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GEMINI PROGRAM MISSION REPORT

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GEMINI V

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AUGUST 1965

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION □ MANNED SPACECRAFT CENTER

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GEMINI SPACECRAFT FLIGHT HISTORY			
Mission	Description	Launch date	Major accomplishments
GT-1	Unmanned 64 orbits	Apr. 8, 1964	Demonstrated structural integrity.
GT-2	Unmanned suborbital	Jan. 19, 1965	Demonstrated heat protection and systems performance.
GT-3	Manned 3 orbits	Mar. 23, 1965	Demonstrated manned qualifications of the Gemini spacecraft.
Gemini IV	Manned 4 days	June 3, 1965	Demonstrated EVA and systems performance for 4 days in space.
Gemini V	Manned 8 days	Aug. 21, 1965	Demonstrated long-duration flight, rendezvous radar capability, and rendezvous maneuvers.

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MSC-G-R-65-4

GEMINI PROGRAM MISSION REPORT

GEMINI V

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MANNED SPACECRAFT CENTER

HOUSTON, TEXAS

OCTOBER 1965

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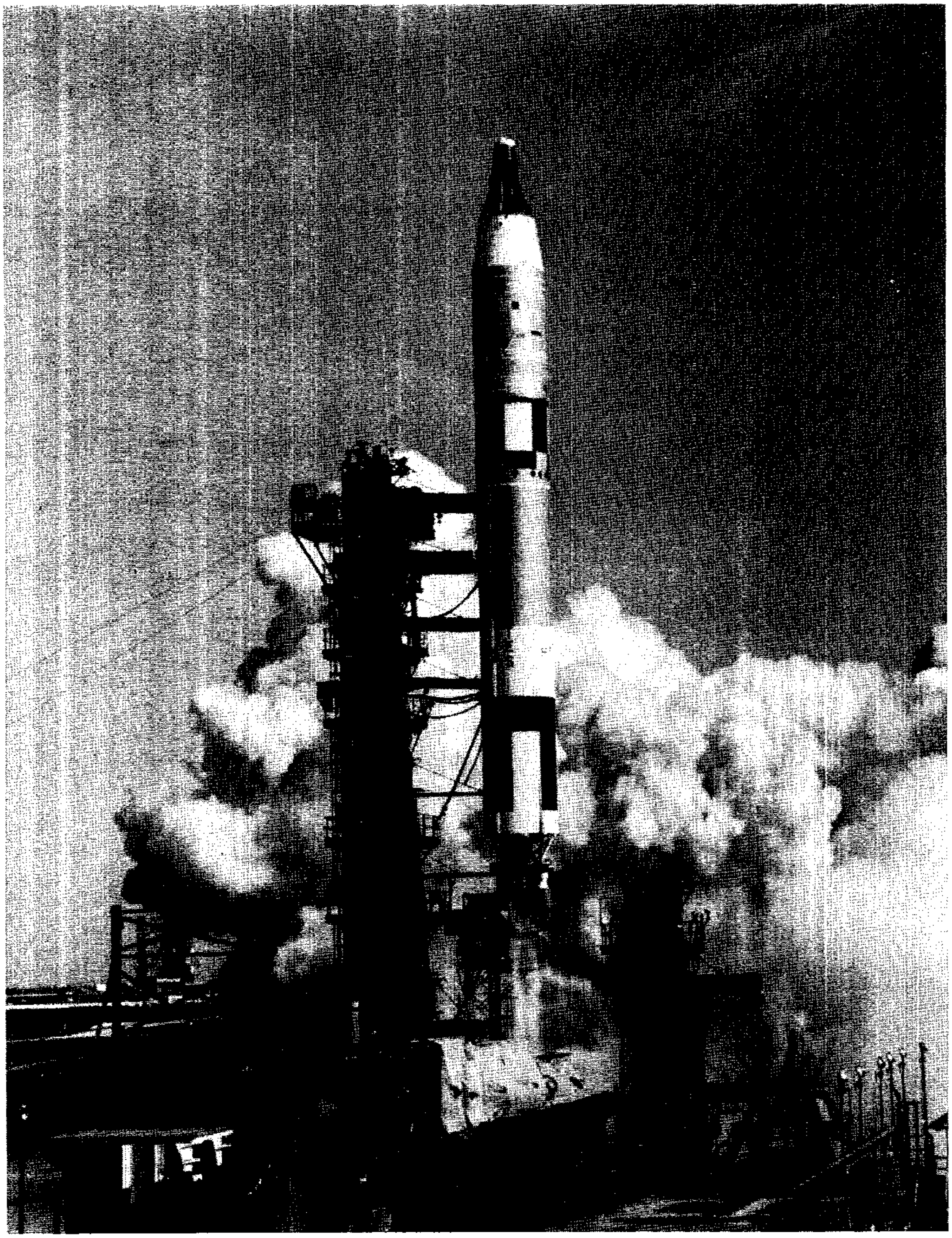

Gemini V space vehicle at lift-off

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1.0 MISSION SUMMARY

The third manned mission of the Gemini Program, designated Gemini V, was launched from Complex 19, Cape Kennedy, Florida, at 9:00 a.m. e.s.t., on August 21, 1965. The flight was successfully concluded on August 29, 1965, with the recovery of the spacecraft by the aircraft carrier U.S.S. Lake Champlain, at 29°52.5' N. latitude, 60°50.8' W. longitude. This 8-day long-duration flight was launched 11 weeks after the completion of the Gemini IV 4-day flight. The spacecraft was manned by Astronaut L. Gordon Cooper, Jr., command pilot, and Astronaut Charles Conrad, Jr., pilot. They completed the mission in excellent physical condition and demonstrated full control of the spacecraft and competent management of all aspects of the mission.

The major objectives of the Gemini V mission were to demonstrate manned orbital flight for approximately 8 days, evaluate the performance of the rendezvous guidance and navigation system, and evaluate the prolonged exposure of the flight crew to the space environment in preparation for missions of longer duration. In addition, it was desired to demonstrate a controlled reentry to a predetermined landing point, evaluate the fuel-cell performance under flight electrical load conditions, demonstrate all phases of guidance and control system operation necessary to support a rendezvous mission, evaluate the capability of either the pilot to maneuver the spacecraft in orbit to a close proximity with another object, evaluate the performance of the rendezvous radar, and execute 17 experiments.

The Gemini launch vehicle performed satisfactorily in all respects. The entire countdown was nominal, resulting in a launch precisely at the scheduled time. The first-stage flight was normal except for a short period of higher-than-expected longitudinal oscillation. Staging and second-stage flight were normal, and the accuracy with which the spacecraft was inserted into orbit was the best yet achieved in the Gemini Program. During the first two orbits, all spacecraft systems were checked, a nominal perigee adjust maneuver was conducted, and the rendezvous evaluation pod was ejected on schedule. The rendezvous guidance and navigation system evaluation proceeded in a satisfactory manner for about 45 minutes when the pressure in the fuel-cell oxygen supply tank decreased to a level well below the specified limit. The crew decided to power down the spacecraft and abandon the radar evaluation with the rendezvous evaluation pod at that time. Concentrated activities were begun by ground personnel to establish an operating mode that would allow continuance of the mission. It was determined that the fuel cells were receiving adequate oxygen to produce the necessary electrical power to continue the mission. From this point in the mission, the flight plan was continuously scheduled in real time to conduct experiments and other activities.

The spacecraft was flown in the powered-down configuration until revolution 7 when the spacecraft was powered up slowly to increase the electrical load. At the end of revolution 17, the spacecraft was powered up to a high load condition, and a successful rendezvous radar test was conducted by tracking a transponder on the ground at Cape Kennedy. Four radar tests were conducted during the mission to evaluate the system for rendezvous missions in lieu of the rendezvous evaluation pod exercise.

At various times during the second day, it was found feasible to approach a full-power configuration. During the third day, a simulated Agena rendezvous was conducted at full electrical load. The apogee adjust, phase adjust, plane change, and coelliptical maneuver were performed using the orbital attitude and maneuver system. It was determined from ground tracking that the simulated rendezvous would have been successful in placing the spacecraft within 0.3 mile of an Agena target vehicle. A concentrated program of operational and experiment activities was conducted throughout the third and fourth days. During the fifth, sixth, and seventh days, attitude thruster problems were encountered; however, experiment and operational activities continued to be conducted on a limited basis. These activities included such things as visual acuity tests, special communications tests, rendezvous radar tests, and cloud and terrain photography. During the last 2 days of this period, close management of the electrical load was necessary to assure adequate power to complete the mission.

The flight continued into the eighth day, the planned duration of the mission. During the latter part of the day, preparations commenced for reentry and recovery operations. The reentry control system was powered up during revolution 119 to provide attitude control in preparation for retrofire and reentry. All checklists and stowage were completed and retrofire occurred exactly on time at 190:27:43 g.e.t. for a landing in the West Atlantic Ocean, the planned landing area for revolution 121. The retrofire operation was completely nominal, and the reentry and landing were satisfactory, except that the landing point achieved was about 89 miles short of that desired. This undershoot was the overall result of incorrect navigation coordinates transmitted to the spacecraft computer from the ground network.

During the course of the mission, 16 of the 17 planned experiments were conducted. A high percentage of the desired data was realized and is being analyzed by the experimenters. Evaluation of the overall results obtained from the Gemini V mission shows that, with three exceptions, all primary and secondary objectives were met.

2.0 INTRODUCTION

A description of the Gemini V mission, as well as a discussion of the evaluation results, is contained in this report. The evaluation covers the time from the start of final countdown of the actual launch (at fueling) to the date of publication of the report. Any reference in this report to the attempted launch on August 19, 1965, is for the purpose of clarifying a particular point of interest.

Detailed discussions are found in the major sections related to each major area of effort. Some redundancy is found in various sections, but this is necessary for a logical discussion of that area.

Only selected segments of the data were reduced and evaluated because of the large amount of spacecraft telemetry data received and recorded by the ground stations during the course of the mission. The major emphasis on data reduction was in the areas of known interest. These data included data transmitted from the spacecraft, onboard recorded spacecraft data and biomedical data, and ground-based radar tracking data. In evaluating launch vehicle performance, all available data were reduced and evaluated. The evaluation of spacecraft and launch vehicle data consisted of analyzing flight test results as well as comparing them with those from ground tests and previous missions.

Section 6.1, flight control, may appear to contain certain redundancies and contradictions because the information contained in this section is based upon observations and evaluations made in real time, and consequently do not reflect the results obtained from the detailed postflight analysis. A brief description of the experiments flown on this mission with the results and conclusions is found in section 8.0.

The following objectives, as set forth in the Mission Directive, formed the basis for evaluation of the flight test and were of paramount consideration during the preparation of this report.

(a) Evaluate the performance of the rendezvous guidance and navigation system using the rendezvous evaluation pod (REP).

(b) Demonstrate manned orbital flight in the Gemini spacecraft for approximately 8 days.

(c) Evaluate the effects of exposing the two-man crew to long periods of weightlessness in preparation for missions of even longer duration.

The second-order mission objectives for the Gemini V mission were as follows:

(a) Demonstrate controlled reentry guidance to a predetermined landing point.

(b) Evaluate the performance of the fuel cell under flight electrical load conditions.

(c) Demonstrate all phases of guidance and control system operation necessary to support a rendezvous mission.

(d) Evaluate the capability of either pilot to maneuver the spacecraft, in orbit, to a close proximity with another object.

(e) Evaluate the performance of rendezvous radar.

(f) Execute 17 experiments. (See table 8-I for a list of these experiments.)

As this report is being published more detailed analyses of data on the performance of the launch vehicle and the performance of the radio guidance system are continuing. Also, analyses of spacecraft performance are continuing in the areas of performance of the inertial guidance system and performance of the rendezvous radar system.

Supplemental reports, listed in section 12.4, will be issued as required to provide a complete and detailed evaluation of the performance of the launch vehicle and certain systems of the spacecraft, and to report major anomalies not resolved at the time of publication of this report.

Results of previous Gemini missions are found in references 1 through 4.

3.0 GEMINI V VEHICLE DESCRIPTION

The space vehicle for the Gemini V mission consisted of Gemini spacecraft 5 and Gemini launch vehicle 5 (GLV-5). Section 3.1 of this report describes the spacecraft configuration, section 3.2 describes the GLV configuration, and section 3.3 provides space-vehicle weight and balance data. The major reference coordinates for the space vehicle are shown in figure 3.1-1.

3.1 GEMINI SPACECRAFT

Except for the addition of the fuel cell power system, the rendezvous radar, and the rendezvous evaluation pod (REP); the structure and major systems (see fig. 3.1-2) of spacecraft 5 were basically the same as those used for spacecraft 4; consequently, only the significant differences are described in this report (refer to table 3.1-I). Descriptions of spacecraft 5 systems are contained in reference 5, and a description of spacecraft 4 is given in reference 4.

3.1.1 Spacecraft Structure

The primary structure was of the same basic configuration as that of spacecraft 4.

3.1.2 Major Systems

3.1.2.1 Communication system.- The communication equipment was the same as that installed in spacecraft 4 except that two switches were added to the voice control center. The silence switch could be used to turn off either headset (command pilot's or pilot's) during sleep periods. The record switch permitted the flight crew to record and transmit simultaneously. This switch replaced the record position previously incorporated in the mode switch which did not permit radio transmissions.

In addition, a fourth telemetry transmitter and separate ultra high frequency (UHF) whip antenna were added for transmitting experiment data. This transmitter was installed in the equipment adapter section and, except for the operating center frequency (244.3 megacycles), was the same configuration as the real-time, delayed-time, and standby transmitters.

3.1.2.2 Instrumentation and recording system.- The instrumentation and recording system was the same as the spacecraft 4 system except for the addition of a tape recorder for use with experiments.

3.1.2.3 Environmental control system.- The environmental control system (ECS) was functionally the same as that used on spacecraft 4. The lithium hydroxide (LiOH) canister was of the long-duration flight configuration.

Drinking water was stored in two tanks located in the adapter assembly (see fig. 3.1-3(a)). These tanks were the same configuration as the orbital attitude and maneuver system (OAMS) propellant tank with the normal diaphragm installed. Each tank had a capacity of 150 pounds of water; however, tank A was serviced with 30 pounds of water and pressurized with oxygen to 8 psi, and tank B was serviced with 126 pounds of water and was pressurized in flight with water produced by the fuel cell power system. Water from the fuel-cell power system entered tank B on one side of the diaphragm and forced drinking water, prestored on the other side of the diaphragm, out of the tank (see fig. 3.1-3(b)).

3.1.2.4 Guidance and control system.- The guidance and control systems were similar to those used on spacecraft 4 except for the energizing of the platform attitude-hold mode and the addition of the rendezvous radar and the REP (see fig. 3.1-4).

3.1.2.4.1 Control system: The platform attitude-hold mode was activated in the attitude control electronics (ACE) system. The purpose of this mode was to maintain spacecraft attitude automatically, in all three axes, to within 1.1° of the platform attitude.

3.1.2.4.2 Guidance system: The rendezvous radar was mounted on the forward face of the rendezvous and recovery (R and R) section and utilized an interferometer antenna system. The purpose of the rendezvous radar is to supply range, azimuth, and elevation relative to the target vehicle during rendezvous maneuvers. The radar consisted of four dual-spiral antennas, a transmitter, a receiver, power supplies, necessary electronics for the computer, and the cabin-display and power-input interfaces.

One of the antennas is a transmitting antenna while the other three are the azimuth, elevation, and reference receiving antennas. The azimuth and elevation antennas, using the reference antenna as a common element, measure the target bearing angle. When the radar is tracking a target, the azimuth and elevation antennas rotate to follow the target's changing position. The amount these antennas are rotated is a measure of the target's relative angular displacement from the spacecraft axes. This information, combined with range, is relayed to the onboard computer for computation of rendezvous maneuvers.

The encoder was installed but not used on the Gemini V mission. It will be used on later rendezvous missions with the Agena target vehicle (ATV). The encoder allows the flight crew to transmit commands to the ATV. These commands may be used at any time after the rendezvous radar has locked on the transponder in the ATV docking adapter, and are transmitted by pulse position modulation of the radar transmission. After docking, the command message will be routed through a hardline umbilical to the ATV.

The REP simulated the ATV for the Gemini V mission. The REP contained a transponder, a dipole antenna, two dual-spiral antennas, and two flashing beacon lights all of which were similar to those to be installed in the ATV. In the ATV, the transponder and the beacon lights will obtain electrical power from the vehicle power supply, but in the REP power was supplied by two 24-volt silver-zinc batteries. The spiral antennas provided spherical coverage about the REP while the dipole provided omnidirectional coverage.

Prior to ejection, the REP was mounted in the equipment adapter section with a silvered fiber glass cover for protection from solar radiation. When the flight crew depressed the pod-eject switch, a cartridge-actuated cable cutter released two spring assemblies which ejected the cover. A second pyrotechnic system ejected the REP.

A 4000-beam candlepower rendezvous and docking light was mounted on the retrograde section of the spacecraft adapter assembly and was intended to provide a 6° cone of light for observation of the REP during the terminal phase of the planned rendezvous maneuvers with the REP.

3.1.2.5 Time reference system.- The time reference system was the same as the spacecraft 4 system.

3.1.2.6 Electrical system.- The electrical system was the same as the spacecraft 4 system except that the adapter battery module used on spacecraft 4 was replaced by a reactant supply system (RSS)/fuel cell module (see fig. 3.1-5). The fuel cell power system consisted of two separate sections which could be operated independently to convert reactants (hydrogen and oxygen) into electrical energy and water. The RSS consisted of two tanks for storing cryogenic supplies of hydrogen and oxygen, and the necessary heaters, regulators, valves, heat exchangers, and plumbing for supplying proper pressure gaseous reactants to the fuel-cell sections. The water produced by the fuel-cell sections was stored in a tank in the ECS (refer to section 3.1.2.3).

Each section of the fuel-cell power system consisted of three interconnected fuel-cell stacks. Each stack contained 32 individual fuel cells made up of two catalytic electrodes separated by a solid-type electrolyte ion-exchange membrane. A small percentage of the reactant gases was purged periodically from the fuel-cell power system to insure that impurities did not restrict reactant flow to the cells.

3.1.2.7 Propulsion system.- The propulsion system was the same as the spacecraft 4 system. (See figs. 3.1-6 and 3.1-7.)

3.1.2.8 Pyrotechnic system.- The pyrotechnic system was the same as the spacecraft 4 system with the following exceptions:

(a) The magnetometer-boom lock-release guillotine required for spacecraft 4 was not installed on spacecraft 5.

(b) Three equipment-release cable-cutter guillotines associated with experiments D-4 and D-7 were installed.

(c) The nose fairing was ejected by a pyrotechnic-driven piston, whereas, on spacecraft 4 the nose fairing was released by spring action after the retention cable was cut.

(d) The REP and its protective cover were ejected by devices similar to the horizon-sensor ejector.

3.1.2.9 Crew station furnishings and equipment.-

3.1.2.9.1 Instrument panels and controls: The basic configuration of the instrument panels and controls (see fig. 3.1-8) was the same as that used for spacecraft 4 except for the following changes:

(a) The fuel-cell power-system monitor was installed in place of the previous ammeter and voltmeter. This instrument consisted of a pressure indicator (inoperative), three dual ammeters, and an ac-dc voltmeter (see fig. 3.1-5). The ammeters monitored individual fuel-cell stack current (1A through 2C). The dc voltmeter, used in conjunction with a selector switch, displayed individual fuel-cell stack voltages, as well as common control bus, OAMS squib buses 1 and 2, main bus, and individual main battery voltages. The ac voltmeter was inoperative.

(b) A fuel-cell pressure differential (FCΔP) light for each fuel-cell section indicated a malfunction when the pressure differential between the hydrogen and oxygen or the oxygen and product water exceeded preset tolerances (see fig. 3.1-5).

(c) An annunicator panel was installed and contained the following lights: an oxygen high rate (O_2 HI-RATE) light to indicate the suit circuit should be on the high flow rate 5 minutes prior to retrofire; a reentry control system heater (RCS HTR) light to indicate the heaters should be turned on in the RCS; a recorder end-of-tape (RCDR TAPE) light to indicate the tape should be changed in the voice tape recorder; an FCΔP light to indicate that the pressure differential between the fuel-cell reactants was out of limits; and an ECS heater (ECS HTR) light to indicate that the heater in the primary breathing oxygen container had been manually activated.

(d) A radiometric selector switch panel was installed for use with experiments D-4 and D-7.

(e) A maneuver hand control was added to the right wall of the cockpit to enable the pilot also to perform spacecraft maneuvers.

(f) A range and range-rate indicator was installed for display of the target range and range-rate data provided by the rendezvous radar.

(g) An Agena control panel was installed on the right switch/circuit-breaker panel. The pod-eject switch was used to eject the REP. The docking-light switch controlled the docking light (mounted on the adapter assembly). The other switch positions on this panel will be required for future rendezvous missions and were not used for the Gemini V mission.

3.1.2.9.2 Space suit: The G4C space suits worn by the flight crew were of the same configuration as those used on the Gemini IV mission except that the overvisor and special cover layer used for extravehicular activity were not included.

3.1.2.9.3 Spacecraft stowage facilities: Containers for stowage of flight-crew equipment are shown in figure 3.1-9. Table 3.1-II lists the major items of equipment stowed in the containers at launch.

3.1.2.10 Landing system.- The landing system was the same as the spacecraft 4 system.

3.1.2.11 Postlanding and recovery systems.- The postlanding and recovery equipment was the same as that used on spacecraft 4 except that the ECS snorkel was redesigned to clear the RCS propellant-tank mounts.

TABLE 3.1-I.- SPACECRAFT 5 MODIFICATIONS

System	Significant changes incorporated in spacecraft 5 from spacecraft 4 configuration
Reentry assembly structure	No significant change
Adapter assembly structure	No significant change
Communications	Additional telemetry transmitter and UHF whip antenna added for transmitting experiment data.
Instrumentation	Additional tape recorder added for recording experiment data.
Environmental control	(a) Long-duration LiOH canister installed.
	(b) Drinking water stored in two tanks in adapter assembly (one tank used also for storing product water from the fuel cell power system).
Guidance and control	(a) Platform attitude-hold mode activated in ACE system.
	(b) Rendezvous radar installed.
	(c) REP added.
	(d) 4000-beam candlepower rendezvous and docking light added.
Time reference	No significant change.
Electrical	Fuel-cell power system replaced adapter battery module.
Propulsion	No significant change
Pyrotechnics	(a) Magnetometer-boom lock-release guillotine removed.

TABLE 3.1-I.- SPACECRAFT 5 MODIFICATIONS - Concluded

System	Significant changes incorporated in spacecraft 5 from spacecraft 4 configuration
Pyrotechnics	(b) Three equipment-release cable-cutter guillotines installed for use with experiments D-4 and D-7.
	(c) Nose fairing ejected by a pyrotechnic-driven piston instead of by spring action.
	(d) REP ejector installed.
Crew station furnishing and equipment	(a) Fuel-cell power-system monitor installed.
	(b) A second maneuver hand control was added to enable pilot to perform spacecraft maneuvers.
	(c) Range and range-rate indicator installed for use with rendezvous radar.
	(d) Annunciator panel installed.
	(e) Radiometric selector switch installed for experiments D-4 and D-7.
	(f) Agena control panel installed.
	(g) G4C space suit did not have overvisor and special cover layer required for EVA.
	(h) Additional stowage containers provided for flight-crew equipment.
Landing	No significant change.
Postlanding and recovery	ECS snorkel redesigned to clear RCS propellant-tank mounts.

TABLE 3.1-II.- CREW STATION STOWAGE LIST

Stowage area (See fig. 3.1-9)	Item	Quantity
Centerline stowage container	16-mm camera (with film magazine, 18-mm lens, and 75-mm lens)	1
	70-mm camera (with film magazine)	1
	35-mm camera back (with film cassette and film)	3
	35-mm camera (with film)	1
	Cloud top spectrometer	1
	1270-mm lens and filter	1
	200-mm lens and filter	1
	Telescope	1
	Tissue dispenser	4
Left-hand aft stowage container	Food	14 man days
Left-hand sidewall stowage containers	Pilot preference kit	1
	Humidity sensor	1
	Suit repair kit	1
	Postlanding kit assembly	1
	Urine receiver and hose system	1

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

Stowage area (See fig. 3.1-9)	Item	Quantity
	Urine sampling bag	3
	CO_2 tape	24
	3-oz drinking water bag	1
	Suit repair kit	1
Right-hand aft stowage container	Food	$3\frac{1}{3}$ man days
	Launch day urine bag	2
	Defecation device	24
	Waste container	4
	Voice recorder tape cartridges	23
	35-mm camera (with film and bracket)	1
	Inflight exerciser	1
	Personal hygiene towel	12
Right-hand sidewall stowage containers	Pilot preference kit	1
	16-mm film magazine	3
	70-mm film magazine	3
	Personal hygiene towel	12
	Vision tester bite board	2
	Dual utility cord	1

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

Stowage area (See fig. 3.1-9)	Item	Quantity
	Lightweight headset	2
	Isolation cap	1
	35-mm film cassette (with film)	6
	Photo event indicator	1
Pouch on pedestal wall	World map	1
	Map booklet	1
	3-oz drinking water bag	1
	Celestial display, mercator	1
	Celestial display, polar	1
Foot wells	Flight data book	2
	Helmet stowage bag	2

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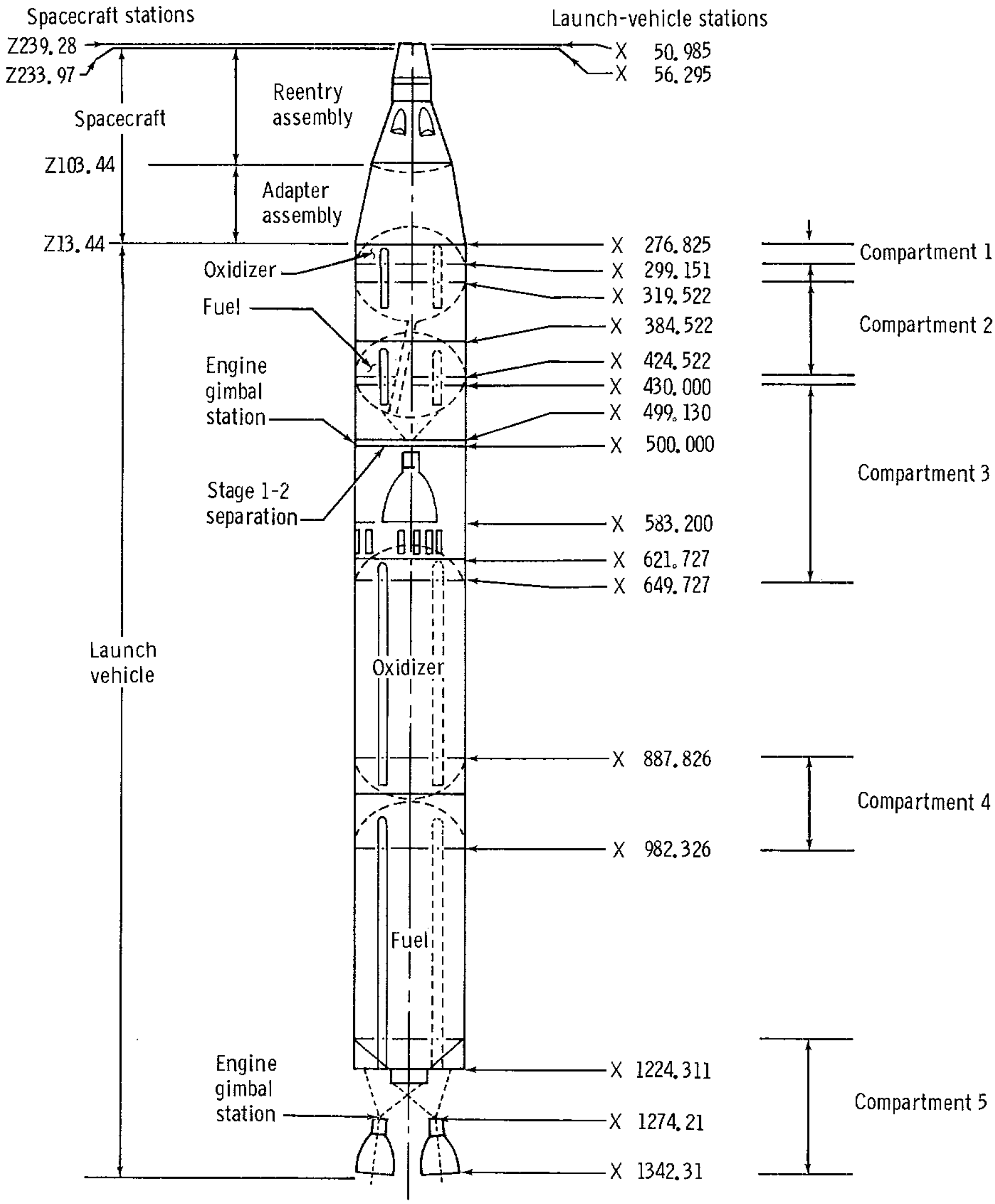

(a) Launch configuration

Figure 3.1-1. - GLV - Spacecraft relationships.

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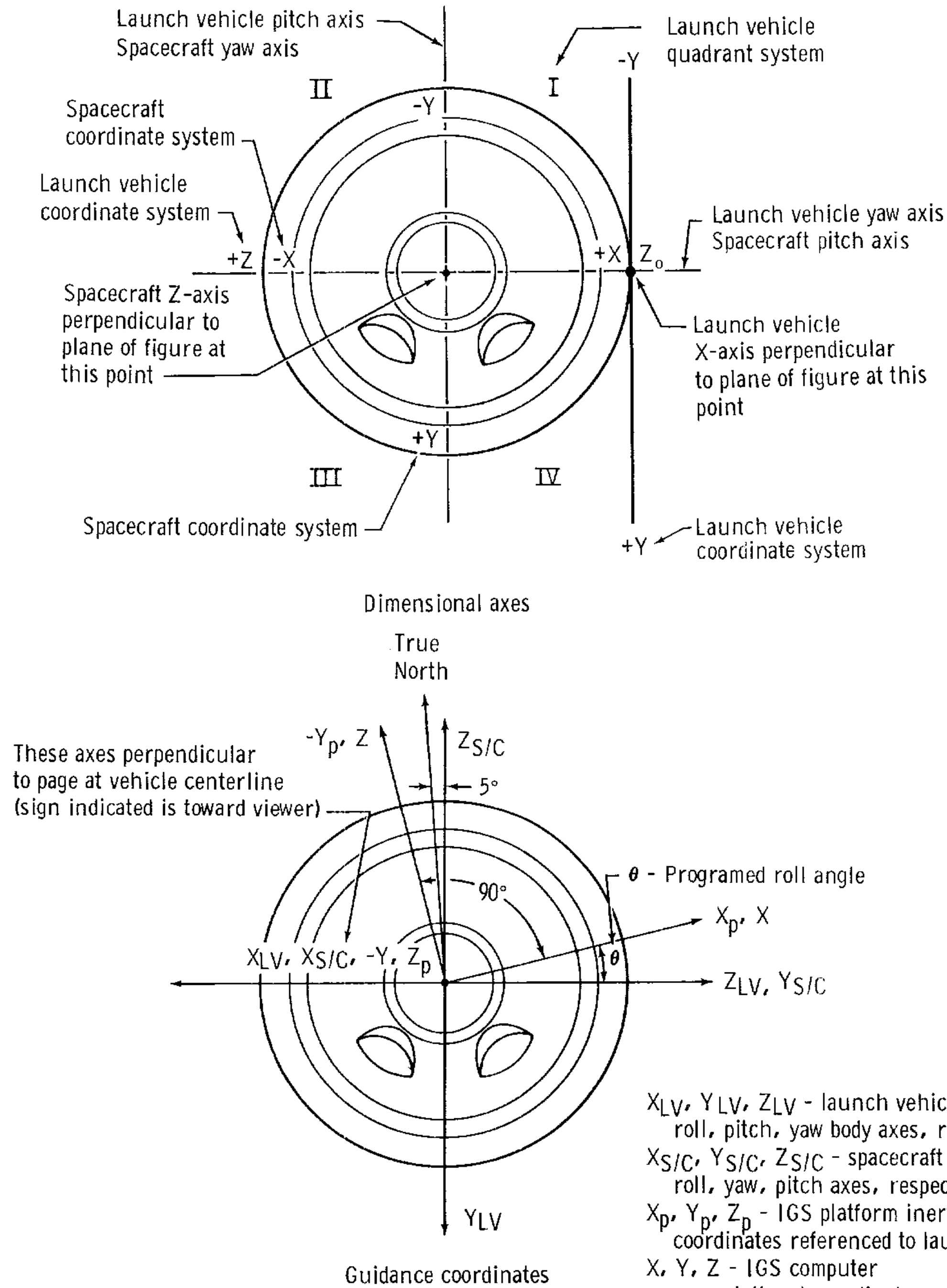

X_{LV}, Y_{LV}, Z_{LV} - launch vehicle roll, pitch, yaw body axes, respectively
$X_{S/C}$, $Y_{S/C}$, $Z_{S/C}$ - spacecraft roll, yaw, pitch axes, respectively
X_p, Y_p, Z_p - IGS platform inertial coordinates referenced to launch stand
X, Y, Z - IGS computer computational coordinates

(b) Dimensional axes and guidance coordinates
Figure 3.1-1. - Concluded.

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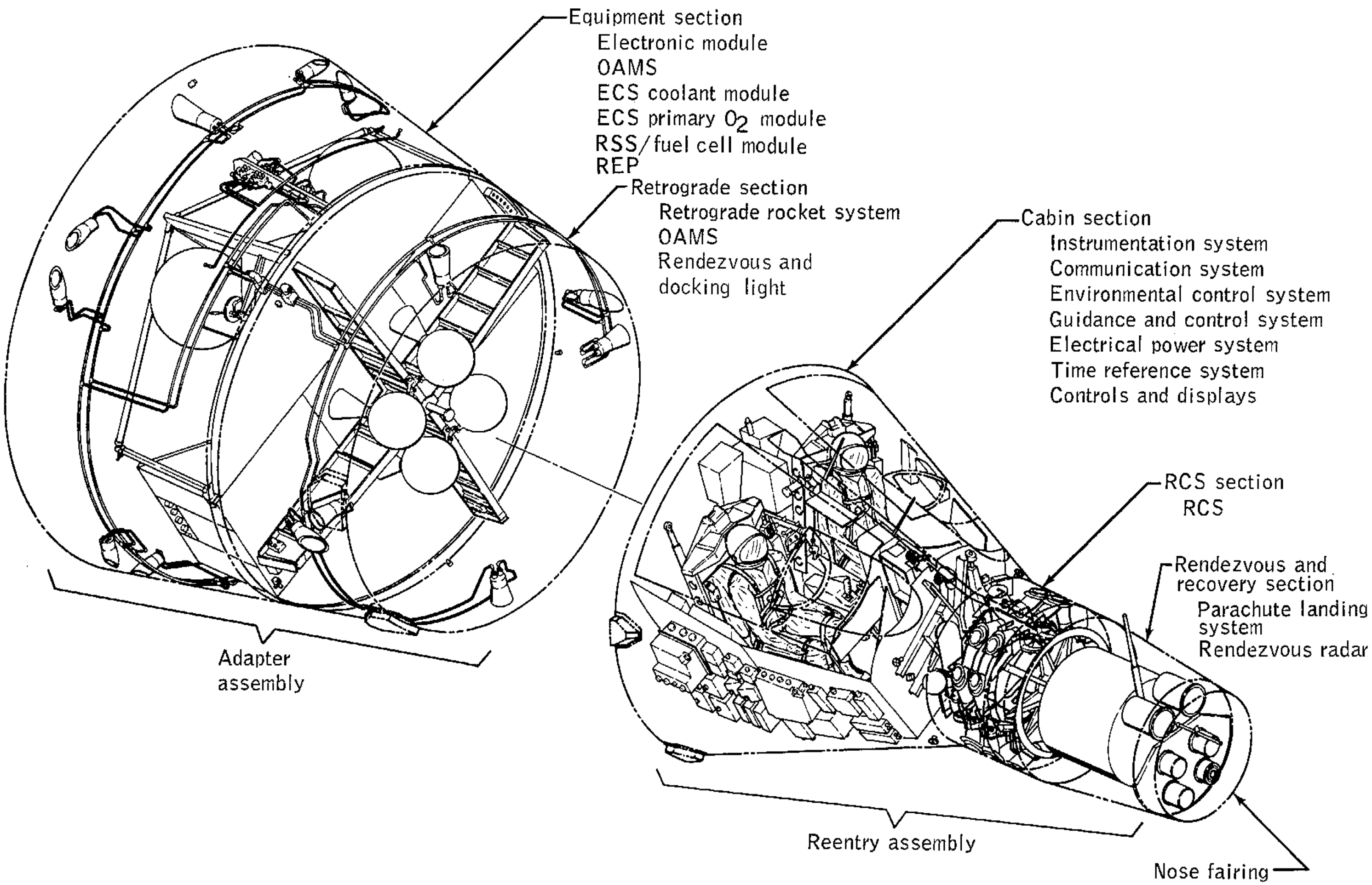

Figure 3.1-2. - Spacecraft arrangement and nomenclature.

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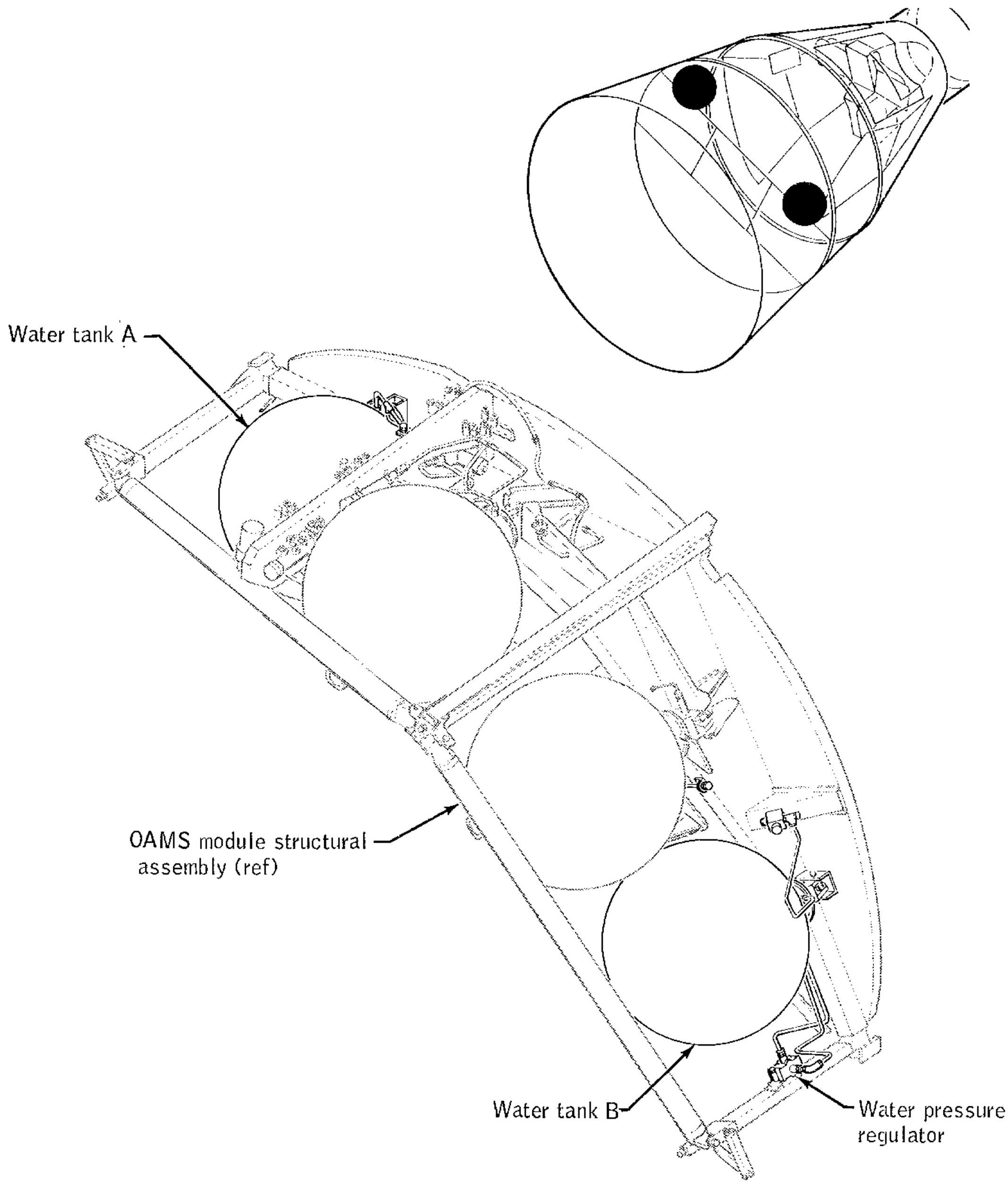

(a) Location in spacecraft

Figure 3.1-3. - Water storage system.

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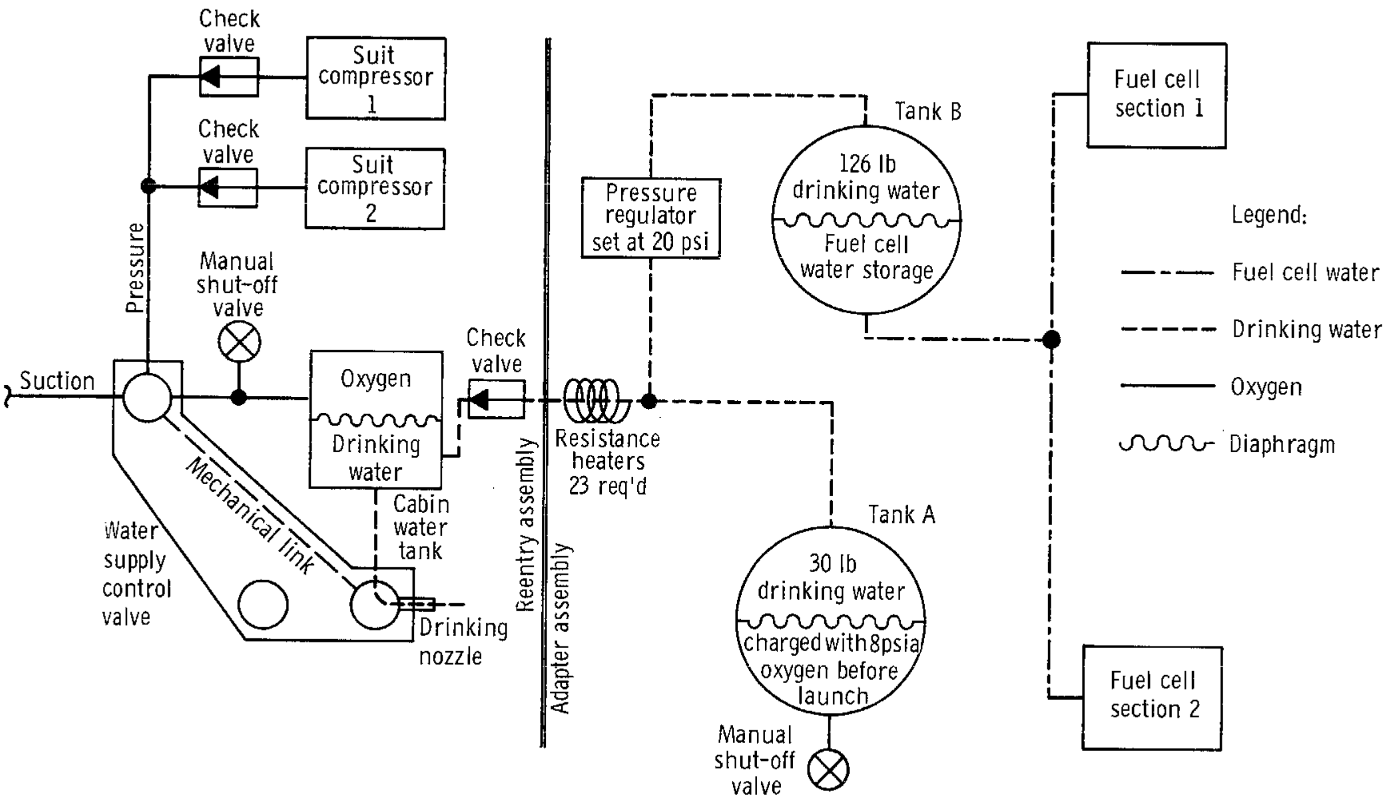

(b) Schematic of drinking water system

Figure 3. 1-3. - Concluded.

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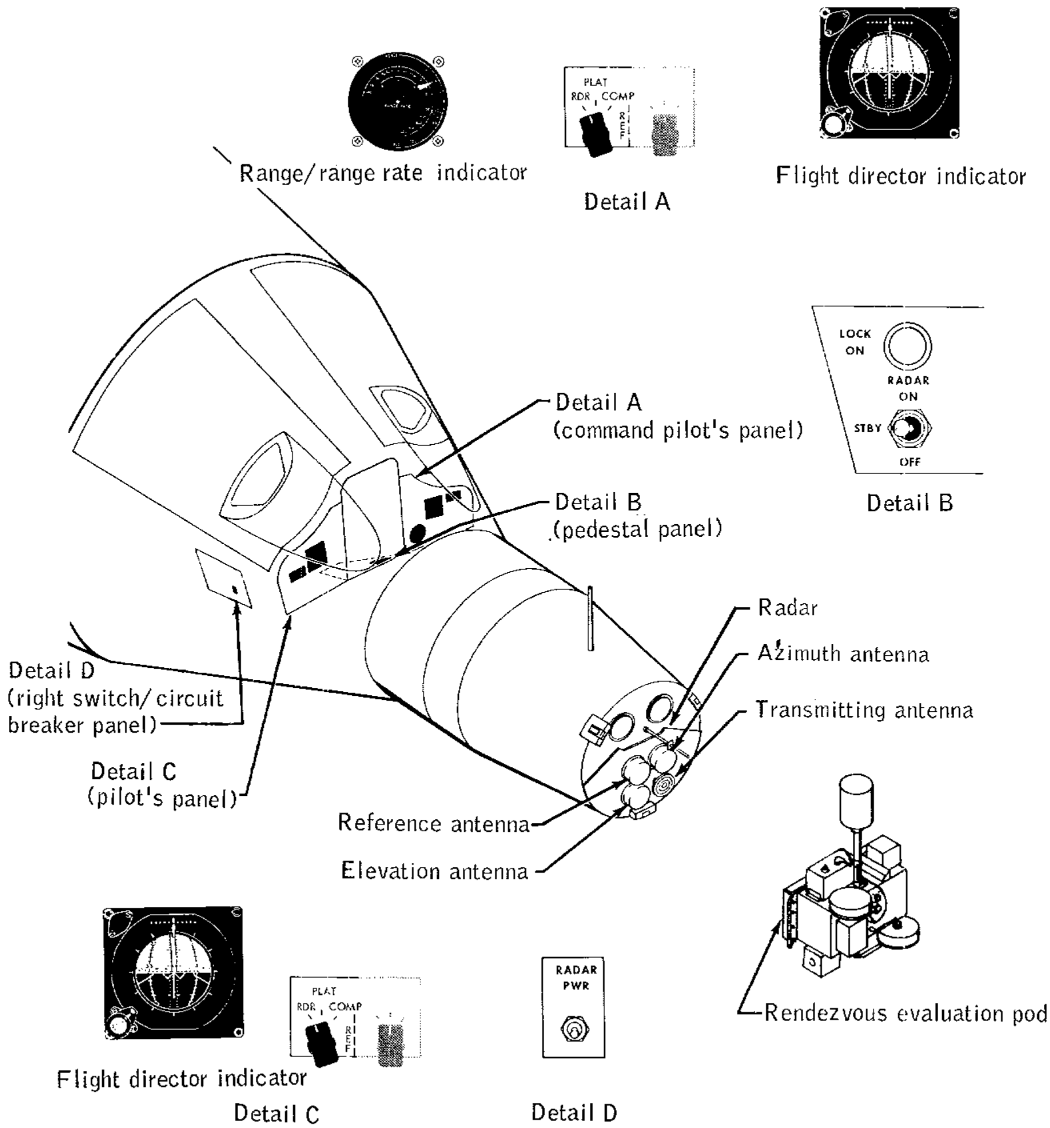

(a) System components

Figure 3.1-4. - Rendezvous radar system.

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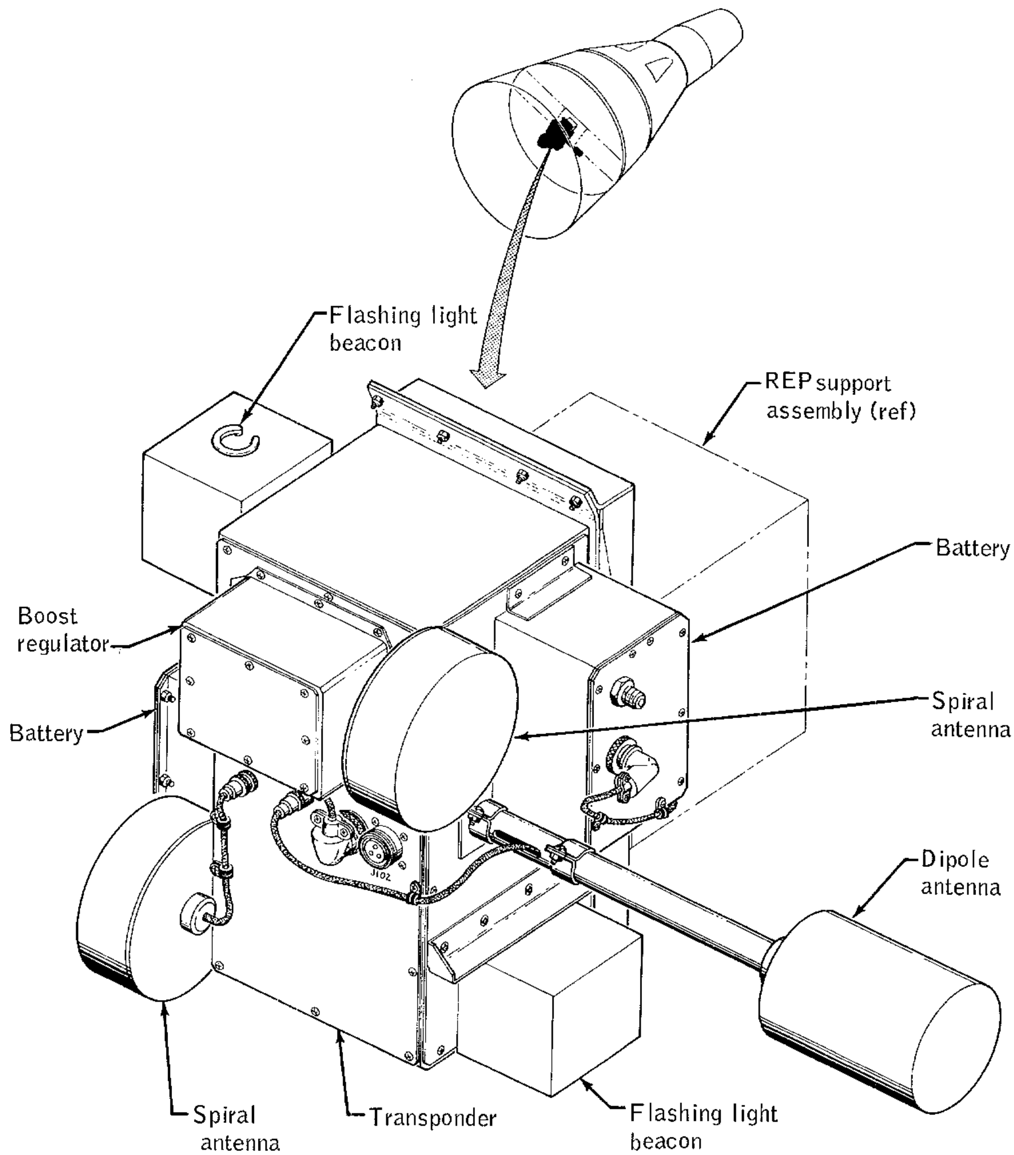

(b) Rendezvous evaluation pod

Figure 3.1-4. - Concluded.

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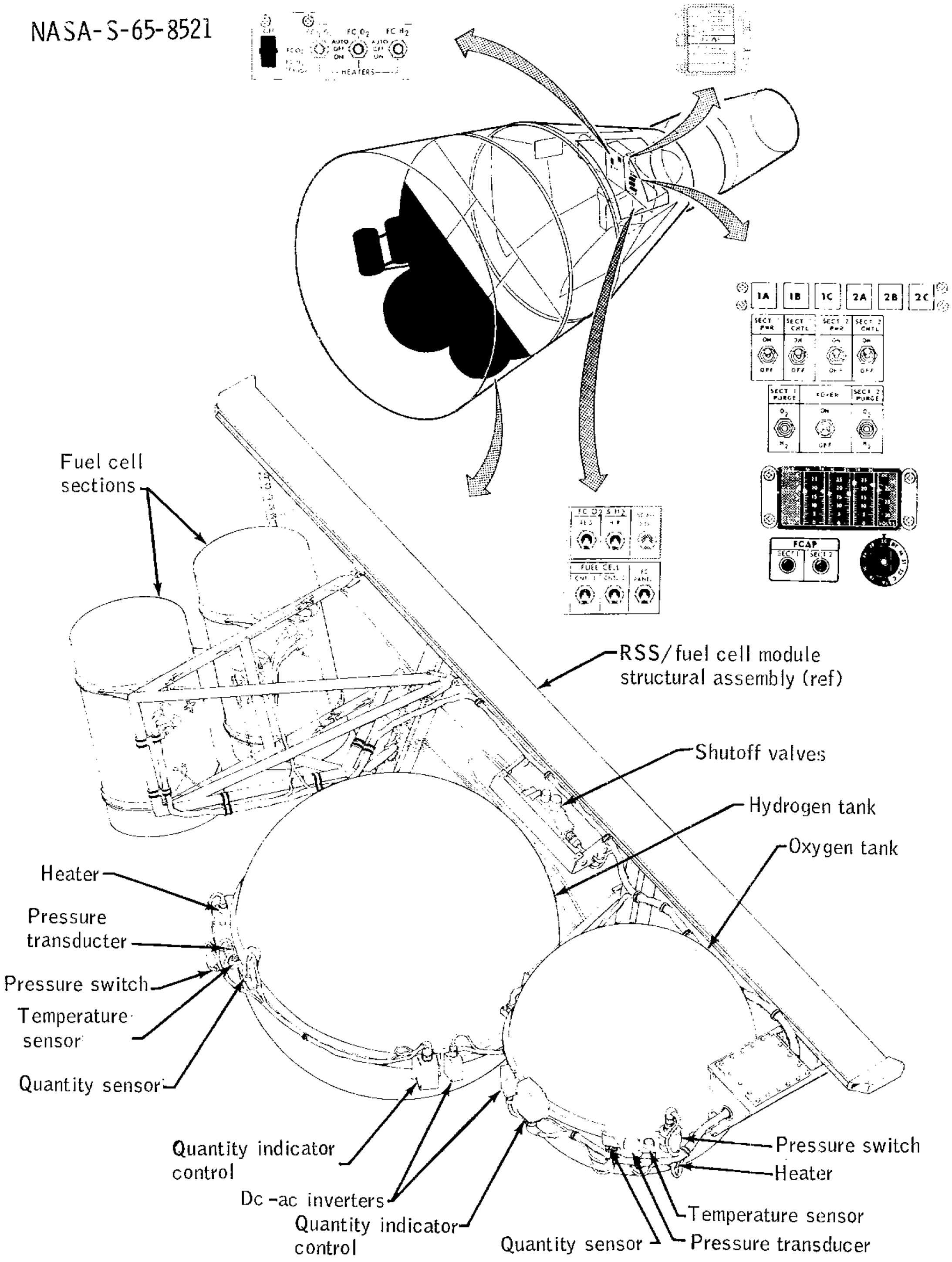

Figure 3.1-5. - Fuel cell power system.

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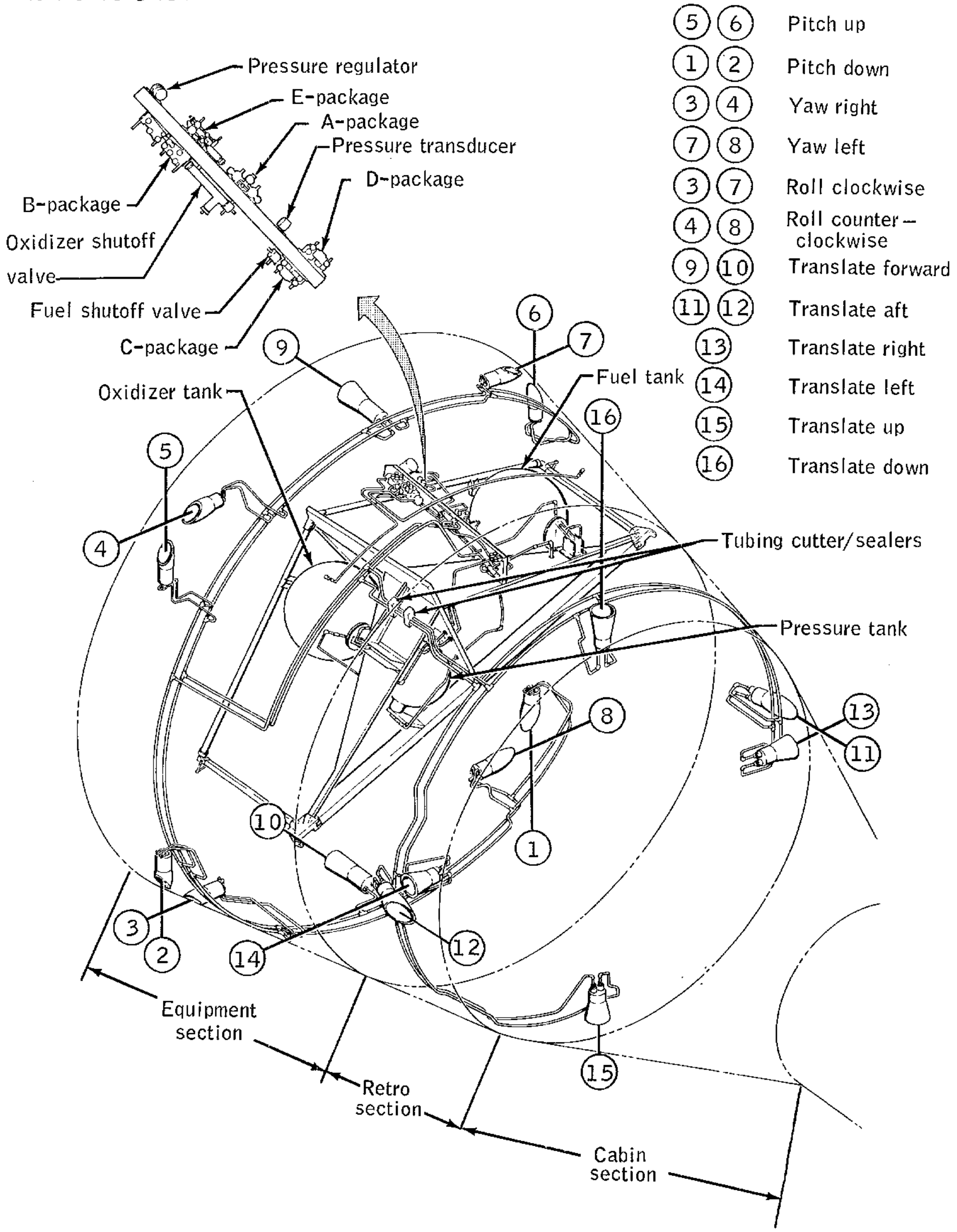

Figure 3.1-6. - Orbital attitude and maneuver system.

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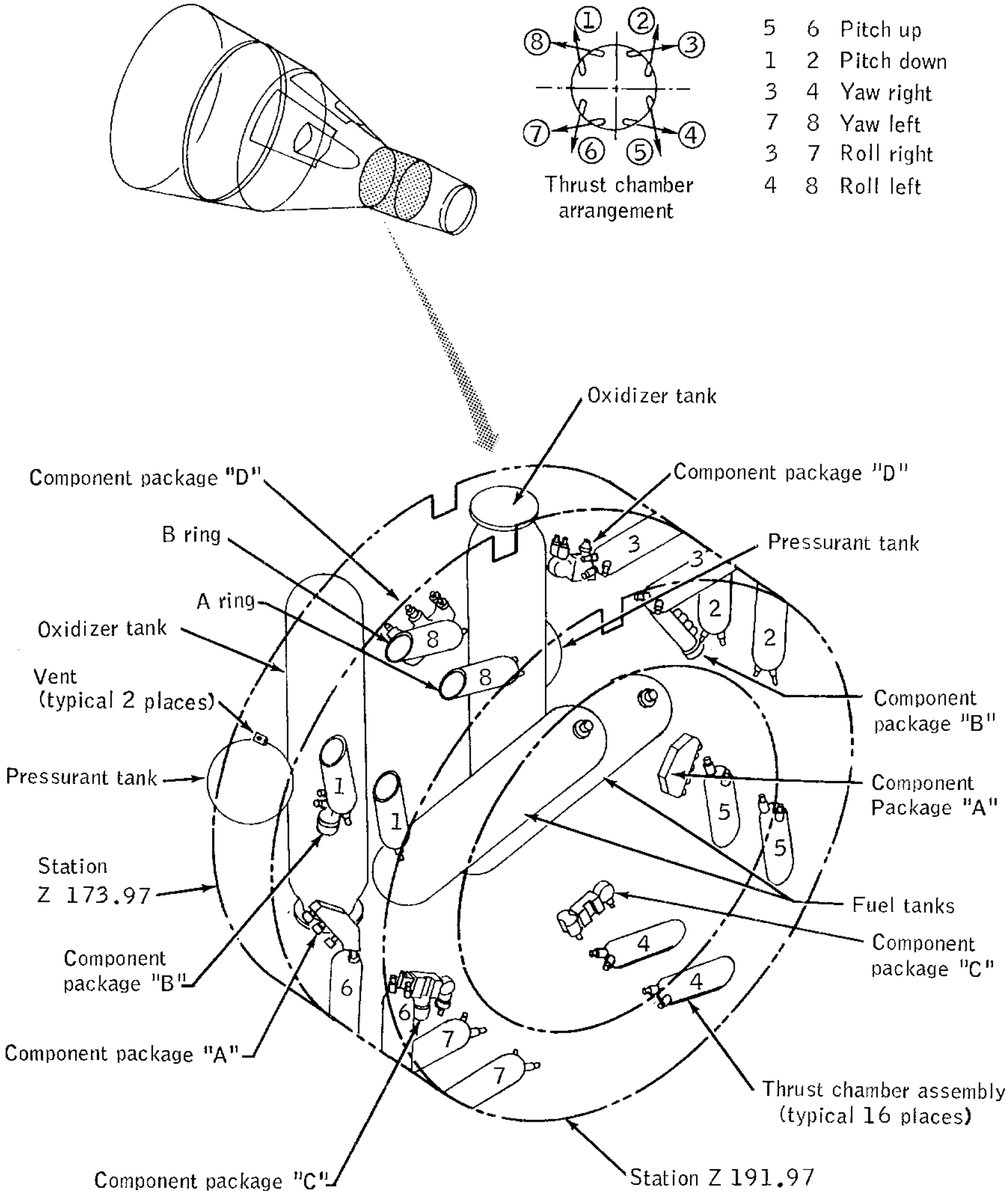

Figure 3.1-7. - Reentry control system

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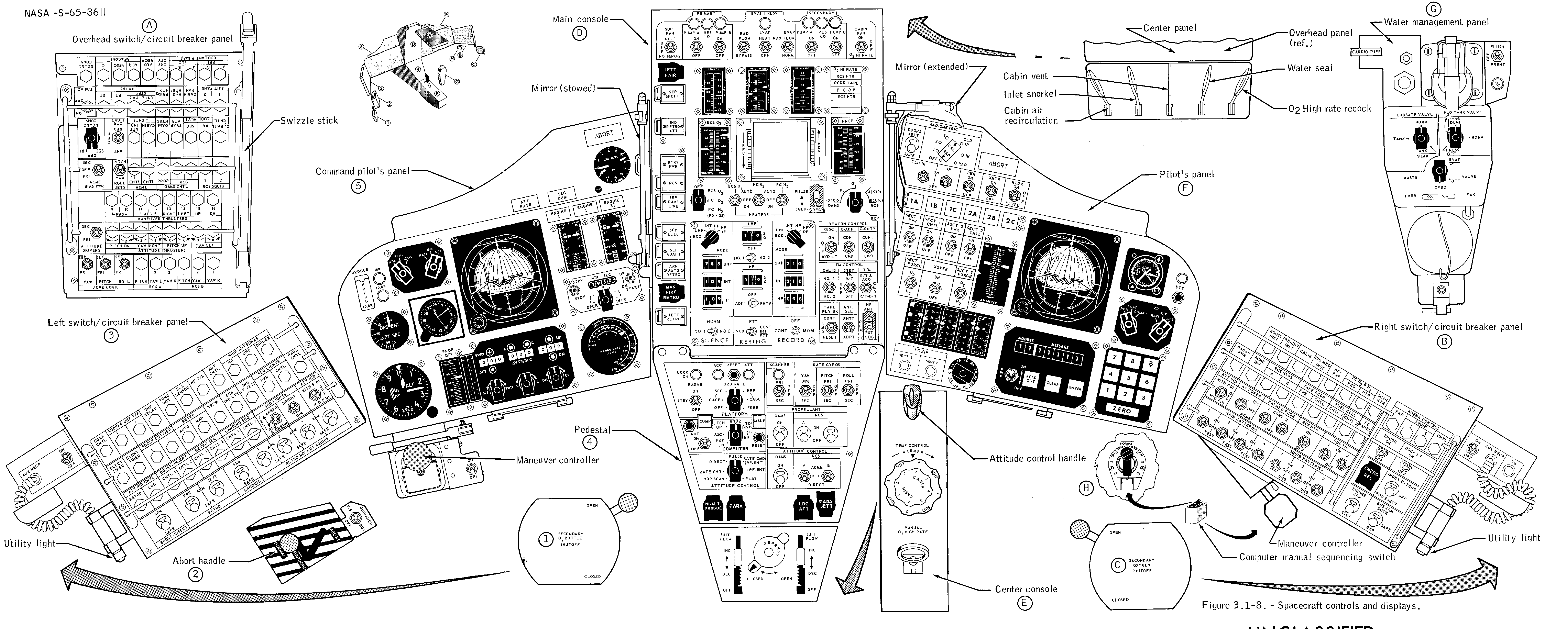

Figure 3.1-8. - Spacecraft controls and displays.

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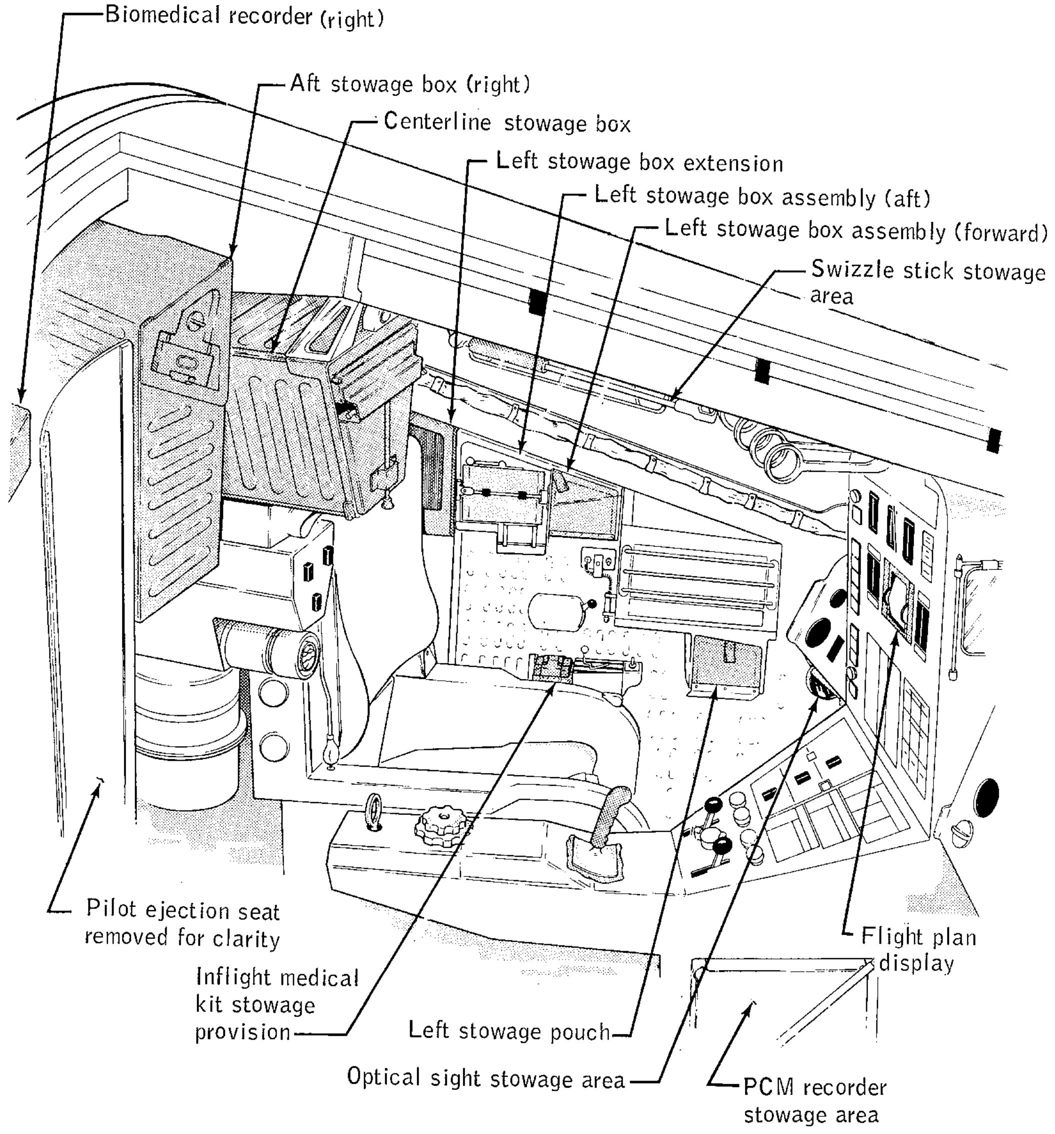

(a) View looking into command pilot's side

Figure 3.1-9. - Cabin stowage areas.

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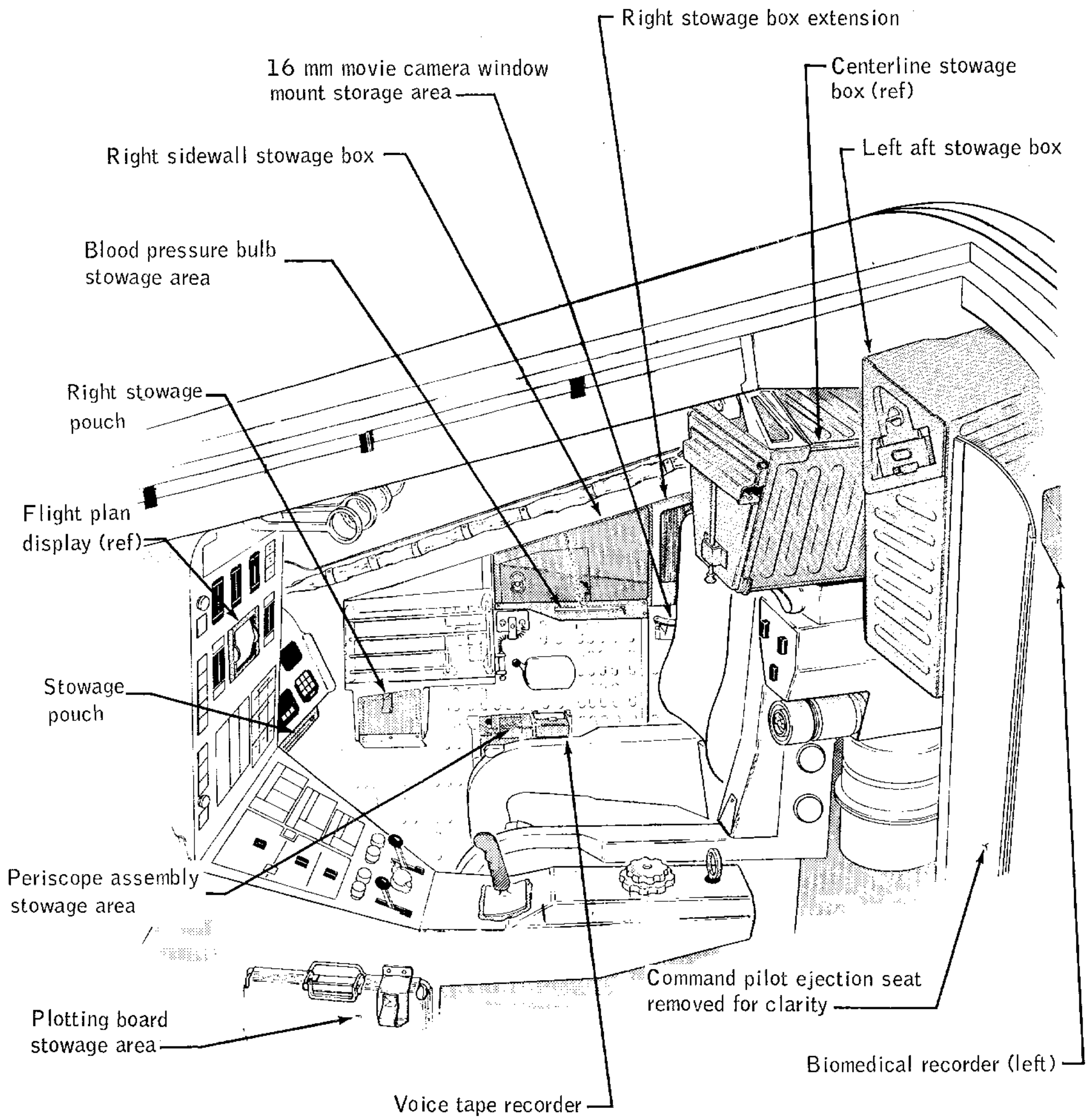

(b) View looking into pilot's side

Figure 3.1-9. - Concluded.

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3.2 GEMINI LAUNCH VEHICLE

Except for minor changes, the Gemini launch vehicle (GLV-5) was of the same basic configuration as GLV-4. Table 3.2-I lists the significant differences between GLV-5 and GLV-4.

3.2.1 Structure

The stage II fuel-tank conduit was fabricated with butt-welded circumferential joints instead of the lapped joints used on GLV-4. The supports and brackets used for the vernier engines on Titan II missiles were removed from GLV-5. The compartment 3 air-conditioning provisions (doubler and skin cutout) were deleted. Two sound pressure level microphones were removed.

3.2.2. Major Systems

3.2.2.1 Propulsion system.- The redundant high-level sensors were removed from the propellant tanks.

3.2.2.2 Flight control system.- The flight control system was the same as the GLV-4 system.

3.2.2.3 Radio guidance system.- The radio guidance system was the same as the GLV-4 system.

3.2.2.4 Hydraulic system.- A hold/kill pressure setting of the pressure switch in the secondary hydraulic system was changed from 2800 psi to 2500 psi.

3.2.2.5 Electrical system.- The flashing beacon light system used in the station-keeping exercise during the Gemini IV mission was not installed on GLV-5. All spare wires and "pigtail" leads were omitted from electrical connectors, relays, and motor-driven switches.

3.2.2.6 Malfunction detection system.- The malfunction detection system was the same as the GLV-4 system.

3.2.2.7 Instrumentation system.- The FM/FM telemetry system, the airborne tape recorder, and 38 PCM and FM measurements (transducers, wiring, and associated brackets) were removed.

3.2.2.8 Range safety system.- The range safety system was the same as the GLV-4 system.

3.2.2.9 Ordnance system.- The ordnance system was the same as the GLV-4 system.

TABLE 3.2-I.- GLV-5 MODIFICATIONS

System	Significant changes incorporated in GLV-5 from GLV-4 configuration
Stage I structure	No significant change.
Stage II structure	(a) Supports and brackets for vernier engines removed from stage II fuel tank aft skirt.
	(b) Compartment 3 air-conditioning provisions (doubler and skin cutout) deleted.
	(c) Oxidizer feed line conduit circumferential welds changed from lap weld to butt weld.
	(d) Two sound pressure level microphones deleted.
Propulsion	Redundant high level sensors removed from propellant tanks.
Flight controls	No significant change.
Guidance	No significant change.
Hydraulics	Secondary system pressure switch setting changed from 2800 psi to 2500 psi.
Electrical	(a) Flashing beacon light system deleted.
	(b) Spare wires and "pigtail" leads removed from connectors, relays, and motor-driven switches
Malfunction detection	No significant change
Instrumentation	FM/FM telemetry system and airborne tape recorder deleted.
Range safety	No significant change.
Ordnance	No significant change.

3.3 GEMINI V WEIGHT AND BALANCE DATA

Weight data for the Gemini V space vehicle are as follows:

Condition	Weight (including spacecraft), lb (a)	Center-of-gravity location, in. (b)		
		Y	Z	X
Ignition	344 685	0.0	-0.1	776.4
Lift-off	341 163	.0	-.1	776.7
Stage I burnout (BECO)	84 675	-.4	-.1	442.8
Stage II start of steady-state combustion	72 699	-.09	-.16	344.1
Stage II engine shutdown (SECO)	13 633	-.5	-.6	291.0

[a]Weights obtained from Aerospace Corporation.

[b]X-axis referenced to GLV station 0.00 (see fig. 3.1-1). Y-axis referenced to buttock line 0.00 (vertical centerline of the horizontal vehicle). Z-axis referenced to waterline 0.00 (60 inches below the horizontal centerline of the horizontal vehicle).

Spacecraft weight and balance data are as follows:

Condition	Weight, lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Launch, gross weight	7947.17	-0.71	0.66	105.61
Retrograde	5549.20	.33	-1.61	129.57
Reentry (0.05g)	4733.79	.19	-1.55	136.43
Main parachute deployment	4355.02	.18	-1.65	129.96
Touchdown (no parachute)	4244.75	.19	-1.71	127.87

[a]Z-axis reference was located 13.44 inches aft of the launch-vehicle-spacecraft mating plane (GLV station 290.265). The X- and Y-axes were referenced to the centerline of the vehicle.

4.0 MISSION DESCRIPTION

4.1 ACTUAL MISSION

A comparison of the planned and actual mission is shown in figure 4.1-1. Lift-off of the Gemini V mission occurred on August 21, 1965, at 13:59:59.518 G.m.t., 0.482 second earlier than planned. The vehicle rolled at the planned rate and to the planned flight azimuth. The flight profile was well within the 3σ trajectory boundary; however, the first-stage flight was slightly lofted because of low-pitch program rates, headwinds, and the first-stage thrust being higher than expected.

Staging was initiated at LO + 153.6 seconds, and separation had begun by LO + 154.3 seconds, approximately 1.3 seconds earlier than predicted. The stage II thrust was slightly higher than nominal, and, as in stage I, engine shut-down occurred earlier than predicted. The lofted trajectory was corrected by steering commands from the radio guidance system (RGS). The RGS steering rates experienced a slight oscillation in pitch due to noise in the radar data. (See section 5.2.5.)

Spacecraft separation from the launch vehicle occurred 23.6 seconds after SECO. Separation was smooth with low angular rates. The aft-firing thrusters provided a velocity increment of 7.6 ft/sec. The orbital path, resulting from the launch vehicle insertion conditions plus the separation thrust, had a perigee of 87 nautical miles and an apogee of 189 nautical miles.

After separation, the flight crew completed the actions listed in the insertion checklist and prepared the equipment necessary for the rendezvous evaluation pod (REP) exercise and experiments. At 56 minutes g.e.t., the perigee adjust maneuver was performed which provided a velocity increment of 9.7 ft/sec. The orbit resulting from this maneuver had a perigee of 92 nautical miles and an apogee of 189 nautical miles. After the perigee adjust, the flight crew continued to prepare for the REP exercise, performed a radar verification test, and conducted other system checks.

Just prior to REP ejection, as the final platform alinement was being made, the crew reported that the flight director indicator (FDI) skewed off. This was about 30 seconds before planned REP ejection and necessitated a return to platform cage and a quick realinement. The crew expressed uncertainty as to the quality of this quick alinement because they thought that the primary horizon sensor had possibly caused a problem.

The REP ejection was commanded at 02:07:15 g.e.t. and was followed by spacecraft turnaround and radar lock-on. The REP appeared to move nearly straight out from the spacecraft (270° indicated). Radar tracking continued. After reaching a maximum range of 7721 feet, the REP moved behind and above the spacecraft. Before the REP crossed behind the spacecraft, the crew noticed that the fuel-cell cryogenic oxygen supply pressure was falling. The pilot cycled the heater switch and circuit breaker several times, but he was unable to correct the situation. This fall in pressure occurred just prior to the Carnarvon pass on the second revolution. The pressure continued to fall, necessitating power-down of the spacecraft equipment and the termination of the REP exercise.

For the next several hours (until time for the revolution 6, area 4 landing) the spacecraft was in drifting flight, and only that equipment which was absolutely necessary remained on. During this time, specialists on the ground undertook a concentrated investigation of the problem, and the flight planners quickly devised plans for an alternate REP exercise in case the power could be brought back on. (See section 5.1.7.2 for a description of the fuel-cell cryogenic oxygen-supply problem.)

At approximately 5 hours g.e.t., the section 2 fuel cell and the secondary coolant loop were turned off line (operating open circuit with no coolant flow). Mission Control Center, Houston (MCC-H), decided to monitor the oxygen pressure until time for the revolution 6, area 4 landing-decision point, which was about 2 hours away; and if the pressure stabilized by that time with satisfactory electrical power, the flight was to be continued. The digital command system (DCS) update to the spacecraft computer for revolution 6, area 4 landing, was sent from Texas on the fourth revolution, and MCC-H continued to monitor the situation.

During the next revolution, it was decided that the pressure had stabilized, and the flight could continue in the powered-down configuration. During the Hawaii pass on the fifth revolution, a decision was reached to continue toward a revolution 18, area 1 landing. Drifting flight continued until the situation was thoroughly understood, and during the Hawaii pass on the seventh revolution, a powering-up procedure was started. As spacecraft equipment was powered up, the fuel-cell oxygen pressure was monitored very closely; and because no problems were encountered, the flight plan was again altered to include certain experiments and systems checks which required more and more power.

At 20 hours g.e.t., the flight crew was asked to concentrate on their sleep schedules, since, because of their concentrated activity, they were behind on their total hours of sleep.

Reentry control system (RCS) temperature warning lights begin illuminating at the end of the first day. Activation of the RCS heaters for short periods of time extinguished the lights, but they continued to come on. Finally, the heaters were left on under control of the thermostats. At the end of the first day, the section 2 fuel cell and the secondary coolant pump were brought back on line.

During the second day, the flight plan was continually altered in order to reschedule various experiments and systems tests, eating, and sleeping, and to conform to work preparation periods. Two radiometry runs were made early in the day, and a number of operational photographs were taken. A radar test was performed during the pass over Cape Kennedy on revolution 17. Lock-on occurred at 27:04:02 g.e.t. and the readouts were good. Photographs of a large storm were taken at the beginning of revolution 18, and two sequences of cloud-top spectrometer readings were made at about 30 hours g.e.t. Several vision tests and other experiments were run during the second day. Other activities included fuel-cell purges, planned landing-area updates, cryogenic quantity readouts, medical passes, and flight-plan rescheduling. A REP exercise was considered for the second day but was cancelled because it required lowering perigee, which would have reduced the orbital lifetime.

Equipment problems encountered during this day included erratic operation of the primary horizon sensor and an apparent malfunction of the optical sight. (See section 5.1.5.3.2 for sensor problems and section 5.1.10.4.4 for an explanation of the optical sight malfunction.)

During the third day, a series of four maneuvers was performed to simulate the Gemini VI rendezvous maneuvers. The first two maneuvers were performed using entries into the airborne computer from ground control via the digital command system, and the last two were performed using entries by the crew through the manual data insertion unit (MDIU). The first maneuver was a height adjustment performed blunt-end-forward (BEF) at 50:49:57 g.e.t. to lower apogee. Aft-firing thrusters were used for these maneuvers, because it was thought that a two-phase condition existed in the fuel-cell oxygen supply tank, and that a sustained maneuver with the forward-firing thrusters would allow gas to be extracted at a high rate with an accompanying large decrease in pressure.

A photographic sequence was attempted after the height-adjust maneuver, and the crew was able to acquire objects visually but could not find them in the boresighted aiming telescope or in the reflex viewfinder in the camera, primarily because of a malfunction in the power to the reticle of the optical sight at the end of the second day. (See section 5.1.10.4.4.) At 51 hours 20 minutes g.e.t., the platform was alined in preparation for the second simulated rendezvous maneuver which was a phase adjustment. The maneuver was initiated at 51:34:31 g.e.t. and was performed small-end-forward (SEF) using the platform mode. This mode was previously used for the perigee adjust maneuver during the first revolution and with good results; however, the crew reported out-of-plane components during this maneuver. (See table 5.1.5-VI for accuracy.)

At approximately 51 hours 50 minutes g.e.t. the platform was alined SEF in preparation for the third maneuver which was out of plane (yaw left 90°). The maneuver consisted of a 15 ft/sec velocity increment performed in the rate command mode and was very close to nominal. A visual acuity experiment sequence was obtained after this maneuver.

The final maneuver of the simulated rendezvous sequence was made on the third day at 53 hours 4 minutes g.e.t. This was a coelliptic maneuver of 17.2 ft/sec and was made SEF.

The spacecraft was powered down after the rendezvous maneuvers and remained down until very near the end of the third day. During this period two photographic sequences were obtained, along with an Apollo landmark run, a cabin lighting survey, an electrostatic charge (plasma measurement) experiment run, two visual acuity sequences, and one human otolith experiment run. A radiometry experiment run was also possible because the optical sight had been repaired.

The fourth and fifth days included various experiment runs, fuel-cell purges, planned-landing-area updates, systems tests, and other necessary activities. A zodiacal-light photographic run was made at the beginning of the fourth day and, at 74 hours 40 minutes g.e.t., the crew tracked a Minuteman missile being launched from the Air Force Western Test Range. A radar test and two platform tests were made early in the fourth day. The platform tests were in conjunction with the primary-horizon-sensor problem encountered earlier. A visual acuity run occurred in revolution 48, in which the crew was able to see smoke at the Laredo site and make several experiment sightings. There was considerable usage of fuel during this pass over the United States, and after the pass, an onboard quantity readout showed about 29-percent fuel remaining.

A sequence of photographs of nearby objects was attempted about halfway through the fourth day, but it was unsuccessful because the platform was not up at the time. On the evening of the fourth day, the pilot requested that activity be kept to a minimum to allow the crew some uninterrupted sleep.

Early in the fifth day, five radiometry sequences were made of sled runs at the White Sands Missile Range, and a visibility test of a ship was performed. A radiometry sequence of a missile launch during revolution 62 was attempted; however, the crew was able to see the missile but could not track it continuously.

During the middle of the fifth day, the crew's activities consisted of numerous experiments and systems tests including a special rendezvous radar test at about 117 hours g.e.t. At approximately 118 hours g.e.t., the crew reported that the orbital attitude and maneuver system (OAMS) was sluggish and thruster 7 was inoperative. All experiments requiring fuel were cancelled, and the spacecraft was powered down. During the next several hours, various fixes for the OAMS were tried, but none were successful. Late in the fifth day, it became apparent that the low OAMS fuel quantity and the remaining fuel-cell water storage would require close management in order to complete the planned 8-day mission.

Early in the sixth day, attitude thruster 8 became inoperative, and the rest of the system was gradually becoming more erratic. The spacecraft remained in drifting flight, and the thrusters were used only for damping when the spacecraft rates became excessive. Occasionally the spacecraft would be in the right attitude at the right time, and an experiment could be performed as planned.

Section 2 fuel cell was again powered down at 123 hours 20 minutes g.e.t. in order to conserve hydrogen and minimize water production. The crew continued to perform thruster tests but were unable to determine the cause of the failures associated with thrusters 7 or 8. Further attempts to clear the OAMS were unsuccessful. At the end of the sixth day, the attitude thrusters that were still operating were causing cross-coupling because of the unsymmetrical degradation of thrust between pairs.

A ground radar interference test was run during revolution 93, and no interference occurred.

A drifting mode of flight continued through the seventh day with an occasional power up for rate damping and a few experiments. The thrusters continued to degrade. (Refer to section 5.1.8.1.3 for a detailed description of the attitude thruster problem.)

Fuel-cell hydrogen stopped venting at the beginning of the eighth day, thus rate build-ups ceased. The laser experiment over White Sands Missile Range (WSMR) was attempted, and another visual acuity sequence was performed over Laredo, Texas. Two short fuel-cell tests were conducted at about 186:57:00 g.e.t. and 187:31:00 g.e.t. in an attempt to determine the capability of section 2 to carry a heavy load after being operated open-circuit for extended periods of time. (See section 5.1.7.1 for detailed performance of the fuel cells.)

The preretrofire checklist was performed starting at about 20 hours into the eighth day. Rate gyros and computers were turned on, and RCS

A- and B-rings were actuated. The platform was alined using the RCS system (A-ring only) and a very good alinement was performed.

On revolution 120 over the United States, a DCS update was sent and verified, and preparations for retrofire continued. Because of a weather condition, a decision was made to use revolution 121, area 1 landing area, instead of revolution 122, area 1.

Prior to the Carnarvon pass, final tracking data showed the DCS update was off in retrofire time and a decision was made to correct the error with an update from Carnarvon. When this one was sent, the pilot reported that the message acceptance light did not illuminate. (See section 5.1.10.2.2.) Key memory cores were then checked and found to be correct, validating the update.

Retrofire occurred in darkness at an elapsed time of 190:27:43 g.e.t.; both RCS rings were on during this sequence. RCS B-ring was turned off after retrofire and was not turned back on until approximately 65 000 feet. The command pilot stayed in single-ring pulse mode until 400 000 feet, then switched to single-ring direct mode until the spacecraft reached 260 000 feet, at which time single-ring rate command mode was selected for the remainder of the reentry.

The command pilot held the spacecraft at full-lift to 400 000 feet and rolled the spacecraft to 53° at guidance initiate. The spacecraft computer had received incorrect initial navigation coordinates for reentry because of omitting a term in the ground computer entry. The overall effect of these incorrect coordinates was a spacecraft landing approximately 89 miles short and 17 miles off track of the planned landing point.

At guidance initiate, the FDI indicated an off-scale overshoot; however, the cross range indicator was indicating in the expected manner. The command pilot correctly analyzed the guidance system performance and banked the spacecraft toward the desired track at 90° (zero-lift) in an attempt to shorten the indicated range and get closer to the desired track. When the downrange error display did not respond, the command pilot returned to the backup bank-angle technique and flew this reentry until drogue parachute deployment which occurred at 69 000 feet. (Section 5.1.5.2.3 provides a detailed description of reentry, and section 6.2.2.2.1 includes a discussion of the incorrect coordinates which were transmitted to the spacecraft.) The RCS propellant valves were shut off at 30 000 feet, and the main parachute deployment was initiated at 10 600 feet. The main parachute opened in the reefed condition and disreefed at the required time. Shortly thereafter, the command pilot actuated the necessary circuitry to reposition the spacecraft to the two-point suspension attitude. Post main checklists were completed, and a very soft water landing occurred at 190:55:14 g.e.t. Recovery of the flight crew was effected in a nominal manner, and they arrived aboard the recovery ship at 192 hours 26 minutes g.e.t. The crew was found to be in excellent physical condition during the preliminary medical examination. The flight was successfully completed at 194 hours 50 minutes g.e.t. when the spacecraft was hoisted on board the U.S.S. Lake Champlain, the prime recovery ship. The crew spent the succeeding days in extensive medical examinations, technical debriefings, consultations with the Mission Evaluation Team concerning the launch vehicle and spacecraft systems, and debriefings with the experimenters. The mission was completed at the end of these activities on September 9, 1965.

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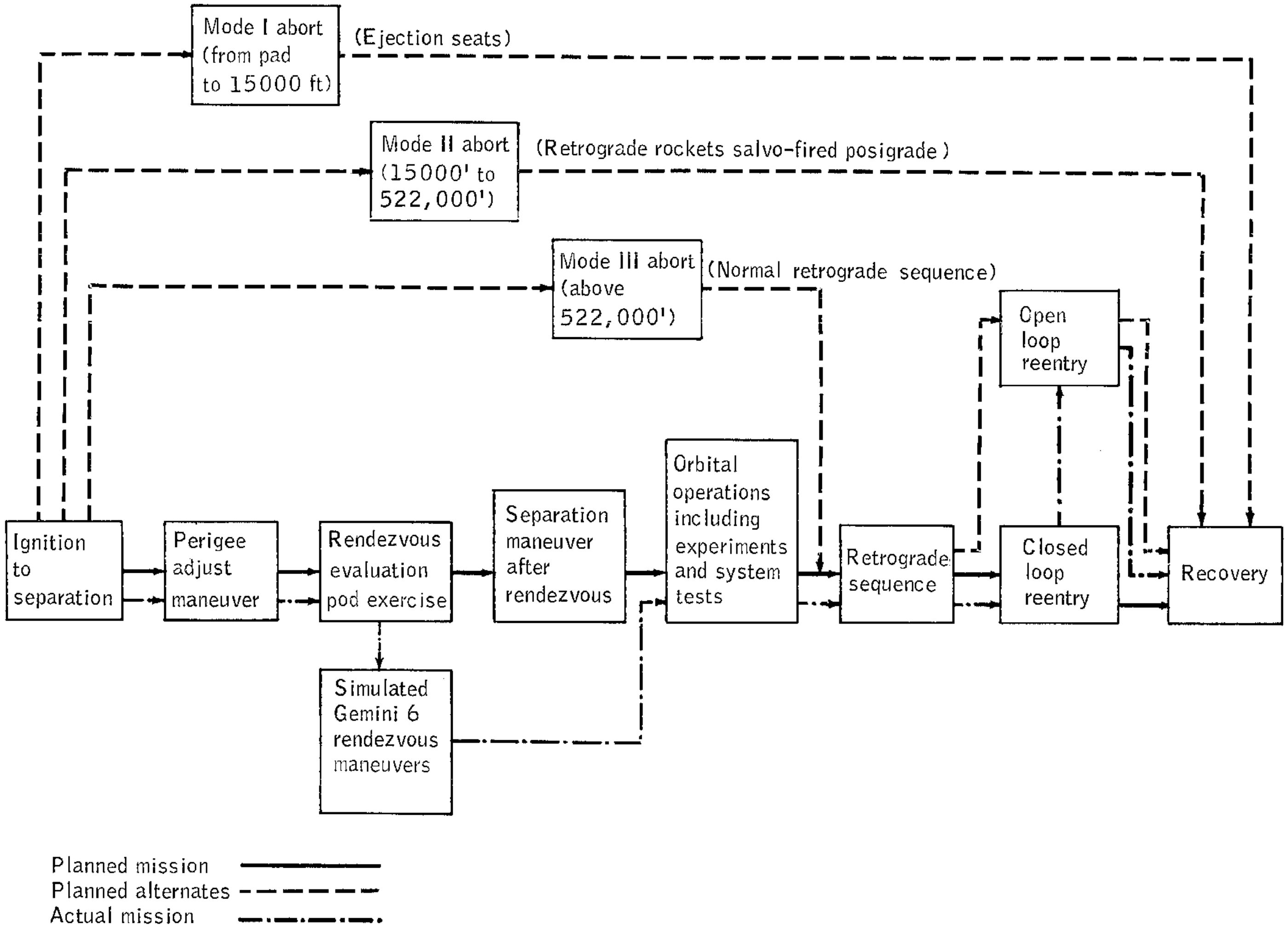

Figure 4.1-1. - Planned and actual mission with planned alternates included.

4.2 SEQUENCE OF EVENTS

The times at which major events were planned and executed are presented in table 4.2-I. All events were completed as scheduled and were within the expected tolerances, indicating a satisfactory flight.

TABLE 4.2-I.- SEQUENCE OF EVENTS

Event	Planned time, g.e.t.	Actual time, g.e.t.	Difference, sec
Launch phase, sec			
Stage I engine ignition signal (87FS1)	-3.40	-3.33	.07
Stage I MDTCPS makes subassembly 1	-2.30	-2.41	-0.11
Stage I MDTCPS makes subassembly 2	-2.30	-2.36	-0.06
TCPS subassembly 1 and subassembly 2 make	-2.20	-2.30	-0.10
Lift-off (pad disconnect separation) (13:59:59.518 G.m.t.)	0	0.00	0.00
Roll program start	10.16	10.13	-0.03
Roll program end	20.48	20.45	-0.03
Pitch program rate no. 1 start	23.04	23.09	0.05
Pitch program rate no. 1 end, no. 2 start	88.32	88.35	0.03
Control system gain change no. 1	104.96	104.97	0.01
No. 1 IGS update sent	105.00	105.00	0.00
Pitch program rate no. 2 end, no. 3 start	119.04	119.06	0.02
Stage I engine shutdown circuitry armed	144.64	144.65	0.01
No. 2 IGS update sent	145.00	145.00	0.00
Stage I MDTCPS unmakes subassembly 1	154.75	153.50	-1.25
Stage II MDTCPS unmakes subassembly 2	154.75	153.49	-1.26
BECO (stage I engine shutdown (87FS2))	154.83	153.55	-1.28
Staging switches actuate	154.83	153.55	-1.28
Signals from stage I rate gyro package to flight control system discontinued	154.83	153.55	-1.28
Hydraulic switchover lockout	154.83	153.55	-1.28
Telemetry ceases, stage I	154.83	153.55	-1.28
Staging nuts detonate	154.83	153.55	-1.28
Stage II engine ignition signal (91FS1)	154.83	153.55	-1.28
Control system gain change	154.83	153.55	-1.28
Stage separation begin	155.53	154.29	-1.24
Stage II engine MDFJPS make	155.73	154.28	-1.45
Pitch program rate no. 3 ends	162.56	162.61	0.05
Radio guidance enable	162.56	162.59	0.03
First guidance command signal (decoder output)	169.00	168.40	-0.60
Spacecraft horizon sensor cover jettisoned	199.83	207.00	7.17
Spacecraft radar cover jettisoned	199.83	207.00	7.17
Stage II engine shutdown circuitry armed	317.44	317.45	.01
SECO (stage II engine shutdown (91FS2))	336.93	333.28	-3.65

TABLE 4.2-I.- SEQUENCE OF EVENTS - Concluded

Event	Planned time, g.e.t.	Actual time, g.e.t.	Difference, sec
Redundant stage II shutdown	336.93	333.32	-3.61
Stage II MDFJPS break	337.23	333.44	-3.79
Spacecraft separation (shape charge fired)	356.93	356.91	-0.02
*OAMS on	356.93	356.11	.82
OAMS off	363.43	367.01	+3.58
Orbit phase, hr:min:sec			
Perigee adjust maneuver initiate	00:56:00	00:56:00	0
REP ejection maneuver initiate	02:07:00	02:07:15	15
Height adjustment initiate	50:50:00	50:49:57	-3
Phase adjustment initiate	51:34:42	51:34:31	-11
Plane change initiate	52:06:16	52:06:26	10
Coelliptic maneuver initiate	53:04:02	53:04:04	2
Reentry phase, hr:min:sec			
Equipment adapter separation	190:27:13	190:26:47	-26
Initiate retrorocket 1	190:27:43	190:27:43	0
Initiate manual retrofire	190:27:44	190:27:44	0
Initiate retrorocket 3	190:27:48	190:27:49	1
Initiate retrorocket 2	190:27:54	190:27:54	0
Initiate retrorocket 4	190:27:59	190:28:00	1
Retroadapter separate	190:28:28	190:28:30	2
Begin blackout	190:44:01	190:44:06	5
End blackout	190:48:18	190:47:58	-20
Drogue parachute deployment	190:50:09	190:49:19	-50
Pilot parachute deployment/main parachute initiate	190:52:01	190:51:16	-45
Landing	190:56:42	190:55:14	-88
Parachute jettison	-	190:55:17	-

*OAMS thrusters were off from 361.26 to 363.34 sec due to switchover from direct mode to rate command by command pilot.

