in the initial training were employed in obtaining the baseline data. Using the hand-held sextant the pilot made five consecutive measurements of the angle obtained when viewing the same simulated star through both sextant lines of sight to establish an instrument-operator measured zero bias. Subsequently, ten consecutive measurements of the angle between the selected sighting targets were made. Measurements were made with the helmet off (normal-eye-relief eyepiece) and helmet on, visor down (long-eye-relief eyepiece). The standard deviation of the measurements about their mean value and the mean measurement bias error for all sessions are summarized in table 8.14-I. Additional baseline data obtained using real-world targets are summarized in table 8.14-II.

The standard deviation of the measurements obtained in both the simulator and using actual stars are substantially below the ±10 arc second level and agree well for both helmet configurations. The mean measurement bias errors are also small except for the helmet-on, visor-down configuration.

8.14.4.3 Inflight data.- The Gemini XII pilot's inflight sextant measurement data were obtained on November 14 and 15, 1966. The measurements were made from the stabilized spacecraft, with the pilot looking through the right-hand window. A summary of the standard deviation of the measurements and the measurement bias error is presented in table 8.14-III.

The standard deviation of the measurements for all sighting conditions is below ±10 arc seconds, agreeing well with the baseline data. The measurement bias errors of the inflight data are generally small except for the helmet-on, visor-down configuration. This is in agreement with baseline data. It should be noted that the measurement bias error presented here is preliminary in nature and should be used with caution since it is uncorrected for window-induced measurement errors, for errors caused by the difference in index of refraction of the light transmitting media within and outside the spacecraft, and for measured zero bias. A more detailed analysis of the data, including these factors, will be published at a later date.

8.14.5 Conclusions

The standard deviation of the inflight measurements was ±9.0 arc seconds or below, indicating that the hand-held sextant may be suitable for making navigation measurements during the midcourse phase of lunar or interplanetary space flight.

The hand-held sextant with a long-eye-relief eyepiece can be used to make accurate navigation measurements ($1\sigma<10$ arc seconds) with the pressure suit helmet on and visor down.

The pilot's performance as indicated by the baseline data obtained both in a simulator and using real stars from earth-based observatories was virtually the same as that obtained in the space-flight environment, thus validating the usefulness of simulators and earth-based observatories in evaluating space navigation measurement techniques.

No operational difficulties were encountered which were associated with the space environment.

TABLE 8.14-I.- PILOT BASELINE DATA FROM AMES SIMULATOR

Sighting targets	Helmet configuration	No. of sightings sessions	No. of measurements per session	Standard deviation of measurements, arc seconds	Mean measurement bias error, arc seconds[a]
Star/Star	Helmet off	7	10	±5.2	+1.8
Star/Star	Helmet on Visor down	3	10	±7.4	-1.6

[a]Measurement bias error = Mean measured target angle[b] minus calibrated target angle.

[b]Mean measured target angle = Mean measured angle corrected for instrument calibration and measured zero bias.

TABLE 8.14-II.- PILOT BASELINE DATA - REAL TARGET

Sighting targets	Helmet configuration	No. of sighting sessions	No. of measurements per session	Standard deviation of measurements, arc seconds	Mean measurement bias error, arc seconds[a]
Altair/Vega	Helmet off	1	10	±4.7	0.0
Altair/δ Cyg.	Helmet off	1	10	±3.4	+1.2
Altair/δ Cyg.	Helmet on Visor down	1	10	±8.2	-9.8

[a]Mean measurement bias error = Mean measured target angle[b] minus computed target angle[c].

[b]Mean measured target angle = Mean measured angle corrected for instrument calibration and measured zero bias.

[c]Computed target angle = The angle between the targets corrected for annual aberration and atmospheric refraction.

TABLE 8.14-III.- PILOT INFLIGHT DATA

Sighting targets	Pressure suit configuration	Revolutions	Ground elapsed time (approx.), hr:min	Time, sec	No. of measurements	Standard deviation of measurements, arc sec	Measurement bias error, arc sec[a]
Betelgeuse/Rigel	Helmet off Gloves off	40	63:15 to 63:47	32	13	±9.0	+5.8
Betelgeuse/Rigel	Helmet off Gloves off	48	75:17 to 75:50	33	14	±5.4	+1.4
Betelgeuse/ Bellatrix	Helmet off Gloves off	54	85:45 to 86:20	35	15	±7.6	-1.1
Betelgeuse/ Bellatrix	Helmet off Gloves off	55	87:15 to 87:48	33	15	±4.5	-1.8
Betelgeuse/Rigel	Helmet on Visor down Gloves off	56	88:45 to 89:20	35	15	±7.5	+9.7

[a]Measurement bias error = Mean measured target angle minus computed target angle[b].

[b]Computed target angle = The angle between the targets corrected for annual aberration.

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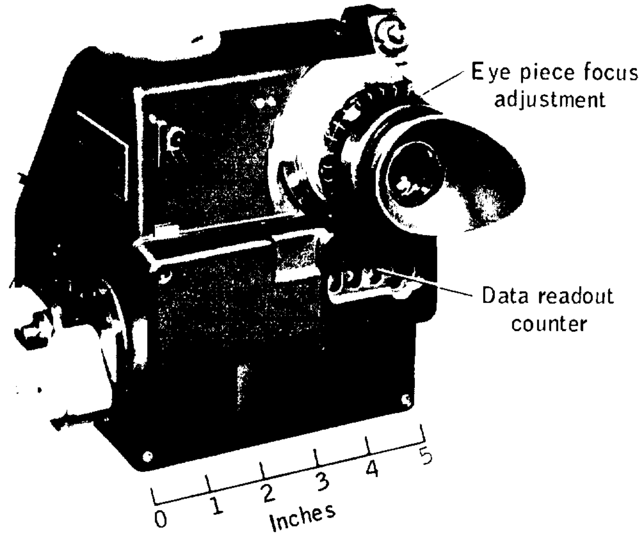

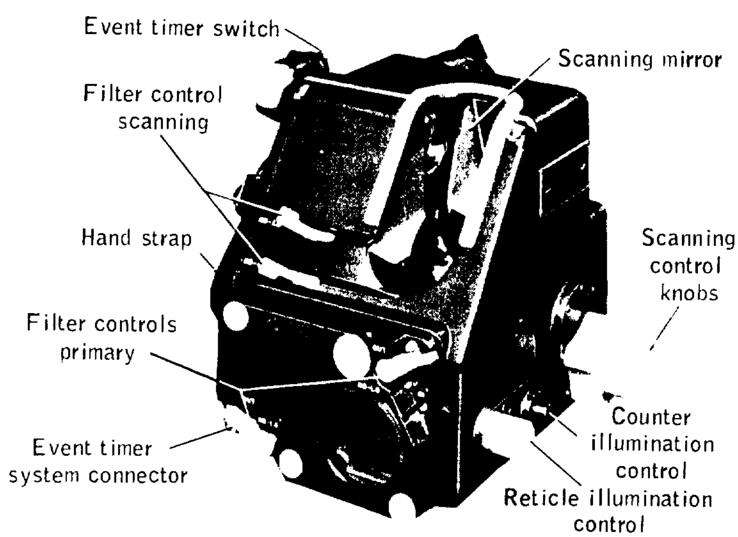

Figure 8.14-1. - Experiment T002, sextant configuration and operating controls.

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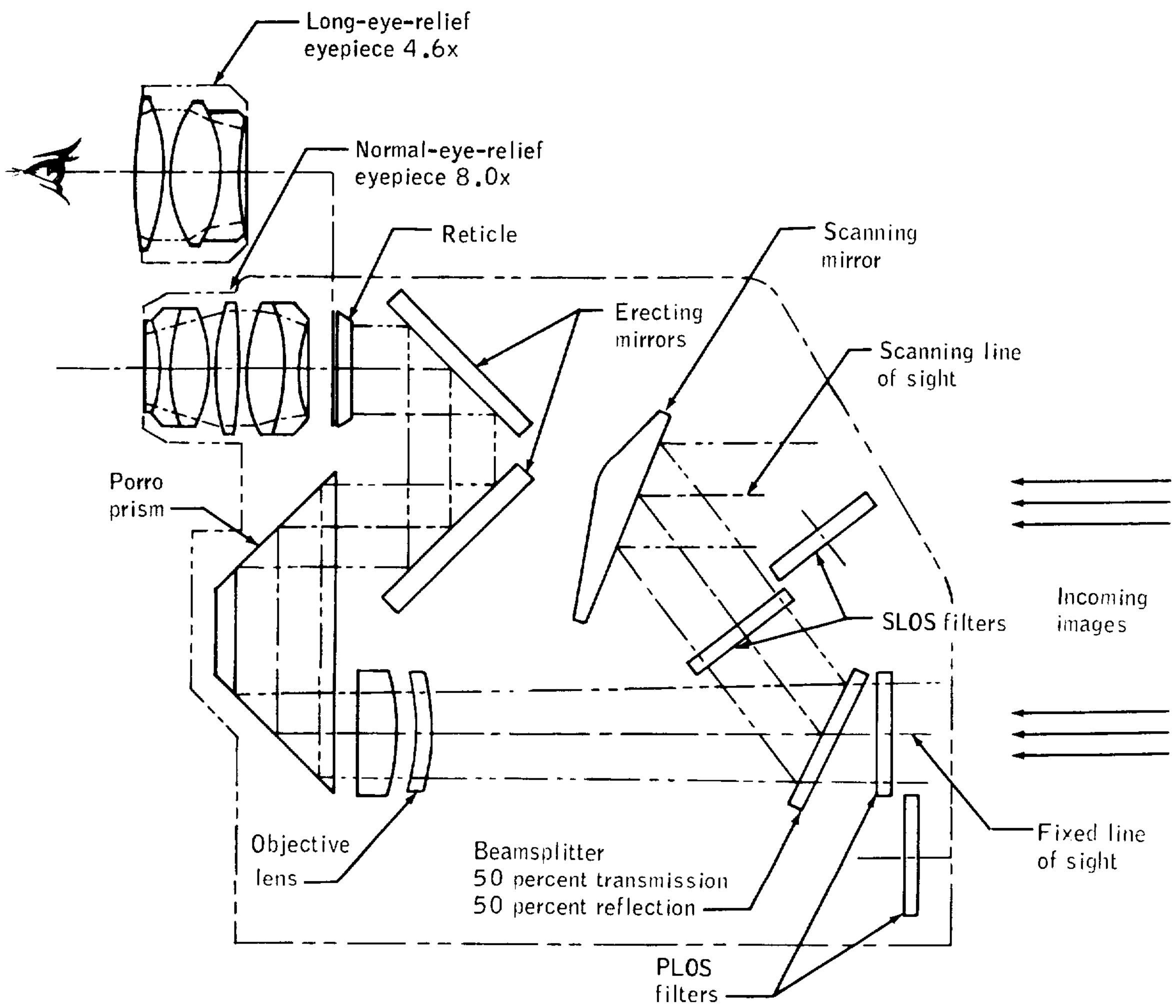

Figure 8.14-2.- Optical schematic of T002 space sextant.

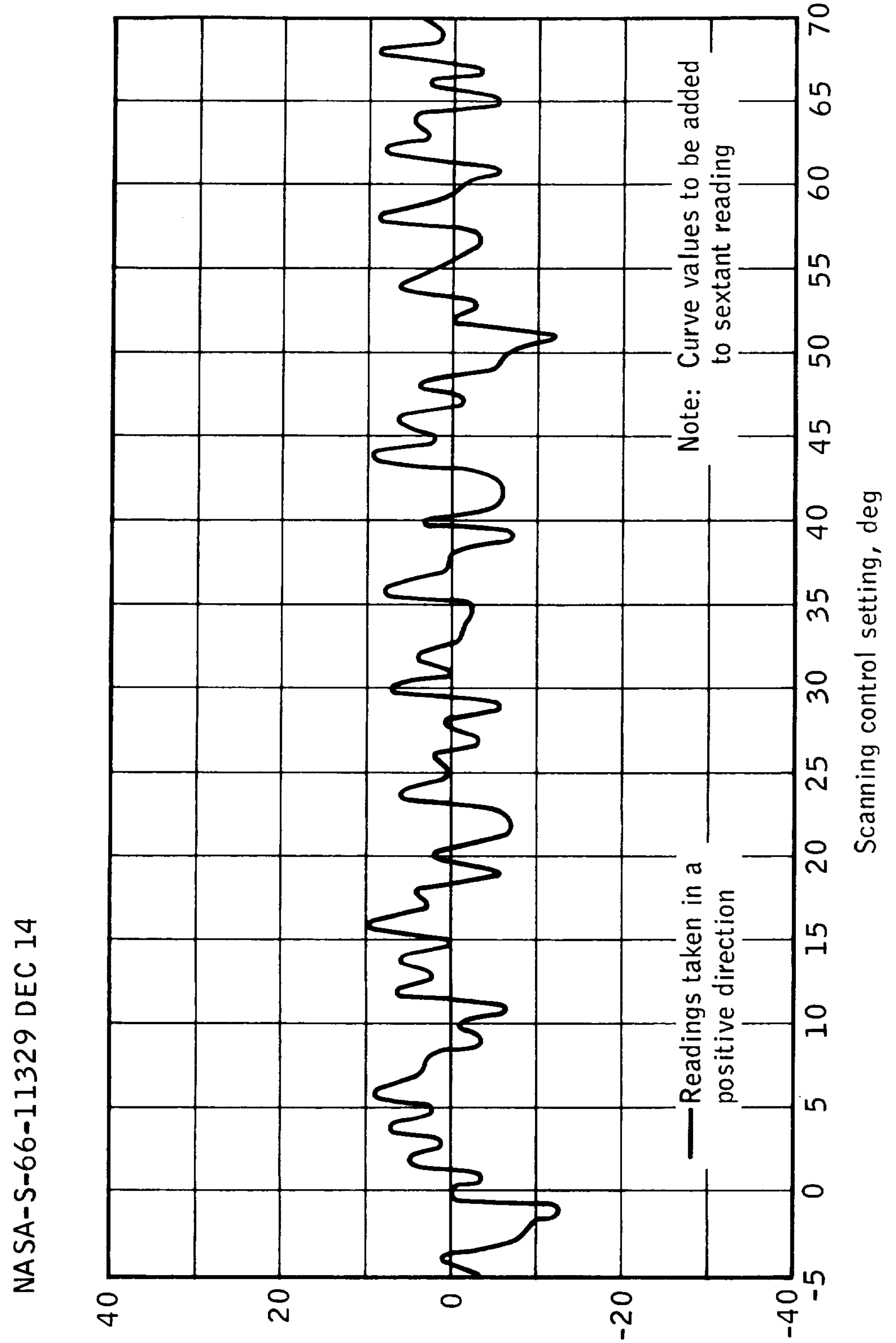

Figure 8.14-3. - Preflight T002 sextant calibration.