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4.3 FLIGHT TRAJECTORIES

The launch and orbital trajectories referred to as planned are either preflight calculated nominal trajectories from references 6 and 7, respectively, or trajectories based on nominal outputs from the real-time computer complex (RTCC) 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. The Patrick Air Force Base atmospheres were used below 25 nautical miles, and 1959 ARDC model atmospheres were used above 25 nautical miles for all trajectories except the actual launch phase which used the atmosphere up to 25 nautical miles at the time of launch. The earth model for all trajectories contained geodetic and gravitational constants representing the Fischer Ellipsoid. A ground track of the first four and last three revolutions is shown in figure 4.3-1. Launch, orbit, rendezvous evaluation pod (REP) test, simulated Agena rendezvous, and reentry trajectory curves are presented in figures 4.3-2 to 4.3-7.

4.3.1 Gemini Spacecraft

4.3.1.1 Launch.- The launch trajectory data shown in figure 4.3-2 are based on the real-time output of the range-safety impact prediction computer (IP 3600) and the Guided Missile Computer Facility (GMCF). The IP 3600 used data from the missile trajectory measurement system (MISTRAM), FPS-16, and FPQ-6 radars. The GMCF used data from the GE Mod III radar. Data from these tracking facilities were used during the time periods listed in the following table:

Facility	Time from lift-off, sec
IP 3600 (FPQ-6)	0 to 11
GMCF (GE Mod III)	11 to 348

The actual launch trajectory, as compared with the planned launch trajectory in figure 4.3-2, was high in altitude and flight-path angle and low in velocity during stage I powered flight. After BECO the radio guidance system (RGS) corrected the trajectory and guided the second stage to a nominal insertion. At BECO the altitude and flight-path angle were high by 5380 feet and 0.89°, respectively, and velocity was low by 155 ft/sec. At SECO the altitude, flight-path angle, and velocity were slightly low by 583 feet, 0.02°, and 1 ft/sec, respectively. A comparison of planned and actual BECO and SECO conditions is shown in table 4.3-I. Actual SECO is based on inertial guidance system (IGS) corrected data. At spacecraft separation, the actual altitude and velocity were low by 53 feet and 2 ft/sec, respectively, when compared with the planned conditions. There were no measurable dispersions in flight-path angle. Table 4.3-I contains a comparison of planned and actual conditions for spacecraft separation (SECO + 23.6 seconds). The preliminary conditions were based on integrating the Canary vector back to separation through the planned attitudes and spacecraft velocity changes (ΔV). The same procedures were used to get the final conditions; however, actual attitude and applied ΔV's were used for the backward integration. The GE Mod III and MISTRAM tracking radar data after SECO are used to compute a go—no-go for spacecraft insertion by averaging 10 seconds of these data starting at SECO + 5 seconds. The go—no-go conditions obtained from these sources indicated that the velocity and flight-path angle were 15.5 ft/sec high and 0.2° low, respectively, when compared with the orbital ephemeris data. Figure 5.2-3 shows the GE Mod III and MISTRAM radar tracking data in the go—no-go region after SECO. It should be noted that the quality of the GE Mod III data rapidly decayed during this period, starting just prior to SECO, because of the low elevation angles; however, because of the smoothing and editing of the data from the ground guidance computers, these degraded data had no effect on the accuracy of the insertion parameters.

4.3.1.2 <u>Orbit.</u>- A comparison of the planned and actual apogees and perigees in reference 7 is shown in figure 4.3-3. The actual apogees and perigees were obtained by integrating the best Gemini tracking network vectors throughout the mission to the apogee and perigee that followed. Table 4.3-II contains a comparison of the planned and actual elements. Preliminary elements are outputs from the real-time computer complex (RTCC) during the mission and are measured over a spherical earth; final elements are measured over an oblate earth. At insertion, the oblate measurement is approximately 0.8 nautical mile greater than the spherical measurement. The apsidal advancement during the mission, however, moved apogee and perigee nearer the equator, thus increasing the earth radius and allowing the spherical measurement to be approximately 2.5 nautical miles higher than the oblate measurement toward the end of the mission.

On Gemini IV, using the 1959 ARDC atmosphere, an atmospheric K factor of 0.72 was required to obtain the lifetime based on a tumbling spacecraft reference area. This is equivalent to a K = 1.01 for a small-end-forward (SEF) or blunt-end-forward (BEF) stable attitude, which indicates a 1-percent uncertainty in the $C_D A \rho$ term. On Gemini V, the ACR initially computed a K factor of 0.75, based on a tumbling vehicle. This is equivalent to a K = 1.05 for a SEF or BEF attitude, indicating a 5-percent uncertainty in the $C_DA\rho$ term. However, after the simulated rendezvous, the K factor computed by ACR for a tumbling vehicle decreased to 0.55, or K = 0.77 for a stable vehicle, resulting in an approximate 23-percent uncertainty in the $C_DA\rho$ term. There are two possible reasons for this large uncertainty. First, this uncertainty did not develop until after perigee was raised 17.6 nautical miles, indicating that the upper atmosphere and coefficient of drag around 108 nautical miles are not known as well as around 87 to 92 nautical miles. Second, the spacecraft attitude control and the hydrogen and oxygen continued venting throughout most of the flight, which could have added some energy to the orbit. This $C_DA\rho$ uncertainty is a major problem in exact orbit determination, and current plans are being made to investigate this parameter in more detail.

4.3.1.2.1 REP exercise: The rendezvous evaluation pod (REP) was released in the second revolution. The planned and actual maneuvers prior to and including REP ejection are shown in table 4.3-III. A time history and relative trajectory profile between the spacecraft and REP are shown in figure 4.3-4 from the time of REP ejection until approximately 5 hours thereafter. The REP trajectory was determined by integrating the spacecraft Bermuda vector through the actual ejection velocity and attitude, as determined by onboard radar and telemetry, and then comparing that trajectory to the spacecraft trajectory also determined by the Bermuda vector. The REP range for 30 hours is shown in figure 4.3-5. The REP range for the first 16 hours in this figure was computed from the integrated trajectory, and after 16 hours was computed using tracking data from North American Air Defense Command (NORAD) Space Acquisition Detection and Tracking System (SPADATS) radars.

In revolution 31, NORAD predicted a REP orbit lifetime of 5.58 days, and in revolution 57 this prediction changed slightly to 5.71. The REP actually lasted 5.71 days and was tracked by radar in Turkey as it reentered during revolution 87. Projected impact was in the South Pacific, north of New Zealand at latitude 31°48' S. and longitude 175°12' E.

Based on preflight REP aerodynamics, the ACR initially computed a 3.5-day lifetime. In order for the ACR to achieve the actual lifetime, the ballistic parameter W/C_DA had to be increased from 11.2 lb/ft^2 to 18.9 lb/ft^2. This reflects a $C_DA\rho$ uncertainty of 41 percent.

4.3.1.2.2 Simulated rendezvous: During the third day of the mission, rendezvous midcourse maneuvers were executed in order to evaluate the techniques to be used for the first Gemini-Agena rendezvous mission. A flight plan involving a simulated Agena target-vehicle orbit had been developed, where the nominal Gemini V maneuver points were scheduled for a rendezvous number M of 4. The maneuvers were, in order, a height adjustment N_H, a phase adjustment N_{CL}, a plane change N_{PC}, and a coelliptic maneuver N_{SR}. The initial simulated Agena orbit had apogee and perigee altitudes of 184.2 and 123.8 nautical miles, respectively, whereas the spacecraft orbit had initial apogee and perigee altitudes of 181.8 and 91.1 nautical miles, respectively. The phase angle at spacecraft insertion was 91.01° and the initial phase rate was 2.6° per orbit.

Figure 4.3-6 shows the time history and relative trajectory profile from the spacecraft to the imaginary Agena. This figure was determined by integrating the Carnarvon vector in revolution 32 through the actual maneuvers described in table 4.3-IV and then comparing the resultant trajectory to that of the imaginary Agena. The planned spacecraft maneuvers in table 4.3-IV were generated to create an imaginary Agena orbit, consistent with the allowable spacecraft fuel expenditure determined in real time. The ground-computed maneuvers were generated by the ACR. These maneuvers were calculated from network tracking vectors after the orbit had been redefined subsequent to each maneuver. The height adjustment was based on the Carnarvon 32 vector, N_{CL} and N_{PC} on the Merritt Island 32 vector, and N_{SR} on the California 33 vector. The actual maneuvers in table 4.3-IV were determined by telemetry where possible and by crew reports of the incremental velocity indicator (IVI) display of the output of the onboard computer. This is the first time rendezvous midcourse maneuvers have been attempted. When these maneuvers were completed, the spacecraft was in the planned coelliptic orbit with a 15-nautical-mile differential altitude. If ground-computed terminal maneuvers had been performed with the same accuracy as the midcourse maneuvers, the spacecraft would have been 0.1 nautical mile from the Agena at the docking maneuver initiation, and the docking maneuver time would have changed less than 2 minutes from nominal.

4.3.1.3 Reentry.- The planned and actual reentry phase of the trajectory is shown in figure 4.3-7. The planned trajectory was determined by integrating the Canary vector in revolution 120 through planned retrofire sequences determined by the RTCC and then by flying a half-lift reentry according to Math Flow 6 described in reference 8. The actual trajectory was obtained by integrating the Canary vector in revolution 120 through actual retrofire attitudes and sequences and then integrating the White Sands vector through actual roll angles and parachute deployment sequences. The trajectory obtained with the Canary vector and actual retrofire data agreed with the postretrofire trajectory obtained with the White Sands vector. Table 4.3-I contains a comparison of the planned and actual reentry dynamic parameters and landing points.

The actual landing point was 89 nautical miles short of the predicted landing point. The major part of this dispersion can be attributed to the computer update error described in section 6.2.2.2.1. Reconstructing a reentry trajectory through the actual roll angles and using the 1959 standard atmosphere charts gave a landing point 19 nautical miles downrange of the actual landing point. In order to fit the trajectory to the actual landing point, the 1959 atmosphere density was increased 13.6 percent between altitudes of 386 000 and 69 000 feet (drogue parachute deployment). This indicates an uncertainty in the $C_D A \rho$ term during the critical dynamic region of reentry. However, the actual density profile as shown in section 12.2 was approximately 15 percent greater than that in the 1959 atmosphere. The landing point obtained with this trajectory was latitude 29°44' N. and longitude 69°45' W. This landing point agrees within 1 nautical mile of the landing coordinates recorded by the Sea Air Rescue (SAR) unit, and within 3 nautical miles of the composite landing point calculations which combined the SAR data and data from the recovery aircraft shown in table 4.3-I.

This reconstructed reentry trajectory agrees very well with the actual trajectory. Communication blackout conditions, deceleration, and drogue parachute deployment altitude are in close agreement with actual event times and magnitudes as recorded by instrumentation.

The reentry curves below drogue parachute deployment are based on nominal parachute-force coefficients (fig. 4.3-7). Because the drogue parachute was deployed in a supersonic region, the data on the curves are displaced in altitude from that reported in section 5.1.11.

4.3.2 Gemini Launch Vehicle Second Stage

The second stage of the Gemini launch vehicle was inserted into an orbit with apogee and perigee altitudes of 182.7 and 87.4 nautical miles, respectively. The Gemini network tracking radars were able to skin-track the second stage during the ensuing 3-day orbit lifetime. Tracking was obtained during reentry in revolution 48 and the Pretoria tracking station reported visual observation of reentry breakup. Estimated impact point was latitude 24° S. and longitude 108° E. in the Indian Ocean.

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

Condition	Planned (a)	Actual	
		Preliminary	Final
BECO			
Time from lift-off, sec	154.83	Not computed	153.58
Geodetic latitude, deg North	28.77		28.77
Longitude, deg West	79.64		79.67
Altitude, feet	210 170		215 550
Altitude, n. mi.	34.6		35.5
Range, n. mi.	50.9		52.4
Space-fixed velocity, ft/sec	9 988		9 833
Space-fixed flight-path angle, deg	18.98		19.87
Space-fixed heading angle, deg	75.07		74.42
SECO			
Time from lift-off, sec	336.93	Not computed	333.28
Geodetic latitude, deg North	30.55		30.52
Longitude, deg West	72.05		72.25
Altitude, feet	530 881		530 298
Altitude, n. mi.	87.4		87.3
Range, n. mi.	461.6		448.4
Space-fixed velocity, ft/sec	25 721		25 720
Space-fixed flight-path angle, deg	0.0		-0.02
Space-fixed heading angle, deg	77.75		77.90
Spacecraft separation			
Time from lift-off, sec	356.93	353.26	356.91
Geodetic latitude, deg North	30.84	30.84	30.89
Longitude, deg West	70.56	70.82	70.55
Altitude, feet	531 121	531 039	531 068
Altitude, n. mi.	87.4	87.4	87.4
Range, n. mi.	540.4	524.8	539.6
Space-fixed velocity, ft/sec	25 807	25 807	25 805
Space-fixed flight-path angle, deg	0.0	-.01	0.0
Space-fixed heading angle, deg	78.54	78.38	78.52

[a]The planned values are for spacecraft separation at SECO + 20 seconds; whereas the actual values are for spacecraft separation at SECO + 23.6 seconds.

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

Condition	Planned (a)	Actual	
		Preliminary	Final
Maximum conditions			
Altitude, statute miles	220.0	217.4	217.4
Altitude, n. mi.	191.2	188.9	188.9
Space-fixed velocity, ft/sec	25 817	25 812	25 812
Earth-fixed velocity, ft/sec	24 501	24 497	24 497
Exit acceleration, g	7.3	--	7.6
Exit dynamic pressure, lb/sq ft	748	--	766
Reentry deceleration, g (tracking data).	4.9	6.5	6.4
Reentry deceleration, g (telemetry data)	--	--	7.1
Reentry dynamic pressure, lb/sq ft . . .	320	422	414
Landing point			
Latitude, deg:min North	29:43	29:58	29:47
Longitude, deg:min West	68:00	69:39	69:45

[a]The planned values are for spacecraft separation at SECO + 20 seconds; whereas the actual values are for spacecraft separation at SECO + 23.6 seconds.

TABLE 4.3-II.- COMPARISON OF ORBITAL ELEMENTS

Revolution	Condition	Planned	Actual	
			Preliminary	Final
Insertion	Apogee, n. mi.	191.2	189.0	188.9
	Perigee, n. mi.	87.0	86.8	87.4
	Inclination, deg	32.53	32.59	32.59
	Period, min	89.64	Not available	89.59
2 (after perigee adjust)	Apogee, n. mi.	191.2	188.3	188.9
	Perigee, n. mi.	93.1	91.9	92.4
	Inclination, deg	32.53	32.63	32.59
	Period, min	89.70	Not available	89.68
5	Apogee, n. mi.	189.0[a]	187.3	187.8
	Perigee, n. mi.	95.2	91.6	92.1
	Inclination, deg	32.53	32.63	32.59
	Period, min	89.73	Not available	89.65
16	Apogee, n. mi.	185.5[a]	184.8	185.5
	Perigee, n. mi.	94.1	91.2	91.9
	Inclination, deg	32.53	32.59	32.59
	Period, min	89.64	Not available	89.57
32 (Before simulated rendezvous)	Apogee, n. mi.	179.4[a]	180.8	181.8
	Perigee, n. mi.	93.0	90.4	91.1
	Inclination, deg	32.53	32.60	32.59
	Period, min	89.50	Not available	89.48
34 (after simulated rendezvous)	Apogee, n. mi.	178.6[a]	168.4	168.8
	Perigee, n. mi.	92.9	108.0	108.7
	Inclination, deg	32.53	32.62	32.61
	Period, min	89.48	Not available	89.57
48	Apogee, n. mi.	172.6[b]	166.4	167.4
	Perigee, n. mi.	91.8	107.9	108.5
	Inclination, deg	32.54	32.65	32.61
	Period, min	89.36	Not available	89.55

[a]Planned elements reflect REP rendezvous maneuvers which were not performed.

[b]Planned elements do not reflect simulated Agena rendezvous maneuvers which were performed.

TABLE 4.3-II.- COMPARISON OF ORBITAL ELEMENTS - Concluded

Revolution	Condition	Planned	Actual	
			Preliminary	Final
64	Apogee, n. mi.	165.0[b]	164.6	164.7
	Perigee, n. mi.	90.6	107.0	107.5
	Inclination, deg	32.54	32.62	32.61
	Period, deg	89.19	Not available	89.51
80	Apogee, n. mi.	154.6[b]	162.2	162.1
	Perigee, n. mi.	89.2	107.0	107.1
	Inclination, deg	32.55	32.63	32.61
	Period, min	89.00	Not available	89.44
96	Apogee, n. mi.	146.3[b]	160.4	158.4
	Perigee, n. mi.	87.4	107.1	106.4
	Inclination, deg	32.55	32.63	32.61
	Period, min	88.78	Not available	89.40
112	Apogee, n. mi.	133.7[b]	158.2	156.4
	Perigee, n. mi.	84.9	107.0	106.0
	Inclination, deg	32.56	32.62	32.61
	Period, min	88.50	Not available	89.35
120	Apogee, n. mi.	124.4[b]	157.3	154.8
	Perigee, n. mi.	83.0	107.2	106.0
	Inclination, deg	32.56	32.61	32.61
	Period, min	88.30	Not available	89.32

[a]Planned elements reflect REP rendezvous maneuvers which were not performed.

[b]Planned elements do not reflect simulated Agena rendezvous maneuvers which were performed.

TABLE 4.3-III.- COMPARISON OF PLANNED AND ACTUAL MANEUVERS DURING THE REP EXERCISE

Condition	Planned	Actual
Perigee adjust		
Maneuver initiate, hr:min:sec, g.e.t.	00:56:00	00:56:00
ΔV, ft/sec	10.0	9.7
Pitch, deg	0.0	0.0
Yaw, deg	0.0	0.0
Thrust duration, sec	12	13
REP eject		
Maneuver initiate, hr:min:sec, g.e.t.	02:07:00	02:07:15
ΔV applied to REP, ft/sec	5.0	[a]4.6
Spacecraft attitude, pitch, deg	0.0	[a]-1.0
Spacecraft attitude, yaw, deg	90.0	[a]88.1

[a]Spacecraft attitudes reflect the angles required to eject the REP directly off the Z-axis of the spacecraft into the orbit as determined by matching spacecraft radar data and ground-based radar data.

TABLE 4.3-IV. - RENDEZVOUS MANEUVERS

Condition	Planned	Ground directed	Actual
Apogee adjustment $(N_H = 1.50)$			
Maneuver initiate, hr:min:sec, g.e.t. . . .	50:50:00	50:49:57	50:49:58
ΔV, ft/sec	-20.5	-21.1	-20.9
Pitch, deg	0.0	0.0	1.9
Yaw, deg	0.0	0.0	0.0
Thrust duration, sec	27	28	27
Phase adjustment $(N_{CL} = 1.5)$			
Maneuver initiate, hr:min:sec, g.e.t. . . .	51:34:42	51:34:31	51:34:31
ΔV, ft/sec	15.1	15.2	15.7
Pitch, deg	0.0	0.0	0.0
Yaw, deg	0.0	0.0	0.0
Thrust duration, sec	20	20	20
Plane change $(N_{PC} = 2.5)$			
Maneuver initiate, hr:min:sec, g.e.t. . . .	52:06:16	52:06:26	52:06:26
ΔV, ft/sec	15.0	14.6	15.0
Pitch, deg	0.0	0.0	-0.8
Yaw, deg	-90.0	-90.0	-89.2
Thrust duration, sec	19	19	19.7
Coelliptical maneuver $(N_{SR} = 3.0)$			
Maneuver initiate, hr:min:sec, g.e.t. . . .	53:04:02	53:04:04	53:04:04
ΔV, ft/sec	16.4	17.4	17.2
Pitch, deg	13.0	15.7	15.2
Yaw, deg	0.0	0.0	-0.3
Thrust duration, sec	21	22	22.5

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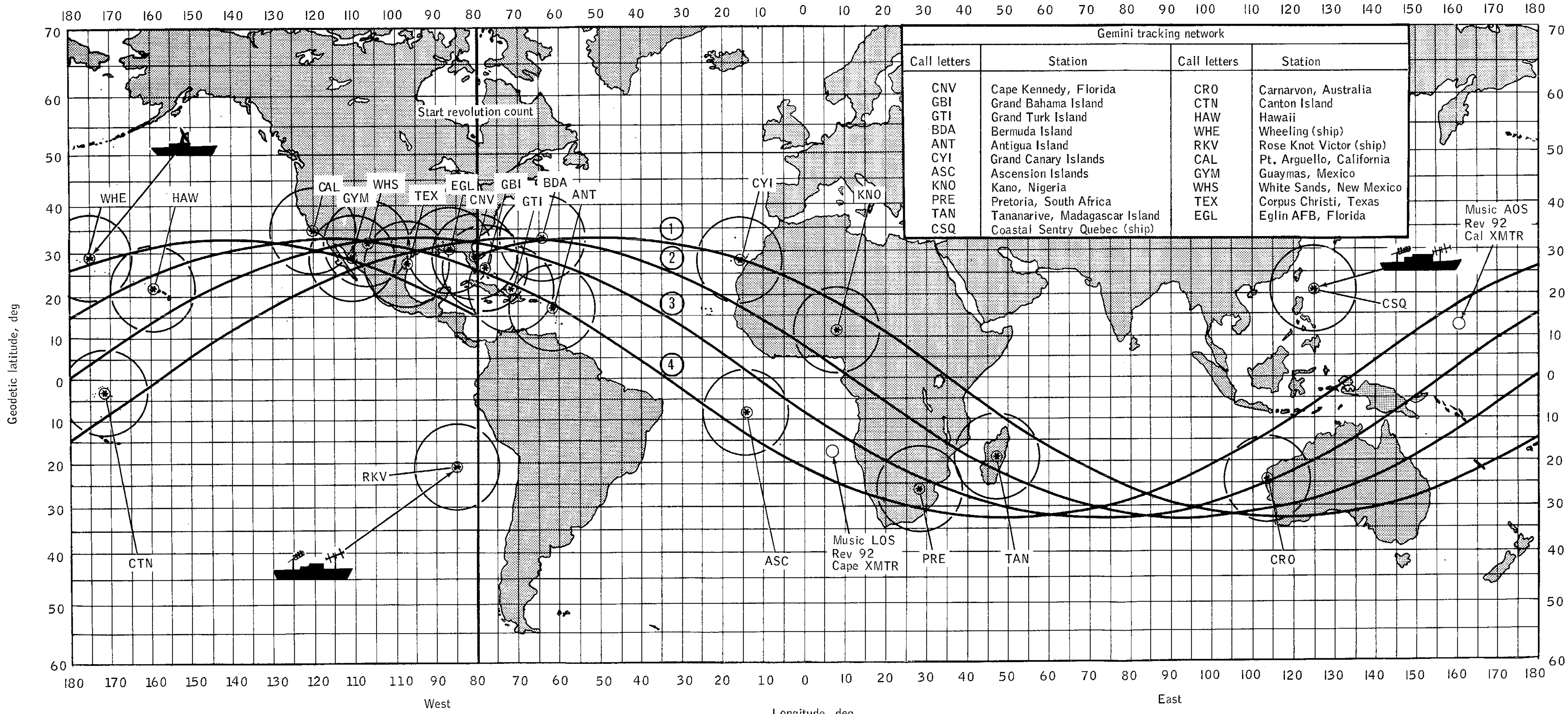

Call letters	Station	Call letters	Station
CNV	Cape Kennedy, Florida	CRO	Carnarvon, Australia
GBI	Grand Bahama Island	CTN	Canton Island
GTI	Grand Turk Island	HAW	Hawaii
BDA	Bermuda Island	WHE	Wheeling (ship)
ANT	Antigua Island	RKV	Rose Knot Victor (ship)
CYI	Grand Canary Islands	CAL	Pt. Arguello, California
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
CSQ	Coastal Sentry Quebec (ship)		

(a) Revolutions 1 through 4

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

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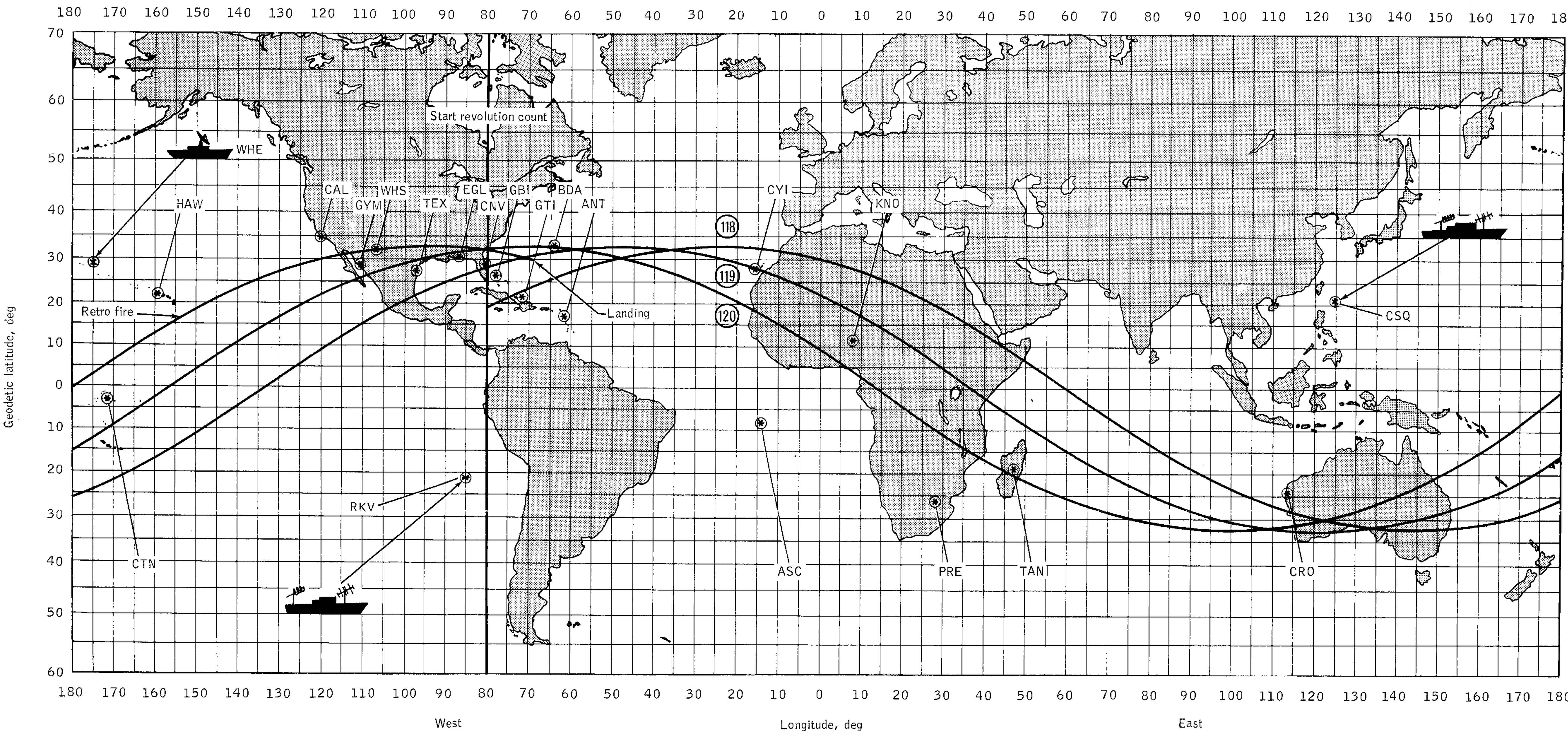

(b) Revolutions 118 through 120

Figure 4.3-1. - Concluded.

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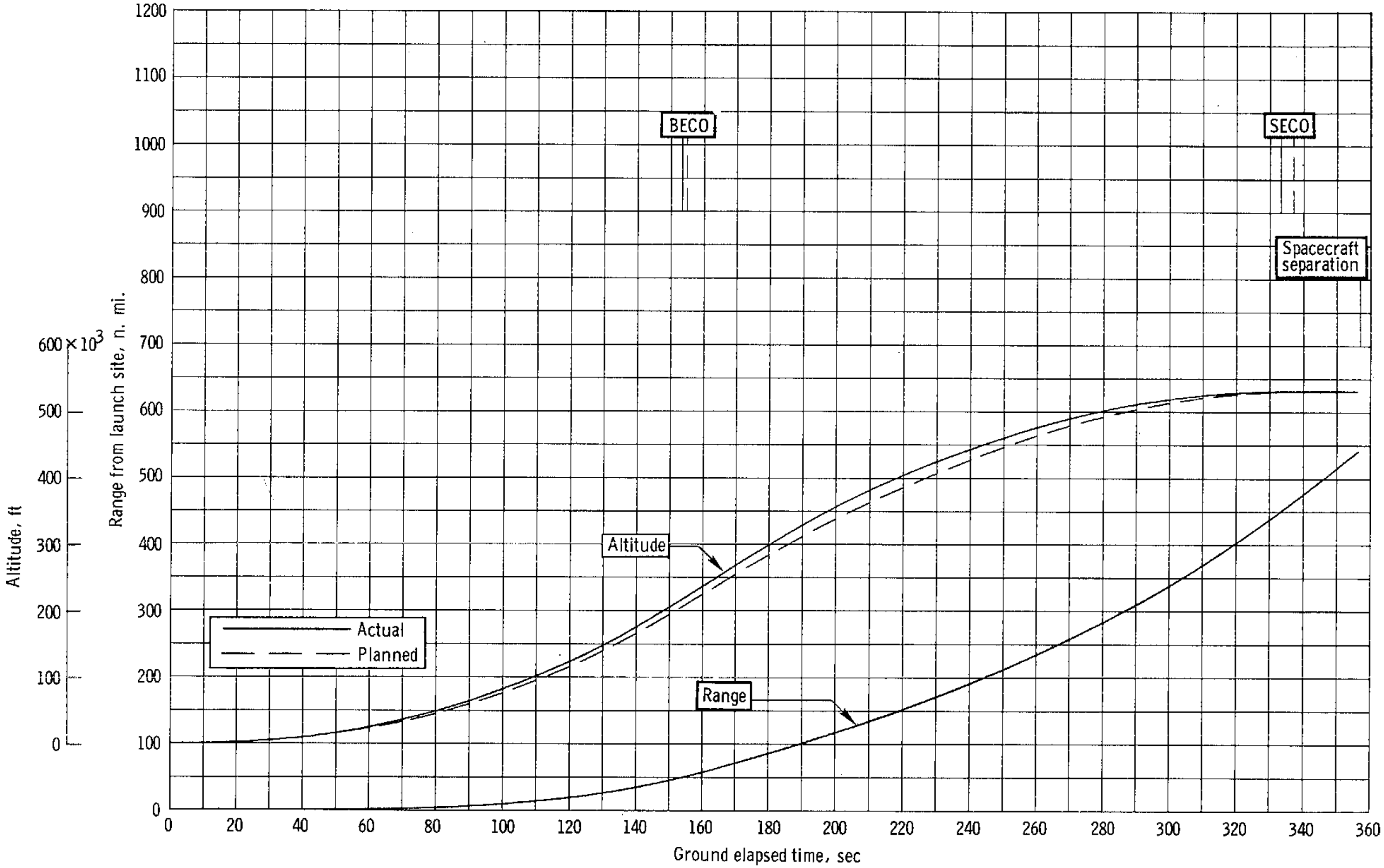
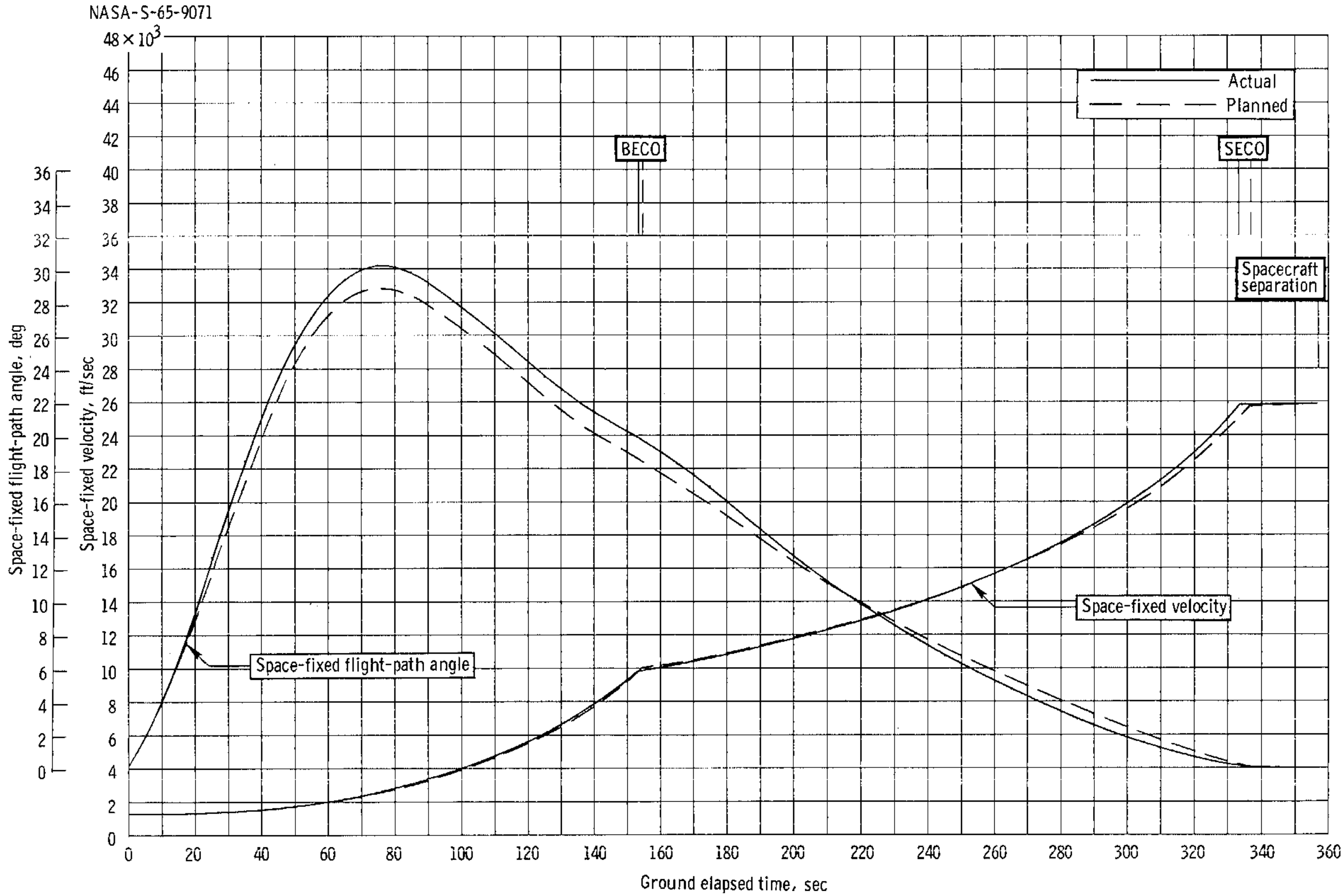


(a) Altitude and range.

Figure 4.3-2. - Trajectory parameters for Gemini V mission launch phase.



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

Figure 4.3-2. - Continued.

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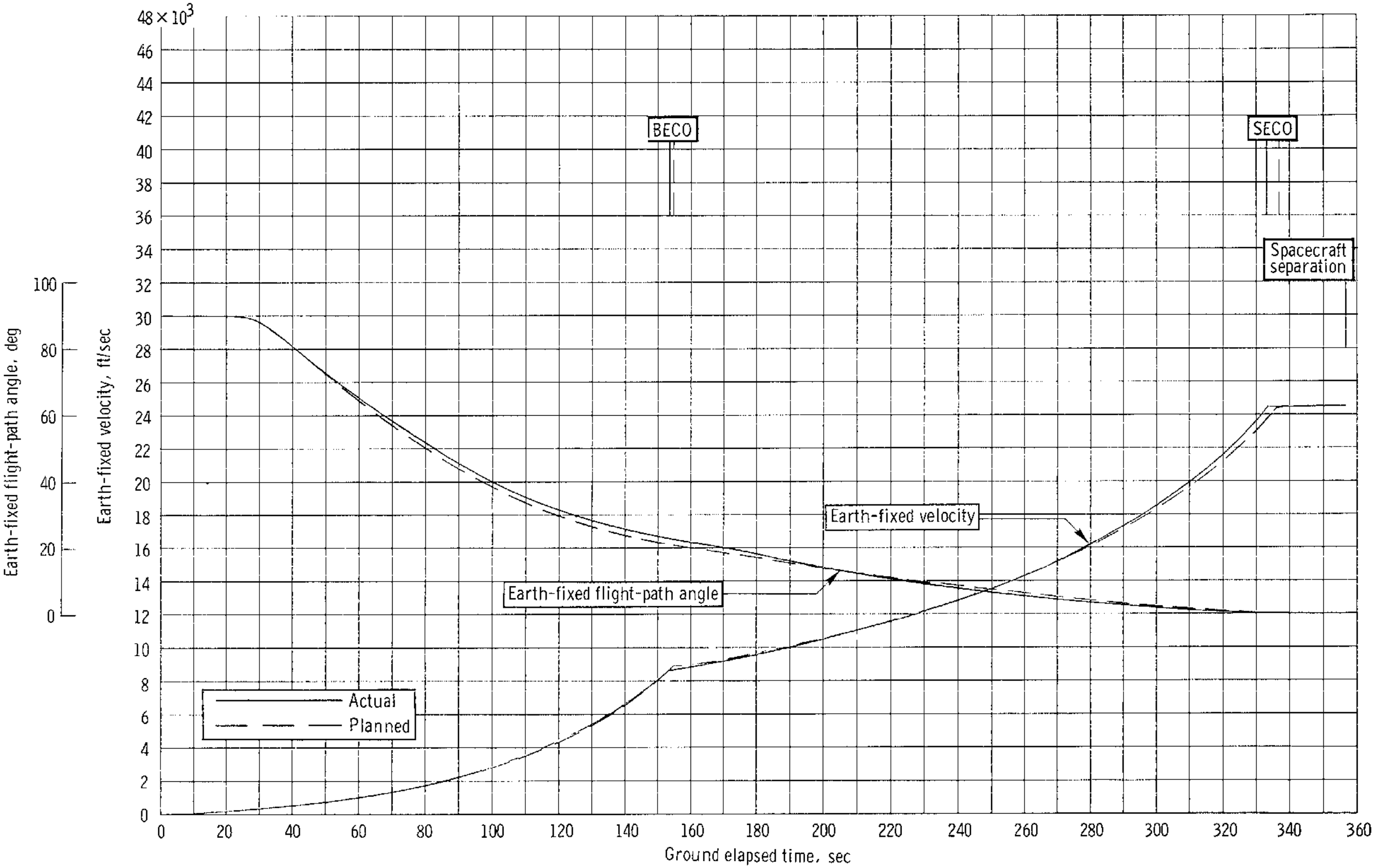

(c) Earth-fixed velocity and flight-path angle.

Figure 4.3-2. - Continued.

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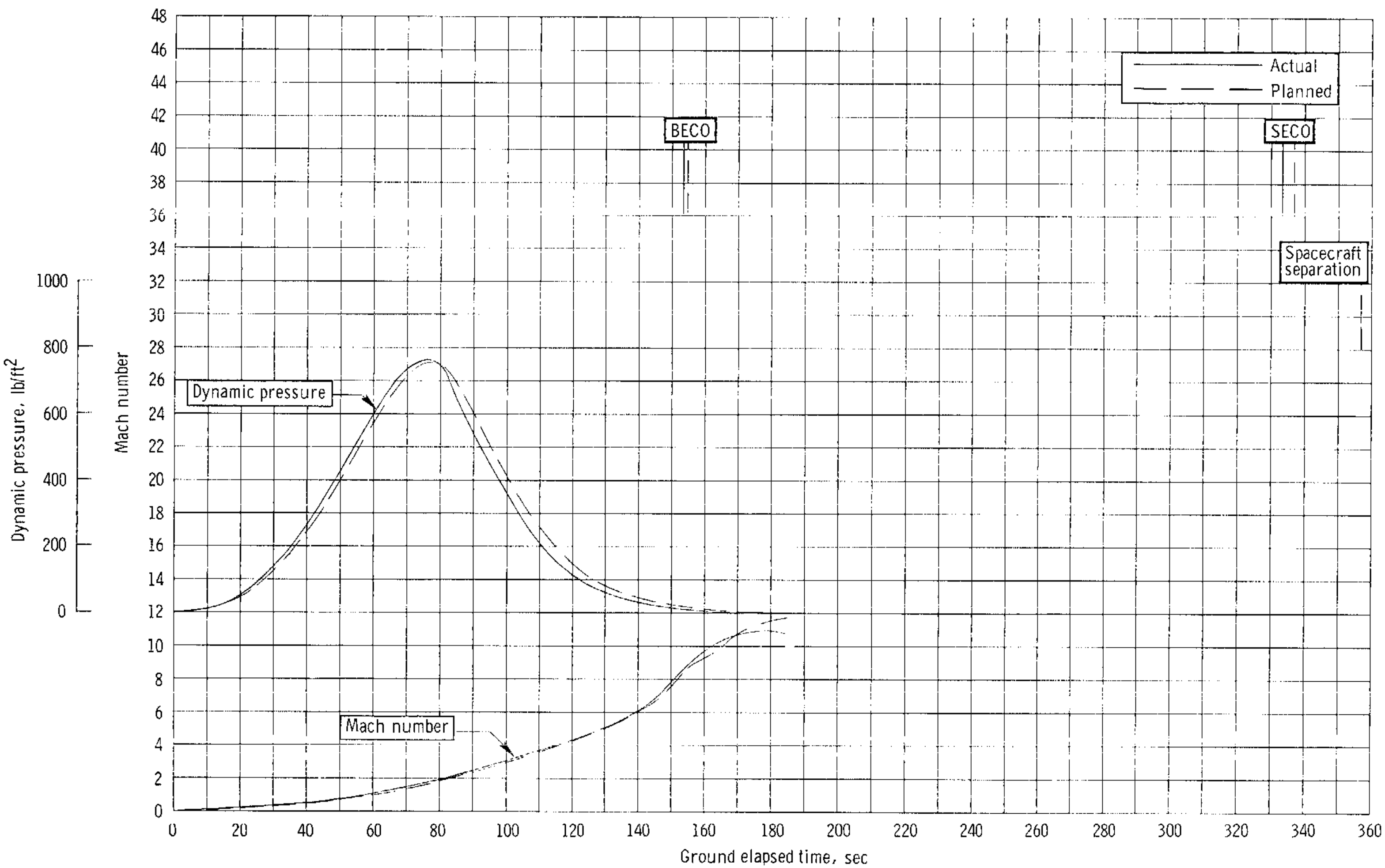

(d) Dynamic pressure and Mach number.

Figure 4.3-2. - Continued.

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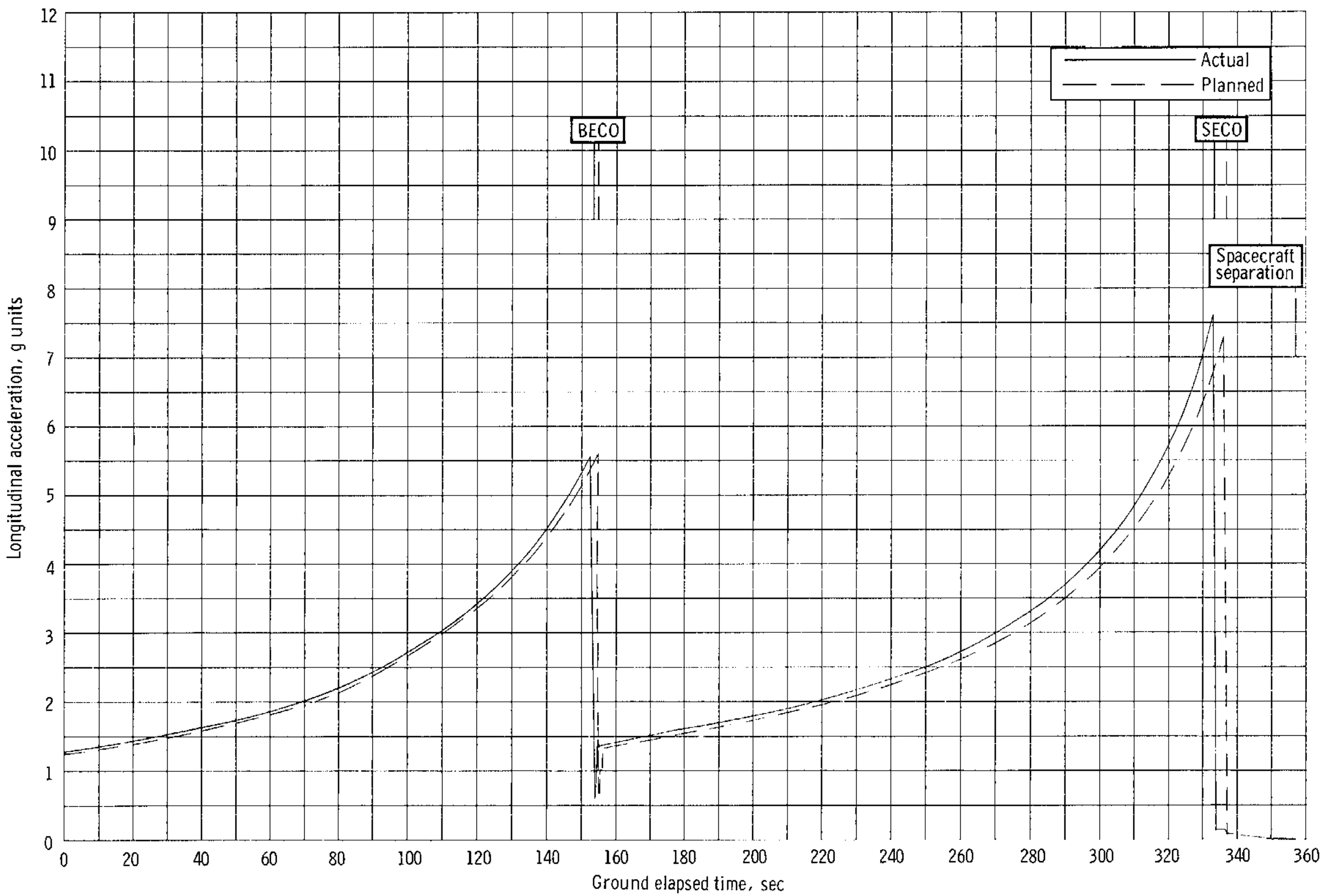

(e) Longitudinal acceleration.

Figure 4.3-2. - Concluded.

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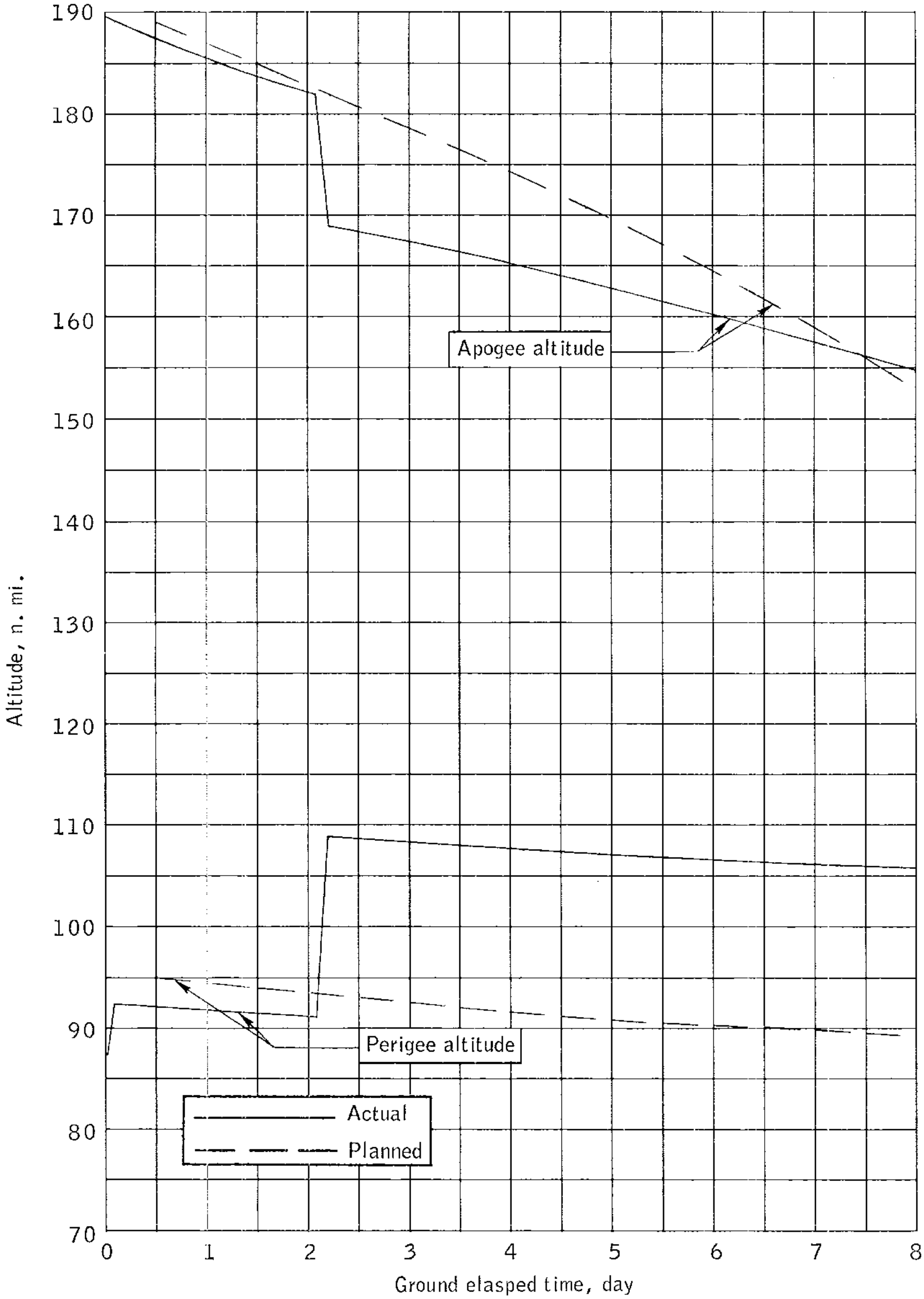

Figure 4.3-3. - Apogee and perigee altitudes for the Gemini V mission.

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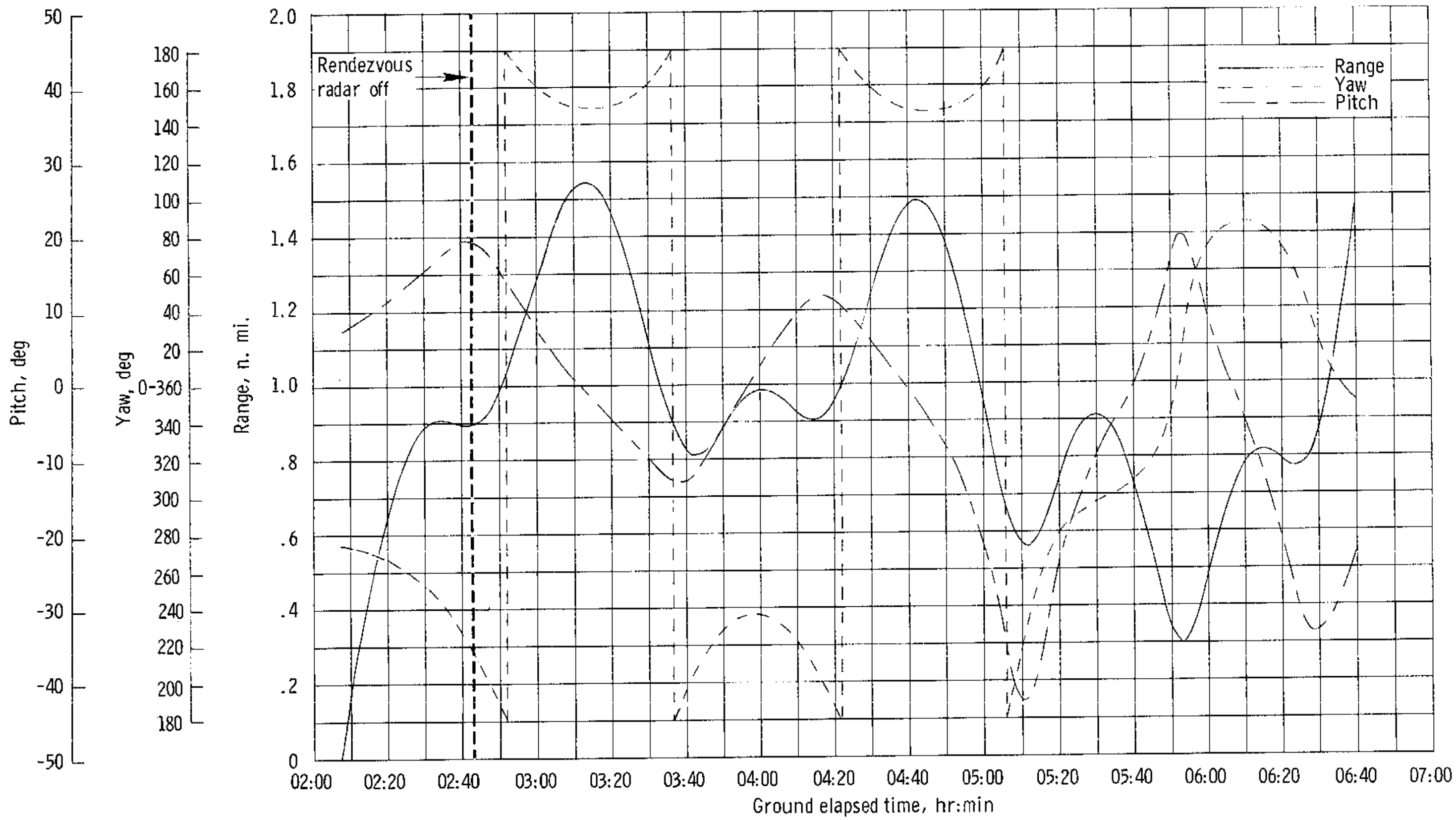

(a) Range, yaw, and pitch referenced to spacecraft.

Figure 4.3-4. - Relative trajectory from spacecraft to REP between 02:07:15 and 06:40:00 (hr:min:sec) g.e.t.

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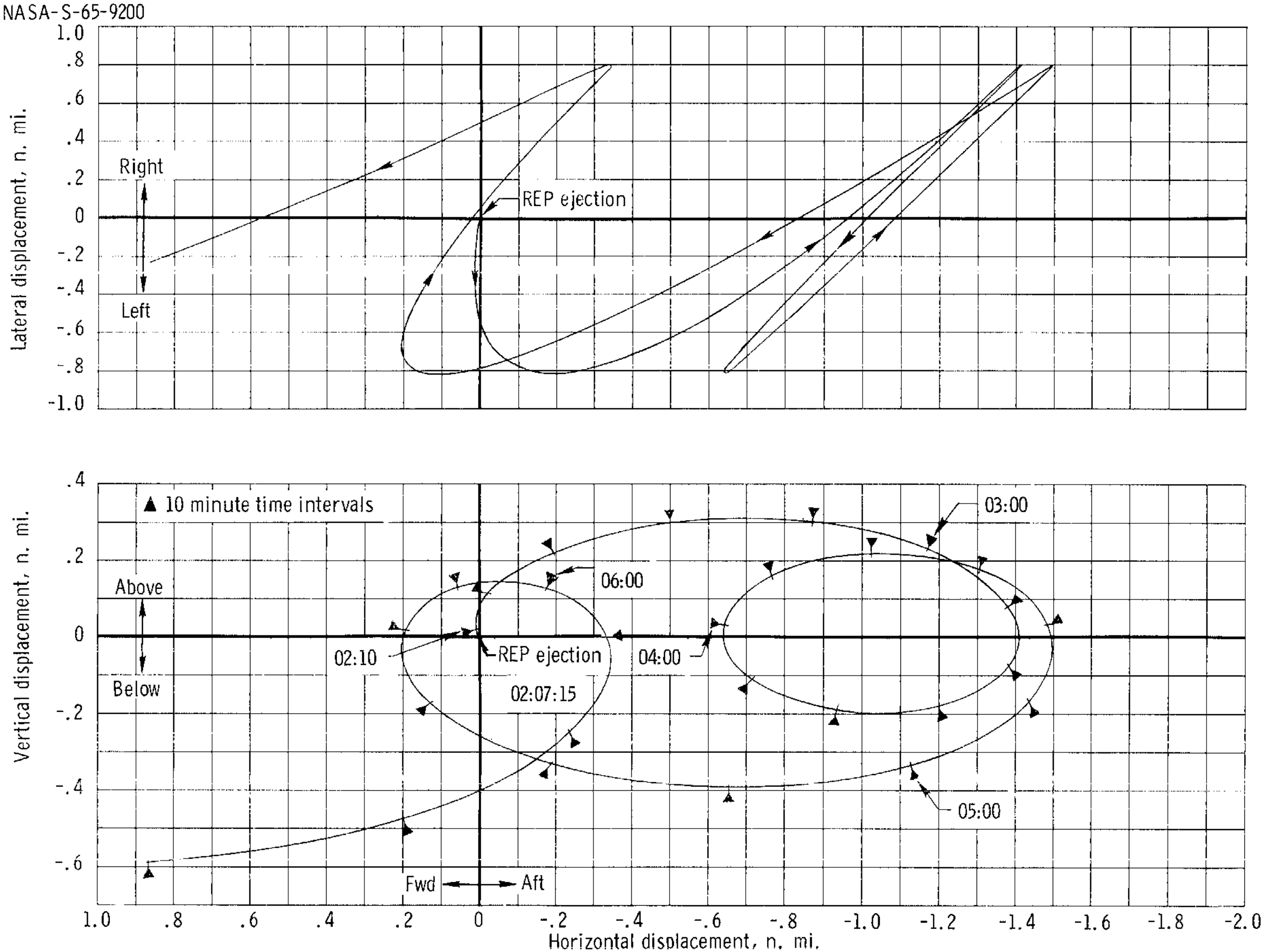

(b) Relative trajectory profile referenced to spacecraft.

Figure 4.3-4. - Concluded.

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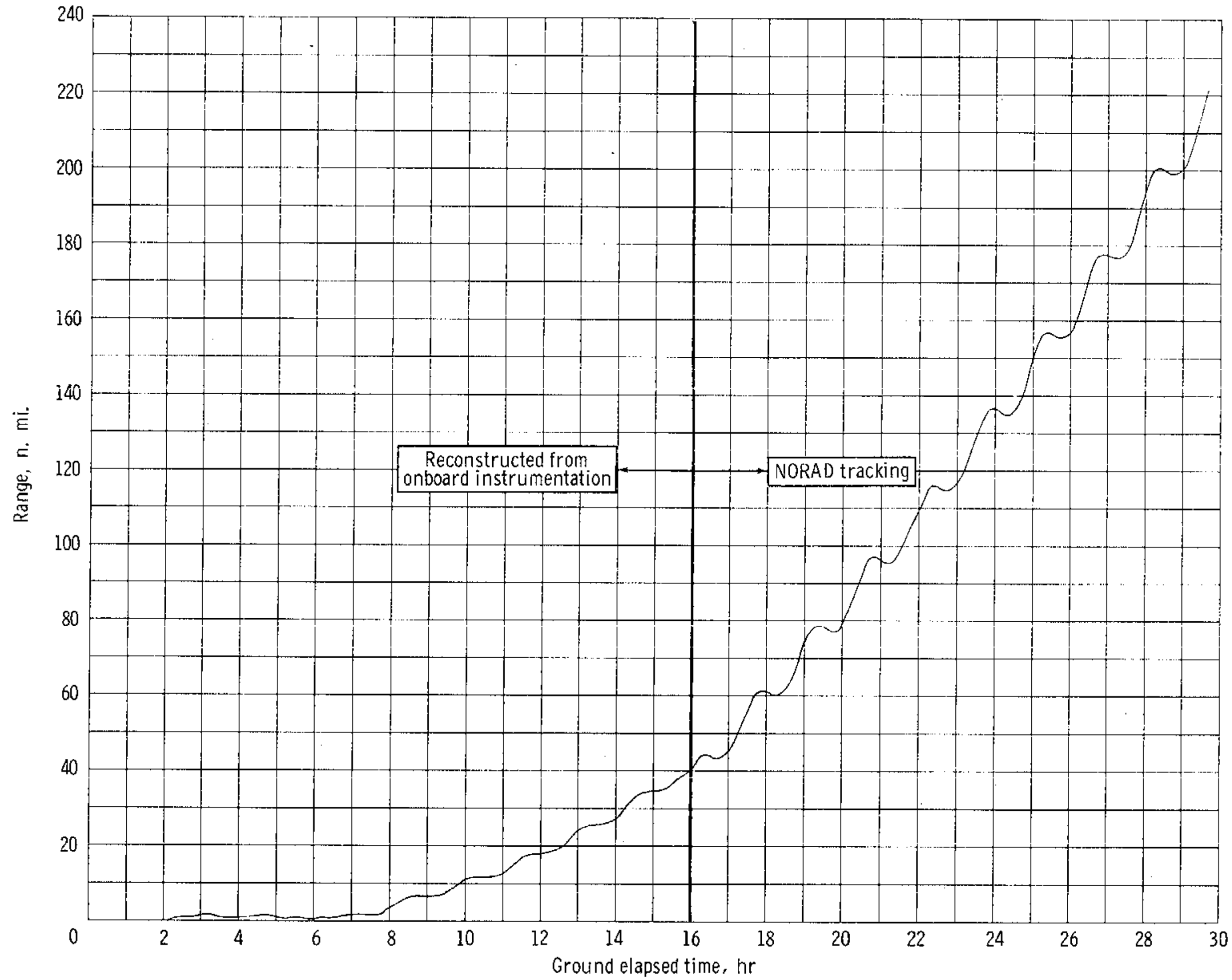

Figure 4.3-5. - Separation range between the REP and the spacecraft.

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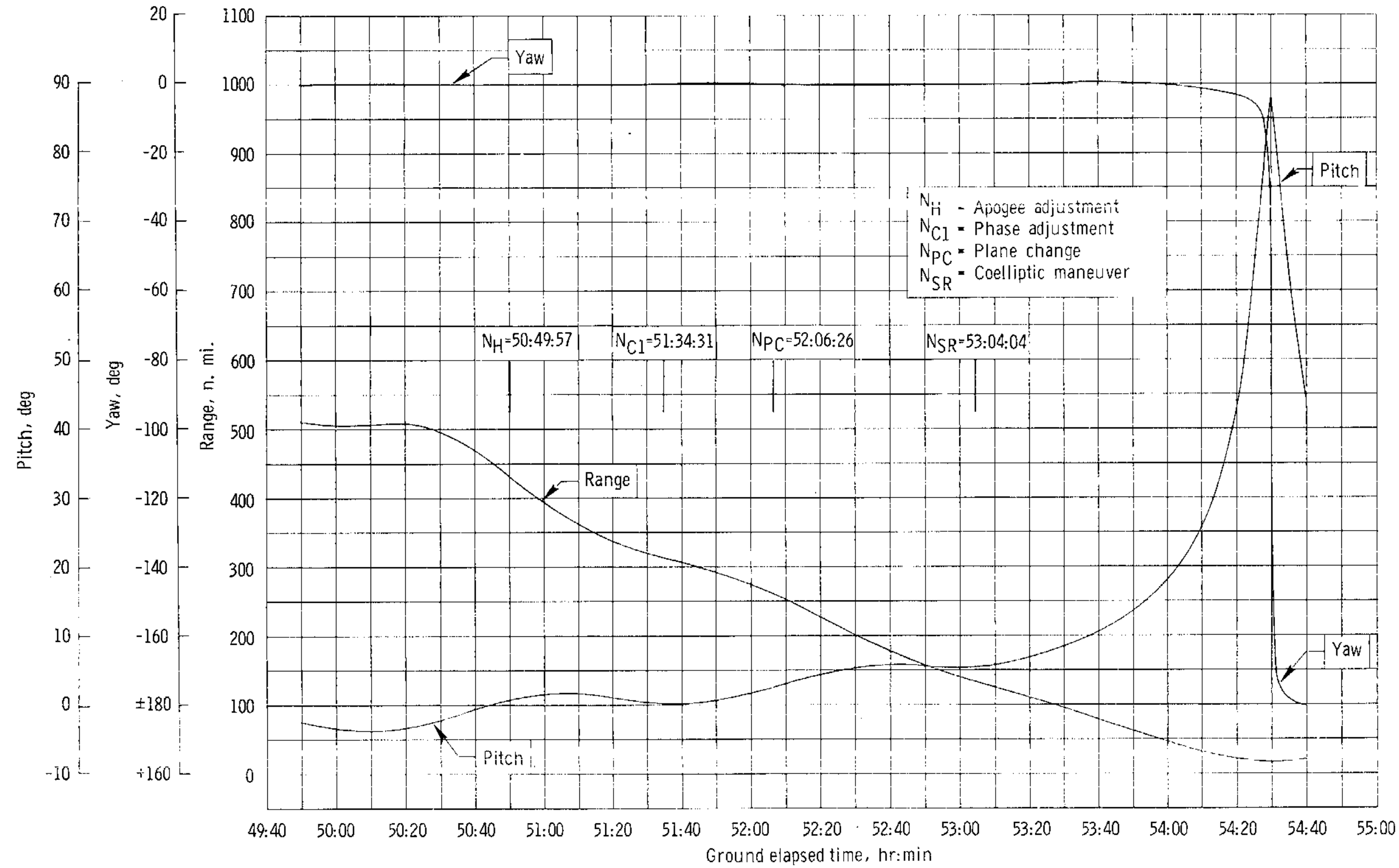

(a) Range, yaw, and pitch from Gemini V to simulated Agena.

Figure 4.3-6. - Simulated rendezvous during the Gemini V mission.

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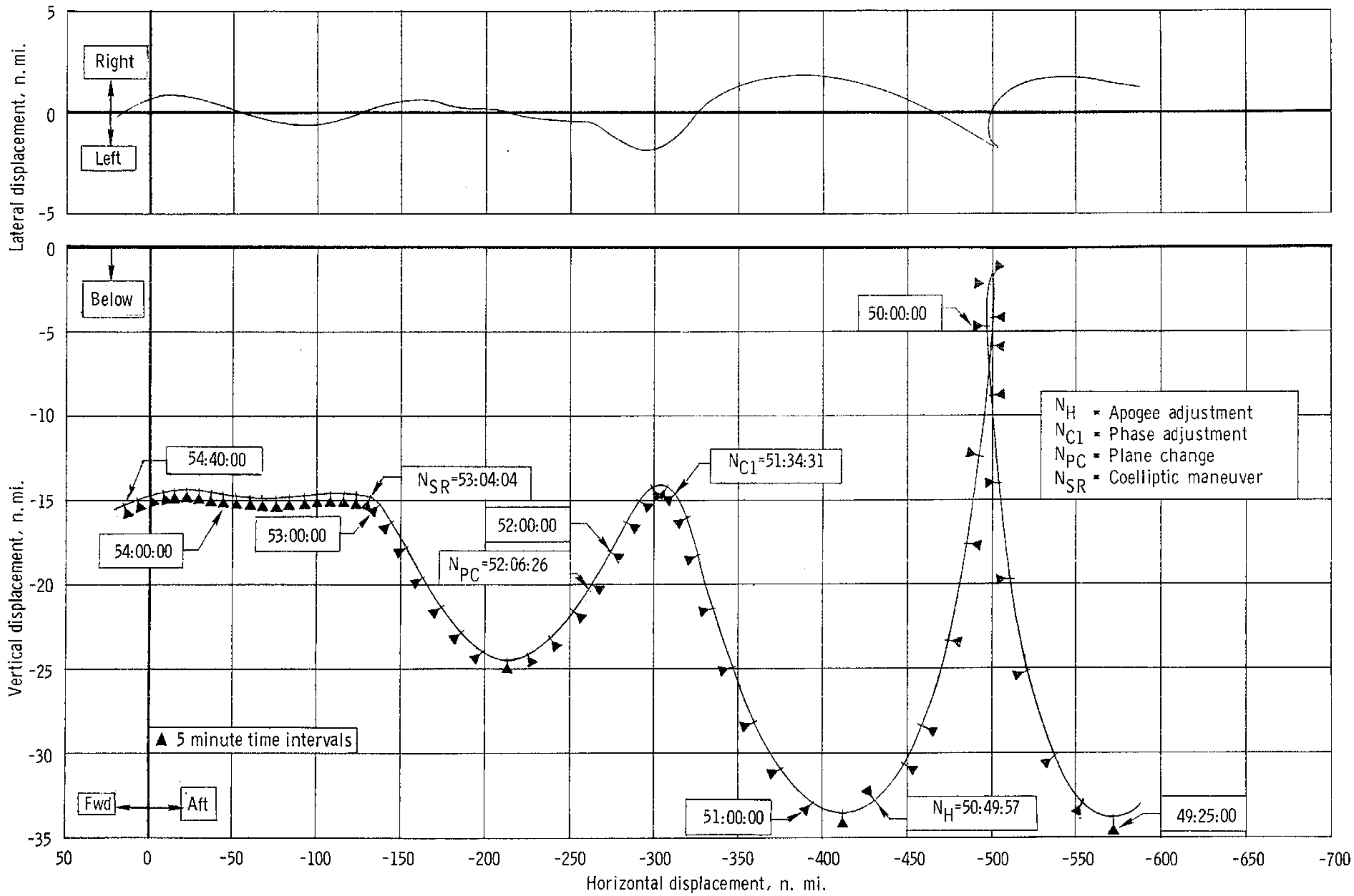

(b) Relative trajectory profile, measured from simulated Agena to Gemini V in curvilinear coordinate system.

Figure 4.3-6. - Concluded.

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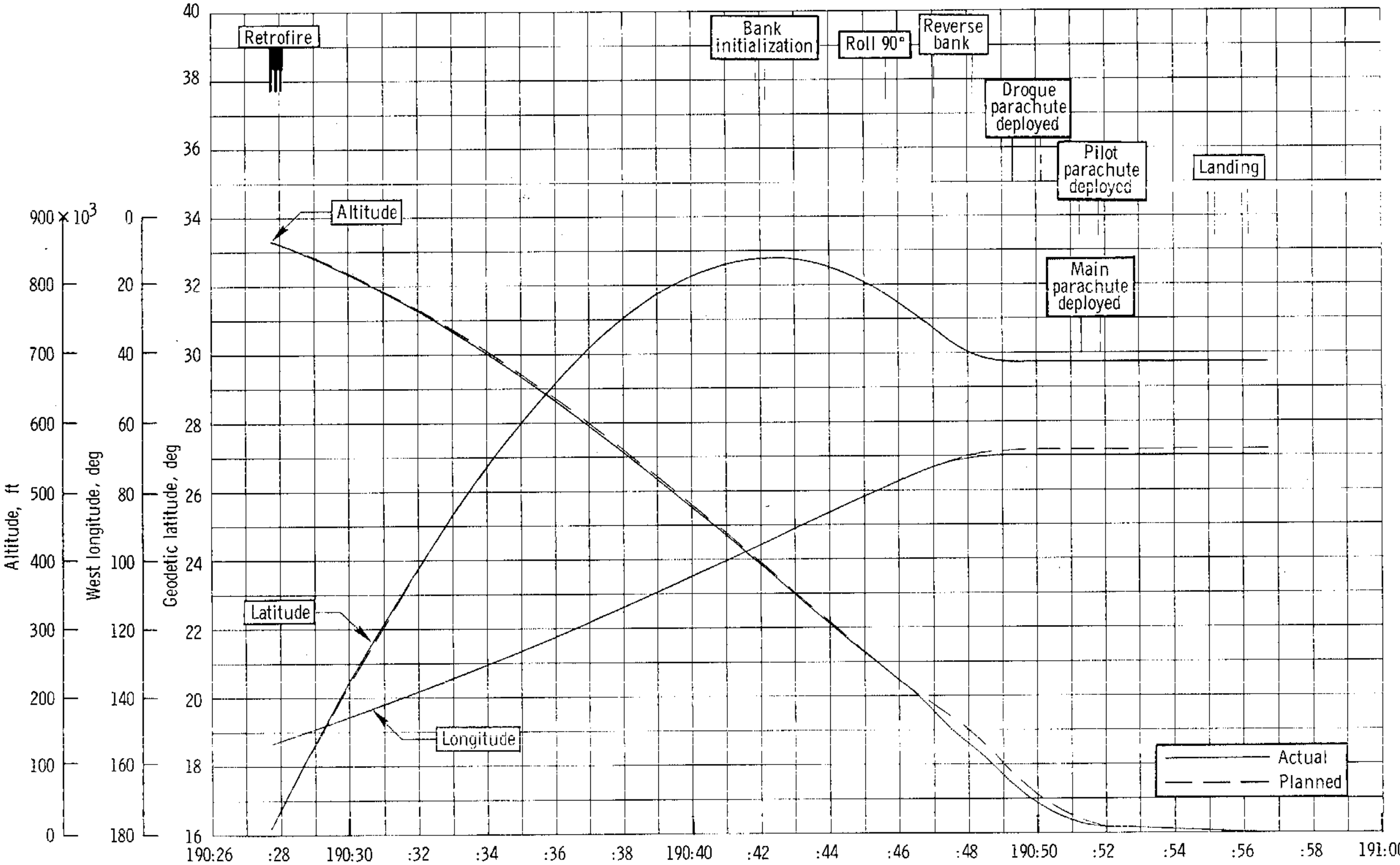

(a) Latitude, longitude, and altitude.

Figure 4.3-7. - Trajectory parameters for Gemini V mission reentry phase.

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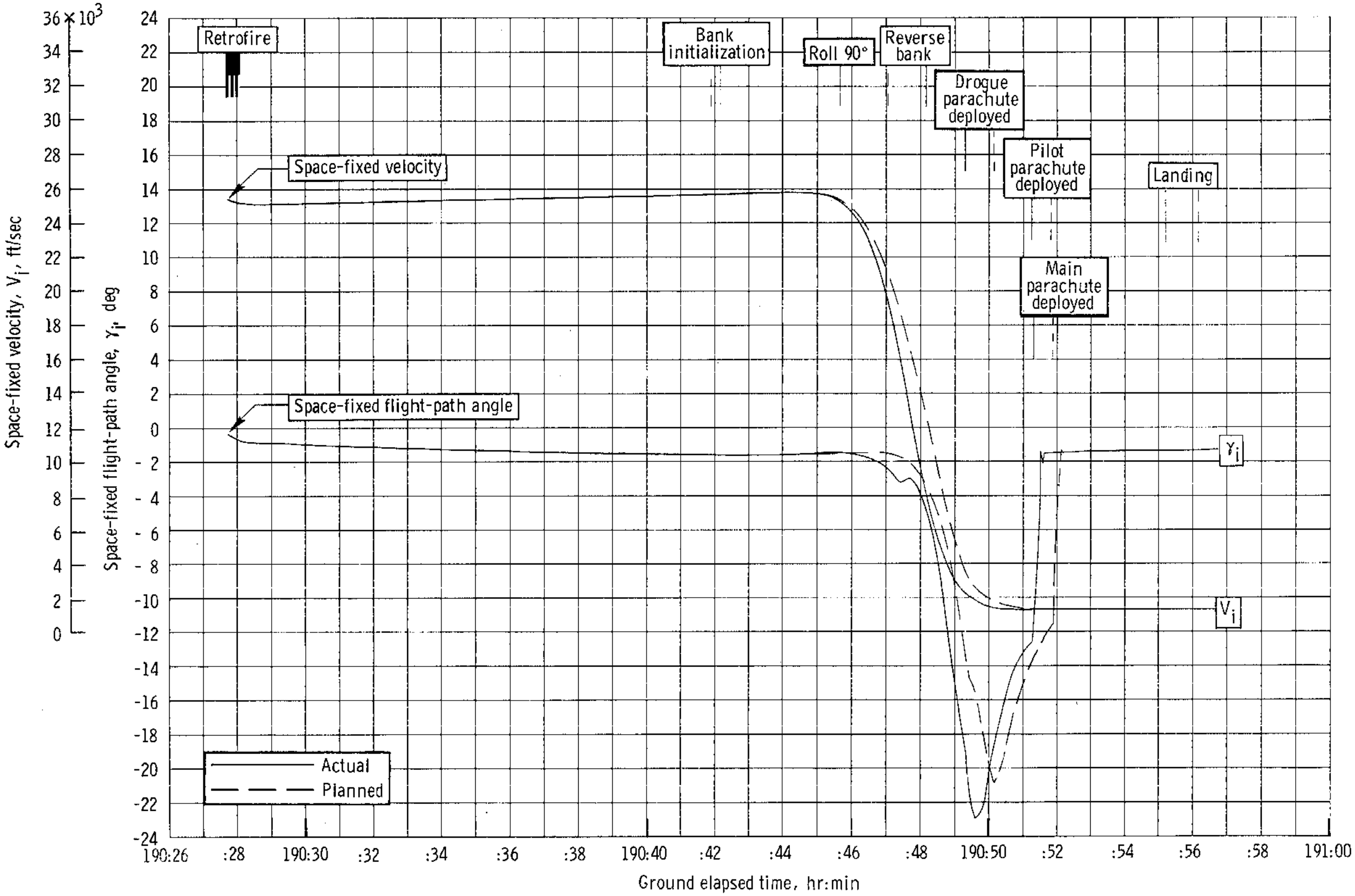

(b) Space-fixed velocity and flight-path angle.

Figure 4.3-7. - Continued.

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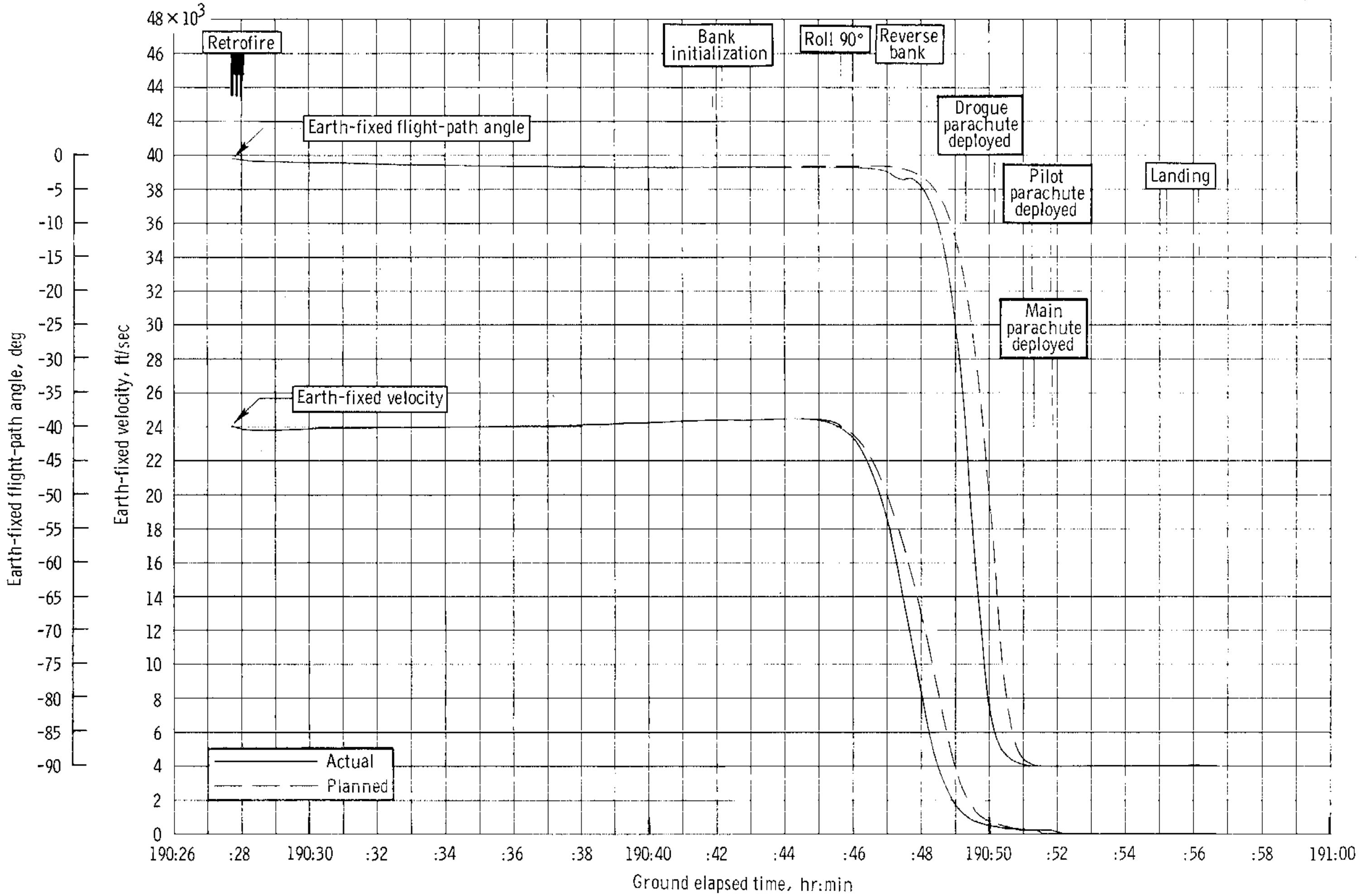

(c) Earth-fixed velocity and flight-path angle.

Figure 4.3-7. - Continued.

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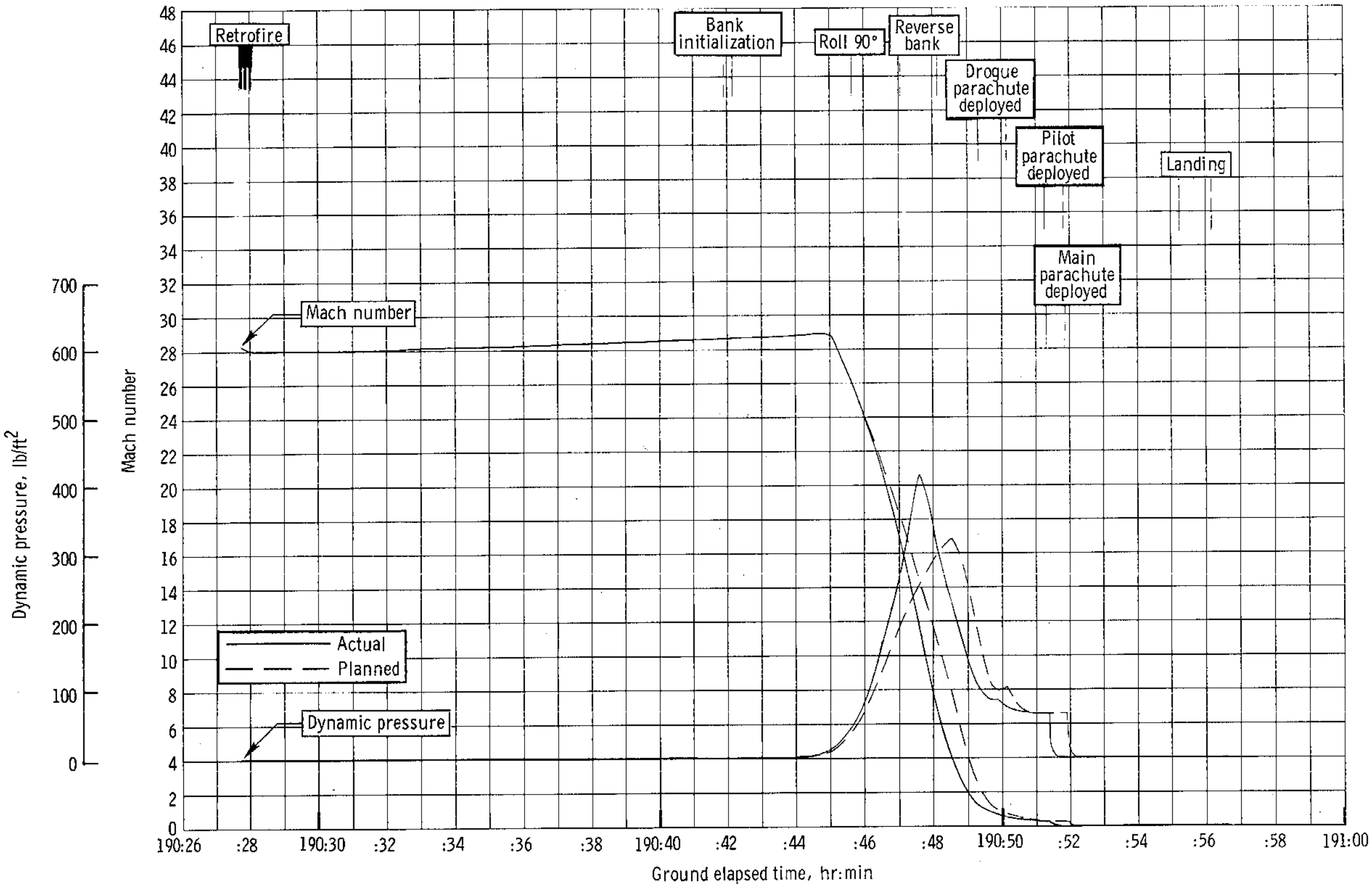

(d) Dynamic pressure and Mach number.

Figure 4.3-7. - Continued.

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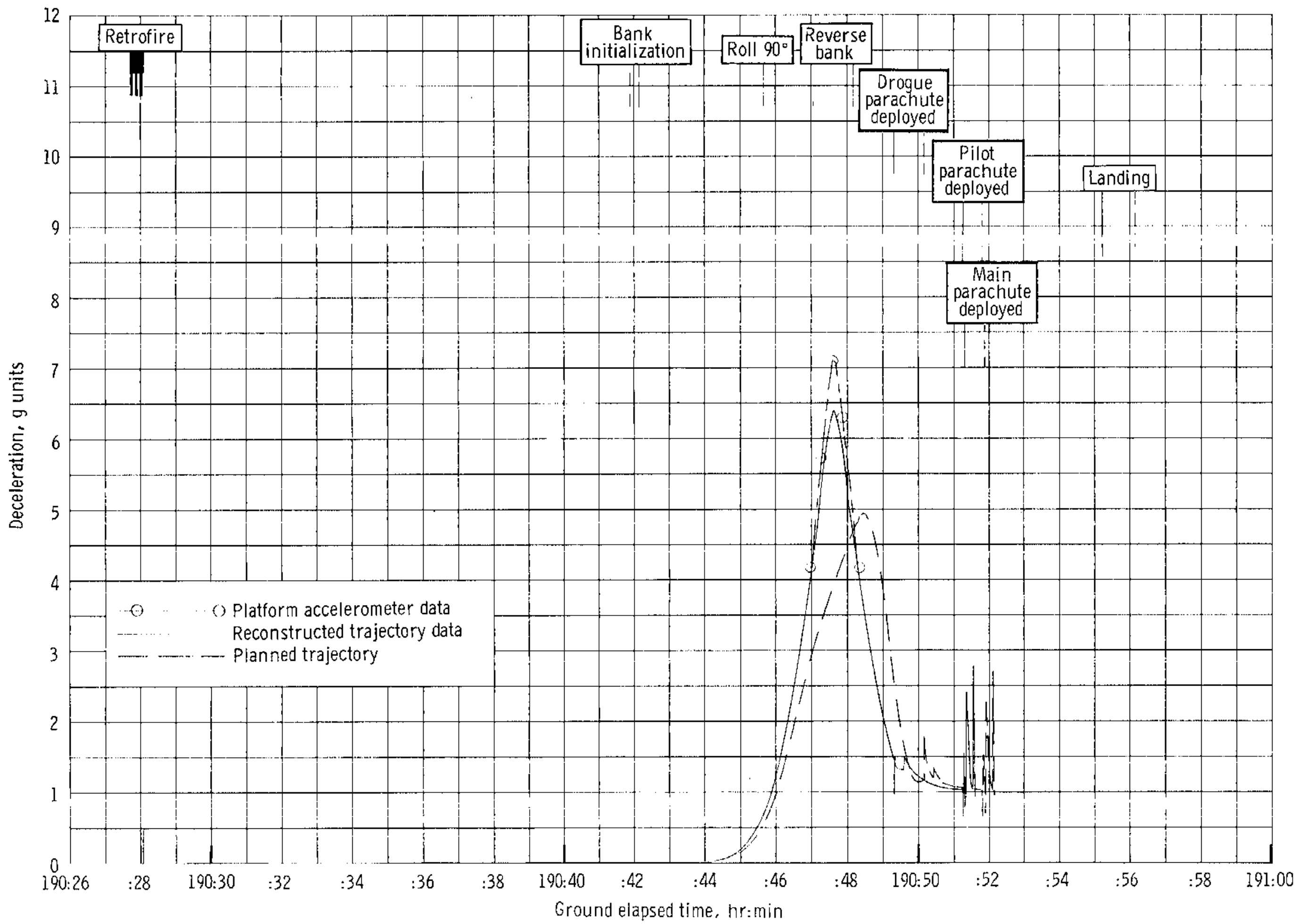

(e) Longitudinal deceleration.

Figure 4.3-7. - Concluded.

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5.0 VEHICLE PERFORMANCE

5.1 SPACECRAFT PERFORMANCE

5.1.1 Spacecraft Structure

The Gemini V spacecraft structure performed as expected for the flight in sustaining all loads, vibration, and heating in a satisfactory manner. Although the longitudinal oscillation (POGO) was approximately double that experienced on previous missions, the structural effects were inconsequential. Deployment of the drogue parachute at 69 000 feet, however, resulted in loading the frangible bolts of the rendezvous and recovery (R and R) section to a higher degree than would have existed at the normal deployment altitude of 50 000 feet. Normally, these bolts do not experience high loading until pilot parachute deployment. This occurs 2 seconds prior to the normal pyrotechnic separation of the R and R section for main parachute deployment (10 600 feet). Failure under the normal maximum load would not be catastrophic because the main parachute would not be jeopardized.

The paragraphs that follow describe the reentry aerodynamics and reentry heating.

5.1.1.1 Reentry aerodynamics.- As with the previous flights, the reentry trim angle of attack and lift-to-drag ratio for Gemini V were computed from gimbal-angle, accelerometer, and tracking data. The preliminary data agree quite well with preflight predictions, as shown in figure 5.1.1-1; however, these data disagree substantially with observations of the apparent stagnation point obtained from the postflight ablative pattern of the heat shield. The ablative pattern indicates the stagnation point was 12.6 inches down from the center line of the spacecraft, whereas the computed trim angle indicates that it should have been about 22 inches down.

The comparisons of heat-shield pattern stagnation-point measurements with computed trim-angle stagnation-points distances are shown in figure 5.1.1-2 for the four Gemini reentries. The data from the rolling ballistic reentries of GT-2 and Gemini IV correlate very well, whereas the data from the bank-angle reentries of GT-3 and Gemini V do not correlate. It is not known at this time why the heat-shield pat- pattern stagnation point varies so widely from the computed trim-angle stagnation point for Gemini V. However, Gemini V reentered from a higher altitude and through a different Reynolds number regime from that of the other flights as shown by figure 5.1.1-3. The ablative patterns that were observed on the recovered heat shields are the result of an integration of varying aerodynamic flow, convective heating, and instantaneous angle-of-attack conditions. Therefore, it is suspected that the rolling ballistic reentries show better correlation because the heating and dynamic pressure usually peak within a relatively short time of each other as compared with that of lifting reentries.

5.1.1.2 Reentry heating.- The Gemini V spacecraft was recovered in excellent condition after reentry heating. Inasmuch as there were no outer skin thermocouples on the reentry assembly afterbody or heat shield, only qualitative heating results based on detailed postflight physical observation are reported.

Afterbody shingles are clean and undamaged and in excellent condition, but show slight discoloration in the area behind the most windward spacecraft-adapter interconnect fairing as on previous flights.

The postflight condition of the heat shield is excellent and shows the white oxide appearance as was noted on GT-3 and Gemini IV. A preliminary examination of the heat shield indicates a char depth of 0.26 to 0.27 inch, which is nominal. The weight loss of the heat shield after drying was measured as 15.54 pounds.

The maximum zero angle-of-attack stagnation-point heating rate was calculated as 56.7 Btu/ft^2-sec, very close to the value of 57.8 Btu/ft^2-sec calculated for Gemini IV. Total reference stagnation-point heating was calculated to be 8660 Btu/ft^2. Total heat for Gemini IV was 8260 Btu/ft^2.

The windows of Gemini V were coated similarly to that experienced on GT-3 and Gemini IV. In addition to the thin coating of what is suspected to be ablation products, as experienced on previous flights, the Gemini V flight crew noticed several particles of gray putty-like material impinge on the windows during nose fairing and horizon sensor fairing jettison.

5.1.1.3 Miscellaneous structural items.- The crew reported that the nose fairing appeared to break up when it was jettisoned. This is believed to be the appearance given by small pieces of aluminized tape, ablative material, and possibly small superficial fiber glass parts which are bonded to the fairing. Such parts are too light to damage the spacecraft. The basic structure of the fairing has a margin of 41 percent above the 36 percent factor of safety, and limit load has been applied to it 12 times in qualification firings without failure. After the fairing had charred from launch heating, it is probable that most of the debris seen was char which had been jarred loose by the

3000-pound force of the ejector. The aluminized tape (around the lip of the R and R section on spacecraft 5 to obtain desired radar performance) will be replaced by a more durable RF gasket on later spacecraft.

The postflight inspection team at Cape Kennedy discovered evidences of water having been in the ECS well of the spacecraft. There were water stains on the inside of the ECS door and on the lithium hydroxide canister lid. An investigation was made to determine if there were possible leak paths through the door seal or through cabin purge valves which are installed in the door. No such leak paths were found, which would indicate there was no leakage through the structure while the spacecraft was on the water. From this indication, it must be inferred that the liquid which produced the stains was introduced into the interior of the spacecraft. Whether this occurred prior to flight, during the mission, or after the flight cannot be determined.

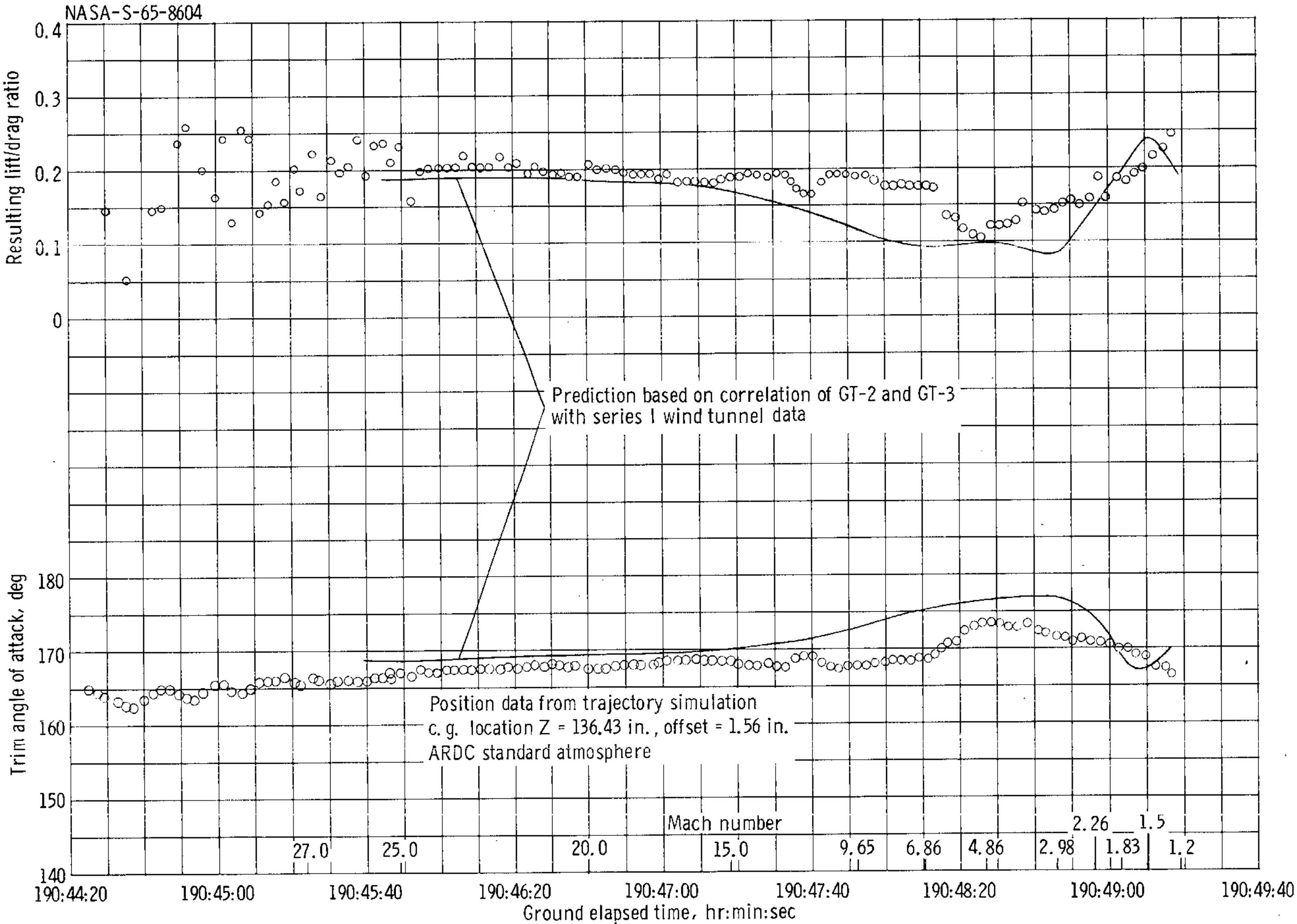

Figure 5.1.1-1. - Preliminary reentry angle of attack and lift-to-drag ratio.

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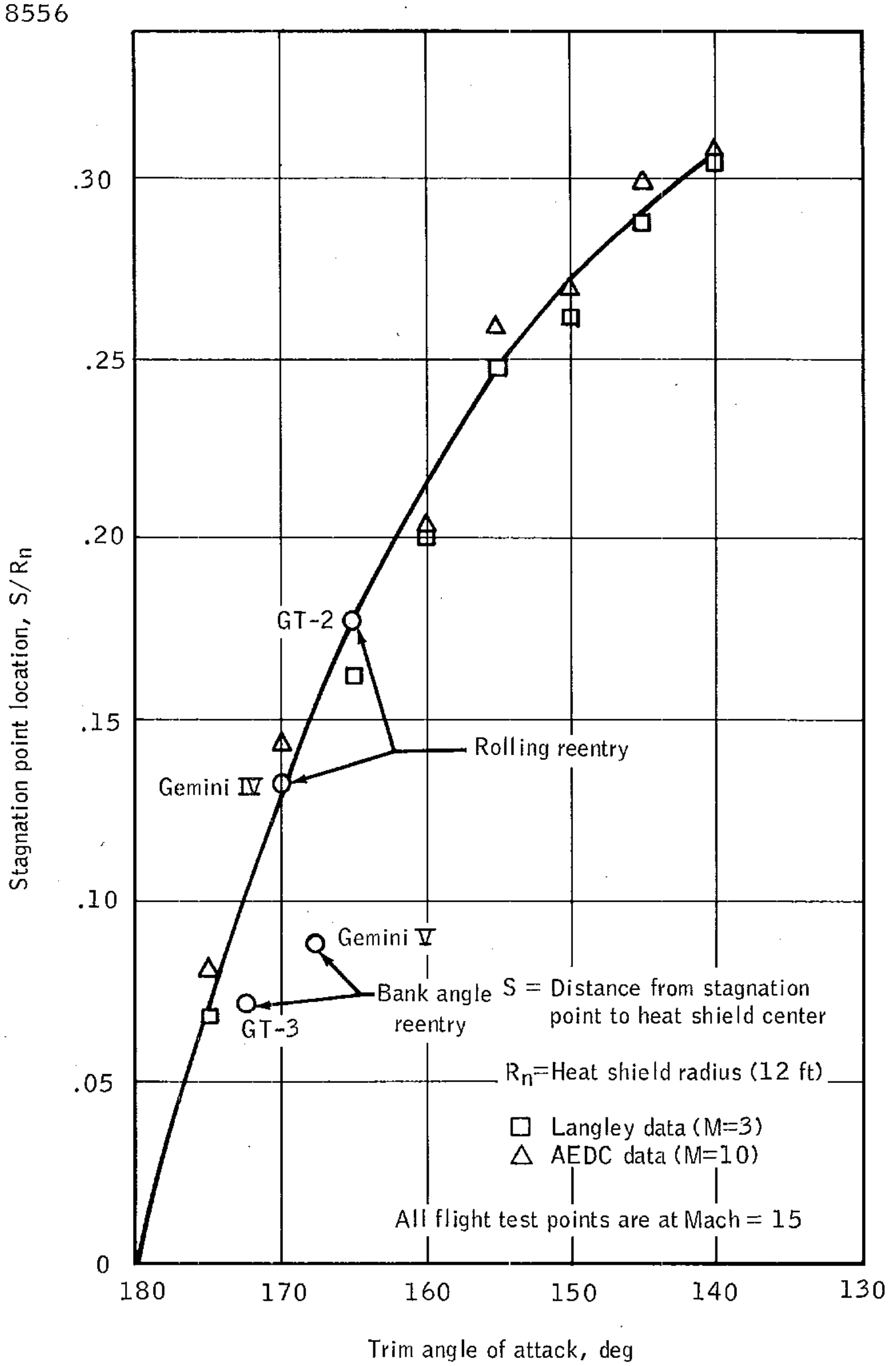

Figure 5.1.1-2. - Gemini stagnation point location.

NASA-S-65-9211

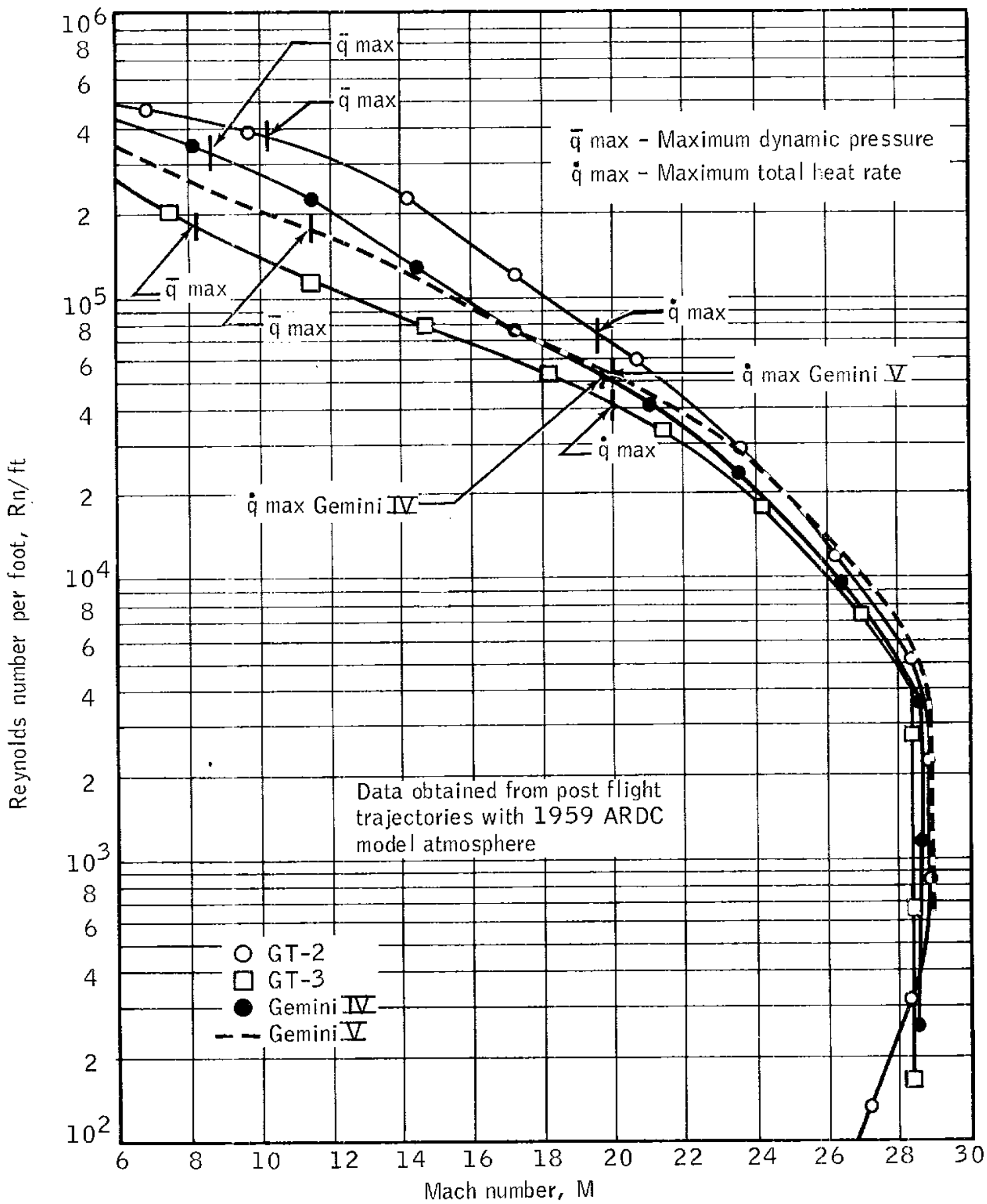

Figure 5.1.1-3. - Gemini Mach-Reynolds number environment during reentry.

5.1.2 Communications Systems

The Gemini V spacecraft communications equipment performed according to design without evidence of malfunction. The few problems which occurred are mainly attributed to operational errors or problems with the ground equipment discussed in the following paragraphs. The voice tape recorder completed recordings on five rolls of tape and then ceased to function. It was returned to the vendor for analysis where it was determined that the capstan drive belt was worn and broken in tension. The voice quality was nominal on the five tapes recorded.

Two new switches were added to the voice control center, effective with spacecraft 5. These were a voice tape recorder start switch that was separate from the communications mode switches, and a "sleep" switch to mute the earphones of either the pilot or the command pilot. Both were satisfactory to the crew. The continuous intercom push-to-talk keying mode was used throughout the mission.

New style lightweight headsets were used by the crew in orbit during this mission with satisfactory results. The crew reported, however, that the molded earpieces were uncomfortable when worn in the ear for long periods of time; therefore, they were left dangling near the ear for part of the time for relief.

There were some instances of voice communications degradation indicative of improper microphone placement. Breath noise was noticeable in the pilot's transmissions at intervals during the prelaunch phase of both the attempted launch and launch. This is usually caused by a microphone being located too close to the front of the mouth rather than at the side. During later revolutions, after the helmets were removed, the command pilot's transmissions usually contained a little higher background noise level than the pilot's transmissions. Background noise in this type of noise-cancelling microphone usually results from placing the microphone too far in front and too near the center of the mouth. These instances did not interfere greatly with normal communications; however, they could have been responsible for some of the few transmissions that were unreadable.

5.1.2.1 Ultra-high frequency voice communications.- The excellent ultra-high frequency (UHF) voice communications experienced during the Gemini V mission were indicative of normal simplex spacecraft equipment operation, together with a high order of support from the more complicated, remotely keyed, duplex ground network. One interruption occurred during the launch phase when the MCC-H spacecraft communicator was unable to hear the spacecraft. The remoted transmitter at MCC-C (Mission Control Center-Cape) was locked on because of an operational error, thus blocking the MCC-C receiver. The spacecraft replies, however, were recorded at Houston on the "Prime GOSS" tape, reaching Houston via some station other than MCC-C, although they were not heard by the spacecraft communicator, who was remoted through MCC-C. Communications were satisfactory during reentry; however, shortly after landing, a nearby recovery aircraft was unable to hear the spacecraft although the spacecraft was receiving and replying. This problem is attributed to other than spacecraft equipment because both the spacecraft and the recovery aircraft transmissions were recorded at Houston, as noted in the air-to-ground voice transcript. There were a few operational problems such as the MCC-H spacecraft communicator being unable to contact the spacecraft through a station which had not been instructed to go remote. Communications blackout caused by plasma attenuation during reentry occurred over a 3-minute, 52-second period from 190:44:06 g.e.t. to approximately 190:47:58 g.e.t.

5.1.2.2 High frequency voice communications.- The high frequency (HF) voice communications system, an emergency and back-up system, was not used for contingency purposes during the mission but was subjected to several tests. These tests consisted of one revolution each of ground-to-air and air-to-ground tone and voice. Analysis of the data and correlation between the time and position for the air-to-ground test is incomplete at this time. The tone was received at Cape Kennedy, Hawaii, and Texas; tone and voice were received at Guaymas; and nothing was received at California or Ascension. The ground-to-air test, with Hawaii transmitting, was heard by the flight crew only twice, once approaching Hawaii and once later near the Canary Islands. They are unable to fix the time; therefore, further analysis is impossible. The times were to have been entered on the spacecraft voice recorder, which, unknown to the crew, had failed; therefore, they were not entered in the flight log. HF voice communication was used several times within line-of-sight distance of a network station with results comparable to UHF. Music was played at MCC-H and was received by the spacecraft on the HF voice link when remoted through various network stations. Although this was not a planned test of the HF system, and very little data were recorded, it is of interest that the crew reported satisfactory reception during more than half of a revolution when the music was being transmitted from the Cape Kennedy and California stations. Figure 4.3-2 shows the point near the west coast of Africa where the crew reported loss of signal for music transmitted from Cape Kennedy on revolution 92. It also shows where they reported acquisition of signal off the east coast of Australia for music broadcast from California later in the same revolution. The antenna was not extended after landing; therefore, neither the HF voice nor the direction-finding tone was transmitted while on the water.

5.1.2.3 Radar transponders.- The radar transponder configuration was similar to that of Gemini IV and consisted of a C-band transponder in the adapter for orbital use and a second C-band transponder in the reentry assembly for launch and reentry use. Transponder operation was very satisfactory with no evidence of abnormal performance. The spacecraft was skin-tracked in many instances, and these data were used for ephemeris calculation as required.

5.1.2.4 Digital command system.- Reports from network station personnel indicate that the spacecraft digital command system (DCS) performed nominally throughout the mission. Shortly after launch, telemetry data indicated a low DCS power supply voltage. The operation of the DCS was not affected, and the measurement returned to normal immediately following each telemetry calibration. The trouble was apparently in the telemetry system and cleared after about 4 days, after which the measurement continued to indicate the correct voltage. See section 5.1.3.3 for an explanation of this anomaly.

On one occasion during the fourth revolution, a series of stored program commands (spacecraft computer update) was remoted from MCC-H through the Texas network station. The series was not fully received and stored by the spacecraft computer. At the time the commands were sent, the telemetry ground station had lost synchronization with the telemetry bit stream, and an oscillograph made from the telemetry video tape showed that the telemetry ground station sent many incorrect message acceptance pulses (MAP's) to the ground acceptance logic circuitry. This resulted in a continuous MAP and is believed to have caused the command words to be sent to the spacecraft at a faster rate than could be accepted by the DCS. Normally, if the words are sent too fast, each word is simply repeated until a valid MAP is received. This is believed to be a problem associated with the airborne telemetry and ground station DCS and is being investigated.

5.1.2.5 Telemetry transmitters.- The available data examined thus far indicate normal operation of all telemetry transmitters. The standby transmitter was switched over to the real-time telemetry during the launch phase and early portion of the first revolution, when a problem existed in the pulse code modulation (PCM) portion of the telemetry system. Later it was discovered that both transmitters operated equally as well with real-time modulation. The problem was identified as a spacecraft telemetry equipment problem and is discussed in section 5.1.3.1.

5.1.2.6 Antenna systems.- Judging from the performance of the communications systems, all UHF antennas deployed properly at the correct time and operated normally. The performance of the C-band radar adapter slot antenna and the launch and reentry helix systems was satisfactory as evidenced by radar performance. The HF antenna failed to deploy after landing. During tests after retrieval and shipment to Cape Kennedy, the antenna deployed properly when energized from the normal spacecraft power source. It was also given a leak test and found normal. The pilot stated that he may have inadvertently deenergized the control bus, which supplies power to the antenna extend mechanism, before trying to deploy the antenna. The HF orbit antenna mounted on the adapter deployed properly and was used extensively.

Transmission tests were run during this mission to aid in determining which of two systems, the UHF reentry antenna mounted on the rendezvous radar ground plane or the adapter-mounted UHF antenna, was more suitable for use in certain orbital conditions such as drifting flight or low elevation-angle station passes. The analysis of the test results is incomplete at this time; however, a few qualitative results are evident. (See table 5.1.2-I.) Further data and analyses are necessary to separate the effect of antenna switching on signal strength. As far as voice communications are concerned, either antenna may be used for normal station passes during drifting flight. For low-angle passes below about 5°, the reentry antenna is superior for drifting flight, and the adapter antenna is better with pitch and roll angles controlled to 5° or less from a 0°, 0° attitude.

UHF test no. 1 was planned as a worst-case look angle for the reentry antenna, with the reentry antenna pointed away from the tracking station at the point of closest approach. UHF signals were switched from adapter to reentry antenna at 20-second intervals. At the present time the spacecraft attitude angles have not been reduced from the telemetry data; therefore, it has not been determined whether this test was representative of a worst case. The telemetry signal strength recorder charts have not been analyzed. The UHF voice frequency signal strength varied from 20 to 400 microvolts through the pass. The station operator's log lists delayed-time telemetry signal strength as 4 microvolts peak, 1.3 microvolts average, and real-time signal strength as 72 microvolts peak, 25 average. This difference in maximum strength is explained in that the delayed-time transmitter was shut off very early during the pass while the signal strength was still low. A signal strength of 1.3 microvolts is marginal under average noise conditions; however, these particular data were excellent. More data must be examined before antenna selection can be recommended.

UHF test no. 2 was planned as a worst-case look angle for the adapter antenna. It was planned that the spacecraft attitude be held constant with the adapter antenna pointed away from the station at the point of closest approach. UHF signals were switched from the adapter to the reentry antenna at 20-second intervals. This test provided only limited information for the following reasons. The roll angle varied as much as 30° from the planned angle, the pitch angle varied 6.5°, and the yaw angle varied 17.5°. Therefore, this was not a worst case for the adapter antenna. The test was started about 1 minute later and ended about 2 minutes earlier than planned; therefore, it was not run at the longer ranges and lower tracking antenna-elevation angles. The operator's log listed delayed-time peak telemetry signal strength as 13 with an average of 7 microvolts, and a real-time peak of 40 with an average of 13 microvolts; however, the signal strength recorder chart was not available for verification of these values. UHF voice frequency signal strength was 250 microvolts peak with a 55 average, which is adequate.

UHF tests 3 and 4 were planned for passes where the network-station tracking antenna maximum-elevation angle would be low. Orbit attitude in pitch, roll, and yaw was to be maintained at $0° \pm 5°$. Test 3 utilized the adapter antenna with a maximum tracking antenna elevation of about 2°. Test 4 utilized the reentry antenna with a maximum tracking antenna elevation of about 6°. The telemetry signal strength recorder charts have not been analyzed. UHF voice signal strength was 50 microvolts average for test 3 and 30 microvolts average for test 4. The adapter antenna was clearly superior under these test conditions; however, either antenna provided acceptable communications. Both real and delayed-time telemetry signal strengths were low during these tests, as reflected by the station operator's log. The signal strength recorder charts have not yet been examined.

5.1.2.7 Recovery aids.- All communications recovery aids operated normally during the Gemini V mission. The flashing light extended normally, was turned on by the crew, and was operating at normal intensity when observed from the aircraft carrier. The external intercommunication jack provided communications with the rescue personnel before the hatches were opened. The recovery beacon was received at distances of 80 and 120 miles by aircraft. One recovery aircraft requested and received three direction-finding transmissions from the spacecraft UHF voice transmitter. The rescue packs were not opened; therefore, the rescue beacon transceivers were not used.

TABLE 5.1.2-I.- GROUND RECEIVED SIGNAL STRENGTHS DURING

THE SPECIAL UHF VOICE COMMUNICATIONS TESTS

UHF antenna system	Test no.	Maximum tracking elevation angle, deg	Logged signal strength, microvolts					
			Voice		Delayed-time telemetry[a]		Real-time telemetry	
			Peak	Average	Peak	Average	Peak	Average
[b]Adapter and reentry	1	76	400	200	4	1.3	72	25
[b]Adapter and reentry	2	48	250	55	13	7	40	13
Adapter	3	2	-	50	4	4	7	6
Reentry	4	6	-	30	13	7	7	6

[a]Delayed-time telemetry had been utilized for recorder playback over MCC-C and was commanded off early during each Bermuda pass.

[b]Data examined thus far does not permit separation of the effects of switching the antennas.

5.1.3 Instrumentation and Recording System

An examination of the real-time and delayed-time telemetry data available revealed that the following anomalies occurred during the mission:

(a) Intermittent loss of high- and low-level multiplexer data during the ascent and reentry phases.

(b) Delayed-time data losses during revolutions 30 through 45.

(c) Intermittent operation of the DCS -18 V dc monitor.

5.1.3.1 High-level and low-level multiplexer data losses.- During the ascent phase of the mission, all low-level and high-level multiplexer data were lost from lift-off to 4.6 seconds and from 36.7 seconds to 95.5 seconds. Postflight testing of the spacecraft wiring revealed a loose connection in the -24 V dc return line from the PCM programer. A loose nut on terminal 4 of terminal block 8 was found to cause as much as 50 000 ohms resistance in the line between the -24 V dc power source and the PCM programer. In vendor tests conducted to simulate this fault, the low-level multiplexer channels dropped out first, the high-level multiplexer channels dropped out next and then, in some cases, the programer synchronization would be lost as higher values of resistance were inserted into the -24 V dc line to the programer. It is also very probable that data dropouts at mission times during which thruster firings produced any noticeable acceleration, at equipment adapter separation, during retrofire, at drogue deployment, and at inversion to landing attitude were caused by this same loose connection. It is also very probable that the high number of resets during this mission (11 were confirmed on revolution 3) was caused by the noise pickup brought on by the impedance change in the -24 V dc line.

5.1.3.2 PCM tape recorder poor dumps.- The quality of the delayed-time PCM data obtained from the PCM tape recorder for orbits 30 through 45 was poor. This poor quality was caused by damaged tape for the first 90 minutes of recording time and a partially magnetized record-playback head. The partially magnetized head was used throughout the mission for both the good and bad portions of the tape; and since good quality data were obtained from the good tape portion, it can be concluded that the partially magnetized head alone did not result in poor data, but when it was combined with the bad tape portion, poor or marginal data resulted.

An investigation by the vendor revealed that throughout the first part of the bad tape portion, the iron oxide was missing from the tape in such a manner that transparent spots were present. The tape is a mylar-base material with an iron oxide coating. It was also discovered that several raised spots on the record-playback and erase heads were present. These raised spots were no larger than a pinhead, were extremely hard, and could not be removed by normal cleaning of the heads with freon. The fine scratch marks on the tape, present in the direction of tape motion, have been attributed to scratching of the tape by the two raised spots on the record-playback head. A chemical analysis of these raised spots was attempted by the vendor, but was unsuccessful because of the minute quantity of material available. Further investigation by the vendor into all work, procedures, et cetera, at the vendor plant prior to the reacceptance of this recorder on July 11, 1965, revealed that the record-playback head was realined prior to the acceptance test and "Loctite" was used to hold the head in place. Further testing has shown that this same "Loctite" will weaken the binder which adheres the iron oxide to the mylar-base tape and the iron oxide will then peel away. The vendor concludes that "Loctite" must have been inadvertently splashed on one of the rollers and blotted off as the tape passed over the roller, until all "Loctite" was transferred to the tape. Repeated use of the tape from acceptance test through revolution 29 of the mission caused sufficient iron oxide to work loose, partly cake on the heads, and result in poor data starting with the revolution 30 dump. The Cape Kennedy telemetry station number 2 (TEL II) noticed a gradual degradation in the delayed-time data signal starting with the revolution 14 data dump through the revolution 18 data dump, even though the quality of the overall data was still good. This supports the conclusion reached by the vendor that the "Loctite" weakened the iron oxide binder and the first portion of the tape became degraded as the iron oxide gradually peeled away from the spots which had been in contact with the "Loctite."

The partial magnetization of the record-playback must have resulted from the use of a magnetized wrench, screwdriver, or other tool either at the vendor or at the launch site. Procedures at both the vendor's plant and at the launch site are being reviewed to prevent a recurrence of this problem. By recording both revolutions 46 and 47 and then dumping only revolution 47, thereby shifting operation to a relatively unused portion of the tape, good quality dump data were obtained after revolution 46.

5.1.3.3 <u>Intermittent operation of the DCS -18 V dc monitor</u>.- The DCS -18 V dc monitor read -12.6 V dc throughout most of the first half of the mission instead of the nominal -18 V dc value. The DCS would not operate if this voltage value were valid, and because proper DCS operation was being obtained, an inflight calibration of the instrumentation system was made when this first appeared. The value returned to -18 V dc immediately after calibration and then returned to the -12.6 V dc value after a period of time. After several calibrations, the voltage reading remained constant at the -18 V dc level for the remainder of the flight. It would appear that the instrumentation package no. 1 calibrate relay contacts in the -18 V dc monitor circuit were dirty or making poor contact. By repeated calibrations, either the relay contacts were cleaned or the contact flexures restored the circuit to normal operation. This equipment was located in the adapter section, and, therefore, failure analysis of this intermittent operation cannot be performed.

5.1.3.4 Delayed-time data quality.- Even though the ground telemetry stations reported that the dump data were degraded to only 50 percent usable during revolutions 30 through 45, the edit program data from computer processing of the video tapes recorded at these stations confirm a total data loss per revolution on the order of only 5 percent. Data for revolutions 30, 32, 33, 34, 35, 36, and 37 were computer-reduced, and resulted in complete loss of 1.0, 0.5, 3.1, 3.7, 4.9, 4.5, and 2.9 percent, respectively. Data were not uniformly degraded over the entire revolution, the first portion being far more degraded than the last. (That is, on revolution 37, of the first 15 minutes of recorded data, 24.9 percent of the data was extracted by existing programs; and the maximum that would have been possible to extract, by extremely difficult manual operations, was probably on the order of 75 percent.) This process was not attempted.

During the period when the onboard recorder was operating over the bad tape portion, the aforementioned small data losses were tempered considerably by the recorded bit stream degradation which was not revealed by the computer-processed data edits. This degradation made normal processing of the data very difficult because bits in the synchronization word were often affected, and the raw data could not be recognized for formatting. A few of these periods involved data which were crucial to evaluation of several spacecraft systems performance (that is, revolutions 32, 33, 34 data for evaluation of the simulated Agena rendezvous). The quality of the recorded tape on revolutions 33 and 34 was such that MCC-C failed to format it after repeated attempts, and the spacecraft contractor was not able to recover significant portions of the data from revolution 33. It was only after repeated attempts and concentrated manual control of reduction equipment that the data from some of these tape segments were recovered at MSC, Houston. This effort continued intermittently over a 2-week period before it was successfully concluded. It should be recognized that most of the delayed-time data for these revolutions was only partially usable and in some cases completely unusable for playback in immediate support of the mission.

The delayed-time data received by TEL II, Texas, Hawaii, Antigua, and the Rose Knot Victor (RKV) telemetry ground stations, as well as the data recovered from the onboard PCM tape recorder, are summarized in table 5.1.3-I. This represents the computer processing of 56 data dumps out of the 116 dumps actually made. For all the ground stations listed, as well as the onboard PCM recorder data, the usable data exceeded 98.67 percent, and for the onboard PCM recorder alone, the usable data recovered were 99.60 percent. The slightly lower figure for the data dumped to the ground stations is attributed to the aforementioned tape-recorder problem. The PCM system and the recorder continued to operate until 190:00:28.4 g.e.t. or 5 minutes 14.6 seconds after landing when the tape ran out as planned. Data were recorded up to that time.

5.1.3.5 Real-time data quality.- Table 5.1.3-II lists the real-time data received by the ETR telemetry range station (TEL II) for various mission phases. From the columns of total losses and valid data, it can be seen that the usable real-time data are more than 99.46 percent for all cases. These figures were also obtained from the computer-processed edit program to determine usable data.

In this mission, there were a total of 285 parameters monitored, and data were received on each parameter.

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TABLE 5.1.3-I.- PERCENT OF USABLE DELAYED-TIME DATA FROM SELECTED STATIONS

Station	Revolutions	Total data received		Total losses (a)		Usable data, percent
		Duration, hr:min:sec	Prime subframes	Prime subframes	Percent	
Tel II	14, 15, 16, 17, 18, 19, 32, 33, 47, 48, 49, 59, 60, 73, 74, 75, 76, 77, 106, 112, 117, 118	19:49:50	713 898	14 137	1.98	98.02
Texas	1, 2, 18, 19, 34	06:01:51	217 101	2 048	0.94	99.06
RKV	9, 11, 23, 53, 54, 55, 112,	12:33:00	451 800	6 738	1.49	98.51
Hawaii	5, 7, 8, 20, 21, 22, 35, 36, 37, 79, 81, 111	15:21:26	552 858	7 031	1.27	98.73
MCC	Launch, 1, 2, 3, 4, 30, 31 32, 33, 118, 119	09:06:08	327 677	1 226	0.37	99.63
Antigua	28, 115	02:38:13	94 925	1 438	1.52	98.48
Onboard recorder	74, 120, reentry	03:44:27	134 670	533	0.40	99.60
	Total	69:14:55	2 492 929	33 151	1.33	98.67

[a]Based on a computer search of the raw data for usable PCM telemetry 8-bit binary words.

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TABLE 5.1.3-II.- PERCENT OF USABLE REAL-TIME DATA RECEIVED BY TEL II

Revolution	Total data received		Total losses (a)		Usable data, percent
	Duration, sec	Total master frames	Master frames	Percent	
Launch	184	5 756	2	0.03	99.97
1-2	374	14 975	7	0.05	99.95
2-3	340	13 611	4	0.03	99.97
14-15	354	14 138	2	0.01	99.99
15-16	428	17 137	4	0.02	99.98
16-17	409	16 339	13	0.08	99.92
17-18	429	17 138	2	0.01	99.99
18-19	438	17 502	7	0.04	99.96
29-30	396	15 855	1	0.01	99.99
30-31	430	17 181	7	0.04	99.96
31-32	414	16 540	14	0.08	99.92
32-33	431	17 221	93	0.54	99.46
33-34	411	16 455	2	0.01	99.99
43-44	377	15 059	15	0.10	99.90
44-45	477	19 085	82	0.43	99.57
45-46	461	18 418	2	0.01	99.99
48-49	400	15 983	49	0.31	99.69
58-59	437	17 463	11	0.06	99.94
73-74	478	19 102	0	0.0	100.00
74-75	486	19 422	18	0.09	99.91
89-90	323	12 924	0	0.00	100.00
102-103	365	14 614	63	0.43	99.57
107-108	387	15 494	32	0.21	99.79
[b]120-121	100	3 980	1	0.03	99.97

[a]Based on a computer search of the raw data for usable PCM telemetry 8-bit binary words.

[b]Preblackout.

5.1.4 Environmental Control System

The performance of the environmental control system (ECS) was within specification during all phases of the mission. The cabin pressure regulator relieved at 5.8 psid during launch and closed off at 5.6 psid. Cabin pressure then decreased slowly to 4.9 psid shortly after insertion and was automatically controlled at this value for the rest of the mission. The launch cooling heat exchanger performed as expected with no apparent freezing. The spacecraft yaw resulting from the water boiler exhaust observed on GT-3 flight was experienced during Gemini V until 45 minutes g.e.t., at which time the radiator outlet temperature dropped to a sufficiently low value for the launch cooling heat exchanger to automatically stop functioning. Suit inlet temperatures varied between 47° F and 60° F; cabin temperature was 89° F at lift-off and varied slowly between 70° F and 80° F during orbital flight.

The space radiator and coolant loop maintained excellent thermal control throughout the mission. When the reactant supply system (RSS) problem dictated a severe reduction in power, resulting in abnormally low thermal loads, the temperature control valves maintained normal coolant temperatures. This demonstrated the control capability of the coolant system over an extremely wide variation of thermal conditions.

When suit inlet temperatures were below 50° F, the crew reported being so cold that they had to restrict cooling from the suit heat exchanger to a minimum and change the suit configuration in order to maintain comfort. Minimum suit heat-exchanger cooling was accomplished by combined use of the coolant control valve and the suit-flow control valve. Adjustment of these control valves corrected the discomfort of the pilot but not the command pilot. Donning of the wrist dams by the command pilot so as to have the same suit configuration as the pilot (that is, helmet and gloves off with wrist and neck dams on) resulted in satisfactory comfort level for both crewmen.

The only moisture observed in the cabin during flight was on the window pane. This occurred only when the crew exhaled in the near proximity of the window and when the spacecraft was tumbling. Cabin relative humidity indicated between 53 and 72 percent throughout the mission. For these readings, however, the wet bulb readings varied from 58° F to 67° F with an average depression below dry bulb of 12.4° F. Large depressions of this magnitude in wet-bulb temperatures are difficult to obtain, and the depressions were probably greater than indicated; thus the true cabin humidity was probably less than that reported. This was substantiated by the excessive drying of the skin, in particular, the finger nails and scalps of the crew.

The drinking water system provided the crew with a sufficient amount of cool drinking water; however, despite efforts to perform vacuum servicing with deaeriated water, the water still contained gas (although much less than on Gemini IV). Servicing procedures will be reviewed prior to Gemini VI in an effort to further reduce the amount of gas inclusion.

The crew reported a reading of 1 mm Hg on the carbon dioxide partial-pressure indicator for a short period. This change from zero is within the limits of accuracy for this sensor, and the actual CO_2 partial pressure was verified as being less than 4 mm Hg by the use of hand-held tapes. The indicator later returned to 0 mm Hg for the remainder of the mission. (The 2 mm hand-held tape window malfunctioned.)

In an effort to explain the eye irritation experienced by the Gemini IV crew, examination of the flight clothing indicated small amounts of lithium from the lithium hydroxide container. The Gemini V crew reported no eye irritation. A partial examination of the crew's underwear revealed only traces of lithium. Tests are being conducted to permit an estimate of the total amount of lithium hydroxide which came out of the lithium hydroxide canister. Results to date indicate the quantity will be below the specification limit of 0.18 mg/hr.

5.1.5 Guidance and Control System

5.1.5.1 Summary.- Table 5.1.5-I lists events significant to the guidance and control system. Inertial guidance system (IGS) performance was excellent throughout the flight, and none of the anomalies or malfunctions experienced on previous missions were exhibited. The onboard radar performed nominally during the rendezvous evaluation pod (REP) exercise and on the first pass over the ground transponder located at the Merritt Island Launch Area (MILA). On three subsequent passes, the digital range readout failed to indicate correctly (see section 5.1.5.4.2) although nominal lock-on indications and pointing data were received. Performance of the control system was excellent with the exception of an apparent failure of the primary horizon sensor (see section 5.1.5.4.1) and the progressive loss of thrusters late in the mission. No evidence of incorrect performance of the system in the new "platform" mode (as reported by the crew in section 7.1.2) has been found in the data available. The reentry miss was caused by an incorrect ground update of the onboard computer.

5.1.5.2 IGS performance evaluation.-

5.1.5.2.1 Ascent phase: The IGS pitch, yaw, and roll steering signals are shown in figure 5.1.5-1. Superimposed on these quantities are the steering signals from the primary guidance system along with the upper and lower IGS limits which were generated by assuming nominal operation of the primary guidance system. The following is a brief discussion of the steering signals with respect to stage I and stage II flight. IGS performance during the ascent phase was excellent.

The difference in the roll steering commands between the two guidance systems just prior to BECO was about 1.2°. Gimbal cross-coupling contributed at least 0.6° and roll misalinement or programer deviations about 0.2° more. The remaining difference of 0.4° was probably a Gemini launch vehicle (GLV) three axis reference system (TARS) roll gyro drift. The offset of the roll steering command from the primary guidance system of 0.4° to 0.6° during stage I indicates an engine misalinement on the launch vehicle.

The difference in the yaw steering commands between the two guidance systems was about 0.5° at BECO. Gimbal cross-coupling again contributed at least 0.3°, with the remaining 0.2° probably caused by initial misalinement and TARS gyro drift. The effect of an offset center of gravity was very pronounced on this flight, as indicated by a 1.0° shift at staging from both systems.

At BECO there was a 1.2° difference between the two pitch steering commands which included an initial misalinement of about 0.2° between the two systems and another 0.2° due to a 0.25-second early IGS pitch step. The remaining 0.8° was probably a pitch programer deviation or a TARS pitch gyro drift, or both. The 0.5°-shift at BECO of both steering signals again indicates a large center-of-gravity offset.

The behavior of the IGS pitch steering command during stage II was near nominal. The deviations observed at about 225 seconds and 285 seconds were normal reactions to changes in steering logic at these times. The IGS yaw steering command was also near nominal until about 320 seconds when it started a slow deviation to about -1.0° at SECO. This was probably the effect of an out-of-plane velocity deviation between the two systems since the IGS indicated a 10 ft/sec out-of-plane velocity at SECO. The steering command was derived by dividing the out-of-plane velocity by an effective time-to-go to SECO. As the vehicle approached SECO, this ratio diverged as did the steering command. At about 336 seconds (during sustainer tailoff) the IGS yaw command shifted approximately 2.0° from about -2.6° to about -0.6°. This was the result of a Math Flow no. 6 programing error and has been corrected for all subsequent flights.

After the gimbal cross-coupling and other deviations during stage I were subtracted from the roll steering command in stage II, the remainder of the difference between the two guidance systems was representative of a TARS linear drift of about 5.0 deg/hr.

Both azimuth updates were received with flight reconstruction simulations indicating the following values for platform misalinement:

Platform release, deg	0.04
After first update, deg	-0.27
After second update, deg	-0.26

This misalinement is well within the specified 3σ value of 0.75°.

If guidance switchover had occurred early in stage II operation, the SECO conditions would have shown the following deviation from nominal: 3.0 ft/sec in velocity, 0.02° in flight-path angle, and 700 feet in altitude. This deviation would have resulted in an apogee of 192.7 nautical miles and a perigee of 86.9 nautical miles. The programing error in the IGS computer mentioned previously essentially eliminated the effect of yaw gimbal-angle movement in determining the yaw attitude error between SECO + 3.5 seconds and SECO + 20 seconds. As a result, if a switch to backup guidance had occurred, vehicle yaw attitude and yaw rate at separation would have been incorrect and would have caused approximately a 1.0 ft/sec out-of-plane velocity error.

The incremental velocity indicator (IVI) display, as actually computed by the onboard incremental velocity adjust routine (IVAR), was reconstructed by using IGS navigational and gimbal-angle data. The crew reported readings of 2 ft/sec forward, 13 ft/sec right, and 2 ft/sec down, which approximate the readings calculated near the end of the roll maneuver. Table 5.1.5-II shows the values of the reconstructed IVAR parameters in their final computation cycle as compared with the actual final values obtained in the prelaunch mode through the data acquisition system (DAS). The increase in the computed required incremental velocity along the $Y_{S/C}$ axis was the result of the spacecraft being inserted with a measured out-of-plane velocity (combination of slight errors in both the RGS and IGS) and an increasing out-of-plane IVAR correction being required to achieve the desired orbit plane.

If the IVAR had been used on this flight, following the separation maneuver, the IVI's would have displayed 1.7 ft/sec forward and 11.5 ft/sec out-of-plane velocity corrections in component form. When the pitch and roll attitude errors had been nulled, the IVI's would have displayed 2 ft/sec forward and 12 ft/sec right with the yaw attitude error needle indicating a limited yaw-right maneuver. To null the yaw attitude error, the spacecraft would have yawed about 80° right, and the resultant correction of 12 ft/sec forward would have appeared on the fore and aft window. Driving this 12 ft/sec reading to zero would have changed the in-plane velocity about 1.7 ft/sec resulting in an apogee of about 1.0 nautical mile higher than actually achieved and would have "corrected" the erroneous out-of-plane error by about 11.5 ft/sec. Relatively no velocity change at apogee V_{gp} would have been required to reach the desired perigee of 87 nautical miles.

Performance of the inertial measurement unit (IMU) was excellent during the ascent phase, with none of the malfunctions that had occurred on previous flights. Telemetry data were of usable quality except for a 38-second drop-out starting at LO + 48 seconds. The GE final tracking data were adequate for quick-look analysis until LO + 280 seconds, at which time the vertical component became very noisy because of the decreasing elevation angle. The missile trajectory measurement (MISTRAM) 10K tracking data agreed with the GE data within 1 ft/sec up to LO + 280 seconds when it also became noisy, particularly in the vertical component. MISTRAM 100K data were poor since the P-calibrate channel, used to correct the vertical velocity component, did not continuously update and appeared to drift after BECO.

As a result of the noisy tracking data the velocity errors at SECO were difficult to estimate, especially since the IMU contribution was lower than the noise level. The present best estimate of these errors is given in table 5.1.5-III. These quantities were obtained from position and velocity comparisons by using the present best estimate of the trajectory as a reference. In this table the IMU error is made up of gyro and accelerometer errors. The navigation error results from various approximations within the airborne computer. The vertical velocity navigation error was larger than on previous flights because of a scale-factor timing error of some significance. An estimate of orbital injection parameters at SECO + 20 seconds from the IGS and other sources is given in table 5.1.5-IV.

The telemetry drop-out, tracking data noise, and small IMU errors mentioned previously made retrieval of IMU error coefficients difficult and questionable. Figure 5.1.5-2 contains velocity comparisons between scaled and biased IMU accelerometer count data and external tracking sources. The indicated errors can be largely accounted for by using the error coefficients obtained in preflight testing. The error coefficients shown in figure 5.1.5-3 are relatively stable, especially those which induce large velocity errors. A preliminary engineering estimate of gyro and accelerometer error sources which caused the velocity errors at SECO are given in table 5.1.5-V, along with those resulting from a preliminary error coefficient recovery program (ECRP) run. The large values of accelerometer bias and scale factor listed in the ECRP column are compensating and probably result from an inexact mathematical error model. Preflight and inflight calibrations of these parameters show values within specifications. Also included in table 5.1.5-V are those errors which the preliminary analysis indicates are contributed by the ground trackers.

5.1.5.2.2 Orbital phase: Approximately 40 hours of operation were accrued on the IMU on this flight with no evidence of anything but nominal operation. Twice during the flight, accelerometer or attitude malfunction indications were received by way of telemetry. Because either of these indications turns on a warning light in the cockpit which must be reset, and since the crew neither saw nor reset either of the lights at any time, the occurrences have been attributed to erroneous bi-level telemetry discretes.

Inflight tests to determine the three accelerometer biases were conducted over different tracking stations. The tests consisted of counting the accelerometer accumulated pulses (counts over a period of time) and are shown in figure 5.1.5-4, along with the envelope within which the bias is considered acceptable. The X_p and Y_p accelerometer bias values were very stable during preflight testing and during flight. The Z_p bias varied about the compensated value approximately $\pm \frac{1}{2}$ the parameter shift specification. This erratic behavior of the Z_p accelerometer bias was observed during preflight testing, with variations of approximately the same magnitude as those noted during the flight.

A summary of platform alinements is presented in table 5.1.5-VI, where the significant performances of the platform and horizon sensors as controlled by the crew are shown. The results presented in this table, combined with the absence of any torquing currents at the conclusion of the operations, indicate the accuracy of the alinements. Sensor and gimbal-angle data may be directly compared in pitch and roll; however, the yaw alinement accuracy may only be determined by observing the effect of orbital travel. At 90° of orbital travel from the alinement termination, any yaw misalinement will propagate into a roll error. This method was used following the termination of the 00:17:25 g.e.t. and 00:55:20 g.e.t. alinements to determine yaw misalinements of -3° and 1°, respectively.

All arrangements of the gimbals were exercised during this mission, that is, for the small-end-forward (SEF) configuration, both 0°, 0°, 0° and 180°, 180°, 180° gimbal orientations were checked. From the data available, no evidence has been found to substantiate reported crew statements of poor alinement in the platform mode. Several alinements were made in this mode and all available results indicate that alinement accuracy was comparable to SEF or blunt-end-forward (BEF) alinements. During the final orbit, prior to retrofire, the platform was continuously alined in BEF by using the reentry control system. The accuracy during this alinement, as determined both from telemetry data and crew observation, indicates that no problems were associated with the use of this mode.

A summary of translation thrusting activity is included as table 5.1.5-VII. The applied velocity changes were calculated from accelerometer data in all cases except the coelliptic maneuver during which the telemetry data were unreadable. Agreement between the applied velocity changes (accelerometer readings), the IVI readings, and the planned quantities is shown to be close except during separation from the launch vehicle when no attempt to be precise was made. Again, as on Gemini IV, a larger than nominal acceleration was experienced at separation, caused by the "pop gun" effect as the spacecraft moved away from the launch vehicle. Two of the translations, the first apogee adjust and the phase maneuver, were performed in the platform control mode. Attitude was held within 1.5° which is comparable to the control maintained in rate command for the other maneuvers. An attitude error of this magnitude, if held constant over a 20 ft/sec thrust, would result in a cross-axis ΔV of the order of 0.5 ft/sec. Additional cross-axis ΔV would accrue from attitude thruster activity counteracting disturbance torques caused by the offset center of gravity. The platform was accurately alined prior to the translations; therefore, the cross-axis ΔV's reported by the crew must have been caused by a combination of these effects.

The flight crew report of counting of the IVI's has been verified as normal operation in the circumstances involved. At some time previous to the occurrence, the computer mode had been allowed to dwell in the "Ascent" position longer than 1.25 seconds which caused the computer to calculate an IVAR correction. Subsequently, when the "start comp" button was pushed after switching to the "catch-up" mode, the IVI's displayed this computed quantity which was meaningless.

The onboard computer operated properly throughout the flight and responded correctly to the inputs received. No evidence of difficulty in turn on or turn off was noted.

On the fourth pass over Texas at 6 hours 15 minutes g.e.t. an attempt was made to update the computer through the digital command system (DCS) with the revolution 6, area 4 reentry load. Subsequent verification indicated that four of the nine memory locations involved (every other word) had failed to update and still contained the previous load. Preliminary investigation reveals that the update message was most probably sent continuously, which is too fast to be properly received and transferred to the computer, therefore allowing only every other word to be entered. A loss of synchronization in telemetry caused an improper message acceptance pulse (MAP) to be retained in the ground station computer, thus allowing each word after the first to be transmitted before the preceding word had been transferred from the DCS to the on-board computer. Because of the danger of attempting a reentry with an erroneous update, a change in procedure or logic is indicated which would preclude such an occurrence or reduce the probability to an acceptable level.

The radar demonstrated normal operation when turned to standby prior to REP ejection. REP ejection was normal with a predicted slow tumble rate observed. When the radar was turned on, the REP was immediately acquired with the range, range rate, and angle-measuring functions of the radar performing properly. Range and range-rate information was properly displayed on the indicator, and azimuth and yaw angles to the REP were displayed on the flight director indicators (FDI). The radar determined digital range and angle information and correctly transferred this information to the spacecraft computer. Telemetry provisions were not included in the REP; however, based on the review of the radar data, all REP functions were performed. There was no evidence that the radar system functions were disrupted or degraded due to any ground-based non-intentional interference sources. The radar continued to function normally until the system was turned off at 2 hours 43 minutes g.e.t. when the REP exercise was terminated.

Figure 5.1.5-5 shows the telemetered rendezvous radar range, azimuth and elevation outputs starting at 2 hours 16 minutes g.e.t. (Prior to this time the computer was in prelaunch mode in which the digital radar data are not telemetered.) Also shown is an estimate of range based on relative trajectory calculations. This estimate includes the effect of ΔV's accrued by the spacecraft during the 180° yaw maneuver performed after REP eject and during the first 9 minutes of tracking. Agreement between the estimated and calculated ranges is seen to be excellent for the first 14 minutes. After the computer was switched to catch-up mode the effect of ΔV's accruing from attitude control could no longer be accounted for because the "Start Comp" button was not pushed and the telemetered accelerometer outputs were held to zero. The divergence of the telemetered and calculated ranges after 2 hours 30 minutes g.e.t. is probably caused by these unaccounted for ΔV's. The azimuth and elevation angles are seen to stay near zero until about 2 hours 25 minutes g.e.t. when rather large excursions, first in azimuth and then in elevation, occur. These excursions are reflected in the gimbal angles for this period indicating that the REP was allowed to drift off boresight. It is probable that the crew became preoccupied with the fuel-cell oxygen supply problem and could no longer give full attention to the tracking exercise.

The rendezvous radar was also exercised under long range conditions by tracking a transponder located at the KSC Radar Boresight Facility. Table 5.1.5-VIII lists the lock-on and loss of target times and ranges for the four tests performed. During the first pass the platform was powered down and, therefore, accurate angle data are unavailable. Figure 5.1.5-6 shows radar range for the first pass superimposed on range computed from trajectory data and indicates performance accuracy within design limits. On the three subsequent passes, the radar digital range output was inoperative (see section 5.1.5.4.2 for discussion); however, angle data were obtained. Figure 5.1.5-7 compares azimuth and elevation angles for the second pass based on radar and gimbal angles with those generated from trajectory data. The differences shown can be attributed to normal servo-dynamic lags which occur when spacecraft angular accelerations are present. Analysis has shown that the lags expected on a rendezvous mission will cause no difficulty.

The relatively low acquisition range on the second pass is unexplained but could have been caused either by an improper pointing angle or by ground-based interference. The spacecraft at that time was near the point of closest approach. Prior to this time intermittent lock-on occurred similar to that which would be produced from interference. After solid lock-on was achieved, the angle function operated normally. Ground-based interference is considered an operational problem peculiar to this type of test and not a hazard for a normal rendezvous mission. During the third pass, the local test environment was closely observed, and measures were taken to reduce interference sources. As seen, lock-on was achieved at 350 nautical miles.

Because of the transponder set-up, a 3 db signal-to-noise advantage was realized. Lock-on range adjusted for this advantage would have been about 250 nautical miles.

On the third pass, the radar was turned on and off five times in an attempt to determine whether the radar was locked on to a false null or to an interfering signal. It was subsequently verified that the transponder was being tracked on the true null.

The fourth and final pass was performed in drifting flight. Results were normal for this type flight with lock-on occurring within the expected angular limits. As shown, the relatively low loss-of-track range was due to the radar being turned off early.

Throughout the periods of radar operation, circuit, power, and temperature indications were nominal. System pressure, however, indicated an excessive leak rate as discussed in section 5.1.5.4.2.

One other test to determine the effect of outside ground interference was conducted by tracking the spacecraft with a space acquisition detection and tracking system (SPADATS) tracking radar while the rendezvous radar was operating. Although all telemetry data are not available as yet for verification of this test, the flight crew reported no cockpit indications of interference.

5.1.5.2.3 Retrofire – reentry phase: The IGS operated properly throughout the retrofire-reentry phase. Retrofire velocity was close to nominal as indicated in table 5.1.5-IX and caused a footprint shift of approximately 5 nautical miles.

From retrofire to an altitude of 400 000 feet, a 0° bank angle (maximum lift) trajectory was flown as planned. At the proper time (190:42:04 g.e.t.) the computer commanded a 60° right bank angle, and at 190:44:20 g.e.t. began to generate a predicted half-lift range. The density altitude parameter at this time was 8.75786 (nondimensional) which is the value associated with an acceleration of 1.0 ft/sec^2 and indicates a proper entry into the guidance logic.

At this time the computer commanded a 90° bank angle which was followed for a time by the flight crew. This angle was generated properly by the computer in response to an erroneous update (see section 6.2.2.2.1) prior to retrofire. As a result, at retrofire the longitude used by the computer was 7.89° east of the actual spacecraft longitude. This longitude error caused the computer to continuously predict an overshoot in excess of 250 nautical miles. The command for a 90° bank angle was displayed until 190:48:21 g.e.t. when an overflow occurred in the computer and the commanded angle went to zero. The overflow resulted when the normalized downrange error used in the bank-angle command equation exceeded the overflow value of ±128.0 and was caused by the erroneous update. This overflow would not occur under normal operating conditions.

The 90° bank angle was flown until approximately 190:47:20 g.e.t. when the flight crew, seeing no movement of the downrange needle, correctly assumed that the computer was giving invalid indications, and began to fly ground-computed backup bank angles. By this time, however, the maneuvering capability remaining was insufficient to overcome the downrange error already accrued. At 190:48:58 g.e.t. and a density altitude of 4.6132 the computer correctly terminated guidance.

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

The IGS computed position (compensated for the update error) at drogue deployment was 4.8 nautical miles from the actual touchdown coordinates obtained from recovery. The IGS navigated altitude at drogue parachute deployment was approximately 3.78 nautical miles lower than that obtained from ground tracking. These navigation errors are within the variation expected because of initial condition uncertainty, IMU misalinement, and IMU component errors. The spacecraft landed 89.25 nautical miles short and 19.67 nautical miles to the right of the ground track.

The flight crew reported that the downrange error needle on the FDI indicated full scale on the low range but something less than full scale on high range. Because the telemetered computer output of the quantity used to drive this display was such that full-scale deflection should have occurred on both ranges, a possible discrepancy exists. It has been determined that "full scale" on the FDI is represented by a deflection of 0.875 inch, whereas the mechanical stop occurs at something more than 1 inch. It is probable that the low range indication was actually more than "full scale" and caused an apparent difference in readings between the ranges. Tests are being conducted on the flight hardware to determine deflection when driven by the actual voltages indicated by telemetry.

5.1.5.3 Control system performance evaluation.-

5.1.5.3.1 Attitude control and maneuvering system: The attitude control and maneuvering system was activated 357 seconds after lift-off with the firing of the two aft-firing translation thrusters. Actual spacecraft separation commenced 0.4 second later with attitudes of -21°, -2.5°, and 90° in pitch (referenced to the launch pad horizontal), yaw, and roll, respectively. The spacecraft body rates and attitude control commands during this period are shown in figure 5.1.5-8. The crew switched to "rate command", and then thrusted forward again for 3.5 seconds. The rate command mode immediately damped the 1.1 deg/sec pitch rate, the only significant spacecraft rate existing at that time. At 364 seconds after lift-off, 0.5 second after the second translation thrust was initiated, the roll to heads-up attitude maneuver was initiated. No contact with the launch vehicle was experienced and a clean separation was achieved. A control mode check lasting 14 seconds was performed at 00:26:05 g.e.t. which verified the operation of the "direct" attitude control mode.

Since the retrofire maneuver and reentry were to be performed using the ring A and B thrusters, an RCS control mode check lasting 15 seconds was performed beginning at 188:28:55 g.e.t. Rate command mode was checked by using first the ring A thrusters, then the ring B thrusters. Proper control system performance was verified in each case. No control torques could be calculated for either check because of the short firing times of the thrusters, telemetry dropouts, and/or noisy data; however, in each case, the correct thrusters fired and spacecraft response was proper for every command.

As mentioned previously, the operation of the control system in "platform" mode appeared normal. At least two good platform alinements were performed in this mode as well as two translation maneuvers. No evidence of the crew report of "sloppiness" could be seen. To check further on this discrepancy, however, a postflight test was conducted on the attitude control electronics (ACE) and the dead bands were within the specification limits.

The first definite indication of a thruster failure occurred during pulse mode attitude control at 75:16:31 g.e.t. The command pilot commanded a roll right (TCA 3 and 7) but the rate gyro signals indicated the spacecraft response to the command was a roll right and a yaw left. Following the roll-right command, yaw-left, roll-right, and yaw-left commands were sequentially generated and rate gyro indications for each command indicate similar thrust forces from TCA 7. Because the TCA firing indications on telemetry are actually measurements of voltage to the thruster solenoid valve drivers, it is apparent that the control system was operating correctly.

During revolution 75, beginning at 117:41:20 g.e.t., a series of at least 12 commands were generated while the attitude control and maneuver electronics (ACME) were operating in the pulse mode. Results summarized in the following tabulation indicate that thrusters 6 and 7 were not firing, or were firing at a very low thrust level.

Command	Spacecraft response	Proper TCA operation	Actual TCA thrust
Pitch up	Pitch up and roll right	5 and 6	5
Pitch down	Pitch down	1 and 2	1 and 2
Roll right	Roll right and yaw right	3 and 7	3
Roll left	Roll left	4 and 8	4 and 8
Yaw right	Yaw right	3 and 4	3 and 4
Yaw left	Yaw left and roll left	7 and 8	8

A similar investigation was made beginning at 142:59:00 g.e.t. Attitude control mode was "pulse", and yaw logic was used to generate roll commands. Results summarized in the following table indicate that TCA 4 was not firing and TCA 2 and TCA 6 were firing with less than full thrust, but TCA 6 greater than TCA 2.

Command	Spacecraft response	Proper TCA operation	Actual TCA thrust
Pitch up	Pitch up and small roll right	5 and 6	5 > 6
Pitch down	Pitch down and small roll right	1 and 2	1 > 2
Roll right	(No roll right commands were generated)	–	–
Roll left	Roll left and small pitch up	2 and 6	6 > 2
Yaw right	Yaw right and roll right	3 and 4	3 only
Yaw left	(No yaw left commands were generated)	–	–

The retromaneuver was performed in the rate command mode using both RCS rings. Pitch attitude was held within ±2°, yaw between 0° and -5°, and roll between 0° and -8.6°. At 190:29:55 g.e.t. (2 minutes 12 seconds after retrofire) control was switched to pulse mode, RCS ring B was turned off, and the initial reentry attitude was assumed. This state was held until 190:41:53 g.e.t. (approximately 400 000 feet altitude) when the ACME mode switch was placed in rate command and the ring A ACME-Direct switch was placed in Direct. Manual damping of pitch and yaw oscillations was performed with maximum rates of approximately 1 deg/sec until about 190:45:06 g.e.t. (approximately 0.3g and 260 000 feet altitude) when the oscillations began to increase and the ring A ACME-Direct switch was returned to ACME. Rate command mode was utilized throughout the remainder of the reentry. Maximum rates experienced prior to drogue parachute deployment were approximately 2.5 deg/sec in both pitch and yaw, a much lower value than those experienced in previous reentries. Control authority was more than adequate throughout, even though only one RCS ring was energized and the relatively tight rate command dead bands were in force. After drogue parachute deployment, RCS control was switched to both rings A and B where it remained until power down. Figure 5.1.5-9 summarizes significant control parameters during the period prior to drogue parachute deployment.

5.1.5.3.2 Horizon sensor: The horizon sensor control mode was used extensively throughout the flight, and generally exhibited excellent performance. Some cases of sun interference were identified, as were cases of loss-of-track due to high frontal clouds, and some around a typhoon in the Pacific Ocean. As mentioned previously, sensor outputs were used to aline the platform several times with good results in each case. The primary sensor apparently failed sometime after the second day (see section 5.1.5.4.1 for discussion) when spurious pitch-down pulses were reported by the crew. Operation was continued using the secondary sensor for the remainder of the flight. No further difficulty was experienced.

5.1.5.4 Anomalies.-

5.1.5.4.1 Horizon sensor malfunction: During the third day, the crew reported that the primary horizon sensor appeared to cause a 15° pitch-down platform alinement. Secondary sensor operation at this time was normal. A special test was conducted on revolution 62 to attempt to establish sensor operational status. After alining carefully to 0° pitch, 0° yaw, and 0° roll, the primary sensor was turned on and the control mode switched to HORSCAN. The spacecraft pitched down to -35° when the loss-of-track light came on. The crew then took over and started a slow pitch-up rate toward the horizon. The HORSCAN mode was tried again, and the same action was repeated. Subsequent analysis of telemetry data during this test shows that the sensor outputs remained near null throughout the test and did not follow the gimbal rotations as they would in normal operation. Large variations from null occurred only during periods of loss or reacquisition of track. The postflight data examined to date indicate that the instrument may have failed as early as revolution 30 when the outputs resembled those during the test just described.

The two sensor electronic packages (the sensor heads were jettisoned prior to reentry) were returned to the vendor for failure analyses. After cleaning, drying, and a thorough resistance check, the primary unit was subjected to a modified pre-delivery acceptance test. A qualification unit sensor head was mated to the electronics package. To date the tests have not revealed any fault in the primary electronics.

The fact that both pitch and roll axes were affected tends to absolve the signal-processing loops. The apparently normal operation of the track loops reduces the malfunction area to those components affecting sensor head azimuth motion. A geometrical study is underway which will help determine whether sensor head azimuth motion was improper and caused the system to act as a point tracker.

5.1.5.4.2 Radar range malfunction: During the second and subsequent passes over the transponder located at KSC, the radar failed to read out digital range above a count of 24 800 feet. Analog range indications were normal although somewhat inconclusive because the actual range was greater than the maximum analog range indication of 300 000 feet.

The rendezvous and reentry section was not recovered; therefore, the failure analysis must be purely analytical. Possible failure causes examined and rejected include outside interference, spacecraft induced RFI, and spurious oscillation from the crystal oscillator in the range counter circuit. The onboard computer interface was also considered and ruled out because range, azimuth, and elevation are processed serially over the same circuits and all but the range data was normal. The most likely cause has been determined to be a failure in the tenth stage of the shift register which functions as a counter to measure range and as a shift register to transfer angle data. The maximum range count obtainable from nine stages of this register is 24 800 feet, the value seen in each of the irregular passes. Tenth-stage failures which could have caused the malfunction include a grounded clock-input transistor, an open in one of five soldered or welded connections between the ninth and tenth stages, or an open in one of two diodes, resistors, or capacitors in the tenth-stage multi-vibrator circuitry. One of these possible failures, the grounded clock-input transistor, was incorporated into a digital pack by the radar vendor and the resulting performance matched that during the mission. Component reliability and hardware test procedures are being examined in an attempt to preclude recurrence of this failure.

A second, unrelated anomaly has been detected in the radar data and is being analyzed. The maximum allowable pressure leak rate of 0.2 psi/day was exceeded after 72 hours g.e.t. when an increase to 1.0 psi/day was noted. The cause is unknown at this time.

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

Ground elapsed time, sec			Event	Component Status					Remarks
Planned	Actual RGS	Actual IGS		ACME	Computer	IMU	Horizon sensor	Radar	
10.16	10.13	10.34	Start roll program	IGS backup	Ascent	Free	Search (primary)	Off	
20.48	20.45	20.28	Roll program complete	IGS backup	Ascent	Free	Search (primary)	Off	
23.04	23.09	22.80	Start no. 1 pitch rate	IGS backup	Ascent	Free	Search (primary)	Off	
88.32	88.35	88.03	End no. 1 pitch, Start no. 2 pitch	IGS backup	Ascent	Free	Search (primary)	Off	
104.96	104.97	104.72	No. 1 gain change	IGS backup	Ascent	Free	Search (primary)	Off	
105.00	105.00	104.91 to 107.44	No. 1 IGS update	IGS backup	Ascent	Free	Search (primary)	Off	
119.04	119.06	119.28	End no. 2 pitch Start no. 3 pitch	IGS backup	Ascent	Free	Search (primary)	Off	
145.00	145.00	143.70 to 146.21	No. 2 IGS update	IGS backup	Ascent	Free	Search (primary)	Off	
154.83	154.55	--	No. 2 gain change	IGS backup	Ascent	Free	Search (primary)	Off	
162.56	162.61	162.48	End no. 3 pitch program	IGS backup	Ascent	Free	Search (primary)	Off	
169.00	168.40	168.00	Guidance initiate						
199.83	207.00	207.00	Horizon sensor and radar covers jettisoned	IGS backup	Ascent	Free	Search (primary)	Off	
336.93	333.32	333.23	SECO	IGS backup	Ascent	Free	Search (primary)	Off	

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

Ground elapsed time, sec		Event	Component status					Remarks
Planned	Actual		ACME	Computer	IMU	Horizon sensor	Radar	
356.93	356.11	Start separation thrust	Direct	Ascent	Free	Search (primary)	Off	Prior to separation: Rate, deg/sec Pitch +0.1 Roll +0.4 Yaw +0.9 Gimbal angle, deg Pitch 340.1 Roll 87.8 TCA's 9 and 10 fire: Time, min:sec On / Off 05:56.11 06:01.26 06:03.34 06:07.01 ΔV = 7.6 ft/sec IVI = 6.5 ft/sec Planned = 5.0 ft/sec
356.93	356.91	GLV-spacecraft separation (shaped charge fire)	Direct	Ascent	Free	Search (primary)	Off	
	360.04	Roll to heads-up position	Rate command	Ascent	Free	Search (primary)	Off	
363.43	367.01	End separation thrust						

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

Ground elapsed time, hr:min:sec		Event	Component status					Remarks
Planned	Actual		ACME	Computer	IMU	Horizon sensor	Radar	
00:06:14	00:07:33	Platform alinement	Platform	Ascent	SEF	Search (primary)	Off	Gimbal angle, deg Pitch = 0.8° Roll = 0.4° Yaw =-1.0° Horizon sensor minus gimbal angle, deg Pitch = +0.2° Roll = +0.5° Completed at 55:20 g.e.t.
00:15:00	00:26:05	Control mode check	Direct	Prelaunch	Orbital	Search (primary)	Off	Completed at 26:19
00:56:00	00:56:00	Perigee adjust translation	Platform	Catchup	Orbital	Search (primary)	Off	ΔT = 12.80 sec ΔV = 9.7 ft/sec Planned = 10.0 ft/sec
	02:03:37	Platform alinement	Pulse	Catchup	SEF	Search (primary)	Off	Three attempts — scanner loss of tracks Gimbal angle, deg Pitch = 0.3° Roll = 0.5° Yaw = 0.1° Horizon sensor minus gimbal angle, deg Pitch = -0.2° Roll = -0.3°
02:07:00	02:07:15	REP eject	Pulse, then direct	Catchup	Orbital	Search (primary)	On	Yaw gimbal angle = 88.1°
	27:04:01	First radar pass over Cape Kennedy	Pulse	Catchup	Off	Search (primary)	On	

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

Ground elapsed time, hr:min:sec		Event	Component status					Remarks
Planned	Actual		ACME	Computer	IMU	Horizon sensor	Radar	
	47:31:11	Platform alinement	HORSCAN	Off	SEF	Search (primary)	Off	<u>Gimbal angle, deg</u> Pitch = -1.8° Roll = +1.8° Yaw = +4.0° <u>Horizon sensor minus gimbal angle, deg</u> Pitch = -1.1° Roll = +2.4°
	50:41:15	Platform alinement	Pulse, Rate command, Platform	Catchup	BEF	Search (primary)	Off	<u>Gimbal angle, deg</u> Pitch = +0.6° Roll = 0.0° Yaw = -0.1° <u>Horizon sensor minus gimbal angle, deg</u> Pitch = +0.1° Roll = -0.3°
50:50:00	50:49:58	Apogee adjust maneuver	Rate command	Catchup	Orbital	Search (primary)	Off	ΔT = 26.88 sec ΔV = -20.9 ft/sec IVI = -21.5 ft/sec Planned = -21.1 ft/sec
51:34:42	51:34:31	Phase adjust maneuver	Platform	Catchup	Orbital	Search (secondary)	Off	ΔT = 20.0 sec ΔV = 15.7 ft/sec IVI = 15.7 ft/sec Planned = 15.2 ft/sec
52:06:16	52:06:26	Plane change	Rate command	Catchup	Orbital	Search (secondary)	Off	ΔT = 20.0 sec ΔV = 15.0 ft/sec IVI = 15.0 ft/sec Planned = 14.6 ft/sec
53:04:02	53:04:04	Coelliptic maneuver	Rate command	Catchup	Orbital	*	Off	ΔT = 22.5 sec ΔV = * IVI = 17.2 ft/sec Planned = 17.4 ft/sec

*Indicates data [illegible]t available

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

Ground elapsed time, hr:min:sec		Event	Component status					Remarks
Planned	Actual		ACME	Computer	IMU	Horizon sensor	Radar	
	74:46:54	Second radar pass over Cape Kennedy	*	Catchup	*	*	Stand-by	
	75:16:31	First indication of TCA malfunction	Pulse	Pre-launch	*	*	Off	TCA 7 fires at less than full thrust
	96:32:41	Platform alinement	Pulse	*	SEF	*	Off	Gimbal angle, deg Pitch = 1.9° Roll = -6.0° Yaw = +0.2° Horizon sensor minus gimbal angle, deg Pitch = +1.7° Roll = +5.2°
	97:07:00	Trouble shooting scanner	Pulse	*	*	Search (primary)	Off	
	98:16:54	Platform alinement	Pulse	*	SEF	Search (secondary)	Off	Gimbal angle, deg Pitch = 1.9° Roll = 6.0° Yaw = +0.2° Horizon sensor minus gimbal angle, deg Pitch = -2.8° Roll = -6.2°
	117:43:10	Third radar pass over Cape Kennedy	Pulse	Catchup	*	*	Stand-by	
	117:41:20	TCA failure	Pulse	Catchup	Orbital	*	Off	TCA 6 and 7 do not fire
	142:59:36	TCA failure	Pulse	*	*	*	Off	TCA 2 - less than full thrust TCA 4 - not firing
	146:15:00 to 146:15:30	First SPADATS test over Cape Kennedy	*	*	*	*	Stand-by	

*Indicates data not available

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

Ground elapsed time, hr:min:sec		Event	Component status					Remarks
Planned	Actual		ACME	Computer	IMU	Horizon sensor	Radar	
	160:45:50	Second SPADATS test over Cape Kennedy	*	*	*	*	Stand-by	
	168:29:56	Fourth radar pass over Cape Kennedy	*	*	*	*	On	
	188:28:55	RCS control mode check	Rate command	*	Orbital	*	Off	
	188:33:46	Platform alinement	Pulse	Prelaunch	BEF	Search (secondary)	Off	Gimbal angle, deg Pitch = +2.1° Roll = -0.40° Yaw = -2.9° Horizon sensor minus gimbal angle, deg Pitch = 5.9° Roll = 4.0° Yaw = -
190:27:43	190:27:43	Retro fire	Rate command	Reentry	Free	Off	Off	ΔV = 324.5 ft/sec IVI = 324.4 ft/sec
190:42:04	190:42:06	400K	Rate command	Reentry	Free	Off	Off	
190:44:27	190:44:20	Guidance initiate	Rate command	Reentry	Free	Off	Off	
190:49:11	190:48:59	Terminate guidance	Rate command	Reentry	Free	Off	Off	
190:50:09	190:49:19	Drogue deploy	Rate command	Reentry	Free	Off	Off	

*Indicates data n[illegible] vailable

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TABLE 5.1.5-II.- IVAR COMPARISONS

	Calculated	Telemetry
Velocity to be applied at apogee, V_{gp}, ft/sec . . .	0.030	0.031
Velocity to be applied at perigee, V_{ga}, ft/sec . .	0.959	0.969
Radial velocity, V_p, ft/sec	126.363	126.359
Inertial velocity, V, ft/sec	25 794.120	25 794.703
IVI fore-aft, $\Delta V_{X_{S/C}}$, ft/sec	0.330	0.349
IVI right-left, $\Delta V_{Y_{S/C}}$, ft/sec	27.068	26.051
IVI up-down, $\Delta V_{Z_{S/C}}$, ft/sec	0.126	0.141
Time to apogee, T_{AP}, sec	3 041.266	3 041.184

TABLE 5.1.5-III.- GUIDANCE ERROR AT SECO

	Position, ft			Velocity, ft/sec		
	X	Y	Z	X	Y	Z
IMU error	550±50	330±50	-127±5.0	0.5±0.5	5.0±3.0	-4.0±1.0
Navigation error	123	232	-32	1.6	2.2	0.15
Total guidance error	673±50	562±50	-159±50	2.1±0.5	7.2±3.0	-3.85±1.0

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

System	Inertial velocity, ft/sec	Inertial flight-path angle, deg	Inertial velocity components (computer coordinates), ft/sec		
			X	Y	Z
Nominal	25 812	0.016	25 402	4580	15
IGS	25 808	-0.01	25 413	4498	-96
STL preliminary BET	25 805	+0.0008	25 411	4491	-92
MISTRAM 10K	25 809	-0.02	25 412	4505	-91
GE MOD III	25 808	-0.01	25 413	4497	-96
Goddard GE MOD III	25 821	-.19	-	-	-
MISTRAM IP	25 820	-.18	-	-	-
Reconstructed from Bermuda first orbital pass	25 805	-.18	-	-	-

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

	IMU component errors															
	Accelerometer bias, $g \times 10^{-6}$			Accelerometer scale factor, ppm			Gyro mass unbalance along input axis, deg/hr/g			Gyro mass unbalance along spin axis, deg/hr/g			Constant gyro drift, deg/hr			Azimuth misalinement, sec
Platform axis	X_p	Y_p	Z_p	X_p	Y_p	Z_p	X_p	Y_p	Z_p	X_p	Y_p	Z_p	X_p	Y_p	Z_p	
Engineering estimate	19.5	-68.4	-98.8	10	0	10	-0.041	-0.017	0.085	0.12	-0.54[a]	0.157	0.119	-0.091	0.043	(b)
Error Coefficient Recovery Program (ECRP)	-316	58	-961	96	34	-417	-0.24	+0.028	-0.24	(b)	(b)	(b)	(b)	(b)	(b)	15
Uncertainty in ECRP estimates	±25	±41	±31	±11	±160	±23	±0.1	±0.035	±0.027	-	-	-	-	-	-	±2
Specification values		360			300			0.5			0.5			0.3		60

System	Tracking radar errors					
	Range bias, ft	P bias, ft	Q bias, ft	Azimuth, radians	Elevation, radians	Refraction, n units
GE (final data)	78.3	N/A	N/A	2.2×10^{-5}	4.0×10^{-5}	-3.1
MISTRAM 10K	37 before SECO 18 after SECO	0.326	0.438	N/A	N/A	-21.9

[a]Contributes less than 0.4 ft/sec at SECO

[b]No significant errors attributed in the quantity using process indicated.

TABLE 5.1.5-VI.- PLATFORM ALINEMENT ACCURACY

Start time, g.e.t., hr:min:sec	Length of cage, min:sec	Length of alinement, min:sec	Mode		Pitch error (sensor minus gimbal angle)		Roll error (sensor minus gimbal angle)		Yaw error[b]	Remarks
			ACME	Platform	Start	Finish	Start	Finish		
00:07:33	01:13	08:34	Platform	SEF	2.14	0.06	-0.05	0.36	-3.0	
00:49:51	0.0	05:29	Platform	SEF	-1.09	0.25	0.14	0.49	1.0	
02:01:09	00:33	01:31	Platform	SEF	-5.00	0.06	-3.70	-0.17		Switched to secondary horizon sensor at 2:03:18.7 g.e.t. Switched back to primary horizon sensor at 2:05:25.2 g.e.t.
02:03:37	00:30	01:10	Pulse	SEF		-0.22		0.25		Sensor data noisy at start of alinement
[a]47:31:11	06:44	01:14	Horizon scan	SEF	-4.18	-1.13	5.05	2.43		
[a]50:41:15		7:51	Pulse, rate command, and platform	BEF	-0.22	.07	-24.27	-0.26		Caging data missing
[a]51:00:20	0.0	≈33:00	Platform	SEF	7.10	-2.2	-0.2	1.0		Uncertain when alinement stopped
[a]51:36:54	0.0	1:04	Platform	SEF	0.65	1.1	0.89	1.79		
[a]51:50:29	0.0	12:04	Pulse	SEF	-0.6	0.7	0.47	0.21		
[a]73:18:53		10:17	Pulse	SEF	-1.13	0.60	21.12	-6.02		Caging data missing
[a]73:34:19	0.0	12:37	Pulse	BEF	1.67	0.23	3.22	3.22		
[a]74:13:48	0.0	16:01	Pulse	SEF	-12.14	0.23	5.28	5.28		
[a]74:56:57	0.0	9:03	Pulse	SEF	-2.28	-1.68	-5.60	-5.60		
[a]76:00:40	0.0	11:15	Pulse	SEF	-3.48	1.90	5.84	4.83		
[a]93:26:41	6:32	48:27	Horizon scan and pulse	SEF	0.47	1.67	0.04	-6.25		Alinement stopped between 93:51:00 and 93:56:00 g.e.t.

[a]Analog data used, less resolution than digital data (2.0° in gimbal angles; 0.2° in sensor)

[b]Yaw error determined 90° of orbit travel later from analog data.

TABLE 5.1.5-VI.- PLATFORM ALINEMENT ACCURACY - Concluded

Start time, g.e.t., hr:min:sec	Length of cage, min:sec	Length of alinement, min:sec	Mode		Pitch error (sensor minus gimbal angle)		Roll error (sensor minus gimbal angle)		Yaw error[b]	Remarks
			ACME	Platform	Start	Finish	Start	Finish		
[a]94:39:46	0.0	2:38	Pulse	SEF	0.69	1.67	7.13	5.75		
[a]95:21:36	0.0	2:17	Pulse	SEF	1.82	1.67	5.36	5.10		
[a]95:52:06	0.0	2:54	Pulse	SEF	1.67	1.67	9.7	6.25		
[a]96:32:41	0.0	19:11	Pulse	SEF	-2.92	1.67	-4.10	5.23		
[a]98:16:54	0.0	5:59	Pulse	SEF	-4.08	-2.77	-4.00	-6.20		
[a]188:33:46	0.0	1:53	Pulse	BEF	0.79	2.10	-14.62	-3.95		
[a]188:36:03	0.0	3:21	Pulse	BEF	-2.17	2.30	-2.33	4.65		
[a]≈190:20:41			RCS	BEF		-1.67		-0.47		Data missing on initiation of alinement

[a]Analog data used, less resolution than digital data (2.0° in gimbal angles; 0.2° in sensor)

[b]Yaw error determined 90° of orbit travel later from analog data.

TABLE 5.1.5-VII.- TRANSLATION MANEUVERS

Burn	Accelerometer data period		ΔV ft/sec		
	From g.e.t., hr:min:sec	To g.e.t., hr:min:sec	Accelerometer integration	IVI readings	Planned
Tailoff	00:05:33	00:05:55	86.3	(a)	85.5
Separation	00:05:55	00:06:09	7.6	6.5	5.0
First apogee	00:55:59	00:56:13	9.7	(a)	10.0
Height maneuver	50:49:58	50:50:31	-20.9	-21.5	-21.1
Phase maneuver	51:34:31	51:34:47	15.7	15.7	15.2
Plane change	52:06:26	52:06:41	15.6	15.0	14.6
Coelliptic	53:04:04	53:04:21	(a)	17.2	17.4

[a] Data not available

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TABLE 5.1.5-VIII.- RENDEZVOUS RADAR AIR-TO-GROUND TEST SUMMARY

Acquisition of target		Loss of target		Tracking time, min:sec
Time, g.e.t., hr:min:sec	Range, n. mi.	Time, g.e.t., hr:min:sec	Range, n. mi.	
27:04:01	291.0	27:06:00	288.9	01:59
74:46:54	115.0	74:48:38	388	01:44
117:43:10	350.0	117:45:59	358	02:49
168:29:56	384	168:31:40[a]	106	01:44

[a]Radar turned off

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TABLE 5.1.5-IX.- RETROFIRE VELOCITY COMPONENTS

ΔV, ft/sec	Predicted	IVI display	Telemetry data
X	277.8 (aft)	269.0 (aft)	269.31 (aft)
Y	0	10 (left)	9.94 (left)
Z	168.5 (down)	181.0 (down)	180.7 (down)
Total	324.5	[a]324.4	324.46

[a]Calculated postflight, not displayed

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TABLE 5.1.5-X.- COMPARISON OF COMPUTER TELEMETRY REENTRY PARAMETERS WITH POSTFLIGHT RECONSTRUCTED COMPUTER DATA

Parameter	Time = 190:42:03.32 g.e.t. Altitude = 400 000 ft			Time = 190:44:25.039 g.e.t Guidance initiate +3 secs			Time = 190:49:08.14 g.e.t. Guidance termination ($D = D_0$) +10 secs		
	Telemetry	MAC	IBM	Telemetry	MAC	IBM	Telemetry	MAC	IBM
Time in mode, sec . . .	2472.11	2472.11	2472.11	2613.84	2613.84		2896.934	2896.934	
Radius vector, ft . . .	21 295 323	21 295 956	21 295 420	21 194 411	21 195 236		20 963 295	20 964 880	
Velocity, ft/sec . . .	24 393 823	24 393 404	24 393 902	24 493 166	24 491 988		1 284.117	1 278.179	
Flight-path angle, deg	-1.66570	-1.66329	-1.665	-1.65781	-1.65452		-44.5382	-44.4482	
Downrange error, n. mi.	NA	NA	NA	-448.6	-448.0941		-270.8	-268.172	
Crossrange error, n. mi.	NA	NA	NA	-93.496	-93.6152		-192.956	-192.865	
Bank angle command, deg	-60.0	-60.0	-60.0	90.0	90.0	90.0	0.0	0.0	0.0
Latitude, deg	32.62503	32.62685	32.627	32.20128	32.20303		29.61758	29.622047	
Longitude, deg	272.45438	272.46205	272.493	283.54266	283.54326		298.02438	298.01548	
Density altitude factor	NA	NA	NA	8.71690	8.709782		4.497	4.49422	
Half-lift range predictor, n. mi. . .	NA	NA	NA	902.091	901.3865		5.884	5.6268	
Range to target, n. mi.	NA	NA	NA	463.356	463.363		314.470	314.0067	
Spacecraft heading, deg	89.26749	89.27114	89.290	95.90925	95.90808		128.85064	128.8438	

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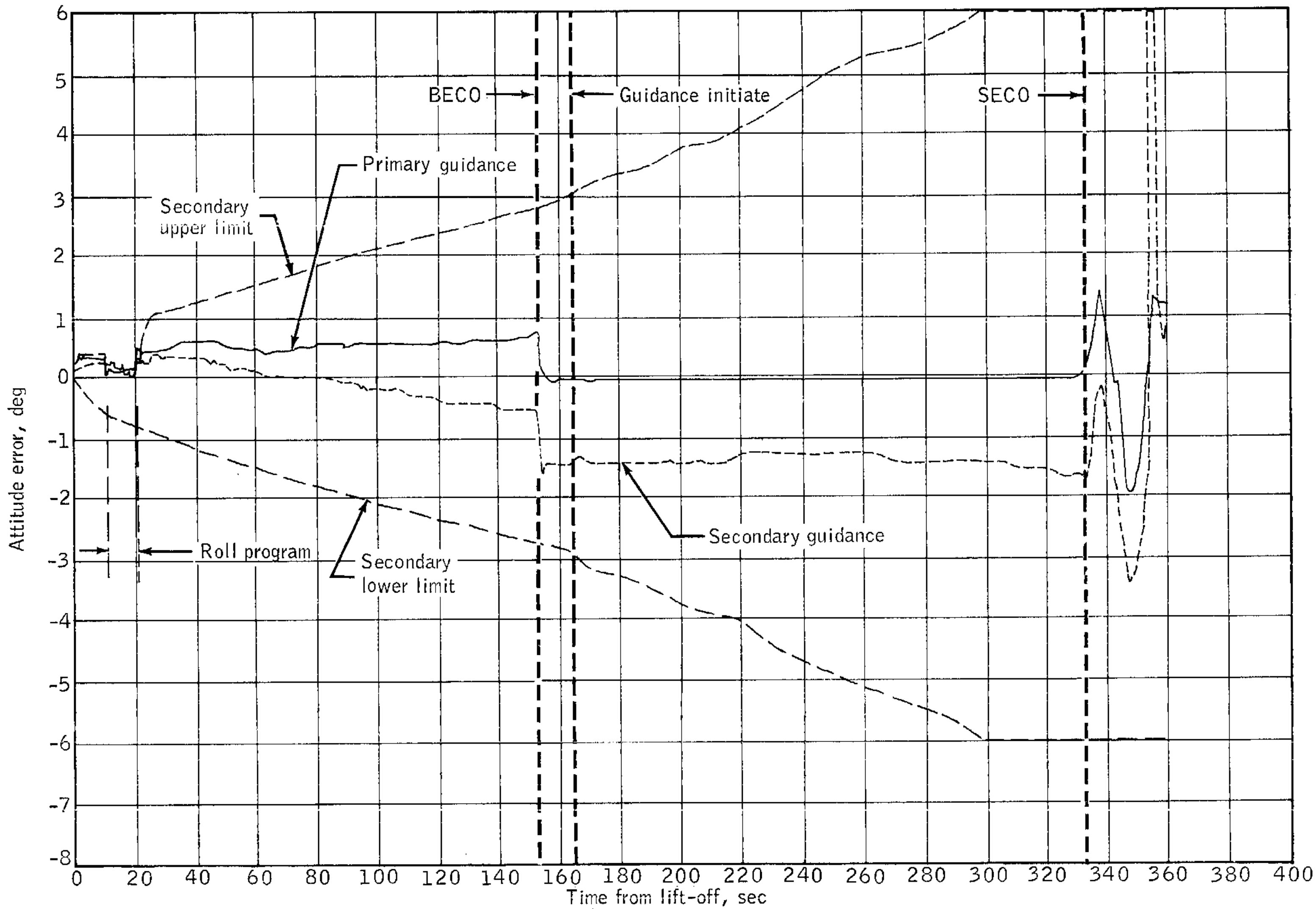
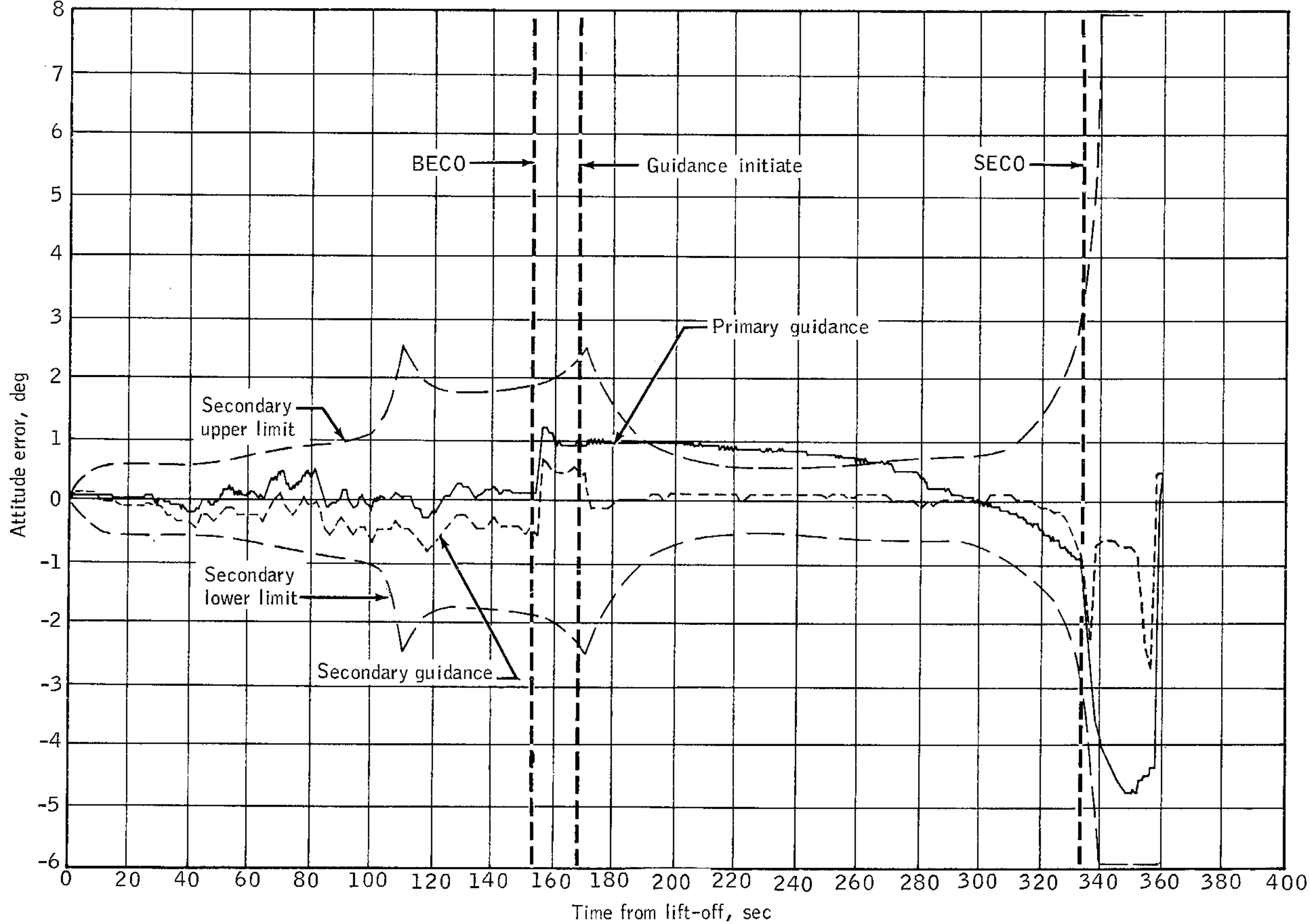
NASA-S-65-8534



(a) Roll attitude error

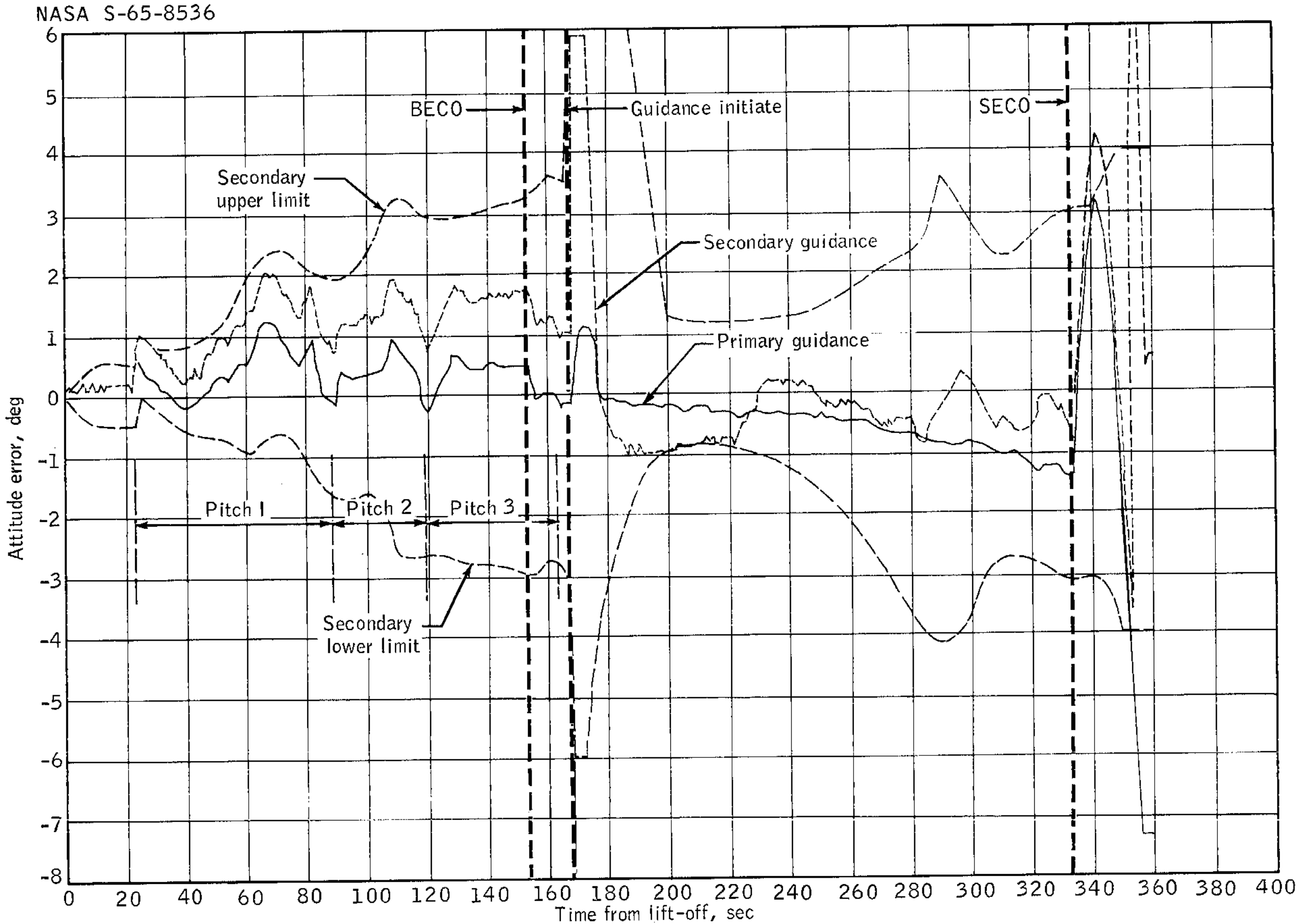
Figure 5.1.5-1. - Comparison of steering commands during launch with preflight value.

NASA S-65-8535



(b) Yaw attitude error

Figure 5.1.5-1. - Continued.



(c) Pitch attitude error

Figure 5.1.5-1. - Concluded.

NASA-S-65-8549

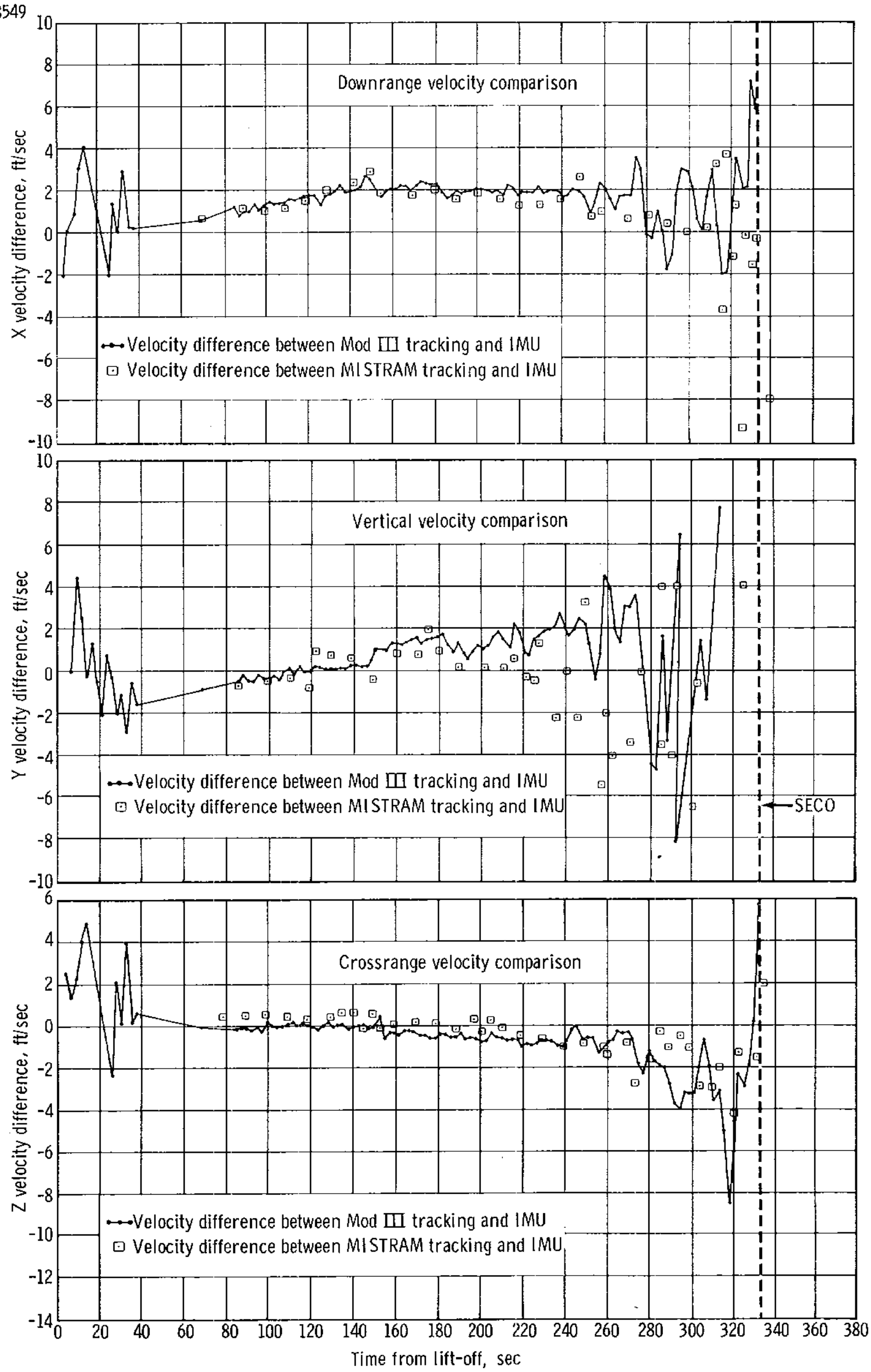


Figure 5.1.5-2. - Velocity comparisons.

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NASA-S-65-8551

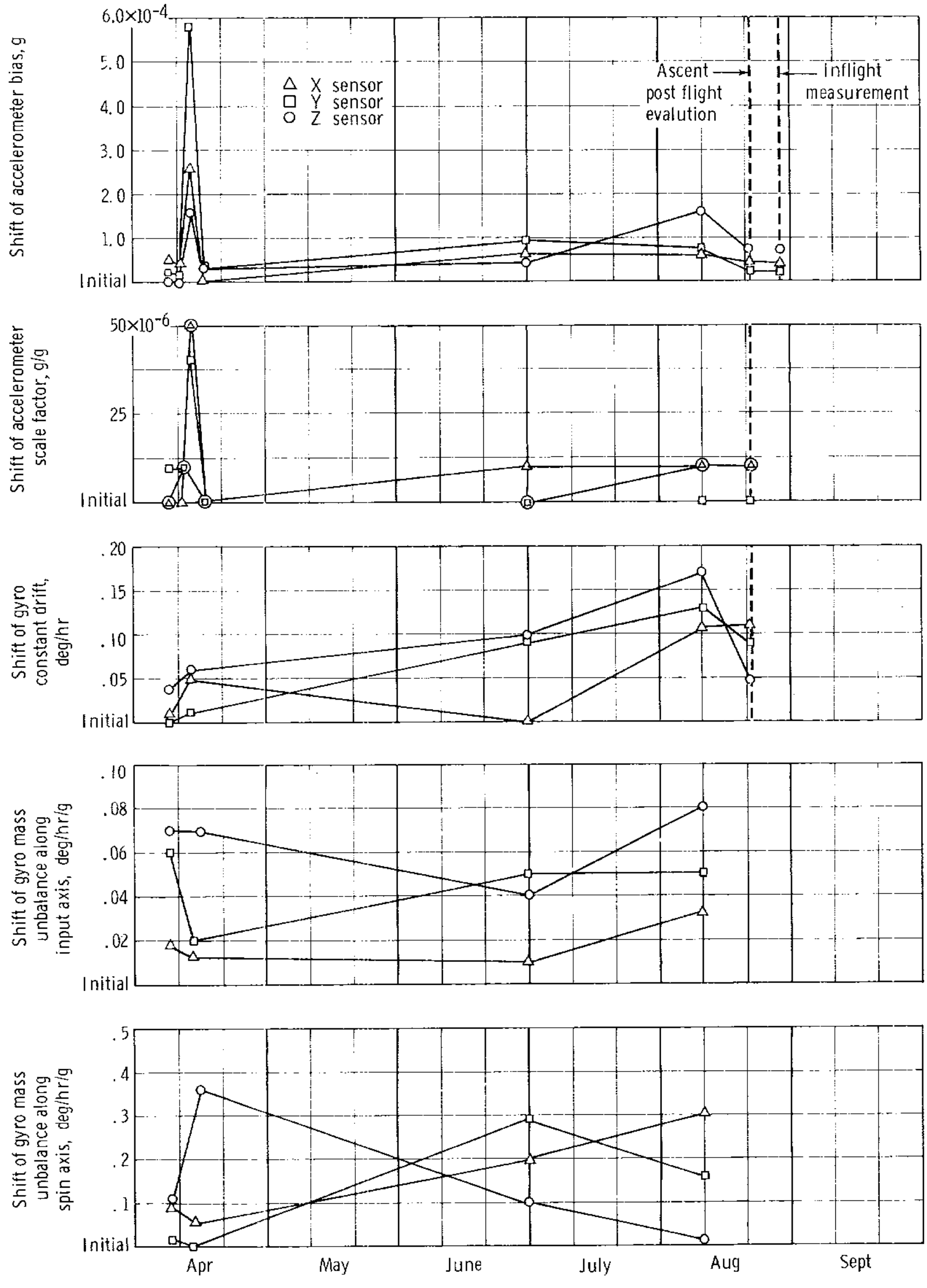

Figure 5.1.5-3. - IMU error coefficient history.