8.15 OBJECTS OF OPPORTUNITY - ULTRAVIOLET PHOTOGRAPHY OF UPPER ATMOSPHERIC DUST CLOUDS

8.15.1 Objective

The objective of the Objects of Opportunity experiment was to photograph the earth's upper atmosphere in the ultraviolet wavelength regions of 1000 to 2500Å, as a means to detect cosmic dust particles and to measure their relative regional concentration.

Recent theoretical and experimental programs have yielded information indicating the high possibility for the existence of dust clouds in the upper atmosphere. Rocket experiments show concentrations on the order of 5×10^{10} particles per square meter for particles 0.05 of a micron and larger in size. Computations of the dynamic response of small particles entering the earth's upper atmosphere with cosmic velocities show that the "effective braking layer" for these particles extends from 70 to about 110 kilometers. It would be expected that a sharp increase in the concentration of cosmic dust at these altitudes would be observed when viewed from above.

The absorption characteristics of the upper atmosphere change radically with altitude in the 1000 to 3000Å range. A dust-free atmosphere should look black in this range of wavelengths when viewed from above. Because dust particles approximately 0.1 of a micron in size scatter strongly in the ultraviolet range, regions of high concentrations should register bright patches of light on an ultraviolet photograph superposed on a dark background.

This experiment is expected to yield data pertaining to the following questions:

(a) What is the number of dust layers in the upper atmosphere, and is there a dust layer in the 140-kilometer region?

(b) Do noctilucent clouds exist only at high latitudes in the summer months and during twilight conditions?

(c) Do noctilucent clouds appear as often in the southern as in the northern hemisphere?

(d) Are the high-altitude dust particles concentrated in patches or are they continuously distributed?

8.15.2 Equipment

Photographs of the atmosphere will be taken with a 70-mm general-purpose camera, equipped with an ultraviolet quartz lens and a filter allowing only the 2000 to 2500Å scattered radiation to actuate the film.

The equipment used will be the same as that used for Experiment S013, with the exception of the film and a simple ultraviolet filter in place of a prism or grating.

8.15.3 Procedures

During the second standup EVA at approximately 67 hours g.e.t., the pilot took 42 ultraviolet photographs of star fields and sunrise. The crew indicated that all sequences were performed as planned.

8.15.4 Results

Preliminary evaluation of the photographic data shows that the carbon-dioxide cartridge within the ultraviolet film magazine did not eliminate the static electricity caused by film movement. All exposures show intense fogging by this internal radiation and it is extremely doubtful that useful information can be extracted from any photographs.

9.0 CONCLUSIONS

The two launch vehicles, the spacecraft, the flight crew, the Gemini Agena Target Vehicle, and the mission support were satisfactory for all phases of the mission. This flight, the last of the Gemini series, contributed significantly to the knowledge of manned space flight, particularly in the areas of extravehicular activity, tethered vehicle operations, rendezvous, and controlled reentry.

1. Approximately five and one-half hours of extravehicular activity were successfully conducted during the Gemini XII mission. The tasks completed during these periods demonstrated the feasibility of conducting a variety of extravehicular tasks without encountering unsatisfactory metabolic heat loads. Significant factors involved in this demonstration were the use of adequate restraints, provision of adequate training, and careful control of work levels and rates.

2. Scheduling simple extravehicular tasks first, such as in a standup extravehicular activity (EVA), permits the flight crew to become acclimated to the extravehicular environment and enables them to subsequently accomplish more difficult tasks. This sequence of events is highly desirable for crews who have not previously performed EVA.

3. Foot restraints of the type used for Gemini XII will provide good body positioning control for EVA tasks within a limited area. Waist tethers like those used for Gemini XII will provide good body position control for EVA tasks for which fixed foot restraints are not practical. Restraints of these types eliminate the effort and concern for maintaining contact with the work task. Plug-in pip-pins of the type used in Gemini XII are useful as EVA handholds or tether attachment points provided they are restrained from rotating.

4. Underwater simulations provide a high-fidelity duplication of the actual EVA environment. Good correlation was shown between the character and difficulty of EVA tasks in orbit and in underwater simulations. The use of underwater simulations for equipment validation, procedures validation, timeline determination, and crew training contributed significantly to the success of the umbilical EVA on this mission.

5. The effort required to overcome the mobility forces of the Gemini extravehicular space suit was reported to be a significant factor in the total EVA workload. This applied primarily to the hands, the arms, and the shoulders, and not to the waist, lower torso, and legs. The glove mobility forces induced hand fatigue after significant periods of use.

6. Standard bolt heads are adequate in height for use with a ratchet wrench during EVA. The type of internal-wrenching bolt head which holds the wrench in place is desirable but not required if adequate body restraints are provided. In the zero-g environment associated with EVA, the low frictional forces of bolts make them susceptible to free rotation in the mating holes. During the Gemini XII mission this condition was aggravated by the high friction in the ratchet wrench used. Removal and replacement of bolts is an undesirable task for repetitive EVA applications. Twenty-five pounds appears to be a reasonable force for an extravehicular pilot to apply with one hand when using foot restraints. Twenty pounds appears reasonable for one hand when using waist tethers only.

7. Portable handholds using Velcro for attachment tended to peel off in the direction of the short dimension (three inches). There was no tendency for the handholds to peel off in the direction of the long dimension (eight inches).

8. During the Gemini XII EVA, several types of electrical connectors were feasible for one-handed disconnect/connect operations; however, it may be concluded that a connector specifically designed for EVA applications is easier to use.

9. Thrust from the Extravehicular Life Support System chestpack (outflow gas pressure forces) or any other extraneous forces acting on the extravehicular pilot were found to be negligible.

10. From the heart-rate data collected during programmed exercise periods, there appears to be no significant physiological difference in exercise performed before and during the flight.

11. The tether evaluation provided evidence that gravity gradient stabilization could be accomplished between two vehicles connected by a 100-foot flexible tether. The following conclusions were reached:

(a) It was possible to maneuver with the precision necessary to gain a gravity-gradient capture despite a spacecraft attitude control problem caused by degraded thrusters.

(b) Even with small rates and oscillations involved, the tether became taut fairly rapidly and remained taut without further oscillations.

(c) Although fairly large spacecraft oscillations occurred, the crew was confident that the tethered vehicle system was stabilized.

12. The Gemini XII rendezvous demonstrated the effectiveness of onboard backup procedures in the presence of a radar failure. The consistent success of Gemini rendezvous operations conducted under a wide

variety of conditions should provide significant data to future programs which involve operations of this type.

13. Onboard determination of plane-change and coelliptic maneuvers is feasible when the spacecraft has a near-nominal trajectory.

14. The radar problems experienced during the Gemini XI and Gemini XII missions were both the result of a failure within the radar transponder located in the Target Docking Adapter (TDA). The failure may not have been the same in both cases, but appeared to have resulted from the same basic cause. This failure probably resulted from arcing induced at very low pressures and may have occurred in a pressurized component which lost pressure, or in a component which was not properly protected against the low pressure condition.

15. The fuel-cell power system successfully supplied all necessary electrical power to the spacecraft despite the failure of two of the six stacks. Accumulation of product water in the fuel cells appears to have led to these failures.

16. A progressive loss of thrusters in the Orbital Attitude and Maneuver System (OAMS) made spacecraft control extremely difficult but did not prevent completion of the mission objectives. This problem has appeared to some degree on most of the Gemini missions, and the degradation has varied from slight to almost total loss of thrust. The number of attitude thrusters affected varied from flight to flight and there appeared to be no consistency as to which ones were affected.

17. Spacecraft window contamination was less severe on this mission than on any previous mission. This reduction was accomplished by curing or conditioning all nonmetallic components to prevent excessive outgassing in the immediate area of the windows and by employing the previously used disposable covers during the launch phase. Since there still was a slight contamination, it appears that, to a much lesser degree, materials were deposited on the windows during the orbital and reentry phases of the flight.

18. The feasibility of a controlled reentry to a selected landing site was successfully demonstrated on the Gemini XII mission. This was the seventh consecutive mission wherein the spacecraft was landed very near the selected landing point. The success of these controlled reentries has provided the necessary confidence for future space programs to use this technique.

19. Increased attention is required to ensure adequate coordination between the experimenters and experiments personnel and the flight crew and operations personnel in developing onboard experiment procedures

which take into consideration operational considerations as well as the limitations of the spacecraft.

20. Although many excellent photographs were taken for the various photographic experiments, the 70-mm general-purpose still cameras and the 16-mm sequence cameras again malfunctioned on a number of occasions, which has been the experience on several previous flights.

10.0 RECOMMENDATIONS

The following recommendations are made as a result of engineering analyses and crew observations of the Gemini XII mission:

1. Future flight plans including extravehicular activity (EVA) should be scheduled to provide an initial period of simple tasks to allow the flight crew to become acclimated to the extravehicular environment. Mission-critical tasks should not be scheduled until after a reasonable period of acclimatization.

2. Future EVA missions should be planned in such a way as to assure that the workloads are not excessive. The EVA flight plans should include a specific allowance for rest periods and contingencies.

3. Underwater simulation should be used, where applicable, for validation of equipment and procedures and for crew training for future EVA missions and tasks. This simulation should be complemented by ground training and selected zero-gravity aircraft simulations. Tasks for which an underwater simulation is valid should not be attempted in EVA unless they can be performed readily in underwater simulation. Tasks for which an underwater simulation is not feasible should be analyzed carefully to ensure that the simulation technique used is valid.

4. Use of waist tethers as body restraints should be considered for future EVA missions involving work tasks on the external surface of a vehicle.

5. Foot restraints of the type used in Gemini XII should be considered for use in future EVA missions having high activity in a localized area. Consideration should also be given to use of this type of foot restraint inside vehicles or modules with significant interior volume.

6. Handholds, handrails, or similar devices should be provided for all EVA transit over the exterior surface of a spacecraft.

7. High priority efforts should be devoted to minimizing the spacesuit mobility forces in the gloves, arms, and shoulders for future extravehicular space suits.

8. The use of standard threaded bolts should be avoided for future EVA tasks, particularly those tasks which are repetitive. Quick release fasteners should be used in preference to standard bolts.

9. Tasks requiring force applications greater than 20 to 25 pounds should be avoided in future EVA missions when using waist tethers as the principle restraint.

10. During underwater simulation, extravehicular crewman should attempt to train in the art of conserving energy and applying minimum effort for each task.

11. Experiment and operations personnel should jointly participate in planning inflight procedures for all experiments, taking into account operational considerations and spacecraft limitations.

12. Continued emphasis should be placed on the method for rapid isolation and diagnosis of inflight system failures in order to effect proper corrective action and minimize the impact on mission performance and mission objectives.

13. A thorough investigation of the thruster problems which occurred throughout the Gemini flights should be conducted to determine the cause, so that action may be taken to eliminate this problem on future programs.

14. Design reviews should be held on the 70-mm general-purpose still camera and the 16-mm sequence camera, and the design weaknesses corrected before further use in space flight.

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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 problem areas.

12.1.2 Gemini Launch Vehicle Histories

The Gemini Launch Vehicle (GLV) history and significant manufacturing activities at the contractor's facilities in Denver, Colorado, and in Baltimore, Maryland, are presented in figure 12.1-5. The GLV history at Cape Kennedy is presented in figure 12.1-6. This figure also includes the problems which were concurrent with the normal GLV launch-preparation activities.

12.1.3 Gemini Agena Target Vehicle and Target Docking Adapter Histories

The Gemini Agena Target Vehicle (GATV) history, prior to final shipment to Cape Kennedy, Florida, is shown in figure 12.1-7. The history of the GATV and Target Docking Adapter (TDA) and the significant events that occurred after delivery to Cape Kennedy are shown in figure 12.1-8.

12.1.4 Target Launch Vehicle History

The Target Launch Vehicle (TLV) history at the contractor's facility in San Diego, California, is shown in figure 12.1-9. Figure 12.1-10 includes significant problems that were encountered during testing at Cape Kennedy.

12.1.5 Extravehicular Life Support System History

Figure 12.1-11 is a history of the Extravehicular Life Support System (ELSS). The figure also identifies the significant problems that occurred during testing at Cape Kennedy.