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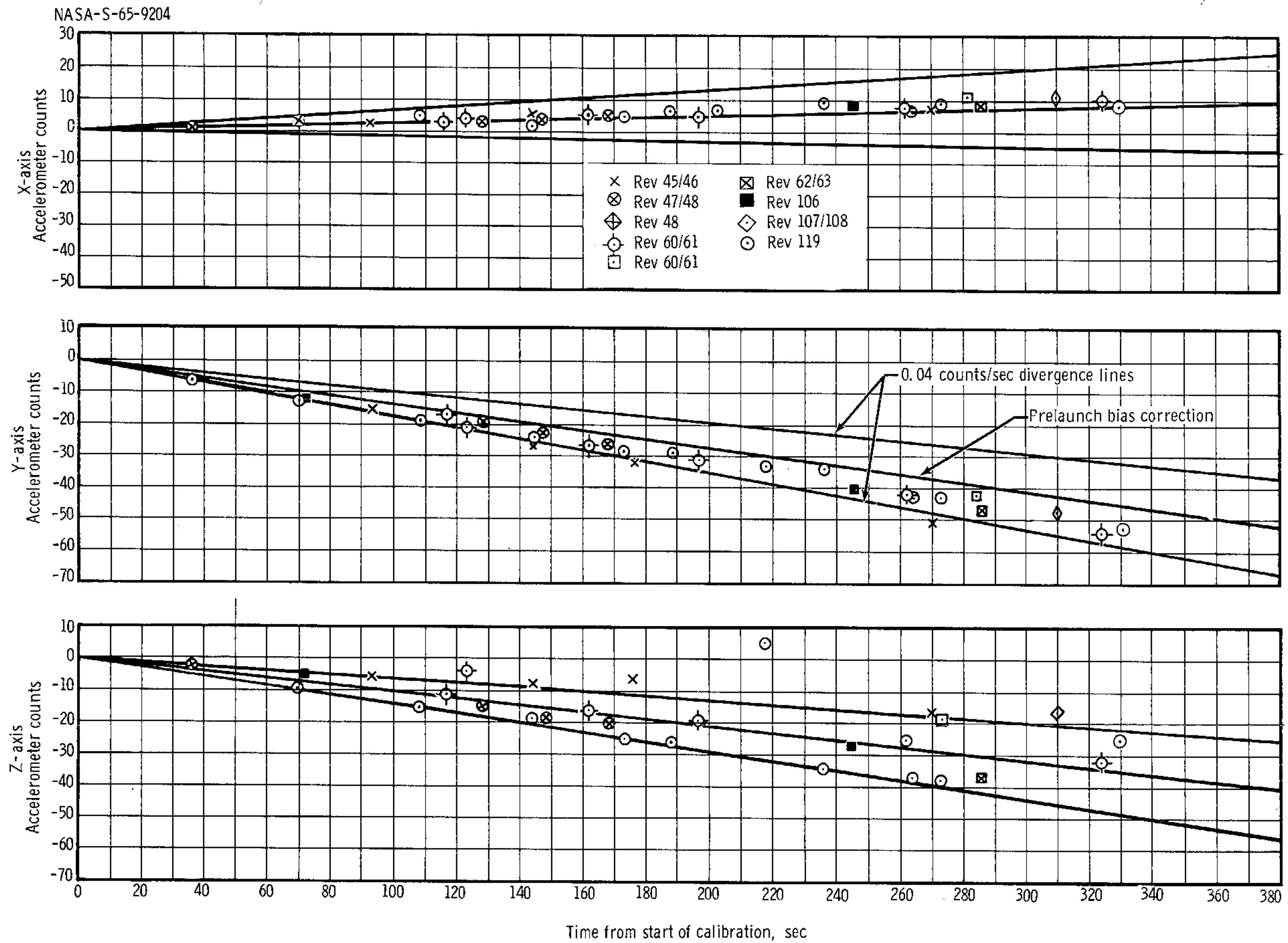

Figure 5.1.5-4. - Inflight accelerometer bias calibration data.

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NASA-S-65-8553

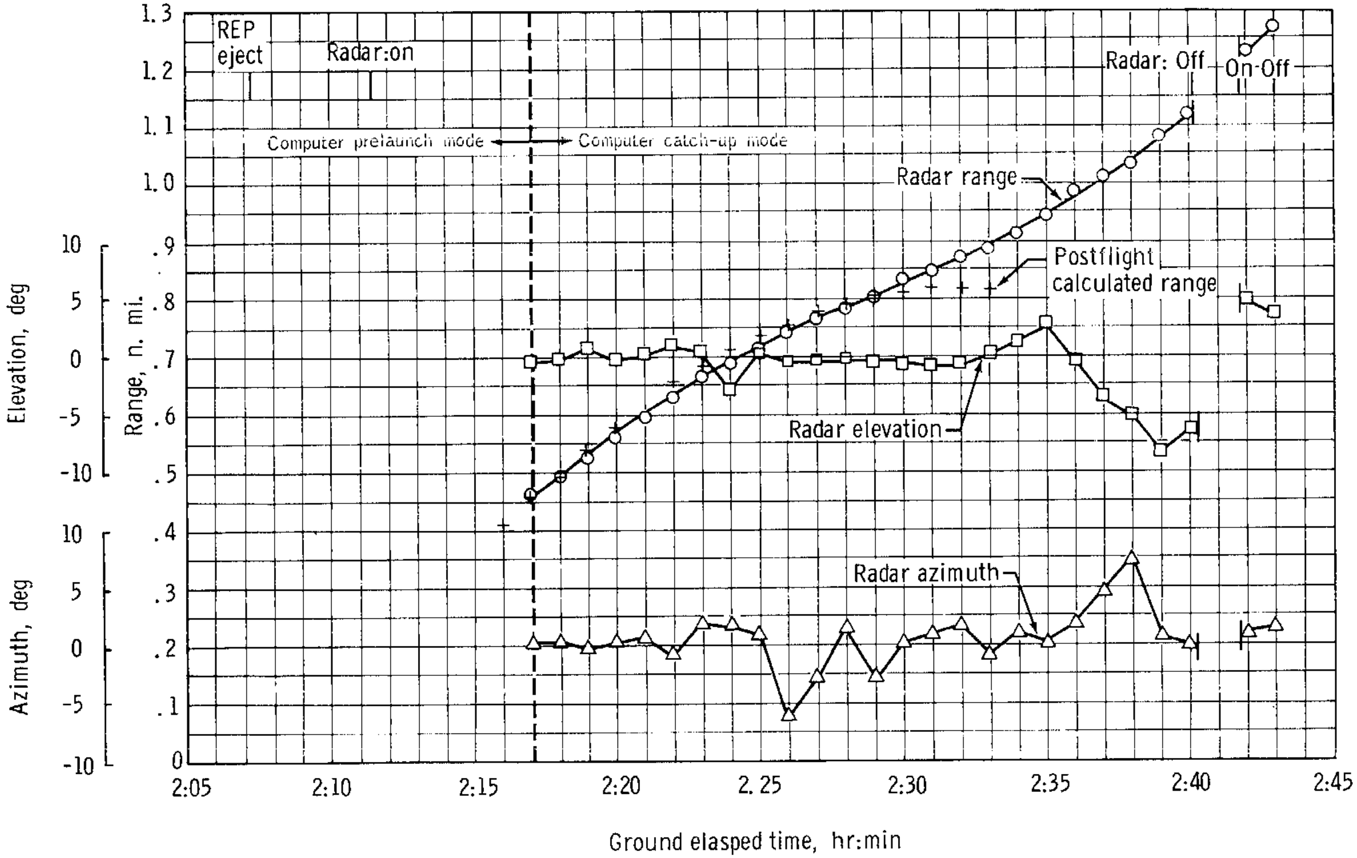

Figure 5.1.5-5. - Rendezvous radar data, uncorrected for spacecraft attitudes, during test with the REP.

NASA-S-65-8548

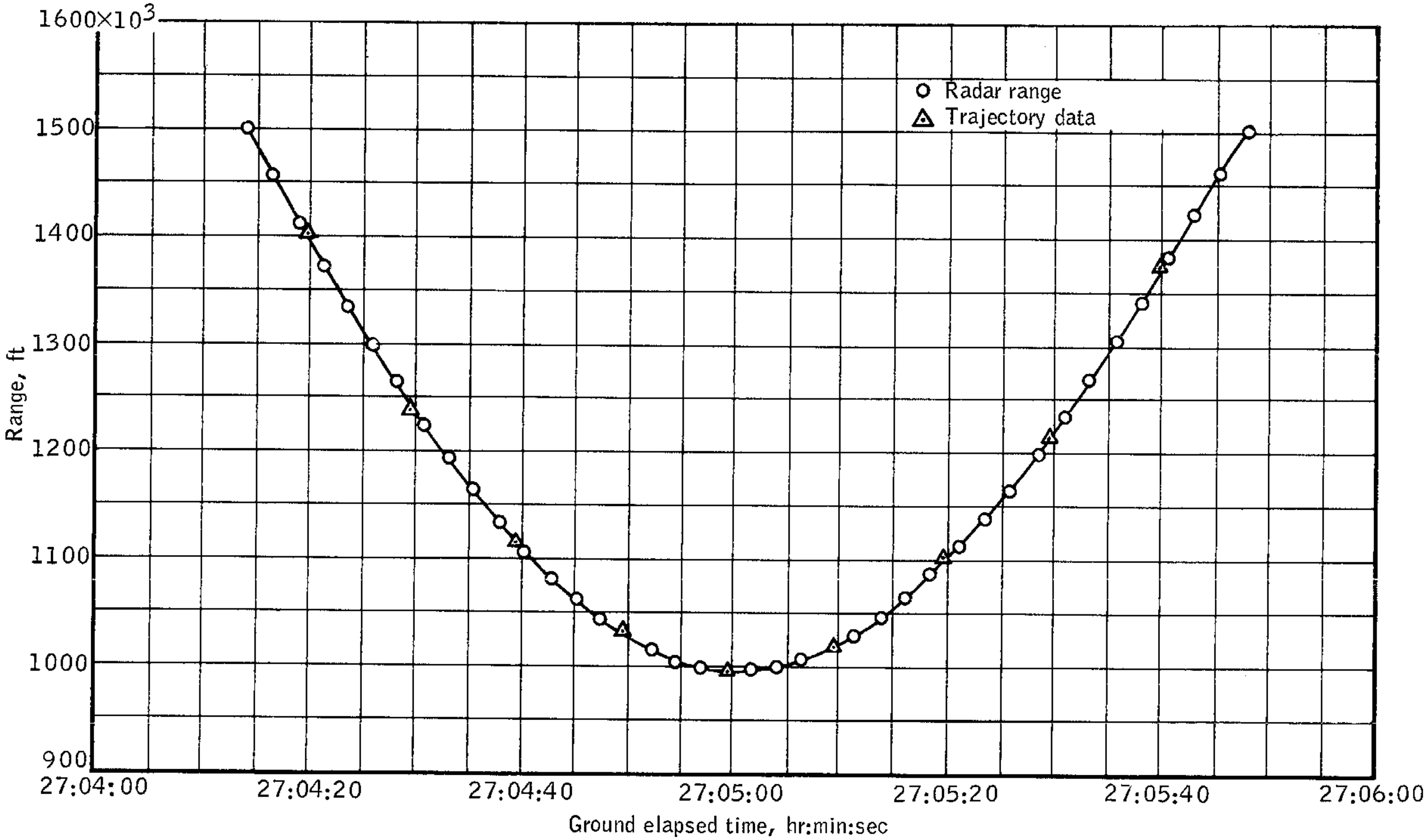

Figure 5.1.5-6. - Rendezvous radar range performance during first test over KSC.

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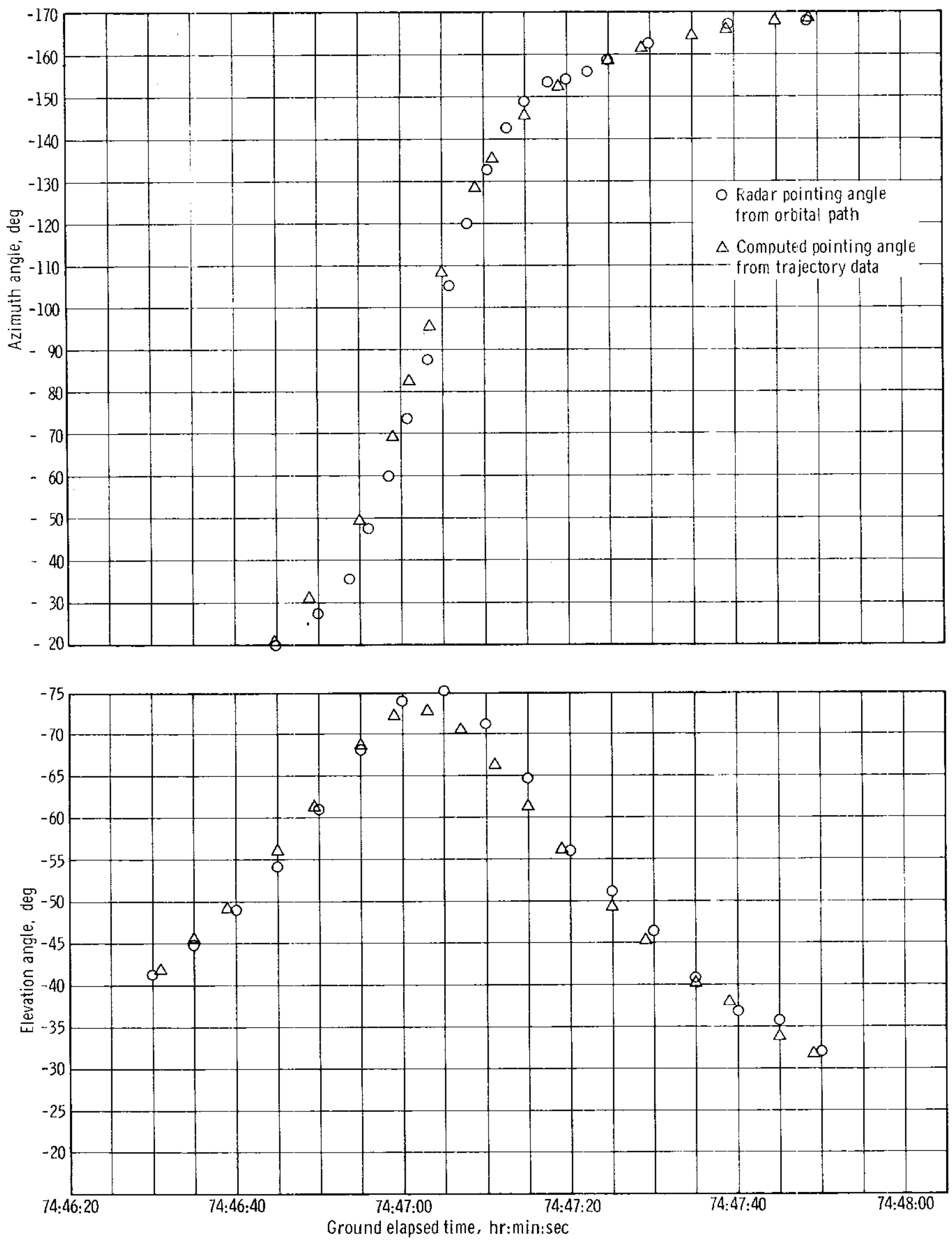

Figure 5.1.5-7.- Rendezvous radar pointing performance during second test over KSC.

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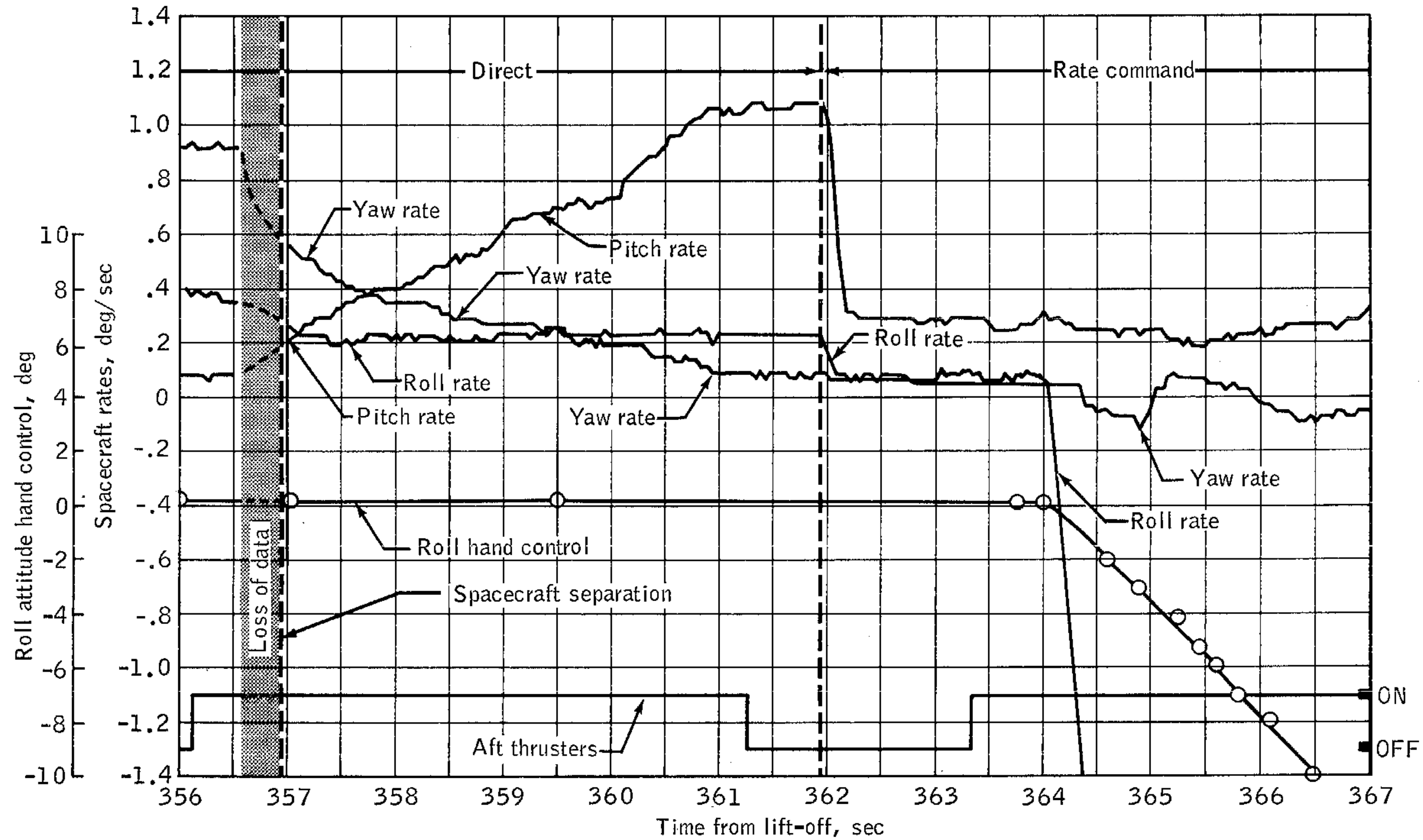

Figure 5.1.5-8. - Separation sequence.

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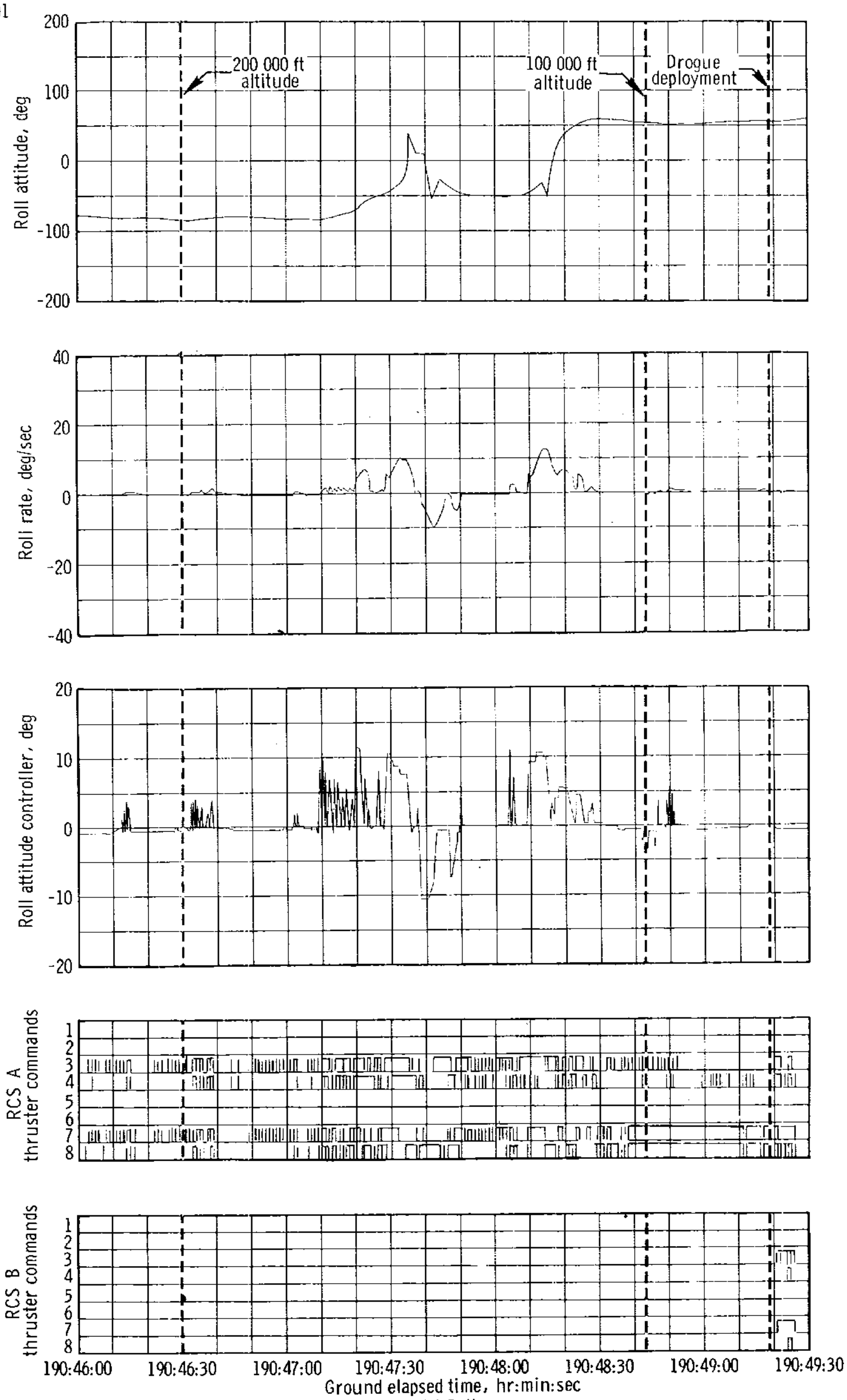

(a) Roll

Figure 5.1.5-9. - Typical reentry time histories.

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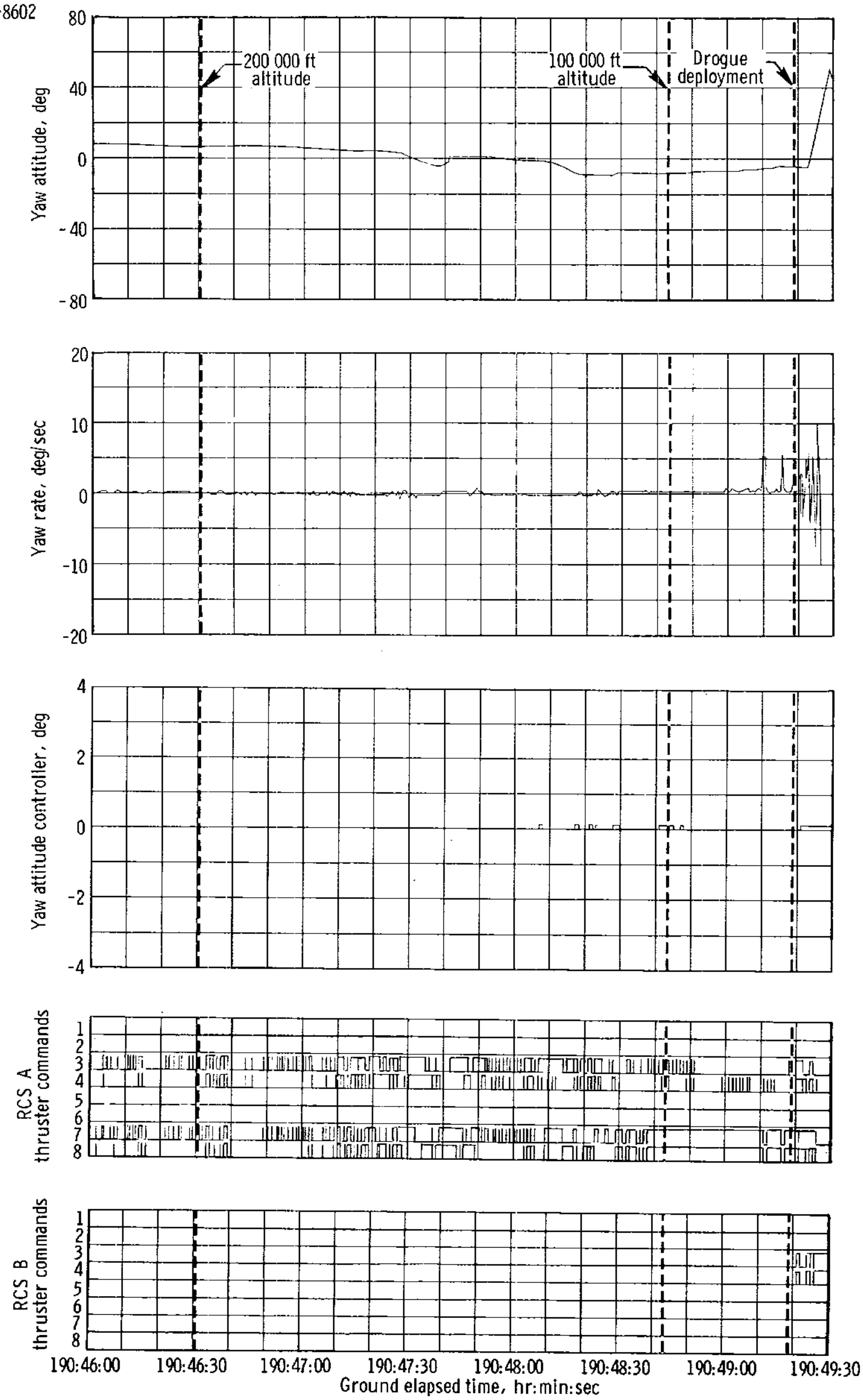

(b) Yaw

Figure 5.1.5-9. - Continued.

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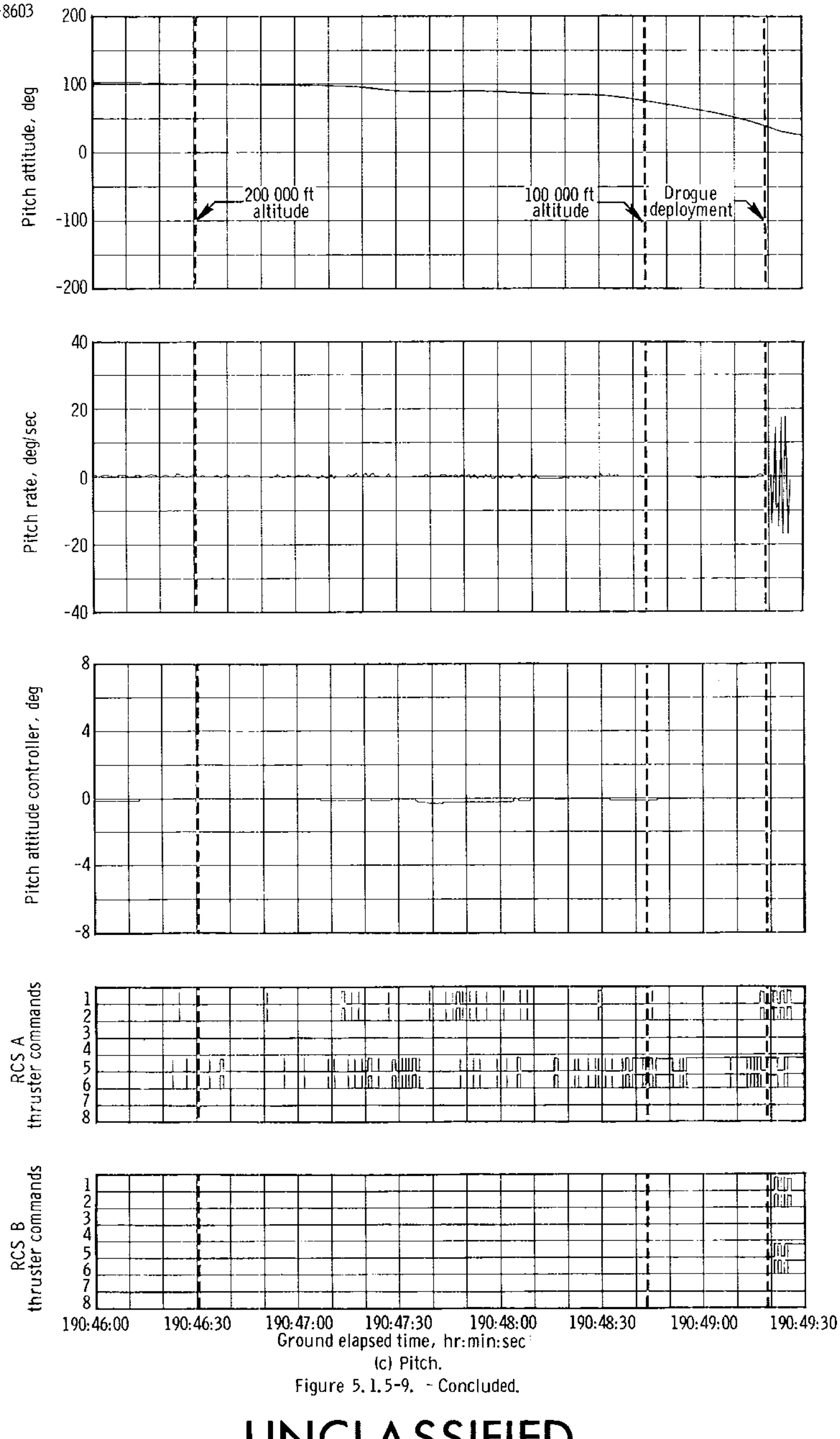

(c) Pitch.

Figure 5.1.5-9. - Concluded.

5.1.6 Time Reference System

All available data indicate that the Gemini V time reference system components performed according to specifications during the entire mission. The electronic timer maximum error was 442 milliseconds or 0.65 part per million per day, which is well within the specification of 35 parts per million per day. The electronic timer correctly initiated the automatic retrofire sequence. The event timer was used several times during the mission and found correct when checked against other sources. The right-hand 8-day G.m.t. clock was reset $2\frac{1}{2}$ minutes at $6\frac{1}{2}$ days after launch and was approximately 1 minute fast after recovery. The flight crew reported that the left-hand G.m.t. battery-operated clock required about 5 seconds correction in 8 days. Correct and recoverable timing was recorded in the onboard voice and biomedical tape recorders indicating that the time correlation buffer operated properly.

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5.1.7 Electrical System

5.1.7.1 Fuel cell.- During the ascent phase, the fuel cells supplied approximately 86 percent of the overall main bus load. During the orbit phase, the main batteries were switched off the bus and the fuel cells provided 100 percent of the main bus power. The maximum load required from the fuel cells was 47.2 amperes at which they maintained 25.5 volts. The minimum load was 11.1 amperes which was supplied at 27.6 volts. During the period in which the reactant supply system (RSS) oxygen-heater fault tripped the circuit breaker (25 minutes 51.2 seconds g.e.t.), the main bus voltage momentarily depressed to 25.08 volts.

5.1.7.1.1 Performance variation: No anomalies were observed in the electrical performance of the fuel cell. The performance was within specification and variations observed were consistent with variations during the extensive ground test program.

The performance of section 1 was plotted discarding all data taken when the inlet coolant temperature to the section was below 70° F (fig. 5.1.7-1). At a load of 15 amperes, approximately a 0.4-volt decline was observed between the section's second activation on August 18, 1965, and the performance on August 21, 1965, the first day of flight. Continuing operation showed a gradual increase in performance until the eighth day of flight, when the performance was approximately equal to that experienced at the second activation.

The performance of section 2 was also plotted, discarding all data taken when the inlet coolant temperature was below 70° F (fig. 5.1.7-2). However, because of the varying coolant temperature, the data taken on the first day were limited to a period of 4 hours 45 minutes whereas the data for the eighth day were plotted for two periods of approximately 3 hours each. These data at 15 amperes show a decline of approximately 0.60 volt between the second activation on August 18, 1965, and the first day of flight. The data show an additional decline of 0.66 volt over the 8 days of flight, most of which occurred during the three periods of open circuit. When the effects of varying coolant temperatures were taken into account, the degradation during the 100-hour period was approximately 0.13 volt at 15 amperes, while a 0.31-volt improvement was realized during the last operational period.

5.1.7.1.2 Effect of coolant temperature: The data were tabulated in terms of current at constant voltage and coolant inlet temperature for section 1 only. Section 2 was not considered because of the many complexities associated with open-circuit operation and stagnant coolant.

Because of an abundance of data points, 27.6 volts was selected for the low load condition. The data for intermediate and high load conditions were sparse and erratic, necessitating analyses over the ranges of 26.7 ± 0.1 V and 25.6 ± 0.1 V. The data indicate that there is a temperature compensation factor of 0.17 and 0.19 A/°F at the low and intermediate load conditions (see fig. 5.1.7-3). The mission load conditions were not suitable to indicate the same smooth pattern at the high loads, but the trend was similar.

5.1.7.1.3 Open circuit effects: During the flight, section 2 was placed on open circuit, without coolant flow, for three periods of approximately 19 hours each. Open circuit operation with the coolant loop shut down was deemed desirable to conserve the ampere-hours drawn by the coolant pump. When the effect of coolant temperature variation was taken into account, the voltage degradation, compared at 8 amperes for each of these three periods, was 0.27 volt. Comparing only the performance which occurred during the periods of operation following each open circuit period shows a net rise of 0.15 volt in section 2 performance.

5.1.7.1.4 Purge sensitivity: The purge sensitivity exhibited during the mission was found to be normal. An average recovery of 0.1 volt resulted from the oxygen and hydrogen purge sequences.

5.1.7.1.5 Differential pressure warning light indications: Differential pressure warning light indications occurred three times during the mission: during launch, during the first hydrogen purge of section 1, and during an attempt to purge section 1 without opening the crossover valve.

Launch: Because acceleration pressure heads on the spacecraft 5 water system (on the fuel cell side of the absolute pressure water regulator) low oxygen-to-water differential-pressure warning lights can be expected at accelerations of approximately 2g or greater, and oxygen-to-hydrogen lights (low O_2 to H_2 on section 1, high O_2 to H_2 on section 2) can be expected for short periods following BECO and SECO. Such indications were observed during the launch. While the exact pressure conditions experienced cannot be determined, worst-case analysis indicates that reverse pressures of between 0.5 and 3 psi, depending on the amount of gas in the B tank (see section 3.1.2 for description), might have been imposed across the water separator plates. The design-proof pressure of the water separators in the reverse direction is 2.0 psi. However, no apparent damage was caused by the pressure conditions which actually existed. The reverse pressure experienced could be minimized by minimizing the amount of gas in the fuel cell side of the B tank and could be eliminated after a slight spacecraft modification by closing the water valves during launch.

Hydrogen purge: Pressure drops across the dual regulator, line, and manifold normally cause the warning light to come on after the start of a hydrogen purge and off immediately after completion. The warning lights did not illuminate during hydrogen purge conditions on this flight after the first purge of section 1. The cause for this cannot be determined with the information obtained; reduction in dual-pressure-regulator differential pressure due to reduced oxygen-tank pressure, on the basis of ground tests at the spacecraft contractor's plant, could not solely account for the lack of a high oxygen-to-hydrogen differential-pressure warning light indication.

Oxygen and hydrogen purge with crossover closed: During the purge occurring at approximately 29 hours 30 minutes g.e.t., the crew initially neglected to open the crossover valve, which resulted in warning light indications and abnormal differential pressure across the cells. The magnitude and duration of the differential pressure excursions were estimated to be in excess of 6.5 psia (O_2 greater than H_2 for 10 seconds) and zero to slightly negative for approximately 5 seconds. While no apparent damage was done to the fuel cell by these pressure excursions, differential pressures of the same magnitude but of longer duration could prove damaging on future flights, and care must be taken to avoid this condition.

5.1.7.1.6 Load sharing: Load sharing of the six fuel cell stacks is shown in table 5.1.7-I. The specification requirement of ±1.5 ampere per stack for $\frac{1}{3}$ of total section current was met. While the inflight performance of section 2 declined, the performance of section 1 improved, resulting in a shift in load sharing between the two sections. Past experience has shown that load sharing is partially a function of individual stack coolant temperatures (which naturally varies because of the series coolant flow through the stacks) and manufacturing tolerances, as well as relative performance decay.

5.1.7.1.7 Cryogenic usage rate and water production rate: Since the fuel cell was flown for the first time, it is important to future mission planning that the cryogenic usage rates be determined for this mission. The water-production rate and water-separator oxygen leakage are also important as long as they are delivered into a fixed storage volume. Should this volume fill up, the water would then back up into the fuel cell sections and gradually reduce the performance to zero.

The data from the first 40 hours were used to determine usage rates because the heat leakage into the hydrogen tanks was sufficient to cause venting after 43 hours. Using the flight hydrogen-quantity data (accounting for hydrogen loss and purging) and postflight computer-summed ampere-hours, the pounds of hydrogen per ampere-hour were obtained (table 5.1.7-II). The cryogenic oxygen heater circuit failed after about 26 minutes of flight, causing a decay in the tank pressure. When the temperature and pressure reached a combination of -265° F and 70 psia, the oxygen passed from the single-phase state to the two-phase state (liquid and gas) causing the oxygen-quantity readouts to become meaningless. Therefore, the usage rates in table 5.1.7-II were calculated from hydrogen data, applying the ratio of 8 to 1 for the chemical combinations of O_2 and H_2.

The water-production rate was a difficult calculation of somewhat questionable reliability. One of the necessary parameters was the quantity of water drunk by the flight crew. This quantity was measured by totaling the 1-ounce swallows for both crew members. The water generation rate of the fuel cell was determined from the water consumed by the crew members, the pressure and volume of water in tank A, and the original loading of the water system. Another estimate of the water production rate was made combining H_2 and O_2 usage rates, assuming all gases used produced water.

All factors taken into consideration, table 5.1.7-II shows good agreement between the measured rates from flight data, theoretical values, and ground-test data. Estimates of these quantities made during the mission were high, mainly because the running estimate of ampere-hours used could only be approximated, using real-time telemetry data taken over ground stations.

5.1.7.2 Reactant supply system.- The reactant supply system (RSS) provided gaseous supplies to the fuel cell throughout the entire mission. A failure in the heater circuitry of the RSS oxygen storage tank occurred at 25 minutes 51 seconds g.e.t., resulting in a loss of tank pressure control. This loss required a severe reduction in spacecraft power consumption until ground tests, theoretical calculations, and inflight tests showed that the system would support normal fuel-cell operation for the power requirements of the mission. The RSS hydrogen storage tank operated as predicted throughout the mission.

5.1.7.2.1 RSS oxygen: The RSS oxygen tank was filled with 178.2 pounds (99 percent of design load), and pressurized to 815 psia at launch. The internal heater was in the AUTO mode, thus allowing the pressure switch to energize the heater to increase tank pressure toward the pressure switch cut-off point of 875 to 910 psia. Tank pressure increased to 853 psia at 25 minutes 51 seconds g.e.t. when the heater circuitry failed. Calculated pressure-rise rates during the period from 10 to 25 minutes after launch indicate that the heater was active.

The pressure then declined gradually until stabilization occurred at approximately 70 psia around 4 hours 22 minutes g.e.t. Flight-crew observance of this pressure trend led to a check of the heater switch and the circuit breaker, at which time the fuel-cell oxygen and hydrogen heater circuit breaker was found to be in the off, or tripped, position. This circuit breaker was reset by the pilot at approximately 50 minutes g.e.t.; however, all efforts to reestablish oxygen pressure control by means of the heater were unsuccessful. Cycling of the RSS oxygen heater switch from off to the automatic or manual mode did not trip the circuit breaker, but at the same time the cycling failed to produce the expected current increase associated with normal heater operation. Examination of fuel-cell stack currents and main bus-voltage data (figs. 5.1.7-4 and 5.1.7-5) revealed a current spike sufficient to trip the oxygen heater circuit breaker and a voltage depression of the main bus to 25.1 volts at approximately 25 minutes 51 seconds g.e.t. Immediately following the fault, the total system current was 2.8 amperes less than the system level prior to the fault. This is the magnitude of current required by the RSS oxygen heater.

It appears very likely that the fault current was caused by a short in the oxygen heater circuit and was responsible for tripping the circuit breaker. It may also be deduced that the fault cleared itself but left the heater inoperative for the remainder of the flight. The portion of the heater circuitry recovered with the reentry assembly has been checked and found to contain no faults.

The pressure decline from 853 to 70 psia is shown in figure 5.1.7-6. Analysis indicates that the fluid state at the 70-psia point was coincident with the saturated liquid line on the primary-enthalpy curves for oxygen. Subsequent extraction from the tank to support the fuel cell electrical load resulted in penetration of the two-phase (liquid and vapor) region for operation during the remainder of the flight. The energy balance between extraction and ambient heat leak permitted a gradual pressure increase to 260 psia at the end of the mission. The mission was completed with an estimated 40.5-percent (73 pounds) oxygen remaining in the tank. The two-phase mixture at the end of the mission was approximately 50 percent liquid by volume. Within the two-phase region and with the heater inactive, the tank performance was sensitive to the liquid-to-vapor mixture extracted to supply the fuel cell. For this particular tank, if the extracted fluid had been all high energy vapor, the maximum extraction rate without a decrease in pressure would have been equivalent to a 12-ampere load. For pure liquid extraction, the flow rate to support in excess of a 100-ampere load would have still allowed the tank pressure to increase. Analysis of flight data indicates that at all times during the mission, the extracted fluid was more than 60 percent liquid (by weight) and the tank pressure was always increasing. Using best estimates of the tank ambient heat leak and extraction rates, a detailed analysis was made to determine the effect of extraction rate on the percent liquid in the removed fluid. These calculations conclude that increased extraction rates result in larger percent liquid ratios and thereby insure continuous rise of vessel pressure. This analysis indicates that if this self-regulating effect had been known in advance, the RSS oxygen vessel could have been used to support the intended high electrical loads early in the mission. If the electrical load had been left at the 30 to 40 ampere level, however, the initial pressure decline from 26 minutes to 4 hours g.e.t. would have been approximately twice as rapid because the oxygen was still in the single-phase region.

Two factors assured adequate supplies to the fuel cell section in spite of the low tank pressure. First, for liquids with wetting characteristics of liquid oxygen and hydrogen, tests at the Lewis Research Center have shown that in a two-phase regime (vapor and liquid) the vapor becomes a central bubble while liquid remains around the walls of a spherical container in the weightless state. Also the outlet port of the tank was located adjacent to a deflector for vortex elimination and to the capacitance probe so that the liquid oxygen meniscus to the deflector and probe covered the outlet port. These effects certainly aided in assuring liquid-rich extraction and attendant pressure rise after entering the two-phase region. Second, the dual-regulators which regulate the inlet pressures of the fuel-cell reactants were capable of within-tolerance regulation at a far lower supply pressure than the specification minimum. Special tests were conducted at the spacecraft contractor's facility during the first day of the mission to determine regulator sensitivity to inlet pressure. These tests demonstrated that a 50-ampere load could be supported with only 55-psia inlet oxygen pressure without simultaneous purge.

5.1.7.2.2 RSS hydrogen: The RSS hydrogen tank was filled to 103.8 percent (23.1 pounds) and 150 psia at launch. Filling to over 100 percent (reading on the special gage for design load and ullage) was accomplished by reducing the ullage below the design value. Overfilling was necessary in order to satisfy the predicted venting in addition to the power requirements of the planned mission. Prelaunch testing showed this tank had an excessive ambient heat leak and provided data for an accurate prediction of inflight performance. The tank pressure was maintained as low as possible at launch and thereafter so as to delay the start of venting. The tank heater was used only during the early portions of the mission, when power consumption was high, to maintain a minimum tank pressure of 100 psia. The combination of subsequent reduction in power consumption and ambient heat leak increased the tank pressure to the vent level of 350 psia at 43 hours g.e.t. Venting continued to 167 hours g.e.t. with a brief period of venting at approximately 177 hours g.e.t. Peak venting rate was calculated at 0.155 pound per hour at 120 hours g.e.t. The relief valve performed adequately by using the pilot portion exclusively, except for two main poppet actuations at approximately 118.5 and 122.5 hours g.e.t. Main poppet actuations were normal. The quantity remaining at the end of the mission was 6.8 percent (1.51 pounds).

The hydrogen venting from 43 hours g.e.t. to 167 hours g.e.t. caused a small yaw-left rate acceleration of the spacecraft. The flight crew elected to damp out these rates when they had built up to approximately 6 to 8 deg/sec. The venting configuration will be modified on future spacecraft to vent through the c.g. of the spacecraft, thus producing little or no rates when venting.

5.1.7.3 Power system.- Nominal electrical power was supplied during all but the postlanding phase of the mission. Twenty-five seconds after landing, a high current drain was recorded by the PCM tape recorder and observed by the flight crew. The currents and voltages recorded were erratic but continually rising to the end of the tape at 5 minutes 15 seconds after landing. The peak current recorded was 41 amperes and the lowest main bus voltage was 23.2 volts. No explanation can be given at this time for this condition. An investigation is in progress and any necessary corrective action will be determined and made effective as necessary.

Postflight inspection of the Gemini V spacecraft revealed that seven fusistors and one fuse were open. The fusistors were blown as a result of slag formation in the pyrotechnic cartridges during firing, which caused an electrical short circuit to the case of the pyrotechnic. A similar reaction involving fusistors occurred on GT-3 and Gemini IV. The urine-tube heater fuse was found to be open. An investigation is in progress to determine if the urine-tube heater circuitry and components are in a normal condition. The urine-tube heater was operative during the entire flight.

The squib batteries handled the added currents caused by the short circuits to the pyrotechnic cases in all instances, with a minimum recorded voltage during the transient of 19.31 volts.

Flight battery discharge after the mission showed that 7.3 percent of the main and 59.2 percent of the squib batteries rated capacities were used during the mission.

5.1.7.4 Sequential system.- The performance of the sequential system during the mission was nominal. The major electrical sequential spacecraft events and times of occurrences may be found in table 4.2-I.

TABLE 5.1.7-I.- FUEL CELL LOAD SHARING

	Bus volts, 25.8				
	Day 1		Change percent bus, amp	Day 8	
	I, amp	Percent bus, amp		I, amp	Percent bus, amp
Stack 1A	7.02	16.70	+3.69	8.25	20.39
Stack 1B	6.45	15.35	+1.82	6.95	17.17
Stack 1C	7.65	18.20	+2.15	8.23	20.35
Section 1	21.12	50.2	+7.7	23.43	57.9
Stack 2A	6.65	15.82	-2.45	5.42	13.37
Stack 2B	6.63	15.77	-1.92	5.62	13.85
Stack 2C	7.65	18.21	-3.34	6.02	14.87
Section 2	20.93	49.8	-7.7	17.06	42.1
Total	42.05	100	--	40.49	100

	Bus volts, 27.3				
	Day 1		Change percent bus, amp	Day 8	
	I, amp	Percent bus, amp		I, amp	Percent bus, amp
Stack 1A	4.15	17.36	+5.02	5.06	22.38
Stack 1B	3.58	14.90	+2.58	3.95	17.48
Stack 1C	4.44	18.50	+3.35	4.93	21.85
Section 1	12.17	50.7	+11.0	13.94	61.7
Stack 2A	3.80	15.81	-3.97	2.67	11.84
Stack 2B	3.60	14.98	-1.83	2.97	13.15
Stack 2C	4.45	18.53	-5.26	3.00	13.27
Section 2	11.85	49.3	-11.0	8.64	38.3
Total	24.02	100	--	22.58	100

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TABLE 5.1.7-II.- FUEL CELL WATER USAGE RATES

Time, hr	Hydrogen, lb/A-hour	Oxygen, lb/A-hour (a)	Water	
			$H_2 + O_2$ (b)	Gulps (c)
15	0.0028	0.0224	0.0252	0.0243
24	.0027	.0216	.0243	.0256
30	.0028	.0224	.0252	.0234
34.5	.0027	.0216	.0243	- -
Theoretical	0.0027	0.0212	0.0238	
Ground test	0.0029	0.0252	0.0253	

[a] Calculated from H_2 ($O_2 = 8 \times H_2$)

[b] Calculated from $H_2 + O_2$

[c] Required flight crew water consumption by gulps (1 gulp = 1 oz)

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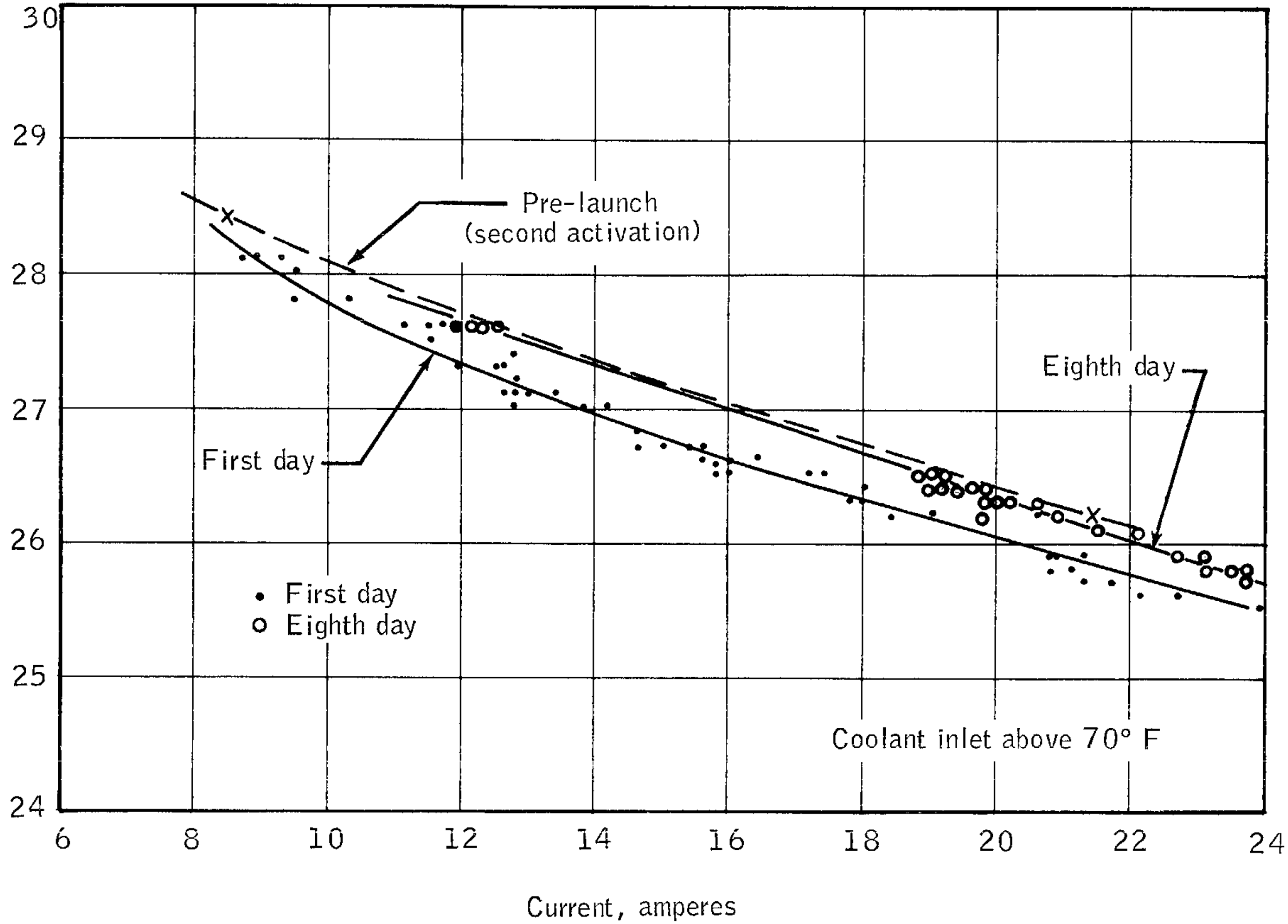

Figure 5.1.7-1. - Fuel cell section 1 performance.

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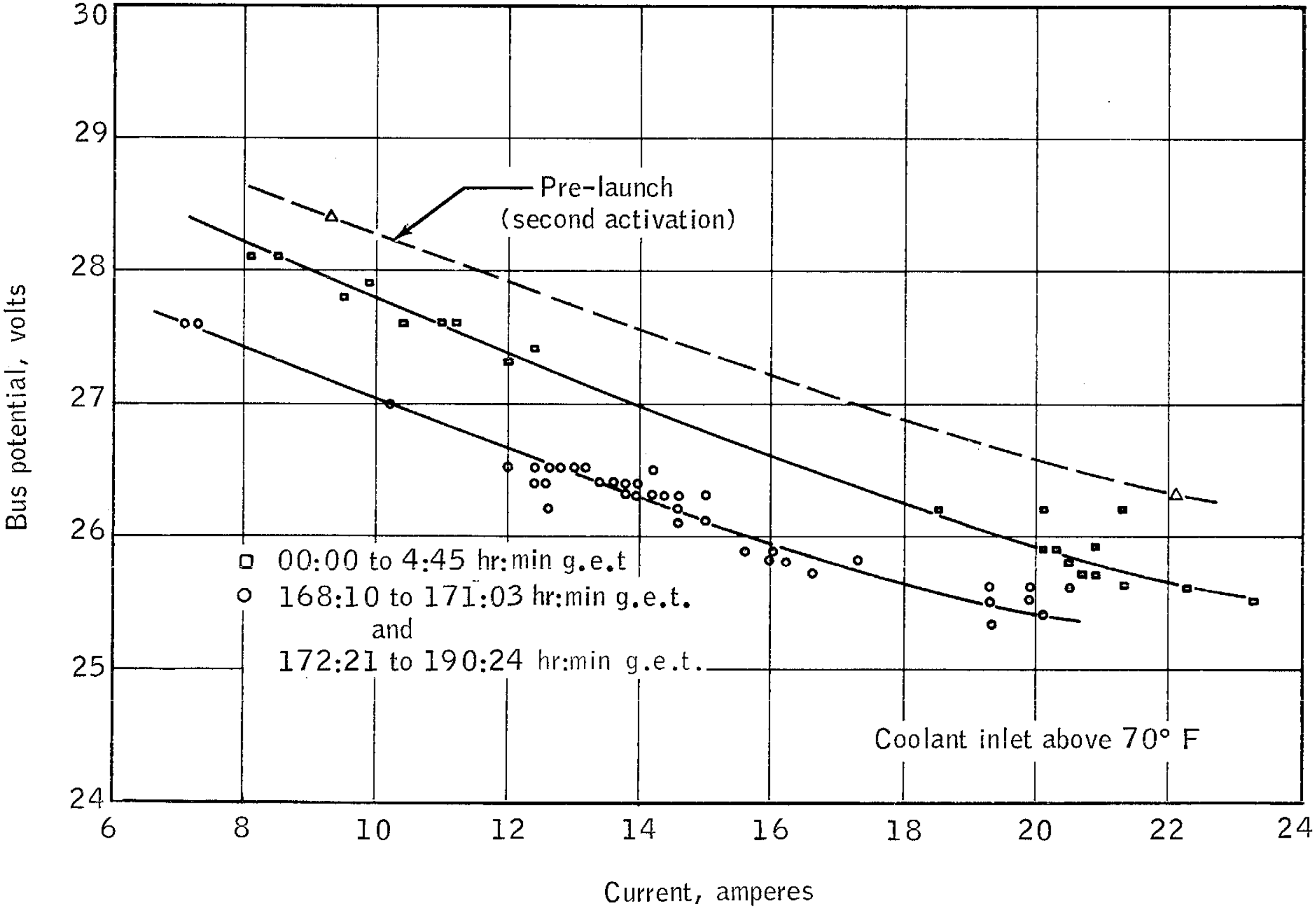

Figure 5.1.7-2. - Fuel cell section 2 performance.

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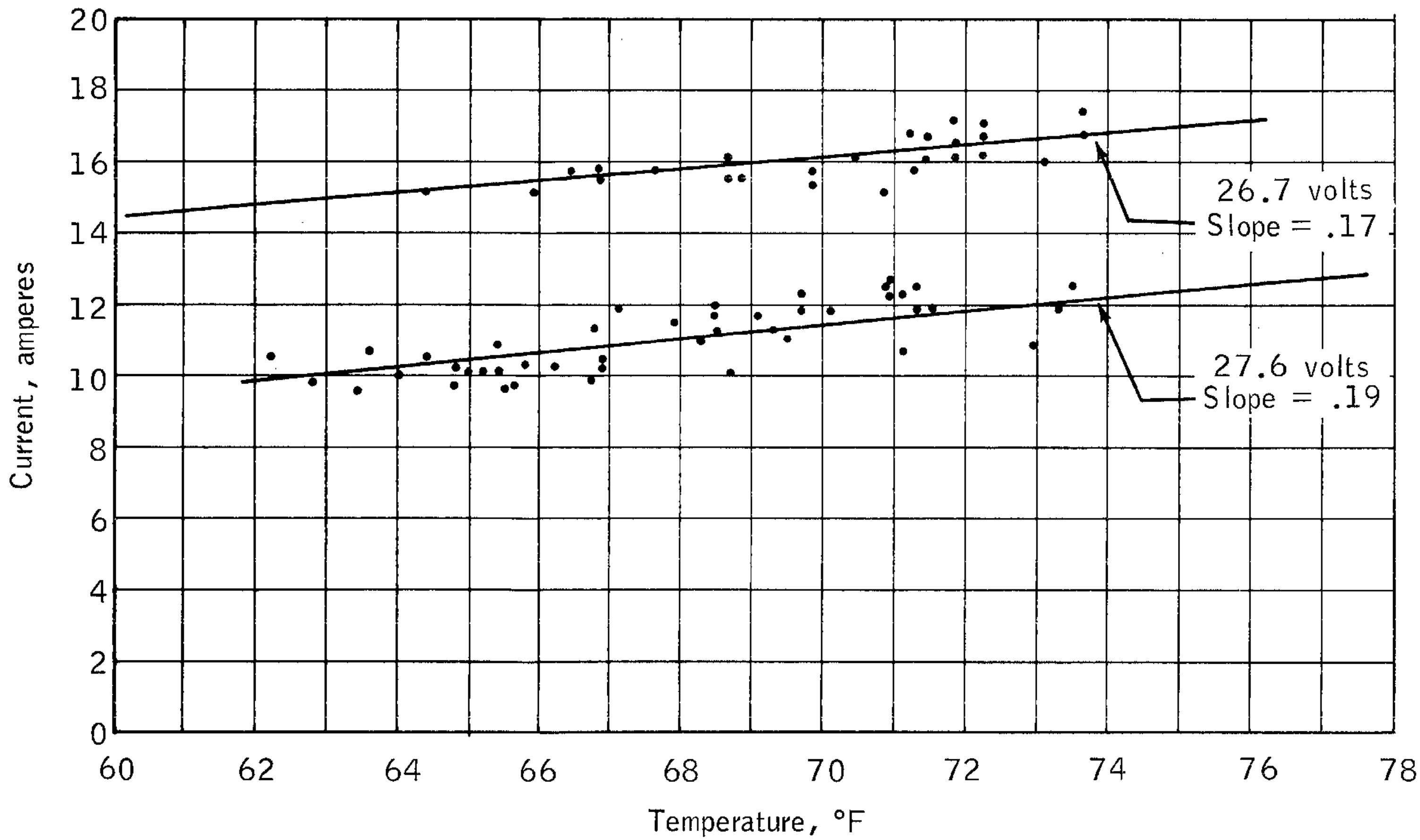

Figure 5.1.7-3. - Effect of temperature on fuel cell section 1 performance.

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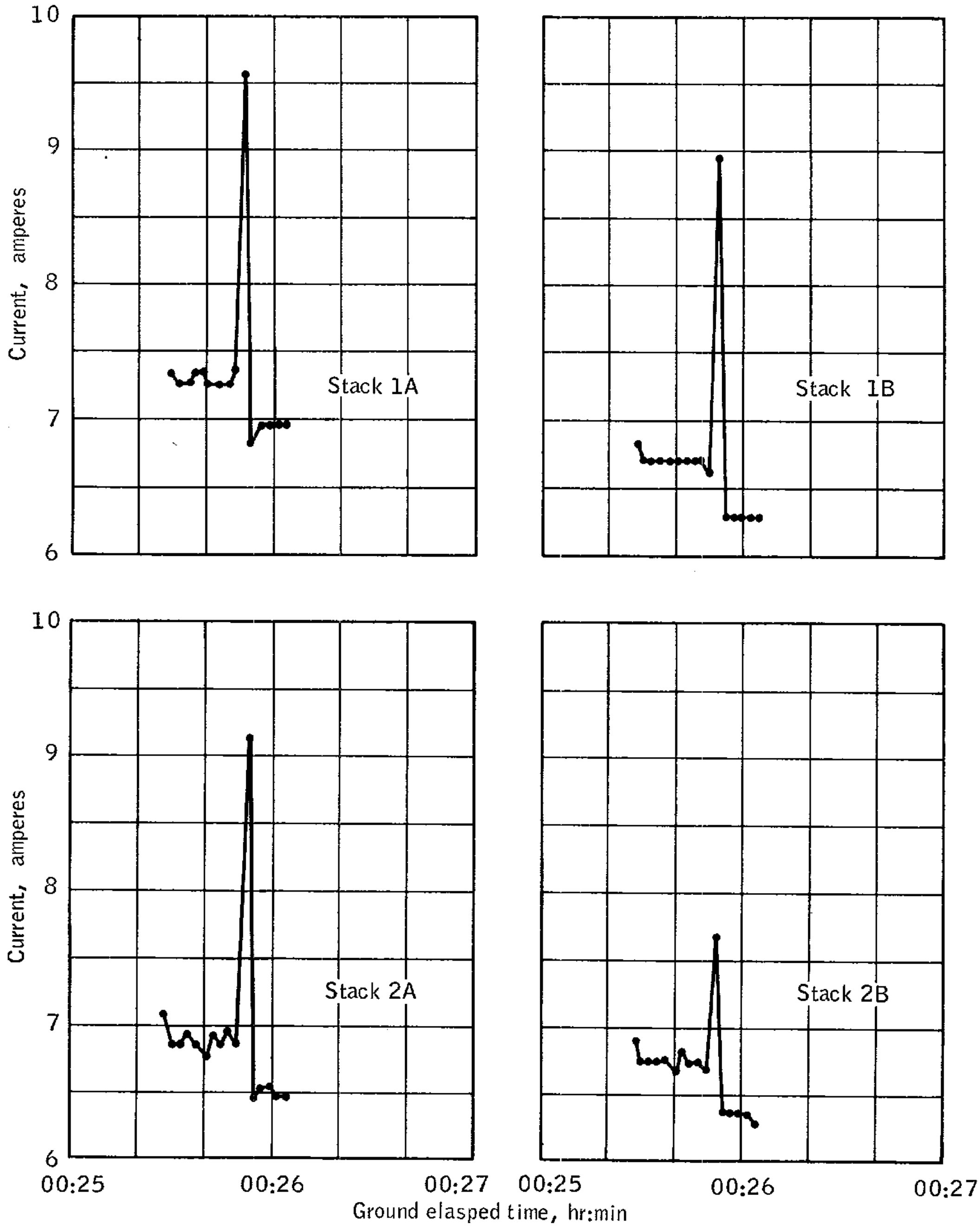

Figure 5.1.7-4. - Stack currents during time of RSS oxygen heater failure at 00:25:51 g.e.t.

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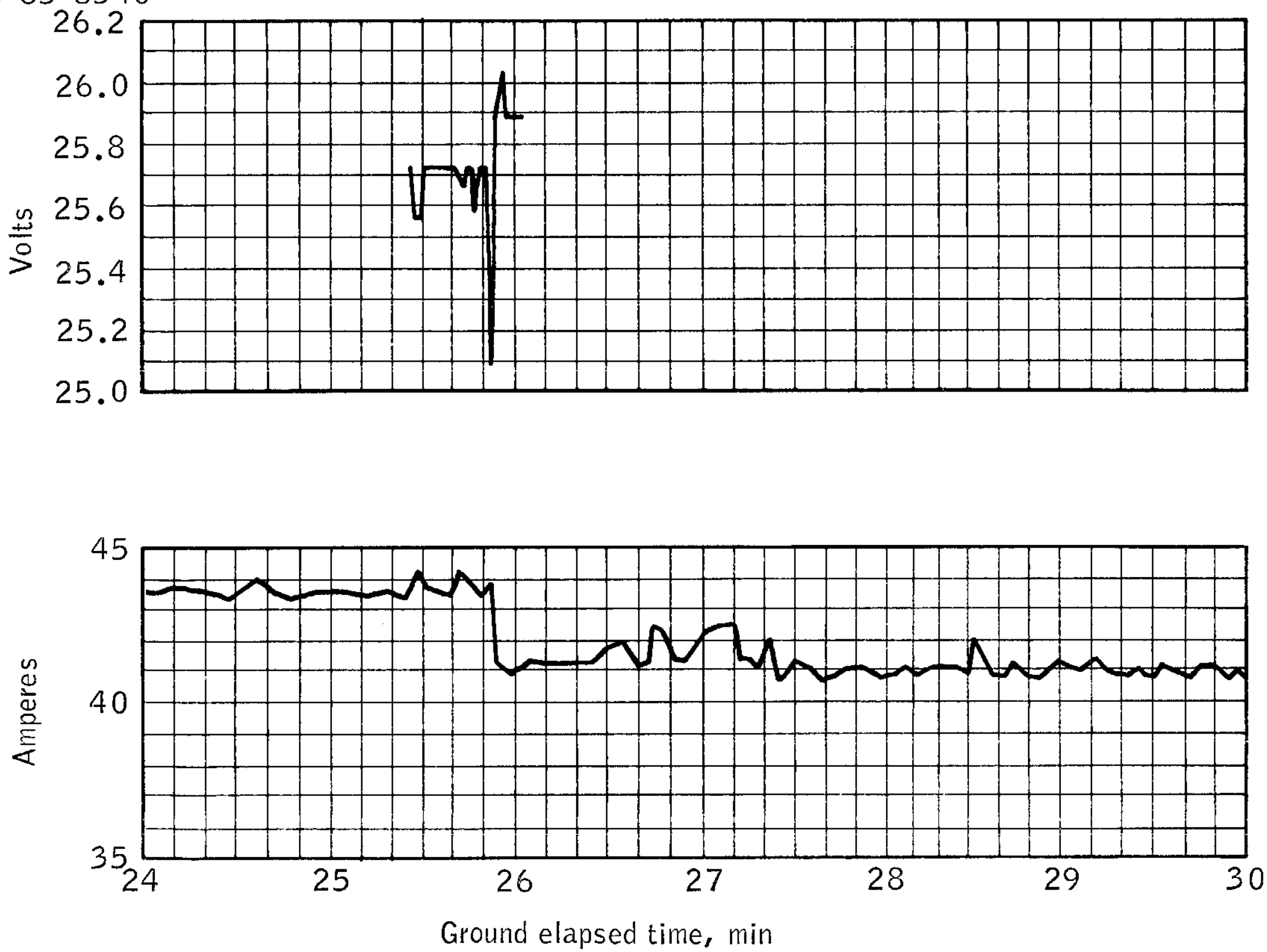

Figure 5.1.7-5. - Main bus voltage and current during time of RSS oxygen heater failure at 00:25:51 g.e.t.

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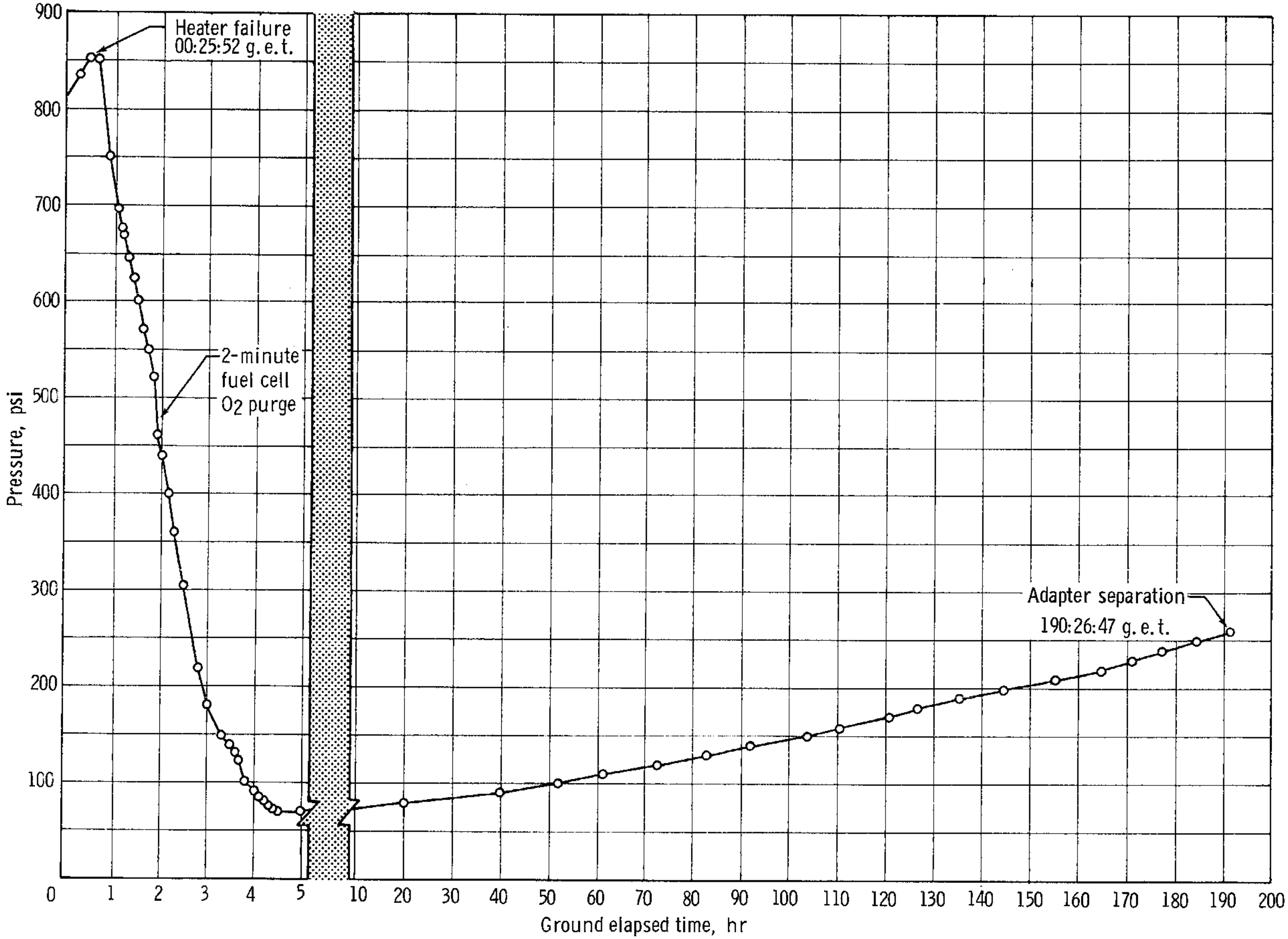

Figure 5.1.7-6.- RSS oxygen supply pressure.

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5.1.8 Spacecraft Propulsion Systems

5.1.8.1 Orbital attitude and maneuver system.-

5.1.8.1.1 Preflight: Propellant servicing of the orbital attitude and maneuver system (OAMS) was performed 21 days prior to lift-off, and the helium source pressure tank servicing was completed 11 days before launch. Table 5.1.8-I compares the planned and actual quantities of pressurant and propellant. These loadings constitute an available overall system mixture ratio of 1.31.

Activation of the system occurred at approximately 18 minutes before lift-off. With the exception of an increase in temperature of the oxidizer feed system and a higher than expected pressure drop of the helium pressurization gas, all parameters were within expected limits. The temperature in the vicinity of the oxidizer tank on the oxidizer feed line had increased 30° F within 30 seconds after activation. This increase appears to be reflected, but to a lesser extent, in the tank's skin temperature. The temperature history from activation to stabilization of the fuel, oxidizer, and pressurant tanks is presented in figure 5.1.8-1, and the subject is discussed further in section 5.1.8.1.3.

The static firing of all eight attitude thrust chamber assemblies (TCA's) provided a final end-to-end verification of system operation and the expulsion of gas entrapped in the propellant manifolds. In order to obtain satisfactory visual indications of thruster operation, all attitude TCA's were fired three times for an accumulated activation time of approximately 12 seconds.

5.1.8.1.2 Flight: System performance is discussed in this section under four basic functional groups: the attitude and maneuver TCA's, the propellant supply system, the pressurization system, and the heaters. The results of special tests which were conducted and were pertinent to the operation of the OAMS are also discussed briefly. At 00:26:05 g.e.t., all attitude TCA's were checked out in the direct mode. Spacecraft rates produced by all attitude TCA's at this time were satisfactory.

The first attitude TCA malfunction noted in the postflight analysis occurred at 75:16:31 g.e.t. during operation of the system in the pulse mode. In response to a roll-right command (TCA's 3 and 7), the spacecraft rolled right at a very low angular acceleration (noticeably less than the nominal 5.8 deg/sec^2) and produced a slight yaw-right acceleration, indicating that the thrust produced by TCA 7 was less than that of TCA 3. Also evidenced in the postflight analysis was improper performance of TCA 7 at 75:16:41 g.e.t. during a yaw-left command (TCA's 7 and 8) which produced a left roll couple. The crew reported TCA 7 inoperative at 118:32:01 g.e.t. The OAMS heaters were turned on when the crew observed "sluggish" system performance shortly before TCA 7 was reported inoperative. Prior to that time, the heater circuit had been activated only intermittently in order to hold the spacecraft electrical load to a minimum. Within another revolution, the crew reported TCA 8 inoperative; however, data for that period of time were unavailable for analysis and detailed performance could not be determined. The crew stated that, although the thruster had visible combustion, there was little or no thrust from it.

Rate data at 117:41:35 g.e.t. show pitch-up coupled with roll-right activity in response to a pitch-up command (TCA's 5 and 6) indicating poor performance from TCA 6. Checks of subsequent rate data showed that this TCA later became operative; however, other TCA's failed to operate properly later in the mission. A number of tests were conducted during the mission in an attempt to determine the exact nature of the problem with TCA's 7 and 8, without any positive results. The data necessary to define accurately the characteristics of failure and TCA performance changes during the mission are not presently available. The malfunctions, except TCA 8, are discussed in section 5.1.5.3.1. Possible causes of TCA malfunctions are discussed in section 5.1.8.1.3.

The use of maneuver TCA's was primarily restricted to six basic maneuvers with the aft engines. The radial engines were operated only in nulling lateral and vertical velocity components introduced during use of the aft TCA's. The forward-firing engines were not fired because of the problem with the fuel-cell oxygen-supply vessel. The crew reported no propulsion problems associated with any of the maneuvers. Specific data relating to the performance of the engines during the maneuvers are presented in table 5.1.8-II. Information relating to the maneuvers was not all available because the telemetry data were intermittent.

Figure 5.1.8-2 shows the percent of propellant remaining throughout the 8-day mission. Increases of propellant quantity after periods of extensive engine activity may be noted. These increases result from the system's operational principle of gas expansion. It is clear from this figure that discretion must be used in determining propellant remaining after periods of heavy TCA activity until the system has had time to stabilize.

The propellant-remaining quantities were calculated from the pressure decay of the helium pressurization gas and were corrected to account for variations in the ratio of expended oxidizer to expended fuel. This system mixture ratio is a variable quantity because the OAMS is composed of engines which operate at different mixture ratios (oxidizer to fuel (O/F)): 0.7 O/F, 23-pound thrust attitude TCA's; 1.2 O/F, 91-pound thrust aft TCA's, and 1.6 O/F, 91-pound thrust radial TCA's. Corrections for these variations are required because the densities of the fuel and oxidizer are not equal. The calculated cumulative expended oxidizer-to-fuel ratio is presented in figure 5.1.8-3. This curve provides an index of OAMS propellant utilization. The positive slopes result from use of the larger mixture ratio engines whereas negative slopes occur during periods of attitude engine activity. The cumulative expended oxidizer-to-fuel ratio curve is based upon the gross quantity of propellant consumed and that quantity burned by the maneuver TCA's. The propellant consumed in maneuver TCA's was selected because the firing durations of these engines are relatively long, and the maneuver-time instrumentation tolerance of ±0.2 second is less significant. Essentially all of the maneuver TCA propellant was consumed at a mixture ratio of 1.2 since only 5 seconds of radial TCA firing time was accumulated. Flow-rate data obtained from engine acceptance testing were assumed for these calculations. The information supplied by these two figures indicates that all usable fuel had been consumed near the end of the mission (181.5 hours g.e.t.) and that 79 pounds of usable oxidizer were remaining.

The onboard propellant quantity indicator values reported by the crew are shown in figure 5.1.8-2. A comparison of these values with ground-computed values shows agreement within the 7.5 percent accuracy of the indicator. When the indicator calibration curve was constructed, an average mixture ratio of 1.12 was assumed because that was the planned preflight value based on mission requirements. The helium pressurant loading was established to minimize variations between the propellant quantity indicator values and actual values near propellant depletion. Thus, at system activation, the gage should have indicated 91 percent for a 0-percent reading at propellant depletion. However, because the pressure dropped 103 psi more than expected, the gage showed 87 percent propellant remaining, which is the correct value for the pressure drop realized. This indicates satisfactory performance of the indicator and a good probability that proper propellant quantities were loaded into the tanks. The indicator values on figure 5.1.8-2 were not corrected for deviations from the 1.12 O/F because there were no means of making flight adjustments.

From the propellant quantity calculations, an overall mission mixture ratio of 0.815 was realized and 289 pounds of propellant were expended. From an average engine specific impulse (I_{sp}) measured in acceptance testing, the total impulse delivered to the spacecraft was 85 500 lb-sec. This impulse is based on an I_{sp} of 259 $\frac{\text{lb-sec}}{\text{lb}}$, for the attitude TCA's, 273 $\frac{\text{lb-sec}}{\text{lb}}$ for the aft TCA's, and 300 $\frac{\text{lb-sec}}{\text{lb}}$ for the radial TCA's.

Pressurization system – The excessive helium source pressure drop at system activation may be attributed to the following error sources: variations in system ullage from the nominal, instrumentation inaccuracies, less-than-planned quantities of propellant loaded, higher-than-nominal regulated pressure, and low system pad pressure. A loss of 20 psi can be attributed to variations in tank ullage if it is assumed that tanks manufactured with maximum volume tolerances were installed. Pressure instrumentation inaccuracy can result in a 40-psi error. Propellant servicing may have added a slight error. The higher-than-nominal regulated pressure accounts for 20 psi, and the low system pad accounts for 8 psi. The performance of the pressure regulator was satisfactory throughout the mission. The regulated pressure increased from an initial value of 300 psia to 308 psia near the end of the mission. During the 189th hour of the mission, the cartridge valves which isolate the source pressurant tanks from the system were actuated by the crew. The crew subsequently pulsed from the manual solenoid pressure valve, increasing the regulated pressure from 308 to 322 psia, thereby verifying its operations.

OAMS heaters – The possibility exists that some elements of the OAMS heater system connected to the TCA solenoids became inoperative as the mission progressed. The sequence of heater operation is presented in table 5.1.8-III. The current and voltage data indicate that the heaters were functioning properly during revolution 3. When the heaters were turned off, a current decrease of 2.35 amperes was recorded, indicating an intact heater system. In revolution 54 when the heater circuit was energized, there was a current increase of only 2.18 amperes. Closure of the heater circuit breaker in revolution 75 caused a 2-ampere current increase. These current changes are based on the sum of two parameters. Short-term instrumentation accuracy for small changes of this nature can be considered ±0.2 ampere on each parameter. Each thruster heater required only 0.062 ampere; therefore, the loss of 1 or 2 heaters would probably not be detected with such gross accuracy.

5.1.8.1.3 Failures and anomalies: The cause of a 30° F oxidizer temperature rise at activation is unknown. Possible explanations considered were: compression of gas in the line, rapid compression of the oxidizer, heat from the cartridge valve firing, the temperature sensor reacting as a strain gage, reactions within the system, and instrumentation error. However, from a detailed review of data and considerations of the hardware, it has been concluded that the cause is not likely attributable to any of these possibilities.

The temperatures measured (105° F maximum) were within qualification limits (110° F maximum), but it is quite possible that the actual heat source of this temperature was considerably hotter. A review of ground tests and previous flight experience does not show comparable results. The maximum tank temperature increase experienced previously was on the order of 7° F. By similarity of the systems involved, this information eliminated explanations of a temperature rise due to compression of gas or oxidizer in the manifold or tank, and a temperature rise resulting from the cartridge valve firing. The temperature changes should be accurate within 1.2° F for skin temperature parameters and 0.68° F on the manifold skin tank outlet temperatures.

Possible causes for the low thrust described in section 5.1.8.1.2 are categorized as: frozen or slushy propellants, TCA malfunctions, and propellant contamination. Thermal considerations encompass propellant line or propellant valve freeze-up, or both, either as a result of heater malfunction or improper operation of the heaters. The heaters normally provide a small heat output with continuous operation. Because of the problem with electrical power availability, the heaters were cycled on and off in accordance with temperatures of the injector on the aft TCA 10. This type of operation may have allowed other valves to cool to such an extent that the heaters, when turned on, were incapable of warming the valves enough to insure proper operation. Because TCA 7 failed, the crew observed "sluggish" system performance. The sluggish operation is described as a very slow response and apparent low thrust. This indication caused the crew to turn the heaters on immediately. Proper operation of all TCA's except TCA 7 resulted. This indicates that a large part of the attitude TCA's were approaching the freezing point. Unfortunately, there was no TCA propellant valve instrumentation which could confirm or refute this supposition.

The heater circuit breakers were test cycled to provide comparisons of current changes with those previously measured. This test was performed at 93 hours 49 minutes g.e.t. when the injector temperature of TCA 10 was not following previous temperature trends. Because the heating circuit consists of series-parallel line heaters in parallel with parallel valve heaters, some heater elements could fail, causing associated TCA's to become inoperative. The data are discussed in section 5.1.8.1.2.

Because TCA's 1 and 8 are located in the same area on the spacecraft and TCA's 6 and 7 are also close to each other, a line temperature problem probably would have been reflected in the operation of TCA's 1 and 6. No operational problem on TCA 1 has been found. Evaluation of TCA command data showed a cycle usage ratio of 2.5:1 on TCA 7 as compared with

TCA 6 (320 pulses against 130) and a ratio of 4.0:1 on TCA 7 as compared with TCA 1 (320 pulses against 80). Hence, from a usage standpoint, TCA 7 should have been warmer than either TCA 1 or TCA 6.

Investigations of system testing at the contractor's facility and at Cape Kennedy revealed that problems with TCA's 7 and 8 were encountered. During system testing at the contractor's facility, the TCA 7 oxidizer valve would not open during the initial valve simultaneity tests. The valve opened on the fourth cycle at nominal voltage and appeared normal on all subsequent tests. A minor problem with the TCA 8 fuel valve was encountered during final systems testing at Cape Kennedy 11 days prior to launch. An opening time of 8.8 milliseconds was measured on the first simultaneity test, and the maximum allowable by specification is 6.5 milliseconds. Four subsequent test reruns all gave opening times within specification limits. This type of malfunction of a self-correcting nature has also been experienced at the contractor's facility, but never has there been an occurrence in an extensive ground-test program where a valve of this design failed to operate on a repeated command.

Problems of particulate contamination within the engine system have not been encountered in any of the system gas-flow tests at the spacecraft contractor's facility or at Cape Kennedy. Contamination from residue is not believed attributable to spacecraft propellant because assays of the loaded propellants were within specification limits as shown by table 5.1.8-IV. The possibility that the flight problem was due to some flow decay phenomenon as experienced during qualification tests by the vendor is considered extremely remote because those failures were experienced after considerably longer flow times as compared with the relatively few, short pulses used in flight.

Six special flight tests were conducted in an attempt to solve the problem. Two tests were performed after the heaters had been turned on, the second being performed 6.5 hours after heater reactivation. In another test, the voltage drop was measured when TCA's 7 and 8 were individually operated. Results of this check showed electrical circuit continuity. Three other tests were associated with an attempt to heat the valves on TCA 7 and 8 by applying a voltage to the TCA solenoid coil for 10 minutes. Ground tests have shown that 4 to 6 minutes are required to thaw a TCA which has a temperature of 0° F at test initiation. The crew did not observe any significant thrust during the subsequent TCA tests or any of the following checks which were performed to insure that the propellant feed lines had been thoroughly bled. Evaluation of other TCA thrust data, when available, will establish whether or not the system was running out of propellant at that time.

5.1.8.2 Reentry control system.-

5.1.8.2.1 Preflight: Propellant servicing of the reentry control system (RCS) was completed 21 days prior to lift-off. Fuel loadings of the A-ring and B-ring were 15.88 and 15.86 pounds, respectively. Both rings were loaded with 20.2 pounds of oxidizer. The respective A-ring and B-ring nitrogen source pressurant tanks were pressurized to 3075 psia at 79.9° F and 3080 psia at 80.0° F, 44 days before launch. Planned loads are compared with actual loads in table 5.1.8-I.

5.1.8.2.2 Flight performance: The RCS heater warning light first indicated that some components of the system had cooled to 44° ± 4° F approximately 24 hours after lift-off. The heaters were subsequently turned on to automatic control and left there for the remainder of the flight. From data recorded, no cold temperature problem was experienced in the RCS. Following heater actuation, the A-ring source temperature remained within the range of 61° to 71° F; the B-ring source temperature remained in the range of 57° to 74° F; and the A-ring oxidizer feed temperature remained within the range of 49° to 74° F until system activation at 188:28:10 g.e.t. From activation until landing, these temperatures ranged from 30° to 67° F, 40° to 69° F, and 64° to 103° F, respectively, which was within the system design limits.

After system activation, regulated pressures of the A-ring and B-ring stabilized at 294 and 298 psia, respectively. Throughout reentry and until the propellant motor valves were closed at approximately 65 000 feet, the regulated pressure of the A-ring remained within the range of 292 to 298 psia, and that of the B-ring from 298 to 300 psia.

Source pressure leakage over the 52 days from servicing to system activation was negligible. The A-ring source pressure just prior to system activation was 3010 psia at 68° F. This compares closely with the serviced pressure of 3014 psia corrected to the flight temperature at activation. Similar values for the B-ring were 3011 psia at 69° F and 3017 psia corrected to the flight temperature at activation.

Spacecraft angular accelerations occurring shortly before retrofire were determined from the changes in pitch and yaw rates when the thrust chamber assemblies from both rings were fired. Correlation between these values and preflight predicted quantities indicated that all thrusters were operating within expected limits. Examination of the spacecraft rates in all three axes during additional periods of TCA activity revealed no unusual rate disturbance. The capability of the system to hold attitude after drogue parachute deployment is discussed in section 5.1.11.

This mission marks the first attempt at single ring reentry. After completion of the A-ring and B-ring checkout, the B-ring was turned off until retrofire. From 190:29:50 g.e.t. when the retro section of the adapter was jettisoned, single-ring reentry control (A-ring) was utilized until drogue parachute deployment after which the B-ring was reactivated, and the reentry was completed with dual-ring operation.

Figure 5.1.8-4 presents the RCS propellant consumption during reentry. Propellant usage prior to this time from the B-ring consisted of less than 1 pound for checkout and from the A-ring approximately 2.5 pounds for system operation checks, platform alinement, and attitude control. The relatively high propellant usage rates shortly before and after drogue parachute deployment are attributed to the tight control inherent in the rate command mode, drogue parachute deployment at a higher altitude than planned (69 000 ft instead of 50 000 ft), and the normally higher usage occurring from manually damping disturbances while in the direct control mode. The overall mission mixture ratio of the B-ring, based on total serviced propellant quantities and the quantities removed from the system during deservicing, was calculated at 1.36. The A-ring mixture ratio cannot be determined precisely because the exact time of fuel depletion is unknown.

5.1.8.2.3 Postflight deservicing and testing: Fuel and oxidizer quantities removed from the A-ring during deservicing at Mayport, Florida, were 0.00 and 0.08 pound, respectively. The propellant expelled from the B-ring consisted of 4.63 pounds of fuel and 4.90 pounds of oxidizer. A chemical analysis was performed on propellant samples taken from the B-ring. The results of this analysis (table 5.1.8-IV), however, are inconclusive because of the unknown cleanliness condition of the tanks used in deservicing the system.

No leakage from the propellant valves could be detected by portable propellant vapor detectors prior to deservicing, nor was any liquid leakage noted after the propellants were replaced with flush fluids and a 50-psi system pad pressure applied. The condition of the system appeared normal with the exception of a green substance observed around the nozzle of TCA 3B. This substance was chemically analyzed at Cape Kennedy and found to be completely foreign to the thruster materials and is believed to have become attached after landing.

5.1.8.3 <u>Retrograde rocket system</u>.- In approximately 130 hours, the temperature of retrorocket motor no. 4 decreased to 35° F from the lift-off value of 74° F. A maximum excursion of 90° F was observed during the remainder of the mission, and that occurred at the time of motor no. 1 ignition. Performance of the system was nominal as shown by table 5.1.8-V.

TABLE 5.1.8-I.- OAMS AND RCS SERVICING DATA

System	Pressurant at reference temperature of 70° F, psia		Propellant at reference temperature of 70° F, lb		Propellant quantity indicator, percent
	Preactivation	Postactivation	Oxidizer	Fuel	
OAMS					
Planned	2915^{+30}_{-0}	2650 ±10	$220.9^{+2.2}_{-0}$	$164.0^{+0}_{-.8}$	91
Actual	2911 ±20	2510 ±20	220.5 ±1	165.5 ±1	87
RCS					
A-ring					
Planned	3030^{+30}_{-0}	–	$20.2^{+.2}_{-0}$	$15.8^{+.16}_{-0}$	–
Actual	3030 ±20	2588 ±20	20.17 ±.1	15.99 ±.08	–
B-ring					
Planned	3030^{+30}_{-0}	–	$20.2^{+.2}_{-0}$	$15.8^{+.16}_{-0}$	–
Actual	3021 ±20	2569 ±20	20.17 ±.1	16.03 ±.08	–

TABLE 5.1.8-II.- OAMS MANEUVER TRANSLATION PERFORMANCE SUMMARY

Maneuver	Time, g.e.t.	Maneuver time, sec		Velocity change, ft/sec	
		Planned	Actual	Planned	Actual
Separation	00:05:57	6	7.9	5	7.6
Perigee adjust	00:56:00	13	12.8	10	9.7
Apogee adjust	50:49:57	28	26.9	21.1	21.1
Coelliptical	51:34:31	20	Data unavailable	15.2	15.7
Out of plane	52:06:26	19	Data unavailable	14.6	14.7
Reverse coelliptical	53:04:04	22	Data unavailable	17.3	17.3

TABLE 5.1.8-III.- OAMS OXIDIZER LINE AND PROPELLANT VALVE HEATER CIRCUIT BREAKER POSITIONING

Ground elapsed time, hr:min:sec	Revolution	Circuit breaker position	ΔT, hr:min	Heater
00:00:00	1	Closed		On
			04:27	
04:26:45	3	Open		Off
			09:12	
13:39:05	9	Closed		On
			10:07	
23:46:24	15	Open		Off
			14:30	
38:15:57	24	Closed		On
			00:48	
39:03:56	25	Open		Off
			46:49	
85:53:13	54	Closed		On
			07:56	
93:49:09	59	Open		Off
			<00:01	
93:49:19	59	Closed		On
			04:31	
98:19:28	62	Open		Off
			20:08	
118:27:38	75	Closed		On
			10:19	
128:46:51	81	Open		Off
			00:01	
128:48:16	81	Closed		On

TABLE 5.1.8-IV.- RCS PROPELLANT ANALYSIS

	Specification	Preservice sample	Postservice sample	RCS postflight removed from ring B RCS
Fuel				
Purity, percent	98.0 min	98.3	98.4	96.6
Water equivalent, percent	2.0 max	1.7	1.6	3.4
Density at 77° F, g/ml	0.872 ± 0.004	0.870	0.870	0.868
Transmittancy, percent	90.0 min	96.0	95.0	88.0
Oxidizer				
Purity, percent	99.5 min	99.9	99.9	94.05
Water equivalent, percent	0.1 max	0.03	0.001	4.94
Chloride as nitrosyl chloride, percent	0.08 max	0.04	0.04	0.025
Non-volatile ash, percent	0.01 max	None	None	None
Total filterable solids, mg/100 ml	1.0 max	None	None	None
Spectrograph emission				Minor — silicon, iron, AL, and CR

TABLE 5.1.8-V.- RETROGRADE ROCKET SYSTEM PERFORMANCE

Motor number	1	2	3	4
Firing sequence	first	third	second	fourth
Ignition time, g.e.t. .	190:27:43.3	190:27:54.0	190:27:48.8	190:28:00.1
Total burn time[a], sec .	6.4	6.0	6.1	6.2

Parameter	Predicted	Actual	Deviation, percent
Total impulse, $\frac{\text{lb-sec}}{\text{lb}}$	56 725	56 709	-0.03
ΔV, ft/sec	325.0	324.5	-0.15
Prefire weight, lb	5 542	5 549	+0.13

[a]Total burn time is defined as the time interval from motor ignition to the end of detectable thrust.

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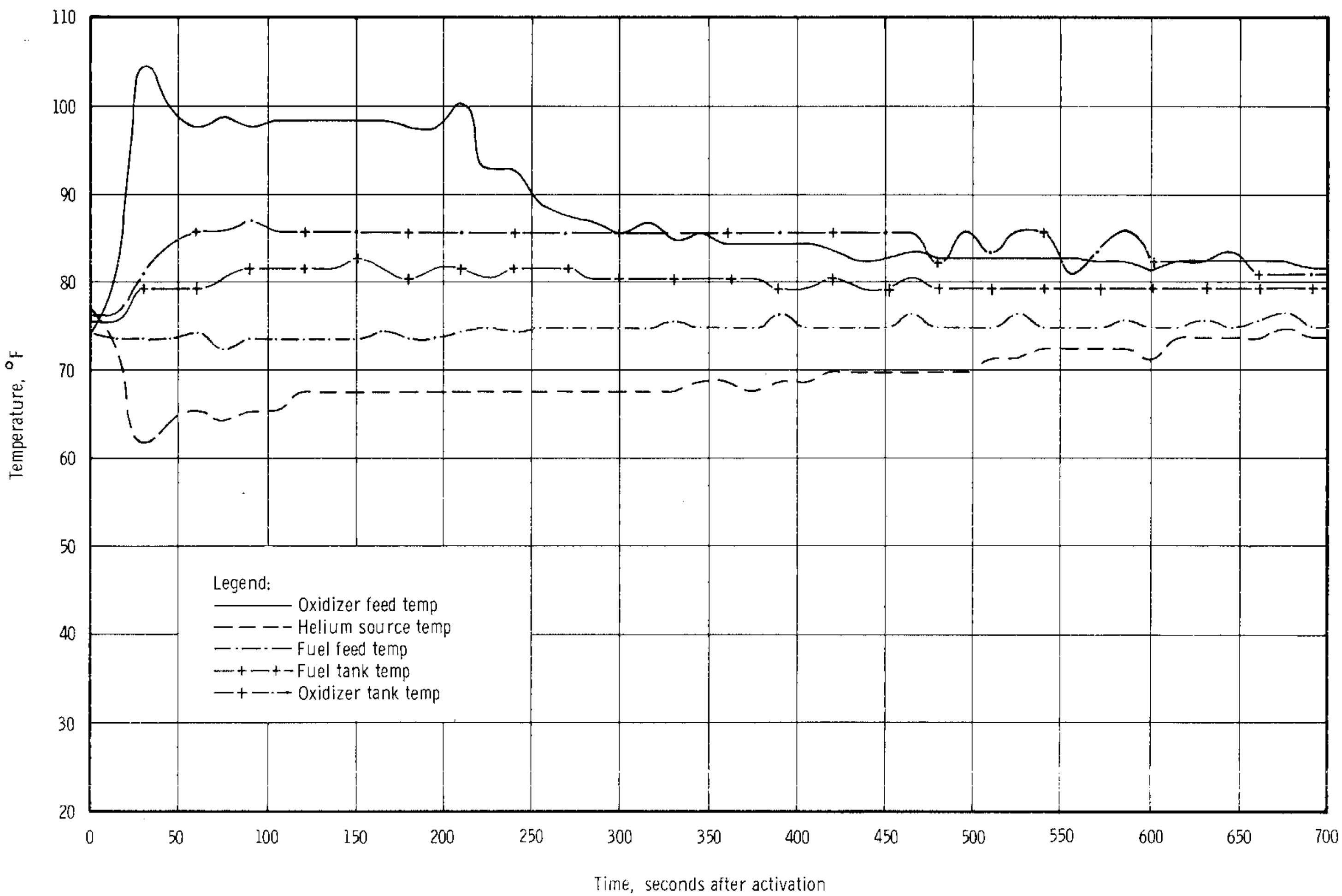

Figure 5.1.8-1. - Post activation OAMS temperatures.

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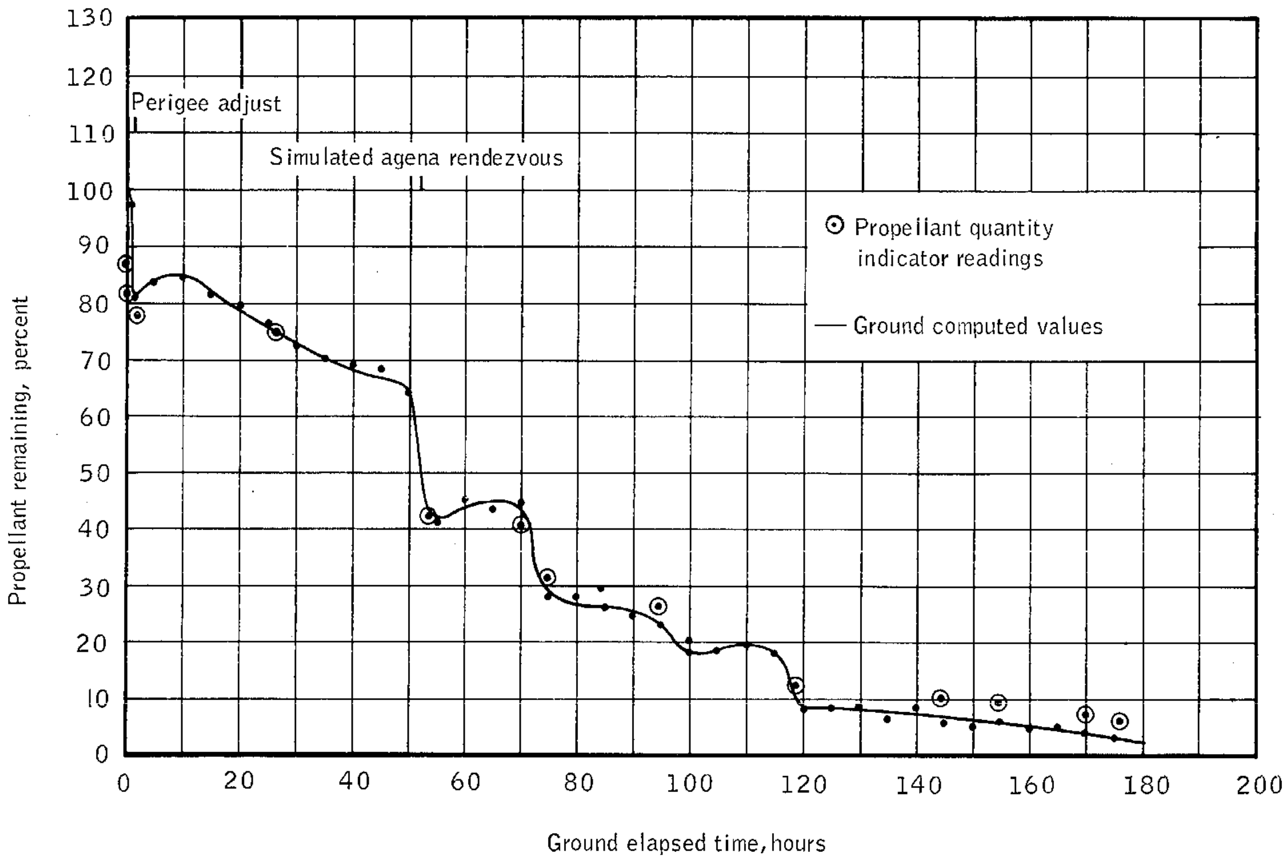

Figure 5.1.8-2. - OAMS propellant consumption.