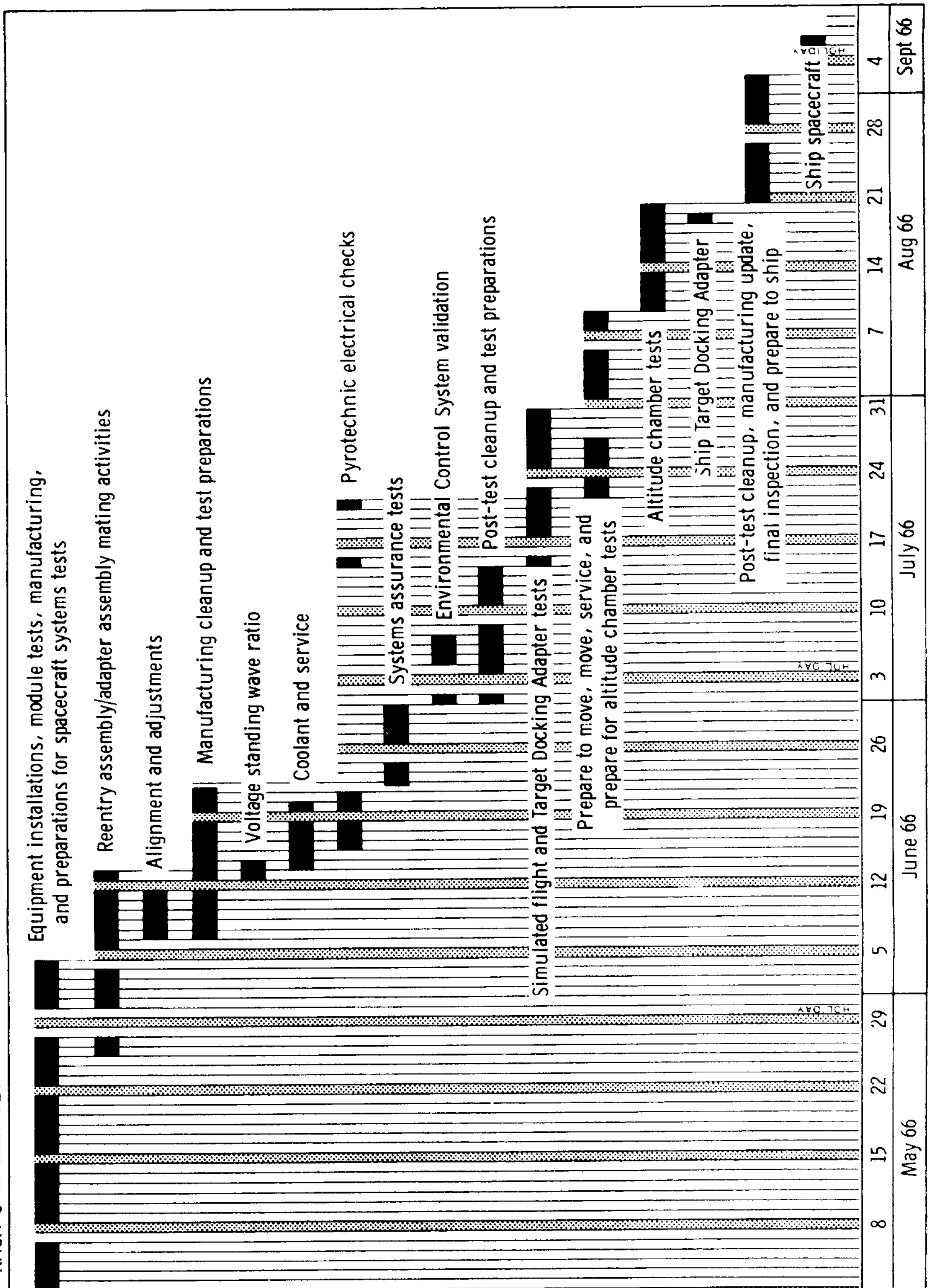

Figure 12.1-1. - Spacecraft 12 test history at contractor facility.

UNCLASSIFIED

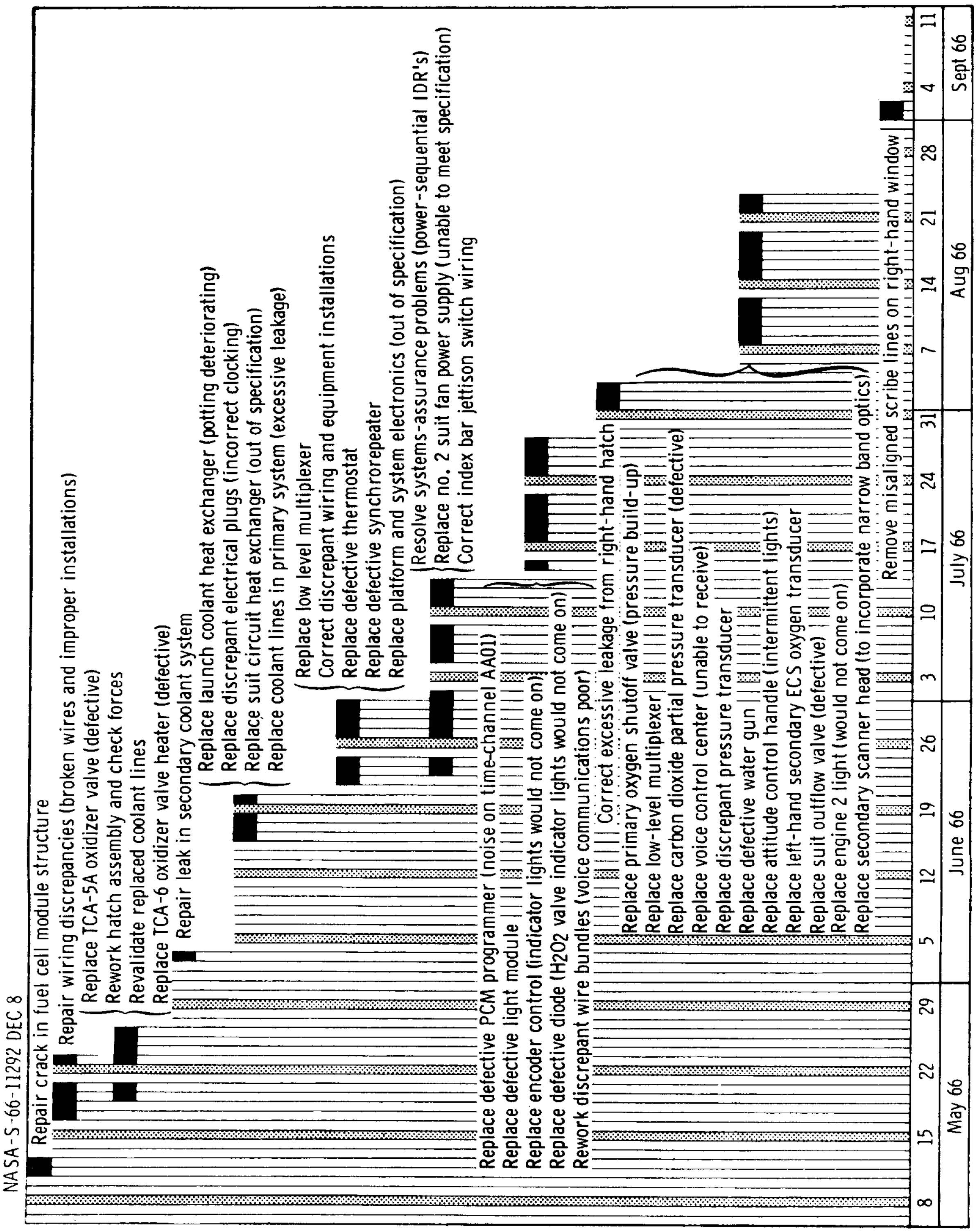

Figure 12.1-2. - Spacecraft 12 significant problems at contractor facility.

UNCLASSIFIED

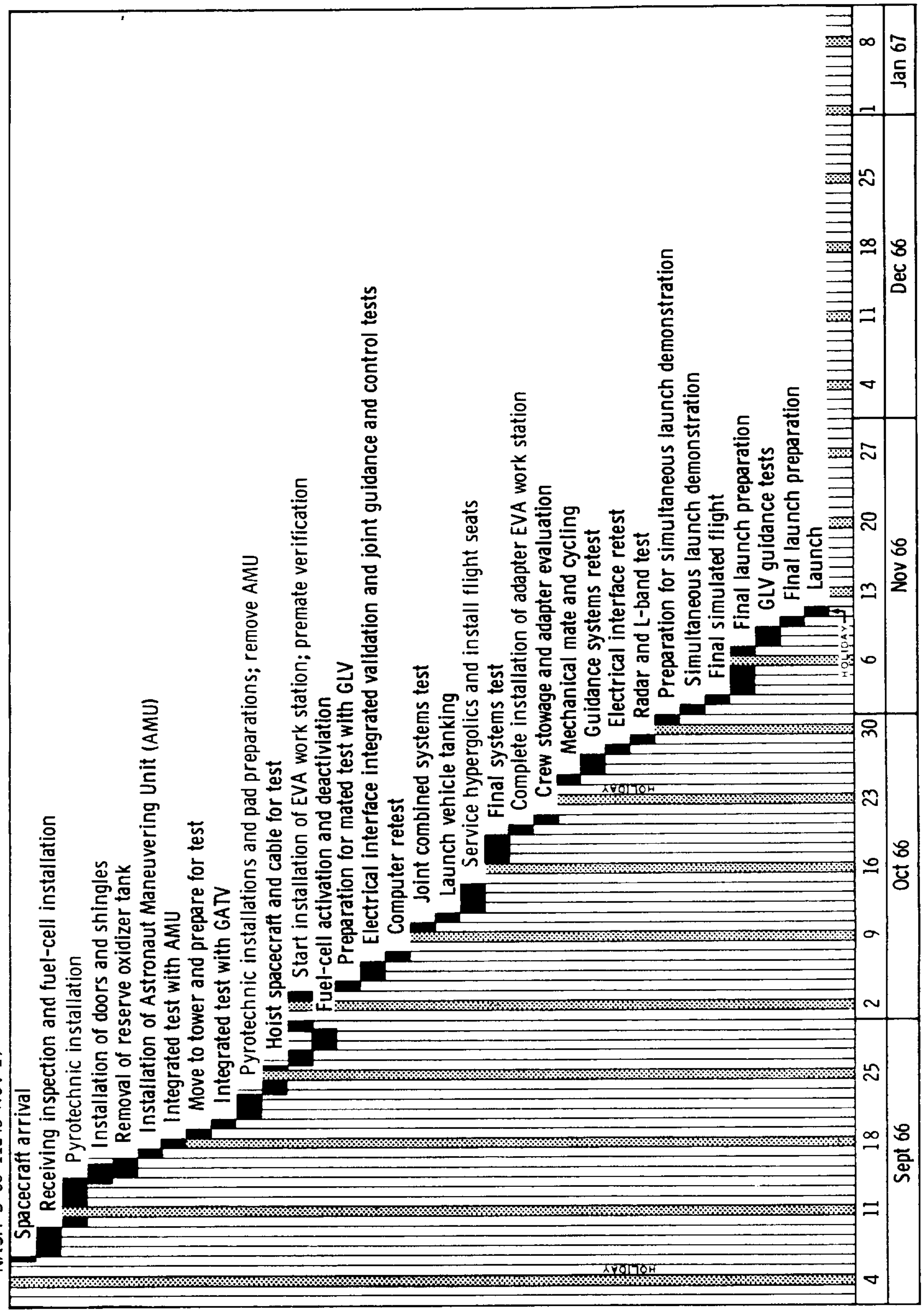

Figure 12.1-3. - Spacecraft 12 history at Cape Kennedy

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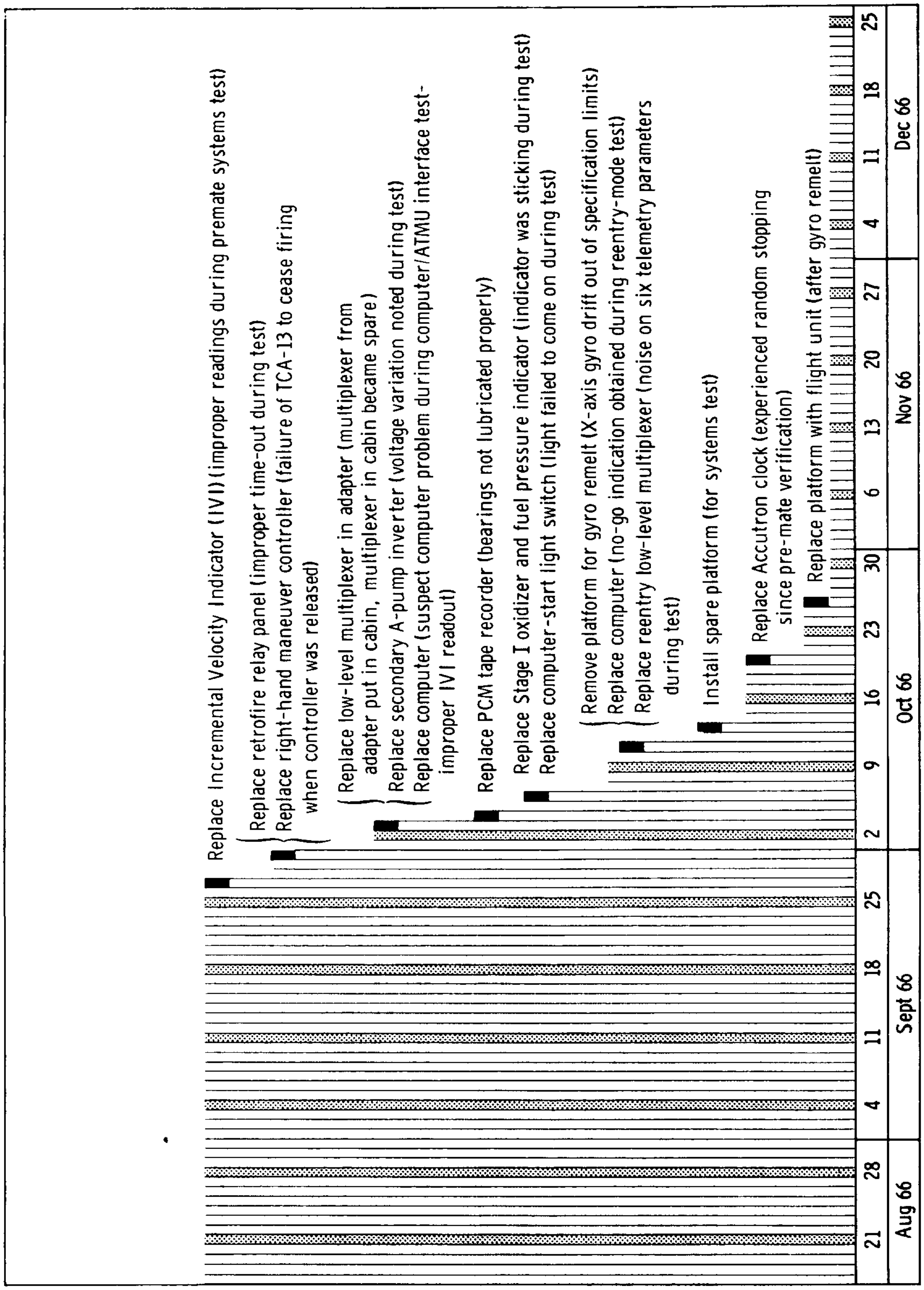

Figure 12.1-4. - Spacecraft 12 significant problems at Cape Kennedy.

NASA-S-66-11250 NOV 29

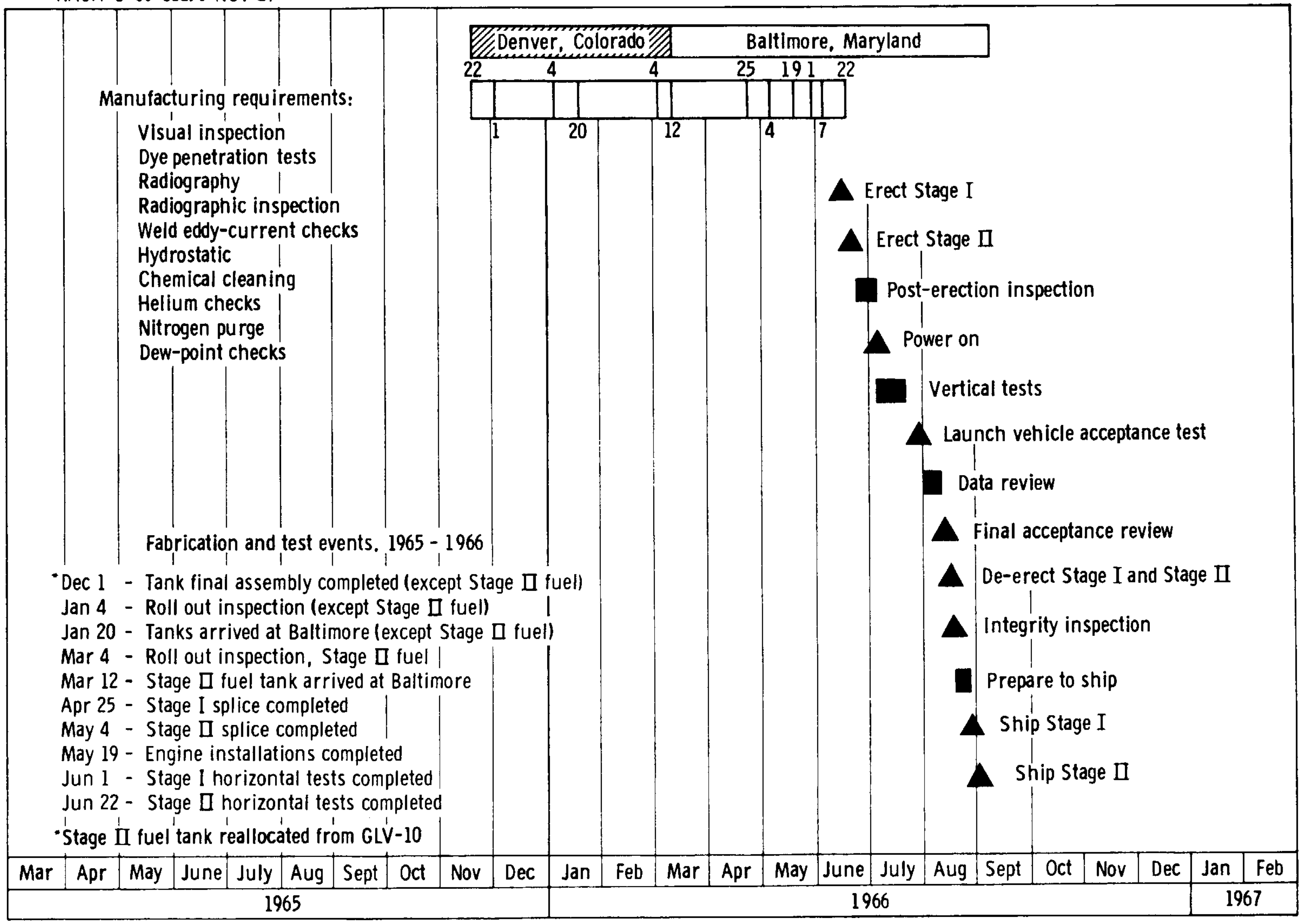

Figure 12. 1-5. - GLV-12 history at Denver and Baltimore.

UNCLASSIFIED

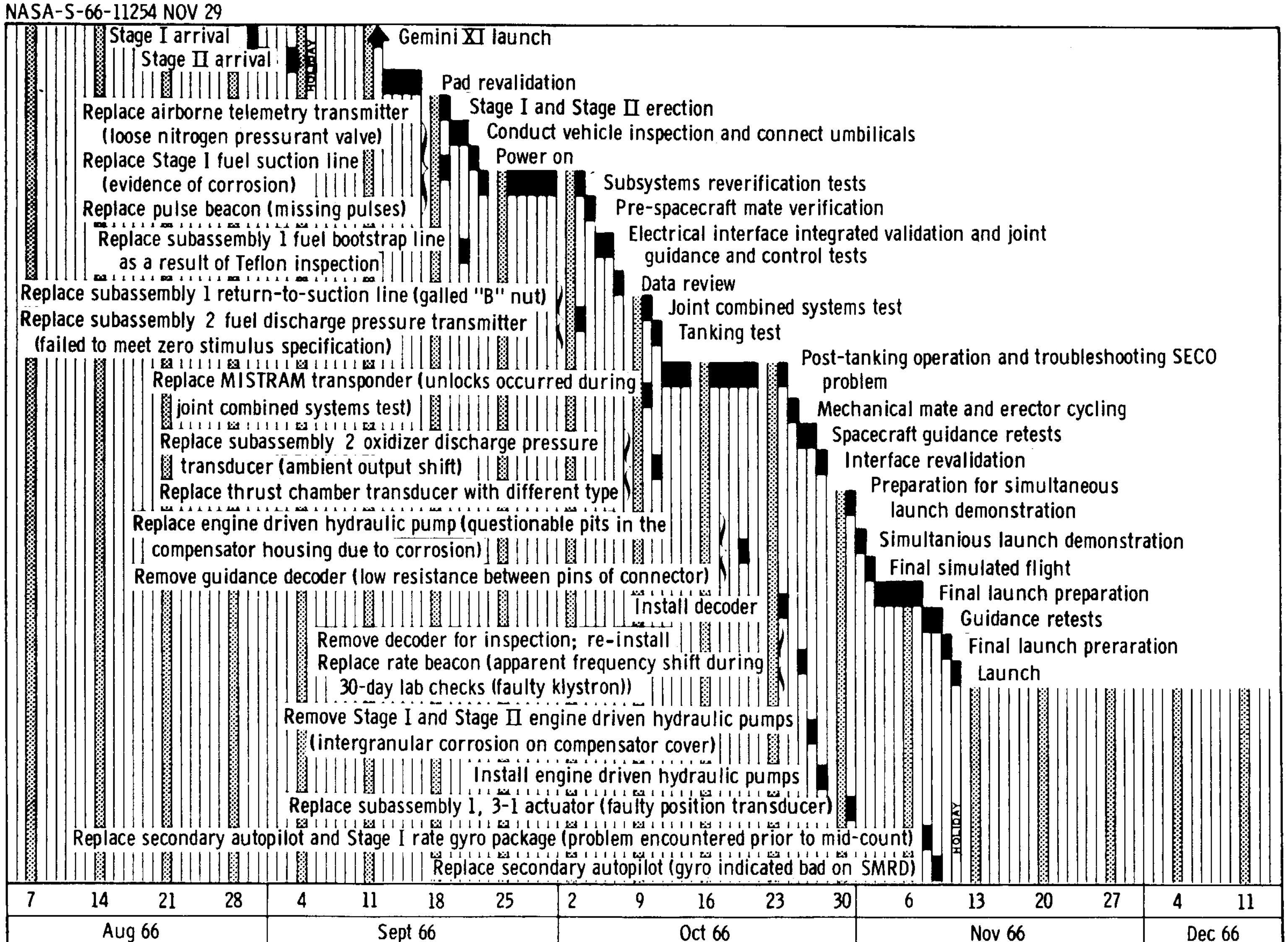

Figure 12.1-6. - GLV-12 history at Cape Kennedy.

UNCLASSIFIED

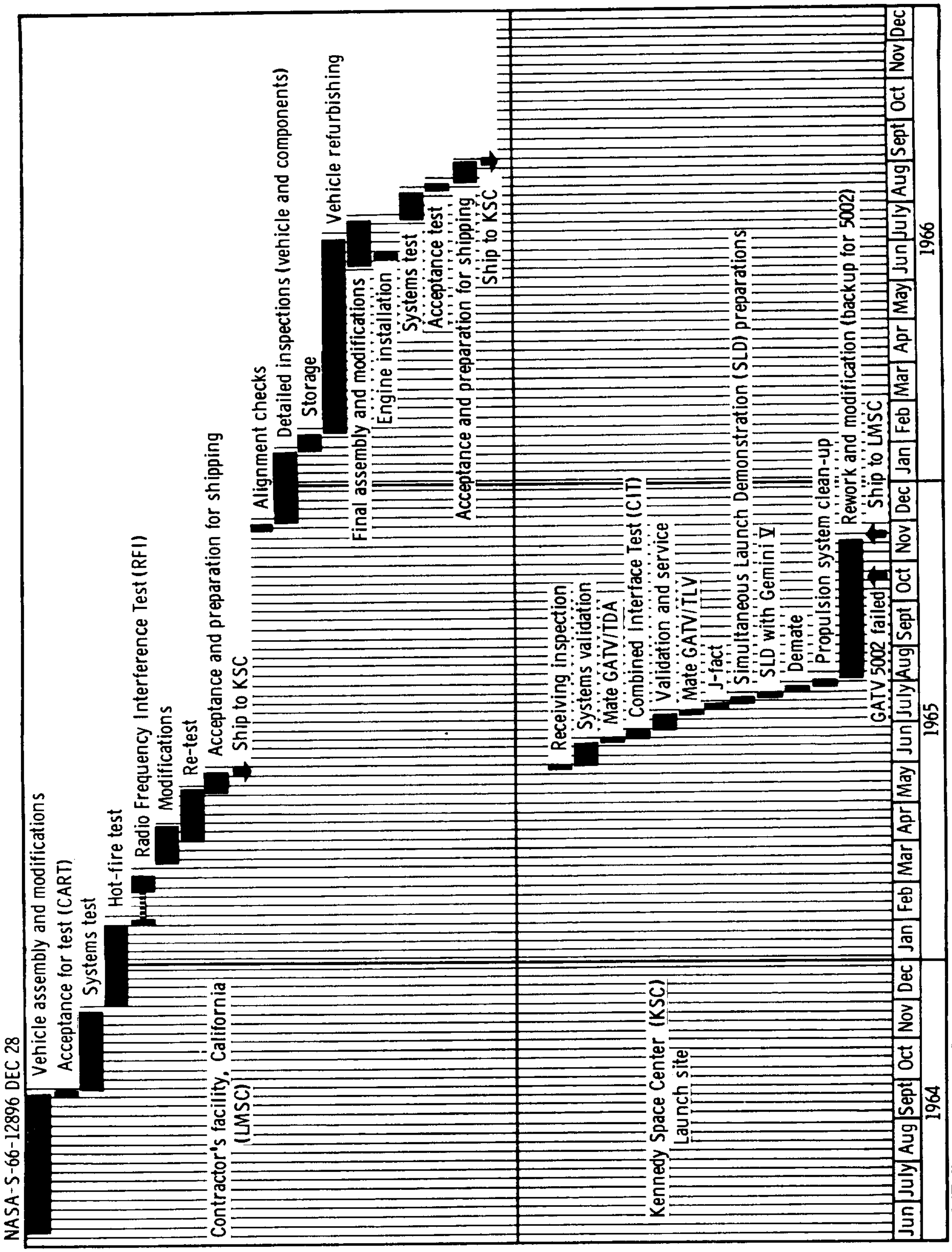

Figure 12.1-7. - GATV 5001 history.

NASA-S-66-11253 NOV 29

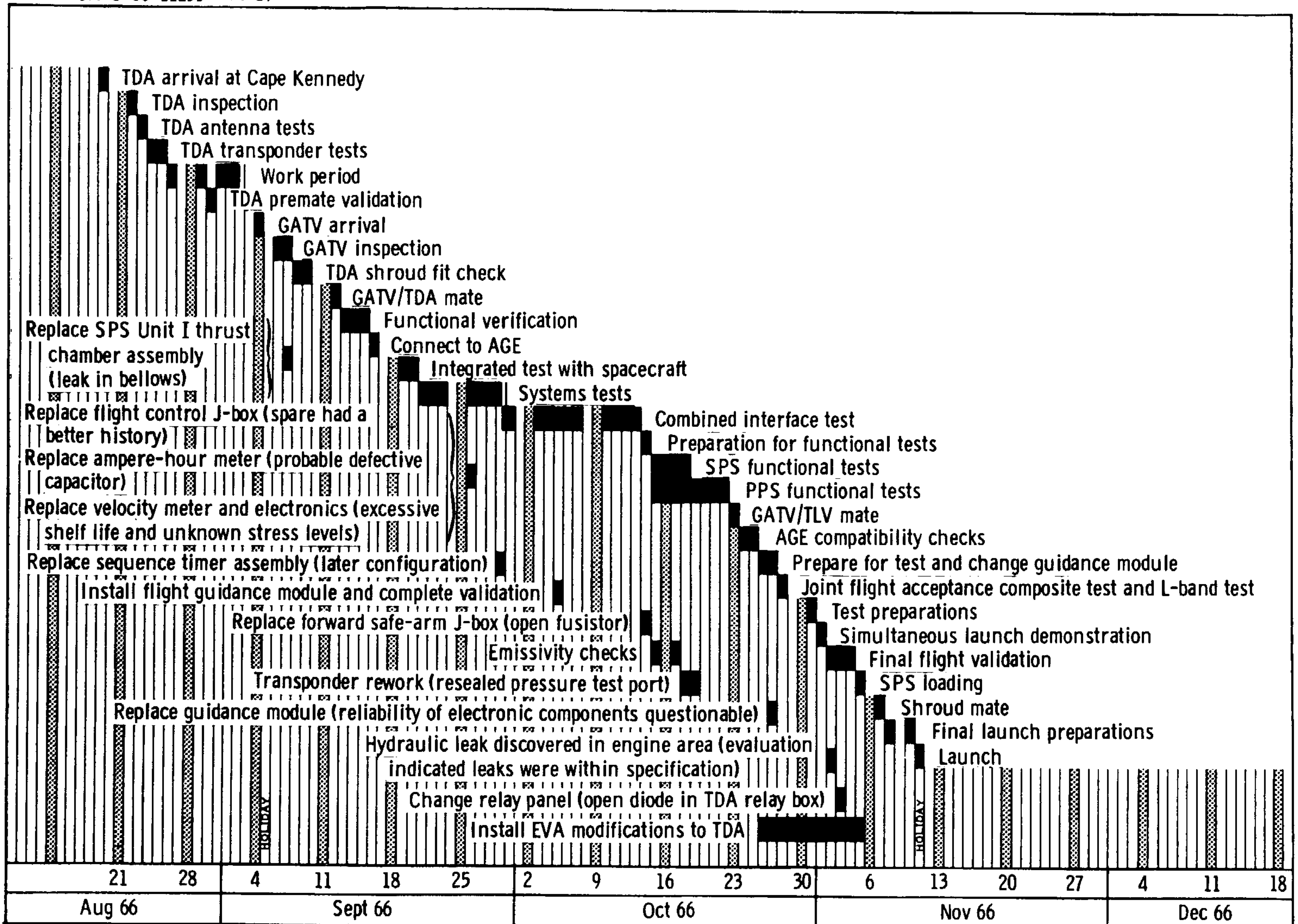

Figure 12.1-8. - GATV 5001 and TDA-7A test history and significant events at Cape Kennedy.