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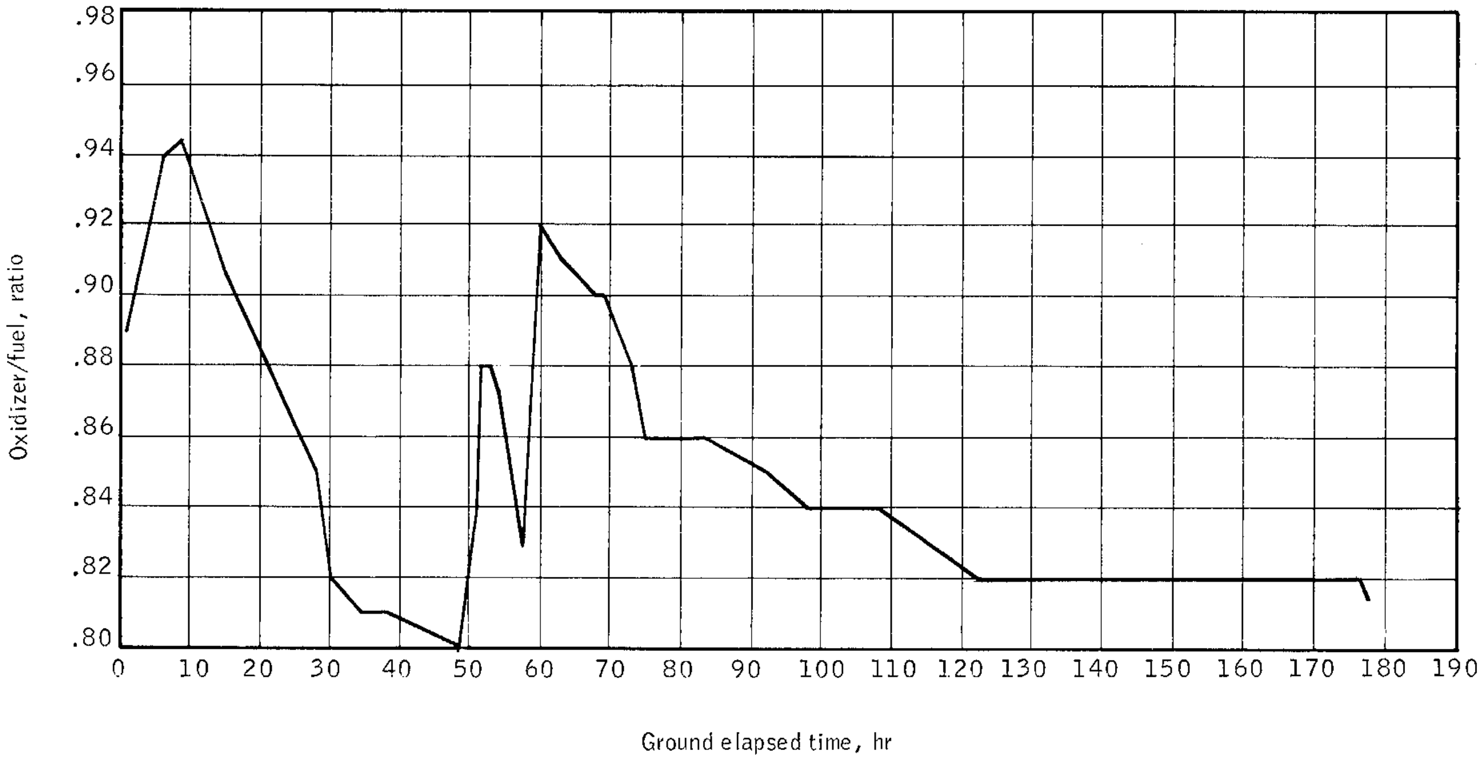

Figure 5.1.8-3. - Calculated accumulative expended oxidizer to fuel ratio.

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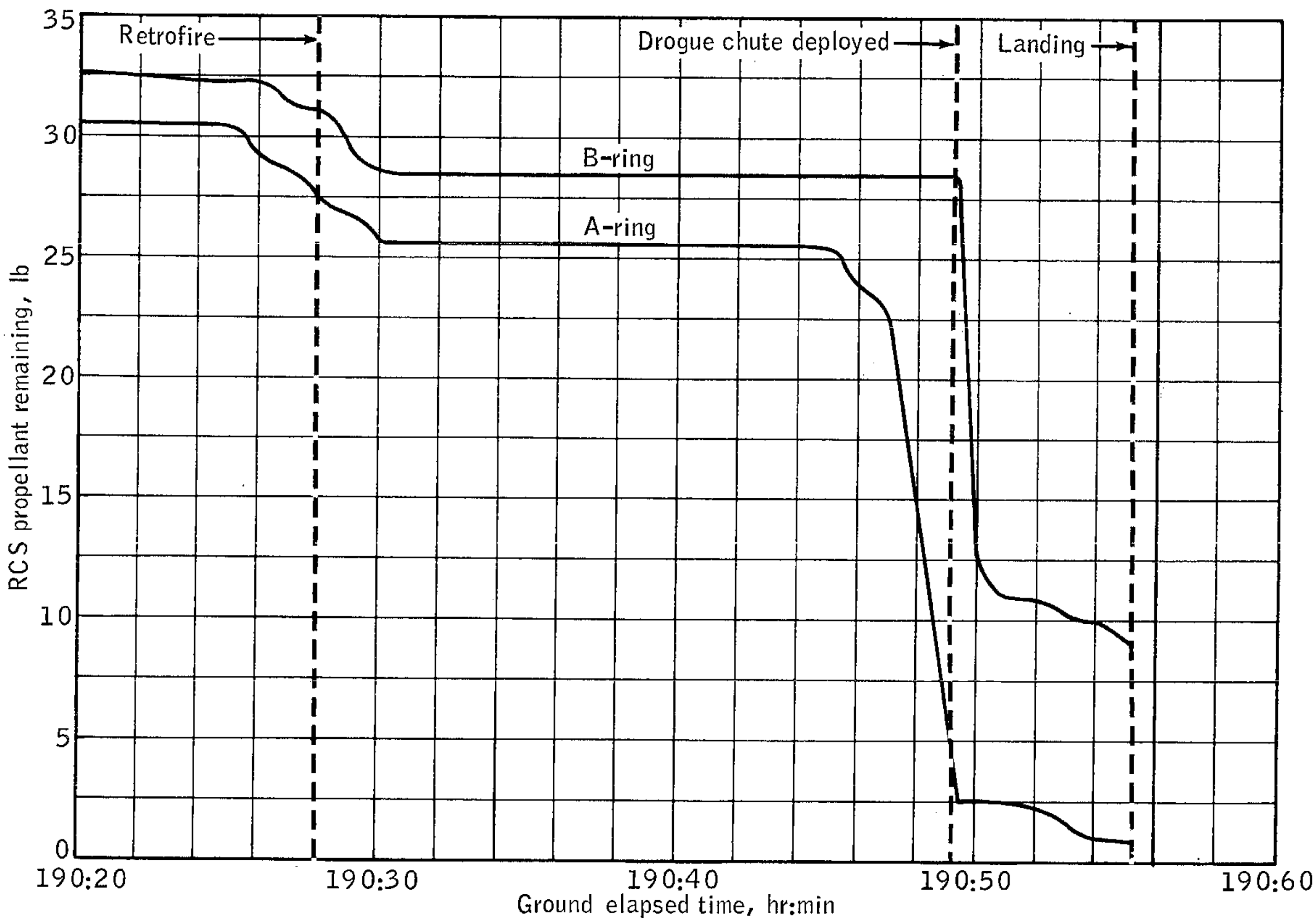

Figure 5.1.8-4. - RCS propellant consumption.

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5.1.9 Pyrotechnic System

Based on a successful mission and all available related data, it may be deduced that the pyrotechnic system performed all required functions during the Gemini V mission in a satisfactory manner.

A postflight evaluation of the ejection seat ballute deploy-release and the drogue-mortar aneroid mechanisms was conducted. All four of these devices functioned within design limits. The test results are listed in the following table:

Nomenclature	Design firing altitude, ft	Firing altitude, ft		
		Test 1	Test 2	Test 3
Right-hand drogue aneroid	5700 ± 600	5700	5700	5700
Left-hand drogue aneroid	5700 ± 600	5350	5350	5350
Right-hand ballute aneroid	7500 ± 700	7200	7250	7200
Left-hand ballute aneroid	7500 ± 700	7375	6950	7400

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5.1.10 Crew Station Furnishings and Equipment

5.1.10.1 Crew stations design and layout.- The basic design of the crew station was satisfactory for the Gemini V mission. A few anomalies were noted, and these are discussed in the following paragraphs. On this mission, one important fact was learned about the basic crew station design, in that the pilot reported seeing the ground from the cabin prior to lift-off by holding the detachable mirror near the bottom of the window. The crew also reported seeing the earth after reentry while in two-point suspension on the parachute. Each pilot could see the water out of his window or the opposite window prior to landing by leaning forward.

5.1.10.1.1 Equipment stowage: All loose equipment except four packages of food was stowed in the normal stowage containers for launch. These food packages were carried in the helmet stowage bags in the footwells. The stowage plan for the mission was to remove all the food and equipment from the right aft stowage box and stow these items in fabric containers over the seat headrests. The right aft box was then to be used for waste. Prior to reentry all the equipment removed from the aft boxes and the food remaining was to be restowed in the aft boxes. This plan was carried out successfully by the crew except that a certain amount of time and effort had to be devoted to the daily task of housekeeping.

The four dry-waste stowage bags mounted on the outboard walls of the footwells were damaged during the flight. The fabric tore in several locations. The elastic top of one bag failed, and the fabric on one bag wore through from being rubbed by the pilot's leg.

The command pilot reported that the stowage location of the optical sight under the left instrument panel was difficult to reach and, because the stowage mount also could not be seen, it was difficult to stow the sight.

5.1.10.1.2 Long-duration habitability: The habitability of the crew station was satisfactory for the 8-day mission. The principal limitation was the length of the crew station which prevented either pilot from straightening his body to full length. Some discomfort apparently resulted from the cramped quarters; however, frequent use of the exerciser helped alleviate this problem.

During the periods when the spacecraft was powered down, the noise level in the crew station was very low. Accordingly, the slightest noise was noticeable to the crew and disturbed them during sleep periods.

The flight crew reported being chilled during the early part of the flight when the spacecraft was powered down in drifting flight. With a suit inlet temperature of 46° to 48° F and a cabin temperature of 70° F the crew shivered, and frost formed on the inside of the window from their breath. When the spacecraft attitude was controlled, the cabin suit temperatures rose 5° to 10° F, and the crew was comfortable.

Light polarized window filters, which could be rotated relative to each other to block light through the windows completely or adjusted as necessary, were carried for the first time on this flight. These filters were effective for shutting out the sunlight during sleep periods.

No crew station design feature or characteristic limited the duration of the mission, and the results of the mission indicate that the crew station is satisfactory for flights of even longer duration.

5.1.10.1.3 Crew station furnishings: The ejection seats were used only for support and restraint on this mission. No discomfort was reported for the prelaunch, orbit, or reentry phases of the mission. The drogue mortar safety pins and receptacles, which had been redesigned prior to the mission, were found to be very satisfactory. The crew was able to install the pins without difficulty. The command pilot reported that the safety pin lanyard for the drogue-mortar automatic firing mechanism on his seat occasionally snagged on his right shoulder and pulled the pin out. The lanyard had been lengthened since the GT-3 mission to improve accessability.

5.1.10.1.4 Cabin lighting: The cabin lighting was adequate for the mission, although the crew reported several lighting deficiencies. The lighting on the center instrument panel was poor for darkside operation. In order to illuminate this panel adequately, it was necessary to turn the center cabin light so bright that it interfered with visibility outside of the spacecraft. If the center light was turned low enough to be compatible with outside visibility, the center panel instruments and markings were difficult to read.

The crew reported that all three cabin light assemblies tended to overheat when they were operating for 30 to 40 minutes or more. The crew smelled the odor of scorched paint from the lights. The condition is being investigated, and corrective action will be taken, if necessary.

The crew accidentally broke the right-hand utility light while removing it from the stowage bracket. Postflight inspection revealed a weak point where the lens was attached to the body of the light making the assembly susceptible to handling damage. This utility light was made from a new design which incorporated a parabolic reflector to concentrate the illumination. The left-hand light was used throughout the flight and found to be satisfactory.

The COMPUTE light on the center pedestal was a frequent annoyance during darkside operation because it could not be dimmed or extinguished. This light illuminated frequently during programed maneuvers and when these maneuvers were required on the darkside, the brightness interfered with the command pilot's ability to see the stars that will be necessary for navigation and tracking during the optical rendezvous procedures.

The previous Gemini flight crews reported difficulty in seeing the inside of the cockpit after looking at either the sunlit earth or the sun. This severe contrast in illumination was reduced to a satisfactory level by use of the polarized window filters which the crew kept in the windows at least one-half of the time. The knob on the rotating disk on the left-hand window filter broke off in flight. The knob had been bonded to the plastic and the joint came apart. Numerous scratches were noticed on the rotating disk and the corresponding stationary part of the right-hand window filter resulting in slightly impaired visibility. Except for these minor deficiencies, the crew described the filters as necessary and very useful.

5.1.10.2 <u>Controls and displays</u>.-

5.1.10.2.1 Controls: The basic attitude and maneuver controls were satisfactory. The other controls were satisfactory except as follows: the oxygen purge control switches for the fuel cell were spring-loaded momentary switches which the pilot had to hold "on" for approximately 2 minutes every few hours. The spring-loaded feature was unsuitable for the frequency and duration of this control function. The inconvenience was aggravated by the high spring forces on the switches and the lack of gravity on the body to assist in overcoming the switch forces.

The command pilot was unable to engage the fabric retention loop on the left-hatch lock-release lever on several attempts before opening the hatch for recovery. After the hatch was opened, the lock-release lever worked freely. The condition of this control lever and its associated cable and linkage is being investigated for possible discrepancies.

5.1.10.2.2 Displays: The radar indicator and the fuel-cell power system monitor indicator were used for the first time on a manned flight, and both were satisfactory.

One of the redundant pointers for the launch vehicle stage II fuel-tank pressure indicator failed intermittently during the ascent phase of the flight. This pointer, driven by the launch vehicle instrument power supply, moved to full scale before the end of stage I operation and indicated intermittently during stage II operation. The pilot correctly referred to the redundant pointer for stage II fuel-tank pressure, and there was no inflight action taken as a result of the intermittent indication. A review of the flight data indicated an open circuit failure in the instrument or associated wiring. A complete failure analysis was conducted on the instrument, the wiring in the recovered reentry section was checked, and the fault was not revealed. GLV telemetry indicated that the proper signal was sent to the launch-vehicle spacecraft interface. This leaves only the interface electrical connectors, which were redundant, and the spacecraft adapter wiring as the probable sources of the trouble.

The cabin temperature gage failed $56\frac{1}{2}$ hours after launch. The indicator needle dropped to the minimum scale reading of 40° F at this time. Later in the flight, the gage indicated intermittently. Postflight analysis showed that a bent pin in the temperature sensor connector caused the intermittent failure. The crew used the hand-held humidity sensor and ground readings of cabin temperature for reference for the remainder of the flight.

The crew reported that the digital command system (DCS) indicator light did not illuminate after the DCS update of the computer at Carnarvon during the last orbit. Postflight tests conducted on the spacecraft have revealed no faults in either the light or in the reentry module wiring. A part of the circuit was in the adapter and could not be checked. It was found that the light is not visible in normal cabin light (dayside) when the dimming iris is in the fully closed position. This was the position in which it was found at the start of the test. It was also found that the mechanism moves very freely and could conceivably have vibrated to that position accounting for the report that the light did not illuminate.

Three time displays were provided to the crew for Greenwich mean time (G.m.t.) and elapsed time. The G.m.t. clock on the right instrument panel was accurate within approximately 25 seconds per day. The 24-hour battery-driven clock on the left instrument panel was in error by less than 1 second per day. The event timer was powered down to conserve electrical power, and the accuracy of this device was not determined in flight. The lack of a mission-elapsed-time indicator in the crew station dictated that G.m.t. be used as the principal time reference after midnight on the first day. Postflight analysis of the flight data was hampered by the simultaneous use of G.m.t. and elapsed time. As in Gemini IV, the flight crew reported that mission elapsed time would have been a better time reference if an onboard indicator had been provided.

The window markings intended for use as pitch or roll attitude indications when using the horizon for visual attitude reference were confusing. The markings were not labeled to identify the corresponding bank angles, and differentiation between the numerous markings was difficult. The crew reported that the scribe lines across the window were distracting throughout the flight.

5.1.10.3 Space suits and accessories.-

5.1.10.3.1 Basic space suits: The G4C space suits operated satisfactorily throughout the flight. The helmets and gloves were removed at approximately 7 hours 40 minutes g.e.t., and they were left off until just before retrofire. The crew used their wrist dams and neck dams to maintain normal ventilation flow within the suits. It was discovered when the command pilot was not wearing the wrist dams that he received substantially more of the total ventilation flow. As a result, the command pilot felt colder than the pilot. Subsequently, the command pilot donned his wrist dams, and the ventilation flow balance was restored. The crew removed their ventilation inlet hoses occasionally in order to warm up.

The suit comfort was acceptable. The crew reported good ventilation, including the ventilation to their feet. In the latter half of the mission, the crew became increasingly aware of pressure points and discomfort. The principal pressure points were around the inlet, exhaust, and communication fittings.

5.1.10.3.2 Space suit accessories: The wrist dams and neck dams functioned in a satisfactory manner, except that the neck dams were susceptible to tearing. Each crew member carried two neck dams and the pilot damaged both of his while donning them.

The dual port manifold block which contained the blood-pressure cuff port and the cardiovascular cuff port on the pilot's suit came loose at approximately 29 hours g.e.t. The result was leakage of about 10 percent of the oxygen supply for the cardiovascular cuffs and a potential leak in the blood-pressure line. The pilot recognized the difficulty after a short period and repaired the leak by tightening the manifold-block retaining screw. Inspection of the manifold block showed that there were no positive means for keeping the retaining screw tight.

The isolation cap, which contained ear cups for noise suppression and an eye shade for light shielding, was unsatisfactory as a sleeping aid. The ear cups were too small, and the cap was uncomfortable. The crew used the light-polarized window filters for light control instead.

Two "O-ring" seals on the blood-pressure inflator fitting failed early in the mission. Postflight inspection of the two space suits showed that the blood-pressure port of each suit had a sharp radius at the outer edge. This sharp radius probably damaged the seals when the blood-pressure inflator was inserted in the port. The seals were replaced with spares from the space suit repair kit. No further failures of these seals occurred for the remainder of the mission.

The cardiovascular cuffs worn within the pilot's suit became a source of discomfort in the last half of the mission. This discomfort was caused by the close fit and the lack of ventilation under the cuffs. After the cardiovascular-cuff oxygen supply was depleted, the pilot removed the cuffs from inside the suit by cutting them off with the scissors. Access to the cuffs on the legs was gained by partially removing the suit torso.

5.1.10.4 Flight crew operational equipment.-

5.1.10.4.1 Still camera (70-mm): The 70-mm still camera, with an 80-mm focal length lens and four 70-mm film magazines, was used successfully to take approximately 235 general purpose and experiment photographs. The quality of the pictures was excellent, and they included some subject material not obtained on previous flights.

5.1.10.4.2 Sequence camera (16-mm): The 16-mm camera mounted on the right-hand window functioned normally throughout the flight. Two of the four film magazines were exposed completely, and the remaining two were partially exposed. Picture quality was less than nominal for some of the picture sequences. Postflight analysis of camera and magazines indicated normal operation.

5.1.10.4.3 Photo event indicator: The photo event indicator was used only with the 35-mm camera for this flight. A problem was encountered in flight with the photo event indicator-film transport adapter system. Postflight analysis revealed that the film transport adapter had not been properly set for the photo event indicator.

The photo event indicator was removed from the 70-mm still camera prior to flight. The flight crew maintained a hand-written log of photographs taken which resulted in a complete postflight photographic identification.

5.1.10.4.4 Optical sight: The optical sight was satisfactory for tracking of ground, orbital, and celestial targets of varying light intensity except when unwanted reflections from the collimating mirror interfered with the view through the sight. When the sunlight struck the mirror, it reflected into the command pilot's eyes and prevented him from seeing any other object. Similarly, during darkside operations, there were reflections from the mirror caused by the center cabin light. These deficiencies indicated that the mirror must be shielded in order for the sight to be usable with the sun in the left window or with the center cabin light on.

At 47 hours 44 minutes g.e.t., the crew reported that the optical sight had burned out. Subsequently, the crew was given instructions for disassembling the sight to use the utility light to illuminate the reticle pattern. The crew had essentially completed this operation when the utility electrical cord was found to have a broken wire which was the actual cause of the reported sight failure. After reassembling the sight, the command pilot substituted another electrical cord and the sight operated normally. A failure analysis was performed on the failed cord, and it was determined that one of the three wires in the assembly was shorter than the other two. This caused undue strain, and the shorter wire was pulled apart during normal handling. A strain relief will be incorporated in both ends of similar cords for future missions.

A voltage regulator cable adapter was provided for reducing the input voltage to the sight if additional dimming was required. Although this item was not used, postflight inspection revealed a significant discrepancy. The integral connector housings used for connecting the cable adapter at either end were loose and rotated within the adapter body. As a result, it was impossible to disconnect or connect the utility cable without risking internal damage to the adapter.

5.1.10.4.5 Lightweight headsets: Two new design lightweight headsets with molded ear plugs were used throughout most of the flight. In addition, one contractor-furnished headset was carried as a backup item but was not used. The crew reported that the new design headsets were satisfactory for communications and were very comfortable except for minor irritation from the molded earpiece. The pilot wore his headset continuously from revolution 6 to revolution 119, approximately 170 hours, without discomfort or difficulty.

5.1.10.4.6 Flight data books: The flight data books were excellent for this mission. The previous problems of pages tearing out and rings coming loose were corrected, and the crew was satisfied with the manner and content of flight data presentation.

5.1.10.5 Flight crew personal equipment.-

5.1.10.5.1 Food: Rehydratable and bite-size foods similar to those eaten during the Gemini IV mission were provided for Gemini V. A total of 54 man-meals (27 meals per man) was carried on this flight. The crew's food log indicated that each crew member ate 19 total meals.

Each meal consisted of two or three rehydratable items, one or two bite-size items, and usually one rehydratable fruit juice. Meals were eaten at approximately 8-hour intervals throughout the flight with the exception of the first 24-hour period. During this first day, the crew was limited to snack-type eating because of the high level of activity. Most of the items consumed were rehydratable foods and juices. Bite-size pieces were only partially eaten, and, after the third day of flight, none of the bite-size items were eaten.

Four food bags failed during flight, three rehydratable food bags and one juice bag. Each of these failures can be attributed to the application of hand pressure to the filled bags which caused the heat seal to fail. Excessive hand pressure was necessary because the bag feeder ports would not open sufficiently to allow passage of the bag ingredients into the mouth. This discrepancy in the bags was caused by the fabrication process of heat-sealing the feeder port material to the bag material and later evacuating the bag, causing hard creases and weak lines in the bag material.

The amount of dry or nearly dry waste which resulted from the food was larger than anticipated. After the flight, the crew estimated that the waste-to-stowed food ratio was approaching 2 to 1. They attributed this apparent increase in volume to the bulk of the overwraps. The waste volume was increased still more when the crew did not elect to eat all of the food items in an individual meal. As a result of the unpredicted amount of food waste, the crew stowed some waste items behind the seats rather than in the normal stowage containers.

5.1.10.5.2 Drinking water dispenser: The pistol-configured water dispenser was utilized in the rehydration of foods and for inflight and postlanding drinking. This device operated in a satisfactory manner with no leakage or improper operation noted.

5.1.10.5.3 Launch day urine collection device (UCD): The crew did not remove the UCD's until late during the first day of flight. No leakage was noted during the removal process. Each UCD was emptied by attaching it to the urine transport system followed by dumping overboard through the spacecraft overboard urine dump system.

5.1.10.5.4 Urine disposal system: The new urine disposal system, effective for spacecraft 5, functioned throughout the flight without noticeable leakage or other major discrepancies. The flight crew noted that the new roll-on receivers showed considerable deterioration during the flight and required frequent cleaning. Investigation of the latex formula revealed that a chemical reaction probably was taking place. This reaction of urine with the latex attributed to the gradual deterioration of these receivers. Although urine deposits were noted on the spacecraft quick disconnect, no problems were encountered in the overboard dump sequence.

5.1.10.5.5 Defecation device: Four defecations occurred during the flight. The prime problem encountered was stowing the used bags, because of the volume required. Use of these bags was accomplished satisfactorily.

5.1.10.5.6 Personal hygiene items: Wet pads furnished with each food pack and with each defecation device were used by the flight crew for face and hand cleaning as well as for cleaning of the urine system. The large personal hygiene towels were utilized for purposes ranging from instrument glass wipers to being wrapped around the neck in scarf fashion to augment the neck dam.

5.1.10.5.7 Oral hygiene items: The toothbrushes on Gemini V were utilized by the command pilot only. He reported that the toothbrush bristles were too stiff. No dental floss was used, although some was carried onboard. The chewing gum provided in each meal pack was used occasionally.

5.1.10.5.8 Carbon dioxide (CO_2) sensing tapes: One CO_2 detection tape was utilized when a 1-mm partial pressure of CO_2 was noted. The tape did not register on the 4-mm indicator circle. The 2-mm indicator circle had remained covered and therefore showed no color change.

5.1.10.5.9 Humidity sensor: The hand-held humidity sensor was utilized three to four times daily beginning with the second day of flight. Operation of the dry bulb and wet bulb portions of the humidity sensor appeared satisfactory. The wet-bulb wick was rehydrated frequently during flight in order that valid wet-bulb readings could be obtained. The surface temperature indicator, when used, registered relatively high temperatures. These high readings may have resulted from failure to allow sufficient time for the surface temperature probe to reach equilibrium.

5.1.10.5.10 Survival equipment: The individual survival kits were not opened during the postlanding mission phase. The life vests remained on the personal parachute harness and were not activated by the flight crew.

5.1.10.5.11 Water measuring bags: Two 3-ounce water measuring bags were carried aboard the spacecraft to measure water intake but were not used.

5.1.10.5.12 Postflight equipment condition: An inspection of each item of personal equipment subsequent to recovery indicated that all components were exceptionally clean.

5.1.10.6 Bioinstrumentation equipment.-

5.1.10.6.1 Bioinstrumentation sensors and signal conditioners: No difficulties were encountered with the sensors or signal conditioners. All equipment was recovered in satisfactory working condition.

5.1.10.6.2 Blood-pressure manual inflator: In addition to the "O-ring" seal damage reported in paragraph 5.1.10.3, there was a significant failure of the inflator assembly in the postlanding phase of the mission. The inflator did not relieve the pressure properly, and the pilot was unable to obtain satisfactory blood-pressure readings after landing. The cuff could be pumped up normally, but the inflator had to be removed from the suit port to release the pressure. A failure analysis is being conducted on the inflator and corrective action will be taken if necessary.

5.1.10.6.3 Data retrieval: High quality bioinstrumentation data were retrieved for the complete mission except for the blood-pressure problems described previously. The analysis of the bioinstrumentation data is included in section 7.2.

5.1.11 Landing System

The parachute landing system accomplished the basic function of providing a safe water landing for the Gemini V crew. All system events occurred as commanded within established tolerances of the commands. Figure 5.1.11-1 illustrates the major sequences with respect to ground elapsed time and pressure altitude as they occurred.

Following reentry on this flight, the drogue parachute was inadvertently deployed at an altitude of 69 000 feet instead of the normal altitude of 50 000 feet. At this time the spacecraft was at a Mach number of approximately 1.20. This supersonic deployment resulted in the following:

(a) A higher-than-normal snatch load - Snatch load is defined as the short duration load following first line stretch. This load rises to a peak in less than 50 milliseconds and immediately falls as the parachute "bounces." However, interpolation of the data indicates that the load was at least 3000 pounds. During the development and qualification test programs for this parachute the snatch load was consistently between 1000 and 1200 pounds.

(b) A failure of the drogue parachute to inflate for approximately 10 seconds after line stretch - One of the significant factors affecting the ability of a parachute canopy to inflate in a supersonic flow is the porosity. Since the Gemini drogue parachute was not designed for supersonic conditions, it has a very low basic porosity (17 percent as opposed to 26 percent for the Mercury parachute). In addition, the Gemini parachute stays reefed for 16 seconds and the geometric porosity is reduced further. Accordingly, during the Gemini V mission, the drogue parachute squidded instead of inflating until the velocity became subsonic. Inflation occurred at an altitude of approximately 55 400 feet at a Mach number of 0.95. The parachute during this 10-second period of time was ineffective as a stabilizing or drag-producing device.

(c) Severe canopy pulsation and ribbon flutter - Although this phenomenon was not measured or reported by the crew, conical ribbon canopies usually exhibit this characteristic beginning at a Mach number of about 1.15. Canopy pulsations were encountered at much lower Mach numbers of 0.7 to 0.8 during the development test program of this parachute. This problem was successfully overcome for normal Gemini conditions by the use of 69-percent permanent skirt reefing. Above a Mach number of 0.8, however, canopy pulsations are still to be expected.

(d) Reynolds number effect on drag coefficient - At the time of drogue parachute deployment during the Gemini V mission, the Reynolds number was approximately 4×10^6. A ribbon-type canopy operating in the transonic and low supersonic speed regimes is significantly affected with respect to the drag coefficient at this Reynolds number. (The drag coefficient is approximately one-half its subsonic value.)

In view of the above factors, the observed performance of the drogue parachute closely followed the theory and test results associated with this parachute. The deployment of the drogue parachute at a supersonic velocity had no catastrophic effect; nor did it improve the overall performance. It should not be concluded that it is safe to deploy the drogue parachute at supersonic velocities, particularly in view of the lack of sufficient test data related to paragraphs (a) and (c) previously discussed. It can be concluded that the drogue is ineffective as a stabilization device at supersonic speeds in view of paragraphs (b) and (d). Further, it can be restated confidently that the established altitude of 50 000 feet for deployment of the Gemini drogue parachute is the optimum value for nominal reentries from orbit.

Although the crew reported that oscillations during the descent on the drogue parachute were less than ±5°, the data indicate that these oscillations were greater in magnitude. The amplitudes during reentry control system (RCS) activity were approximately ±10°. Within 20 seconds following RCS shutdown the oscillations built up to ±20° and remained near this value until drogue parachute release. This performance was compared with the descent of Gemini IV on the disreefed drogue parachute. Gemini IV did not exceed ±10° on the disreefed drogue parachute, and for at least half of the time had essentially no oscillation.

Even though the Gemini V landing system held the spacecraft within the specification stability band of ±23°, performance on the drogue should have approached that exhibited by Gemini IV. The most probable cause for the relatively degraded performance of Gemini V is a failed bridle cable or attachment, with descent being effected on only two legs of the bridle rather than the three. Two factors appear to indicate that this did occur:

(a) The spacecraft oscillations on the drogue were predominantly in the spacecraft pitch plane. Normally, oscillations will "walk around" the bridle; that is, they will move progressively around the planes described by each pair of bridle cables.

(b) The crew observed that there always appeared to be slack in the cable on the top side of the rendezvous and recovery (R and R) section as viewed by the crew. This particular cable is the one to which the static line to the pilot parachute is attached. If this cable had failed near the attachment point to the R and R section or if the attachment fitting had failed, the leg would still have been attached to to the R and R section through the static line to the pilot parachute mortar. Under this condition, this leg could not have picked up load, and would have appeared slack to the crew.

If this cable or its attachment failed, it probably occurred as a result of the high snatch load immediately following drogue parachute deployment. As previously noted, the Gemini V snatch load exceeded 3000 pounds. The cable attachment fitting is capable of taking an ultimate load of 6050 pounds. It would only have required a dynamic shock factor of approximately 2 to exceed the ultimate load capability of the fitting. The cable is a $\frac{9}{32}$-inch diameter cable with a minimum breaking strength of 7800 pounds. A dynamic shock factor of about 2.5 would have been required to exceed this. Either of these factors could have been attained at a loading rate of 3.6×10^6 pounds per minute. However, it is more likely that the attachment fitting failed, since its ultimate load capability is significantly lower than the cable breaking strength.

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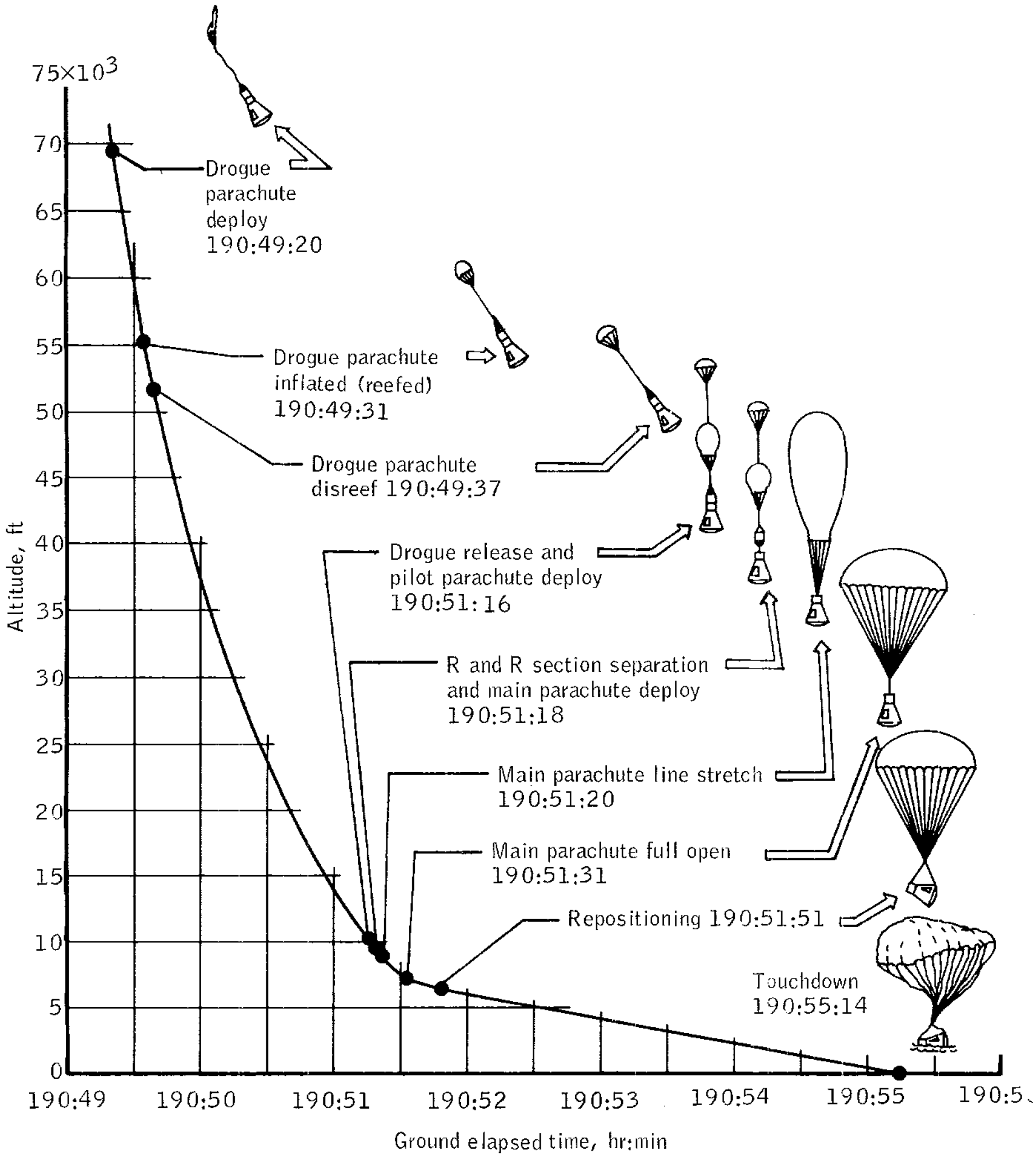

Figure 5.1.11-1. - Landing system performance.

5.1.12 Postlanding

The ultra-high frequency (UHF) descent and recovery antennas automatically deployed when the spacecraft was repositioned during the descent phase of the mission. The sea dye marker was automatically dispensed upon landing, and shortly thereafter the flashing recovery light and recovery hoist loop extended when the main parachute was jettisoned. Satisfactory deployment of these recovery aids is evidenced in many of the recovery photographs. The high frequency (HF) antenna failed to extend when commanded by the crew. See section 5.1.2 for a discussion of this item. The operation and effectiveness of the recovery aids are covered in the communications and recovery operations sections of this report.

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5.2 GEMINI LAUNCH VEHICLE PERFORMANCE

The performance of the Gemini launch vehicle was satisfactory in all respects except one. As reported by the crew, the duration and magnitude of longitudinal oscillations (POGO) were greater than normal. The cause of these greater values has been traced to an improper charge in the oxidizer standpipes (surge suppression chambers).

5.2.1 Airframe

The maximum launch vehicle loading, occurring in the pre-BECO region of flight, was 79 percent of design ultimate load. This compares with the lowest value of 76 percent on GT-1 and the highest of 81 percent which occurred on the Gemini IV flight.

5.2.1.1 Longitudinal oscillation (POGO).- The flight crew reported objectionable longitudinal oscillations (POGO) during the boost phase of flight, stating that panel gages could not be read to a desired degree of accuracy and that speech was difficult. Analysis of the flight data verifies that the onset of POGO occurred at LO+92 seconds, had a duration of 46 seconds, and reached a maximum amplitude of ±0.38g at the spacecraft-launch vehicle interface. The POGO amplitude was higher than the Gemini launch vehicle design goal of ±0.25g for approximately 13 seconds. POGO oscillations during this flight period were sustained oscillations as compared with the intermittent characteristic of the suppressed responses exhibited on previous vehicles (see fig. 5.2-1). A discussion as to the cause of this high value is given in section 5.2.2.2 of this report.

5.2.1.2 Structural loads.- Ground winds were approximately 5 mph during the Gemini V countdown, and the resulting structural loads were not significant.

Estimated structural loads are shown in the following table for the Gemini V flight. These data indicate that maximum loading occurred at station 320 in the pre-BECO region of flight.

Launch-vehicle station, in.	Max qα		Pre-BECO	
	Load, lb	Design ultimate, percent	Load, lb	Design ultimate, percent
276	26 000	26	48 000	48
320	133 000	39	273 000	79
935	430 000	59	441 000	61
1188	485 000	72	473 000	70

5.2.1.3 Post-SECO pulse.- A pulse of ±0.015g peak axial acceleration occurred 5.1 seconds after SECO and damped out within 0.3 second. The disturbance was not detected on actuator deflections or on rate gyro telemetry data, but the acceleration response was similar to that experienced on previous flights.

5.2.1.4 First stage recovery.- A substantial portion of the GLV-5 first stage was recovered by the U.S. destroyer DuPont. The recovered portion is shown in figure 5.2-2. As shown, the recovered section includes the entire oxidizer tank barrel, forward skirt, and forward dome. The tank was shipped to the contractor's plant for inspection. The results of this inspection and analysis will be published as a supplemental report.

5.2.2 Propulsion

Performance of the propulsion system was satisfactory. A comparison of preflight predicted with postflight reconstructed engine performance is shown in tables 5.2-I and 5.2-II for stages I and II, and indicates good agreement between predicted and actual performance.

5.2.2.1 Stage I engine performance.- The start transients of both subassemblies were of the expected form and within the range of GLV and Titan II experience. Analysis of landline data shows that the fuel pressurant differential pressure switch of the prelaunch malfunction detection system made momentarily at engine start signal + 0.13 second before making solidly at the expected time of engine start signal + 0.95 second. This phenomenon has been previously seen during the acceptance test of the engine to be installed on GLV-9 or GLV-10, and is believed to be caused by the initial pressurization of the autogenous lines during the start transient.

Engine performance was normal during steady-state operation except for the presence of pressure oscillations in five parameters during the period of LO+114 to LO+135 seconds. (See section 5.2.2.2.)

Visual observation and a review of the launch films showed that several momentary flashes occurred in the stage I exhaust plumes; however, no corresponding perturbations could be found in any of the telemetered vehicle parameters. A film review of previous launches has revealed that similar flashes occurred in GT-2, GT-3, Gemini IV, and several Titan II launches. The exact cause of these flashes is unknown; however, they are thought to be due to the tape used to secure desiccant bags in the turbine exhaust stacks.

Shutdown was initiated by oxidizer exhaustion with approximately 871 pounds of usable fuel remaining. Predicted mean outage was 568 pounds.

5.2.2.2 Pressure oscillations.- The data indicate that pressure oscillations were reflected only in the oxidizer and not in the fuel system. Oscillations in both thrust chamber pressures of 12 psi peak-to-peak at approximately 11 cycles per second were noted in subassemblies 1 and 2. Maximum oscillations of 34 psi peak-to-peak at 11 cycles per second were also noted in subassembly 2 oxidizer pump suction pressure and in both oxidizer pump discharge pressures (subassembly 1 oxidizer pump suction pressure was not instrumented). Comparison of the oxidizer system parameters measured on GLV-5 with previous Gemini and Titan II flight data indicates that the high oscillatory response resulted from improper operation of the oxidizer standpipe.

Postflight analysis and tests show that only about 10 percent of the normal bubble (volume of gas) was in the oxidizer standpipe after the charging procedure was completed. Various explanations of the reason for this $\frac{1}{10}$ size bubble have been advanced including: nitrogen gas absorption, gas escape, displacement of the N_2 gas by N_2O_4 vapor and consequent vapor condensation, and inadequate charging procedures. At the present time, tests are underway by the contractor to establish the prime contributors to the small bubble and, more importantly, to establish correct procedures to insure a proper charge under any conditions.

5.2.2.3 Stage II engine performance.- Performance of the stage II propulsion system was generally as predicted; however, oxidizer and fuel pump discharge pressures were slightly higher than expected. This was reflected in higher chamber pressure and thrust. Shutdown was initiated by radio guidance system (RGS) command and produced a shutdown thrust transient similar to that experienced on GLV-3 and GLV-4. Actual total impulse during the shutdown transient was approximately 36 600 lb-sec as opposed to a predicted value of 37 500 ± 7000 lb-sec.

5.2.2.4 Propellant loading and autogenous system performance.- The following tables provide data on loaded-propellant weight and flight-propellant temperatures. The actual propellant weights given in the first table agree well with requested loads, while the average propellant temperatues, listed in the second table, are lower than predicted.

The propellant temperatures were lower partly because of low wind velocity, but principally because the temperatures are predicted for a launch at the 1.7-hour point in the window.

PROPELLANT LOADING

Component	Stage I		Stage II	
	Requested	Actual	Requested	Actual
Fuel, lb	90 049	90 051	21 952	21 948
Oxidizer, lb	171 972	171 961	37 857	37 865

AVERAGE PROPELLANT TEMPERATURES

Component	Stage I		Stage II	
	Predicted	Actual	Predicted	Actual
Fuel, lb	49.0	42.7	45.0	44.3
Oxidizer, lb	51.7	45.1	51.2	46.9

Comparison of propellant tank pressures during flight with pre-flight predictions shows good agreement, indicating satisfactory autogenous system tank pressurization.

5.2.2.5 Performance margin.- Real-time calculations performed during the launch countdown indicated that the spacecraft weight would exceed predicted launch vehicle negative 3σ performance capability by 126 pounds at lift-off. The postflight reconstruction of vehicle performance indicated actual payload capability to be 8306 pounds. This capability was 359 pounds above the spacecraft weight, but 163 pounds less than the preflight predicted nominal of 8469 pounds. This was the first Gemini launch vehicle in which the achieved payload capability was less than the preflight predicted nominal. The following table gives predicted and actual values.

	Predicted	Actual
Spacecraft weight, lb	[a]7938	7947
Payload, -3σ real-time, lb	7812	N.A.
Payload, -3σ preflight, lb	7802	N.A.
Payload, nominal, lb	8469	8306

[a]Used by contractor in real-time performance calculations.

5.2.3 Flight Control System

The performance of the flight-control system was satisfactory. The primary flight-control system was in command throughout the flight and no switchover to the secondary system was required. The operation of three-axis reference system (TARS) and the inertial guidance system (IGS) was compatible during both stage I and stage II flight. Switchover could have been successfully accomplished at any time during the flight.

5.2.3.1 Stage I flight.- Ignition and lift-off transients were normal. The peak actuator travel and rate-gyro disturbances recorded during the ignition and holddown period are shown in table 5.2-III. The combination of thrust misalinement and engine misalinement at full thrust again initiated a small roll transient at lift-off. The control system responded satisfactorily to correct the roll transient, limiting the roll rate to a maximum of 1.1 deg/sec clockwise at 0.21 second after lift-off. No significant transients were noted in the pitch and yaw channels.

The TARS roll and pitch programs were properly executed. The rates and initiation times were nominal and within the first-stage trajectory requirements. All TARS-initiated discretes were executed as programed. The planned and actual roll and pitch programs are listed in section 5.2.5 (table 5.2-IV).

Analysis of the primary flight-control attitude error signals during stage I flight shows proper response to wind disturbances and to the guidance programs. The maximum rates and attitude errors are shown in table 5.2-V.

The TARS and IGS attitude signals for pitch, yaw, and roll are presented in figure 5.1-4. The stage I dispersions between the primary and secondary systems were caused primarily by gyro drift, errors in the TARS guidance programs, and reference axis cross-coupling effects. The magnitude of these dispersions was well within the primary system limits.

5.2.3.2 Separation.- Stage separation was satisfactory. Thrust-vector control was attained as soon as the stage II hydraulic system was pressurized. The pitch, yaw, and roll rates at stage separation were higher than those experienced during previous missions, and are attributed to the heavier spacecraft. The maximum attitude errors recorded were as follows:

Pitch, deg	+0.47 at BECO + 1.3 sec
Yaw, deg	+1.2 at BECO + 2.5 sec
Roll, deg	-0.3 at BECO + 1.2 sec

The maximum vehicle rates recorded during staging were as follows:

Pitch, deg/sec	-1.7 at BECO + 0.1 sec
Yaw, deg/sec	+1.4 at BECO + 1.2 sec
Roll, deg/sec	-1.7 at BECO + 0.2 sec

5.2.3.3 Stage II flight.- The radio guidance system (RGS) guidance enable command was initiated by the TARS timer at LO+162.59 seconds. The first pitch guidance command was received at LO+168.4 seconds and consisted of a small command followed by a full 2.0 deg/sec pitchdown for 5.8 seconds. Throughout the remainder of the flight, small pitch commands were transmitted to the vehicle to achieve the desired cutoff conditions.

The control system indicated attitude bias in both pitch and yaw during stage II. The yaw bias of +1.2° compares closely with the Gemini IV bias of +1.3°, and is approximately the same as the biases experienced on other Gemini flights. Both pitch and yaw biases were well within the predicted limits. The attitude errors in pitch, yaw, and roll are shown in figure 5.1-4. The biases are caused by engine thrust-vector misalinement, center-of-gravity travel off the vehicle longitudinal axis, and the position of the roll thrust vector off the longitudinal axis.

5.2.3.4 Post-SECO flight.- The vehicle pitch, yaw, and roll rates during the period from SECO through spacecraft separation appear in table 5.2-VI. Again, as on Gemini IV, the vehicle post-SECO rates were less than those experienced on the GT-1 and GT-2 flights. Spacecraft separation was accomplished at 23.63 seconds after SECO.

5.2.4 Hydraulic System

Operation of the hydraulic system was normal. Starting transients and steady-state values are shown in the following table:

System event	Stage I primary system	Stage I secondary system	Stage II system
Starting transient (max)	3880 psia	3380 psia	3760 psia
Starting transient (min)	2700 psia	3040 psia	3000 psia
Steady state	3000 psia	3040 psia	3000 psia
BECO	2760 psia	2960 psia	N.A.
SECO	N.A.	N.A.	2850 psia

5.2.5 Guidance System

The vehicle was guided by the primary Mod III radio guidance system (RGS) which performed satisfactorily throughout the countdown and flight.

5.2.5.1 Programed guidance.- The programed guidance was within acceptable limits, as shown in table 5.2-IV. As discussed in section 4, a slightly lofted first-stage trajectory was flown. The errors at BECO were 164 ft/sec low in velocity, 5383 feet high in altitude, and 0.89° high in flight-path angle.

5.2.5.2 Radio guidance.- The guidance system acquired the pulse beacon of the launch vehicle, tracked in the monopulse automatic mode, and was locked-on continuously from lift-off to 30 seconds after spacecraft-launch vehicle separation. At this time, there was a 12-second period of intermittent lock until final loss of signal at 66 seconds after SECO. Track was maintained to an elevation angle of 2.3° above the horizon. The average received signal strength at the central station during stage II operation was satisfactory. Rate lock was continuous, except for a momentary interruption at staging, from LO+44.4 seconds to LO+380.8 seconds (47.52 seconds after SECO). Rate lock was maintained to an elevation angle of 4.1° above the horizon.

Normal steering commands were issued, as planned, by the airborne decoder at LO+168.4 seconds. At this time, an initial 10-percent pitch-down steering command (0.2 deg/sec) was given for 0.5 second, followed by a 100-percent pitch-down steering command (2.0 deg/sec) for 5.8 seconds. The 100-percent command was given 2.8 seconds longer than nominal because of the slightly lofted first-stage trajectory. The steering gradually returned by 15 seconds later to relatively small and slowly varying pitch-down commands of 0.1 deg/sec. This produced generally negative pitch rates until LO+275.0 seconds; however, the rates, starting at LO+222.0 seconds, became quite oscillatory, varying between pitch-up and pitch-down commands of 0.1 to 0.3 deg/sec, until 2.5 seconds before SECO. Yaw steering started at LO+168.4 seconds. The yaw commands were of very small magnitude, with the commands over the radio-guided portion of flight amounting to positive and negative yaw rates of 0.04 to 0.06 deg/sec.

SECO occurred at LO+333.284 seconds at an elevation angle of 7.26°. The SECO+20 second conditions were well within 3σ limits. The flight-path angle was -0.01°, the velocity was 25 805 ft/sec, and the altitude was 531 025 feet. The planned column in table 4.3-I lists spacecraft separation conditions at SECO+20 seconds which may be compared with these conditions. The flight-path angle was 0.01° low, the velocity was nominal, and the altitude was 96 feet low. These differences do not precisely agree with the differences indicated in table 4.3-I because the actual conditions listed are for spacecraft separation which occurred 3.6 seconds later. Because the shut-down thrust transient was nominal, the small insertion errors were attributable to shut-down timing at SECO and to the noise in the guidance data. At the end of tail-off, vehicle rates were 0.77 deg/sec pitch-down, 0.31 deg/sec yaw-right, and 0.45 deg/sec roll-clockwise.

The computing system, in conjunction with the RGS ground and airborne systems, completed all prelaunch and launch operations in a normal and satisfactory manner. The spacecraft inertial guidance system ascent updates from the computer were sent by way of the spacecraft digital command system and verified by the buffer as follows:

Update sent, LO + sec	Update verified, LO + sec	Value, ft/sec
100.0	105.693	-319.5
140.0	145.693	-204.0

In figures 5.2-3 and 5.2-4, the velocity and flight-path angle are shown in the regions of SECO and tail-off. The launch-vehicle RGS data and the range safety data (MISTRAM I - 10 000-foot base legs) are shown to illustrate the quality of the post-SECO data used by the real-time computing complex (RTCC) at Houston to compute the orbital determination. Both sources of data, as compared with previous Gemini flights, experienced noisier data in the area of cut-off. This random noise is considered worse than the predicted magnitude for this time of the year. It can also be observed in the figures that the flight-path angle and velocity parameters gave distinct differences from those computed by using NASA tracking network information.

5.2.6 Electrical System

The electrical system operated normally throughout the flight. No anomalies were noted in any of the electrical parameters. Variation of load on both the auxiliary power system (APS) bus and the instrumentation power system (IPS) bus reflected electrical functions and sequence of events as expected. Both bus potentials remained within acceptable limits for the noted variations of electrical load. The characteristics of the ac power system as well as the instrumentation power sources remained constant and well within required limits.

5.2.7 Instrumentation System

5.2.7.1 Ground.- All measurements programed for use during the countdown and launch performed as anticipated. There were 121 measurements in use. The wiring associated with the outlet temperature measurement of the oxidizer heat exchanger was damaged during the recycle period and was not reparired for the launch. The umbilical sequence was as planned and complete in 0.760 second.

5.2.7.2 Airborne.- The removal of the FM/FM system, and the reduction in some engine parameters on GLV-5 and future missions reduced the number of measurements by 40. There were 191 measurements programed for this flight. No anomalies occurred in the countdown and launch, and data acquisition was 100 percent. Loss of signal for telemetry was LO+430.5 seconds.

5.2.8 Malfunction Detection System

Performance of the malfunction detection system (MDS) during preflight checkout and flight was satisfactory. All MDS hardware functioned properly with the exception of the stage II fuel tank channel B pressure indicator, which was intermittent. Table 5.2-VII presents MDS parameters.

5.2.8.1 Engine MDS.- The actuation times of the malfunction detection thrust chamber pressure switch (MDTCPS) have been evaluated. The stage I engine subassembly 1 and subassembly 2 (SA1 and SA2) switches actuated at 580 psia and 600 psia, respectively. The stage II malfunction detection fuel injector pressure switch (MDFJPS) pressure cannot be determined, because there was no analog telemetry channel of injector pressure. Switch actuation times and corresponding pressures were as follows:

Switch	Condition, psia	Actuation time from lift-off, sec	Pressure, psia
Subassembly 1 MDTCPS	Make: 540/600 Break: 585/515	-2.41 +153.50	580 520
Subassembly 2 MDTCPS	Make: 540/600 Break: 585/515	-2.36 +153.49	600 535
Subassembly 3 MDFJPS	Make Break	+154.28 +333.44	--- ---

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 indicators.- All tank pressure indicators performed satisfactorily except for stage II fuel B channel which was intermittent. Launch vehicle telemetry indicates that the transducer was looking into an open circuit rather than the 1333 ohms of the meter from LO+84 seconds to LO+151 seconds, LO+154 seconds to LO+318 seconds, and from LO+333 seconds to spacecraft separation. The command pilot reported full-scale indication of the meter during approximately these time periods and agreement with the A channel sensor at other times.

Each of the four GLV propellant tanks have redundant pressure tranducers whose outputs are displayed on paired analog display meters in the spacecraft panel. The primary (A) sensor from each tank is powered from the GLV-APS bus and the four analog needles are nearest the center of the two analog gages. The secondary (B) sensor from each tank is powered from the GLV-IPS bus and the four analog needles are located nearest the outside of the two analog gages.

An indication of power failure of either IPS or APS would register as an immediate full-scale deflection on the respective analog indicators. The malfunction that occurred was on only one analog indicator. However, the command pilot correctly diagnosed the problem as a wire or indicator failure rather than a loss of IPS power. The tank pressure monitor at MCC uses channel A only and was not aware of the malfunction. A failure analysis of recovered spacecraft equipment was made in an attempt to isolate the malfunction.

5.2.9 Range Safety and Ordnance

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

5.2.9.1 Flight termination system.- Both GLV command receivers were looking at minimum signal strengths of between 6 and 10 microvolts at approximately LO+322 seconds. A similar drop in received signal strength at approximately the same position in flight occurred on the Gemini IV mission. Data indicate that, had a command been transmitted during this period, it would have been successfully accomplished by the GLV flight termination system. The Gemini spacecraft command receiver did not experience this drop in signal strength. A review of the ground transmitter power output disclosed no change in transmitted power during this period. This phenomenon is again attributed to the pattern characteristics of the GLV command receiver antenna configuration.

The following command facilities were used:

Time, sec	Facility
LO to LO+66	Cape 600-W transmitter and single helix antenna
LO+66 to LO+115	Cape 10-KW transmitter and quad helix antenna
LO+115 to LO+453	GBI 10-KW transmitter and ESCO 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 (IP) and provided accurate information through insertion. These data were selected for input to the IP for a total of 289.8 seconds.

Prior to lift-off, an unlock of one receiver at the central site occurred. As a result, no calibrated rate data could be obtained from the west 100 000 foot leg and it was necessary to use data from the west 10 000 foot leg throughout the launch. Automatic track in azimuth and elevation was maintained from 22.4 to 382.2 seconds after lift-off. During the first 159 seconds after lift-off, polarization was manually updated and polarization track was inhibited, but track was stable after that period until LO+382 seconds. Approximately 308 seconds of MISTRAM I data were reconstructable for postflight use.

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

5.2.10 Prelaunch Operations

5.2.10.1 Launch attempt.- Problems encountered by the spacecraft forced a delay of 1 hour 27 minutes in initiation of propellant loading. The loading was complete in 3 hours 21 minutes. The launch vehicle continued the split count at T-240 minutes. At T-134 minutes the launch pad crew discovered a leak in the pressure regulator for the remote charging system for the oxidizer standpipe. In the final portion of the launch count (T-34 minutes) a manual charging procedure was initiated, and it was complete in approximately 10 minutes. This operation delayed the lowering of the erector for 10 minutes.

Following a spacecraft anomaly at T-10 minutes, a hold was initiated at 1708 G.m.t. After 5 minutes in the hold, the Supervisor of Range Operations warned of thunderstorm activity in the Cape Kennedy area. The vehicle erector was raised after 10 minutes and at hold plus 31 minutes (1741 G.m.t.) on August 19, 1965, the launch attempt was cancelled.

5.2.10.2 Recycle.- The recycle activities consisted of off-loading of propellants, removing start cartridges and destruct initiators, and special engine inspections to insure that oxidizer was not leaking by the thrust-chamber valves. Electrical power was left on the vehicle during this period.

The off-loading of propellants was accomplished between 2400 G.m.t., August 19, and 0300 G.m.t. on August 20, 1965.

An investigation of the leaking regulator revealed metal chips within the regulator. Because of the possibility of these metal chips passing through the regulator and damaging the electrical/manual ball valve downstream of the regulator, as well as on the launch vehicle, it was decided to use the manual charging procedure on the subsequent launch.

5.2.10.3 Launch.- The split count was initiated through the range sequencer for the rescheduled launch at 1700 G.m.t. on August 20, 1965. Propellant prechill was begun at 0015 G.m.t. and propellant loading was completed at 0340 G.m.t. on August 21, 1965. Loading was accomplished in 3 hours 25 minutes. Because the prevalve remained open during the recycle, a weight correction had to be applied to the loading schedule to compensate for this difference in configuration. A manual oxidizer standpipe charging procedure was accomplished, starting at T-140 minutes.

At T-34 minutes, a hold fire circuit (which monitors parameters associated with correct launch vehicle and AGE status) was found to be inoperative. By placing the selection switch in the "test" position, the circuit was found to function properly. The circuit did function properly, while in the "test" mode, until T-10 seconds when it was manually turned off. (Normally, this parameter is automatically programed on at T-35 minutes and off at T-2 seconds.) It was subsequently determined that the RCA sequencer, which is part of the Master Operations Control Set, had malfunctioned; it had not turned this hold-fire circuit on. Investigation is underway to determine the cause of this malfunction.

No further delays were encountered and launch was successfully accomplished. Pad damage was minimal.

Flight crew debriefing revealed an event which occurred during erector lowering preparations. A vibration was introduced into the vehicle and has not yet been explained. Investigation has been instigated to resolve this vibration and eliminate it if possible.

TABLE 5.2-I.- PRELIMINARY STAGE I ENGINE PERFORMANCE PARAMETERS

Parameter	Preflight predicted	Postflight reconstructed	Difference, percent
Thrust[a] (engine), lb	438 696	438 281	-0.09
Thrust (engine flight average), lb	461 711	464 377	0.57
Specific impulse[a], $\frac{\text{lb-sec}}{\text{lb}}$	260.91	260.84	-0.03
Specific impulse (flight average), $\frac{\text{lb-sec}}{\text{lb}}$	278.10	278.40	0.11
Engine mixture ratio[a]	1.9387	1.9554	0.86
Engine mixture ratio (flight average) . . .	1.9234	1.9359	0.65
Burn time (87FS1 to 87FS2), sec	158.20	156.86	-0.85

[a]Standard inlet condition

TABLE 5.2-II.- PRELIMINARY STAGE II ENGINE PERFORMANCE PARAMETERS

Parameter	Preflight predicted	Postflight reconstructed	Difference, percent
Thrust[a] (engine), lb[b]	101 200	102 717	1.50
Thrust (engine, flight average), lb[b] . . .	101 404	103 099	1.67
Specific impulse[a], $\frac{\text{lb-sec}}{\text{lb}}$[b]	312.35	312.23	-0.04
Specific impulse (flight average), $\frac{\text{lb-sec}}{\text{lb}}$[b]	312.91	312.61	-0.10
Engine mixture ratio[a]	1.7706	1.7629	-0.43
Engine mixture ratio (flight average) . . .	1.7378	1.7387	0.05
Burn time (91FS1 to 91FS2), sec	182.10	179.74	-1.24
Burn time remaining, sec	1.0		

[a]Standard inlet condition

[b]Includes roll control nozzle thrust

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

Actuator designation	Maximum during ignition		Maximum during holddown null check, in.
	Travel, in.	Time from T-0, sec	
Pitch 1_1	-0.04	-2.53	-0.01
Yaw/roll 2_1	+0.03	-2.52	+0.01
Yaw/roll 3_1	+0.09	-2.55	+0.01
Pitch 4_1	-0.10	-2.57	-0.01

Axis	Maximum rate stage I gyro, deg/sec	
	Primary	Secondary
Pitch	+0.30	-0.28
Yaw	+0.21	+0.20
Roll	+0.51	+0.50

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TABLE 5.2-IV.- PLANNED AND ACTUAL LAUNCH VEHICLE EVENT TIMES AND RATES

Event	Planned time from lift-off, sec	Actual time from lift-off, sec	Difference, sec	Planned rate, deg/sec	Actual rate, deg/sec	Difference, deg/sec
Roll program start	10.16	10.13	-0.03	1.25	1.19	-0.06
Roll program end	20.48	20.45	-.03	1.25	1.19	-.06
Pitch program 1 start	23.04	23.09	.05	-.709	-.75	.041
Pitch program 1 end	88.32	88.35	.03	-.709	-.75	.041
Pitch program 2 start	88.32	88.35	.03	-.516	-.56	.044
Pitch program 2 end	119.04	119.06	.02	-.516	-.56	.044
Pitch program 3 start	119.04	119.06	.02	-.235	-.25	.015
Pitch program 3 end	162.56	162.61	.05	-.235	-.25	.015

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

Axis	Attitude error, deg	Time from lift-off, sec
Pitch	+1.31	68.0
Yaw	+0.53	81.3
Roll	+0.79	152.0
Axis	**Rates, deg/sec**	**Time from lift-off, sec**
Pitch	+0.31 -1.0	0.3 83.0
Yaw	+0.21 -0.30	0.4 83.0
Roll	+1.63 -1.10	10.9 153.5

TABLE 5.2-VI.- VEHICLE RATES BETWEEN SECO AND SPACECRAFT SEPARATION

Pitch axis	Rate, deg/sec
Max position rate at SECO+3 sec	+1.03
Max negative rate at SECO+19.1 sec	-0.89
Rate at SECO+20 sec	-.89
Rate at spacecraft separation (SECO+23.63 sec)	-.77
Yaw axis	
Max positive rate at SECO+18 sec	+0.31
Max negative rate at SECO+3 sec	-.39
Rate at SECO+20 sec	+.31
Rate at spacecraft separation (SECO+23.63 sec)	+.31
Roll axis	
Max positive rate at SECO+18 sec	+0.66
Max negative rate at SECO+9 sec	+.35
Rate at SECO+20 sec	+.66
Rate at spacecraft separation (SECO+23.63 sec)	+.45

TABLE 5.2-VII.- GEMINI V MALFUNCTION DETECTION SYSTEM SWITCHOVER PARAMETERS

Parameter	Switchover setting	Maximum or positive	Time from lift-off, sec	Minimum or negative	Time from lift-off, sec
Stage I primary hydraulics	Shuttle spring (1500 psia equiv)	3080 psi	-2.16	2700 psi	-2.43
Stage I secondary hydraulics	None	3370 psi	-2.70	2960 psi	BECO
Stage I tandem actuators					
No. 1 subassembly 2 pitch	±4.0 deg	+0.35 deg	40.0	-0.50 deg	68.0
No. 2 subassembly 2 yaw/roll	±4.0 deg	+0.20 deg	80.5	-0.35 deg	39.0
No. 3 subassembly 1 yaw/roll	±4.0 deg	+0.02 deg	39.0	-0.50 deg	80.5
No. 4 subassembly 1 pitch	±4.0 deg	+0.50 deg	68.0	-0.40 deg	40.0
Stage I pitch rate	+2.5 deg/sec -3.0 deg/sec	+0.20 deg/sec	0.5	-1.04 deg/sec	83.5
Stage I yaw rate	±2.5 deg/sec	+0.22 deg/sec	66.5	-0.28 deg/sec	84.0
Stage I roll rate	±20 deg/sec	+1.60 deg/sec	11.0	-1.7 deg/sec	153.70
Stage II pitch rate	±10 deg/sec	+0.10 deg/sec	321.0	-2.02 deg/sec	172.0
Stage II yaw rate	±10 deg/sec	+1.00 deg/sec	155.5	-0.20 deg/sec	159.5
Stage II roll rate	±20 deg/sec	+0.90 deg/sec	159.2	-0.10 deg/sec	325.5

Note: + indicates up, right, clockwise; - indicates down, left, counterclockwise

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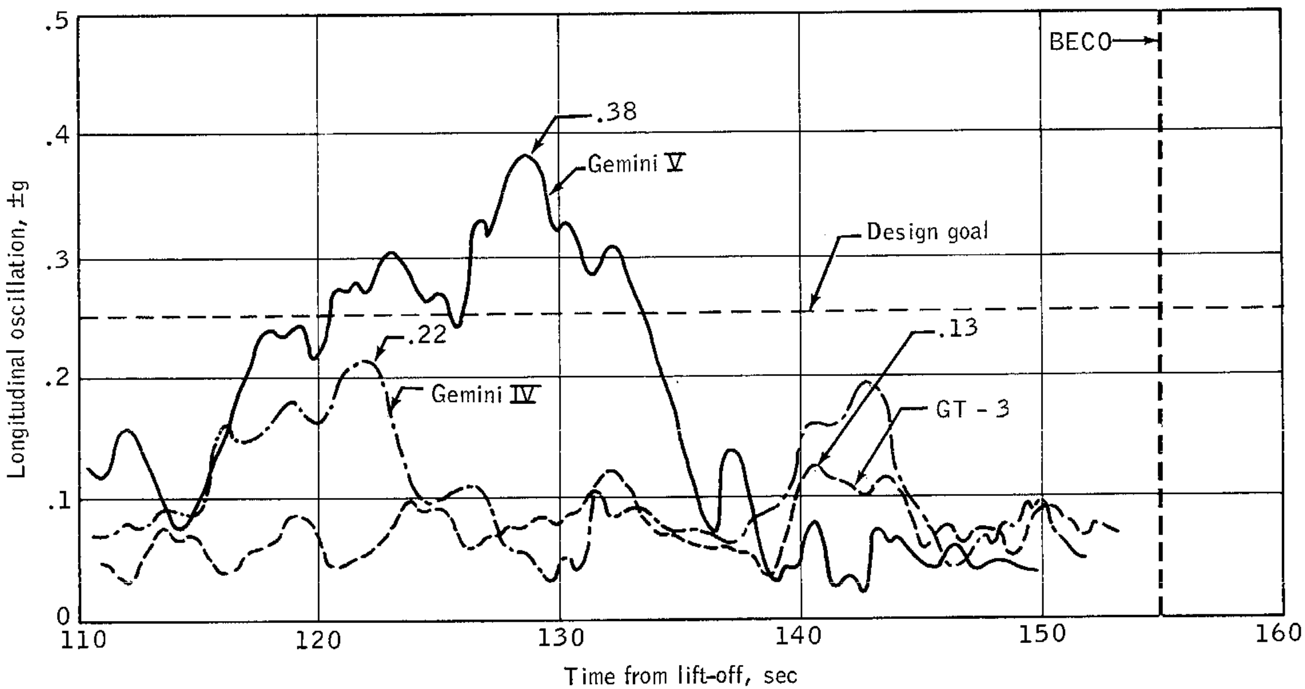

Figure 5.2-1.- GLV longitudinal oscillation.

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Figure 5.2-2. - Recovered GLV-5 stage I oxidizer tank.

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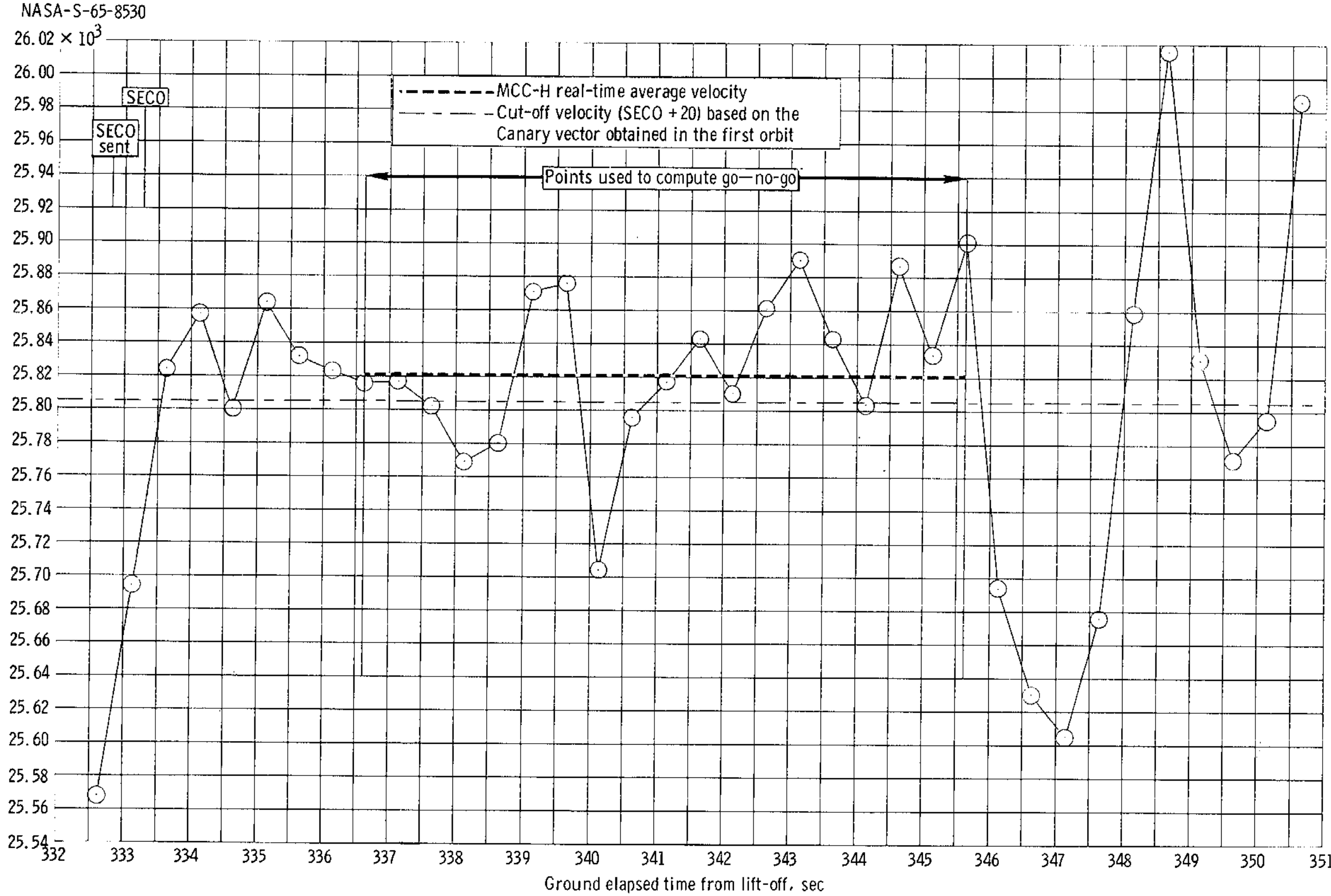

(a) Launch vehicle guidance data.

Figure 5.2-3.- Space-fixed velocity in the region of SECO.

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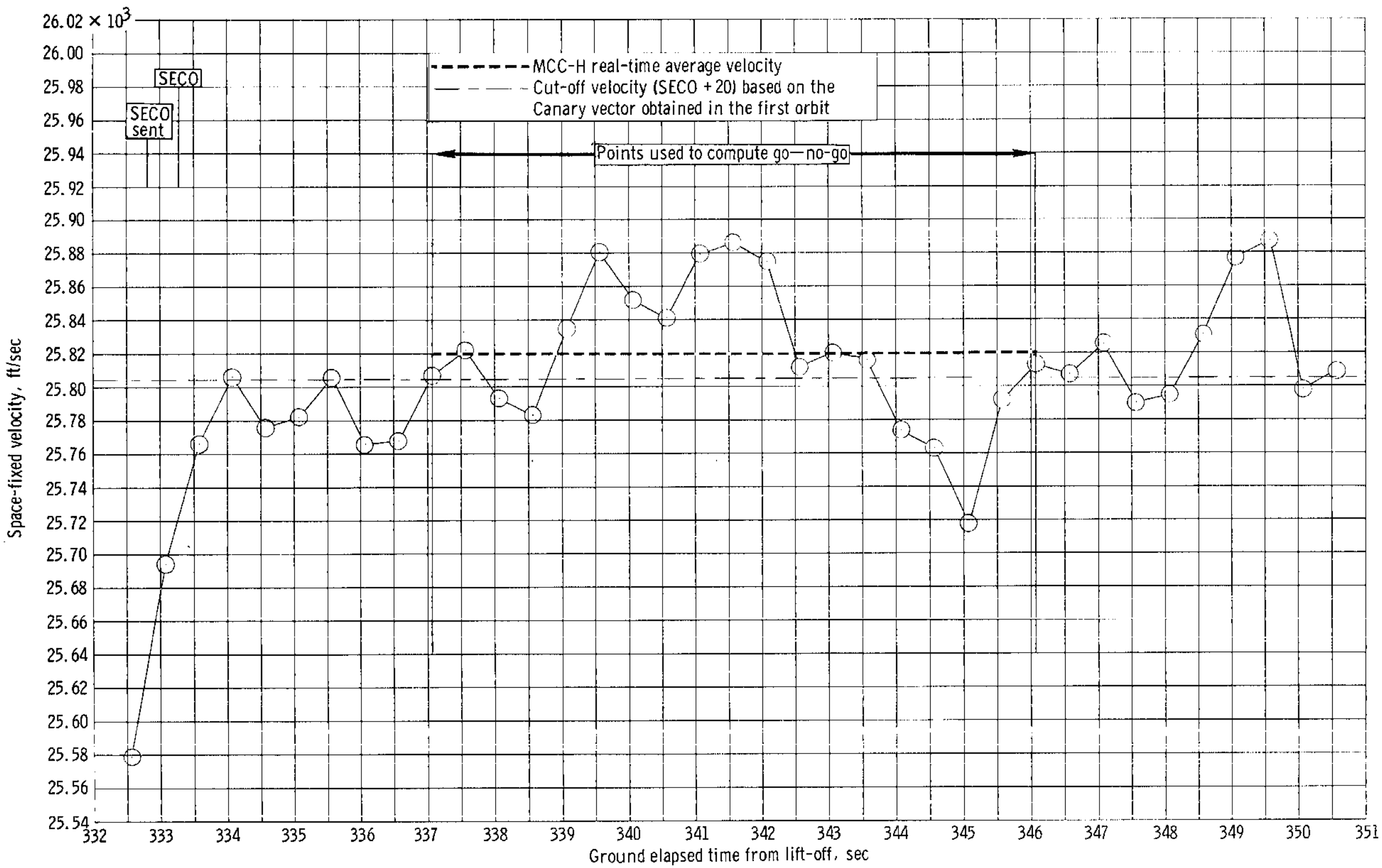

(b) MISTRAM I range safety computer (IP-3600) data.

Figure 5.2-3.- Concluded.

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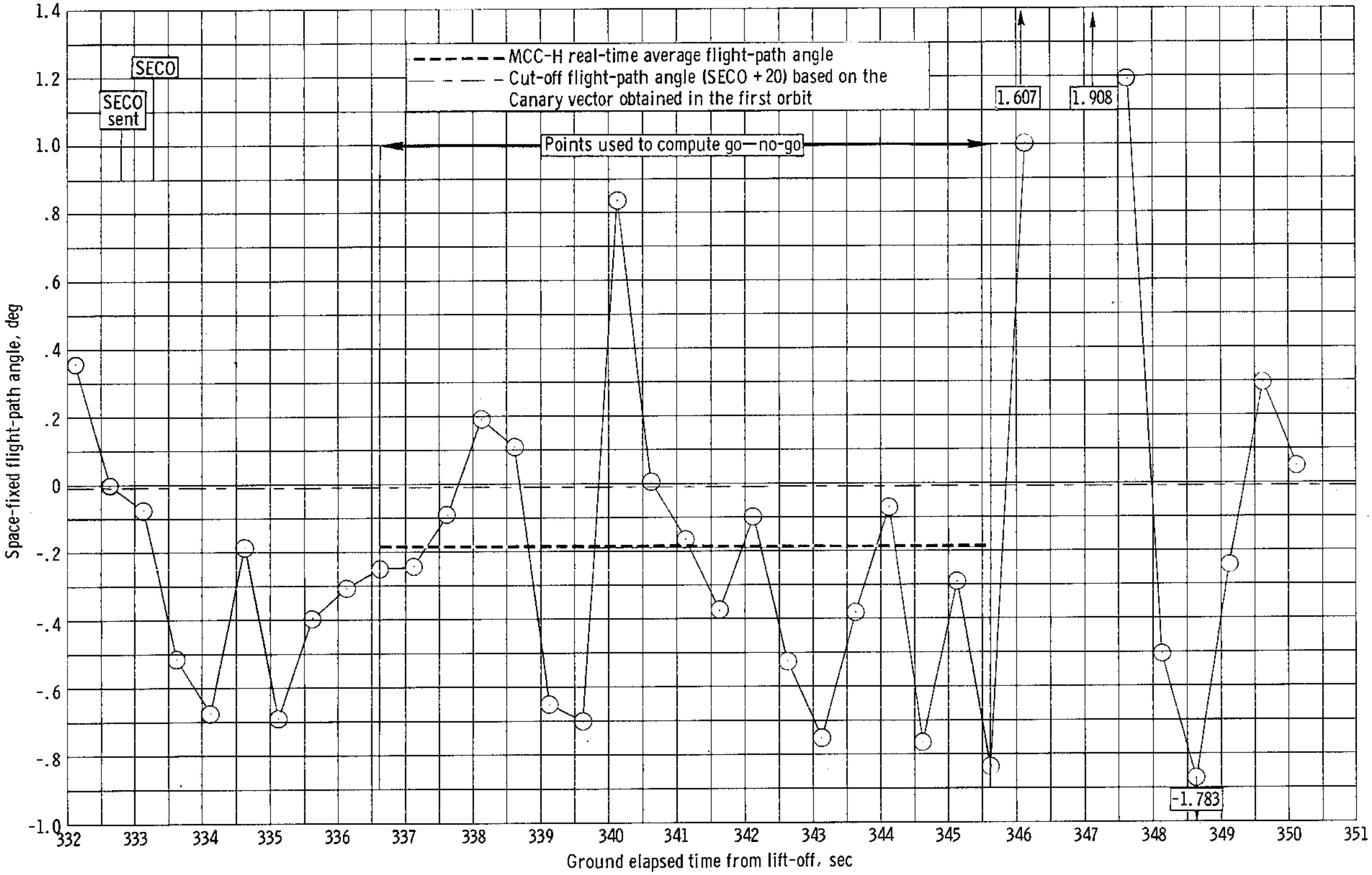

(a) Launch vehicle guidance data.

Figure 5.2-4.- Space-fixed flight-path angle in the region of SECO.

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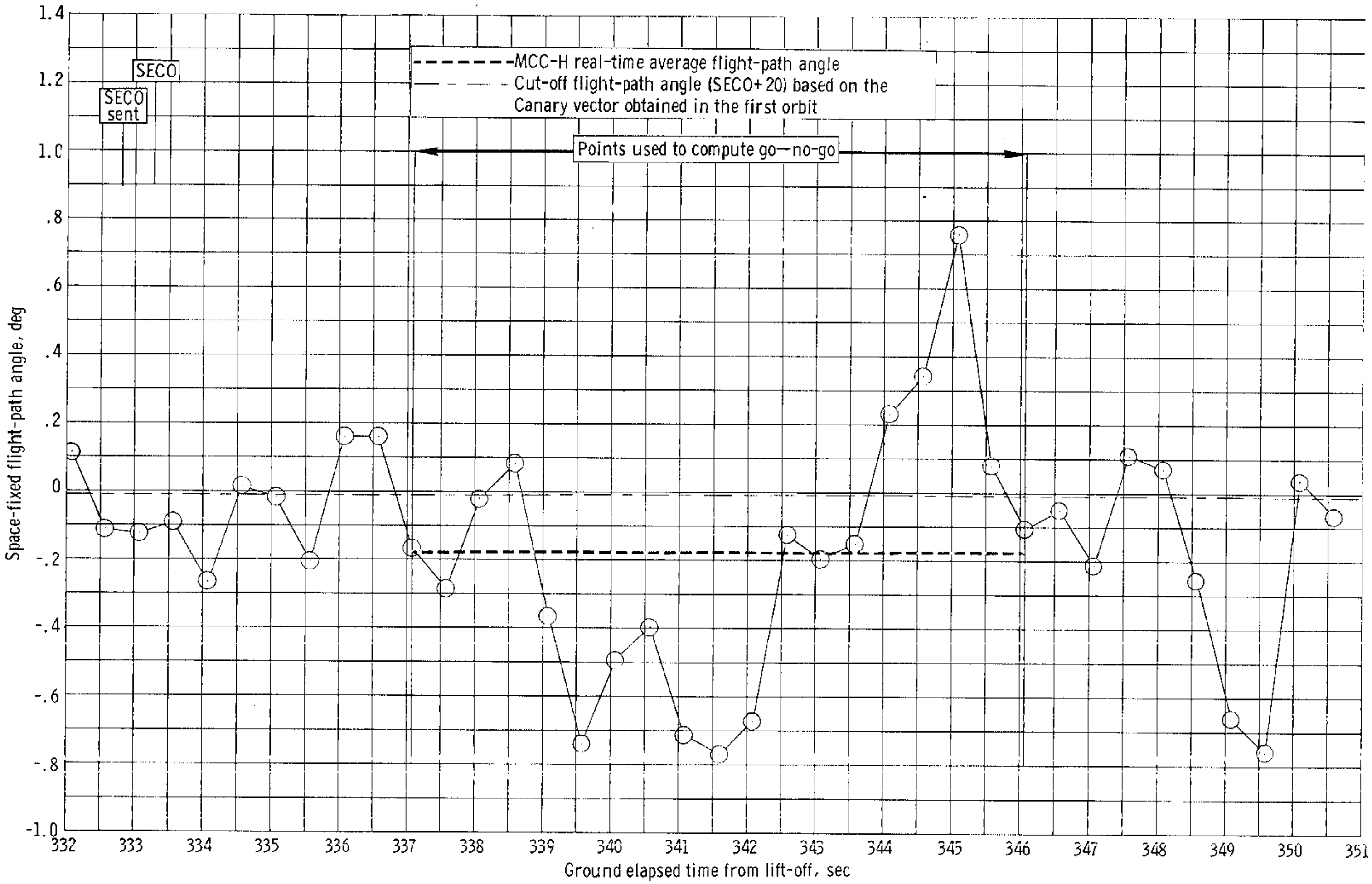

(b) MISTRAM I range safety computer (IP-3600) data.

Figure 5.2-4. - Concluded.

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5.3 SPACECRAFT—LAUNCH-VEHICLE INTERFACE PERFORMANCE

The various aspects of the spacecraft—launch-vehicle interface as defined in reference 9 performed within specification limits. The performance of the electrical and mechanical interfacing systems was derived from the overall performance of the launch vehicle and the spacecraft as determined from instrumentation and crew observation.

The electrical circuitry performed as anticipated except for the intermittent operation of the pressure-indicating system for the stage II B fuel tank. A discussion as to the cause of this intermittent operation is given in section 5.2.8 of this report. All other facets of the electrical interface performed nominally as indicated by the passive condition of the malfunction detection system (MDS) performance and the spacecraft inertial guidance system (IGS) steering signals.

Mechanical interface inspection before and after the final mating of the launch vehicle and spacecraft showed the configuration to be as specified by the interface drawings. The venting and sealing requirements of the spacecraft adapter and the skirt area of the launch vehicle were inspected and determined to be in accordance with the specification drawings.

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6.0 MISSION SUPPORT PERFORMANCE

6.1 FLIGHT CONTROL

The Gemini V mission was controlled from the Mission Control Center at the Manned Spacecraft Center, Houston, Texas (MCC-H). This was the first mission in which the Mission Control Center at Cape Kennedy (MCC-C) was not used as a backup to MCC-H during the launch phase.

This section of the report is based on real-time observations, and may not agree with some of the detailed evaluations in other sections of the report that were derived from postflight analysis of all available data from the flight.

6.1.1 Premission Operations

6.1.1.1 Premission activities.- The flight-control teams at the MCC-H conducted simulations and provided support to the launch-site operations during the premission phase. Support was provided for the Simultaneous Launch Demonstration (SLD), the Final Systems Test, the Simulated Flight Test, and for the launch attempt on August 19, 1965. This support operation provided the flight controllers with the opportunity to monitor telemetry from the launch vehicle and the spacecraft, to send commands to the spacecraft, and to observe systems tests on both the launch vehicle and the spacecraft. This support also resulted in an operational checkout of the equipment in the MCC-H.

6.1.1.2 Documentation.- The documentation for the mission was satisfactory. However, numerous changes in the flight plan were required during the mission as a result of the fuel-cell oxygen supply pressure problem and the associated effect on the mission.

6.1.1.3 MCC/network flight control operations.- The network went on mission status on August 4, 1965, when flight controllers were deployed to the remote sites. The remote sites and MCC-H went through the normal preflight schedule of simulations and checkout. The tests were successful and all sites were ready to support the mission on August 21, 1965.

6.1.1.4 Countdown.- The countdown was completely nominal as seen by MCC-H. MCC-H entered the countdown at T-300 minutes. At T-260 minutes, the Flight Dynamics Officer (FIDO) participated in a trajectory run which was successful. At T-189 minutes, the Retrofire Controller updated the time of retrograde (T_R) by way of the DCS with the revolution 2, area 1 reentry area time. The update was validated. At T-145 minutes, FIDO participated in the second trajectory run which was successful. At various times during the countdown, voice checks were made with the remote site, and in each case the communications were good.

There were no problems at any time during the countdown which could have caused a hold or impaired the MCC-H support of the mission. It should be mentioned, however, that transmission of the auxiliary stage II cut-off (ASCO) command from MCC-H was inhibited because of noise in the lines between MCC-H and MCC-C. This noise had also caused the ASCO command to be inadvertently transmitted during the countdown of the launch attempt on August 19, 1965. The ASCO command, if needed, would have been transmitted by the Range Safety Officer (RSO) at the Air Force Eastern Test Range (ETR).

6.1.2 Mission Operations Summary

6.1.2.1 Powered flight.- At lift-off, the ground time-to-retrofire clock did not start. The clock did start at 3 minutes ground elapsed time (g.e.t.) and was reported to be lagging by 2 seconds, requiring correction. The roll and pitch programs started and stopped at the nominal times. At LO + 30 seconds, the spacecraft telemetry transmissions became intermittent. This problem persisted until late into the stage I thrust, and the real-time telemetry was commanded by MCC-H to the standby transmitter. Switching the transmitters did not solve the problem; however, the telemetry did clear up at some time around staging. Both the 105-second and 145-second inertial guidance system (IGS) updates were correctly sent, received, and verified. At about LO + 2 minutes, the flight-path angle appeared to be a little low; however, the radio guidance system (RGS) steered the vehicle onto the correct flight-path angle during stage II flight. At approximately LO + 100 seconds, the crew reported loss of the stage II fuel tank pressure gage powered by the instrumentation power supply (IPS). This anomaly was not seen on the launch-vehicle telemetry records and was considered to be a gage or circuitry failure rather than a transducer failure. The crew reported feeling longitudinal oscillations (POGO) at approximately 126 seconds. The Booster Systems Engineer saw no indications on telemetry of the excessive POGO.

At cut-off, the following conditions were achieved as indicated at MCC-H:

Data source	Velocity, ft/sec	Flight-path angle, deg
GE/Burroughs	25 820	-0.20
IP 3600	25 819	- .19
Bermuda radar	25 802	- .02

The orbit, based on Bermuda data, was 86.8 nautical miles by 189 nautical miles and required an apogee maneuver of about 10 ft/sec for the rendezvous evaluation pod (REP) exercise.

6.1.2.2 Orbital.- At spacecraft separation, the crew accomplished a 5-ft/sec maneuver and proceeded with their insertion checklist. At Canary Islands, the telemetry was good, and the system was switched back to the primary transmitter. At Carnarvon, the perigee was raised with a 9.7-ft/sec maneuver. Bermuda radar data subsequently confirmed that the orbit had a perigee of 92.1 nautical miles and an apogee of 188.5 nautical miles. At the beginning of the second revolution, the fuel-cell oxygen supply tank pressure had dropped from a value of 810 psi at lift-off to 450 psi under a heavy electrical load and after purging of both sections. This pressure had been read at Canary Islands, the last station able to receive telemetry data prior to the REP ejection. The heater switch had been placed in the manual ON position during revolution 1 and MCC-H was aware of this event. Considering the heavy electrical load, the purging effects, and the fact that the pressure was well above the 200 psi specification minimum inlet pressure of the regulator, REP ejection was recommended. At 02:07:15 g.e.t., the REP was successfully ejected as planned and the radar was providing good readouts of range and range rate. About 10 minutes after REP ejection, the crew reported over Carnarvon that the pressure in the fuel-cell oxygen supply tank was dropping very rapidly. During this pass over Carnarvon, the pressure was determined to be 330 psi, and approximately 18 minutes later over Hawaii the pressure had dropped to 116 psi. In order to maintain the oxygen pressure, the spacecraft was powered down and by revolution 4 the electrical load was reduced to about 13 amperes. The oxygen pressure stabilized at 71.2 psia. It was determined at that time that the oxygen heater had failed. The decision was made to take fuel-cell section 2 off line and to turn off the coolant pump in the secondary loop. The decision was also made to purge only the hydrogen side of the fuel-cell sections and to do so on a normal 6-hour cycle. The quantity and pressure time histories for fuel-cell reactants are shown in figures 6.1-1 and 6.1-2. These trends were plotted and used in real time by the MCC-H.

The pressure began to increase and at about 10:45:00 g.e.t., the pressure was approximately 80 psia. At this time, the decision was made to increase the power load gradually. A short time later, a fuel-cell oxygen purge of 2-minute duration was successfully completed on section 1. At 16:59:00 g.e.t., a second oxygen purge of section 1 was successfully completed and section 2 was brought back on the main bus and performed normally. At approximately 23:00:00 g.e.t., the electrical load had gradually been increased to 20 amperes and the oxygen tank pressure had again increased to 80 psia. At this time in the mission, operations were somewhat normal as far as electrical power usage was concerned, and the crew began to perform flight-plan items and experiments. The oxygen pressure continued to increase but was kept under close surveillance and no further trouble was experienced with it during the mission.

At approximately 94:00:00 g.e.t., the crew reported that the primary horizon sensor malfunctioned and would not maintain the spacecraft in the proper pitch attitude. A test was performed at 98:00:00 g.e.t. which confirmed that the horizon sensor had failed. The secondary horizon sensor was used for the remainder of the mission and performed satisfactorily.

Four rendezvous radar tests were conducted during the mission by using the backup REP transponder at Cape Kennedy. Good radar lock-on was achieved on all four tests, but no digital range or range-rate readouts were obtained on the last three tests. The cause of this lack of digital output information was not determined; however, it was not considered to be a spacecraft computer problem.

The orbital attitude and maneuver system (OAMS) attitude thruster 7 (yaw-left) failed at approximately 120:00:00 g.e.t. The crew reported that when the thruster was fired, vapor could be seen, indicating that oxidizer flow had stopped and fuel was flowing out unburned. A short time later the crew reported that thruster 8 (yaw-left) was producing no thrust, but combustion could be seen. The OAMS heaters were turned on to correct for any possible oxidizer freezing. During the sixth day, the entire OAMS became sluggish. During the last few orbits of the mission, the OAMS had degraded to the point where it was unusable. Tests were performed to determine the cause of the thruster problems, but the results were inconclusive. Possible causes considered at the time were clogging of the lines, freezing of the lines, or a combination of both.

In contrast to the fuel-cell oxygen tanks, both the fuel-cell hydrogen and environmental control system (ECS) oxygen tanks vented at high pressure due to heat leak as anticipated. As shown in figure 6.1-2, the hydrogen pressure varied with demand, reaching a peak vent rate at approximately 120:00:00 g.e.t. The sharp pressure drop at that time was believed to be caused by normal poppet action of the pressure relief valve. ECS oxygen curves in figure 6.1-3 show the rise in the primary oxygen supply pressure at 26:00:00 g.e.t., when the automatic heater was turned on. Subsequently, the pressure rose to the vent pressure of 1020 psia; however, the mass quantity usage curve indicates a negligible amount of oxygen loss.

The water management system was cause for concern during the mission because the fuel cells were apparently producing more water than anticipated. On the fifth day, the spacecraft was powered down to minimize water production of the fuel cells to insure that the remaining water storage space would be sufficient for the planned 8-day mission. Real-time computation of water quantity (fig. 6.1-4) was difficult because of a combination of factors. The water management system—fuel-cell system interface was of prime concern because of the risk of fuel-cell damage which would have resulted had tank B been emptied of drinking water. The most accurate computation was based on tank A quantity change and the quantity of water drunk as reported by the crew. The inaccuracy in determining the latter quantity was on the order of 20 percent. An even less accurate computation of tank B product-water quantity was based on ampere-hours delivered by the fuel cells. This computation was biased by the obvious errors in the oxygen quantity readout from the gaging system and by the uncertainty in estimating gas leakage through the water separator plates in the fuel cells. In addition, the hydrogen venting, which began at approximately 43 hours g.e.t., made this technique unusable after that time.

Beginning on the second day, the delayed-time telemetry data became increasingly noisy, probably because of damage to the recorder tape surface. During the remainder of the flight, different portions of the tape were used with varying results. Toward the end of the mission, some improvement was noted even though that particular segment of tape had been used for 4 days continuously. At the end of the mission, the last recorded tape transmission was timed so as to position the final orbit and reentry data on the final portion of the tape to obtain the cleanest possible record of spacecraft reentry performance.

6.1.2.3 Reentry.- At Carnarvon (CRO) on revolution 120, the T_R of 190:27:43 g.e.t. for a revolution 121, area 1 reentry was transmitted to the spacecraft and confirmed to be synchronized with the ground T_R clock. The reentry command update was also transmitted and was checked by having the crew read out 2 cores with the manual data insertion unit (MDIU). (Incorrect coordinates were transmitted to the spacecraft and for a detailed discussion, refer to section 6.2.2.2.1.)

Retrofire occurred exactly on time over Hawaii. The retrorocket thrust based on the incremental velocity indivator (IVI) appeared to be nominal and the crew reported that the spacecraft attitude was good during retrofire. The Hawaii radar tracking data were of poor quality and were not used. The California and White Sands radar tracking were good, and the guidance officer used these data to determine the back-up guidance quantities of roll left 54°, roll right 68°, and a retrofire-elapsed-time-to-reverse bank (RETRB) of 19 minutes 25 seconds. The back-up quantities were relayed to the crew prior to blackout, which occurred 16 minutes 23 seconds after retrofire. The landing area footprint, based on California and White Sands data, did not shift after retrofire.

The crew reported after blackout that they did not get guidance and had flown the back-up guidance quantities. The crew thought that they would land a little short, and in fact, they were about 89 nautical miles short.

The crew made contact with the carrier at 190 hours 51 minutes g.e.t. and reported that they were on the main parachute and in landing attitude. The spacecraft landed at approximately 190 hours 56 minutes g.e.t. Except for one transmission immediately after landing, no further communications could be obtained from the crew. The cause of the communications failure was assumed to be a battery failure at that time.