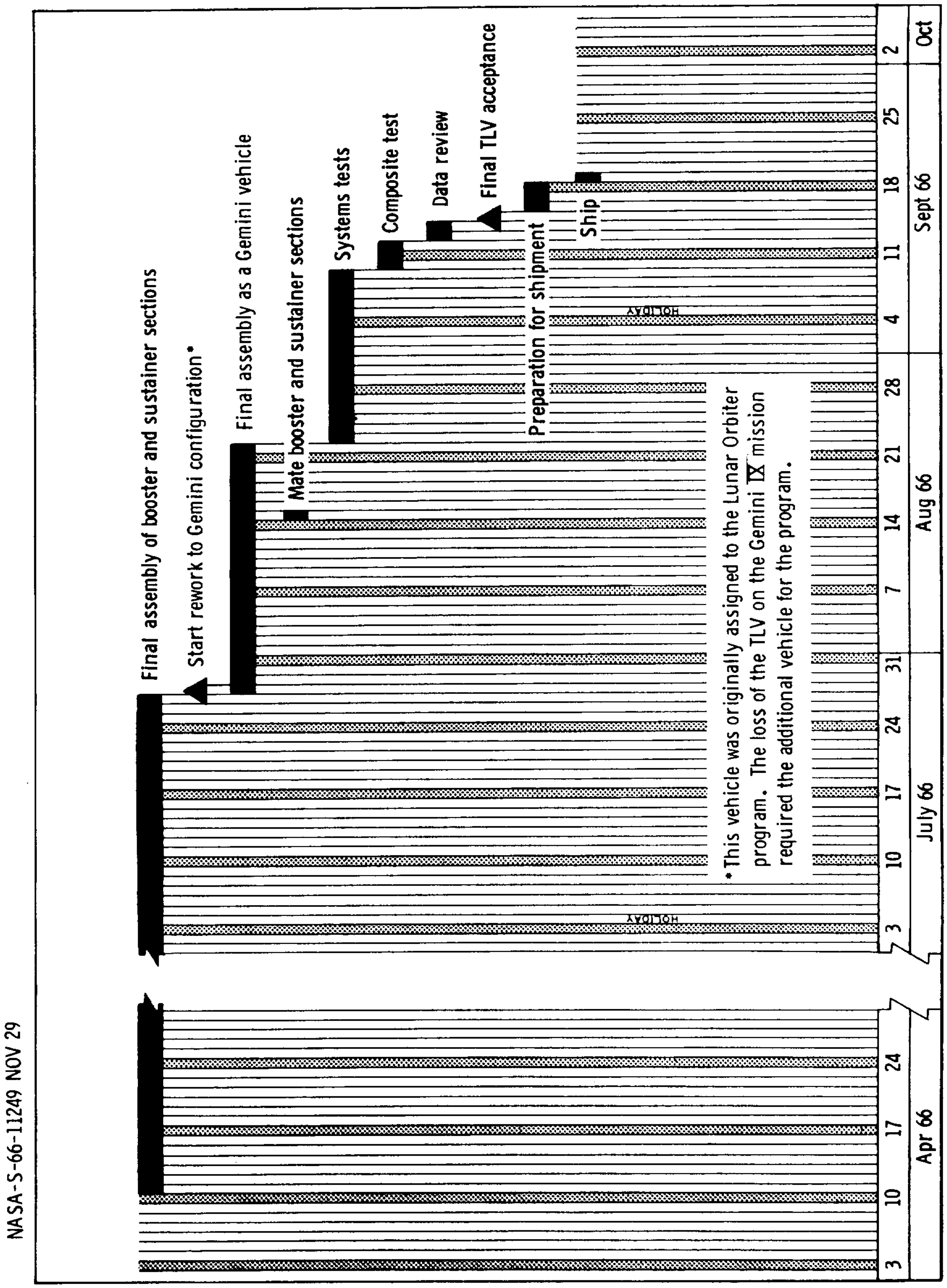

Figure 12.1-9. - TLV-5307 history at contractor facility.

UNCLASSIFIED

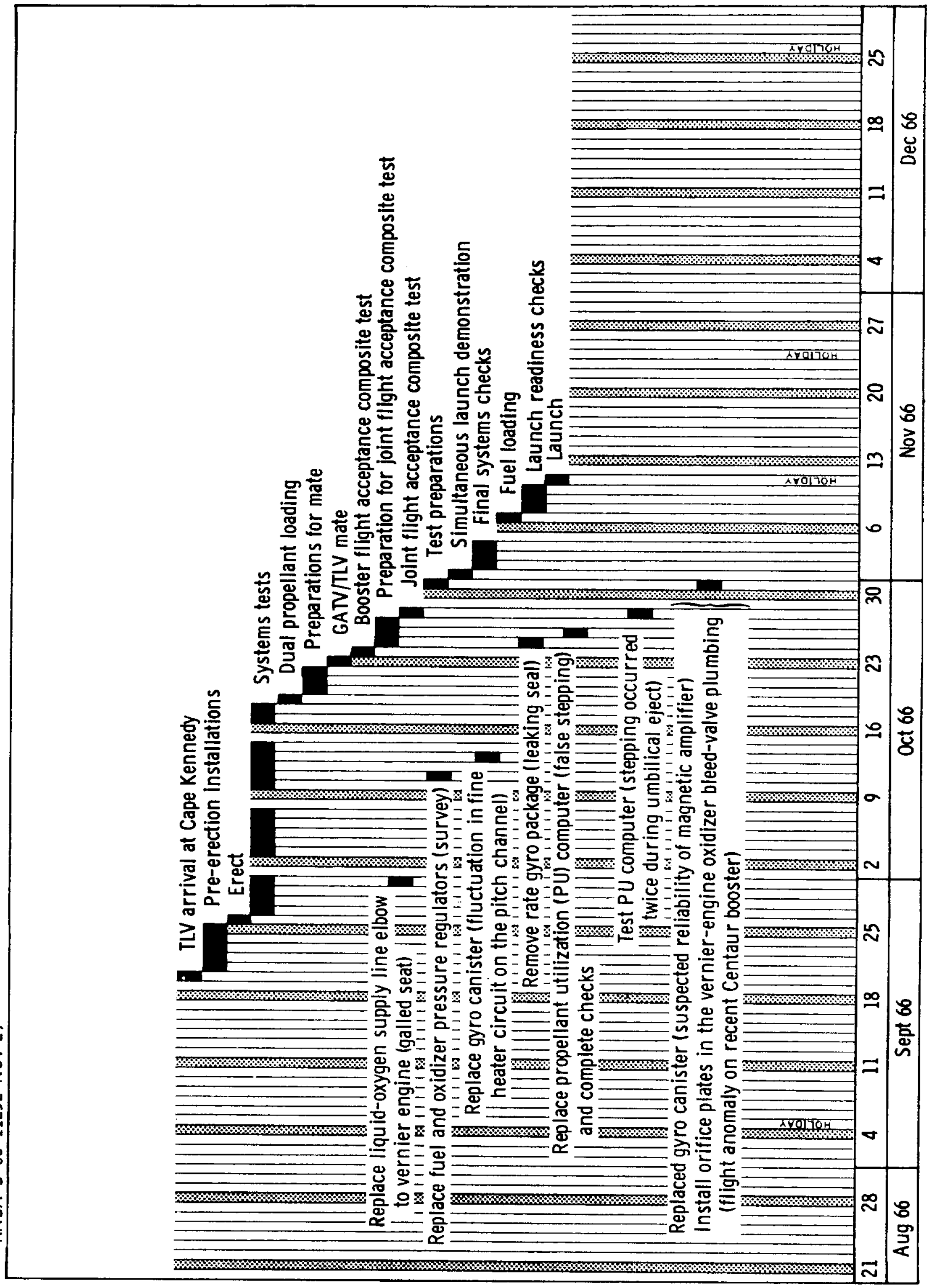

Figure 12.1-10. - TLV-5307 history and significant events at Cape Kennedy.

NASA-S-66-11246 NOV 29

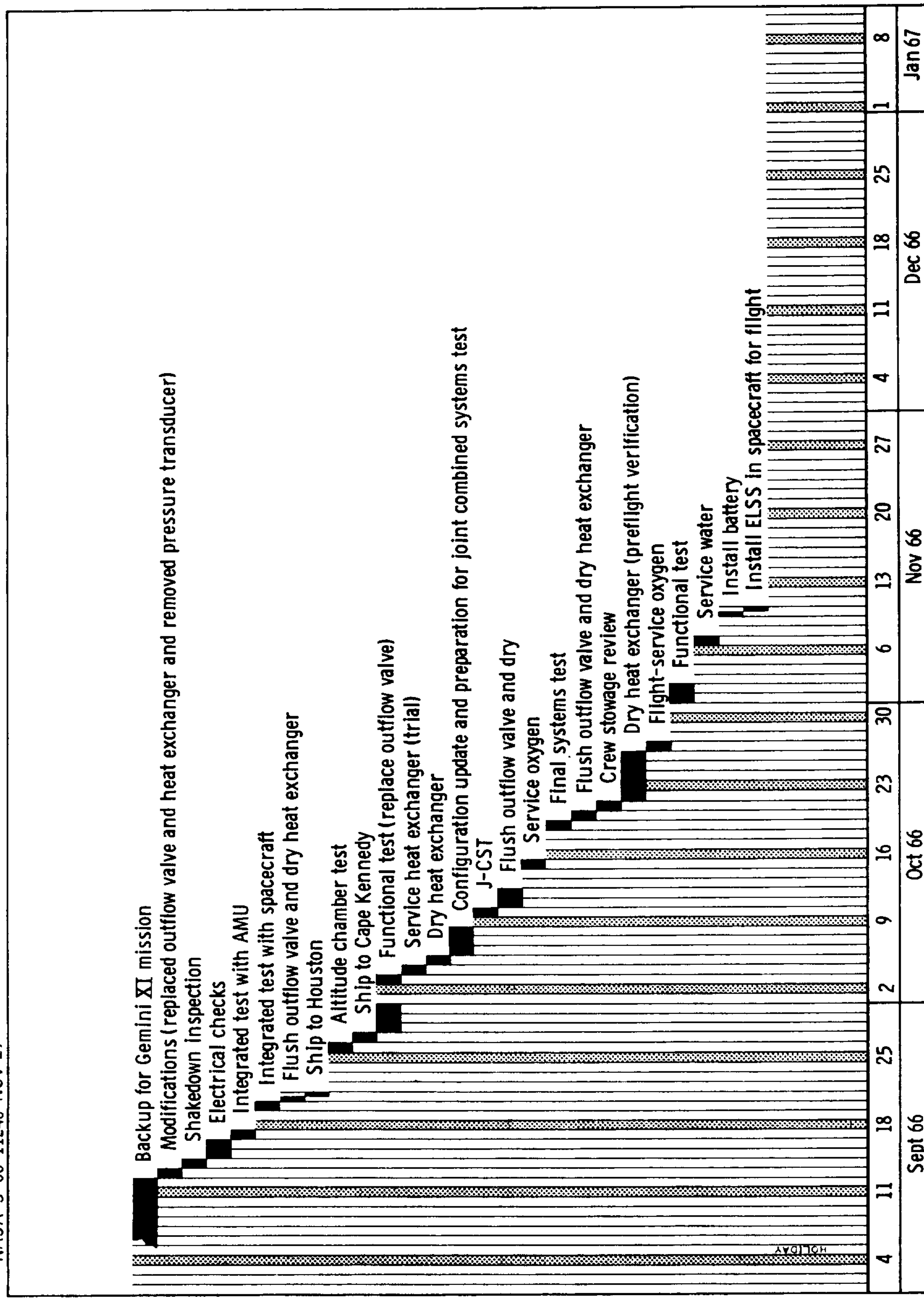

Figure 12.1-11. - ELSS-113 history at Cape Kennedy.

12.2 WEATHER CONDITIONS

The weather conditions at Cape Kennedy were satisfactory for all operations on the day of the launch, November 11, 1966. Surface weather conditions at 3:46 p.m. e.s.t. were as follows:

Cloud coverage	High scattered, 4/10 covered
Wind direction, deg from North	80
Wind velocity, knots	5
Visibility, mi.	10
Pressure, in. Hg	30.09
Temperature, °F	76
Dew point, °F	66
Relative humidity, percent	71

The prime recovery ship for the Gemini XII mission, the U.S.S. Wasp, was stationed at 24 degrees 33.9 minutes north, 69 degrees 56.9 minutes west on November 15, 1966. Weather conditions observed in the area at 20:25 G.m.t. on that day were as follows:

Cloud coverage	2800 feet, scattered; 12 000 feet, broken
Wind direction, deg from North	10
Wind velocity, knots	20
Visibility, miles	10
Temperature, °F	78
Relative humidity, percent	70
Sea temperature, °F	84
Sea state	2-foot waves, 4-foot swells

Atmospheric conditions on the day of launch are shown in table 12.2-I and conditions in the spacecraft recovery area are shown in table 12.2-II. Figures 12.2-1 and 12.2-2 show the launch-area and reentry-area wind velocities and directions plotted against altitude.

TABLE 12.2-I.- LAUNCH AREA ATMOSPHERIC CONDITIONS

AT 22:50 G.m.t., NOVEMBER 11, 1966

Altitude, ft (a)	Temperature, °F (a)	Pressure, lb/ft^2 (a)	Density, $slugs/ft^3$ (a)
0×10^3	78.4	2125.07	2280.6×10^{-6}
5	57.7	1782.55	1997.5
10	48.9	1487.44	1699.9
15	32.7	1234.11	1457.9
20	18.2	1018.15	1242.0
25	1.3	834.16	1055.9
30	-23.1	677.31	904.0
35	-45.8	543.85	765.6
40	-66.6	431.3	639.7
45	-84.6	338.3	525.6
50	-94.9	263.2	420.3
55	-101.4	203.4	330.8
60	-102.5	157.1	256.1
65	-83.2	121.8	188.6
70	-69.5	95.7	142.6
75	-66.1	75.4	111.6
80	-61.4	58.2	85.4
85	-55.8	46.4	66.8
90	-52.6	36.8	52.8
95	-50.6	29.2	41.7
100	-42.9	22.3	31.1
105	-38.7	18.8	26.0
110	-42.3	15.0	21.0
115	-41.4	12.1	16.9
120	-28.1	9.8	13.2
125	-30.3	7.9	10.7
130	-26.5	6.3	8.5
135	-20.6	5.2	6.8
140	-16.1	4.2	5.4
145	-4.9	3.3	4.3
150	5.0	2.7	3.5
155	12.4	2.3	2.9
160	17.6	1.9	2.3

[a]The accuracy of the readings is indicated at the end of the table.

TABLE 12.2-I.- LAUNCH AREA ATMOSPHERIC CONDITIONS

AT 22:50 G.m.t., NOVEMBER 11, 1966 - Concluded

Altitude, ft (a)	Temperature, °F (a)	Pressure lb/ft^2 (a)	Density, slugs/ft^3 (a)
165×10^3	21.2	1.5	1.9×10^{-6}
170	26.2	1.3	1.6
175	29.7	1.0	1.4
180	31.8	0.8	1.0
185	19.8	0.8	1.0
190	5.0	0.6	0.8
195	-3.1	0.4	0.6

[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 120×10^3	1	1	0.8
120 to 165×10^3	4	1.5	1.0
165 to 195×10^3	6	1.5	1.5

TABLE 12.2-II.- REENTRY AREA ATMOSPHERIC CONDITIONS

AT 19:23 G.m.t., NOVEMBER 15, 1966

Altitude, ft	Temperature, °F	Pressure, lb/ft^2	Density, slugs/ft^3
0 × 10^3	72.9	2130.9	2318.7 × 10^{-6}
5	45.5	1780.7	2047.2
10	34.5	1478.3	1741.2
15	16.2	1218.9	1491.9
20	-0.2	998.7	1266.2
25	-21.5	811.0	1078.6
30	-43.6	652.0	913.3
35	-70.3	518.6	756.9
40	-69.0	409.6	610.6
45	-76.5	321.8	489.5
50	-89.0	251.5	395.1
55	-90.8	195.3	308.5
60	-88.8	151.8	238.5
65	-82.1	118.4	182.8
70	-77.1	92.7	141.3
75	-70.2	72.9	109.1
80	-61.2	58.1	85.0
85	-49.4	46.2	65.6
90	-48.6	36.8	52.2
95	-48.1	29.5	41.5
100	-35.9	23.6	32.4
105	-35.7	19.0	26.0
110	-26.9	15.3	20.6
115	-13.2	12.3	16.1
120	-14.3	10.0	13.2
125	-2.9	8.1	10.5
130	10.8	6.7	8.3
135	21.4	5.4	6.6

[a]The accuracy of the readings is indicated at the end of the table.

TABLE 12.2-II.- REENTRY AREA ATMOSPHERIC CONDITIONS

AT 19:23 G.m.t., NOVEMBER 15, 1966 - Concluded

Altitude, ft	Temperature, °F	Pressure, lb/ft^2	Density, $slugs/ft^3$
140×10^3	19.3	4.6	5.6×10^{-6}
145	23.4	3.8	4.5
150	22.6	3.1	3.7
155	32.9	2.5	3.1
160	29.3	2.0	2.5
165	35.6	1.9	2.1
170	35.8	1.5	1.8
175	36.9	1.3	1.4
180	25.4	1.0	1.2
185	19.4	0.8	1.0
190	16.5	0.6	0.8
195	12.9	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 120×10^3	1	1	0.8
120 to 165×10^3	4	1.5	1.0
165 to 195×10^3	6	1.5	1.5

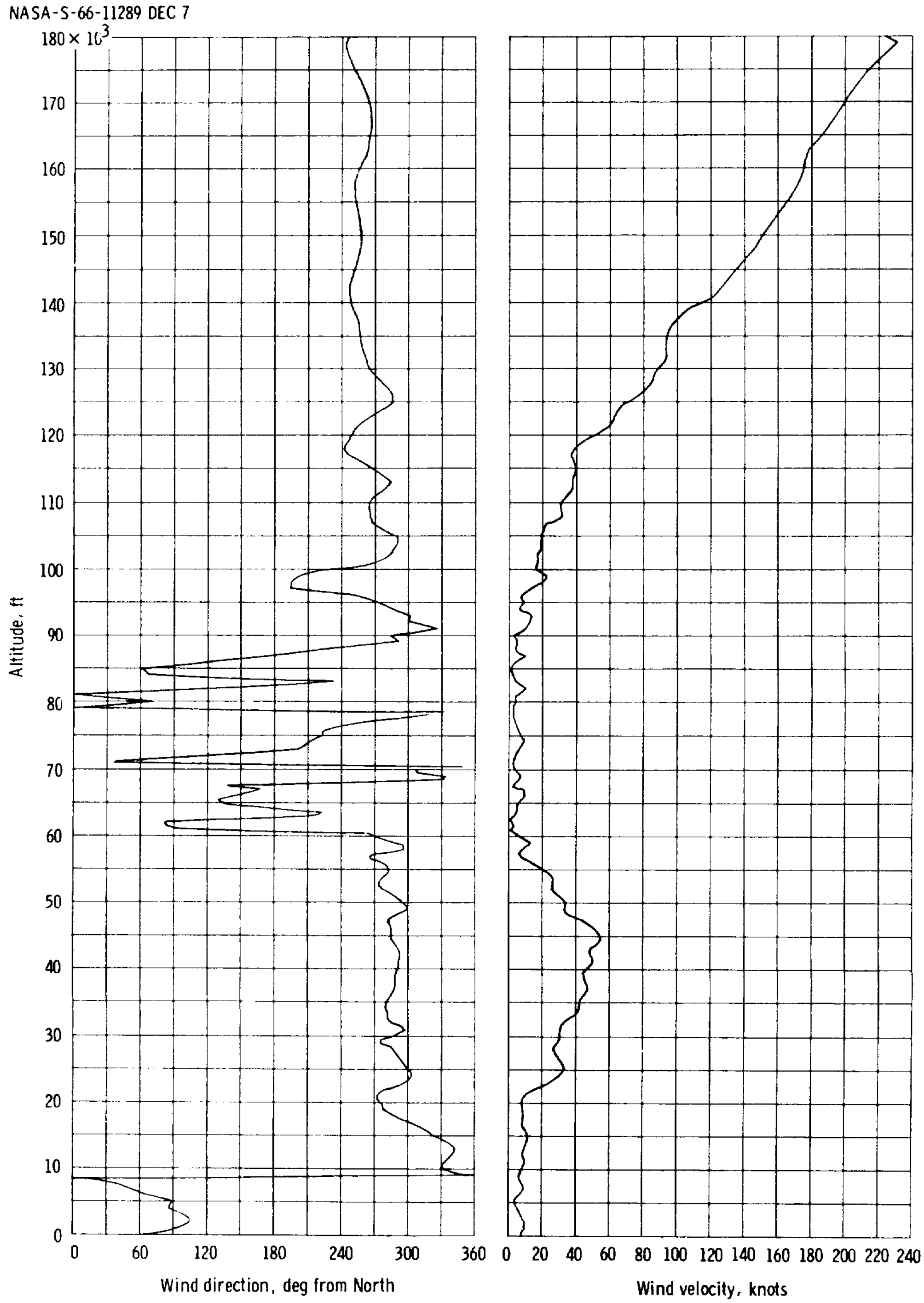

Figure 12.2-1. - Variation of wind direction and velocity with altitude for the Gemini Space Vehicle and GAATV at 20:47 G.m.t., November 11, 1966.

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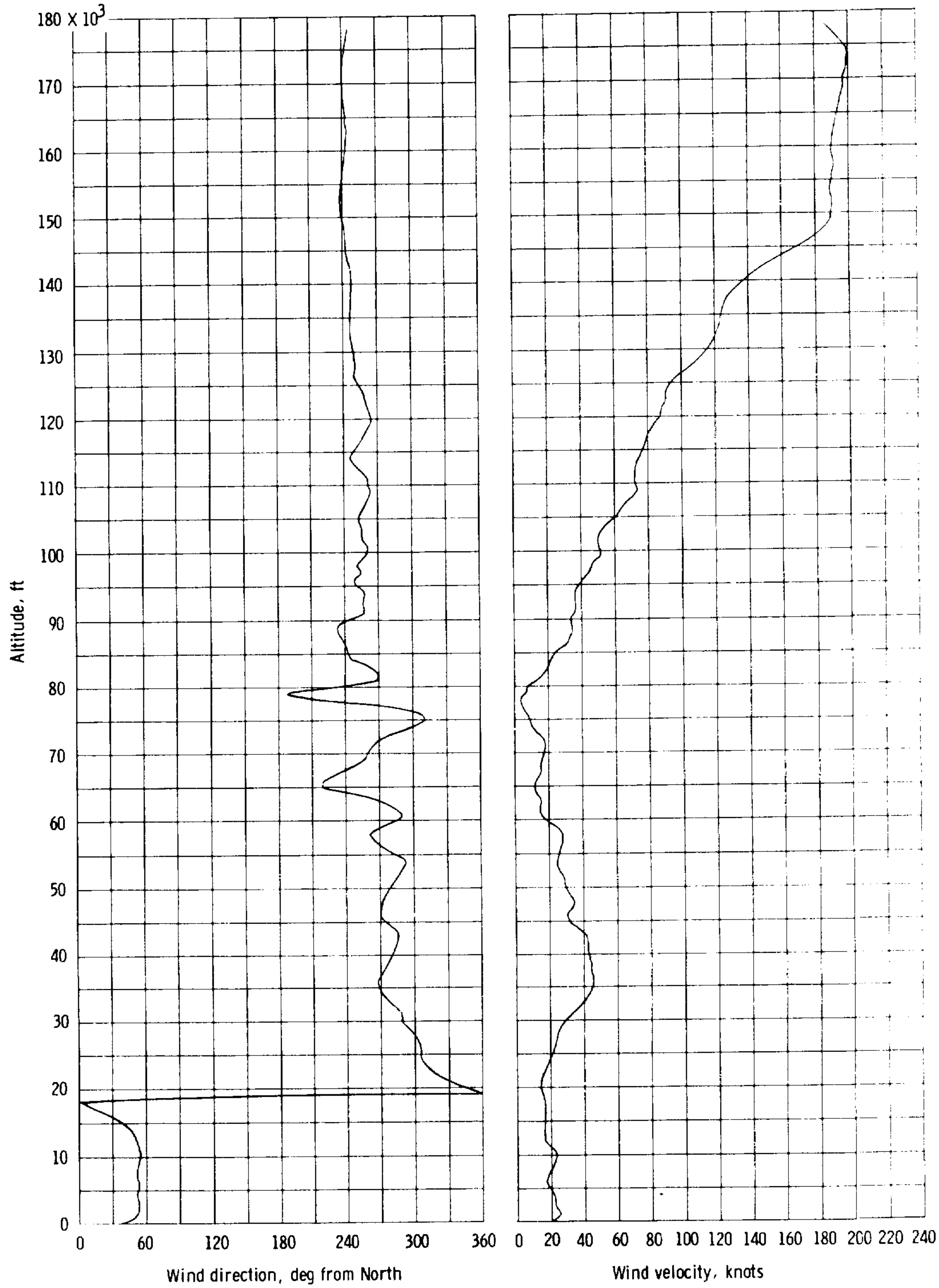

Figure 12.2-2.- Variation of wind direction and velocity with altitude for the Gemini XII reentry area at 17:45 G.m.t., November 15, 1966.

12.3 FLIGHT SAFETY REVIEWS

12.3.1 Gemini Launch Vehicle Technical Reviews

A technical review of Gemini Launch Vehicle-12 (GLV-12) was held during the GLV Coordination Committee Meeting at the Manned Spacecraft Center on October 25, 1966. Updates were presented at the GLV-12 Preflight Status Briefing held at the Air Force Eastern Test Range on November 5, 1966. No significant problems were outstanding.