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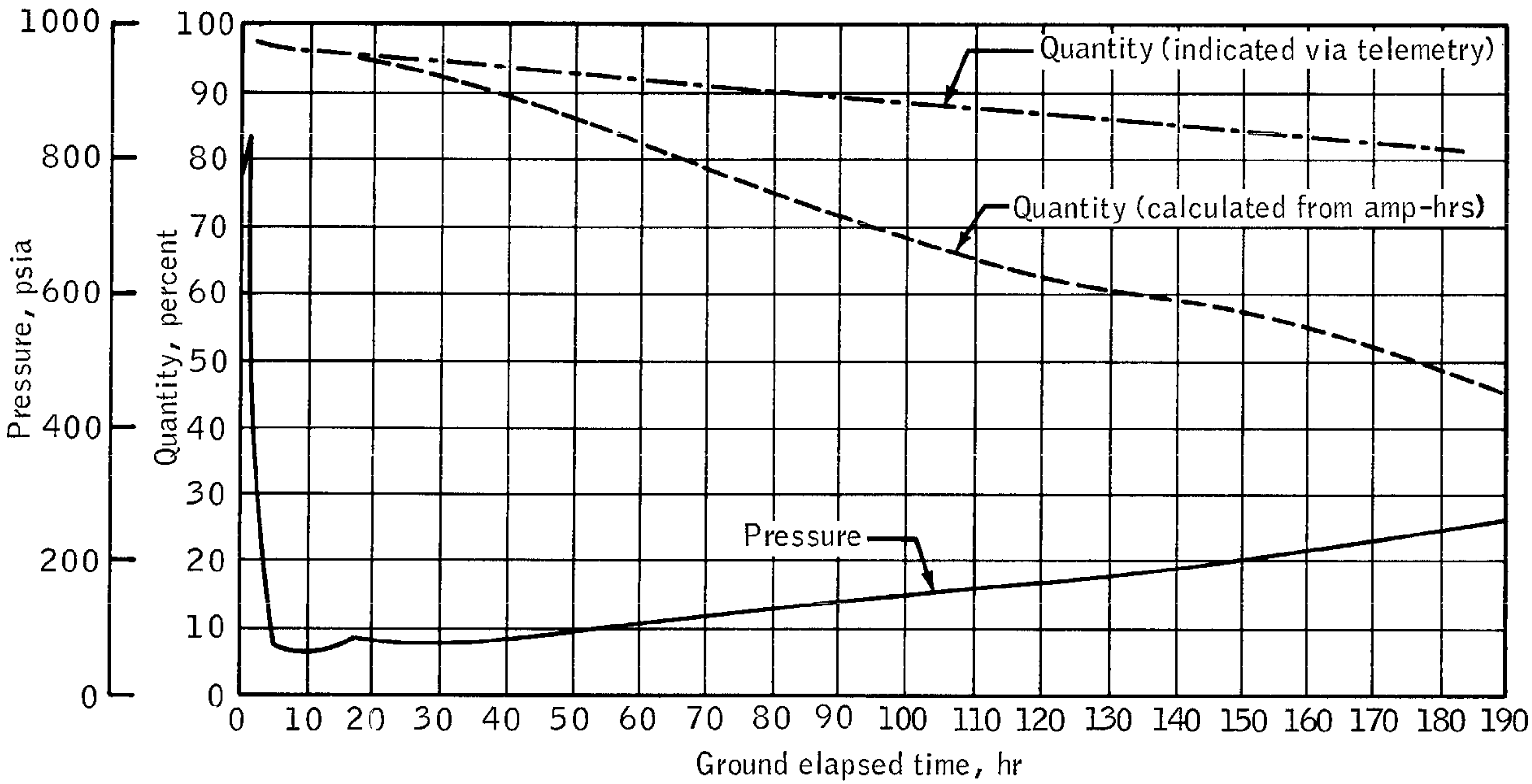

Figure 6.1-1.- Real - time data concerning the fuel-cell oxygen supply.

UNCLASSIFIED

UNCLASSIFIED

NASA - S-65-8580

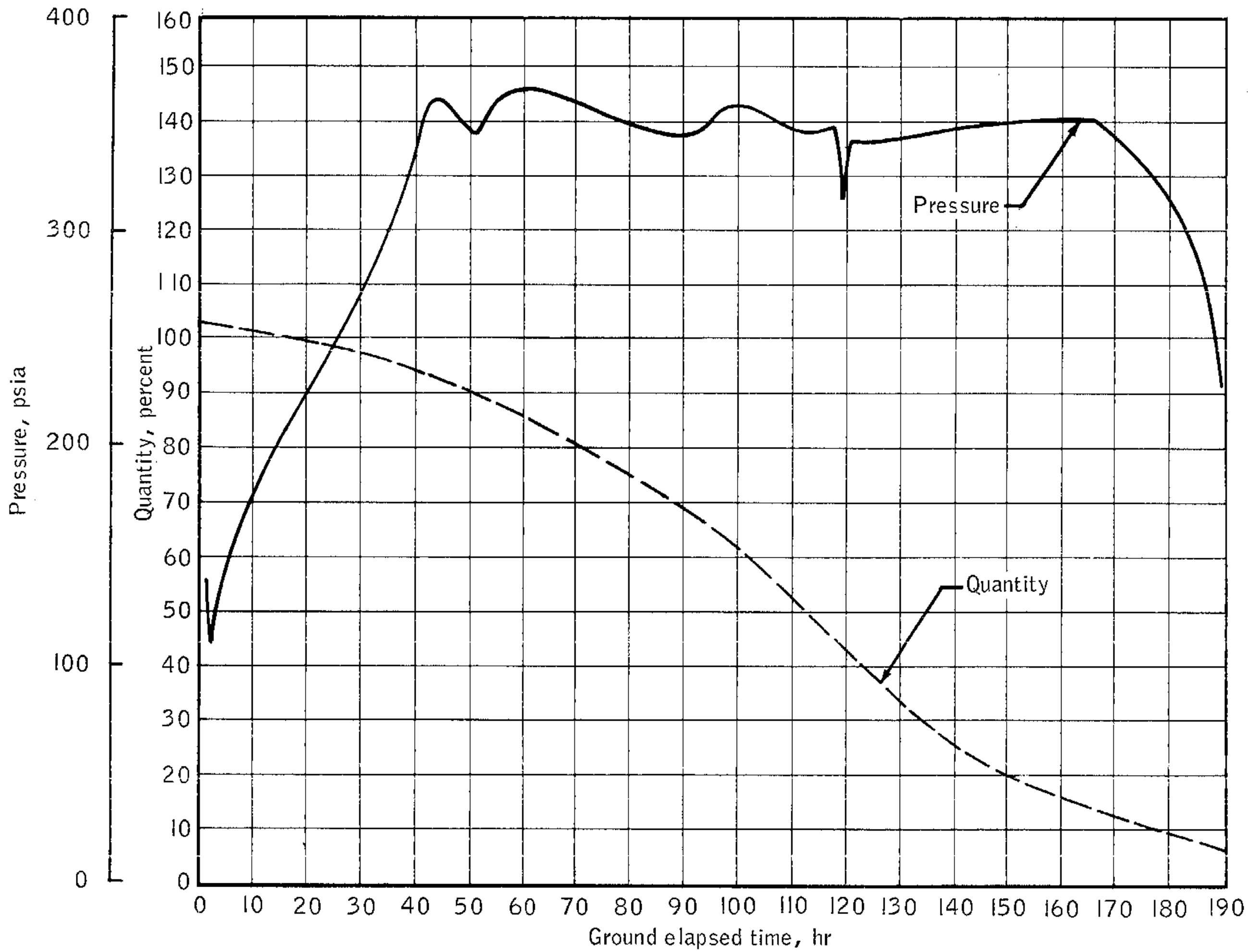

Figure 6.1-2. - Real - time data concerning the fuel-cell hydrogen supply.

UNCLASSIFIED

NASA-S-65-8577

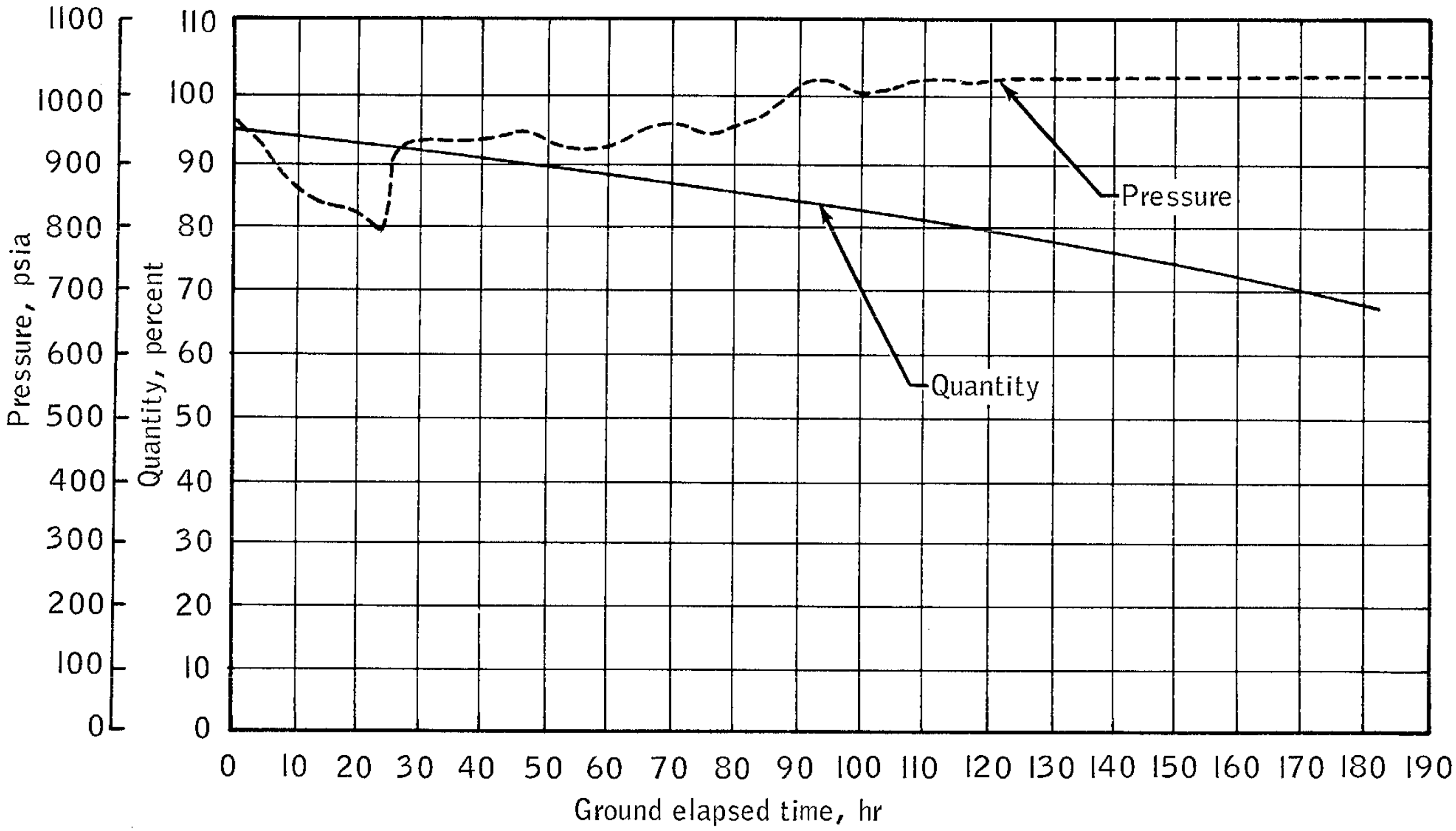

Figure 6.1-3. - Real-time data concerning the ECS primary oxygen supply.

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NASA- S-65-8578

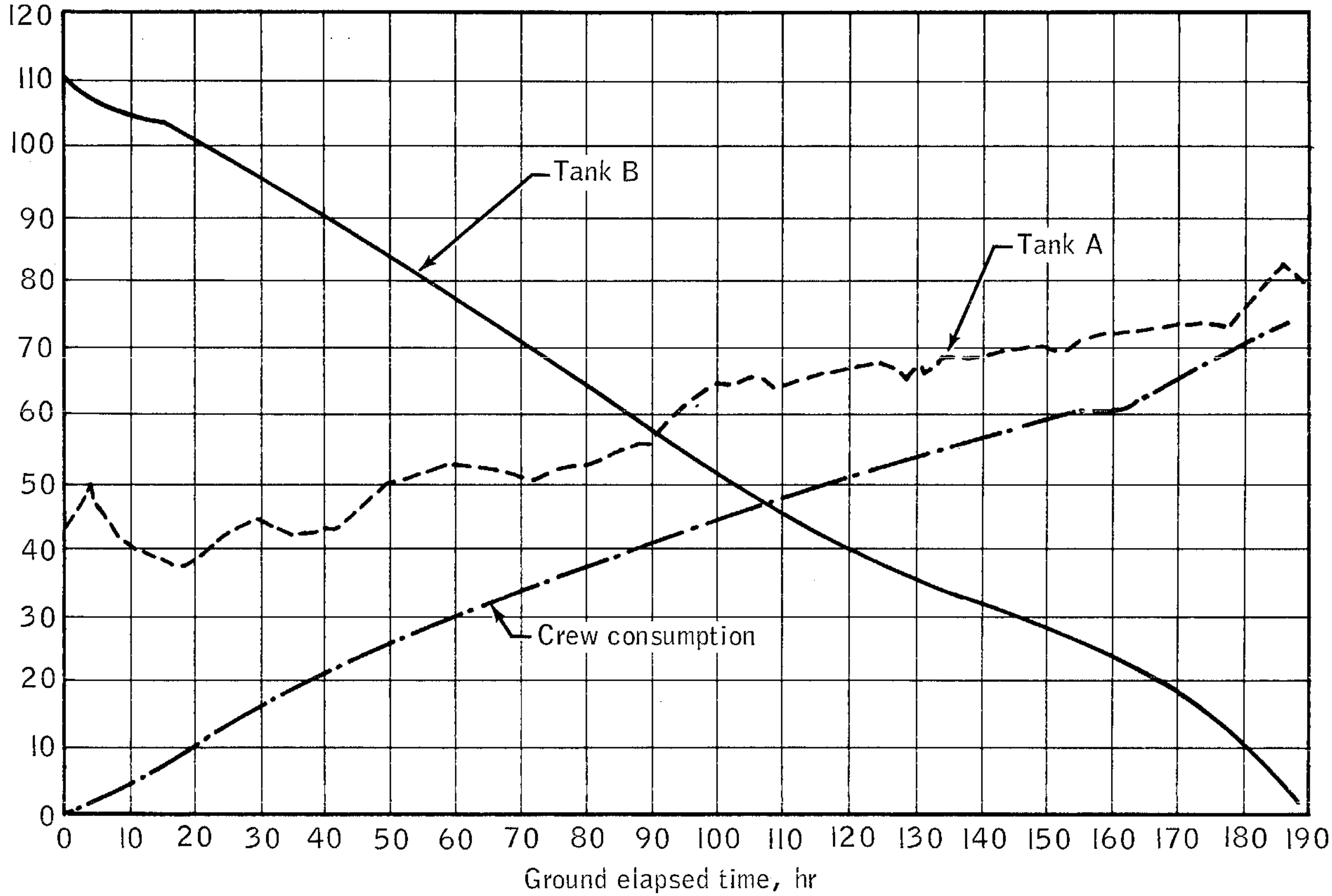

Figure 6.1-4.- Real - time data concerning the drinking water supply.

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6.2 NETWORK PERFORMANCE

The network was placed on mission status for Gemini V on August 4, 1965. The first launch attempt on August 19 was conducted with only minor network problems until T-25 minutes, at which time the Cape Kennedy complex experienced major problems because of an electrical storm. On the rescheduled launch day, August 21, the network was ready to support the mission at lift-off.

6.2.1 MCC and Remote Facilities

The network configuration and the general support required at each station are indicated in table 6.2-I. Table 6.2-II details the type of data collected and processed at each location. Figure 4.3-1(a) shows the network, and figure 6.2-1 illustrates the network complex at Cape Kennedy. In addition, approximately 14 aircraft provided supplementary photographic, weather, telemetry, and voice relay support in the launch and reentry areas. The U.S.S. Wheeling (WHE) was also considered supplementary support and was positioned near Midway Island for backup tracking and voice relay.

6.2.2 Network Facilities

Performance of the network is reported on a negative basis by system and site. All performance not detailed in this report was satisfactory.

6.2.2.1 Remote sites.-

6.2.2.1.1 Telemetry: The telemetry ground stations supporting the mission had no equipment failures to cause total loss of real-time data for any one revolution. California, however, did lose postflight evaluation data for one pass because a magnetic tape recorder was not turned on. Several incidents such as radio frequency interference (RFI), patching errors, and spacecraft recorder problems caused data losses and dropouts.

6.2.2.1.2 Radar: With the exception of two problems associated with the Carnarvon (CRO) radar data and several failures on the U.S.S. Wheeling (WHE), the tracking performance of the network radar was satisfactory. Of the 18 reported radar failures, only 4 resulted in a loss of data.

Good skin tracking of the launch vehicle was obtained by all scheduled network sites throughout its orbital lifetime of 48 revolutions.

Although attempts were made to track the rendezvous evaluation pod (REP) and the kite-shaped cover plate, skin track acquisition was not successful.

During the first day, it was discovered that the CRO data contained a bias, and the data were manually rejected at MCC-H. MCC-H computers were expecting CRO data with a slightly different calibration. The difference in the calibration would have resulted in an error of approximately 138 yards when tracking at a range of 1000 nautical miles. A decision was made to reprogram the 4101 computer at CRO so that all range radar data would be compatible with the MCC-H programs. At 66 hours 55 minutes g.e.t., the program changes were made and checked out. The CRO data were then accepted by the MCC-H computers.

During revolution 78, the WHE radar failed and was not operational for the remainder of the mission. The problems reported with this radar included failures of slip rings, hydraulic pumps, preamplifiers, and a reference voltage supply.

The spacecraft reentry assembly C-band transponder was turned on for the last three revolutions. All stations tracked the spacecraft and supplied data for an orbit computation prior to reentry. The CRO data were manually rejected on each pass because Goddard Space Flight Center (GSFC) computations indicated an intermittent angle bias error. Retrofire was computed without using CRO data. This problem is under study by GSFC.

6.2.2.1.3 Acquisition aids and timing: There were several minor failures during the mission, none of which were significant.

6.2.2.1.4 Command: The command support of the mission prelaunch and orbital phases demonstrated a relatively high reliability. Approximately 33 discrepancies were reported throughout the mission, and only two items resulted in a loss of command support during scheduled passes. A total of three passes was affected by these two items. The remainder of the discrepancies were either identified and corrected between passes, or involved a redundant system, and therefore did not affect mission support.

The most significant command discrepancy noted during prelaunch activities was the sporadic transmission of the auxiliary stage II cutoff (ASCO) function at Cape Kennedy when the MCC-H master digital command system (MDCS) output to the Cape was inhibited. The MDCS inhibit was accomplished by opening the command data line at Houston. This resulted in noise at the input to the MCC-C data routing and error detection (DRED) equipment and the noise was randomly recognized as an ASCO command as well as other valid commands. The decision was made to inhibit the ASCO tone at the Cape. This was accomplished by powering down the tone transmitter.

During the fourth pass over Texas (TEX), an attempt was made to update the spacecraft computer with revolution 6, area 4 reentry information. The DCS commands were transmitted too fast to be accepted by the spacecraft computer. This has been determined not to be a MDCS problem. The telemetry ground station at TEX lost synchronization. The remote site data processor (RSDP) continued to send the last valid data frame, which contained a valid message acceptance pulse (MAP), and continued to recycle this transmission until synchronization was regained. Therefore, when the preretrofire update load was transmitted, it was validated by a MAP contained in the recycled data frame rather than by a spacecraft-transmitted MAP. Consequently, the rate at which commands were transmitted did not include the normal uplink and downlink delay between command transmission and validation.

During revolutions 51 and 52, the Coastal Sentry Quebec (CSQ) digital command system (DCS) was not able to support the mission. Several modules were destroyed due to application of improper voltages. The DCS cables were damaged by ship vibrations and as a result of repeated opening and closing of the DCS drawers. The shorted and open cable wires were repaired and operation was restored.

During revolution 79, the CSQ digital command system RF command did not function. A circuit breaker had malfunctioned (internal short to an arc shield) and caused a loss of power. The arc shield was tightened and the burned contacts were cleaned and polished. The circuit breaker performed normally for the remainder of the mission.

6.2.2.1.5 Missile trajectory measurement (MISTRAM) system: The MISTRAM system supported the launch with no significant problems.

6.2.2.2 Computing.-

6.2.2.2.1 MSC computers: Computer personnel experienced approximately 20 operational problems during the mission. Most of these were associated with programing and interfaces with other systems. Procedures have improved, and the number of problems of this type should decrease in the future. At no time was MCC-H without the required real-time computer support.

On August 23, both the mission operational computer (MOC) and dynamic standby computer (DSC) were inoperative from 57 hours 25 minutes g.e.t. until 57 hours 36 minutes g.e.t. This situation was initiated by the entry of an illegal code at the checkout console. The operator did not realize that the code entered was illegal, and the executive program did not recognize and reject the illegal code. The computer program failed in attempting to implement the coded instruction. A third computer was placed on-line. During the 11-minute period following, the required programs were reloaded, all necessary data were replayed from tape in the communications area, and the computers were returned to their normal condition. The cause of the problem is currently under investigation. The mission program will be changed to reject all illegal codes, and operators will be instructed further on legal and illegal codes.

On August 29 at 189 hours 23 minutes g.e.t., the MOC recommended a 5-second change in retrofire time for revolution 121, zone 1 landing area. Prior to this time, the retrofire time had been fairly constant for several revolutions. Basically, the problem resulted from a lack of data on a segment of orbit over CRO. Without data from this orbit segment, the RTCC program required approximately three revolutions of orbital data over the United States to determine velocity accurately. Had CRO data been used, the orbit determination would have been more accurate. The reasons for not using CRO data are many, but basically the data did not appear to fit with the other tracking data. This apparent misfit was due to an improper weighting of a new station in the differential correction program and a possible erroneous residual summary display. Also, on the first day of the mission, the data were not usable because of a difference in the range granularity constant used by CRO and the RTCC. On the seventh day, GSFC reported a possible 100 to 200 millisecond bias in the CRO data. Postflight replay of all data from the last day of the mission shows the CRO data did contribute to a good solution, and that the final revolution 121, area 1, retrofire time was correct. The orbit determination is currently being reviewed. A change will be implemented to adjust the weighting of new stations, and the residual summary display is being reviewed.

On August 29, at 190 hours 50 minutes g.e.t., the MOC and DSC went into a constant on-line print condition, when a manual entry was made. This appears to be a program problem as the entry was legal. A third machine was put on-line in the event the print condition caused the machine to fail. A second entry was made which caused the MOC and DSC to recover.

It was discovered postreentry that a part of the preretrofire update calculation was in error by approximately 7.89°. This quantity, which references the longitudinal (X_e) axis of the spacecraft coordinate system to Greenwich, was computed incorrectly in the real-time program at MCC-H. The net result of this error was that at retrofire the spacecraft computer was instructed that the spacecraft was at 187.44° west from Greenwich when, in fact, it was at 195.33° west. Thus, to the onboard computer, the spacecraft appeared to be overshooting the target, and the computer displayed corrections for the situation, resulting in a zero-lift indication and an actual undershoot. The MCC-H computer program was correct as written. An earth's rotation rate of 360.98° per day is used in the program which requires that the total elapsed time from G.m.t. midnight, prior to launch, be inserted to derive the spacecraft present position relative to the earth. The 7.89° error resulted from an omission of the elapsed number of days in the G.m.t. of retrofire term. Subsequent Gemini missions will have this calculation checked by an off-line computer which will use the pre-retrofire IGS update and spacecraft trajectory data to calculate a landing point for correlation and validation.

6.2.2.2.2 Remote site data processors (RSDP): The RSDP's performed very well during the mission. The Coastal Sentry Quebec, Rose Knot Victor, Bermuda, and Texas sites experienced some temporary difficulties in generating pulse code modulation (PCM) summary messages. Hardware and procedural problems contributed to these failures.

6.2.2.2.3 GSFC computing: The GSFC real-time computing center supported the mission with no malfunctions. No problems were encountered in generating skin track pointing data for the launch vehicle. The second stage impact point computed by Goddard was:

Date	August 24, 1965
Time, G.m.t.	17:03:10
Revolution	48
South latitude	20.499°
East longitude	117.0691°

The data received from the North American Air Defense Command (NORAD) on the first day track of the REP appeared to contain track of more than one object; therefore, it was not possible to obtain a reliable vector. As the mission progressed, a more reliable vector was obtained but never with the confidence associated with vectors computed for the spacecraft or launch vehicle.

6.2.2.3 Communications.- Communications for the mission were exceptionally effective, both point-to-point and air-to-ground.

6.2.2.3.1 Ground communications: During revolution 2, a cable was cut at Oxford, Alabama, which affected both Houston-Cape circuits and GSFC-TEX circuits. A temporary restoration was made on microwave and all circuits were returned to normal service after 3 hours. All three of the DOD network circuits were plagued with echo problems and at times were unusable.

During the launch attempt on August 19, 1965 the MCC-C was virtually isolated due to a manhole fire in the Department of Defense (DOD) facility at Cape Kennedy. The first outage was noted at 1930 G.m.t. and the last circuit was turned up for service at 1600 G.m.t. on August 20, 1965.

The communications processor at MCC-H had four outages during the mission which, because of system redundancy, caused no loss of data. The cause for these outages was a drum memory lockout resulting from a pulse-shaped circuit that was sensitive to a particular data pattern. On August 24, 1965, 11 incoming messages and 1 outgoing message were delayed for approximately 2 minutes. All messages were recovered and forwarded without any loss of data. There were a total of 46 097 messages processed by the communications processor.

6.2.2.3.2 Air-to-ground: Spacecraft communication during the mission was generally reported satisfactory with improvement over that of the previous missions. Actual failure to communicate with the spacecraft seldom occurred and these few times were due to atmospheric conditions interrupting ground communication links and in most cases were predicted. Two ground HF transmitters failed temporarily during the mission with no significant loss of support.

6.2.2.3.3 Frequency interference: Partially because of the length of the mission, a large number of interference reports were generated. They are approximated in the following table.

Location	C-band radar	Rendezvous radar	UHF voice	HF	Spacecraft HF	Telemetry	Command	Support communication
ETR	2	2	2		24	9	1	3
WTR					2	1		
TEX						9		
GYM						1		
CAL						1		
EGL						1		
CYI						1		
HAW					3			
TF130					3			
WHS					2			
ASC					1			
USN-San Diego					1			
Ft. Hauchuca					10			

HF interference was not always identified because it either disappeared in a short time or did not present any major problem. The Cape FPS-8 radar caused rendezvous radar interference at 1428 Mc; some 1528 Mc interference was not identified, and it disappeared before aircraft could pinpoint the location.

Radio frequency interference (RFI) was not considered excessive, and did not cause a significant loss of support. It did degrade telemetry data in some instances and was a potential threat to mission success.

TABLE 6.2-I.- GEMINI V NETWORK CONFIGURATION

	C-band radar	SPANDAR	R and R TM	Real time TM display	Delayed time telemetry	TM experiment R and R	TM experiment display	High speed telemetry	GLDS	Remote site data processor summary	GLV telemetry	GLV command	Digital command system	Downrange uplink	Data routing and error detection	RF command	Voice (SCAMA)	Teletype (NASCOM)	Horizon sensor radar data	FC manned	Acquisition aid	TM RCV antenna	FC, A/G	Air-to-ground remoting	MISTRAM
MCC-H				X	X			X	X	X	X		(X)				X	X	X	X			X	X	
MCC-C			X		X	X		40.8	X	X	X		X		X		X	X			X	X		X	
MILA	X																		X						
CNV			X								X	X		X		X			X			X		X	
PAT	X																		X						
GBI	X		X		X			40.8			[X]	X		X		X			X			X		X	
GTI	X		X		X			40.8			[X]	X		X		X			X		X	X		X	
BDA	X		X		X			2.0			[X]			X		X	X	X	X		X			X	
CYI	X		X	X	X					X			X			X	X	X		X	X		X	X	
KNO			X			X											X	X			X			X	
TAN			X														X	X			X			X	
CRO	X		X	X	X	X	X			X			X			X	X	X		X	X		X	X	
CTN			X														X	X			X			X	
HAW	X		X	X	X	X				X			X			X	X	X		X	X		X	X	
GYM			X	X	X	X				X							X	X		X	X		X	X	
CAL	X		X			X											X	X			X			X	
TEX			X	X	X	X		2.0		X			X	X		X	X	X			X			X	
WHS	X																X	X			X				
EGL	X																X	X			X				
ANT	X		X		X			40.8						X		X			X			X		X	
ASC	X		X			X													X			X		X	
CSQ			X	X	X					X			X			X	X	X		X	X		X	X	
RKV			X	X	X					X			X			X	X	X		X	X		X	X	
WHE	X		X														X	X			X			X	
A/C			X		X																			X	
WLP		X															X	X							
PRE	X																		X						
VAL																									X
ELU																									X

(X) -Master DCS [X] -Record only

Ship positions: RKV - 21°S 85°W; CSQ - 21°N 125°E; WHE - 29°N 175°W

TABLE 6.2-II.- DATA AVAILABILITY

Data category	Station																												
	MCC-C	MCC-H	CKV	MLA	PAFB	VAL	GBI	ELU	GTK	ANT	ASC	PRE	BDA	CYI	KNO	TAN	CSQ	CRO	I/AC	CTN	HAW	WHE	RKV		GYM	WHS	TEX	EGL	WLP	GSFC
Telemetry																														
Spacecraft real-time PCM record - magnetic tape	X		X				X		X	X	X		X	X	X	X	X	X	X	X	X	X	X	X	X		X			
Spacecraft delayed-time PCM record - magnetic tape	X^1		X				X^1		X^1	X^1			X^1	X^1			X^1	X^1			X^1		X^1		X^1		X^1			
Spacecraft real-time PCM decommutate - display	X	X					X^2		X^2	X^2			X^3	X			X	X			X		X		X		X^4			
Spacecraft real-time paper records	X		$X^{5,6}$																											
GLV FM/FM record - magnetic tape			$X^{5,6}$				X						X																	
GLV PCM record - magnetic tape			$X^{5,6}$				X						X																	
GLV PCM decommutate - display			$X^{5,6}$																											
GLV real-time paper records			X^5																											
Spacecraft high-speed telemetry data							40.8		40.8	40.8			2kb														2kb			
Telemetry operator's logs	X		X				X		X	X	X		X	X	X	X	X	X	X	X	X	X	X	X	X		X			
Telemetry signal strength record	X		X				X		X	X	X		X	X	X	X	X	X		X	X	X	X	X	X		X			
Data quality monitor record	X																													
150-channel events record	X												X	X			X	X			X		X		X		X			
100-channel events record													X	X			X	X			X		X		X		X			
Radar (pulse and CW)																														
C-band radar transponder track				X	X		X		X	X	X	X	X	X				X			X	X		X		X		X		
S-band radar skin track																													X^7	
Tape recording				X	X		X	X	X	X	X	X	X	X				X			X	X		X		X		X	X	
Event record				X	X		X		X	X	X	X	X	X				X			X	X		X		X		X	X	
Function record				X	X		X	X	X	X	X	X	X	X				X			X	X		X		X		X	X	
Plotboard charts	X			X	X																									
Radar operator's logs													X	X				X			X	X		X	X			X	X	
Data sheet					X		X	X	X	X	X	X																		
Acquisition system																														
Acquisition aid record	X								X				X	X	X	X	X	X		X	X			X	X	X	X	X		
Acquisition aid operator's logs													X	X	X	X	X	X		X	X			X	X	X	X	X		
Remote site data processor																														
Teletype RO printout	X												X	X			X	X			X		X		X		X			
Telemetry PCM summary	X												X	X			X	X			X		X		X		X			
Air-to-ground communications (UHF and HF)																														
Magnetic tape	X						X^8		X^8	X^8	X^8		X^8	X	X^8	X^8	X	X	X	X^8	X	X^8	X	X^8	X		X			
S/S record and operator's logs	X						X^8		X^8	X^8	X^8		X^8	X	X^8	X^8	X	X		X^8	X	X^8	X	X^8	X		X			
Command																														
DCS command	X													X			X	X			X		X				X			
Tone command	X						X^9		X^9																					
DRUL			X^{10}				X^{10}		X^{10}				X^{10}																	
30-channel events record	X																													
Teletype																														
Summary messages and/or reports	X	X	X				X		X	X	X		X	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X	
Optics																														
Metric camera film			X																											
Cine fixed engineering sequential film			X																											
Cine tracking engineering sequential film			X																											

See notes on page 6-20

TABLE 6.2-II.- DATA AVAILABILITY - Concluded

Data category	Station																													
	MCC-C	MCC-H	CNV	MLA	PATB	VAL	GBI	[illegible]	GTK	ANT	ASC	PRE	BDA	CYI	IMO	TAN	CSQ	CRO	I/AS	[illegible]	[illegible]	[illegible]	RKV	CAL	GYM	WHS	TEX	MGL	MLF	GSFC
Computer support																														
Univac 1218 computer	X[11]						X[12]		X[12]	X[12]			X[13]	X[14]			X[14]	X[14]			X[14]		X[14]		X[14]		X[15]			
Other computers		X	X[16]																											X
Reduced data (tabular)																														
Position, velocity, and acceleration			X		X																									
Special and aerodynamic parameters			X		X																									
Attitude					X																									
BET					X																									
ETR Flight Test Report					X																									
Spacecraft (onboard recorder)																														
Magnetic tape			X																											
Engineering photographic data		X																												
Other test data																														
Meteorological data			X																											

NOTES

1. Real-time dump recording and post-pass dump data reduction.
2. Spacecraft real-time PCM remoted to MCC-H via MCC-C.
3. Remoted to MCC-H via MCC-C.
4. Remoted to MCC-H only.
5. Cape telemetry no. 2.
6. Cape telemetry no. 3.
7. SPANDAR will skin track the launch vehicle.
8. HF and UHF air-to-ground communications remoted from MCC-H only (no CapCom on station).
9. Tone command (FIDO-generated ASCO) remoted from MCC-C only.
10. DCS command remoted from MCC-H or MCC-C.
11. Three Univac 1218 computers utilized as follows: 2 computers (1 primary and 1 backup) to format 40.8-kbps data for transmission to MCC-H and 1 computer as RSDP for summary message generation.
12. 1 Univac 1218 computer to format 40.8-kbps data for transmission to MCC-H and MCC-C via AFETR subcable.
13. 1 Univac 1218 computer to format 2-kbps data for transmission to MCC-H and MCC-C.
14. 1 Univac 1218 computer as RSDP for summary message generation.
15. 1 Univac 1218 computer to format 2-kbps data for transmission to MCC-H.
16. Range safety plotboards in central control.

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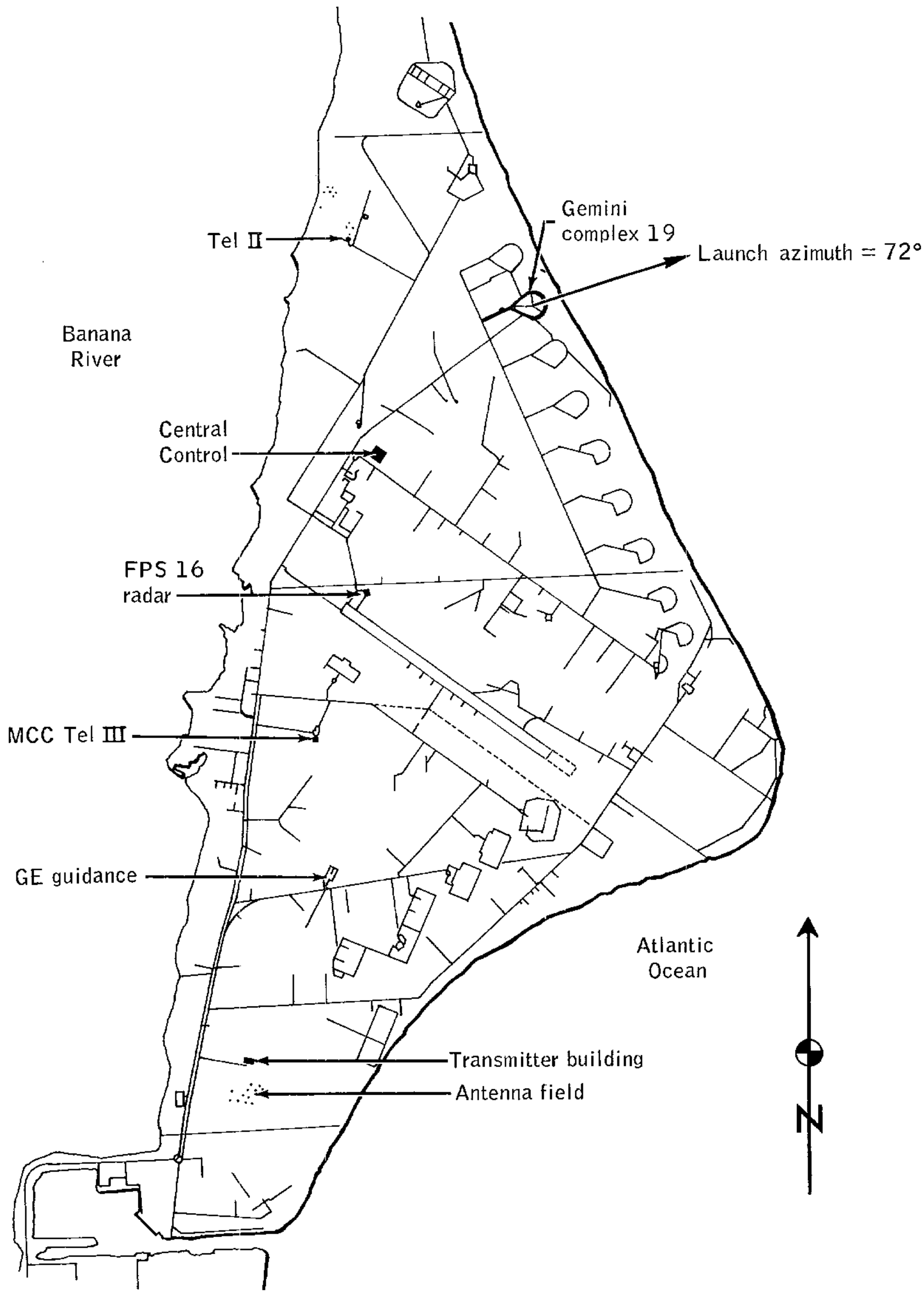

Figure 6.2-1. - Cape Kennedy Air Force Eastern Test Range network stations.

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6.3 RECOVERY OPERATIONS

6.3.1 Recovery Force Deployment

The four categories of planned landing areas designated for the Gemini V mission were:

(a) Primary landing areas (supported by an aircraft carrier and located in the West Atlantic zone).

(b) Secondary landing areas (East Atlantic, West Pacific, Mid-Pacific, and areas within the West Atlantic zone not supported by the aircraft carrier).

(c) Launch site landing area.

(d) Launch abort landing areas.

Data concerning the deployment of ships and aircraft in planned landing areas are provided in table 6.3-I. Figure 6.3-1 shows the deployment of ships and aircraft in the launch abort landing areas. The four worldwide landing zones are illustrated in figure 6.3-2, and the ship support provided for each of the numbered landing areas is listed in table 6.3-I.

The recovery forces were assigned positions in these areas so that any point in a particular area could be reached within a specified access time. The ship and aircraft access times, which varied for the different areas, were based upon the probability of the spacecraft landing within a given area and the amount of recovery support provided in that area.

Twelve ships, 59 fixed-wing aircraft, 10 helicopters, and various special vehicles were positioned for support of the planned landing areas. Forty-nine of the aircraft, with pararescue teams aboard, were deployed around the world on strip alert to provide contingency recovery support and support in the zones described in the preceding paragraphs.

Normal operational contingents of Department of Defense (DOD) ships and aircraft were used for recovery support. Special equipment, such as retrieval cranes, airborne UHF electronic receivers (homing systems), spacecraft flotation collars, and swimmer interphones, was furnished to the DOD by NASA. All aircraft providing contingency and secondary landing-area support carried pararescue teams ready to drop to the spacecraft, install a spacecraft flotation collar, and render assistance to the flight crew. Twin turbine helicopters (type SH-3A) launched from the aircraft carrier provided location support and were used to transport swimmer teams, flotation collars, and photographers to the landing point. Fixed-wing aircraft from the carrier were utilized for communication relay and to transport the "on-scene commander" to the landing point.

6.3.2 Location and Retrieval

The MCC-Recovery Control Center informed all recovery forces of flight progress throughout the mission. As the orbital ground tracks shifted during the mission, possible landing points were passed to all forces, and the position of the recovery ships and aircraft were altered accordingly. During the third revolution, as a result of the fuel-cell oxygen supply pressure problem, recovery forces were alerted and aircraft deployed in the mid-pacific area, zone 4, to reduce access time in the event of an early mission termination. Late in the mission, the decision was made to terminate the flight in landing area 121-1 (revolution 121, zone 1) rather than the planned 122-1 area, because of expected poor weather conditions associated with Hurricane Betsy. Recovery forces were notified of this decision and assumed positions as shown in figure 6.3-3.

On August 29 at 190 hours 31 minutes g.e.t., recovery forces were informed that retrofire was nominal. The destroyers U.S.S. DuPont and U.S.S. Waldron, positioned uprange from the aircraft carrier U.S.S. Lake Champlain, reported radar contact of the spacecraft after blackout and provided range and bearing information to the carrier and air elements, thus initiating movement of recovery forces toward the projected spacecraft landing point. The "on-scene commander", in an S-2F aircraft, received a short count from the Gemini V flight crew on request and obtained UHF-DF bearing information. At approximately 191 hours 16 minutes g.e.t., the on-scene commander reported visual sighting of the spacecraft located approximately 91 nautical miles from the carrier. An Air Rescue aircraft (HC-97) from an uprange position was vectored to the spacecraft and reported over the spacecraft at 191 hours 27 minutes g.e.t. It was decided not to deploy a pararescue team but instead to await the arrival of swimmers aboard helicopters enroute from the carrier. A swimmer team was deployed at 191 hours 38 minutes g.e.t. The spacecraft flotation collar was installed and inflated by 191 hours 45 minutes g.e.t. Voice contact with the flight crew was established by a swimmer using the swimmer interphone. The crew members egressed through the left hatch and were taken aboard the helicopter at 191 hours 58 minutes g.e.t. for transportation to the aircraft carrier. The recovery helicopter landed aboard the carrier (U.S.S. Lake Champlain) at 192 hours 26 minutes g.e.t. The carrier retrieved the spacecraft at 194 hours 50 minutes g.e.t. The position of the spacecraft at pickup was 29°52.5' N., 69°50.8' W., approximately 7 nautical miles from where it had landed (29°47' N., 69°45.4' W.) at 190 hours 55 minutes g.e.t.

Recovery forces in the landing area reported no visual sighting of the main parachute nor the rendezvous and recovery (R and R) section.

6.3.3 Recovery Aids

6.3.3.1 UHF recovery beacon.- Signals from the spacecraft recovery beacon were received by the various aircraft as follows:

Aircraft	Initial time of contact, hr:min g.e.t	Range, n. mi.	Receiver	Mode
Search 1 (SH-3A)	191:24	80	SPP	Pulse CW
Search 2 (SH-3A)	191:03	113	SPP	Pulse CW
Search 3 (SH-3A)	191:00	70	SPP	Pulse
Rescue 1 (HC-97)	191:00	95	SPP	Pulse CW
Relay aircraft (EA-1F)	191:00	87	ECM	Pulse

No aircraft reported reception of recovery beacon transmissions prior to spacecraft landing time. Search 1 was not airborne at spacecraft landing time because of a fuel leak. Search 3 assumed the up-range on-station position originally assigned to Search 1.

6.3.3.2 HF transmitter.- The HF antenna was not erected and HF transmissions were not attempted or reported.

6.3.3.3 UHF transmitter.- UHF voice transmissions were received by aircraft as follows:

Aircraft	Time of contact, hr:min g.e.t.	Range, n. mi.	Receiver
Air Boss (S-2F)	190:54	60	ARA-25
Photo 2 (SH-3A)	190:53	90	ARA-25

Frequent attempts were made from aircraft in the area to communicate with the flight crew on UHF after landing; however, the aircraft reported no response from the spacecraft. Voice tapes recorded at the MCC-H confirm that transmissions were made from the spacecraft as late as 14 minutes after landing.

6.3.3.4 UHF survival radio (voice and CW, 244.0 Mc).- This system was not used.

6.3.3.5 Flashing light.- The spacecraft flashing light erected properly and was activated after landing. Aircraft in the area did not report sighting the flashing light; however, the deployed swimmer team noted that it operated normally. The aircraft carrier reported a sighting range of approximately 500 yards.

6.3.3.6 Fluorescent sea marker.- The sea-dye marker diffusion appeared normal and was observed by all recovery ships and aircraft in the landing area. The maximum range reported was 15 nautical miles from an aircraft at an altitude of 15 000 feet. Dye was still being emitted in small quantities at the time of spacecraft retrieval.

6.3.4 Postretrieval Procedures

Spacecraft postretrieval procedures were performed as specified in references 7 and 8. All onboard film and certain equipment were expedited to Cape Kennedy and Houston by special flights from the carrier.

Visual inspection of the spacecraft disclosed no excessive heating effects. Other observations include the following:

(a) The heat shield appeared very similar to other recovered Gemini spacecraft. Two relatively deep gouges in the heat shield were noted. The cause of the gouges has not been determined.

(b) Both windows contained some moisture between the glass layers, except at the periphery. Protective covers were placed over the windows by recovery personnel.

(c) The hoist loop door functioned properly, as did the recovery light, which was operating when the spacecraft was hoisted aboard the carrier.

(d) Upon retrieval it was noted that the left-hand spacecraft hatch was closed, but in the unlocked position. The right-hand hatch was locked and a torque of 275 in-lb was required to open it. Both hatch seals appeared to be in excellent condition.

(e) The spacecraft interior was exceptionally clean and all equipment was stowed.

(f) All spacecraft power was off with the exception of the recovery light.

(g) Safety pins had been installed in the drogue mortar and the ejection seat D-ring was in the stowed position.

(h) Moisture was noted in both the left and right footwells.

At 1500 G.m.t. on August 30, 1965, the day after recovery, the flight crew departed the U.S.S. Lake Champlain and flew to Cape Kennedy. The spacecraft was off-loaded at Mayport Naval Station, Florida, at 1430 G.m.t., August 30, 1965.

6.3.5 Reentry Control System Deactivation

After the spacecraft was unloaded from the carrier U.S.S. Lake Champlain at Mayport, it was transported by dolly to a previously selected, well-isolated area where deactivation was begun at 3:00 p.m. e.s.t., August 30, 1965, and completed by midnight the same day. Upon receipt of the spacecraft, there was no visual indication of toxic vapors from any of the reentry control system (RCS) thrust chamber assemblies. The RCS shingles had been removed previously onboard the carrier by contractor personnel.

Before the pressurant in each ring was relieved to atmospheric pressure, source pressure and regulated lock-up pressure were measured. Source pressure readings of 1190 psig and 1560 psig (ambient dry bulb temperature of 76° F) were obtained from rings A and B, respectively. Regulator lock-up pressure readings of 285 psig from ring A and 290 psig from ring B were obtained. The pressures in each ring were then relieved to atmospheric pressure. Immediately following the source pressurant draining operation, the pressurant upstream of the propellant bladders and downstream of the system check valves was relieved by venting through separate scrubber units.

Following the above operations, nitrogen pressure of 50 psig was utilized to force the remaining usable propellants of both rings into the proper propellant holding containers. When these steps were accomplished, the propellant motorized valves were still in the closed position so that propellant loss would be minimized. The propellant solenoid valves did not leak vapors or flush-fluids at any time. All the RCS valves appeared to function normally.

TABLE 6.3-I.- RECOVERY SUPPORT

Landing area	Access time, hr		Support
	Aircraft	Ship	
Launch site:			
Pad			4 LARC (amphibious vehicle) 1 LCU (large landing craft) with spacecraft retrieval capabilities
Land	10 min		2 LVTR (amphibious vehicle) with spacecraft retrieval capabilities
Water (ejected)	2 min		3 M-113 (tracked land vehicles)
Water (spacecraft)	15 min		4 CH-3C (helicopters) (3 with rescue teams) 2 MSO (mine sweepers) with salvage capabilities 1 ATF (deep water salvage ship) with spacecraft retrieval capabilities 2 boats (50 ft) with water salvage teams
Launch abort:			
A			1 CVS (aircraft carrier) with onboard aircraft capabilities, 4 DD (destroyers), 1 AO (oiler), and 5 aircraft on station (2 HC-97 and 3 HC-54) (See fig. 6.3-1)
B	3	3	
C	3	14	
D	3	14	
Primary: West Atlantic (end-of-mission area 121-1)	1	4	1 CVS (aircraft carrier) from area A, station 3 1 DD (destroyer) assigned just prior to end of mission 1 DD from station 2 2 HC-97 (search and rescue) 5 JC-130 (3 telemetry and 2 communications relay) 6 SH-3A helicopters (3 location, 2 swimmer, and 1 photo) 2 S-2F (on-scene commander and backup) 2 EA-1F (Navy communications relay - 1 primary, 1 backup) 1 EA-1E (radar search)
Secondary landing areas:			
West Atlantic (zone 1)	5	6	1 CVS (carrier) from station 3 2 DD (destroyer), 1 from station 2 and 1 from station 4 on a rotating basis
East Atlantic (zone 2)	5	6	1 DD (destroyer) from station 7 and 1 AO (oiler) from station 6 on a rotating basis
West Pacific (zone 3)	5	12	1 DD (destroyer)
Mid-Pacific (zone 4)	5	6	1 DD (destroyer) and 1 AO (oiler)
Contingency	18		49 Aircraft on strip alert at worldwide staging bases
Total (including MSO's)			12 ships, 10 helicopters, 59 aircraft

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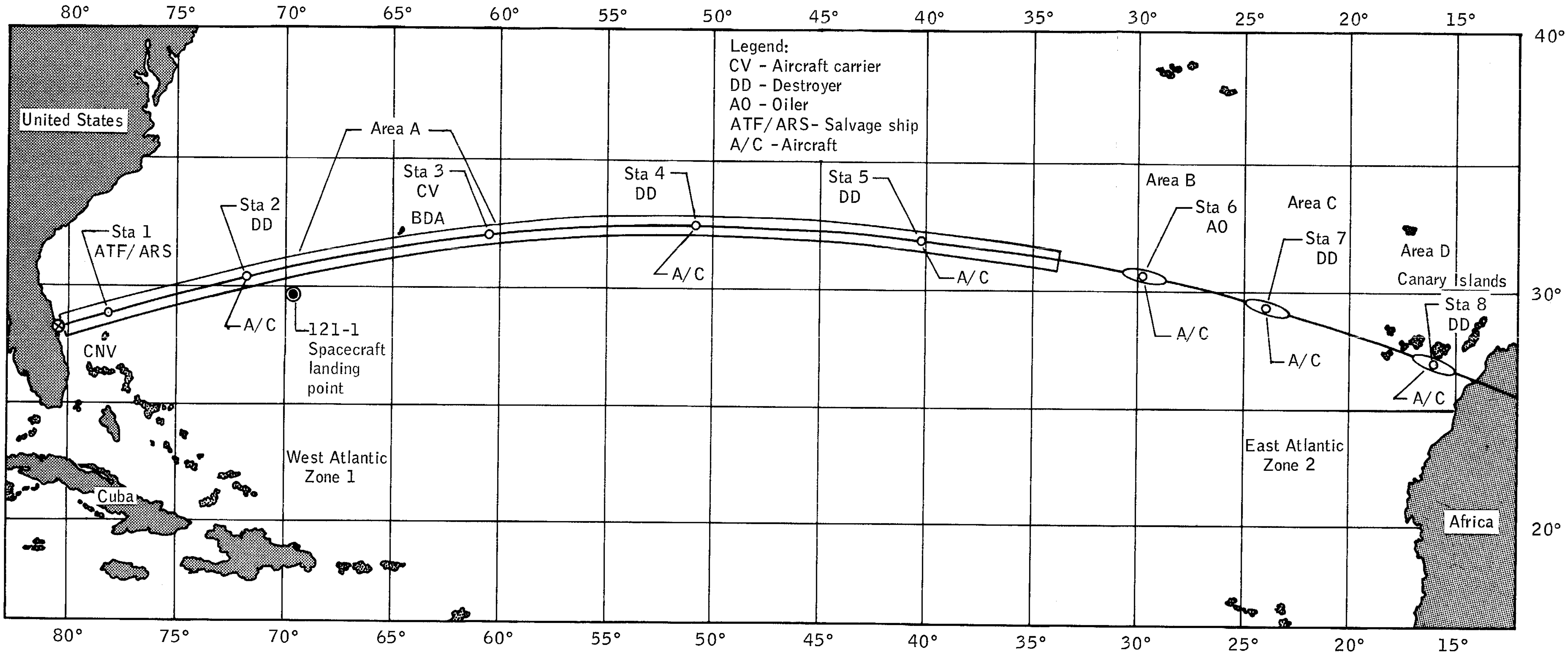

Figure 6.3-1.- Gemini V launch abort areas and recovery force deployment.

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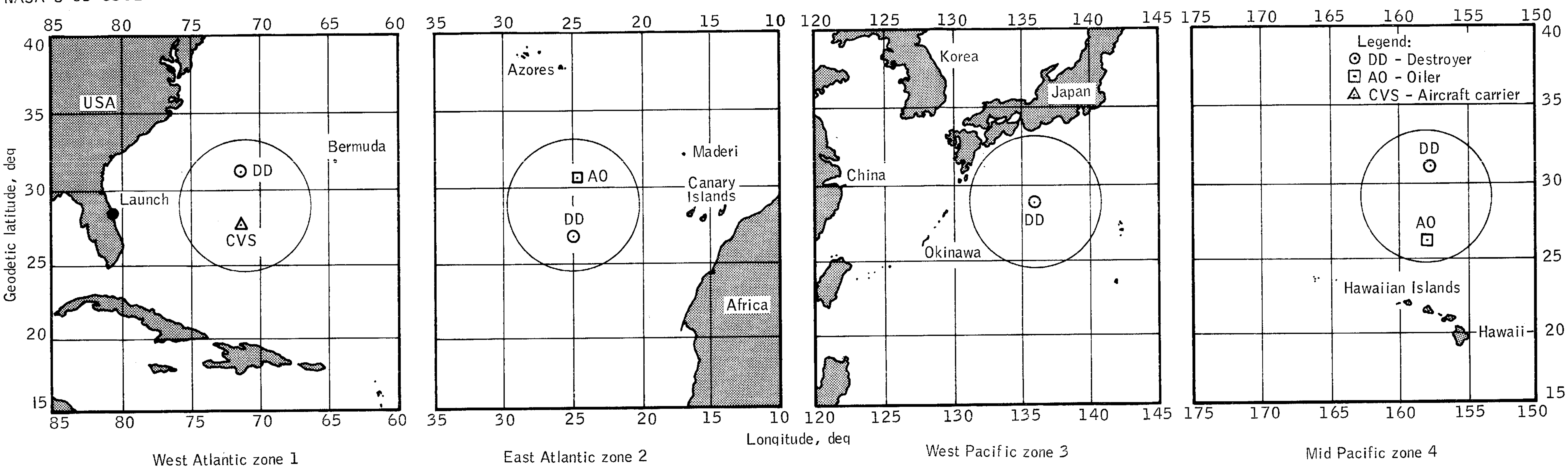

West Atlantic zone 1 — Landing area coverage for revolutions:

Primary (CVS)		Secondary (DD)	
2	62	3	76
15	74	16	90
18	77	17	91
30	89	31	105
33	92	32	106
45	107	46	119
47	121	61	120
60		75	

East Atlantic zone 2 — Landing area coverage for revolutions:

13	43	74	104
14	44	75	105
15	45	87	116
16	58	88	117
28	59	89	118
29	60	90	119
30	72	102	120
31	73	103	

West Pacific zone 3 — Landing area coverage for revolutions:

6	37	81
7	38	82
8	51	83
9	52	95
21	53	96
22	66	97
23	67	110
24	68	111
36	80	112

Mid Pacific zone 4 — Landing area coverage for revolutions:

3	35	79
4	48	80
5	49	92
6	50	93
18	62	94
19	63	95
20	64	107
21	65	108
33	77	109
34	78	

Figure 6.3-2. - Gemini V landing zone force deployment.

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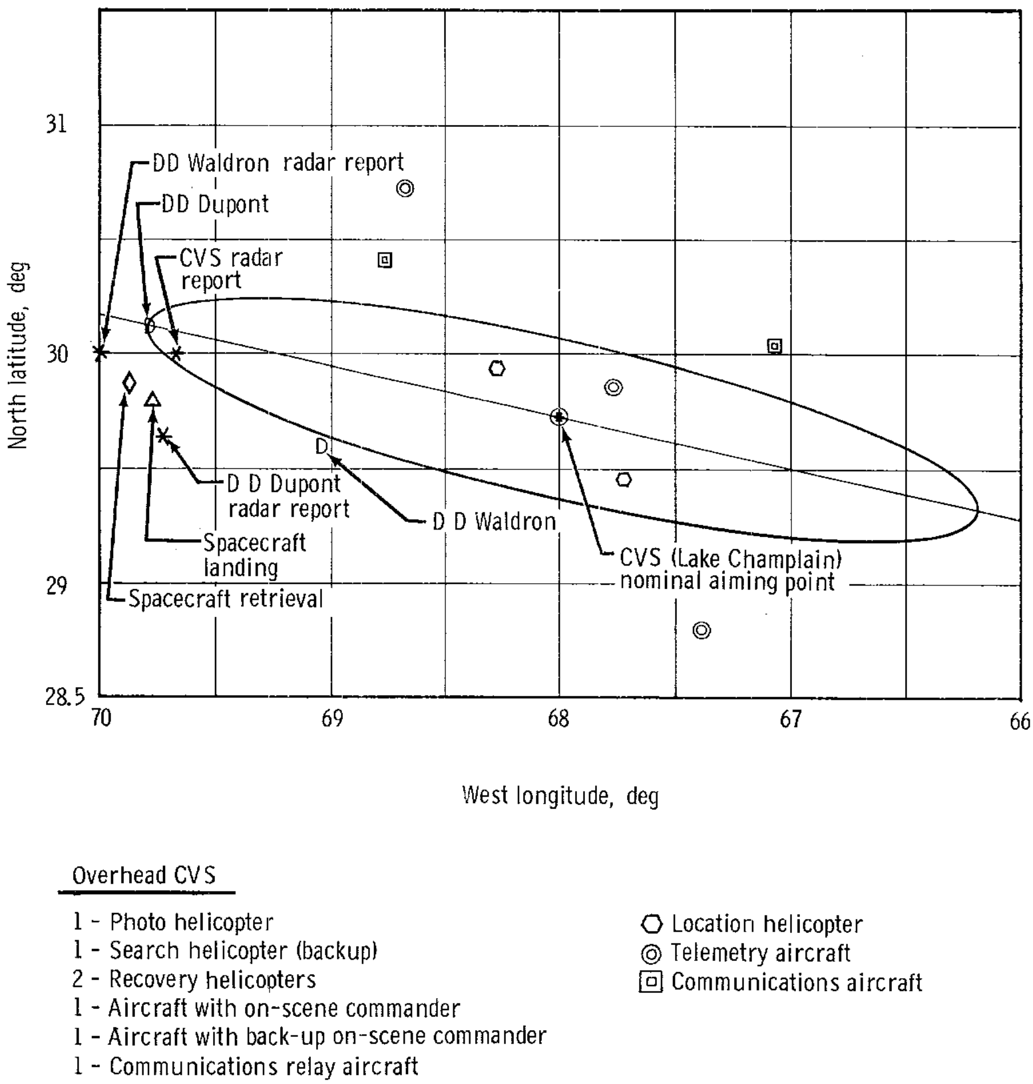

Figure 6.3-3. - Details of primary landing area.

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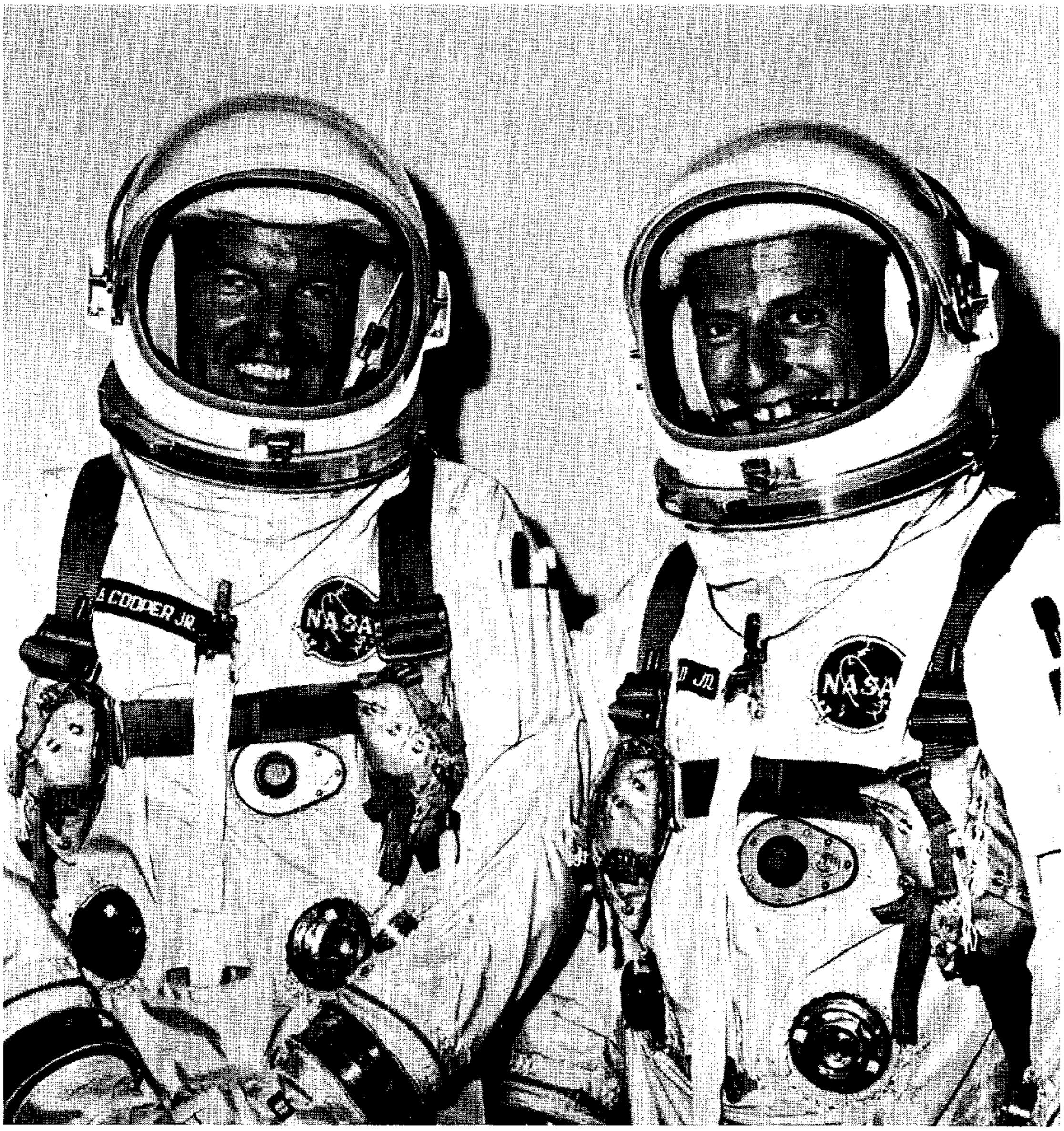

Astronaut L. Gordon Cooper, Jr., Command Pilot
and Astronaut Charles Conrad, Jr., Pilot.

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7.0 FLIGHT CREW

7.1 FLIGHT CREW PERFORMANCE

7.1.1 Crew Activities

The results of the flight indicate that the flight crew satisfactorily performed a series of experiments in addition to the required operational tasks over a period of 8 days within the confines of the Gemini spacecraft. The crew completed most of the major mission objectives in spite of systems problems requiring considerable modification to the flight plan and alteration of practiced flight procedures. Figure 7.1-1 represents the actual summary flight plan for this mission.

The crew correctly analyzed various systems problems and established appropriate alternate procedures. The rendezvous evaluation pod (REP) was ejected on time; however, a failure of the heater in the fuel-cell oxygen supply tank required termination of the REP exercise. The reentry was performed according to techniques established prior to the mission; however, an error in the use of the equation to calculate retrofire coordinates was entered in the ground computer and the effect of this error was extended into the airborne computer by way of the digital command system (DCS). This erroneous update resulted in an error of approximately 90 miles in the landing point. Crew training was adequate but more out-the-window simulation would have been beneficial. No significant physical problems were evidenced. The most significant crew problem emanating from this flight was that of routine crew-station housekeeping.

7.1.1.1 Prelaunch.- The crew entered the spacecraft at the proper time and all prelaunch crew functions were completed on time. The only prelaunch anomalies noted by the crew were: the command pilot's window had foreign matter between the panes, both windows fogged over completely during prelaunch but cleared up reasonably well by lift-off, and the OAMS fuel quantity indicated only 87 percent at lift-off.

7.1.1.2 Powered flight and insertion.- Powered flight was nominal except for three events:

(a) The launch vehicle instrument power supply (IPS) second-stage fuel-pressure indicator failed intermittently to full scale during launch. The pilot properly reported this to be an indicator malfunction. (See section 5.2.8.3.)

(b) Launch-vehicle longitudinal oscillations (POGO) occurred for a short period of time prior to staging and the pilot estimated the amplitude to be ±0.5g (actual amplitude ±0.38g). Talking and panel indicator reading were degraded during POGO (See section 5.2.1.1.)

(c) The no. 1 UHF radio appeared to become inoperative several seconds prior to insertion and the command pilot switched to the no. 2 backup radio before insertion. Radio communication on both UHF radios was excellent after insertion. (See section 5.1.2.1.)

This was the first flight during which the nose and horizon sensor fairings were jettisoned during the launch profile, and the nose fairing exploded and fell away from the spacecraft in many small pieces. (See section 5.1.1.3.) Five or six very small gray splotches hit the command pilot's window as these fairings were jettisoned, and remained there through reentry. As expected, the fuel-cell warning lights came on after lift-off, remained on throughout launch, and went out several seconds after insertion. (See section 5.1.7.1.5.)

Insertion was nominal. The spacecraft came off the launch vehicle with no noticeable angular rates. The pilot read nominal values for all insertion parameters from the inertial guidance system (IGS) before receiving the same information from the ground. The insertion checklist was performed according to the flight plan.

7.1.1.3 Operational checks.- The scheduled operational checks were completed according to the flight plan. These checks, in addition to control systems and platform procedures, are discussed in the following paragraphs.

7.1.1.3.1 Platform alinements: The platform was alined on numerous occasions during the flight, during both day and night conditions. At least 18 alinements were made in the small-end-forward (SEF) mode, and at least five were accomplished in the blunt-end-forward (BEF) mode. The crew attempted two or three platform alinements early in the mission by making use of the platform attitude control mode; however, they believed the mode to be out of tolerance in the yaw channel, and elected not to use it for subsequent alinements. During the critical alinements such as retrofire, or when adequate time was permitted to do a final alinement, final alinement errors were usually less than 1°. In a few cases, misalinements up to 5° occurred when the platform was alined quickly as the spacecraft drifted through the 0°, 0°, 0° position. The primary horizon sensor interrupted alinements during the early phases of the mission, although this was not immediately recognized because of the horizon sensor intermittent tendencies.

The crew preferred BEF alinements when alining with an earth reference, although the SEF alinements were within the same accuracies as the BEF alinements.

7.1.1.3.2 Use of spacecraft controls and displays: Except for periods when the spacecraft was powered down, the orbital attitude and maneuver system (OAMS) was utilized for attitude control and translation maneuvers until the last orbit. During the last orbit, the reentry control system (RCS) was used for final platform alinement in preparation for the retrograde maneuver. This was made necessary by the degradation of the OAMS toward the end of the mission. (See section 5.1.8.1.3.)

The translations accomplished during orbit by using the rate-command mode for attitude control appeared to the crew to be controlled more accurately in attitude than those accomplished while in the platform mode which had a tendency to drift in yaw. Fine tracking in all axes during translations was found to be more precise in the actual spacecraft than that experienced in the Gemini Mission Simulator.

Spacecraft attitude control response in the pulse mode was less than in the simulator, whereas the control authority of the spacecraft in rate command or direct mode was more positive than experienced on the Gemini Mission Simulator. The crew readily adapted to this variance and were able to position and track targets in each of these modes.

The crew used fuel conservatively in attitude hold by remaining in the pulse and horizon scan modes for long periods of the flight. The rate command was primarily used during translation maneuvers, and a combination of direct and pulse mode was used for some tracking tasks when the target was acquired late. The reentry-rate command mode was never checked by the flight crew during the flight. The failure and degradation of the OAMS was easily recognized and analyzed by the crew with subsequent attitude control maintained by a combination of thrusters in the remaining axes. The effect of venting of gases such as the water evaporator, cryogenic oxygen, and hydrogen was recognized and was adequately handled by the crew.

The reentry control system provided the crew with a positive control system for retrofire and reentry. The crew selected single-ring operation after jettison of the retroadapter section. This configuration was used until drogue parachute deployment after which the second ring was turned on.

The crew initially had difficulty in establishing or recognizing the proper spacecraft orientation associated with 0°, 0°, 0° attitude. This was caused by the crew angular position offset and by the lack of dynamic out-the-window simulations prior to flight. The crew, however, was able to obtain reference points on the spacecraft and windows to assist in determining correct alinements after a little experience in flight. Pitch attitude was referenced by establishing a line through the corner of the window, the front RCS yaw thruster, and the airglow on the horizon. Zero roll angle was referenced by orienting the inside vertical edge of the window frame perpendicular to the horizon. Yaw alinement was established by pitching down for a better view and then removing the apparent yaw error by using drift of terrestrial features as a reference.

7.1.1.3.3 Cabin lighting survey: Figure 7.1-2 presents the measurement of ambient light levels taken at several locations on the controls and displays panel together with the related out-the-window levels. Although the number of data points acquired so far is minimal, several preliminary conclusions can be drawn.

There is a significant contrast between the day-side black sky and the earth-shine light levels as seen through the window (roughly 2° above and below the earth's horizon). These intensities differ markedly from those viewed within the cabin, particularly on the center panel instrument and controls group, the radar range display on the command pilot's side, and the dc voltmeter area at the pilot's station. The 4200 foot-lambert reading measured by the pilot on August 24, 1965, suggests the presence of sun shafting and/or a reflection off the window or window frame. The combination of high light levels viewed out the window and low light levels within the cabin presents a problem of vision accomodation for those tasks requiring acquisition of the outside targets simultaneously with the management of systems within the cabin.

The flight director indicators (FDI) in the right and left center instrument panels are the only instruments with individual lighting. The cabin lights were normally on during the day portion of the revolution. The light from these two sources, together with that shining through the window, did not raise the light levels appreciably at the nonindividually lighted instrument locations. Light levels also appear to be too low for those situations involving the cross-monitoring of instruments when one crew member has to read an instrument in the opposite station.

7.1.1.3.4 Apollo landmark investigation: The Apollo landmark investigation on Gemini V consisted of obtaining photographs of preselected landmarks, evaluating the onboard maps used for the acquisition and identification of the landmarks, and obtaining photographs of landmarks selected from orbit.

The landmarks photographed were:

(a) Southern tip of La Palma Island in the Canaries

(b) Western tip of Cape Rhir, Morocco

(c) Northwestern tip of point on western extreme of peninsula, Lake Titicaca, Bolivia

(d) Northern tip of Isla de Panza in Lake Poopo, Bolivia

The 70-mm photographs are excellent and will be used as landmark acquisition-identification aids during the Apollo missions. The crew commented that the Apollo onboard maps were unsatisfactory and did not present enough information for acquiring and identifying landmarks. This will be remedied by use of the Gemini photographs as acquisition-identification aids on the Apollo missions.

The crew also commented that the types of landmarks most desirable for acquiring, identifying, and tracking are predominant coastal areas, lakes, and rivers. They believe that airports are not particularly good landmarks because they do not contrast sharply with the surrounding terrain.

7.1.1.4 Rendezvous evaluation pod (REP) and simulated Agena rendezvous.- The mission proceeded according to the flight plan through REP ejection. A problem was encountered in alining the platform just prior to REP ejection because of apparent intermittent operation of the horizon sensor; therefore, a 30-second alinement was made just prior to the 90° yaw for REP ejection. The REP was ejected approximately on time (15 sec late), the spacecraft was yawed 180°, the radar was turned on, and lock-on was quickly attained. The first digital readout of separation relative velocity was 3.5 ft/sec. Shortly afterwards, the first analog reading of relative velocity on the range—range rate indicator was 5 ft/sec. The REP continued straight out the 270° line for 0.9 mile and the range rate attained was 7 ft/sec. No definite nodal point was reached where the range stopped increasing and the range rate did not decrease noticeably.

The REP started moving behind the spacecraft and as it passed through the 210° bearing, the decision was made to power down and terminate the REP maneuver because of the fuel-cell oxygen supply pressure problem. Visual indications confirmed the radar readings. The crew estimated that the REP tumbled at a rate of approximately 1 deg/sec. The REP remained in the vicinity of the spacecraft for the next five revolutions and illuminated the nose of the spacecraft many times when it was very close to the spacecraft. In many instances, sunlight reflection off the REP was more visible than the blinking lights.

As a substitute for the cancelled REP exercise, four translation maneuvers were made to simulate rendezvous with an imaginary Agena in a different orbit. The information for the first two maneuvers was inserted into the computer by way of the DCS. The first maneuver was a height adjustment using the platform control mode; however, because of problems with this control mode, some out-of-plane change in velocity resulted. The second translation was the phase adjustment maneuver. The information for the third and fourth maneuvers was transmitted to the crew over the UHF voice link and inserted into the computer by the crew. The third maneuver was a plane change, and the last maneuver was a coelliptical maneuver. The last three maneuvers were made in the rate command attitude control mode, and this system satisfactorily held the spacecraft in the proper attitude. The aft-firing thrusters were used for all these maneuvers. In all four maneuvers, the spacecraft was well positioned in space to continue the rendezvous if this had been an actual case.

7.1.1.5 Experiments.- Although the crew was hampered by spacecraft systems failures of the oxygen tank heater, slow degradation of the OAMS, and power and propellant limitations, they did accomplish a large part of all experiments with the exception of D-2, which had to be cancelled because it required a rendezvous with the REP. Additional photography and spacecraft systems tests were also performed.

Electrical power was limited intermittently throughout the mission. The fuel-cell oxygen cryogenic heater failed early in the mission and the fuell-cell water production rate later in the mission constrained the amount of allowable electrical power. A degradation in the OAMS thrusters, beginning with the fifth day of the mission, limited the amount of controllability of the spacecraft, making tracking tasks extremely difficult. A failure of the utility cord powering the reticle in the optical sight curtailed tracking-type experiments over the United States on the third day.

Preestablished methods of acquisition to be used by the crew in accomplishing the tracking task experiments were revised. The fields of view of the telescope and of the 1270-mm lens were too small to be used for acquisition or tracking. The method used by the crew was to acquire the object first with the unaided eye, and then to let the command pilot track it using the optical sight and the pulse attitude control mode while the pilot operated the cameras or equipment controls. Results shown by the photographs indicate very satisfactory tracking by the crew. The pulse attitude control mode was satisfactory for all tracking where acquisition was begun early enough to start tracking at low apparent rates. However, during one of the missile tracking tasks, the visual sighting was lost against the cloud background and when the missile was reacquired against the sky background, the direct attitude control mode was required to get back on the accelerating missile. After the optical sight was again alined on the target, the pulse mode was used successfully to complete the tracking task.

The numerous experiments and operational tests performed during the passes over the United States placed a heavy work load on the crew each day. The stowage, assembly, and disassembly of equipment required to perform these rapidly occurring tasks were handled well by the crew, and a high percentage of these experiments and tests was completed.

Electrical power and propellant constraints prevented attitude control for the accomplishment of some experiments. Nevertheless, the crew was given information in the event they would be in proper attitude for accomplishing certain experiments.

7.1.1.6 Crew housekeeping.- The extended length of the mission, together with the number of experiments requiring pilot-operated equipment in the crew station, made it mandatory that the crew carefully manage housekeeping activities. The most critical tasks were anticipating stowage requirements and being prepared for a contingency reentry.

The flight crew maintained an orderly cabin by planning the requirements for equipment in advance for temporary stowage of each piece of equipment and for a location of the operational data required for each task. The crew made it a policy to stow only a limited number of items in the footwell and unstow limited items of food per man. By careful selection and management of equipment stowage and dry waste stowage, the crew was always in a semi-state of readiness in the event of an early reentry.

The flight data books were made available during the flight by stowing them next to the pilot on the center side of the seat. Continuity of command was provided by the interchange of necessary information and data to the other pilot prior to each sleep cycle. The crew sleep and eat cycles were not satisfactory during the flight. The sleep periods of one crew member were often interrupted by the other pilot's activities, air-ground voice communications, systems operation, and exterior lighting conditions. The crew eventually avoided some of the sleep interruptions by eating at the same time, taking vision tests at the same time, and sleeping at the same time. The crew also found that sleep periods were more successful when scheduled near their normal preflight sleep habits.

The flight crew monitoring of systems operation was satisfactory. The crew initially had detected the loss of the fuel-cell oxygen supply pressure and had turned the heater to automatic prior to the ground control request. The crew, by noting the RCS heater warning light, found that the RCS heaters were required during most of the flight.

The monitoring of critical items on the center instrument panel and water management panel was degraded because of inadequate lighting in this area. Crew efficiency was also hampered by the low light level in the overhead stowage areas. Items in this area are very difficult to stow efficiently and this is compounded by the lack of proper light.

7.1.1.7 Retrofire and reentry.- Stowage of experiment and operational equipment was initiated approximately 16 hours prior to retrofire and was completed about 12 hours later. The platform was alined accurately in BEF over the final $1\frac{1}{2}$ revolutions. Alinement was maintained throughout this period by using out-the-window references during both day and night. Stars and constellations close to the orbital track provided excellent night-side reference. The RCS system was used throughout this period because of the degraded OAMS.

During the last revolution over Carnarvon at T_R-27 minutes, the computer was updated by way of the DCS. This operation was upsetting to the crew because they were unaware that additional tracking data had made this update necessary, and were not expecting it. They were also concerned about quickly switching the computer from the reentry mode to the prelaunch mode to receive this update. In addition, the crew reported that the update verification indicator light did not illuminate (see section 5.1.10.2.2); however, to verify the computer entry, two memory cores were checked by the pilot using the manual data insertion unit. This served to confirm the validity of the update and restore the confidence of the crew for the retrofire and reentry maneuvers.

All preretrofire checklists were completed on or before the scheduled time. Crew reports of the separation events were similar to those of previous Gemini crews. Retrofire occurred precisely on time from the time reference system, and the pilot backed up the retrofire signal with manual retrofire at T_R+1 second. Spacecraft attitudes were held within ±1° of nominal by use of the rate command system with both RCS rings activated. The flight director indicator had to be used for the reference, because retrofire occurred on the dark side of the revolution and the RCS thrusters completely obliterated all external references. The crew reported the retrofire incremental velocities as 269 ft/sec aft, 10 ft/sec left, and 181 ft/sec down, which were close to nominal. Each of the crew members, particularly the pilot, said he experienced vertigo during the retrofire maneuver.

Retropack jettison was nominal after which the spacecraft was rolled to the heads-down position, the pulse mode selected (attitude control to PULSE and RCS to ACME), and the RCS B-ring turned off. At 400 000 feet, the direct mode was selected (attitude control in RATE COMMAND but with RCS to DIRECT) and only small inputs were required to damp the oscillations. The command pilot flew the back-up bank angle of 54°-left using the flight director indicator as a reference until the deceleration started to rise rapidly. The computer was indicating a significant overshoot with a full-scale deflection in the low-range and nearly full-scale in the high-range position. Because the downrange indication did not seem to be changing, the command pilot then rolled left to the 88° bank angle (zero lift), which he held to approximately the time of peak deceleration. At this time he went back to the nominal back-up bank angle of 54°-left and held this for about 40 seconds, after which he rolled to 60° right which he held for the last 33 seconds before drogue parachute deployment. Flying zero lift for a relatively short period of time (about 1 min 30 sec) during the most effective period of reentry lift capability caused the major portion of the 89 miles uprange miss distance. The navigation coordinates that had been entered into the spacecraft computer were subsequently determined to be incorrect. (See section 6.2.2.2.1.)

The command pilot's technique for controlling the reentry was to alternate between control modes and between rate and attitude flight-director-indicator references so as to provide optimum control and monitoring. Landing occurred at a zero altitude reading on the altimeter. The crew reported that the landing was very "soft", with submersion only part way up the spacecraft windows. The crew had completed all of the prelanding checklists, with the exception that the pilot could not stow his D-ring pin until after landing.

7.1.1.8 Recovery.- Recovery weather conditions were optimal and the spacecraft flotation attitude was satisfactory. The cabin and suit inlet temperatures, shortly after landing, were approximately 45° F and 50° F, respectively, providing a comfortable condition for the crew with the suits on and the hatches closed. All postlanding procedures were performed according to the postlanding checklist with one possible exception. The HF antenna, which is powered by the common control bus, would not extend and the crew could not transmit on HF. There remains a question as to whether or not the pilot inadvertently turned off the number 3 squib battery (common control bus), which powers the extend motor, prior to the time of the attempted HF antenna extension. The crew made a routine egress through the left hatch after approximately 1 hour in the spacecraft. The pilot and command pilot were hoisted aboard the helicopter shortly thereafter and transported to the prime recovery ship. Other than the crew reporting that the flight to the carrier was extremely warm (due to helicopter cabin temperature and no suit ventilation), no postlanding physical discomfort or problems were encountered.

7.1.1.9 Training.- The Gemini V crew training was conducted as outlined in reference 10. A summary of the Gemini V crew training is shown in table 7.1-I.

Because of the large number and complexity of experiments scheduled for the first time on this flight, the most time-consuming part of the training program was that concerned with experiments. Although this training was considered adequate, more priority should have been given to providing training hardware to the crew early in the training program so that experiment training could have been essentially complete before the crew reported to Cape Kennedy for the final phase of prelaunch testing.

Gemini Mission Simulator training was adequate with the exception that there was no "out-the-window" feature. Although some "out-the-window" training for the REP rendezvous was accomplished on the engineering simulator at the spacecraft contractor's plant, more training of this type should have been available for normal launch, orbit, retrofire, and reentry operations. As a result, the crew had to spend considerable time in flight learning to correlate spacecraft attitudes with horizon and/or star sightings.

7.1.2 Gemini V Pilot's Report

7.1.2.1 Preflight schedule and training.- The Gemini V flight team was on a rigorous schedule, and it was recognized early in the preparation for flight that in order to meet the scheduled launch time, it would be necessary to make efficient use of the critically short training time. Although efficient utilization of time was observed, it is thought that the training of the Gemini V crew was minimal. However, the flight appeared to go well and training was apparently adequate to complete the flight. The amount of systems training, technical training, and study in the systems area was adequate. The crew had confidence that they were well versed in the spacecraft systems. Water-egress training was well planned and adequate.

7.1.2.2 Countdown.- On the day of the actual flight, the launch countdown was excellent. The operations crew did a good job and everything went smoothly. In particular the flight crew was well pleased with the timing.

7.1.2.3 Powered flight.- The powered portion of flight can be summarized by saying that except for a few minor discrepancies, it was a smooth, powered flight that inserted Gemini V into an almost perfect orbit. Lift-off was smooth, and easily identifiable. The first minor discrepancy occurred when the stage II instrumentation power supply (IPS) fuel-pressure gage failed to the full-scale position at approximately 99 seconds. The noise level and vibration through max q were not objectionable at all, and were at a very low level as compared with the Atlas. The second minor discrepancy was the higher than expected "POGO." It was of a higher level than would be desirable for flight and vision was impaired for a short period; however, it was not overly objectionable because there were no detailed gage readings required in that particular period. Staging was on time, and the launch vehicle had lofted slightly up to the time of staging as anticipated. After staging, the RGS smoothly steered out the lofting, and the radio guidance system and inertial guidance system were nearly synchronized throughout the remainder of the flight. There were no attitude deviations and no noticable angular rates. Insertion was nominal and in the 20-second period immediately after insertion, the angular rates and attitude excursions remained negligible. Communications were lost with MCC-H after the announcement of V/V_R of 0.8, and the loss persisted until immediately following insertion. The command pilot switched to the secondary UHF, and contacted MCC-H immediately after recording the incremental velocity indicator (IVI) readouts. (It was found later, however, that the radio had not failed, and the problem was one of procedures at MCC-C.) The spacecraft separated from the launch vehicle cleanly, with no apparent angular rates.

7.1.2.4 Orbital phase.-

7.1.2.4.1 Control system:

Orbital attitude and maneuvering system - The torquing obtained in the pulse mode by the thrusters was less than in the Gemini mission simulator (GMS). This mode, however, is excellent for precise tracking tasks and uses a negligible amount of fuel.

The direct mode was a precise stream of thrust and was much sharper and more positive than in the GMS. This mode was necessary where a rapid movement was required from one attitude to another but should be avoided if possible because of the associated high rate of fuel consumption. Rate command was a very strong mode and stopped the spacecraft precisely when the stick deflection stopped. It was much more precise than in the GMS. The horizon scan mode had loose tolerances, but is an excellent mode for long-range general attitude hold. The crew thought that the platform mode was not maintaining attitude within the prescribed design limits. (See section 5.1.5 for operation of the platform mode.)

Reentry control system (RCS) - The pulse mode is the same as on the OAMS. Thruster firings were all visible at night. Very precise control was possible. The direct mode was very precise and had excellent control authority. The rate mode had slightly looser limits than the OAMS rate command, but was still very tight. The horizon scan mode was almost identical to the OAMS horizon scan mode. (The platform mode was not used while on the RCS.)

7.1.2.4.2 Experiments: In relation to the length of the Gemini V mission, it is believed that the proper number of experiments was onboard and that there was enough time available to do most of these experiments. The tracking reticle was used without the additional dimming rheostat in the circuit and the dim setting was satisfactory for lower order star tracking. Full bright was ample for day tracking across any type of sky or terrain. The new pattern on the reticle was evenly diffused and was satisfactory for the task. In general, the experiments went well. Some were compromised because of the long periods of drifting flight; however, the majority of experiments were 85 to 100 percent completed.

7.1.2.5 Eating, drinking, sleeping, urination, defecation, and housekeeping.- The Gemini V flight crew found that the bite-size foods were not as appetizing as they had hoped in the early days of testing, and, because of this, they were not eaten after the third day. Rehydratable food, which was also carried on this flight, was eaten for the remainder of the flight. Four rehydratable food bags failed on the Gemini V flight, mostly because of the folds caused by crushing these bags around smaller objects.

It would be difficult to overremphasize the importance of drinking sufficient water. A careful drinking-water log was kept on this flight in order to assure proper water intake. The water on the Gemini V mission was good and it was cold. It still had a great deal of air in it; however, this did not seem to have an adverse effect on the crew. The drink gun worked satisfactorily and did not leak or have valve stoppage or slowdown.

The Gemini V crew had extreme difficulty in sleeping during the periods allotted. The spacecraft was so quiet in flight that anything (conversations with the ground crew, experiments, system management and test, et cetera) done by one crew member interrupted the other crew member's sleep. The polaroid window shields were found to be useful during sleep periods in cutting down distractions from sunlight. At times on this flight, both crew members slept simultaneously.

The urine system worked quite well. Two new procedures established in flight were:

(a) Preheat the system for a minimum of 4 minutes prior to flushing.

(b) After the urine receiver bag is empty, open the valve on the urine receptacle for 30 seconds to flush air through and then cycle two or three times to actuate the flapper valve for drying.

The new rubber receiver worked well but did get very gummy and sticky even when cleaned thoroughly, and each one remained usable for approximately 2 days.

Defecation was performed carefully and slowly. Care had to be utilized to assure that the defecation bag was wide open all the way to the bottom, was firmly glued on, and properly alined. The whole procedure was difficult and time consuming, but possible. For stowage planning purposes, one defecation bag, complete with medical disinfectant bag and tissues, will require approximately the same amount of stowage room as one entire food bag unless there is some change to the equipment or procedures.

It was recognized at a very early date that stowage would be one of the biggest and most critical problems of the Gemini V mission, and the flight proved this to be the case. The spacecraft was badly crowded, but the crew did successfully stow most items for reentry in the places where they belonged, and the spacecraft was landed in a clean, well-stowed configuration. This would not have been the case, however, had not a great many hours been available toward the end of the mission for restowing. For later missions, it should be recognized that approximately 4 hours must be made available for stowage, or all items will not be stowed properly during reentry.

The first step concerning stowage was to remove all of the food from the right-aft food box. This food was stowed about the cockpit above both ejection seats in the red stowage bags and one or two packages on each side of the floor. From this point on, the right-aft food box was used only to store wet waste, defecation bags, and garbage type items. The red bags above both ejection seats were extremely useful. The $8\frac{1}{2}$-inch by 11-inch flight plan and the flight books were kept on the inside edge of each seat between either the command pilot's or the pilot's hip and the seat, depending on which one was using them. The two smaller 5-inch by 8-inch books were kept in the elastic-topped pouches along the side of the center console. The surface photography maps were kept in the regular map and data case in the left-hand footwell. The larger green stowage pouches were found to be unsatisfactory. The lids, with the bungee cord along the top, were too hard to get apart; and anytime that any large items were in the pouches, these pouches stood open at the top and allowed small items to float in and out. These pouches rapidly wore to shreds.

The helmets and gloves were removed very early in the flight, placed in a light-weight helmet bag and fastened with velcro to the floorboard in between the ejection seat foot stirrups. These items remained there throughout the entire flight until just before reentry. On the right side of each helmet there was a convenient place to put a food bag and/or an aluminum food cover to hold garbage or paper trash. On the left side of the helmet, by the squared-off footwell corner, there was room for the exerciser and perhaps one other small item. As the flight progressed, it rapidly became evident that for every food bag removed, twice that amount of room was required for stowage. That is, when the paper and trash and residual food were replaced in the aluminum bag holding the food, it was nearly equal to the volume it had occupied prior to unpacking it. In addition, the defecation bag with the feces and disinfectant bag all rolled into a package very similar in size to a food bag. Each and every day, a very thorough house cleaning was performed in which all garbage and trash were stowed very carefully. If this were not done, it was found that the available free space very rapidly dwindled, making it difficult to find anything.

The preretrofire stowage was conducted over approximately a 12-hour period. A great deal of time and thought was given to this stowage, and it was completed according to the reentry stowage list with a few exceptions: several partially empty defecation bag pouches, the voice tape recorder cartridge belt, one food bag full of paper trash, one of the sponge rubber camera box liners, and one or two other small items of this type were stowed around, behind, and under the left ejection seat; the zodiacal light camera was stowed in the right-aft food box, and a fair amount of food was carefully packed and taped into the window adapter mount in its bracket in front of the keystone stowage box. It was found that each piece of paper trash had to be very carefully folded and reduced in size as much as possible and that each rehydratable food bag had to have the air evacuated, then had to be rolled very tightly, and fastened with a piece of tape or rubber band. These were left temporarily in one of the stowage bags and then restowed, item by item, in the big locker so that they were fitted very tightly into each of the small areas. These small items had to be stowed very tightly at zero g because they kept floating out if friction did not hold them in place. This fact combined with the deep storage boxes was one reason so much time was needed for inflight stowage.

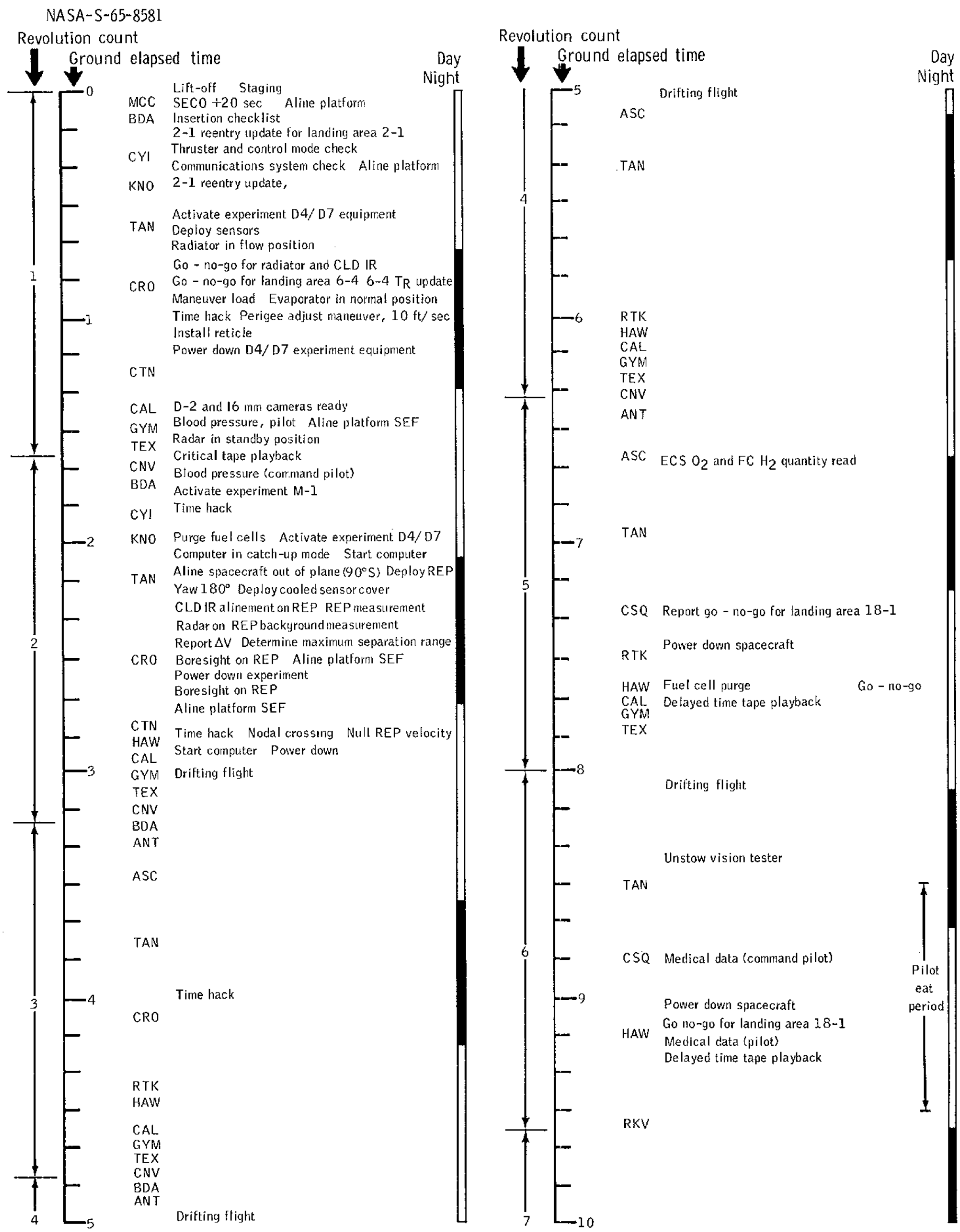
7.1.2.6 Retrofire.- The RCS was activated, checked out, and used to aline the platform BEF for one and one-half orbits prior to retrofire. The platform was alined by using horizon scan and the spacecraft was controlled inside the horizon scan limits with the pulse mode. The needles were finely alined. The T_R and targeting load was sent over Houston, and reentry mode was selected on the computer immediately thereafter. The platform, computer, and all systems checked out good around to Carnarvon. At Carnarvon, a T_R update was sent without warning to the crew. No digital command system (DCS) light was received and the computer was still in the reentry mode just as the update was started. The crew rapidly switched to prelaunch mode, but the DCS light did not illuminate. The crew was not certain that the update had been entered correctly. The ground had received the proper message-acceptance pulses via telemetry, and two memory locations were read out from the computer by the crew to further verify the update to be correct. The clock was set to T_R-26 over Carnarvon, and was counting down correctly. The crew then started going through the preretrofire checklist and had everything checked and double checked prior to the retrofire maneuver. The crew went to retrofire attitude and dual-ring rate command; the alinement appeared excellent, and the T_R time and the spacecraft clock times were exactly synchronized. At T_R, the retrorockets fired and the attitudes were held within ±1°. It was noted that the third retrorocket tended to yaw the spacecraft slightly off to the left and the fourth one tended to yaw it slightly off to the right. However, the dual-ring RCS had ample authority to counteract this offset. Retrofire was accomplished in the middle of the night, and the lights were turned up bright in the cockpit. At the instant of retrofire, the crew had the impression that the whole outside was a fireball as the retrorockets fired and the RCS thrusters began firing directly in front of the spacecraft windows. The crew was apparently somewhat sensitive from being in zero g for 8 days, and as the retrorockets fired, the command pilot felt that the spacecraft had completely stopped and was accelerating back to the west. The pilot felt that it was going into an inside loop.

7.1.2.7 Reentry.- After retrofire, the retro attitude was maintained, the retro jett squib was armed, the retro jett light came on, and the retropack was jettisoned on time. The spacecraft was pitched up to approximately +20° and rolled left to the inverted position. The B-ring was turned off, and A-ring was placed in pulse mode. Single-ring pulse mode was flown to 400 000 feet, and single-ring direct mode was used from 400 000 feet to approximately the time deceleration forces started to rise rapidly. The rates, combined with steering, became too demanding, at which time single-ring rate command was selected to help damp the rates. Guidance commands came in at 280 000 feet; the crossrange needle was off to the right and the downrange needle was pegged full-scale low (approximately half-way on high-scale). At this point, the spacecraft was banked 54° left, according to plan, and held for about 40 seconds. When the downrange needle did not start moving up, the spacecraft was banked 90° left (time of rapid deceleration rise). By the time the deceleration reached 6g, it was realized that the downrange needle was not going to come off the peg, and the spacecraft was rolled back to the nominal bank angle of 54°-left. At this point, it was debated whether to go to the reverse nominal bank angle or to hold the left bank angle. The crossrange needle had moved in during the 90° bank, indicating that it appeared to be working properly and that left bank was still needed to correct the crossrange error. It was decided to fly the 54°-left bank through the effective lift and then bank right. In a short time the deceleration reached 7.5g, and shortly thereafter the altimeter began indicating below the full-scale reading of 100 000 feet. The crew started down the checklist for landing and, as the pilot called off 100 000 feet, the landing squib was armed; however, when the pilot called off 70 000 feet, the command pilot erroneously put out the drogue parachute instead of going to dual-ring RCS. The drogue parachute came out at 70 000 feet, gave a couple of supersonic squids, opened very neatly in the reefed condition, dereefed, and was a very good looking drogue. The crew selected dual-ring RCS at 65 000 feet. The spacecraft appeared extremely stable throughout the entire landing phase. Cabin repressurization and O_2 high rate were selected at 50 000 feet to pressurize the cabin with a positive pressure; however at 27 000 feet the pressure dropped rapidly to zero, and the snorkle and vent were opened at that time. The main parachute was deployed at 10 000 feet and it opened in the reefed condition, perfectly symmetrical, and disreefed symmetrically. There was no swing, oscillation, or roll during descent. The snorkel and the vent valves were closed at 2000 feet, the repressurization and O_2 high rate were left on, and approximately 1 psi of positive pressure built up prior to landing.

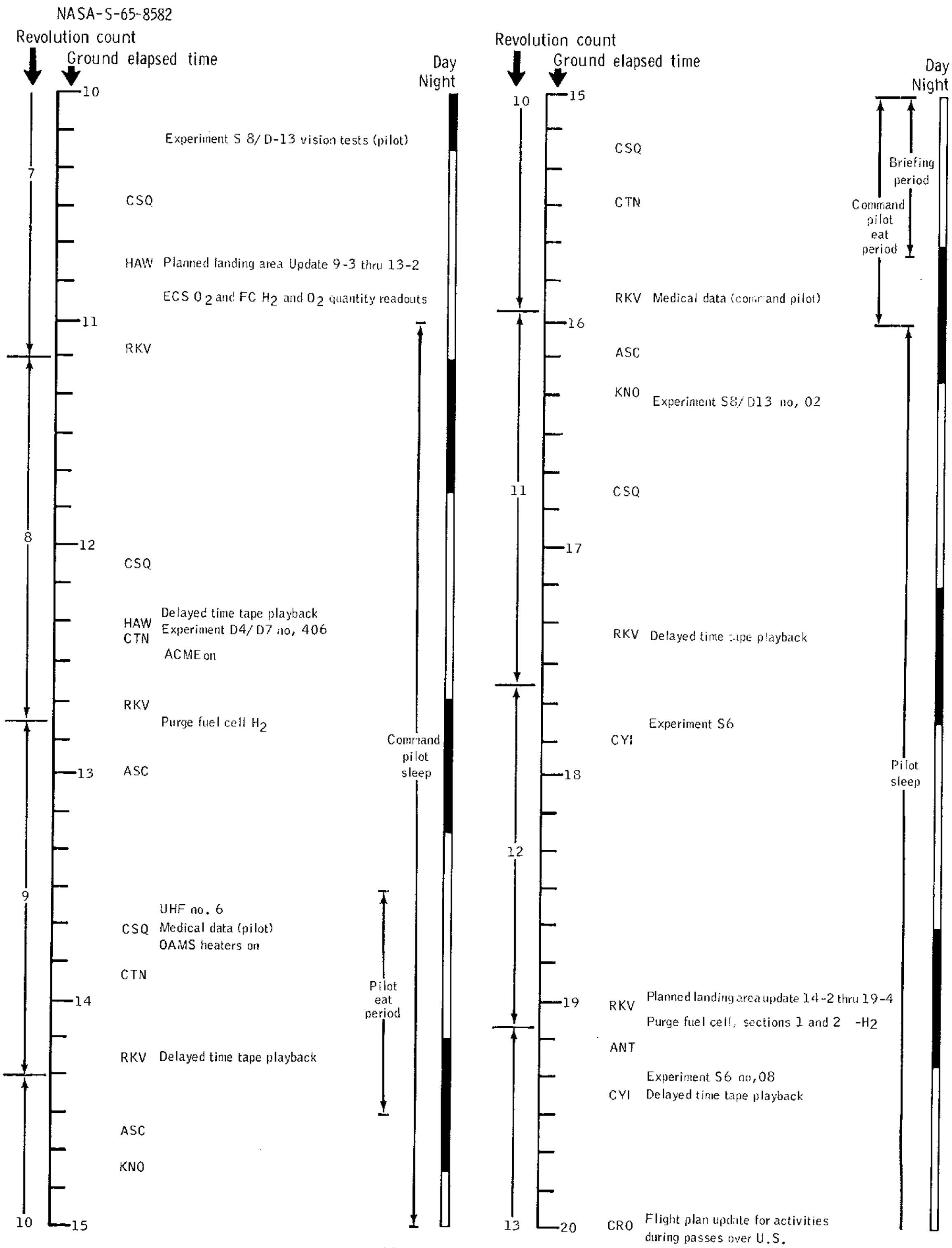
7.1.2.8 Landing and recovery.- Touchdown was very "soft" and no large water splash was noticed. The crew jettisoned the main parachute and proceeded through the postlanding checklist. They could hear the helicopters and "Air Boss" (Navy recovery leader's airplane) calling while on the main parachute. The crew talked to "Air Boss" two or three times, giving him counts so they knew that they were approximately 85 miles from the ship on a bearing of 280°. After the spacecraft landed in the water, the recovery aircraft apparently did not receive any transmissions from the crew, but the crew was receiving from that aircraft in addition to many other aircraft. The first aircraft directly overhead was an Air Force C-54, apparently out of Bermuda, with swimmers and jumpers aboard. "Air Boss" however elected to wait a few minutes for the prime recovery helicopter. The helicopter arrived shortly and the swimmers jumped in and attached the flotation collar to the spacecraft. The spacecraft windows were relatively clear at that time and the swimmers came up and peered into the windows. The crew gave them the "thumbs up" signal, and the swimmers completed putting the collar around the spacecraft and inflating it. After the collar was inflated, the left-hand hatch was opened. The crew proceeded to shut off the ECS system and power down the spacecraft. The crew egressed without incident, closed the hatch, stepped into a Navy raft, and rode the sling up and into the recovery helicopter.

TABLE 7.1-I.- CREW TRAINING SUMMARY

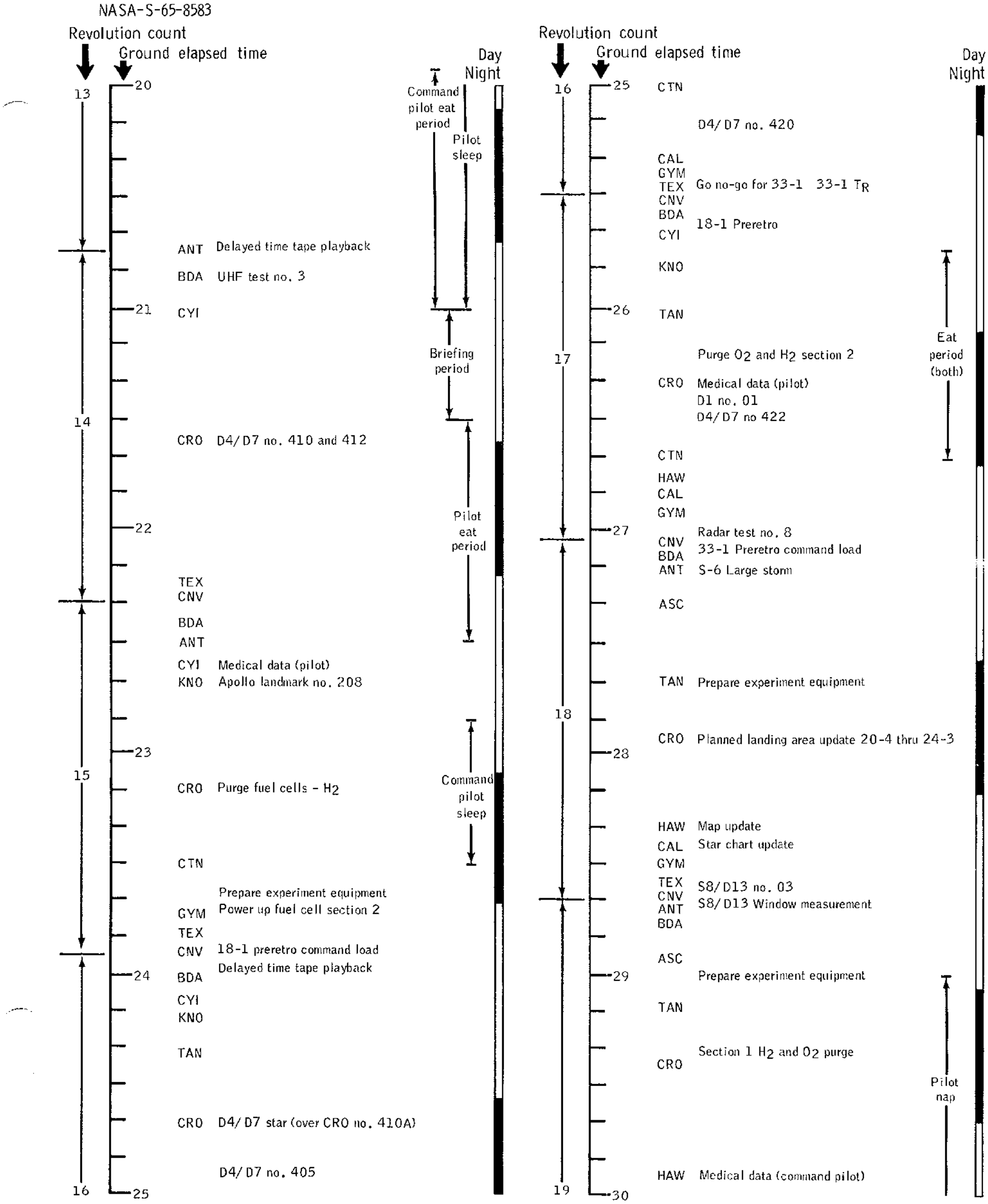
	Command Pilot		Pilot	
	Runs	Hr:min	Runs	Hr:min
Spacecraft tests		53:55		73:28
Gemini mission simulator		105:50		108:40
MAC engineering simulator (launch, rendezvous, and reentry)		38:00		34:00
Parachute	11		11	
Experiments training/briefings		150:00		150:00
Centrifuge	2		8	
Launch abort training	236		172	
Planetarium		34:00		34:00
Zero-g (KC-135)	44 Parabolas		44 Parabolas	
Survival training (water egress)		8:00		8:00
Systems briefings		58:00		58:00



(a) 0 to 10 hours g.e.t.
Figure 7.1-1. - Summary flight plan.



(b) 10 to 20 hours g.e.t.
Figure 7.1-1. - Continued.



(c) 20 to 30 hours g.e.t.
Figure 7.1-1. - Continued.

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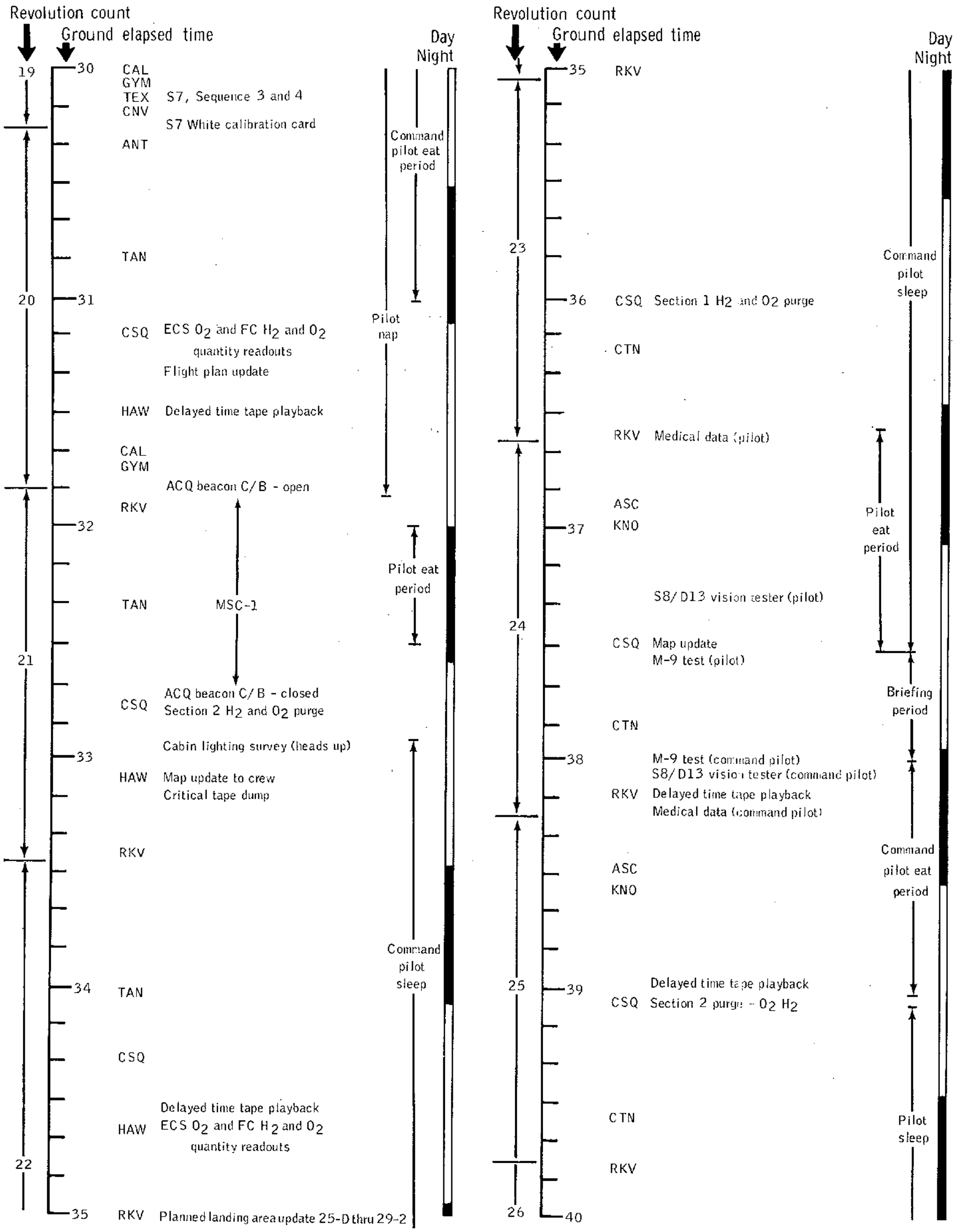
NASA-S-65-8584



(d) 30 to 40 hours g.e.t.
Figure 7.1-1. - Continued.

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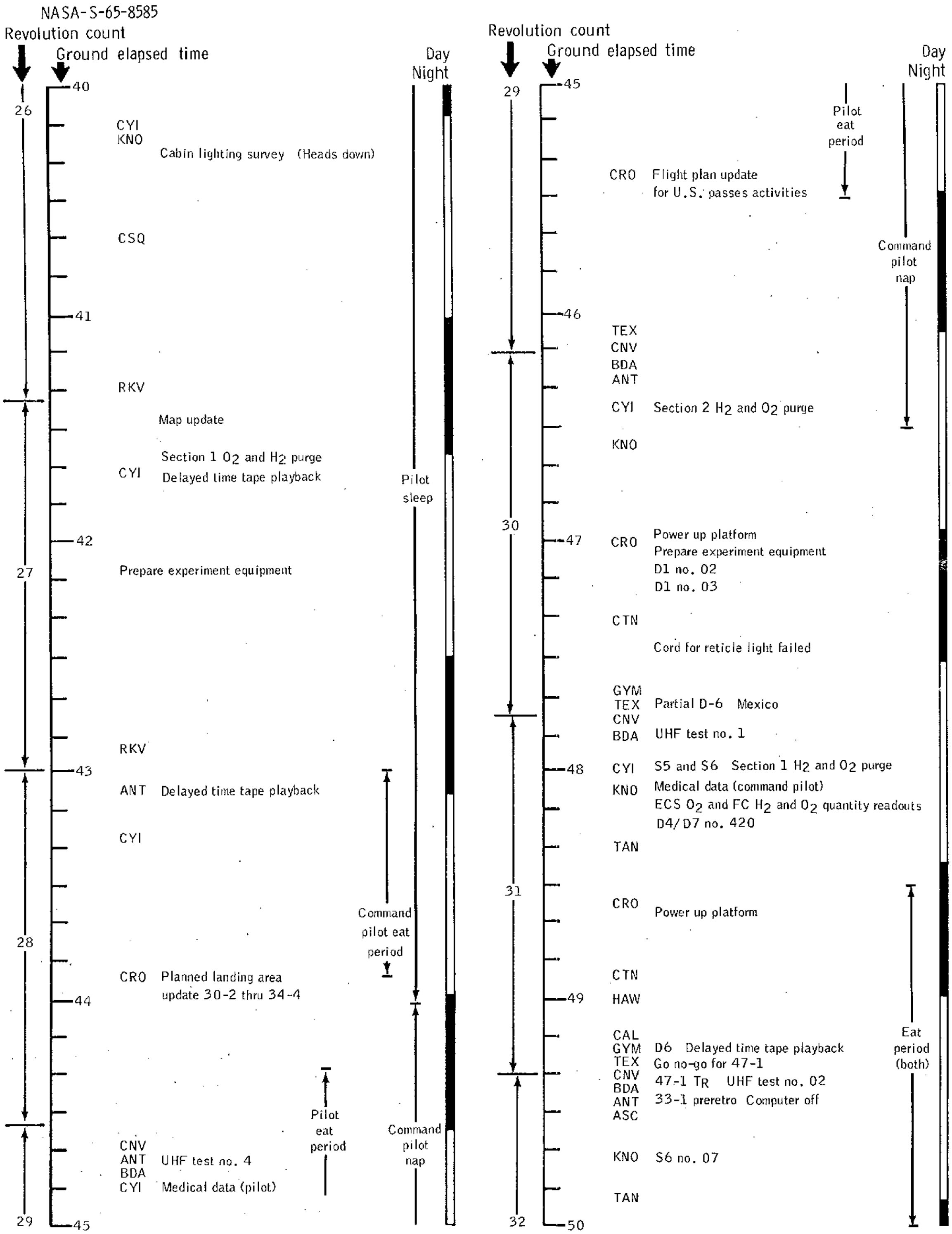
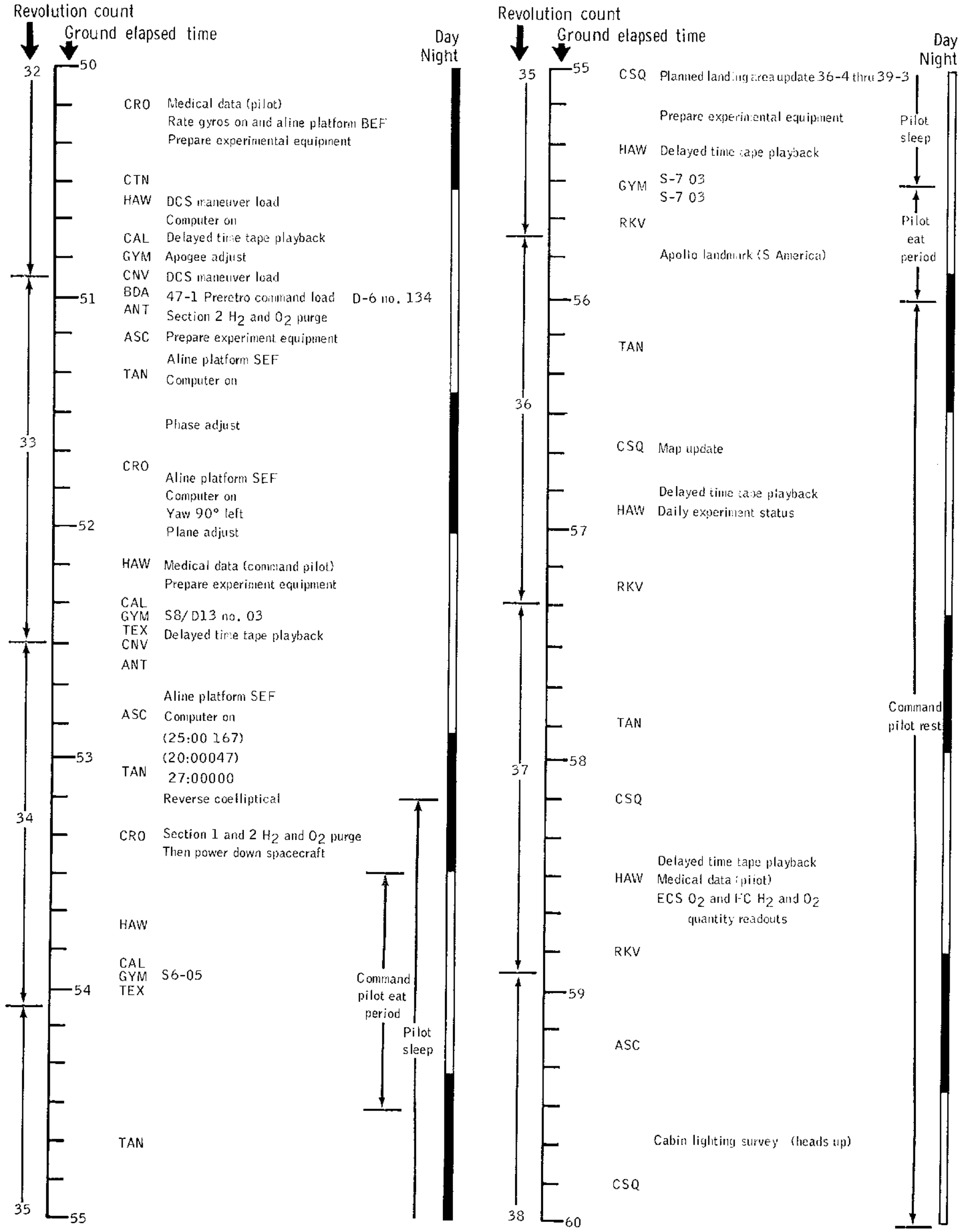
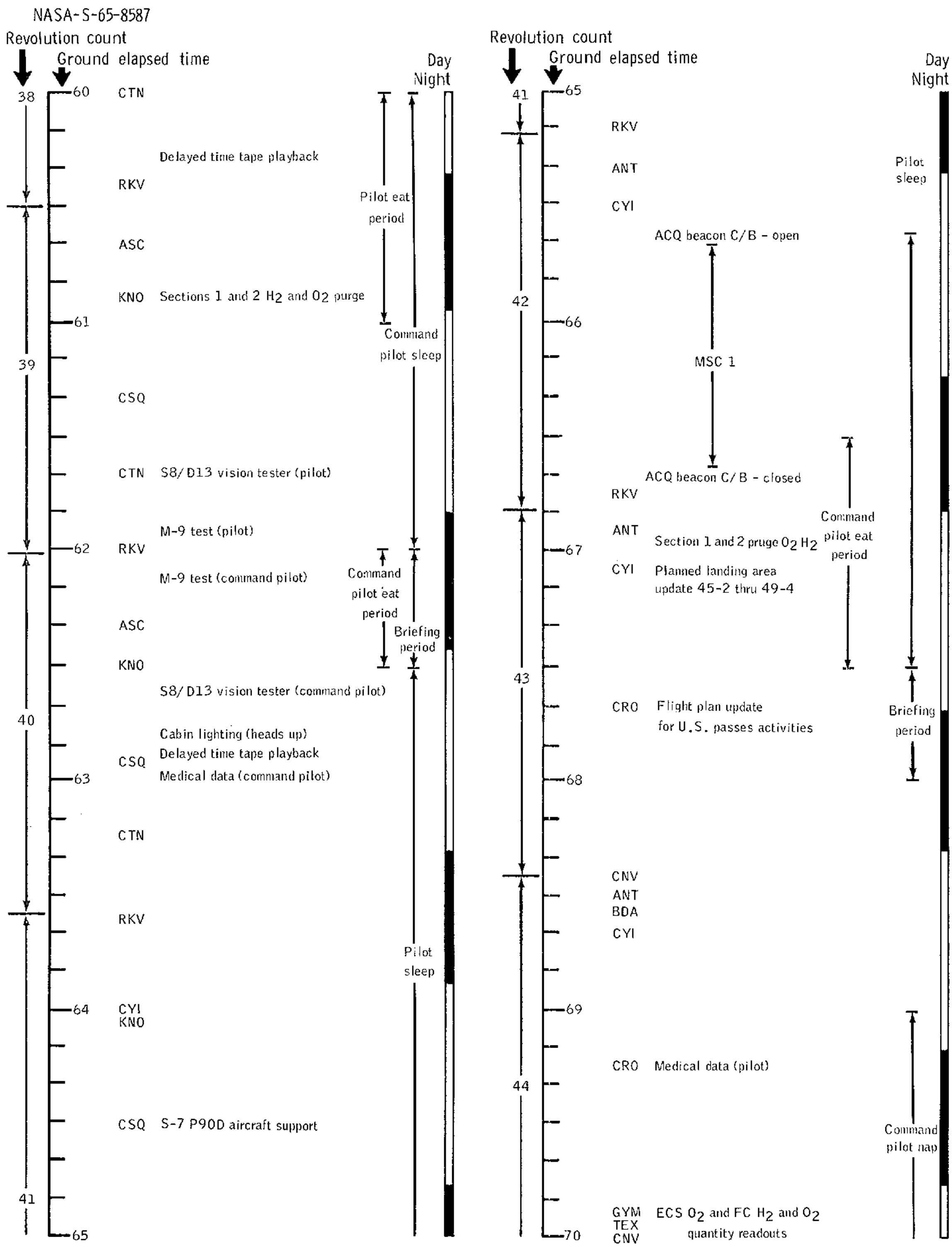


(e) 40 to 50 hours g.e.t.
Figure 7.1-1. - Continued.

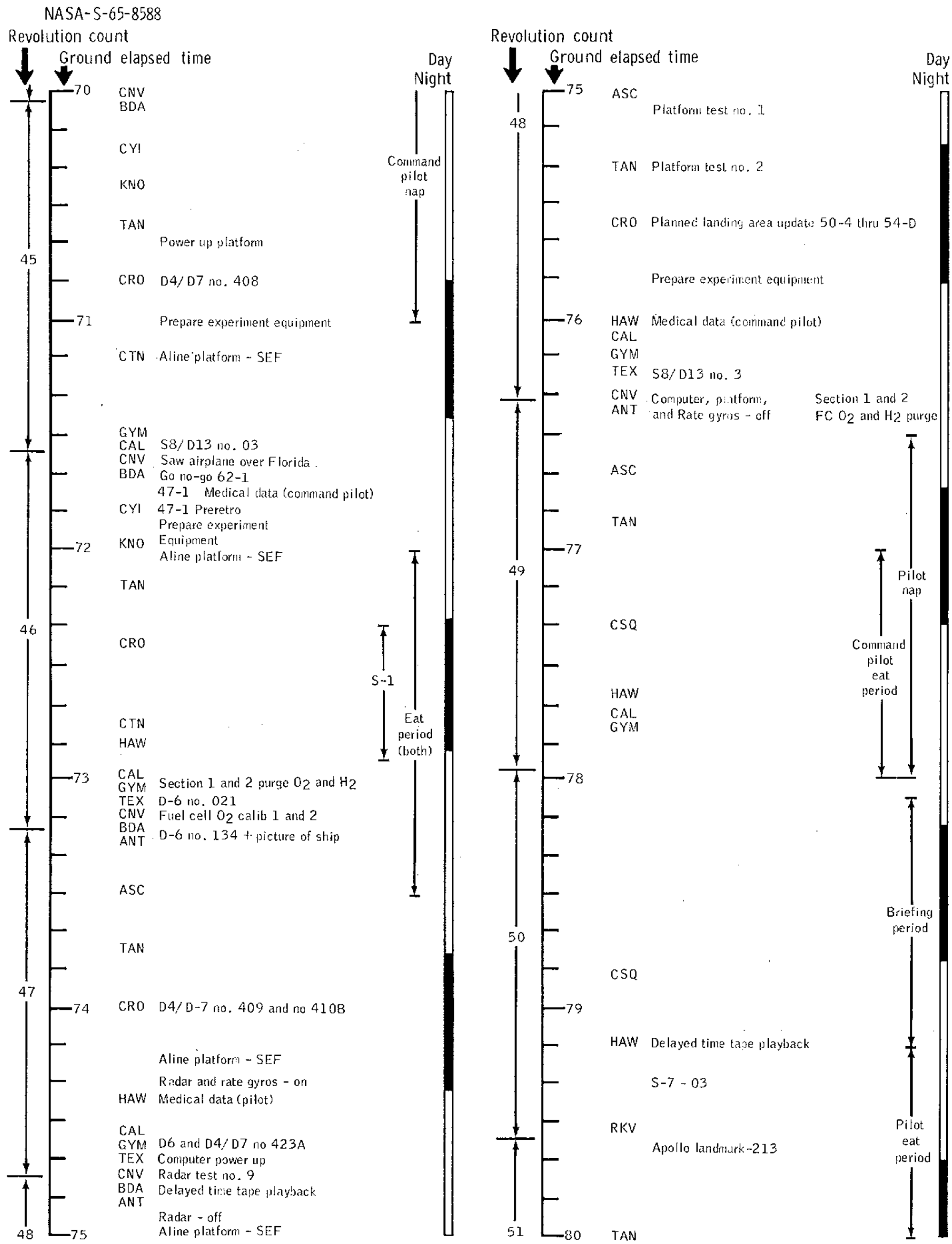
NASA-S-65-8586



(f) 50 to 60 hours g.e.t.
Figure 7.1-1. - Continued.

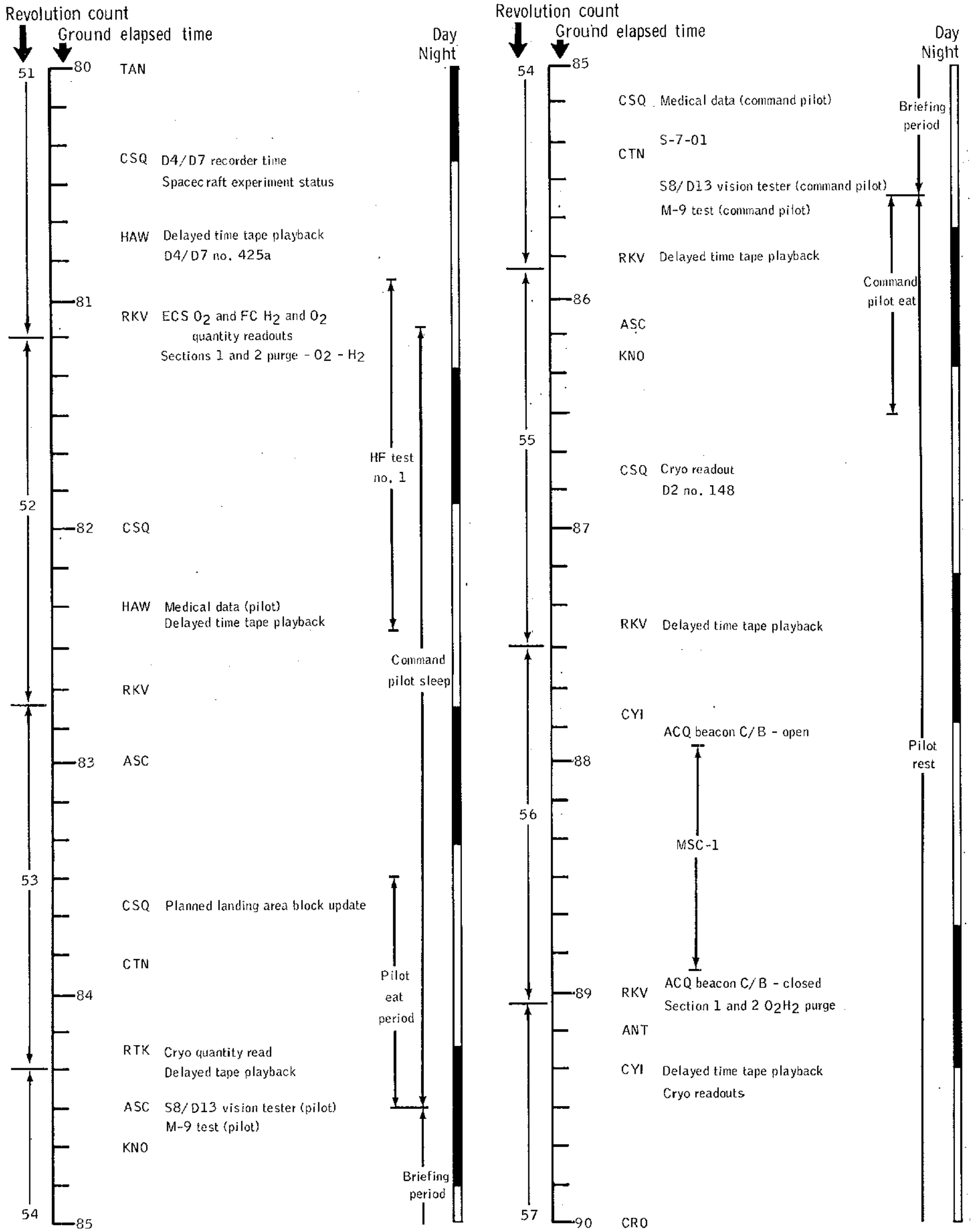


(g) 60 to 70 hours g.e.t.
Figure 7.1-1. - Continued.

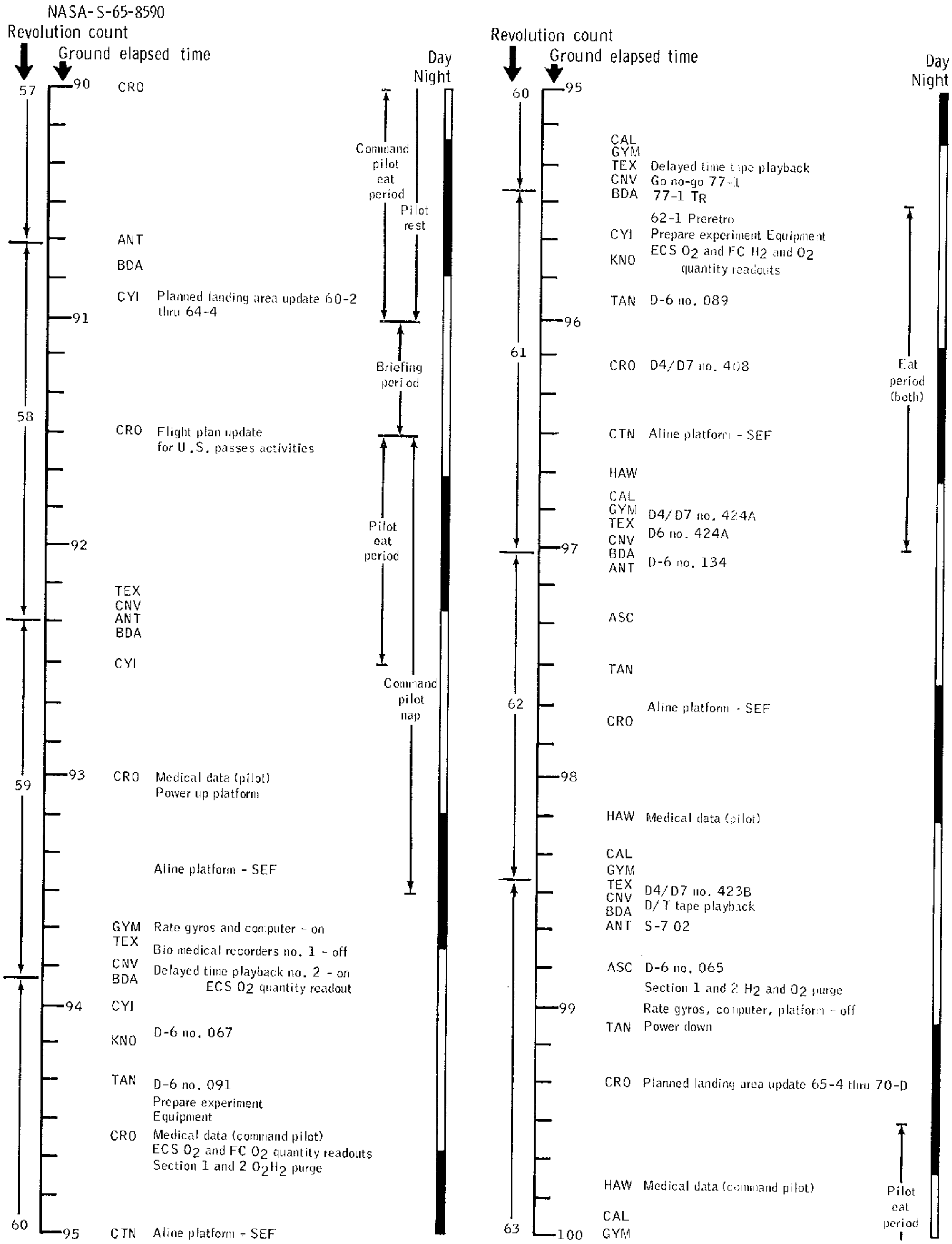


(h) 70 to 80 hours g.e.t.
Figure 7.1-1. - Continued.

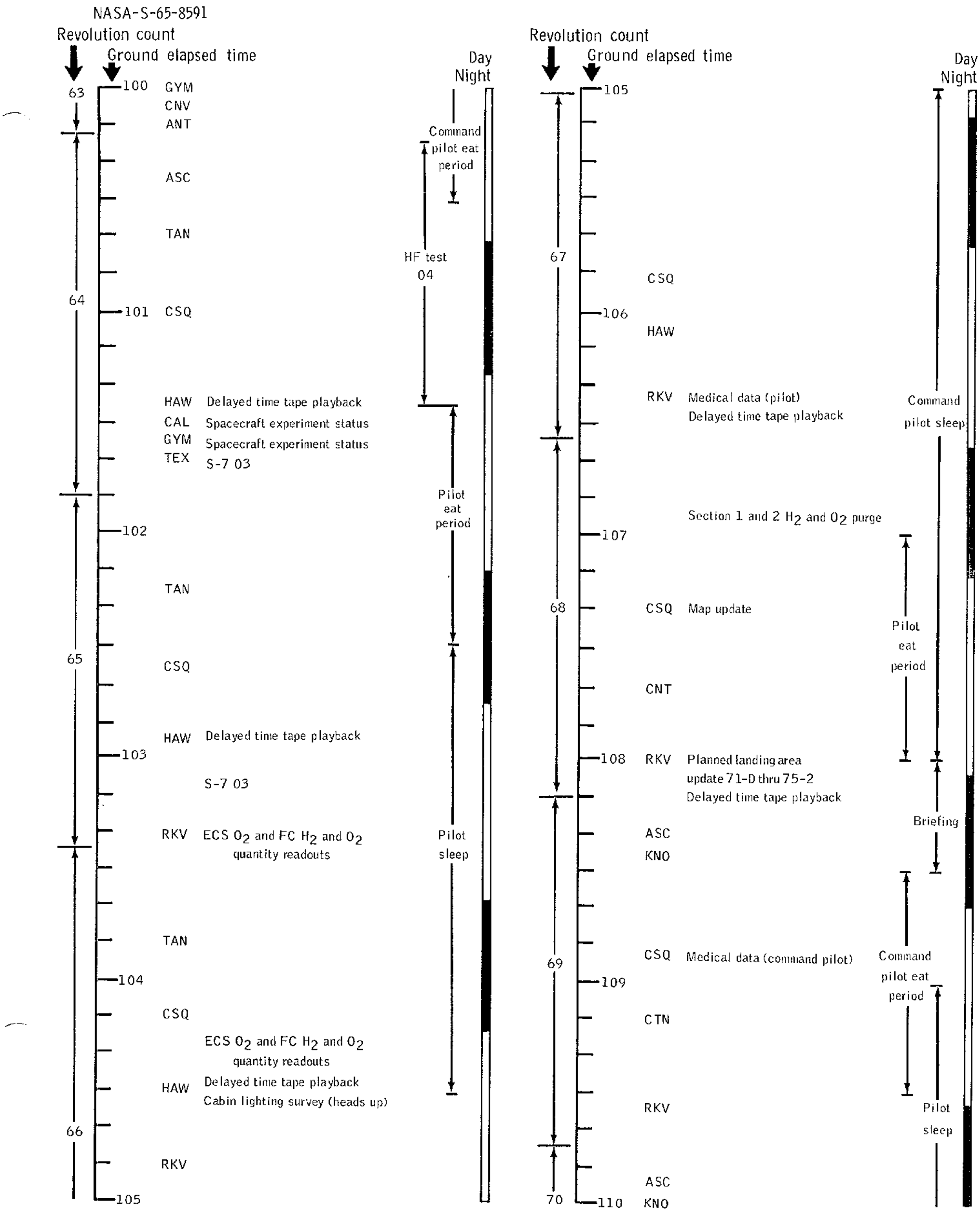
NASA-S-65-8589



(i) 80 to 90 hours g.e.t.
Figure 7.1-1. - Continued.

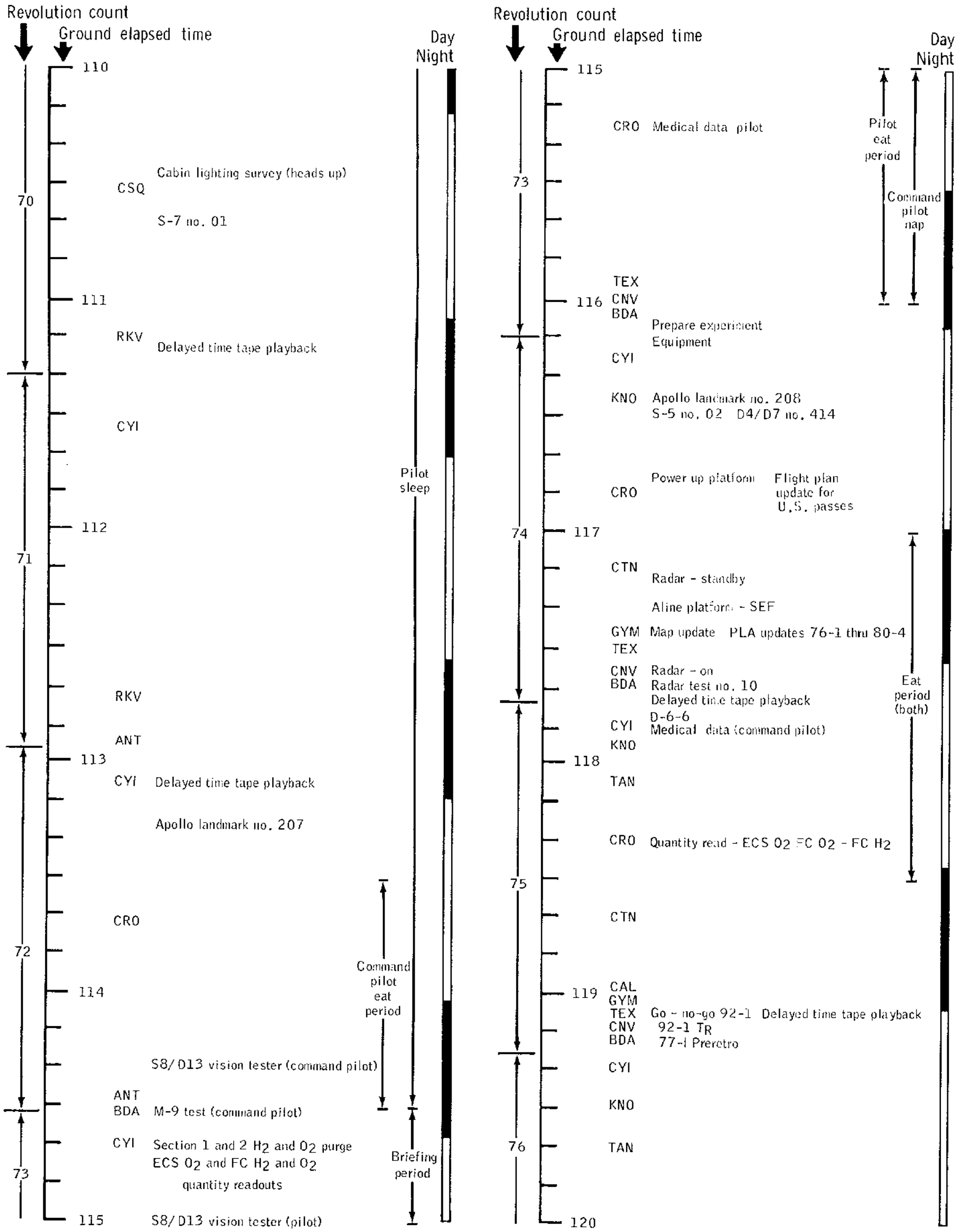


(j) 90 to 100 hours g.e.t.
Figure 7.1-1. - Continued.



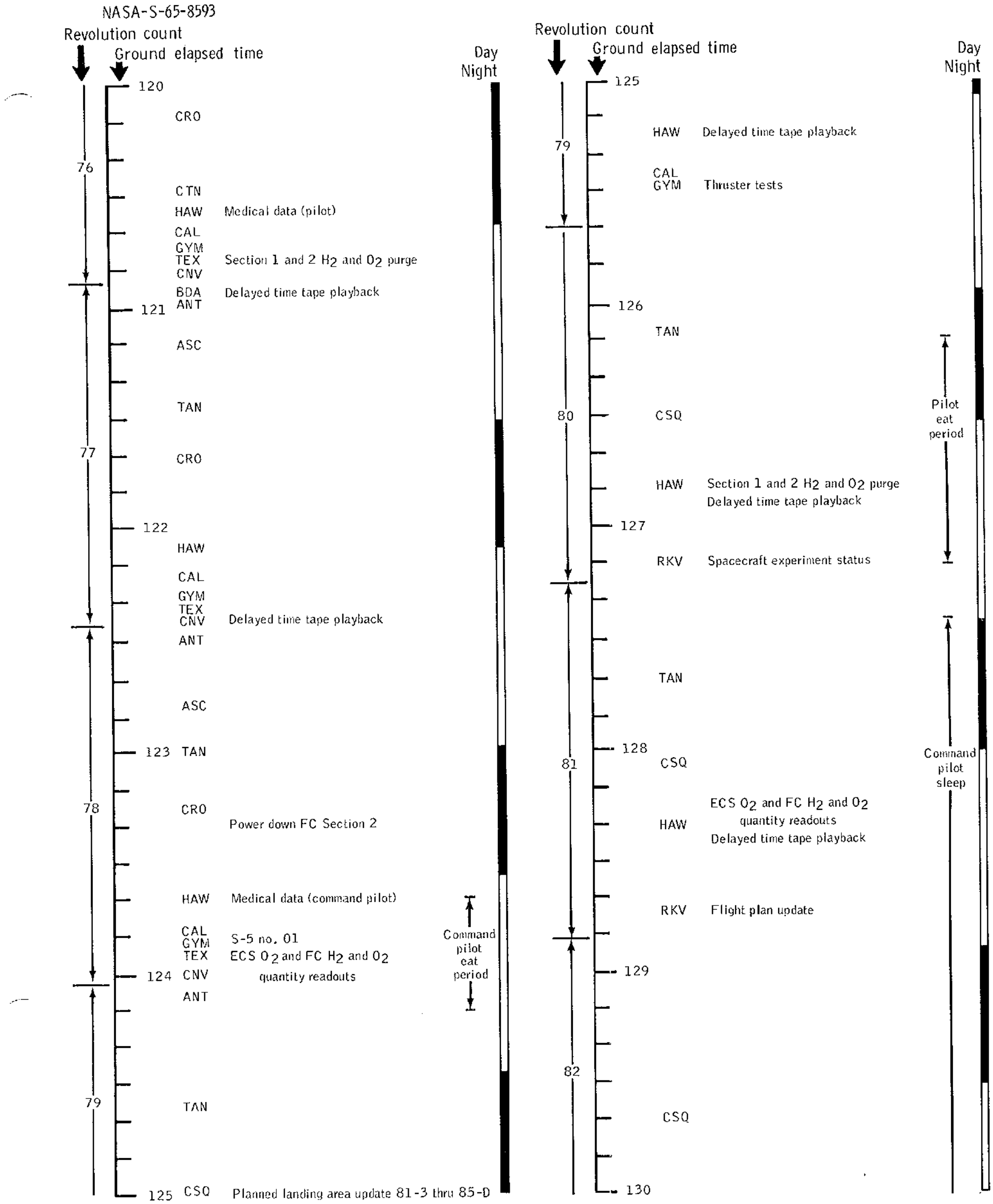
(k) 100 to 110 hours g.e.t.
Figure 7.1-1. - Continued.

NASA-S-65-8592

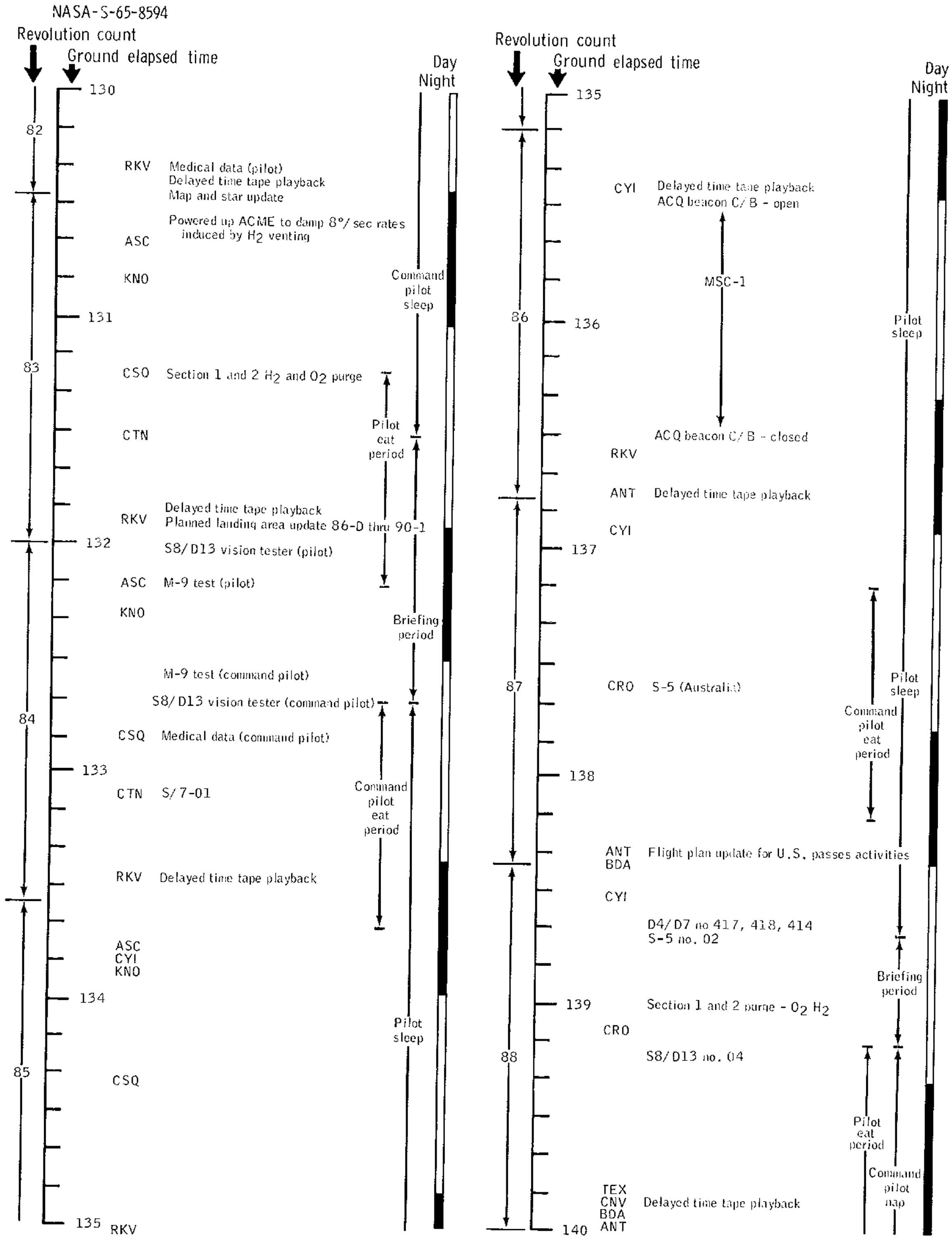


(l) 110 to 120 hours g.e.t.

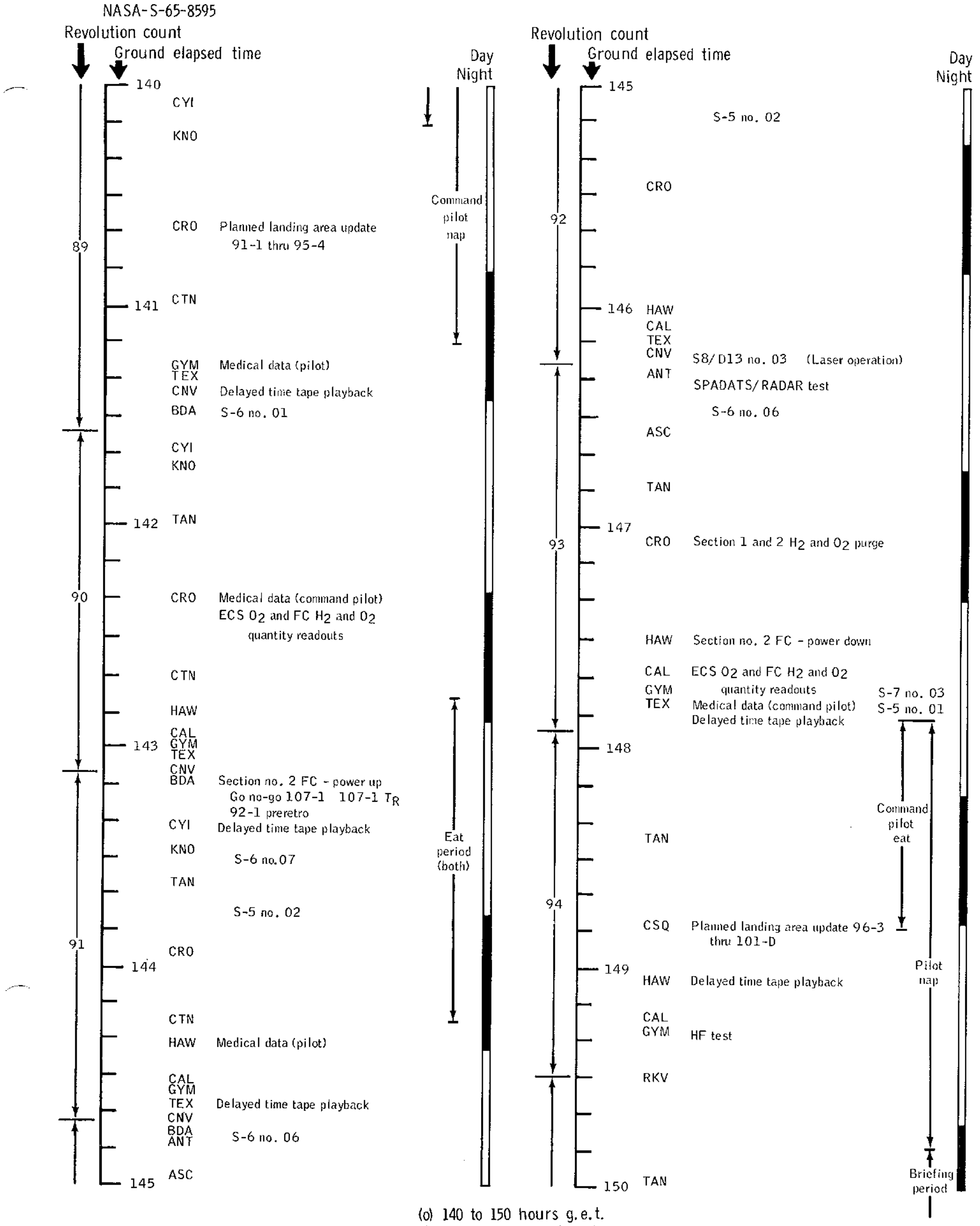
Figure 7.1-1. - Continued.



(m) 120 to 130 hours g.e.t.
Figure 7.1-1. - Continued.



(n) 130 to 140 hours g.e.t.
Figure 7.1-1. - Continued.



(o) 140 to 150 hours g.e.t.
Figure 7.1-1. - Continued.

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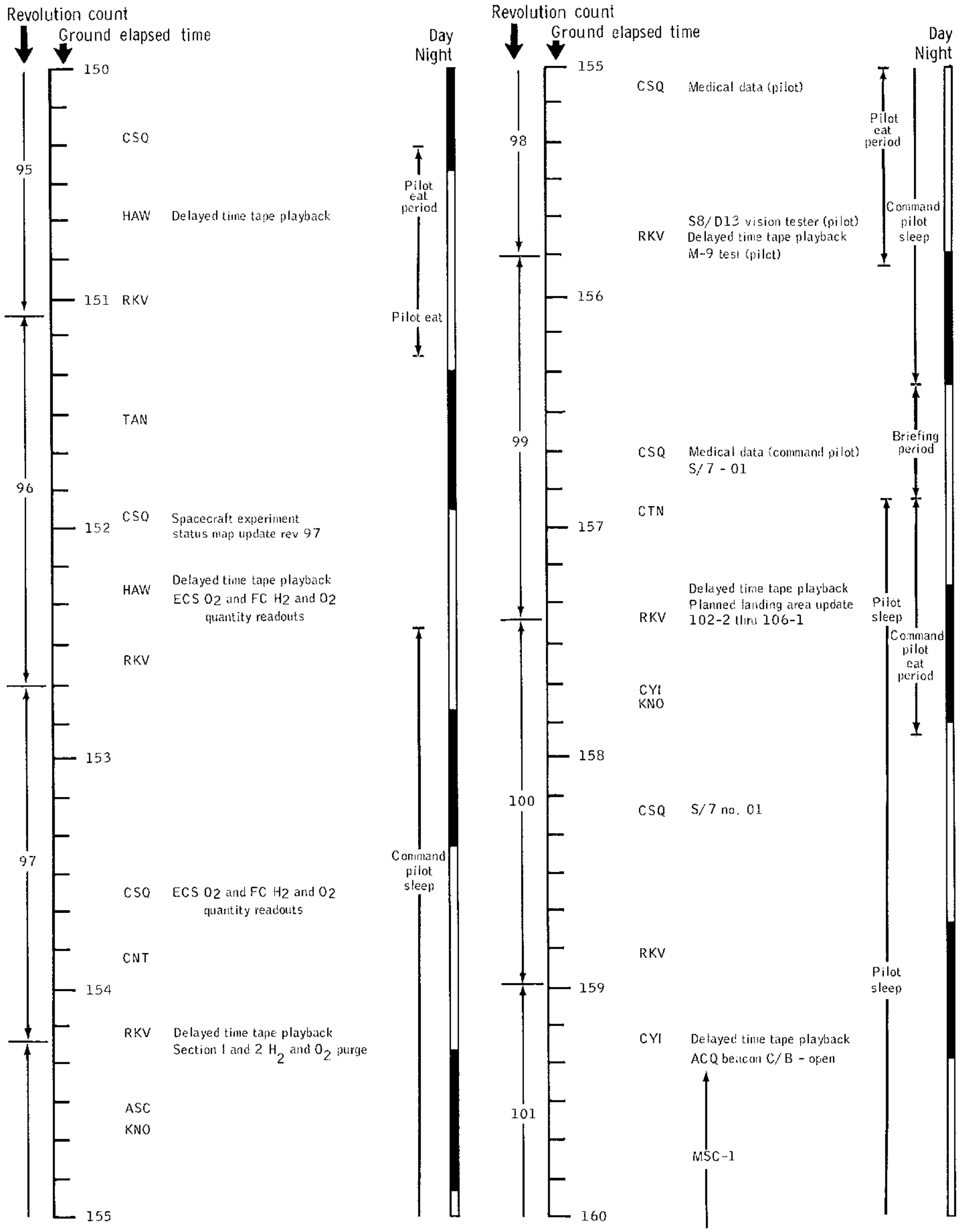
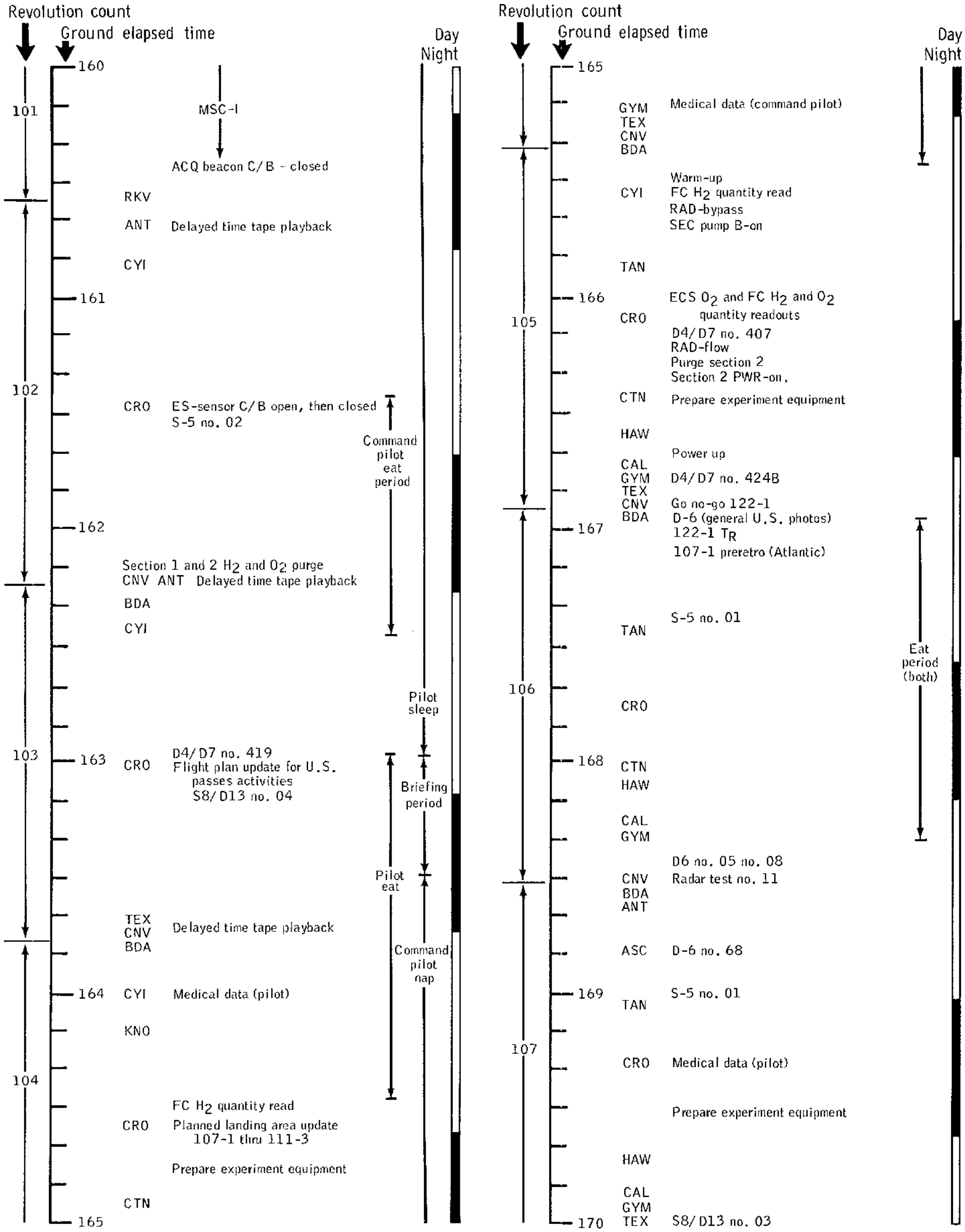
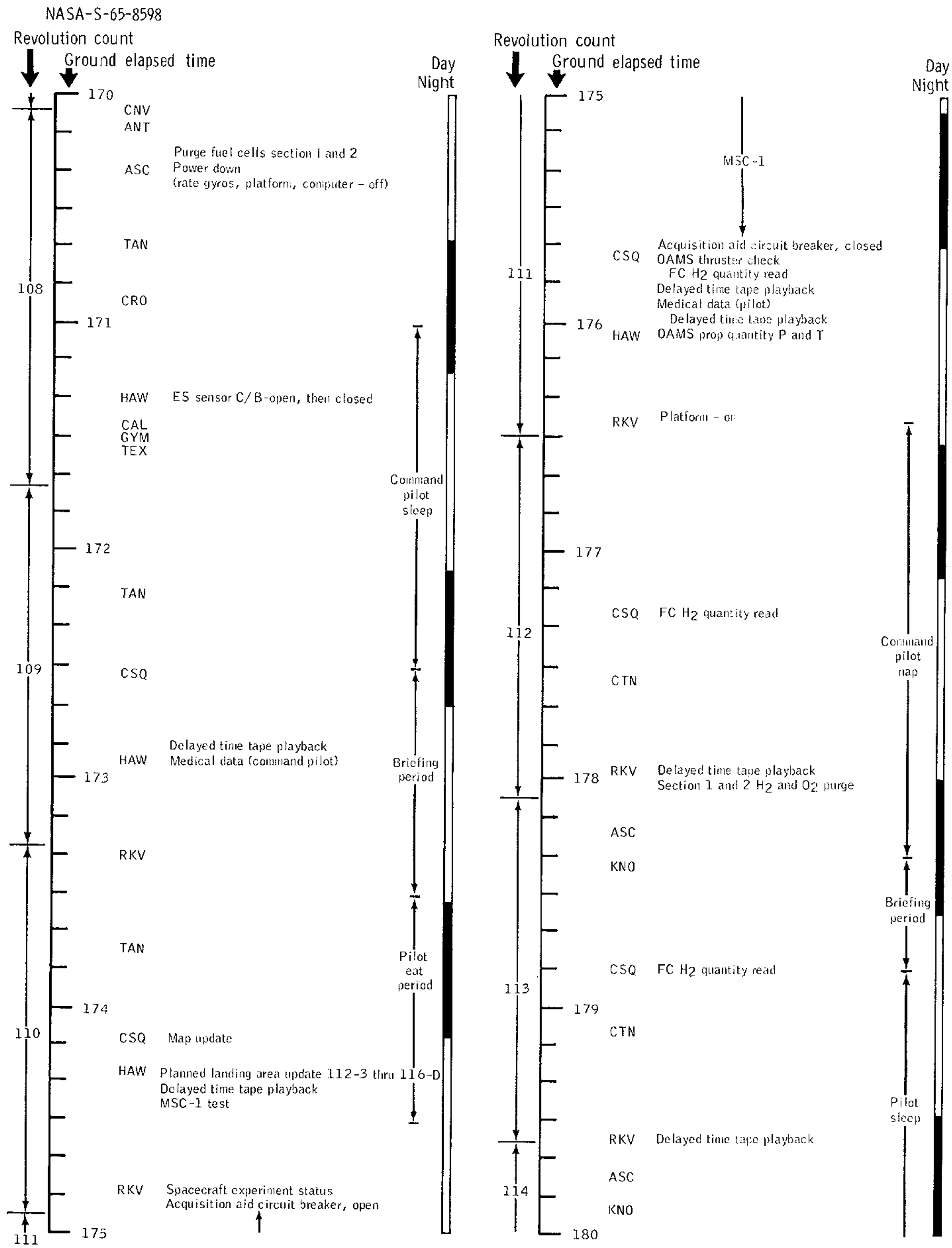


(p) 150 to 160 hours g.e.t.
Figure 7.1-1. - Continued.

NASA-S-65-8597

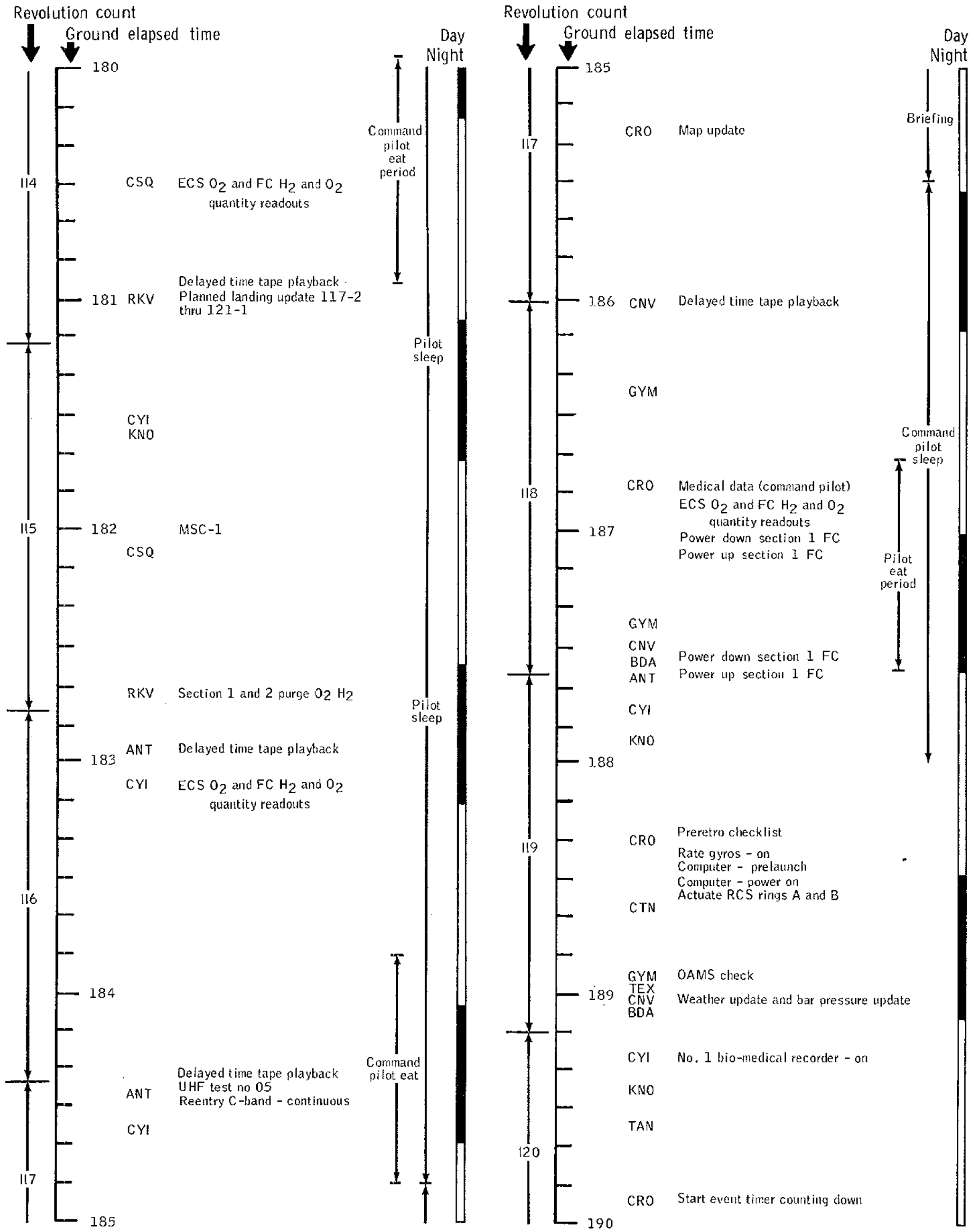


(q) 160 to 170 hours g.e.t.
Figure 7.1-1. - Continued.



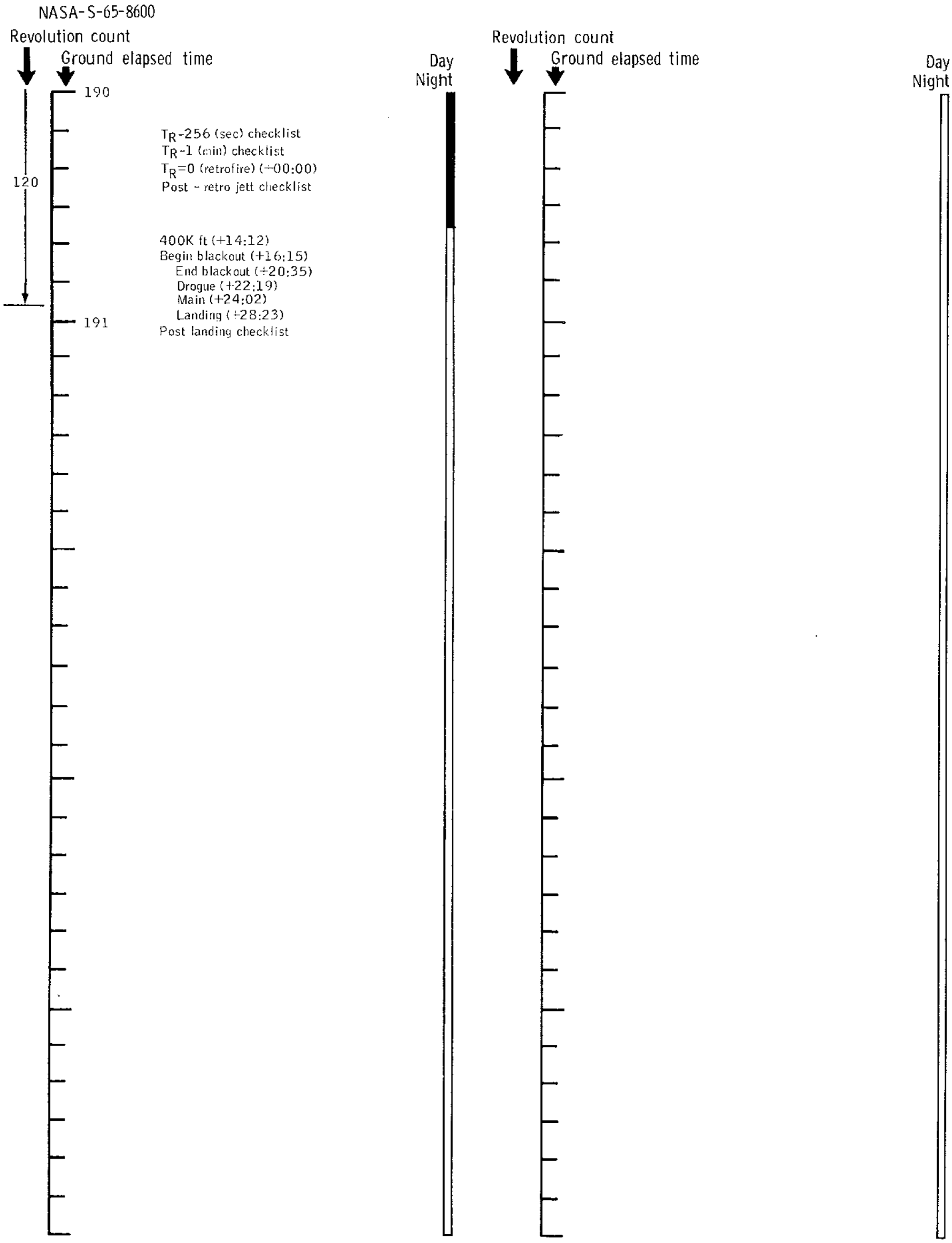
(r) 170 to 180 hours g.e.t.
Figure 7.1-1. - Continued.

NASA-S-65-8599



(s) 180 to 190 hours g.e.t.

Figure 7.1-1. - Continued.



(t) 190 through 191 hours g.e.t.
Figure 7.1-1. - Concluded.

NASA-S-65-9205

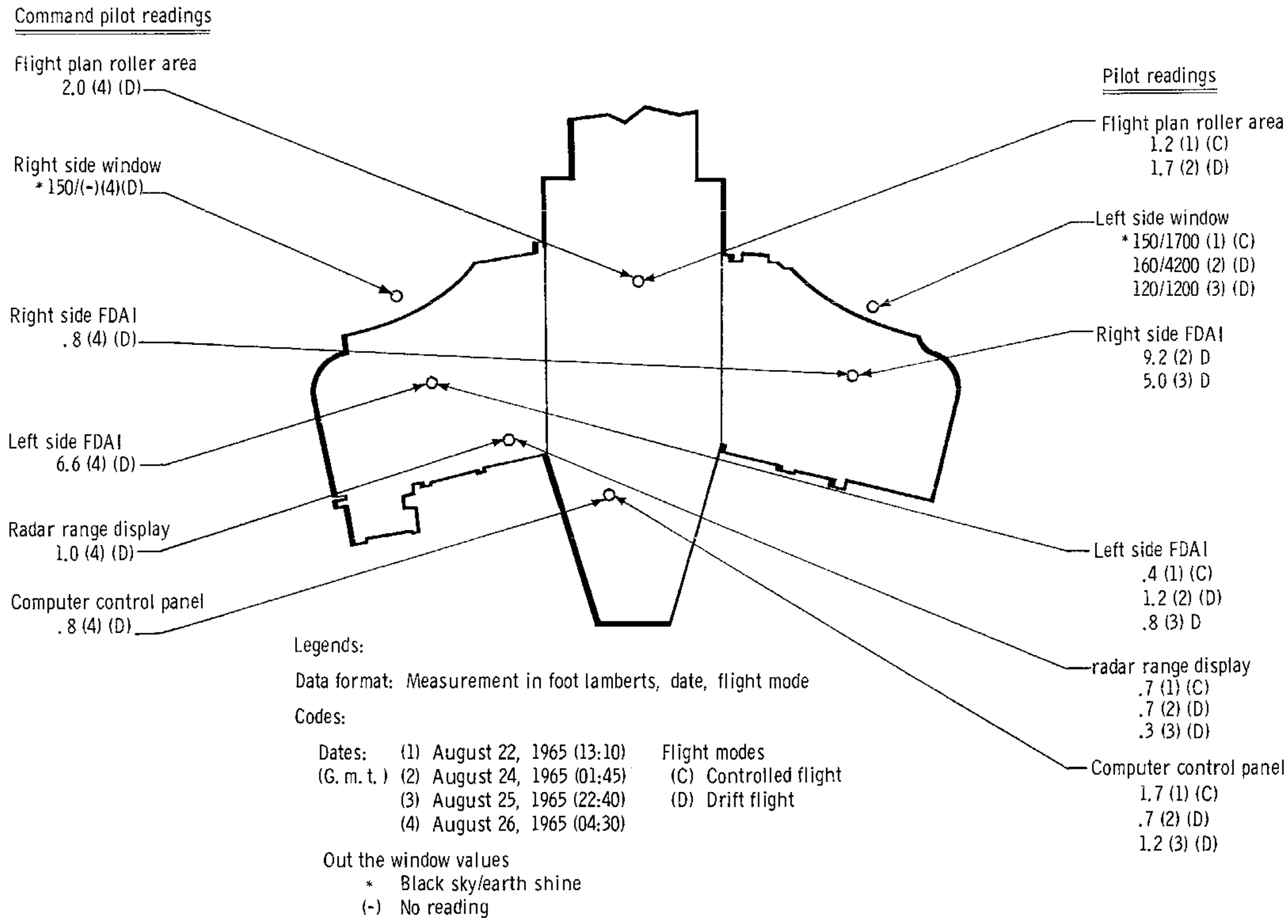


Figure 7.1-2. - Spacecraft cabin lighting survey, with cabin lights on, during day part of revolution.

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7.2 AEROMEDICAL

Gemini V provided an opportunity to study the physiological effects of space flight on two crewmen in the Gemini spacecraft for a period of 8 days. This flight clearly demonstrated the usefulness of man in space and the ability of man to function during prolonged space flight of this duration with no serious decrement in performance. It also demonstrated that man can readapt to normal gravitational forces after a space flight of this duration. There were many interesting aeromedical observations and significant findings during this flight. These data are presented in the following sections.

7.2.1 Preflight

7.2.1.1 Medical histories.- The medical histories from the flight crew consisted of their military health records, records of the medical examinations conducted at the time of their selection as astronauts, and their annual medical examinations since selection. In addition, a considerable volume of data has been collected during simulated flights, centrifuge training runs, and spacecraft systems tests (table 7.2-I).

7.2.1.2 Bioinstrumentation.- The standard Gemini bioinstrumentation system described in previous reports was used during this flight. A microphone placed on the anterior chest wall of each crewman was used to record their phonocardiograms for the M-4 experiment. These microphones were positioned to the left of the sternum at the fourth intercoastal space. In addition to the standard Gemini instrumentation, strain gages were used to measure the difference in circumference of the legs during all tilt-table studies.

7.2.1.3 Preflight tilt-table studies.- Three preflight tilt studies were accomplished on the Gemini V prime crew. These studies were used as a baseline or normal tilt response and postflight responses were compared to these normals.

7.2.1.4 Preflight diet.- Approximately 3 weeks prior to the scheduled mission date, the crew began a low-residue diet. Three days later a diet of the programed inflight food was begun. This subjective evaluation was terminated during the second day because of the onset in both crewmen of a mild upper respiratory illness associated with general malaise and soft stools. As a result of this subjective evaluation and a continued evaluation by the backup pilot, the crew elected to alter their original selection of bite-size food to include a number of rehydratable items.

7.2.1.5 Preflight medical examination.- Ten days prior to the scheduled launch, an examination of both the prime and backup crew members revealed no abnormalities. Signs and symptoms of the above-mentioned illness were found to be completely resolved except for the hematologic changes seen in table 7.2-II. A comprehensive examination of the prime and backup crew was conducted 2 days prior to the scheduled launch. This examination was conducted by the medical evaluation team which included an internist-cardiologist, ophthalmologist, otorhino-laryngologist, a neuropsychiatrist, and a flight surgeon. Due to a delay in the launch date, this examination actually occurred 4 days prior to launch. At this time, no significant abnormalities were found and the crew was considered medically ready for flight. Hematological studies done in connection with these examinations are reported in tables 7.2-II and 7.2-III. The preflight blood volume and red cell mass determinations are reported in table 7.2-IV. A brief preflight physical examination was conducted by the flight surgeons on the morning of the successful launch. No significant changes were found, and the crew was again considered medically fit for flight.

7.2.1.6 Miscellaneous preflight activities.- The flight crew elected to move into the astronauts' quarters in the Manned Spacecraft Operations Building at the Kennedy Space Center approximately 1 month prior to flight. This afforded them the necessary privacy for study and preparation, and it minimized inadvertent exposure to communicable diseases. At the time of the upper respiratory illness mentioned in section 7.2.1.4, an alpha-hemolytic streptococcus organism was cultured in the throat of the backup command pilot. He was treated with antibiotics and his throat culture rapidly reverted to normal. Throat cultures on all other crew members were repeatedly normal.

Operational requirements planned for the first day of flight made it highly desirable to delay defecation until after this critical period. After thorough consideration by the medical director, a mild laxative was given to both crew members 2 days prior to flight.

All crew members were tested for sensitivity to the biosensoring agents and onboard medications. No abnormal reactions were found.

7.2.1.7 Sensoring, suiting, and checkout.- Sensoring, suiting, and checkout were accomplished as for previous flights. The M-1 cuffs described in section 8 were fitted to the pilot and connected to the common blood pressure/M-1 suit fitting. Satisfactory records of the electrocardiogram, impedence pneumogram, blood pressure, and phonocardiogram were obtained prior to departure from the crew preparation trailer for ingress into the spacecraft.

7.2.1.8 Denitrogenation.- Denitrogenation was begun 127 minutes prior to launch as the suits were being purged with oxygen. Oxygen concentration in both suits was determined to be 100 percent, 3 minutes after the helmet visors were closed. The visors were not opened again until after launch.

7.2.2 Inflight

The inflight portion of the aeromedical report includes events from lift-off to spacecraft landing, an elapsed time of 190 hours 56 minutes.

7.2.2.1 Physiological measurements.- Physiological measurements obtained from the Gemini bioinstrumentation system and certain environmental parameters were monitored by physicians at the Mission Control Center, Houston (MCC-H), and at the remote network tracking sites. The electrocardiograms, pheumogram, and blood-pressure tracings on each crewman were relayed to the MCC-H over the voice-data lines either during a pass over the station or immediately after the pass. The quality of the analog data received at MCC-H was satisfactory for clinical analysis; however, there was some decrement in signal evident in the blood-pressure records.

7.2.2.1.1 Electrocardiograms: The rates and patterns of the electrocardiogram on each crewman remained within normal and expected limits. During the flight, a detailed analysis of the electrocardiogram was made during each pass by the remote site physician and/or the physicians at the MCC-H for rates, patterns, and intervals. The rates were transformed into graphs in the Staff Support Room at MCC-H and further analyzed for trends or significant findings. Each crewman's electrocardiogram was also recorded on the onboard biomedical tape recorders. Significant periods of this record were reviewed during the postflight analysis. In addition, a beat-by-beat rate was derived from the onboard tape recorded data and averaged for each 15-minute interval over the entire flight. For the purpose of this report, figure 7.2-1 shows a 4-hour average of rates for the complete flight along with respiration, blood pressure, and oral temperature findings. This record also depicts the Florida day and night cycle, as well as an approximation of each subject's sleep periods. Peak heart rates associated with specific activities or events are also shown in this figure.

7.2.2.1.2 Respiration: The respiratory rates, as measured by the impedence pneumogram, were within the expected normal range and are shown in figure 7.2-1.

7.2.2.1.3 Blood pressure: Eighty-nine blood-pressure measurements were obtained during this flight. Most of these were associated with exercise periods which were a part of the M-3 experiment described in section 8. An O-ring seal on the blood-pressure bulb fitting was damaged twice early in the flight and replaced each time by the pilot. The pilot also repaired (tightened the screws in the flange seal) the suit fitting common to the M-1 and blood-pressure measuring systems during the second day. Other than these minor failures, the blood-pressure measuring system performed satisfactorily and provided a considerable amount of significant information. All blood pressures obtained were completely within normal limits. It was observed, however, that the pilot's blood pressure appeared to be more labile in response to exercise and excitement than the command pilot's. There also appeared to be a narrowing of the postexercise pulse pressure toward the end of the flight on the command pilot's record. This finding was also present on the postflight examination and may relate to the extreme narrowing of pulse pressure during the postflight tilts. It cannot be determined at this time whether this is because of individual differences in the crew or represents a benefit derived from the M-1 experiment.

During this flight, it was found that the blood-pressure readings obtained by the remote site physicians did not correlate with the readings at the MCC-H. The records of all blood pressures obtained at remote sites were returned to Houston and compared with records received at MCC-H. The problem was determined to have been caused by inherent errors in reading the rapidly decreasing post-exercise blood pressure, coupled with a minor calibration and noise problem in the transmission of data over the voice-data lines to MCC-H. For this report, the blood-pressure readings by the remote site physicians have been confirmed and are graphically presented in figure 7.2-1.

7.2.2.1.4 Oral temperature: The oral temperature on both crew members was measured regularly during the medical data passes. A graph of these temperature is shown in figure 7.2-1. Inasmuch as the obvious inaccuracies of oral temperatures are recognized, it is interesting to note that the oral temperature trace very closely followed the circadian rhythm for the first 4 days of flight. These readings on both crew members appeared to stabilize after the fourth day. No such adaptation is seen in the mean-pulse rate and no adaptation is seen in the pilot's sleep pattern. This may serve to indicate the importance of body temperature as a basic tool in the study of circadian rhythm associated with space flight. This study has obvious operational importance in future prolonged manned space flight.

7.2.2.2 Medical observations.-

7.2.2.2.1 Environment: The environmental control system functioned well during the flight. The cabin temperature ranged between 70° and 79° F with a relative humidity of 72 percent or less. The suit inlet temperature ranged between 50° F and 55° F for most of the flight and could be raised to 65° F to 70° F when the coolant flow to the suit heat exchanger was turned to the full off position. Both crewmen were cool during the entire flight except during periods of maximum physical activity. At times, they were uncomfortably chilled. Both crew members flew the entire flight, with the exception of lift-off and reentry, with their helmets and gloves off. They were isolated from the suit flow by neck dams and wrist dams. There was no nose or throat irritation during this flight and there were no persistent disagreeable odors in the cabin.

7.2.2.2.2 Food and water: For this flight, it had been planned that a menu of freeze-dehydrated bite-size food would be utilized in order to save storage space in the spacecraft. However, shortly before the flight, it was found that, although the bite-size food was sufficiently nourishing and tasty when sampled, if eaten for every meal, it became excessively dry and rich and, because it had to be chewed thoroughly, caused actual fatigue in the masseter muscles. For these reasons, the menu was changed prior to flight to give slightly over 2700 calories per man per day. Twenty-seven separate meals were provided for each crewman, each one consisting of a mixture of the bite-size food and the rehydratable food. The crew, by choice, ate very little of the bite-size food. They did notice a lift in energy level after eating, and stated that the rehydratable food was very tasty. Their total food intake was slightly less than 1000 calories per day, which is certainly less than the programed 2700 calories and can account for some of the loss in weight which was experienced.

The crew did not have a satisfactory method of measuring the drinking water consumed; however, they did keep an accurate log of the swallows. Prior to the flight, their normal swallows had been roughly calibrated to equal a fluid ounce. Sixteen of these fluid ounces were taken to equal 1 pound of water. Although the inherent inaccuracies of this system of water measurement are quite obvious, it served to give an adequate check to monitor the crew's fluid intake. Each crewman drank approximately 6 pounds of water per day and approximately 1 pound of water after landing while awaiting recovery.

7.2.2.2.3 Waste: Each crewman urinated approximately four to five times a day and experienced no difficulty with the procedure. There was very little leakage from the new roll-on urine receptale. Each crewman had two bowel movements during the flight. These were relatively soft and difficult to manage. On the seventh day each crewman took medication in a successful effort to prevent additional bowel movements.

7.2.2.2.4 Sleep: It was planned for each crewman to have a 6-hour sleep period and a 2-hour nap period during each 24 hours of flight. These periods were to be separated by a 30-minute briefing period. Early in the flight, it was found that nearly all of a 24-hour period had been allocated as sleep time or nap time for either one crew member or the other, leaving very little time for certain experiments or other activities that required mutual participation. Moreover, it was found that since the suit fan noise and other noises associated with the suit circuit air flow were essentially isolated from the crewmen by the neck and wrist dams, the spacecraft became very quiet. It was impossible for one crewman to do anything without awakening the other. Such minor noises as clicking a camera or turning pages in the flight plan were enough to awaken the other crewman. The crew estimated their sleep time during a medical data pass each day, and while there was no accurate way to measure their actual sleep, it became apparent that, after the first 24 hours, they were averaging approximately 5 to 6 hours each day. It was also apparent that they had naturally reverted to sleeping at the same time. Their sleep periods can be seen in figure 7.2-1 to coincide with the latter part of the Florida night. The diurnal swings in temperature and heart rate coincide roughly with the sleep periods and with the Florida nights.

7.2.2.2.5 Personal hygiene: Prior to the flight the crew had bathed for several days with soap containing hexachlorophene. Their flight undergarments had also been laundered with soap containing this agent. During the flight, they did not feel that body odors were a problem. However, toward the end of the flight, the pilot noticed some disagreeable odors in the suit circuit. Dandruff became a major problem in that when they moved their heads, a cloud of dandruff would appear. This cloud would settle on parts of the spacecraft and at times caused some difficulty reading the instrument panel. Neither crewman complained of disagreeable dental problems. The bristles on the toothbrush were too hard to be effective and chewing gum was used only occasionally. They did not use the dental floss and felt no need for additional oral hygienic measures.

7.2.2.2.6 Reentry: The g-forces of reentry caused no medical problems. There was, as expected, an increase in the heart rate during retrofire and reentry. This increase started earlier than anticipated, especially in the pilot. However, this can be explained by the complicated storage procedure which was begun several hours prior to retrofire. Although no medication was prescribed by the MCC-H surgeon, both crewmen took a stimulant 2 hours prior to retrofire. The highest heart rates during the flight were experienced during reentry and correspond with the time of concern over the reentry guidance error and premature deployment of the drogue parachute. When the spacecraft was commanded to the two-point suspension landing attitude, the crewmen were easily able to brace themselves and did not impact with any part of the spacecraft. Throughout reentry, no symptoms indicative of hypotension were reported. However, after landing the pilot felt that his legs were heavy. After leg exercises, they felt completely normal. Both crewmen state that there were no symptoms of dizziness, lightheadedness, blurring of vision, or nausea during reentry or at any time awaiting recovery.

7.2.3 Postflight

This portion of the report includes medical information gathered after the time of spacecraft landing. These data were obtained during clinical examinations, medical debriefing, and by laboratory examinations of blood, urine, and feces. Postflight deviations from normal were limited to the following: (1) transient reduction in pulse pressure and elevation of heart rate during the postflight tilt procedures which were greater than the preflight normals, (2) mild crew fatigue, and (3) body fluid changes.

7.2.3.1 Recovery activities.- Medical recovery activities were planned in advance of the mission and were modified as dictated by the observed medical responses of the crew.

7.2.3.1.1 Planned recovery procedures: Following recovery and suit removal, a detailed examination by the medical evaluation team who examined the crew preflight was planned. Tilt procedures were planned twice on the day of recovery and daily thereafter until the responses had returned to preflight values.

7.2.3.1.2 Narrative: Significant postflight medical events are listed in table 7.2-V. After landing, the crew reported that they were comfortable in the spacecraft and elected to remain in their pressure suits. The pilot ingested a single 50-mgm cyclizine hydrochloride tablet shortly after water landing. The crew egressed without difficulty and were immediately taken aboard the rescue helicopter where both crew members stood up without difficulty. The NASA physician aboard the helicopter performed a brief examination and found no medical abnormalities in the crew. Immediately after landing aboard the aircraft carrier, the crew walked unassisted below decks to the ship's sick bay where the initial postflight medical examinations were performed. At no time during the recovery or postflight phase of the mission did the crew report any subjective symptoms of low blood pressure, except as noted in section 7.2.2.2.6.

7.2.3.2 Examinations.- A detailed medical examination was conducted by the medical evaluation team and by the NASA physicians aboard as soon as the crew arrived in sick bay. The examination protocol is shown in table 7.2-VI. With the exception of body fluid changes and tilt responses, no significant abnormalities were noted during the examination. The findings are summarized in tables 7.2-II to 7.2-VIII.

Both crew members exhibited a moderate reaction to the micropore tape used to fasten the body sensors in place. There was a moderate amount of dead skin that peeled off at the time of underwear doffing in a manner similar to skin loss following a heavy sun exposure. There was no discomfort associated with this phenomenon. The 8-day beard growth was not matted and, following shaving, the facial skin was normal. The underwear was nearly saturated with perspiration, but appeared to be relatively clean. It was noted that odor was definitely less than on Gemini IV. There was no skin reaction at any sites other than where the biosensors had been attached. Specifically, there was no maceration, no change in skin turgor, and no evidence of pressure points. There was marked scaling noted in the command pilot's scalp and somewhat less in the pilot's. The Gemini V crew appeared to be better rested than the Gemini IV crew. They reported to be fully rested following $10\frac{1}{2}$ hours of sound sleep on the night following recovery. During the day of recovery, from R + 3.5 hours to bedtime at R + 14 hours, the command pilot drank 1530 cc fluid; the pilot consumed 1650 cc in the same period.

7.2.3.3 Tilt-table studies.- The same tilt-table procedure as used on previous flights was used on this mission with the following modifications: (1) the same three individuals performed all of the tilts, (2) the saddle was modified to allow partial deflation for more subject comfort, (3) the leg strain gages were placed in the same position each time by measuring not only the leg circumferences but also the distance cephalad from the medial malleolus, and (4) the tension of these gages was calibrated before each tilt to insure that measurements were obtained where the response of the gage was linear. A total of six postflight tilt studies were performed on each crewman. The first postflight tilt procedure revealed significant elevation of heart rate and decrease in pulse pressure in both crewmen, although no symptoms were noted at any time. This tilt response returned to preflight normals as shown by figures 7.2-2 and 7.2-3. These cardiovascular responses are believed to have occurred because of physiologic alterations, although the individual crewman's tilt responses were influenced by a number of individual, operational, and environmental variables. This physiological change did not in any way compromise the crew's ability to function during the inflight or postflight phases of the mission. The strain gages gave a reliable indication of the increase in leg circumference during the 70° head-up tilt. From these readings a calculation of mean change in leg volume was derived. These changes were significantly higher on both subjects during the postflight tilts as compared with preflight normals. Present information indicates that the increase in leg volume is a reflection of altered physiology which could mean a pooling of blood in the lower extremities.

7.2.3.4 Radioisotope studies.- Plasma volume, circulating red cell mass, and red cell survival determinations were performed shortly before launch and after recovery. All plasma volume measurements were made by the $RISA_{125}$ technique, while the red cell mass and red cell survival times were accomplished with Cr_{51}. The calculated total amount of effective radiation received per crew member for the entire series of isotope tests was 88 millirem. The measured radiation received during the mission is reported in table 7.2-IX. All injections were made intravenously, without extravasation or other untoward consequences. All samples were prepared in duplicate for counting. Analysis of the counting statistics shows excellent reliability. Comparison of expected and observed baseline values exhibit insignificant variations.

In both crewmen the total circulating blood volumes were reduced postflight as tabulated in table 7.2-IV. Quantitatively, this amounted to a 13-percent decrease or 592 cc and 547 cc for the command pilot and pilot, respectively. Analysis of the red cell mass and plasma volume data reveals the circulating blood volume deficit is due primarily to a loss of red cell mass, with only small decreases of the plasma volumes. Actual values for both crewmen show a 20-percent decrease in red cell mass, whereas plasma volume deficits of only 8 percent and 4 percent were observed.

Three possible basic mechanisms exist which could explain the observed decrease in red cell mass: (1) reduced red cell production with a normal red cell destruction rate, (2) normal red cell production with an increased red cell destruction rate, and (3) sequestration of red cells or redistribution of the circulating red cell mass. At this time, direct evidence exists in support of the first two hypotheses. Significantly, reduced reticulocyte counts were observed immediately after the flight interval. Average reticulocyte counts prior to flight were 1.9 percent, whereas postflight counts averaged 0.77 percent. The average red cell survival times showed a decrease from the normal $T\frac{1}{2}$ Cr_{51} range of 22 to 29 days to 18 and 16.6 days for the command pilot and pilot, respectively. Extrapolation to the percent tagged red cells present 8 days postlaunch reveals values of 70 percent and 69 percent as compared with the normal range of 82 percent to 83 percent. In light of the hematologic picture and directly observed red cell mass and red cell survival data, it may be postulated that both decreased red cell production and increased red cell destruction existed during the 8-day flight. Evidence in favor of the third hypothesis is slight; however, comparison of the peripheral hematocrits and the calculated total body hematocrits suggests a redistribution of the circulating red cells or, more accurately stated, a greater percentage of red cells per unit volume in the periphery than centrally. Clinical observations do not support the contention that significant sequestration of red cells exists in processes such as hematoma formation, intracavitary bleeding, et cetera. Examination of all stool specimens produced during flight show less than 0.015 cc of blood per stool. This value is well within the limits of normal.

TABLE 7.2-I.- PREFLIGHT MEDICAL STUDIES AND ASSOCIATED ACTIVITIES

Date, 1965	Activity	Medical study or support
May 18 and 19	Simulated flight (SST)	Examination before and after tests. Biosensors used during test.
June 1	Spacecraft checkout in altitude chamber (SST)	Examination before and after tests. Biosensors used during test.
July 22	Wet mock simulated launch	Brief examination before test. Biosensors used during test.
August 5	Tilt-table test 1; exercise baseline; counter rolling[a]	Biosensors and strain gages used. No instrumentation for counter rolling.
August 11	F-10 day examination by flight surgeon; tilt-table test 2; exercise baseline	Complete physical examination, including blood and urine. Densitometry; biosensors and strain gages used.
August 14	Plasma volume, red cell mass, CBC	Radioisotope studies (I_{125} and Cr_{51})
August 15	Red cell survival	Blood specimen
August 17	F-2 day medical examination; tilt-table test 3; exercise baseline	Examinations by Medical Evaluation Team; biosensors and strain gages used.
August 19	Launch morning examination by flight surgeon. Mission postponed, crew egress	Final brief clinical examination. No hematology.
August 21	Repeat of August 19 examination; launch at 9:00 a.m. e.s.t.	