12.3.2 Gemini Atlas-Agena Target Vehicle Technical Review

Technical reviews of the Gemini Agena Target Vehicle (GATV-5001) and the Target Launch Vehicle (SLV-3 5307) were held during the Atlas-Agena Coordination meeting at the Manned Spacecraft Center on October 26, 1966. Updates were presented at the Gemini XII Atlas-Agena Preflight Status Briefing, held at the Air Force Eastern Test Range on November 5, 1966. No significant problems were outstanding.

12.3.3 Flight Safety Review Board

The Gemini XII Flight Safety Review Board was convened at the Air Force Eastern Test Range on November 8, 1966. Following technical summaries, conclusions, and recommendations by the Air Force and the contractors, the Flight Safety Review Board recommended to the Mission Director that the GLV and GAATV be committed to flight.

Subsequent action was required by the AFSSD Flight Safety Review Board because of two launch delays.

(a) During the prelaunch operations on November 8, 1966, a problem was discovered in the GLV secondary autopilot. The secondary autopilot package and the secondary Stage I rate gyro package were replaced, and the Status Review Team met on November 9, 1966, to review the corrective action. A recommendation was made to the AFSSD Flight Safety Review Board that the GLV be committed to flight. The mission was rescheduled to start on November 10, 1966.

(b) During the precount tests on November 9, 1966, the Stage II secondary rate gyro spin-motor rotation detector indicated another secondary autopilot problem. This problem was resolved by replacing the secondary autopilot package. The Status Review Team met again on November 10, 1966, to review applicable data and recommended that the GLV be committed for flight. The AFSSD Flight Safety Review Board accepted the recommendation and the mission was rescheduled to start on November 11, 1966.

12.4 SUPPLEMENTAL REPORTS

Supplemental reports for the Gemini XII mission are listed in table 12.4-I. The format of these reports will conform to the external distribution format of NASA or to that of the external organization preparing the report. Each report will be identified on the cover page as a Gemini XII supplemental report. Distribution of the supplemental reports will be the same as that of this Gemini Program Mission Report.

TABLE 12.4-I.- GEMINI XII SUPPLEMENTAL REPORTS

Number	Report title	Responsible organization	Completion due date
1	Launch Vehicle Flight Evaluation Report - NASA Mission Gemini/Titan GT-12	Aerospace Corp.	January 13, 1967
2	Launch Vehicle No. 12 Flight Evaluation	Martin Co.	December 30, 1966
3	Manned Space Flight Network Performance Analysis for Gemini XII Mission	Goddard Space Flight Center	January 13, 1967
4	Gemini XII IGS Evaluation Trajectory Reconstruction	TRW Systems	December 30, 1966
5	Gemini XII Postflight Analysis Report	International Business Machines Corp.	December 30, 1966
6	Gemini Agena Target Vehicle 5001 Systems Test Evaluation	Lockheed Missiles and Space Co.	December 30, 1966
7	Gemini PCM Tape Recorder Performance Summary	McDonnell Aircraft Corp.	March 3, 1967
8	Gemini Fuel Cell Performance Analysis	McDonnell Aircraft Corp.	March 3, 1967
9	Gemini TCA Anomaly Investigation	McDonnell Aircraft Corp.	March 3, 1967

12.5 DATA AVAILABILITY

Tables 12.5-I through 12.5-IV list the Gemini XII 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.

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TABLE 12.5-I.- INSTRUMENTATION

Data description
1.- Paper recordings
Spacecraft telemetry measurements of selected parameters (revolutions 1, 2, 3, 4, 5, 10, 11, 12, 13, 14, 15, 16, 25, 26, 27, 28, 29, 30, 31, 32, 33, 34, 39, 40, 42, 43, 44, 45, 46, 55, 56, 57, 58, 59, and reentry)
GLV telemetry measurements (launch)
Telemetry signal-strength recordings
MCC-H plotboards (Confidential)
Range safety plotboards (Confidential)
2.- Radar data
IP 3600 trajectory data (Confidential)
MISTRAM (Confidential)
Natural coordinate system
Final reduced
C-band (launch phase) (Confidential)
Natural coordinate system
Final reduced
Trajectory data processed at MSC and GSFC
3.- Voice transcripts
Air-to-ground
Onboard recorder (Confidential)
Technical debriefing (Confidential)
System debriefing (Confidential)
4.- GLV reduced telemetry data (Confidential)
Engineering units versus time plots
5.- Spacecraft reduced telemetry data
Engineering units versus time
Ascent phase
Bandpass tabulations of selected parameters
Orbital phase
Time history tabulations of selected parameters for selected times for revolutions 1, 2, 3, 4, 5, 23, 24, 25, 49, 50, 53, 54, 55, and 56

UNCLASSIFIED

TABLE 12.5-I.- INSTRUMENTATION - Concluded

Data description	
Orbital phase - concluded Time history tabulations (computer words) for revolutions 1, 2, 3, 4, 5, 10, 11, 12, 25, 30, 31, 32, 33, 39, 46, 55, 58, and 59 Bandpass tabulation of selected parameters for revolutions 1, 2, 3, 4, 5, 10, 11, 12, 23, 24, 25, 26, 27, 30, 33, 34, 39, 48, 49, 50, 53, 54, 55, 58, and 59 Reentry phase McDonnell time history presentations (plots and tabulations of all systems parameters) 6.- Event tabulations Sequence-of-event tabulations versus time (including thruster firings) for ascent, reentry, and revolutions 1, 2, 3, 4, 5, 10, 11, 12, 13, 14, 15, 16, 17, 18, 23, 24, 25, 26, 27, 28, 29, 30, 31, 32, 33, 34, 39, 40, 43, 44, 45, 46, 47, 48, 49, 50, 53, 54, and for selected real-time passes for revolutions 1, 2, and 49	7.- Special computations Ascent phase MISTRAM versus IGS velocity comparison (Confidential) MOD III RGS versus IGS velocity comparison (Confidential) Orbital phase OAMS propellant-remaining, thruster-activity, and thrust-duration computations for revolutions 1, 2, 3, 4, 5, 10, 11, 12, 13, 14, 15, 16, 17, 18, 25, 26, 27, 28, 29, 30, 31, 33, 34, 39, 40, 43, 44, 45, 46, 47, 48, 49, 55, 56, 57, 58, and 59 Reentry phase RCS propellant-remaining and thruster-activity computations Lift-to-drag ratio and auxiliary computations

TABLE 12.5-II.- SUMMARY OF PHOTOGRAPHIC DATA AVAILABILITY

Category	Number of still photographs	Motion picture film, ft
Launch		
TLV/GATV	(a)	[b]967
GLV/spacecraft	(a)	[b]3636
Recovery		
Spacecraft in water	77	1100
Loading of spacecraft on carrier	15	200
Inspection of spacecraft	24	--
Boston, Mass.		
General activities	12	100
Inspection of spacecraft	18	200
Postflight inspection	20	--
Inflight photography		
Rendezvous and docking	21	292
Tether evaluation	57	270
Weather and terrain	291	150
Extravehicular activity	42	486
Eclipse	2	32
Miscellaneous	14	--
Reentry	--	63
Experiment S011, Airglow Photography	18	--
Experiment S013, Ultraviolet Astronomical Camera	27	--
Experiment S029, Libration Regions Photography	(c)	--

[a]Not normally used for evaluation purposes.

[b]Engineering sequential film only.

[c]No useful pictures were obtained due to a camera malfunction.

TABLE 12.5-II.- SUMMARY OF PHOTOGRAPHIC DATA AVAILABILITY - Concluded

Category	Number of still photographs	Motion picture film, ft
Sunrise ultraviolet photography	(d)	--
Experiment S051, Sodium Cloud Photography	(e)	--

[d]No useful pictures were obtained because of too short exposure time.

[e]No useful pictures were obtained due to a camera malfunction.

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
1.2-9	16	50-foot tower, 19-1	GLV launch	140
1.2-10	16	50-foot tower, 19-5	GLV launch	180
1.2-11	16	50-foot tower, 19-5	GLV launch	180
1.2-12	16	50-foot tower, 19-2	Spacecraft launch	46
1.2-14	16	Umbilical tower, second level	GLV Stage II umbilical	100
1.2-15	16	50-foot tower, 19-7A	GLV, engine observation	100
1.2-16	16	East launcher	GLV, possible fuel leakage	200
1.2-17	16	West launcher	GLV, possible fuel leakage	165
1.2-18	16	North launcher	GLV, engine observation	115
1.2-19	16	South launcher	GLV, engine observation	160
1.2-20	16	Umbilical tower, first level	GLV, umbilical disconnect	100
1.2-21	16	Umbilical tower, second level	GLV, umbilical disconnect	80
1.2-22	16	Umbilical tower, fourth level	GLV, umbilical disconnect	80
1.2-23	16	Umbilical tower, fifth level	GLV, umbilical disconnect	125
1.2-24	16	Umbilical tower, sixth level	GLV, umbilical disconnect	130
1.2-25	16	Umbilical tower, sixth level	GLV, umbilical disconnect	210
1.2-26	16	Umbilical tower, top level, no. 1	GLV, upper umbilical disconnect	130
1.2-27	16	Umbilical tower, top level, no. 2	J-bars and lanyard observation	208
1.2-34	16	South of Pad 19	Tracking	336
1.2-36	35	South of Pad 19	Tracking	308
1.2-40	70	Cocoa Beach	Tracking, ROTI	130
1.2-41	70	Melbourne Beach	Tracking, ROTI	128
1.2-42	35	Patrick AFB, Florida	Tracking, IGOR	220
None	70	ALOTS aircraft	Tracking	65

TABLE 12.5-III.- LAUNCH PHASE ENGINEERING SEQUENTIAL CAMERA DATA AVAILABILITY - Concluded

(b) TLV and GATV

Sequential film coverage, item	Size, mm	Location	Presentation	Total length of film, ft
1.2-4	16	East of Pad 14	TLV engine observation	71
1.2-5	16	West of Pad 14	TLV engine observation	71
1.2-6	16	Northwest of Pad 14	TLV engine observation	81
1.2-7	16	Ramp, south of Pad 14	TLV engine observation	66
1.2-8	16	West of Pad 14	TLV launch	50
1.2-9	16	Northwest of Pad 14	TLV launch	41
1.2-10	16	Northwest of Pad 14	TLV vernier-engine heat shield	66
1.2-11	16	Southeast of Pad 14	TLV vernier-engine heat shield	100
1.2-12	16	Umbilical tower, 72-foot level	Upper umbilical disconnect	51
1.2-13	16	Umbilical tower, 72-foot level	Lower umbilical disconnect	80
1.2-19	35	Patrick Air Force Base	Tracking, IGOR	225
None	70	ALOTS aircraft tracking film	Tracking	65

TABLE 12.5-IV.- SUMMARY OF DATA AVAILABILITY ON GEMINI XII GATV

GATV event	Revo-lution	Nearest station	Reduced data (engineering units) and paper recordings[a]							
			Bandpass (b)	Bilevels	Time history plot groups (b)	Time history tab groups (b)	Turbine speed and velocity meter	Programmer memory	Control gas impulse	Paper recordings
Launch		CNV	X	X	1, 4, 6, 8	3, 5, 6, 7, 7A, 8, 9, 10	X	X		X
Radar problem	2/3	ANT	X	X		29				
	3	CRO	X	X		29				
	3	HAW	X	X		29				
	3	GYM	X	X		29				
	3/4	MCC-K	X	X		29				
	3/4	ANT	X	X		29				
	4	HAW	X	X		29				
First docking	4	CSQ	X	X	1, 8			X		
First undock-ing	4	HAW	X	X	8			X		
Second docking attempt	5	RKV	X		1, 8		X	X		
Second docking	5	RKV/TAN	X	X	1, 8			X		X
Second undock-ing	5	HAW	X	X	8			X		X
Secondary pro-pulsion sys-tem firing no. 1	6	TAN/CSQ	X	X	1, 2, 6, 7, 8	11 thru 15	X	X	X	X

[a,b]See notes at end of this table.

TABLE 12.5-IV.- SUMMARY OF DATA AVAILABILITY ON GEMINI XII GATV - Continued

GATV event	Revo-lution	Nearest station	Reduced data (engineering units) and paper recordings[a]							
			Bandpass (b)	Bilevels	Time history plot groups (b)	Time history tab groups (b)	Turbine speed and velocity meter	Programmer memory	Control gas impulse	Paper recordings
Secondary propulsion system firing no. 2	11	KNO	X	X	1, 2, 6, 7, 8	11 thru 15	X	X	X	X
Velocity-meter problem	35	HAW	X	X			X			
	35	CSQ	X	X			X			
	39	RKV	X	X			X			
	43/44	MCC-K	X	X			X			
	62	HAW	X	X			X			
	38	CSQ	X	X			X			
	41/42	ANT	X	X			X			
Final status	75	MCC-K	X	X				X		
Thruster test	1	CRO	X	X	8					
	31	CRO	X	X	8					
Gas consumption	29/30	MCC-H	X	X	29	8			X	

[a,b]See notes at end of this table.

TABLE 12.5-IV.- SUMMARY OF DATA AVAILABILITY ON GEMINI XII GATV - Continued

GATV event	Revo-lution	Nearest station	Reduced data (engineering units) and paper recordings[a]							
			Bandpass (b)	Bilevels	Time history plot groups (b)	Time history tab groups (b)	Turbine speed and velocity meter	Programmer memory	Control gas impulse	Paper recordings
Tether evaluation	30	CYI	X	X	29	8				
	30	HAW	X	X	29	8				
	30/31	MCC-H	X	X	29	1, 8				
	30/31	ANT	X	X	29	8				
	31	CRO	X	X	29	8				
	31	HAW	X	X	29	1, 8				
	34	TAN	X	X	29	8				
	31/32	MCC-H	X	X	29	1, 8				
	31/32	ANT	X	X	29	8				
	32	ASC	X	X	29	8				
	29/30	MCC-H	X	X	29	8				
	32	CRO	X	X	29	8				
	32	HAW	X	X	29	8				
	32/33	MCC-H	X	X	29	1, 8				
	33	HAW	X	X	29	1, 8				
	33	TEX	X	X	29	1, 8				
Primary propulsion system firing no. 1	63	HAW	X	X						X

[a,b]See notes at end of this table.

TABLE 12.5-IV.- SUMMARY OF DATA AVAILABILITY ON GEMINI XII GATV - Continued

GATV event	Revolution	Nearest station	Reduced data (engineering units) and paper recordings[a]							
			Bandpass (b)	Bilevels	Time history plot groups (b)	Time history tab groups (b)	Turbine speed and velocity meter	Programmer memory	Control gas impulse	Paper recordings
Fuel venting	1/2	CNV	X							
	1	CYI	X							
	1	HAW	X							
Structure current	24	CSQ	X							
	45	HAW	X							
Inverter temperature	10	CSQ	X							
	59	CRO	X							
	60	HAW	X							
	61	HAW	X							

[a,b]See notes at end of this table.

TABLE 12.5-IV.- SUMMARY OF DATA AVAILABILITY ON GEMINI XII GATV - Concluded

[a]In addition to the data listed in this table the following data are available: TLV telemetry measurements (launch), MCC-H and Range Safety plotboards (launch), C-band overlapping trajectory (Confidential) (launch) - final reduced coordinate system 2 and 3, Rawindsonde weather data (launch), Sunrise-sunset computations (orbital).

[b]Time history plot and time history and bandpass tabulation group numbers listed in table 12.5-IV are related to GATV systems as shown in the following table. (All bandpass tab groups run unless otherwise indicated.)

Systems	Time history plot groups	Time history and bandpass tab groups
Structural	1, 2	1, 1A, 2, 3, 4A
Primary propulsion	3, 4, 5	5, 6, 7, 7A, 8, 9, 10
Secondary propulsion	6	11, 12, 13, 14, 15
Electrical	7	16, 17, 18, 19
Guidance and control	8	20, 21, 22, 23, 24
Command and communications	9	25, 26, 27, 28, 29, 30

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12.6 POSTFLIGHT INSPECTION

The postflight inspection of the Spacecraft 12 reentry assembly was conducted in accordance with reference 21 and Spacecraft Test Requests (STR's) at the contractor's facility in St. Louis, Missouri, from November 19, 1966, to December 14, 1966. The rendezvous and recovery (R and R) section and the parachutes were not recovered. While the spacecraft was still aboard the recovery ship, the crew-station items defined in STR 12000 were removed and properly disposed of, and, in addition, several items were removed from the equipment bays and treated in accordance with reference 22.

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 surfaces of both hatch windows.

(b) An excessive amount of water (approximately two gallons) was found in the Environmental Control System (ECS) bay well. A considerable amount of residue remained in the well when the water was removed.

(c) The left-hand skid well door had a shingle curled on the corner and a washer missing. The forward lower centerline equipment bay door was deflected inward.

(d) The external umbilical receptacle had elongated pin holes on the left-hand side.

(e) The ends of the hat stiffeners on equipment access doors no. 28 and no. 30 were bent and fractured.

(f) The electrical connector on the C-band transponder static inverter was broken from the case.

(g) An open circuit was found in the pilot's communication system egress kit electrical disconnect.

(h) The heat shield had circumferential score marks in the lower left-hand quadrant.

(i) The pyrotechnic cartridge on the retrorocket-wire pyrotechnic-switch H-1 indicated a possible unfired pyrotechnic.

(j) The insulation retainer under the left-hand center hatch flipper door was damaged.

12.6.1 Spacecraft Systems

12.6.1.1 Structure.- The overall appearance of the spacecraft was good. The appearance of the heat shield was normal except for the circumferential score marks in the lower left-hand quadrant. The heat shield was cored per STR 12506. The heat shield stagnation point was measured to be 2.0 inches to the right of the vertical centerline and 20.0 inches below the horizontal centerline. The heat shield was removed and dried with the reentry assembly. The dry weight of the heat shield was 320.78 pounds.

Residue similar to that found on the exterior surface of the windows of previous spacecraft was noted; however, all other surfaces of the window assemblies were relatively clean. STR 12009 was initiated to analyze the residue and determine the optical condition of the windows.

The left-hand skid well door had a shingle curled on the corner and a washer missing. The forward lower centerline equipment bay door was deflected inward. The damage was attributed to the landing impact.

The ends of the hat stiffeners on equipment access doors no. 28 and no. 30 were bent and fractured. The hat ends had the appearance of being squeezed together to allow fitting of the external shingles.

The insulation retainer under the left-hand center hatch flipper door was damaged. The probable cause was interference of the retainer with a shingle washer during opening of the left-hand hatch.

The external appearance of the shingles, doors, and adapter attach fairings appeared similar to those of previous spacecraft after reentry.

12.6.1.2 Environmental Control System.- The drinking water was removed and prepared for analysis in accordance with reference 21. The total remaining in the system was 4220 cubic centimeters or about a gallon of water. The lithium-hydroxide cartridge was removed from the ECS package and weighed. The cartridge weighed 114.51 pounds, and the center-of-gravity was determined to be 7.92 inches from the bottom.

The ECS well contained approximately two gallons of water. A considerable amount of residue remained when the water was removed. The postlanding checklist of the Gemini XII Flight Plan required the water-seal valve to be closed after landing. The open valve during landing was the probable cause of water leakage into the cabin. A sample of water and residue was removed for analysis (STR 12502).

The secondary oxygen system was deserviced in accordance with reference 21. The left-hand system had a residual pressure of 27 psia and the right-hand system was at 45 psia.

The ECS control handles were actuated in accordance with reference 21, and the maximum forces recorded were 20 pounds on the cabin-vent handle and on the oxygen high-rate recock handle.

An attempt was made to remove a water sample from the condensate lines (STR 12010), but insufficient condensate was obtained for analysis.

12.6.1.3 <u>Communications System</u>.- The external appearance of all communications equipment and antennas was good. The electrical connector on the C-band transponder static inverter was broken from the case. The fractured metal band of the connector was bright and not corroded, indicating that the damage was incurred after landing.

The pilot's communication system was investigated for an anomaly in the helmet right-hand microphone circuit (STR 12015).

12.6.1.4 <u>Guidance and Control System</u>.- While the spacecraft was still aboard the prime recovery ship, the Inertial Measurement Unit (IMU) system and the computer were removed and packaged for delivery to the vendor representatives in Boston, Massachusetts (STR's 12001A and 12002B). The Auxiliary Computer Power Unit (ACPU), the Attitude Control and Maneuver Electronics (ACME), and the horizon-sensor electronics were removed, returned to St. Louis, Missouri, and then sent to the applicable vendor (STR's 12003, 12004 and 12005).

12.6.1.5 <u>Pyrotechnic System</u>.- Pyrotechnic resistance measurements were made of all electrically initiated pyrotechnic devices in the reentry assembly in accordance with reference 21. Tests on the firing circuit of the retrorocket-wire pyrotechnic-switch H-1 cartridge indicated a bridgewire resistance that was near the unfired value. The cartridge was removed for a visual inspection (STR 12504) which revealed that the cartridge had detonated normally. The measured resistance was due to the conductive residue remaining in the cartridge after firing. The same condition was noted on the equivalent cartridge in Spacecraft 10 and 11.

All wire-bundle guillotines, parachute bridle-release mechanisms, and other pyrotechnically operated devices 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 nearly all previous spacecraft and is considered acceptable. Both of the MDF detonators appeared to have had a normal high-order detonation.

12.6.1.6 Instrumentation and Recording System.- While the spacecraft was still aboard the recovery ship, the PCM programmer and multiplexers were removed and packaged for release to the vendor's representative at Boston, Massachusetts (STR 12007A). Instrumentation package no. 2 was also removed, but it was returned with the spacecraft to St. Louis (STR 12006). The PCM tape recorder was removed as soon as possible on the prime recovery ship and returned by special courier to St. Louis for data processing (STR 12008). The dc-to-dc converters were removed and returned to St. Louis (STR 12500). The biomedical tape recorders were removed and carried by courier to MSC for data processing (STR 12000).

The circuitry, detector, and indicator of the carbon dioxide detection system were investigated (STR 12507).

12.6.1.7 Electrical System.- The main and squib batteries were removed and discharged in accordance with reference 21. The following table lists the ampere-hours remaining in each battery when discharged to the level of 20 volts, with the batteries still delivering the current specified in reference 21.

Main battery	Discharge, A-h	Squib battery	Discharge, A-h
1	17.50	1	8.00
2	35.00	2	10.00
3	32.50	3	8.00
4	11.25		

The main and squib batteries were recharged and placed in bonded storage for use in ground tests.

After the spacecraft was dried, no current leakage was detected on the main-bus-to-ground circuits when the main battery switches were actuated in accordance with reference 21.

The fuse blocks were checked for open fuses or fusistors in accordance with reference 21, and the following fusistors were open:

Fuse block	Pin no.	Fuse no.
XF-AF	4	5-120
XF-C	3	4-14
XF-J	3	4-39
XF-AV	4	14-46

The external umbilical receptacle had elongated pin holes on the left-hand side.

The cryogenic-hydrogen heater switch and circuitry were investigated (STR 12503).

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 21. The ejection seats were removed and deactivated in accordance with reference 21. The back-board contours, pelvic blocks, and lap belts were placed in bonded storage at the contractor's plant in St. Louis, Missouri. The seat ballast was shipped to KSC. The survival packs, water metering dispenser, and retractable pencils were removed and sent to MSC (STR 12000).

12.6.1.9 Propulsion System.- The RCS thrust chamber assemblies appeared normal. Prior to shipping the spacecraft to St. Louis, Missouri, the RCS was deactivated at Boston, Massachusetts, in accordance with reference 20. The following amounts of propellants were removed from the RCS tanks at Boston:

Propellant	A-ring	B-ring
Oxidizer, lb	0.25	4.50
Fuel, lb	0.06	2.00

The RCS A-ring pressure regulator was investigated for leakage (STR 12014). RCS thruster 1A was tested for leakage (STR 12016). The fuel valve signatures of RCS thrusters 2A and 4A were checked (STR 12017).

The RCS section was dried with the spacecraft in the 30-foot altitude chamber in accordance with reference 21.

12.6.1.10 Postlanding recovery aids.- The flashing recovery light and the hoist-loop door appeared to have functioned normally. The sea dye marker container was removed from the spacecraft on the recovery ship and returned to St. Louis as a loose piece.

12.6.1.11 Experiments.- The bremsstrahlung spectrometer data processor package and sensor were removed at the contractor's facility in St. Louis and shipped to MSC (STR 12000).

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 no.	System	Purpose
12011	Crew Station	To investigate the failure of the 16-mm EVA sequence camera
12012	Crew Station	To investigate an anomaly in the 16-mm sequence camera
12014	Propulsion	To determine the cause of the leakage of the regulator in the RCS A-ring
12015	Communications	To establish the cause for the failure of the pilot's right-hand helmet microphone
12016	Propulsion	To determine the cause of leakage in the fuel valve of RCS engine no. 1 (noted on the recovery ship)
12502	Structure	To determine the identity of the fluid and residue noted in the ECS well after recovery
12503	Electrical	To determine whether an anomaly exists in the recovered portion of the cryogenic-hydrogen heater circuit
12504	Pyrotechnics	To examine the pyrotechnic cartridges in switch H-1 for proper operation
12505	Guidance and Control	To investigate the failure of the rendezvous radar/transponder
12507	Instrumentation	To investigate a possible anomaly in the carbon-dioxide partial pressure sensor

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Mr. V. R. Rawlings, Vice President Mail No. 14 Martin-Marietta Corporation Baltimore, Maryland 21203	1
Mr. Bastian Hello Director, Lifting Body Programs Mail No. 3070 Martin-Marietta Corporation Baltimore, Maryland 21203	1
Mr. W. D. Smith Director, Gemini Program Mail No. 3134 Martin-Marietta Corporation Baltimore, Maryland 21203	8
Mr. O. E. Tibbs, Vice President Mail No. A-1 Canaveral Division Martin-Marietta Corporation Cocoa Beach, Florida 32931	1

Addressee	Number of copies
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MCDONNELL AIRCRAFT CORPORATION	
Mr. Walter F. Burke Vice President and General Manager Spacecraft and Missiles McDonnell Aircraft Corporation Lambert-St. Louis Municipal Airport P. O. Box 516 St. Louis, Missouri 63166	25

Addressee	Number of copies
Mr. Frank G. Morgan McDonnell Aircraft Corporation 1730 NASA Road 1 Suite 101 Houston, Texas 77058	1

AERONAUTICAL RESEARCH ASSOCIATES OF PRINCETON

Dr. John C. Houbolt Aeronautical Research Associates of Princeton 50 Washington Road Princeton, New Jersey 08540	1

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GEMINI FLIGHT HISTORY

Mission	Description	Launch date	Major accomplishments
Gemini VIII	Manned Three days Rendezvous and dock EVA	Mar. 16, 1966	Demonstrated rendezvous and docking with GATV. Demonstrated controlled landing and emergency recovery. Demonstrated multiple restart of GATV in orbit. Spacecraft mission terminated early because of an electrical short in the control system.
Gemini IX	Manned Three days Rendezvous and dock EVA (Canceled after failure of TLV)	May 17, 1966	Demonstrated dual countdown procedures.
Gemini IX-A	Manned Three days Rendezvous and dock EVA	June 3, 1966	Demonstrated three rendezvous techniques. Evaluated EVA with detailed work tasks. Demonstrated precision landing capability.
Gemini X	Manned Three days Rendezvous and dock EVA	July 18, 1966	Demonstrated dual rendezvous using GATV propulsion for docked maneuvers. Demonstrated removal of experiment package from passive target vehicle during EVA. Evaluated feasibility of using onboard navigational techniques for rendezvous.
Gemini XI	Manned Three days Rendezvous and dock Tether evaluation EVA	Sept. 12, 1966	Demonstrated first-orbit rendezvous and docking. Evaluated EVA. Demonstrated feasibility of tethered station keeping. Demonstrated automatic reentry capability.
Gemini XII	Manned Four days Rendezvous and dock Tether evaluation EVA	Nov. 11, 1966	Demonstrated rendezvous and docking. Evaluated EVA. Demonstrated feasibility of gravity-gradient tethered-vehicle station keeping. Demonstrated automatic reentry capability.