[a]See section 8 for experiment M-9

TABLE 7.2-II.- BLOOD STUDIES - COMMAND PILOT

(a) Chemistries

Determination	Preflight		Postflight		
Date, 1965 Time, e.s.t.	August 11 7:30 a.m.	August 17 8:00 a.m.	August 29 11:00 a.m.	August 29 9:15 p.m.*	September 1 8:45 a.m.*
Blood urea nitrogen (BUN), mg percent	---	22	16	25	---
Bilirubin, direct, mg percent	0.2	---	0.2	0.1	0.1
Bilirubin, total, mg percent	0.7	1.0	0.8	0.5	0.4
Alkaline phosphatase (B-L units)	0.9	1.2	1.1	1.2	1.1
Cholesterol, mg percent	---	255	205	235	205
17-OH corticosteroid, mg percent	17.0	15.0	14.2	17.0	---
Sodium, m Eq/l	145	143	139	146	137
Potassium, m Eq/l	4.9	4.2	3.9	4.6	4.4
Chloride, m Eq/l	106	103	101	106	104
Calcium, mgms percent	9.6	10.5	9.1	9.5	9.0
Phosphate, gm percent	3.7	3.4	3.5	4.2	2.8
Glucose, mgm/100 ml	72	78	94	84	79
Albumen, gm percent	4.7	4.4	4.9	4.7	4.1
Alpha 1, gm percent	0.2	0.1	0.2	0.2	0.2
Alpha 2, gm percent	0.4	0.5	0.6	0.6	0.6
Beta, gm percent	0.9	0.8	0.8	0.8	0.8
Gamma, gm percent	0.6	1.4	1.3	1.5	1.2
Total protein, gm percent	6.8	7.2	7.8	7.9	6.9
Electrophoretic pattern	Normal	Normal	Normal	Normal	Normal

*Non-fasting

TABLE 7.2-II.- BLOOD STUDIES - COMMAND PILOT - Concluded

(b) Hematology

Determination*	Preflight		Postflight	
	F-10 days	F-4 days	R+2 hrs	R+8 hrs
White blood cells /mm^3	4537	6850	7125	9200
Neutrophiles, percent	29 (1316)	34 (2329)	70 (4988)	72 (6624)
Lymphocytes, percent	58 (2631)	61 (4179)	23 (1639)	24 (2208)
Monocytes, percent	11 (449)	5 (343)	6 (428)	3 (276)
Eosinophiles, percent	2 (44)	0	1 (16)	1 (92)
Basophiles, percent	0	0	0	0
Red blood cells, millions/mm3	4.97	5.17	5005	-
Hematocrit, percent	44	47	47	43
Mean corpuscular, volume, M^3	89	91	13	-
Total serum protein, gm percent . . .	6.8	7.2	7.3	7.1
Reticulocyte count, percent	2.2	-	1	-
Hemoglobin, gm/100 ml	15.5	15.5	-	-
Platelets/mm^3	176 000	-	222 500	-

*Figures in parentheses are absolute cell counts

TABLE 7.2-III.- BLOOD STUDIES - PILOT

(a) Chemistries

Determination	Preflight		Postflight		
Date, 1965 Time, e.s.t.	August 11 7:30 a.m.	August 17 8:00 a.m.	August 29 11:45 a.m.	August 29 9:15 p.m.*	September 1 8:30 a.m.*
Blood urea nitrogen (BUN), mg percent	---	24	17	23	19
Bilirubin, direct mg percent	0.1	0.1	0	0.1	0
Bilirubin, total mg percent	0.2	0.8	0.4	0.3	0.2
Alkaline phosphatase (B-L units)	1.3	1.8	1.6	1.4	1.4
Cholesterol, mg percent	---	255	185	235	205
17-OH corticosteroid, mg percent	20.8	25.0	7.1	9.2	15.7
Sodium, m Eq/l	146	148	144	143	136
Potassium, m Eq/l	4.6	4.4	4.4	4.5	4.9
Chlorida, m Eq/l	103	107	106	105	105
Calcium, mgms percent	10.2	9.9	8.8	9.7	9.3
Phosphate, gm percent	3.6	3.0	3.4	3.7	4.2
Glucose, mgm/100 ml	93	100	92	100	100
Albumen, gm percent	4.3	4.6	5.1	4.2	4.6
Alpha 1, gm percent	0.2	0.1	0.2	0.2	0.1
Alpha 2, gm percent	0.6	0.7	0.7	0.8	0.6
Beta, gm percent	1.0	0.8	0.8	0.9	0.7
Gamma, gm percent	1.0	1.2	1.1	1.5	1.2
Total protein gm percent	7.1	7.4	7.9	7.6	7.2
Electrophoretic pattern	Normal	Normal	Normal	Normal	Normal

*Non-fasting

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TABLE 7.2-III.- BLOOD STUDIES - PILOT - Concluded

(b) Hematology

Determination*	Preflight		Postflight	
	F-10 days	F-4 days	R+2 hrs	R+8 hrs
White blood cells/mm^3	6575	9650	11 150	9400
Neutrophiles, percent	33 (2170)	42 (4053)	71 (7917)	61 (+734)
Lymphocytes, percent	53 (3485)	48 (4632)	19 (2119)	33 (3102)
Monocytes, percent	11 (723)	6 (389)	9 (1004)	4 (376)
Eosinophiles, percent	1 (66)	3 (290)	0 (11)	0
Basophiles, percent	1 (66)	1 (96)	1 (112)	1 (94)
Red blood cells, millions/mm^3	5.36	5.32	5.30	-
Hematocrit, percent	44	49	47	44
Mean corpuscular volume, M^3	87	92	88	-
Total serum protein, gm percent	7.1	7.4	7.8	7.1
Reticulocyte count, percent	1.8	-	0.55	-
Hemoglobin, gm/100 ml	15.0	15.9	-	-
Platelets/mm^3	191 580	-	157 250	-

*Figures in parentheses are absolute cell counts

TABLE 7.2-IV.- BLOOD RADIOISOTOPE STUDIES (I_{125} and Cr_{51})

Determination	Preflight			Postflight	
	Normal	Observed	Difference	Observed	Change from preflight
Command Pilot					
Blood volume, cc	4341	4267	-74	3675	-592 (-13 percent)
Plasma volume, cc	2388	2354	-34	2145	-209 (- 8 percent)
Red cell mass, cc	1953	1913	-40	1530	-383 (-20 percent)
Body hematocrit, percent . . .	45	45 (46)[a]	--	42 (47)[a]	--
Pilot					
Blood volume, cc	4232	4306	+74	3759	-547 (-13 percent)
Plasma volume, cc	2328	2300	-28	2194	-106 (- 4 percent)
Red cell mass, cc	1904	2006	+102	1565	-441 (-20 percent)
Body hematocrit, percent . . .	45	47 (48)[a]	--	42 (46)[a]	--

[a]Venous hematocrit, peripheral, percent, in parenthesis

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TABLE 7.2-V. - POSTFLIGHT EVENTS AND MEDICAL ACTIVITIES

Date, 1965	Time, e.s.t.	Activity
August 29	7:55 a.m.	Spacecraft landing, 93 miles from U.S.S. Lake Champlain
	8:45 a.m.	Right hatch open, egress began
	8:58 a.m.	Both crewmen in helicopter
	9:26 a.m.	Arrived aboard U.S.S. Lake Champlain
	10:00 a.m.	Suits doffed
	10:35 a.m.	Began initial medical examination
	12:45 p.m.	First postflight meal (low calcium)
	4:15 p.m.	Completed initial medical evaluation
	7:45 p.m.	Second tilt procedure and blood specimens
	10:00 p.m.	To bed, asleep shortly thereafter
August 30	8:30 a.m.	Awoke, breakfast
	10:00 a.m.	Departed U.S.S. Lake Champlain by way of aircraft
	10:45 a.m.	Arrived launch site
	1:40 p.m.	Third tilt procedure; medical debriefing
August 31	8:00 a.m.	Fourth tilt procedure; third blood specimens; medical debriefing
September 1	8:45 a.m.	Fifth tilt procedure; medical debriefing
September 2	1:00 p.m.	Depart launch site for Houston
September 3	6:30 p.m.	Sixth (final) tilt procedure

TABLE 7.2-VI.- POSTFLIGHT MEDICAL EXAMINATION PROTOCOL

[In order of priority]

Command pilot		Pilot	
	Duration, min		Duration, min
X-ray - Chest X-ray Densitometry Blood - CBC Chemistry Isotopes studies Electrocardiogram	30	Tilt and exercise test	60
Ophthalmology	30	X-ray - Chest X-ray Densitometry Blood - CBC Chemistry Isotopes studies Electrocardiogram	30
Tilt and exercise test	60	Ophthalmology	30
Audiogram	15	Counter-rolling	45
Neuropsychiatry	30	Ear, nose, and throat	30
Counter-rolling	45	Internal medicine	30
Ear, nose, and throat	30	Audiogram	15
Internal medicine	30	Neuropsychiatry	30

Note: Both crewmen had nothing by mouth until the blood specimen had been taken and initial tilt-table procedures were completed.

TABLE 7.2-VII.- SUMMARY CLINICAL EVALUATION

(a) Command Pilot

	Preflight	Postflight		
	(Launch site) August 21, 1965 5:00 a.m. e.s.t.	(Shipboard) August 29, 1965 11:30 a.m. e.s.t.	(Shipboard) August 29, 1965 8:00 p.m. e.s.t.	(Launch site) August 30, 1965 2:40 p.m. e.s.t.
Body weight (nude), lb[a]	152	$144\frac{5}{8}$	150	$149\frac{3}{8}$
Temperature, oral, °F	98	98.6	99.2	98.4
Respirations, breaths/min . . .	14	16	16	18
Skin	Minimal neuro-dermatitis on chest; otherwise clear.	Moderate reaction at biosensor sites, moderate desquamation; otherwise clear, good turgor.	No change at sensor sites, otherwise normal (after shower).	Minimal skin clearing at sensor sites; otherwise normal.
Comments	Well conditioned normal male, fit for flight.	Alert, cooperative, steady, oriented, tired, minimally thirsty, hungry.	No longer hungry nor thirsty (after a meal); otherwise no change.	Rested.

[a]The shipboard scale was calibrated to the launch site scale; weights considered accurate to $\frac{1}{4}$ lb.

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TABLE 7.2-VII.- SUMMARY CLINICAL EVALUATION - Concluded

(b) Pilot

	Preflight	Postflight		
	(Launch site August 21, 1965 5:00 a.m. e.s.t.	(Shipboard) August 29, 1965 10:45 a.m. e.s.t.	(Shipboard) August 29, 1965 7:00 p.m. e.s.t.	(Launch site) August 30, 1965 1:30 p.m. e.s.t.
Body weight (nude), lb[a]	154	$145\frac{1}{2}$	$149\frac{7}{8}$	$150\frac{3}{4}$
Temperature, oral, °F	98.4	99.6	99.2	98.2
Respirations, breaths/min . . .	15	18	16	14
Skin	Clear, no lesions.	Moderate reaction at biosensor sites, moderate quantity of desquamation; otherwise clear, good turgor.	No change at sensor sites; otherwise normal (after shower).	Minimal skin clearing at sensor sites; otherwise normal.
Comments	Well conditioned normal male, fit for flight.	Steady, alert, oriented, cooperative, moderately tired, not thirsty, hungry.	No longer thirsty nor hungry (after a meal); otherwise no change.	Rested.

[a]The shipboard scale was calibrated to the launch site scale; weights considered accurate to $\frac{1}{4}$ lb.

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TABLE 7.2-VIII.- URINALYSIS

(a) Command Pilot

	Preflight					Postflight
	F-10 days	F-4 days	F-2 days (1st voiding)	F-2 days (2nd voiding)	Flight morning	R+4 hours
Volume, cc	300	86	375	70	50	300
Color	amber, clear	yellow, clear	yellow, cloudy	yellow, clear	yellow, clear	amber, cloudy
Specific gravity	1.025	1.025	1.015	1.022	1.020	1.025
pH	5.0	5.5	6.0	6.0	6.0	6.0
Albumen, sugar, occult blood	Negative	Negative	Negative	Negative	Negative	Negative
Microscopic	Pyuria, 4 to 6 white blood cells/hpf	1 to 2 white blood cells/hpf, rare red blood cells, occasional epithelial threads	6 to 10 white blood cells/hpf, 1 to 2 red blood cells/hpf, occasional epithelial and mucous threads	4 to 5 white blood cells/hpf, occasional epithelial, rare red blood cells	6 to 8 white blood cells/hpf, 1 to 2 red blood cells/hpf, infrequent cellular cast	4 to 5 white blood cells/hpf, large amount of uric acid crystals present
Epinephrine, micro gm/l	1.320	2.520	--	--	--	14.690
Norepinephrine, micro gm/l	6.810	6.160	--	--	--	16.490
17-OH corticosteroids, micro gm/l	4.53	5.97	--	--	--	10.67
Sodium, mEq/l	110	116	74	--	--	105
Potassium, mEq/l	24	29.6	18	--	--	44
Calcium, mEq/l	25.6	12.0	13.5	--	--	20.0
Calcium, mgm percent	51.2	24.0	27.0	--	--	40.00
Chloride, mEq/l	86	79.5	39	--	--	77
Phosphate, gm/l	1.12	0.69	0.45	--	--	1.20
Osmolarity, mOsm/kg	814	860	575	--	--	780
Creatinine, gm/l	1.4	1.58	1.94	--	--	3.0
Urea nitrogen, gm/l	14.8	15.7	9.46	--	--	13.0
Total nitrogen, gm/l	15.3	15.9	1P°3	--	--	13.5
Hydroxyproline, mg/l	27	48	51	--	--	78
Creatine, gm/l	--	--	--	--	--	0.27

TABLE 7.2-VIII.- URINALYSIS - Concluded

(b) Pilot

	Preflight				Postflight			
	F-10 days	F-3 days	F-2 days	Flight morning	R+6 hours	R+11 hours	R+24 hours	R+76 hours
Volume, cc	70	367	160	56	340	160	300	380
Color	yellow, cloudy	yellow, clear	yellow, clear	yellow, clear	yellow, clear	yellow clear	yellow, clear	yellow, clear
Specific gravity	1.030	1.029	1.020	1.025	1.020	1.025	1.025	1.020
pH	5.0	5.5	6.0	5.0	5.0	5.0	5.0	5.5
Albumen, sugar, occult blood	Negative	Negative	Negative	Negative	Negative	Negative	Negative	Negative
Microscopic	No cells; large amounts amorphous urates	2 to 3 white blood cells/hpf, rare red blood cells	1 to 2 white blood cells/hpf, occasional mucous tjreads	3 to 4 white blood cells/hpf, infrequent red blood cells, occasional epithelial	1 to 2 white blood cells/hpf, occasional mucous threads	--	--	1 to 2 white blood cells/hpf
Epinephrine, gm/l	1.080	1.810	--	--	17.290	--	1.000	1.000
Norepinephrine, gm/l	9.280	4.680	--	--	16.500	--	20.800	7.130
17-OH corticosteroids, mg/l	7.65	--	--	--	8.82	--	5.67	2.71
Sodium, mEq/l	61	104	99	--	72	--	31	139
Potassium, mEq/l	36	30	26	--	65.6	--	55.2	22
Calcium, mEq/l	26.4	10.4	10.7	--	12.8	--	10.2	15.8
Calcium, mgm percent	52.8	20.8	21.4	--	25.6	--	20.4	31.6
Chloride, mEq/l	52.0	89	65	--	74.0	--	20.0	99
Phosphate, gm/l	1.52	0.48	0.67	--	0.5	--	1.10	0.95
Osmolarity, mOsm/kg	**	940	860	--	625	--	865	874
Creatinine, gm/l	2.6	1.31	1.72	--	3.7	--	2.72	1.84
Urea nitrogen, gm/l	18.8	16.7	14.4	--	9.7	--	--	14
Total nitrogen, gm/l	21.2	18.4	16.4	--	10.2	--	19.9	16
Hydroxyproline, mg/l	qns	63	65	--	90	--	78	78
Creatine, gm/l	--	--	--	--	0.5	--	0.36	0.28

**Sample acidified

TABLE 7.2-IX.- CREW RADIATION

[The Gemini V radiation film badges were read out using a thermoluminescent detector]

Command Pilot

Film badge location	Dose, mr
Right chest pocket	173 ± 17.3
Left chest pocket	190 ± 19.0
Left thigh pocket	183 ± 18.3
Helmet	195 ± 19.5

Pilot

Film badge location	Dose, mr
Right chest pocket	172 ± 17.2
Left chest pocket	140 ± 14.0
Right thigh pocket	186 ± 18.6
Helmet	172 ± 17.2

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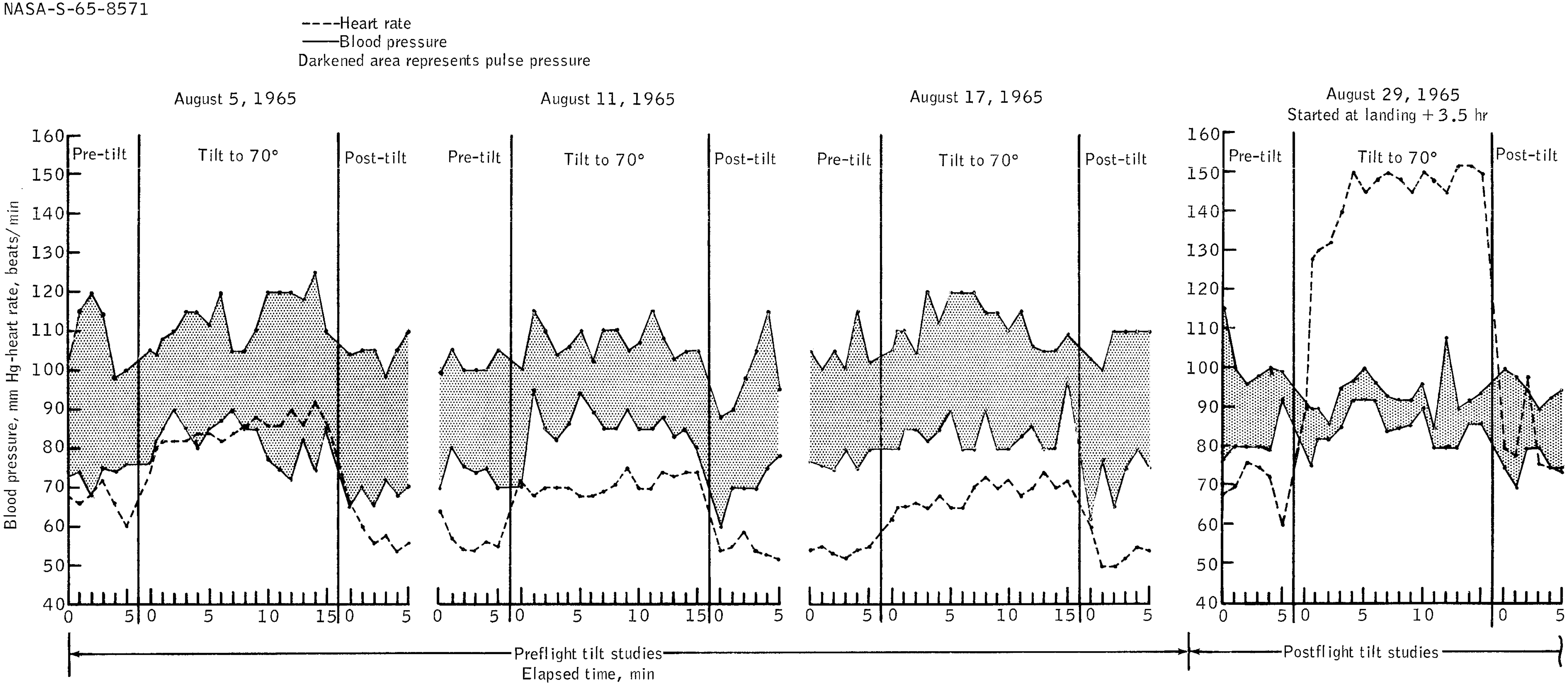
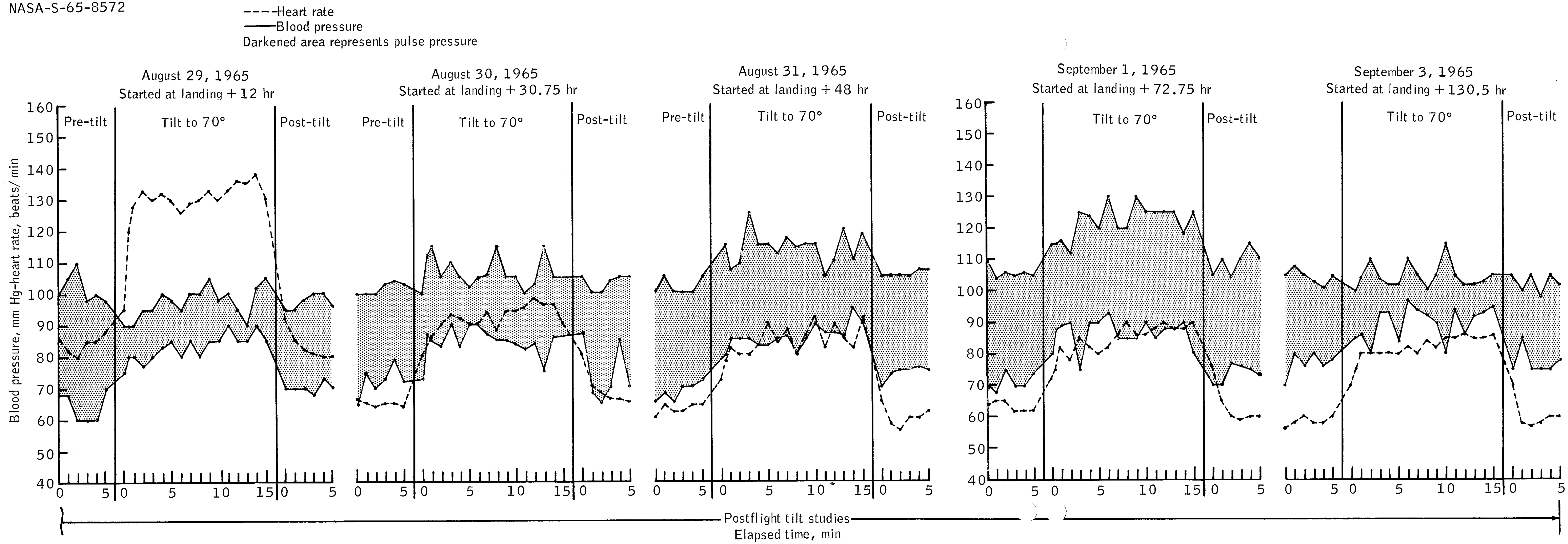


(a) Command pilot.

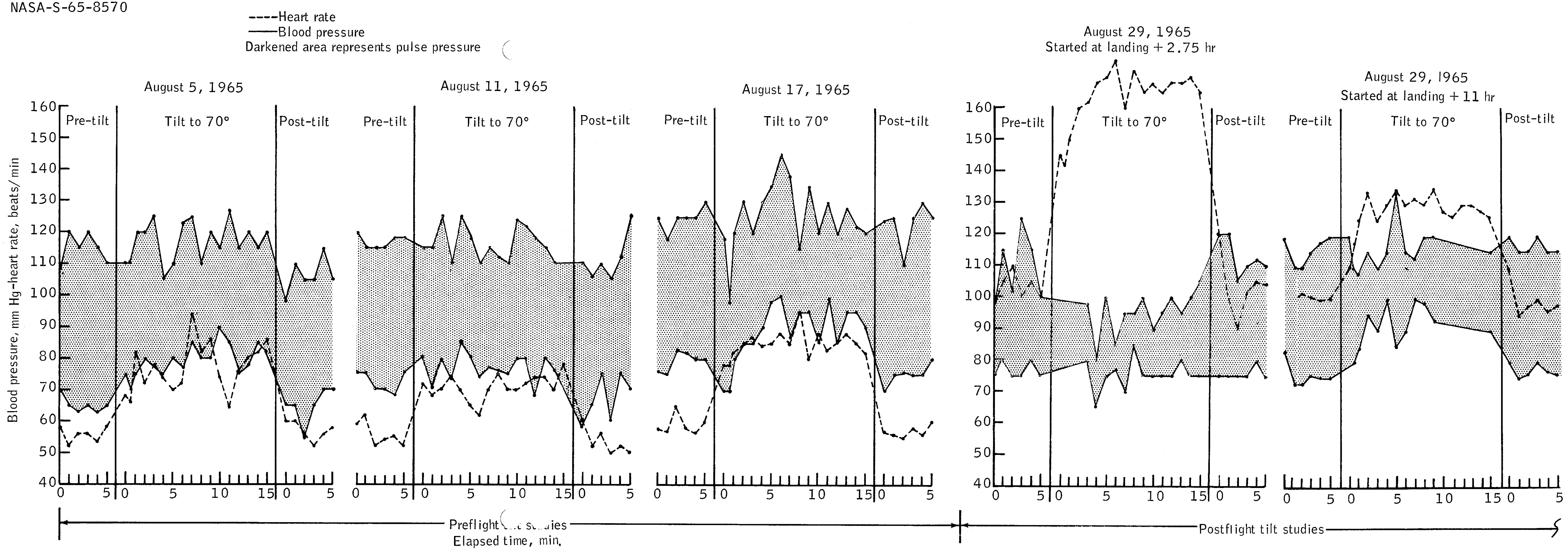
Figure 7.2-2. - Tilt table studies.

NASA-S-65-8572



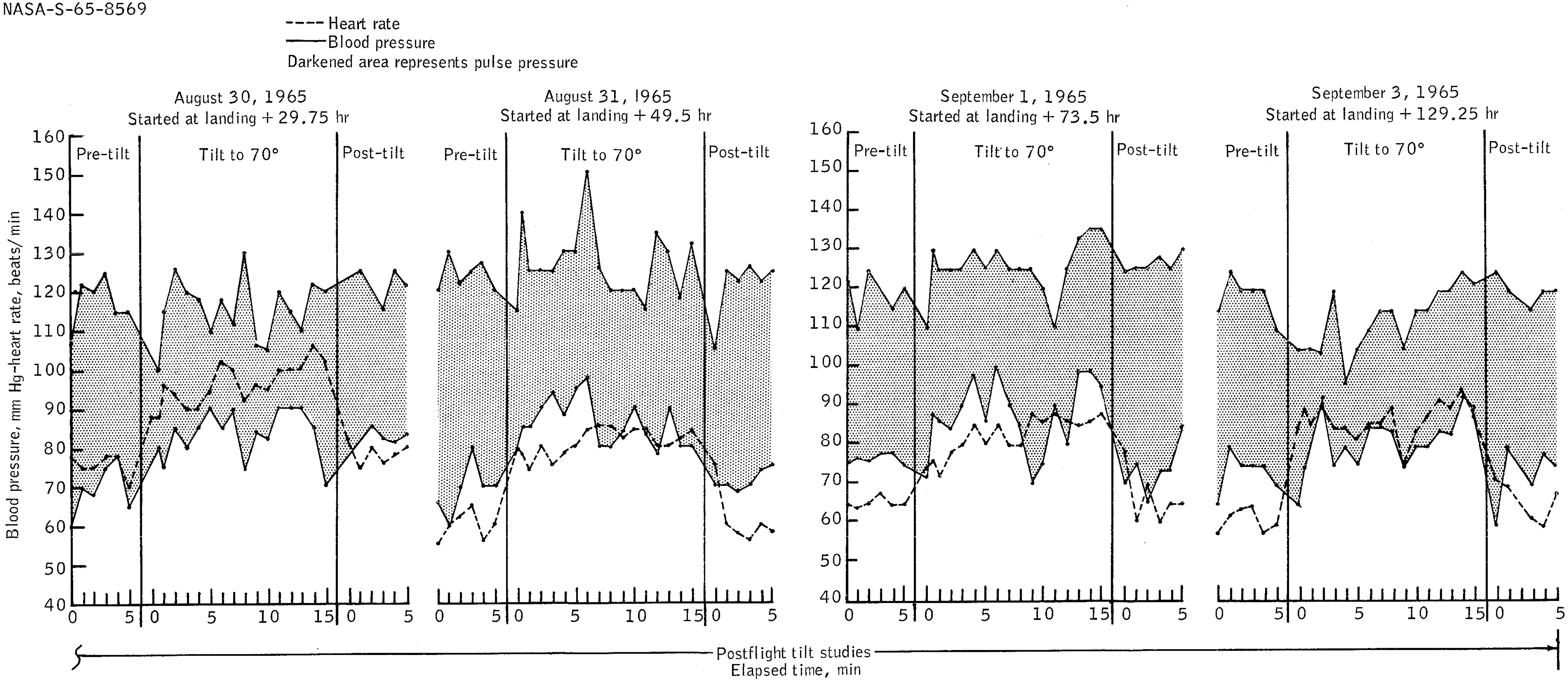
(b) Command pilot.
Figure 7.2-2. - Concluded.

NASA-S-65-8570



(a) Pilot.
Figure 7.2-3. - Tilt table studies.

NASA-S-65-8569



(b) Pilot.
Figure 7.2-3. - Concluded.

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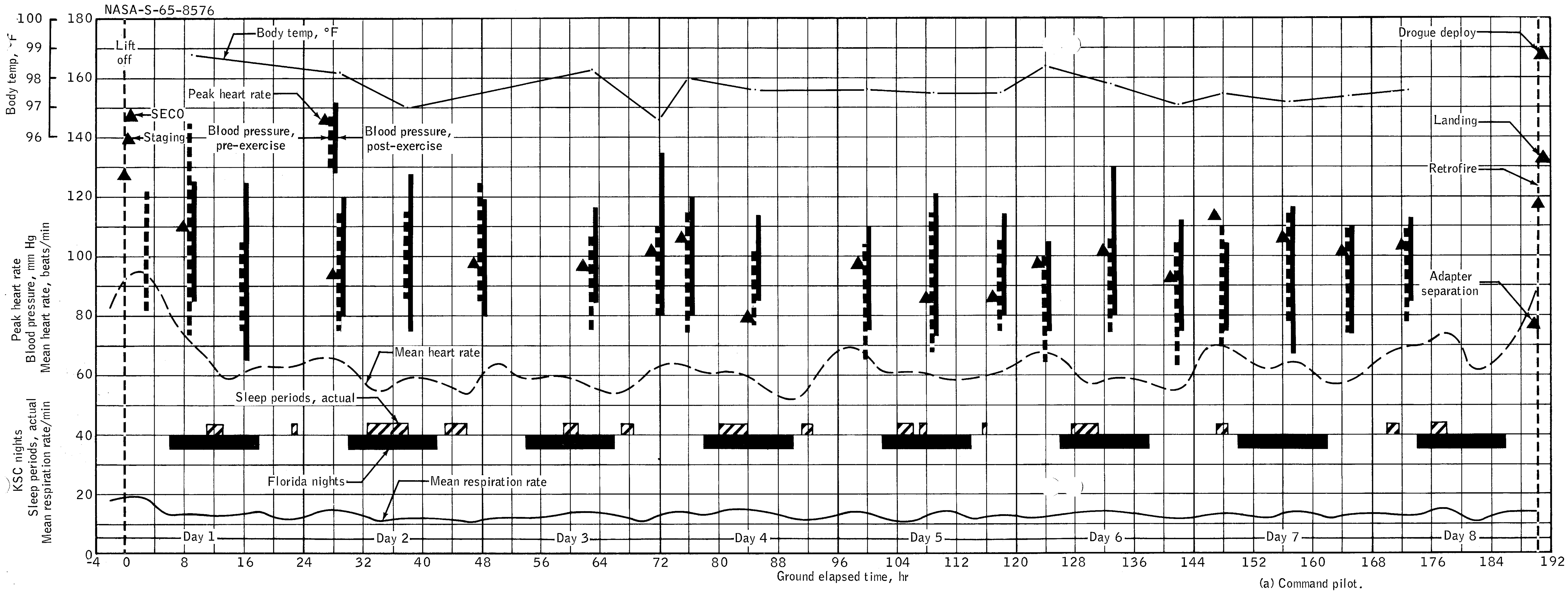
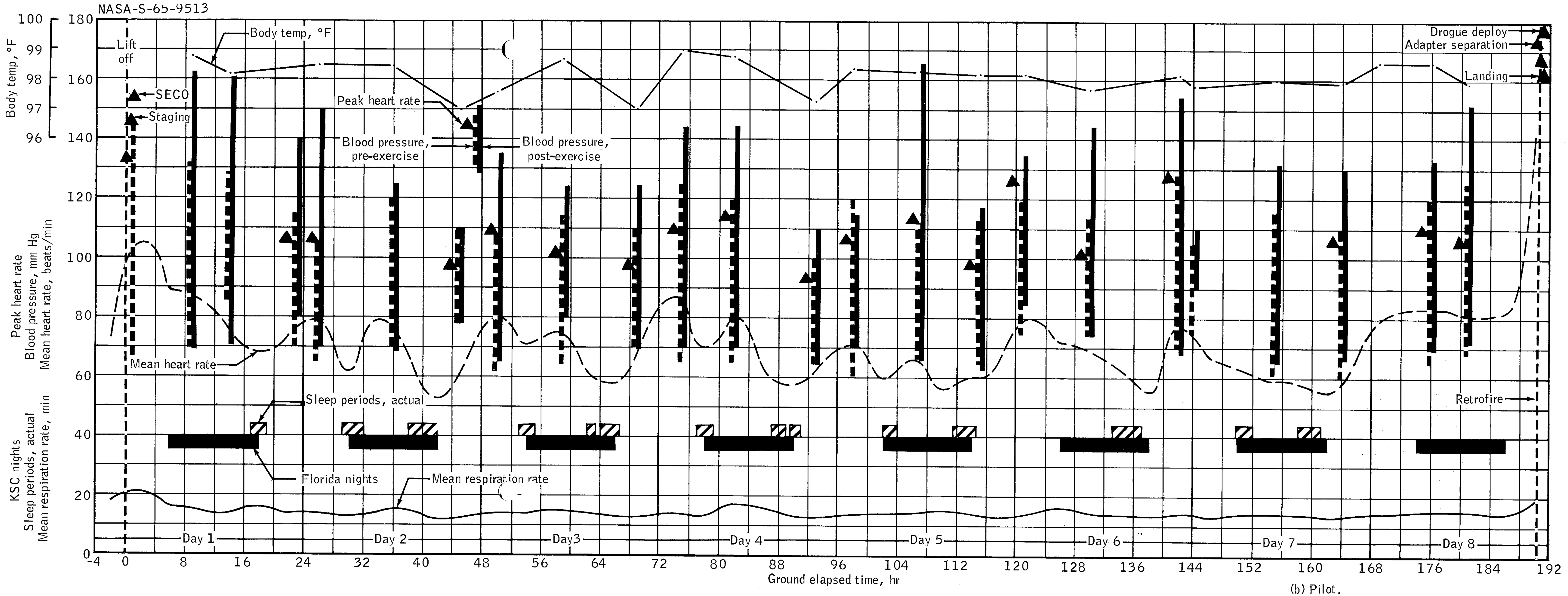


(a) Command pilot.

Figure 7.2-1. - Physiological measurements.



(b) Pilot.
Figure 7.2-1. - Concluded.

8.0 EXPERIMENTS

Sixteen of the seventeen scientific, medical, technological, and engineering experiments were conducted on the Gemini V mission to extend man's knowledge of space and to develop further the ability to sustain life in the space environment. The D-2 experiment was not conducted because rendezvous with the rendezvous evaluation pod was not accomplished. These experiments are listed in table 8-I. Some experiments have been combined for this report because of similar objectives.

Because of the nature of these experiments, only a preliminary evaluation of the experiment results can be presented in this report. In most cases, detailed evaluations and conclusions will be published in separate documents after all data for each experiment have been analyzed.

TABLE 8-I.- EXPERIMENTS

Experiment number	Experiment title	Principal experimenter	Sponsor
D-1	Basic Object Photography	Photographics Branch, Air Force Avionics Laboratory, Wright-Patterson Air Force Base, Ohio	Department of Defense
D-2	Nearby Object Photography	Photographics Branch, Air Force Avionics Laboratory, Wright-Patterson Air Force Base, Ohio	Department of Defense
D-4/D-7	Celestial Radiometry/ Space Object Radiometry	Optics and Radiometry Laboratory, Air Force Cambridge Research Laboratory, Bedford, Massachusetts	Department of Defense
D-6	Surface Photography	Photographics Branch, Air Force Avionics Laboratory, Wright-Patterson Air Force Base, Ohio; and the Naval Reconnaissance and Technical Support Center, Washington, D.C.	Department of Defense
M-1	Cardiovascular Conditioning	Space Medicine Branch, Crew Systems Division, NASA-MSC, Houston, Texas	NASA Office of Manned Space Flight
M-3	Inflight Exerciser	Space Medicine Branch, Crew Systems Division, NASA-MSC, Houston, Texas	NASA Office of Manned Space Flight
M-4	Inflight Phonocardiogram	Space Medicine Branch, Crew Systems Division, NASA-MSC, Houston, Texas	NASA Office of Manned Space Flight
M-6	Bone Demineralization	Nelda Childers Stark Laboratory for Human Nutrition Research, Texas Women's University, Denton, Texas	NASA Office of Manned Space Flight
M-9	Human Otolith Function	U.S. Naval School of Aviation Medicine, Pensacola, Florida	NASA Office of Manned Space Flight
MSC-1	Electrostatic Charge	Radiation and Fields Branch, Advanced Spacecraft Technology Division, NASA-MSC, Houston, Texas	NASA Office of Manned Space Flight
S-1	Zodiacal Light Photography	School of Physics, Institute of Technology, University of Minnesota, Minneapolis, Minnesota	Office of Space Sciences
S-5	Synoptic Terrain Photography	Theoretical Division, NASA-Goddard Space Flight Center, Greenbelt, Maryland	Office of Space Sciences

TABLE 8-I.- EXPERIMENTS - Concluded

Experiment number	Experiment title	Principal experimenter	Sponsor
S-6	Synoptic Weather Photography	National Weather Satellite Center, U.S. Weather Bureau, Suitland, Maryland	Office of Space Sciences
S-7	Cloud Top Spectrometer	National Weather Satellite Center, U.S. Weather Bureau, Suitland, Maryland	Office of Space Sciences
S-8/D-13	Visual Acuity/Astronaut Visibility	Visibility Laboratory, Scripps Institute of Oceanography, University of California, San Diego, California	Office of Space Sciences/Department of Defense

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8.1 EXPERIMENT D-1, BASIC OBJECT PHOTOGRAPHY

8.1.1 Objective

The objective of experiment D-1 was to investigate the ability of man to acquire, track, and photograph space-borne objects, such as the rendezvous evaluation pod (REP), natural celestial bodies, and other objects of opportunity.

8.1.2 Equipment

The experiment equipment consisted of a 35-mm still camera with modifications and adapters necessary for right-hand window mounting and for boresighting along the longitudinal axis of the spacecraft. Two interchangeable lenses were used, one 1270-mm and one 200-mm, f/4. Three interchangeable film backs containing Type 3400 aerial film, Type 3401 aerial film, and Type 8443 infrared aerial film were used. Time correlation was also provided.

8.1.3 Procedure

The sequence of objects to be photographed was updated during the mission and included spacecraft attitude and time values for object acquisition in addition to exposure settings.

The spacecraft attitude maneuvering system in pulse mode was used to acquire the selected object visually within the field of view of the optical sight mounted on the left-hand window, and to track the object visually through the photo-optical system mounted on the right-hand window. At the flight crew's discretion, utilizing estimated minimal tracking rates, a preplanned number of exposures were made.

8.1.4 Results

The photography of celestial bodies was completed as planned. It should be noted that the celestial photographs were made with the command pilot maneuvering the spacecraft and sighting through the optical sight for aiming. The periscope viewer was used as a sight by the pilot who gave verbal directions for alinement correction to the command pilot. The REP photography was not performed.

8.1.5 Conclusion

Proper orientation of the object in the film format indicates that acquiring, tracking, and photographing celestial bodies present no problems.

8.2 EXPERIMENT D-2, NEARBY OBJECT PHOTOGRAPHY

8.2.1 Objective

The objective of experiment D-2 was to investigate the ability of the flight crew to obtain high resolution photographs of the REP with various backgrounds.

8.2.2 Equipment

The experiment equipment consisted of a 35-mm still camera with modifications and adapters necessary for mounting on the right hatch window, easy operation, and boresighting along the longitudinal axis of the spacecraft. Two interchangeable lenses were used – one 1270-mm and one 200-mm f/4. The interchangeable film back containing type 3400 aerial film was to be used. Time correlation was also provided.

8.2.3 Procedure

Subsequent to the planned REP exercise, the experiment D-2 photography of the REP was to be accomplished in conjunction with a spacecraft in-plane maneuver around the REP at a distance of 60 (±10) feet. Exposures were to be made at 30° increments using the 200-mm f/4 lens. The spacecraft was to be maneuvered aft to a distance of approximately 500 feet, at which time an exposure was to be made using the 1270-mm lens.

8.2.4 Results

Due to perturbations in the spacecraft electrical power system subsequent to the ejection of the REP, experiment D-2 was not performed.

8.2.5 Conclusion

None.

8.3 EXPERIMENT D-4/D-7, CELESTIAL RADIOMETRY AND SPACE OBJECT RADIOMETRY

8.3.1 Objective

The objective of experiment D-4/D-7 was to make irradiance measurements in the band from 0.2 to 12 microns on various celestial and terrestrial backgrounds, on rocket plumes, and on cold objects in space.

8.3.2 Equipment

The experiment primary equipment consisted of the following eight units:

Multichannel radiometer

IR interferometer/spectrometer

Cryogenic interferometer/spectrometer

FM/FM transmitter

Electronic unit

Recorder electronics unit

Tape transport

UHF antenna

8.3.3 Procedure

The experiment was to be initiated with the antenna extension and erection of the sensing instruments during the first revolution. A "go" decision was to be made at Carnarvon on the first revolution for the cooled measurement based upon a real-time check of its operating parameters. During the night of the second revolution, far infrared (IR) region measurements were to be made on the rendezvous evaluation pod (REP). Subsequently, measurements were to be made on rocket engine plumes, terrestrial background during day and night, and celestial background including selected stars and the moon.

8.3.4 Results

A total of approximately 3 hours of data was taken in 30 separate measurements. The experiment recorder tape was completely used. Of the 28 planned targets, 23 were accomplished, and 1 of 4 targets of opportunity was accomplished.

All equipment with the exception of the radiometer operated satisfactorily. The filter wheel on the radiometer ceased to advance after approximately 15 minutes of operation and remained on filter position 8. This reduced the radiometer data to information in three spectral bands. However, it was an appropriate position for performing rocket plume measurements. Visual observation of the rocket plumes was possible in all cases. Pulse control mode proved adequate for tracking missiles. The first launch was acquired as the missile emerged from a low cloud deck and was tracked through staging. The second missile was seen at ignition by the pilot. The command pilot did not observe lift-off and, as a result, tracking was not successful. The flight crew made measurements on two of the three selected stars. Measurements of the final rocket sled run, the Milky Way, and the sun were not performed.

8.3.5 Conclusions

Man's capability to observe and track space objects, such as the REP, and other objects, such as rocket engines, is of significance. Data available at this time are quantititive only. The detailed evaluation of the individual measurements will be made when all the spectral response data have been reduced, correlated, and analyzed.

8.4 EXPERIMENT D-6, SURFACE PHOTOGRAPHY

8.4.1 Objective

The objective of experiment D-6 was to study the problems associated with the ability of the flight crew to acquire, track, and photograph terrestrial objects.

8.4.2 Equipment

The experiment equipment consisted of the basic 35-mm still camera with modifications and adapters necessary for mounting on the right hatch window, and for boresighting along the longitudinal axis of the spacecraft. Two interchangeable lenses were used - one 1270-mm and one 200-mm f/4. Three interchangeable film backs containing type 3400 aerial film, type 3401 aerial film, and type 8443 infrared aerial film were used. Time correlation was also provided. Two optical filters in each lens were used to reduce atmospheric resolution degradation.

8.4.3 Procedure

The sequence of objects to be photographed was updated during the mission and included spacecraft attitude and time values for object acquisition in addition to equipment modes. Included were the lens-film combination, acquisition mode, tracking mode, and exposure settings. The acquisition modes were: (1) visual, using the optical sight mounted on the left hatch window and (2) instrument, using the spacecraft attitude instruments. The modes of tracking were: (1) visual, command pilot sighting through the optical sight and pilot sighting through telescope sight or periscope viewer while giving verbal directions to the command pilot, (2) telescope, using the telescope sight mounted on the right hatch window, and (3) periscope, using the reflex viewing capabilities of the photo-optical system mounted on the right hatch window. The command pilot was to maneuver the spacecraft for both of the acquisition modes and for the visual tracking mode. The pilot was to maneuver the spacecraft for the telescope and periscope tracking modes. The spacecraft attitude maneuvering system in pulse mode was to have been used to acquire and track the selected object within the field of view of the sighting device. At the flight crew's discretion, utilizing estimated minimal tracking rates, a series of four photographs were to be taken of the selected terrestrial object beginning with acquisition and ending at the spacecraft nadir.

8.4.4 Results

The experiment equipment performed successfully. The pilot experienced some film transport difficulties during the mission which were alleviated by loosening the knurled knob on the photo event indicator. A postflight analysis revealed that an unmated pair of photo event indicator and film transport adapter had been provided.

The restricted field of view resulting from the experiment equipment being mounted on the right hatch window required that the equipment be assembled and disassembled a number of times during the flight. Assembly and disassembly of the photo/optical system had no effects on its boresight with the optical sight throughout the mission.

The time required for assembly of the equipment placed some constraint on the number of possible experiment D-6 series. A quick response system could have allowed more programed experiment D-6 series and provided a greater probability of success.

Weather was a significant factor affecting the conduct of the experiment. The sequence of experiments to be conducted over the United States was planned approximately 4 to 6 hours in advance. Although the meteorology facilities were adequate for normal weather prognosis, they were inadequate for the conduct of this experiment. Cloud cover with respect to the line of sight between the spacecraft and the terrestrial object must be known for optimum results. A few experiment D-6 series were cancelled while others were attempted but not completed because of cloud cover over the terrestrial object of concern.

Results were limited because only one of five planned combinations of acquisition and tracking modes was accomplished. The one combination was that of a visual acquisition and visual tracking.

8.4.5 Conclusion

Results obtained indicate that visual acquisition with visual tracking can be successfully applied to obtain photographs of a preselected terrestrial object.

8.5 EXPERIMENT M-1, CARDIOVASCULAR CONDITIONING

8.5.1 Objective

The objective of experiment M-1 was to determine the effectiveness of pneumatic venous pressure cuffs worn about the upper thigh during flight in preventing the orthostatic hypotension and tachycardia observed during postflight tilt-table tests after previous Gemini missions. The experiment was based on two premises: First, cyclic inflation of the pneumatic cuffs was expected to prevent the loss of blood plasma by decreasing thoracic blood volume. A reduction of the thoracic blood volume would prevent hypogravic diuresis thus maintaining plasma volume. Second, cyclic inflation of pneumatic cuffs would produce an artificial hydrostatic gradient across the walls of the leg veins. This action would stretch the venous sensory receptors, thus providing the stimuli needed to maintain active venomotor reflexes. This action would theoretically prevent pooling of blood in the lower extremities and increase the effective circulating blood volume while standing in a 1g environment.

8.5.2 Equipment

The cardiovascular conditioning experiment equipment consisted of a pneumatic control system and a pair of pneumatic venous pressure cuffs. The cardiovascular conditioner was an automatic mechanical system (fig. 8.5-1) which alternately inflated and deflated the pneumatic venous pressure cuffs around the proximal attachment of the pilot's thighs. The cardiovascular conditioner was comprised of three basic components:

(a) Pressurized storage vessel charged with oxygen to 3500 psig.

(b) Pneumatic control system to monitor the pressurized storage vessel.

(c) Pneumatic oscillator system to provide the timing function and switching logic for the periodic inflation and deflation of the pneumatic cuffs.

The pneumatic venous pressure cuffs were form-fitted to the proximal attachment of the pilot's thighs. They consisted of a 3- by 6-inch bladder, enclosed in a soft non-stretchable fabric, that was located on the dorsal medial surface of the thigh. The lateral surface of the cuff had a lace allowing the cuff to be adjusted for proper fitting.

8.5.3 Procedure

The pilot wore the pressure cuffs around the proximal attachment of each thigh during the flight. Upon activation of the manual shutoff valve, the cuffs were automatically pressurized to 80 mm Hg for 2 minutes during each 6-minute interval. The system was capable of continuous operation during the flight; however, it could be turned off for sleep periods, if desired. The experiment imposed no operational requirements other than activation and deactivation by the pilot. The experiment utilized the bioinstrumentation system to measure inflight physiological parameters. The data obtained were transmitted to ground stations via the spacecraft PCM telemetry system and also were stored on the onboard biomedical recorder.

8.5.4 Results

Both crew members had normal cardiovascular responses to three passive 70° tilt-table tests prior to flight. Mean values for these responses are shown in the preflight portion of figure 8.5-2 and are summarized in the following table of mean values.

	Pretilt		Tilt		Percent blood volume change cc/100 cc tissue/min
	Heart rate, beats/min	Blood pressure, mm Hg	Heart rate, beats/min	Blood pressure, mm Hg	
Command pilot	58	109/72	75	111/79	3.01
Pilot	59	117/68	78	120/79	2.70

As indicated by these data, the heart rate increased, blood pressure changed very little, and leg volume increased. These values returned to the pretilt levels after tilting back to the horizontal position.

During the mission, the equipment was connected only to the pilot, and was programed for operation for the full 8 days; however, the pneumatic programer stopped cycling after 4 days when the pressure vessel dropped below the operational level, indicating the lack of sufficient oxygen capacity for the 8-day mission (24 hrs/day) as programed.

Postflight tilt-table values appear in figure 8.5-2. Both crew members had increased resting heart rates during the first 2 days after recovery. Increase in maximum heart rate over preflight resting values were observed on the first day postflight tilts (increase by command pilot, 27 beats/min; increase by pilot, 50 beats/min). Postflight resting blood pressure was below the preflight values for the command pilot. The systolic pressure remained 10 mm Hg below the preflight values for 3 days postflight. Diastolic values were 8 mm Hg below the values for 4 days postflight. The pilot exhibited an increased diastolic pressure (3 to 9 mm Hg) for 4 days postflight (fig. 8.5-2), whereas the systolic values were identical with the preflight readings.

During the postflight tilt-table tests both crew members had an increased heart rate and a narrowed pulse pressure. The command pilot exhibited a 78 beat/min increase in heart rate above the postflight resting level during the first tilt (4 hours postflight), and a 54 beat/min increase on the second tilt (8 hours postflight). The pilot exhibited a 46 beat/min increase on the first tilt and a 33 beat/min increase on the second tilt. Both crew members showed a marked decrease in resting and tilt heart rates 24 hours after recovery (resting rates 22 beats/min decrease, tilt rates 34 beats/min decrease for each crew member). During the third, fourth, and fifth day after recovery, both crew members had progressively decreasing resting and tilt heart rates; however, both still exhibited mean values above the preflight values.

Both crew members had a narrower postflight blood pressure range. The command pilot's blood pressure was low for 3 days postflight and the pilot for only 4 to 8 hours postflight.

During the first 70° passive tilts, the blood pressure of each crew member narrowed below the preflight tilt and postflight resting values (command pilot 95/85 mm Hg; pilot 90/76 mm Hg). During the second, third, and fourth postflight tilts, the command pilot still maintained a lower systolic pressure during tilt, whereas the pilot had returned to preflight levels.

The command pilot exhibited an 89 percent increase above the preflight value in leg blood volume and the pilot an 87 percent increase above the preflight levels during the first postflight tilt. During the second tilt, however, the command pilot had increased to 149 percent over the preflight value and the pilot to only 73 percent. If the two first-day tilts are combined, it can be shown that the command pilot pooled 39 percent more blood in the legs than did the pilot.

8.5.5 Conclusions

The pilot exhibited a greater increase in postflight resting heart rate than did the command pilot. The pilot's heart rate and blood pressure returned to preflight levels for all phases of the tilt-table test within 2 days after recovery, whereas the command pilot had not yet returned to preflight values 4 days postflight. The command pilot lost 8 percent of his plasma volume during flight and the pilot lost only 4 percent (table 8.5-I). Both the greater pooling and the loss of blood plasma contributed to a decreased effective circulating blood volume. This accounts in part for the more pronounced postflight response of the command pilot to the tilt-table test. The command pilot pooled 39 percent more blood in the legs than did the pilot during the first day postflight (table 8.5-II). The longer period required to return to preflight baseline levels for the command pilot can be accounted for in part by the increased venous pooling in the legs and a greater loss in plasma volume.

In comparing the Gemini V cardiovascular data with the Gemini IV data, it is evident that the Gemini IV pilot and the Gemini V command pilot who sustained nearly equal weight losses also had similar cardiovasuclar responses to a passive tilt-table test. They also had comparable blood pooling in the lower extremities during the first 2 days postflight (table 8.5-II). The plasma volume changes are also comparable (table 8.5-I). The Gemini IV pilot had a 13-percent decrease in blood plasma volume and the Gemini V command pilot an 8-percent decrease. Both the Gemini IV command pilot and Gemini V pilot had a 4-percent loss in plasma volume during flight.

It is concluded that the objective of the Gemini V M-1 experiment may have been accomplished, at least in part. The pilot of the Gemini V crew exhibited a different postflight tolerance to passive tilt when compared with the command pilot, but whether this was a result of simply individual variation or a reflection of the relative effectiveness of the cuff device is not known at this time. Additional flight experimental data will have to be obtained before any definitive conclusions can be made.

TABLE 8.5-I.- POSTFLIGHT BLOOD VOLUME STUDY

[Comparison of results between Gemini V and Gemini IV]

	Gemini V		Gemini IV	
	Command Pilot	Pilot	Command Pilot	Pilot
△ total blood volume, percent	-13	-13	-	-
△ Plasma volume, percent	-8	-4	-4	-13
△ Red blood cell mass, percent	-20	-20	-12	-13

Note: Minus indicates percent below preflight.

TABLE 8.5-II.- POSTFLIGHT LEG PLETHYSMOGRAPHIC STUDY

[Comparison of results between Gemini V and Gemini IV]

Days after recovery	Postflight change in volume per minute, percent (a)			
	Gemini V		Gemini IV	
	Command Pilot	Pilot	Command Pilot	Pilot
1	+119	+80	+22	+131
2	+44	+25	+27	+61
3	+73	+57	-38	+126
4	+78	+117		
5	+111	+97		

Note: + indicates percent above preflight, - indicates percent below preflight.

(a) Percent change in volume equals cc/100 cc tissue/min

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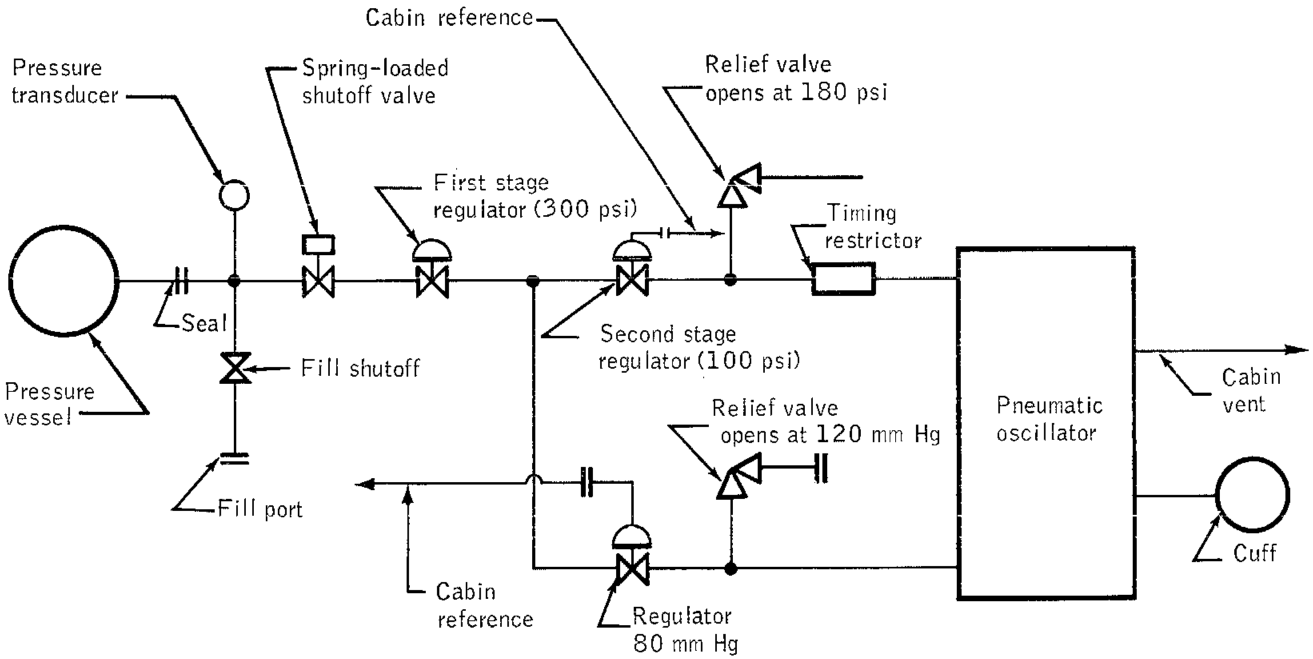

Figure 8.5-1. - Experiment M-1, schematic diagram of cardiovascular reflex conditioner.

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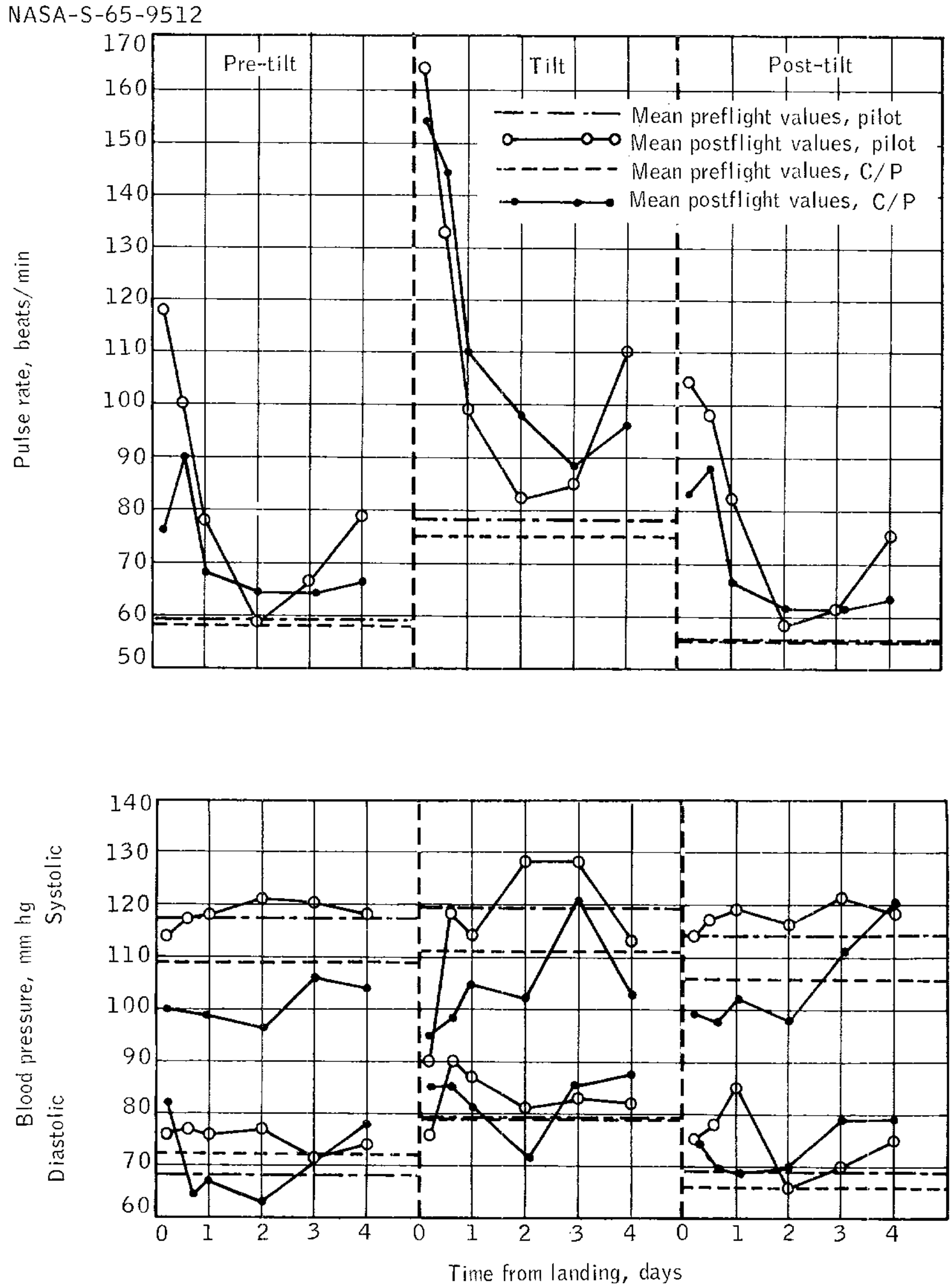

Figure 8.5-2. - Experiment M-1, Gemini V tilt table summary.

8.6 EXPERIMENT M-3, INFLIGHT EXERCISER

8.6.1 Objective

The objective of this experiment was to assess cardiovascular reflex activity in response to a given physical workload (exercise) and to ascertain the general capacity to perform physical work under prolonged space-flight conditions. It should be pointed out that this is not programed as exercise as such, but is a mild cardiovascular stimulus used to monitor reflex activity.

8.6.2 Equipment

The inflight exerciser consisted of a pair of rubber elastic cords attached to a handle at one end and to a nylon foot strap on the other end. The length of extension was limited by a stop-cable, which fixed the workload. The exerciser required 70 pounds to pull to a full extension of 12 inches. The exerciser can be utilized to exercise either the upper or lower extremities by holding the handle fixed and pushing with the feet or by holding the feet fixed and pulling with the handle. The bioinstrumentation system provided the necessary data for support of this experiment.

8.6.3 Procedure

Exercise periods (medical data passes) were scheduled approximately three times a day for each crew member. For these medical data passes, the exercise consisted of pulling the handle one pull per second for 30 pulls. Blood-pressure measurments were made before and after the exercise period. Also, the command pilot was encouraged to exercise his legs between scheduled periods.

8.6.4 Results

The flight crew performed the exercise periods as scheduled. Heart rates were determined by counting 15-second periods for 2 minutes before and after exercise and the first and last 15-second periods during exercise. Comparison of 1-g preflight exercise periods with those obtained during flight indicate little difference in heart rate response. Comparison of the inflight exercise periods from the first to the last day also indicated little difference in heart rate response. Inflight heart rate responses are illustrated in figure 8.6-1. Blood pressures before and after exercises are reported in section 7.2.

After the fourth day of flight, both crew members used the exerciser frequently between scheduled medical data passes. Both felt that exercise was essential and beneficial on long duration flights.

8.6.5 Conclusions

The M-3 experiment on Gemini V can be classified as a success. The crew demonstrated their ability to perform physical work through 8 days of flight. The biomedical data obtained in response to this given workload offered no evidence of cardiovascular reflex decrement during 8 days of flight. It is felt that the medical data passes should be programed throughout the 14-day mission for real-time cardiovascular reflex evaluation, and that additional exercise periods, for exercising both upper and lower extremities, should be scheduled into the flight plan.

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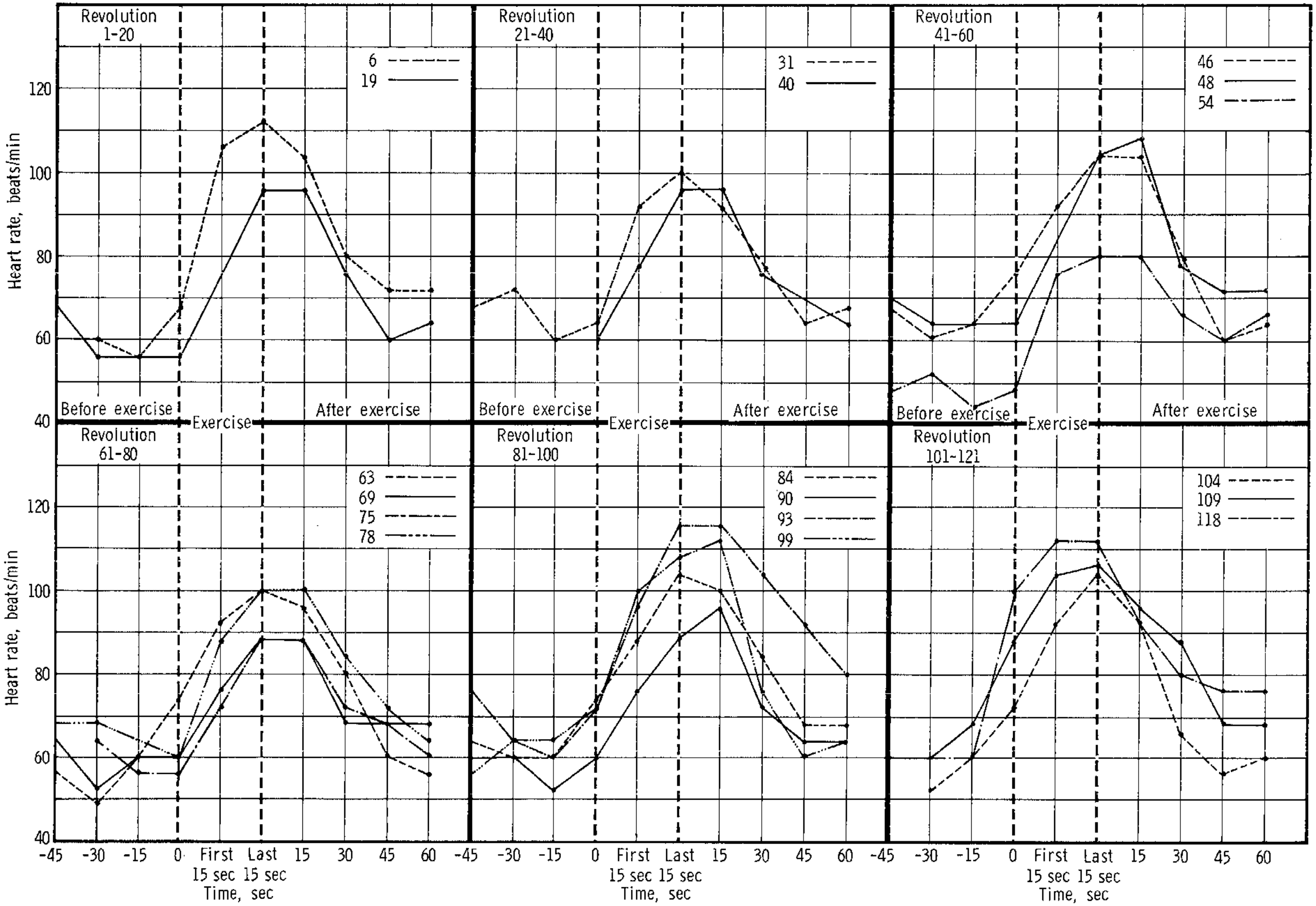

(a) Command pilot

Figure 8.6-1. - Experiment M-3, response to inflight exercise. (Minute heart rate plots before, during and after exercise per 15 second intervals).

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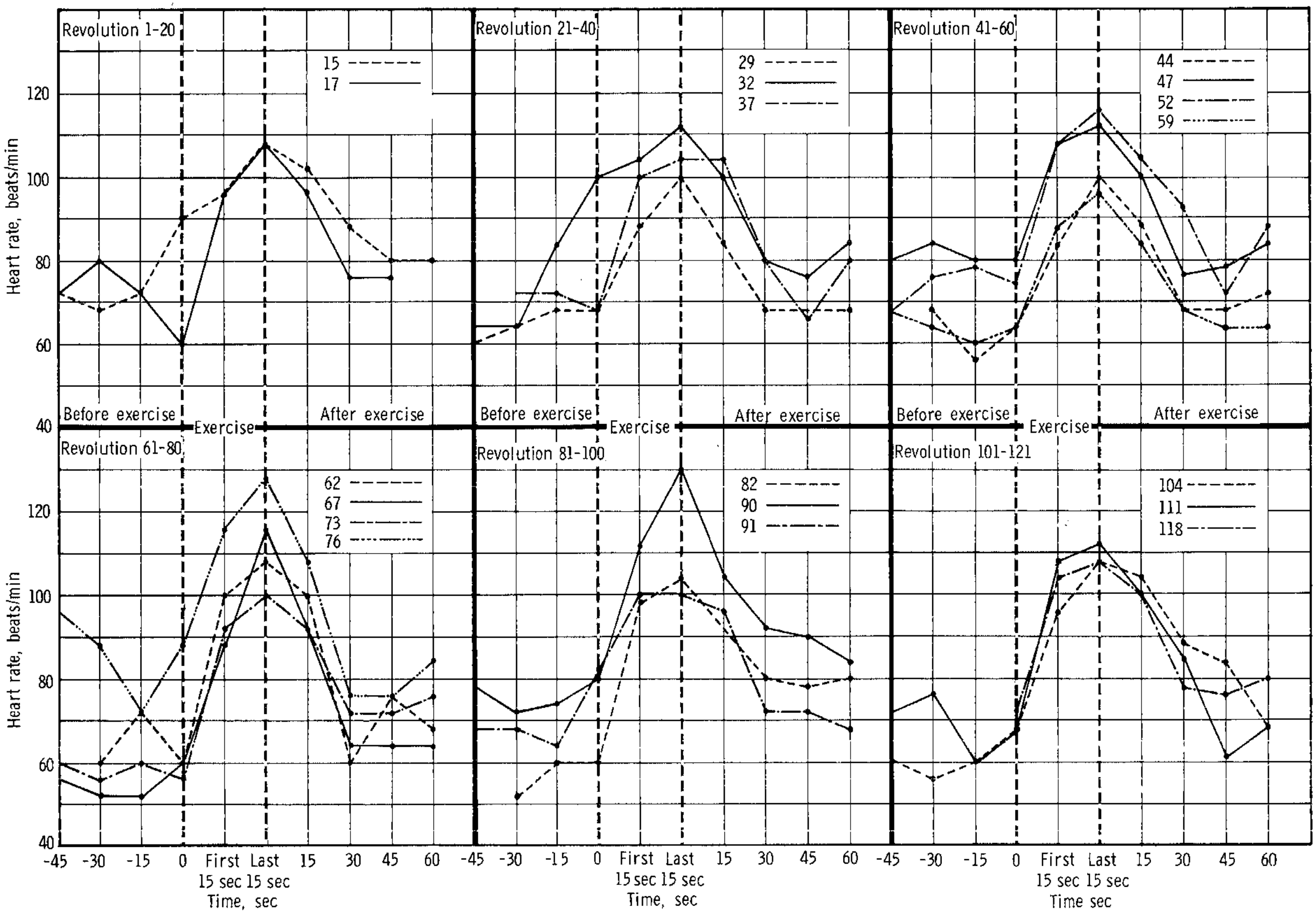

(b) Pilot

Figure 8.6-1. - Concluded.

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8.7 EXPERIMENT M-4, INFLIGHT PHONOCARDIOGRAM

8.7.1 Objective

The objective of experiment M-4 was to measure the time interval between the electrical activation of the heart muscle (myocardium) and the onset of the muscular contraction of a man in space. This time interval is a measure of the functional status of fatigue-state of the muscle. This information will provide some insight into the functional cardiac status of flight crew members during prolonged space flight.

8.7.2 Experiment

The experiment equipment consisted of one transducer and an associated signal conditioner for each flight crew member. The signal conditioner was the same unit as that used for the operational electrocardiogram measurements. The transducer was applied to the chest wall on the sternum of each flight crew member. All heart sounds detected were transmitted through the harness wiring bundle to the biomedical recorder.

8.7.3 Procedure

The phonocardiogram signals were recorded on the appropriate biomedical recorder when it was operating. These data provided information on the duration of mechanical systole and diastole, the duration of the time period between electrical and mechanical systole, and the duration of the complete heart cycle. In addition, the phase of isometric contraction was measured which included the electrical excitation period. From these measurements, an assessment of myocardial function will be made, particularly the effectiveness of cardiac contractility under conditions of space flight.

8.7.4 Results

The biomedical tapes have been reduced to real time at $1\frac{7}{8}$ ips and have been transmitted to the Texas Institute for Rehabilitation and Research for analysis and evaluation.

8.7.5 Conclusions

Postflight determinations indicate that both phonocardiogram transducers were still in satisfactory operating condition. Recognizable phonocardiogram data were received at the Cape Kennedy ground station during the mission. However, since these data have not yet been analyzed, no conclusions can be made in this report.

8.8 EXPERIMENT M-6, BONE DEMINERALIZATION

8.8.1 Objective

The objective of experiment M-6 was to investigate the occurrence and degree of any bone demineralization resulting from prolonged space flights. Bone demineralization has been observed in humans during periods of inadequate calcium intake, during periods of immobilization such as bed rest, and during other situations involving physical inactivity.

8.8.2 Equipment

The equipment used in this experiment was a standard clinical X-ray machine, standard 8 by 10-inch X-ray films, and calibrated densitometric wedges.

8.8.3 Procedure

X-rays were made on the flight crew at Cape Kennedy, Florida, in accordance with the following schedule: (a) launch minus 10 days, (b) launch minus 48 hours, and (c) launch minus 220 minutes. The launch minus 220 minutes X-ray was repeated because of the 48-hour flight delay. Precise X-ray densitometric measurements were made of the heel bone (os calcis) of the left foot and the terminal bone of the little finger (fifth digit) of the left hand.

Three similar measurements were made after completion of the mission according to the following schedule: (a) as soon as possible after recovery, (b) approximately 24 to 48 hours after completion of the mission but prior to the flight crew's departure from the primary recovery vessel, and (c) at the NASA Manned Spacecraft Center, Houston, Texas, approximately 10 days after completion of the mission.

The data obtained will be compared to determine any bone demineralization that occurred during the mission.

8.8.4 Results

All X-rays have been developed and initially analyzed. All of the required scannings of each X-ray have not been completed. Actual analysis of the complete results is dependent on the completion of the analytical phase of the experimental protocol.

Comparison between the currently available preflight and postflight X-rays, to the extent analyzed, indicates actual absorbency changes to have occurred on the foot (os calcis) X-rays of both flight crew members. The hand X-rays have not been completely analyzed. The decrease in absorbency from the last postflight X-ray to the first preflight X-ray of the os calcis is 3 percent for the pilot and 9 percent for the command pilot. This change is in the order of magnitude expected from available bed-rest information in which neuroendocrine stress is also a factor and is very similar to those observed in the Gemini IV crew.

The last postflight X-rays (September 8, 1965) indicate that the observed decrease in os calcis absorbency had not completely returned to preflight levels; however, a gradual return is evident.

The reproducibility of the densitometric analysis of the os calcis X-rays developed at a standard location is excellent and is within 2 to 3 percent when films were developed at different locations. Reproducibility of the data on the X-rays on the hand are not expected to be as good.

8.8.5 Conclusions

Available data and current analysis do not permit any conclusions to be reached at this time. All scheduled experiment X-rays have been completed. The preliminary results indicate the importance of continuing the postflight observations. It is planned to take another X-ray of the Gemini V flight crew at a later date to determine if their absorbency rate has returned to normal. The results also indicate the importance of continuing these types of observations, especially for longer missions.

8.9 EXPERIMENT M-9, HUMAN OTOLITH FUNCTION

8.9.1 Objective

The objective of experiment M-9 was to measure any change in otolith activity, and particularly as might result from exposure to prolonged weightlessness. Prior to the flight, the basic otolith function of each flight crew member was determined by measuring the ocular counterrolling (CR) response to body tilt. These measurements were to serve also as a basis of comparison with postflight CR data in establishing whether changes in otolithic sensitivity had occurred during the 8 days of weightlessness. Evaluation of the effect of any change in otolithic input upon the crew members' behavior was to be accomplished by measurement of the ability to orient visually to environment under standard and zero gravitational conditions.

8.9.2 Equipment

The apparatus for measuring egocentric visual localization (EVL) was incorporated into the onboard vision tester which was part of experiment S-8/D-13. This was only a physical interface; in all other respects experiment M-9 was completely separate.

The inflight vision tester was a binocular instrument with an adjustable interpupillary distance (IPD) but without any focusing adjustment. The instrument was held at the proper position, with the lines of sight coincident with the optic axes of the instrument, by means of a bite-board individually fitted to the astronaut. This assured that at each use the instrument was identically located with respect to the visual axis, providing the subject made the proper IPD adjustment. In this position eyecups connected to the eyepieces of the instrument excluded all extraneous light from the visual field. Power was supplied by the spacecraft utility cord.

The luminous line target was produced in the vision tester by the insertion of an astigmatizer to refract the collimated light from a small central field, with the adaptive field light of experiment S-8/D-13 turned off. The line of light so produced was rotated about its center through a helical gear system by turning a knurled ring located between two numbered cylinders. An index mark on the side of the eyepiece indicated the position of the line in the eyepiece.

8.9.3 Procedure

A long history of experimentation has established EVL as a delicate, reliable, and specific indicator of otolithic input. Furthermore, studies carried out during transient periods of weightlessness demonstrated that EVL is stable and quite accurate under temporary physiological deafferentation of the otolith organs. EVL therefore provided a reliable baseline indicator and one poised to reflect the influence of any unusual otolith activity that might be generated during the Gemini V flight.

The initial testing of inner ear function and EVL was limited to a 90-minute period for each pair of subjects (primary and backup crew) and was accomplished at Cape Kennedy 16 days prior to the mission.

Immediately prior to the preflight and postflight tests of EVL, one drop of 1 percent pilocarpine hydrochloride ophthalmic solution was instilled in the subject's eye opposite to the one used for making visual orientation judgments. The subject was then placed in the CR tilt device, properly adjusted, and secured. The apparatus for measuring ocular CR was essentially a tilt device on which a camera system was mounted. The method of conducting the preflight and postflight EVL test was as follows: the IPD of the vision tester was adjusted and the device was brought into its proper position by inserting the bite-board into the mouth of the subject. By means of the knurled wheel the subject rotated the target clockwise or counterclockwise until it appeared to be alined parallel to the gravitational horizontal. This procedure was repeated until several settings (eight, preflight; and five, postflight) had been made in the upright as well as in several tilt positions (±10°, ±20°, ±30°, ±40°). The angles of body tilt were presented in a random order.

The method of testing inflight was as follows: Immediately after completion of experiment S-8/D-13, the instrument was readied for EVL testing by turning off the adaptive field, occluding the left eyepiece (command pilot) or right eyepiece (pilot) by means of the ring on the eyepiece, and rotating the astigmatizer into its proper position before the opposite eye. The white-line target appearing against a completely dark background was initially offset at random by the observer pilot. The subject pilot's task was to aline the target parallel to the apparent position of the FDAI, pitch axis zero indicator. The subject, when satisfied with his setting, closed his eyes and removed his hand from the knurled ring. This served as a signal to the observer pilot to record the setting and offset the target. This procedure was to be repeated five times. The vision tester was then handed to the other pilot and the same sequence was carried out after completion of the visual acuity test. The pilots were requested to maintain an erect position by alinement with the head rest.

The preflight and postflight measurements of ocular CR were accomplished according to the standard procedure used at the U.S. Naval School of Aviation Medicine. Following the EVL test, the subject remained in the upright position in the tilt device, the vision tester and its bite-board were removed, and preparations were made for recording eye position photographically. The CR bite-board was inserted into the subject's mouth and the position of his appropriate eye was adjusted so that it centered upon the optic axis of the camera system when he fixated the flashing red ring of light. Six photographic recordings were made at this position; then the subject was slowly tilted in his lateral plane to each of four other positions (±25°, ±50°) and the same photographic procedure was repeated.

During the postflight EVL and CR tests, the accelerometer system was used continuously to record motions of the recovery ship around its roll, pitch, and yaw axes.

During the EVL and CR tests, readings of blood pressure, pulse rate, and EKG were carried out by the MSC medical evaluation team. Postflight examinations for pilot A were begun approximately 5 hours after recovery and for pilot B, approximately 6 hours after recovery.

8.9.4 Results

The data received from all phases of this experiment are being analyzed at the time of publication of this report.

8.9.5 Conclusions

A cursory analysis of the available data indicate that experiment M-9 successfully met its stated objectives. Further reduction and analysis of data are necessary before any account of the possible influence of otolithic activity during and following the prolonged period of weightlessness can be presented.

8.10 EXPERIMENT MSC-1, ELECTROSTATIC CHARGE

8.10.1 Objective

The objective of experiment MSC-1 was to obtain measurements of the plasma field potential around the spacecraft and to screen out any effect of the electric field terminating on the spacecraft.

The data obtained were to be compared with those obtained during the Gemini IV mission which included electric field in the measurement.

8.10.2 Equipment

An electrostatic potential meter (EPM), installed in the spacecraft retrograde adapter section, was used for this experiment. The EPM was the same as that flown on the Gemini IV mission, with an external modification which consisted of a screen assembly mounted in front of the sensor unit. The purpose of the screen assembly was to inhibit the electric field incident on the face of the sensor unit.

8.10.3 Procedure

The EPM was to be turned on at insertion and operated continuously during the mission.

For seven preselected periods during the mission the spacecraft was to be configured as follows:

(a) Acquisition beacon - Off

(b) All other beacons and transmitters - Off

(c) No thrusting

These periods were programed into the mission for comparison with periods of either transmission or thrusting, or both.

8.10.4 Results

The experiment procedures were accomplished as planned. Ground tests have shown the Gemini IV configuration of the EPM to be responsive to radiated spacecraft radio frequency (RF) energy and to charged particles incident on the sensing face of the sensor unit. Specific data from the mission that would define the contribution of these phenomena to the EPM output were not available in time to be analyzed for this report. However, quick-look EPM data were obtained for several periods during the mission.

This indicated that during the initial 4 hours of the mission, the EPM operated in the same manner as it did during preflight tests and was similar to the Gemini IV EPM orbital operation. Figure 8.10-1 depicts the Gemini IV and Gemini V EPM outputs versus time from 00:10:00 to 04:20:00 g.e.t.

The Gemini V EPM was screened from measuring electric field; however, the only existing calibration on the EPM is in terms of electric field. For clarity, the nomenclature "output unit" is being used with 1 output unit equal to the output corresponding to an input of 1 volt per centimeter in preflight calibrations.

At approximately 04:04:00 g.e.t., the Gemini V EPM output began to oscillate. An oscillating output was evident on the greater part of all subsequent data.

8.10.5 Conclusions

Data presently available do not give sufficient information to perform detailed comparisons between the Gemini IV and Gemini V data. However, quick-look data for the initial 4 hours of each mission show that the Gemini V output was, in general, slightly higher than the Gemini IV output. This could be interpreted as a gross indication that the contribution of an electrostatic field charge on the Gemini IV EPM was negligible.

Evaluation of the Gemini V data will continue. Correlations between EPM output and RF, and plasma fluxes will be made. Comparisons will be made between Gemini IV and Gemini V EPM outputs under similar projected environmental and spacecraft operational conditions.

Presently available data do not enable determination of the cause of the oscillating EPM output.

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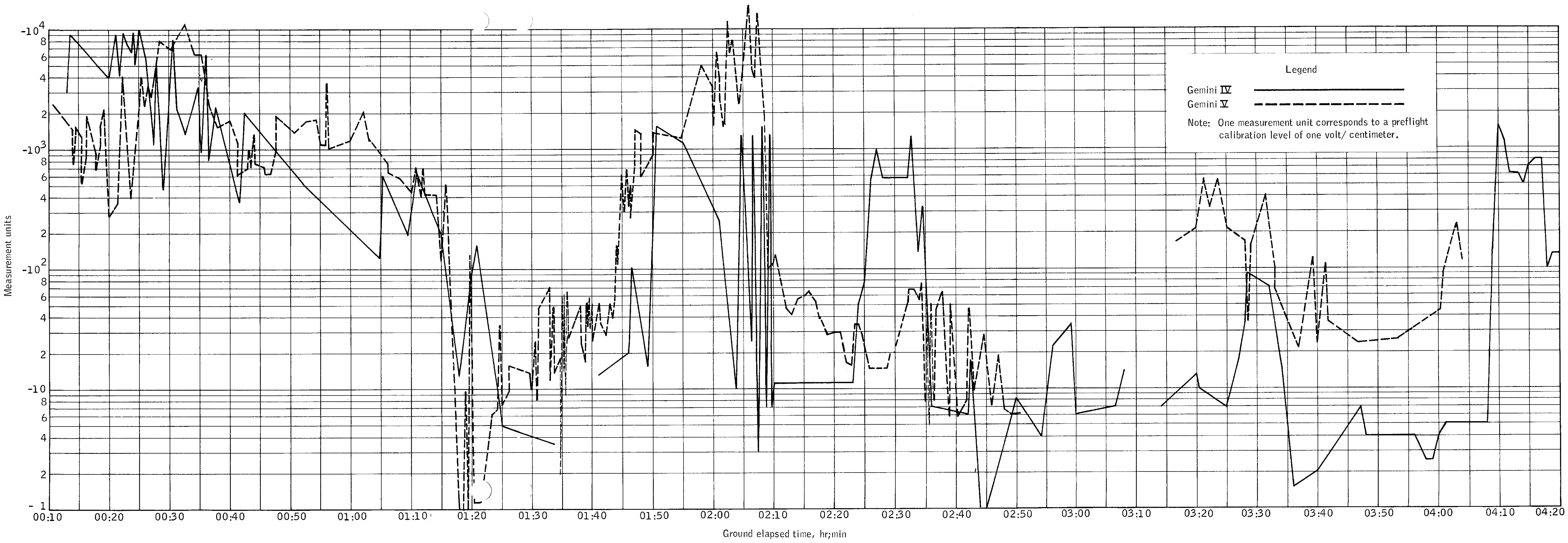

Figure 8.10-1. - Experiment MSC-1, Gemini IV and V instrument measurements during first three revolutions.

8.11 EXPERIMENT S-1, ZODIACAL LIGHT PHOTOGRAPHY

8.11.1 Objective

The objective of experiment S-1 was to obtain photographs of the zodiacal light, the earth's airglow, and other dim light phenomena such as the gegenschein. The gegenschein was postulated to be a dim illumination approximately in the anti-sun direction. No previous photographs of it existed. Detection of it by the human eye is at the limit of visibility and it is obscured from the ground by the earth's airglow, which is much brighter. In the two competing scientific theories concerning the gegenschein, one predicts that it should be directly in the anti-sun position, while the other (which assumes that the earth has a cometary tail) predicts a westerly displacement. In addition to these scientific photographs, it was also planned to photograph and determine the brightness of the spacecraft thruster exhaust.

8.11.2 Equipment

The experiment equipment consisted of a modified 35-mm camera with mounting brackets to position it in the right hatch window. The camera was designed to cover a wide field of view by rotating the short focal length f/1 lens from right to left about its optic center in the camera. This resulted in a field of view larger than 50° by 130°. Internal batteries, solid-state circuitry, and motors sequenced the entire program of exposures automatically once the camera was started. The starting circuit incorporated a photo sensor which prevented activation of the program until sunset. For the zodiacal light sequence, successive exposures were sequenced automatically, starting nominally with a $\frac{1}{2}$-second sweep, and then doubling each exposure until a nominal 2-minute sweep was made. All succeeding exposures were then of 2-minute duration with an interval of 24 seconds between exposures. Standard base tri-X film was used in this experiment.

8.11.3 Procedure

The inflight photographic procedure required the command pilot to acquire two celestial aiming points in the spacecraft optical sight mounted in the left hatch window. The spacecraft orientation could then be properly maintained to photograph the objects of interest with the mounted zodiacal light camera.

8.11.4 Results

The inflight photographic procedures appear to have been performed adequately by the flight crew. The initial aiming point on the Southern Cross was acquired well before sunset during revolution 46. Although the object of interest, the zodiacal light, was very near the sun when viewed from the right hatch window, the left hatch window was shaded from the sun, making acquisition of the aiming point possible. The programed series of exposures was performed as planned and frames 1 through 7 were exposed. In frames 1 through 5, the exposures are high due to twilight but Venus can always be seen, indicating that the spacecraft orientation was proper. In frame 6, many stars are visible as well as a bright central illumination which is not on the ecliptic and is, therefore, not the zodiacal light. The exposure of frame 6 was 39 seconds and the solar depression was 11°. Frame 7 was exposed for 81 seconds at a solar depression of 16°. It shows the zodiacal light, as well as the appearance of the airglow line, above the twilight-illuminated earth's limb. Part 1 of the experiment could be considered a success on the basis of this picture, except quantitative densitometry of the crucial area of the film cannot be accomplished due to a shadow on the film introduced by extraneous means. However, it does show that measurements of the zodiacal light can be made at elongation angles as small as 16°. This is shown in figure 8.11-1.

After 5 minutes of photographing the various phenomena after sunset, the spacecraft was maneuvered by the command pilot to the second aiming point, the constellation Grus. Two-minute exposures were made throughout the rest of the night orbit pointed at the second object of interest, the gegenschein area in the anti-sun direction.

Frames 8 through 17 are 160-second exposures. Frames 8 and 9 both show stars as well as spacecraft thruster exhaust glows (see figure 8.11-2). Together with data on the thruster firing, these photographs can be utilized for quantitative estimates of the thruster light level. Some parts of frame 8 are exposed to levels as high as 10^{-10} of the sun's surface brightness.

Frames 11 through 14 are exposures in the gegenschein direction but with a spacecraft roll of approximately 10° about the direction toward Grus. Frames 15 and 16 are good attitude holds and show spacecraft motion of only a few degrees. Both of these photographs show the gegenschein. Although the gegenschein is weakly exposed because it is so dim, the center of the illumination can be determined within several degrees and is approximately at the position of the star Aquarius. This is only 3° from being directly in the anti-sun direction and shows no measurable westerly displacement of the gegenschein. Figure 8.11-3 illustrates this phenomenon clearly; however, the contrast has been enhanced in this photograph. Figure 8.11-4 is a star chart for identifying the stars in figure 8.11-3. Part II of the experiment can be considered as completely successful.

8.11.5 Conclusions

The first part of the experiment demonstrated that the zodiacal light can be measured at angles as small as 16° from the sun. At this elongation, both the airglow layer and the sunlit horizon are visible. The second part of the experiment lead to the first successful photos of the gegenschein and shows that the gegenschein is within a few degrees of the anti-sun direction.

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This photograph shows the zodiacal light along the ecliptic with the sun 16° depressed. The airglow band and the horizon dimly lit show as parallel lines through a shadow on the film introduced by extraneous means. Antares is at the top of the photograph.

Figure 8.11-1. - Experiment S-1, photograph of the zodiacal light.

NASA-S-65-8564

This frame was exposed during the maneuver from Crux to Grus. The curved lines in the picture are due to stars, the brightest being Venus. The diffused illumination is from the thrusters. The light from the thrusters obscures all but the brightest stars.

Figure 8.11-2. - Experiment S-1, photograph of spacecraft thruster firing.

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This is a print of frame 15 in which the contrast has been enhanced. The gegenschein is clearly evident with its center at θ Aquarius and illuminating an approximately circular area of about 10 degrees diameter.

Figure 8.11-3. - Experiment S-1, photograph of the gegenschein.

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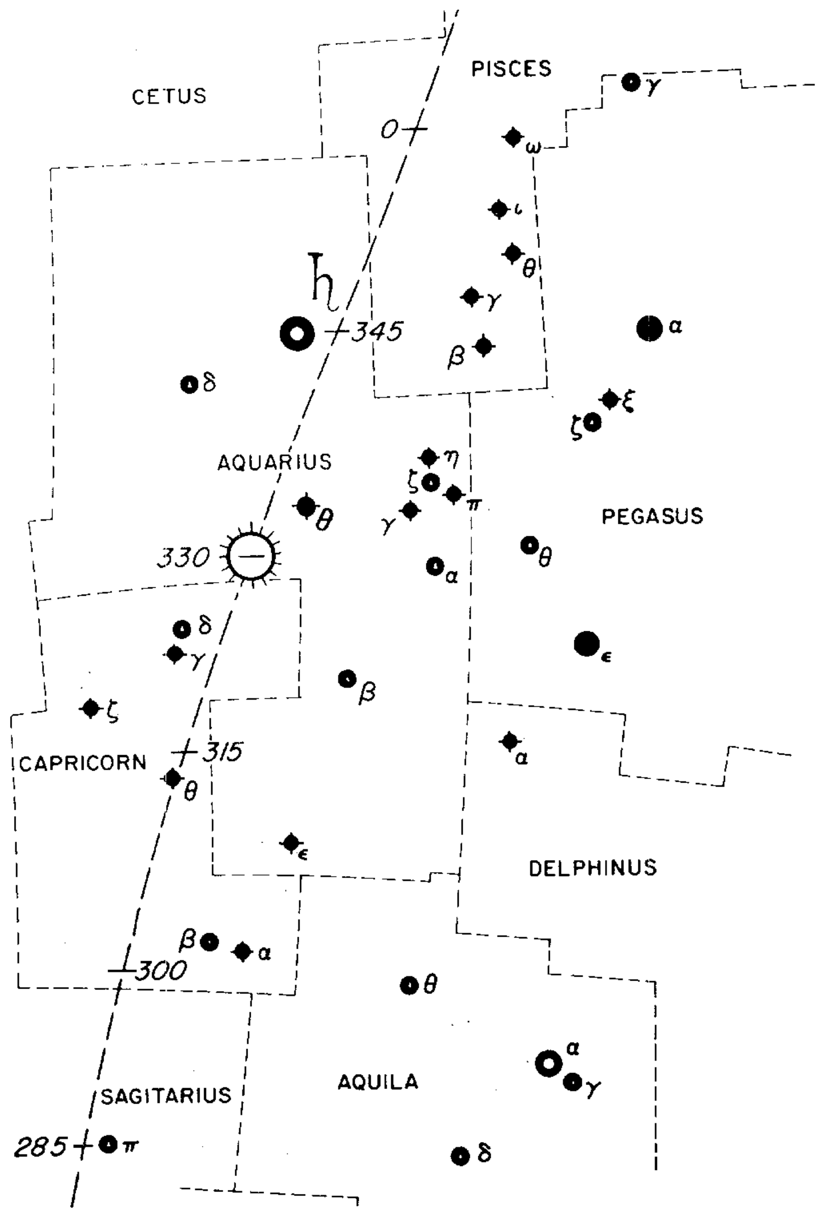

Identification of the stars in gegenschein photograph with the notation of Norton's Star Atlas. The anti-sun direction and Saturn ♄ are indicated.

Figure 8.11-4. - Experiment S-1, star chart.

8.12 EXPERIMENT S-5, SYNOPTIC TERRAIN PHOTOGRAPHY

8.12.1 Objective

The objective of experiment S-5 was to obtain high-quality color photographs of terrain features for geological and geographic purposes. The following two types of pictures were desired:

(a) Pictures of well-known areas, such as the United States, which could serve as standards for interpretation of lesser known areas.

(b) Pictures of remote regions, such as the central Sahara, which are poorly covered by existing photography.

8.12.2 Equipment

Four 70-mm camera film magazines loaded with color film were onboard the spacecraft, each having a 55-frame capability. The equivalent of one-half magazine was allotted for experiment S-5. These magazines were used with a 70-mm still camera also required for other experiments. A haze filter was available to be used at the discretion of the crew.

8.12.3 Procedure

Subject to fuel and power restrictions, the crew was instructed to take vertically oriented, systematic overlapping pictures of Mexico, Africa, Australia, and any other areas showing terrain features of possible interest. As in the previous Gemini IV flight, it was stressed that almost any picture of the earth's surface was valuable, even if the optimum procedure could not be followed exactly.

8.12.4 Results

The crew was hampered in taking terrain photographs by spacecraft fuel and power restrictions which necessitated drifting flight for a portion of the mission. In particular, it was not possible to take systematic strip photography. Most of the pictures obtained were taken as an opportunity arose with random spacecraft orientation. A total of 250 frames of 70-mm film (in 4 magazines) was exposed. Of this unexpectedly large number, 172 are considered usable for terrain study. The quality ranges from excellent to poor, depending on factors such as cloud cover, lighting conditions, and spacecraft depression angle.

Areas previously photographed from space and photographed again during the Gemini V mission include the southwestern United States, Egypt, Tibet, northern Mexico, Florida, the Bahama Islands, and parts of northern Africa and Southwest Asia. In addition, photographs of many areas never photographed before during a manned mission were obtained. These include southern Mexico, Japan, Australia, southern Africa, the Hawaiian Islands, parts of southeast Asia, and several Pacific Islands.

Pending completion of a detailed analysis, a few general comments can be made concerning the photographs obtained. Based on a review by the crew and a comparison with pictures from previous flights, color rendition is generally good, and, in many pictures, is outstanding. Ground resolution is remarkably high; many small roads, canals, pipelines, and similar features are clearly visible. Figure 8.12-1 shows two typical synoptic terrain photographs, one of lower California and the other of Iran.

One especially valuable aspect of the photographs is the abundant coverage of near-shore areas. Considerable bottom topography and water current structure is visible, making these pictures of great value in planning future photography for studies in oceanography and marine geology.

8.12.5 Conclusions

Experiment S-5 can be classified as highly successful. Limitations in quality caused by spacecraft orientation problems were compensated for by the extremely wide variety of terrain features photographed, and by the crew's alertness in taking pictures-of-opportunity. Detailed study of the photographs is expected to provide new insight into many scientific questions and into problems concerning photographic mission planning.

NASA -S-65- 8609

(a) Southern California, the Salton Sea, and the Imperial Valley. View to the southeast showing several major faults.

Figure 8.12-1. - Experiment S-5, typical synoptic terrain photography.

NASA-S-65-8605

(b) Zagros mountains, Iran, 100 miles east of Shiraz. Note geologic structure and intermittent lakes.

Figure 8.12-1. - Concluded.

8.13 EXPERIMENT S-6, SYNOPTIC WEATHER PHOTOGRAPHY

8.13.1 Objective

The principal objective of experiment S-6 was to obtain high-quality color photographs of a variety of cloud systems, in particular, systems that could be compared with similar views obtained from unmanned meteorological satellites. A second objective was to obtain views of the cloud systems on successive orbital revolutions to show the short-period development and movement of clouds and cloud systems.

8.13.2 Equipment

Four 70-mm camera film magazines loaded with color film were on-board the spacecraft, each having a 55-frame capability. The equivalent of one-half magazine was allotted for experiment S-6. These magazines were used with a 70-mm still camera also required for other experiments. A haze filter was available to be used at the discretion of the crew.

8.13.3 Procedure

Prior to the flight, the crew was briefed on the various types of weather systems of interest for the experiment. During the mission, meteorologists from the Weather Bureau's National Weather Satellite Center utilized pictures from TIROS weather satellites and worldwide weather maps to select specific areas likely to contain various cloud patterns of interest. When operationally feasible, this information was communicated to the crew so they could look for, and, when possible, photograph these patterns. In addition, views were to be taken of interesting clouds which the crew observed and had time to photograph.

8.13.4 Results

Despite fuel and power restrictions on orientation of the spacecraft during the flight, a total of approximately 175 pictures containing clouds was obtained. All but a few were of very high quality. A variety of weather systems were photographed, including tropical storm Doreen in the Pacific, typhoon Lucy near Japan, other tropical disturbance areas, eddy patterns in the lee of subtropical islands and peninsulas, the contrasting cloudiness over water and adjacent land areas in many parts of the world, snow and clouds over mountain areas, and drifting smoke from forest fires.

Of particular interest were pictures of Florida taken on three successive revolutions showing the diurnal changes in cloud and thunderstorm activity over the land and the adjacent water area. These photographs are shown in figure 8.13-1.

8.13.5 Conclusions

Experiment S-6 can be classified as a complete success. The photographs obtained will be of great value for meteorological studies. Detailed study of the many photographs will require extensive analysis and evaluation. This information was not available for this report. When all data are available, comparisons will be made among a selection of the experiment photographs and corresponding TIROS views.

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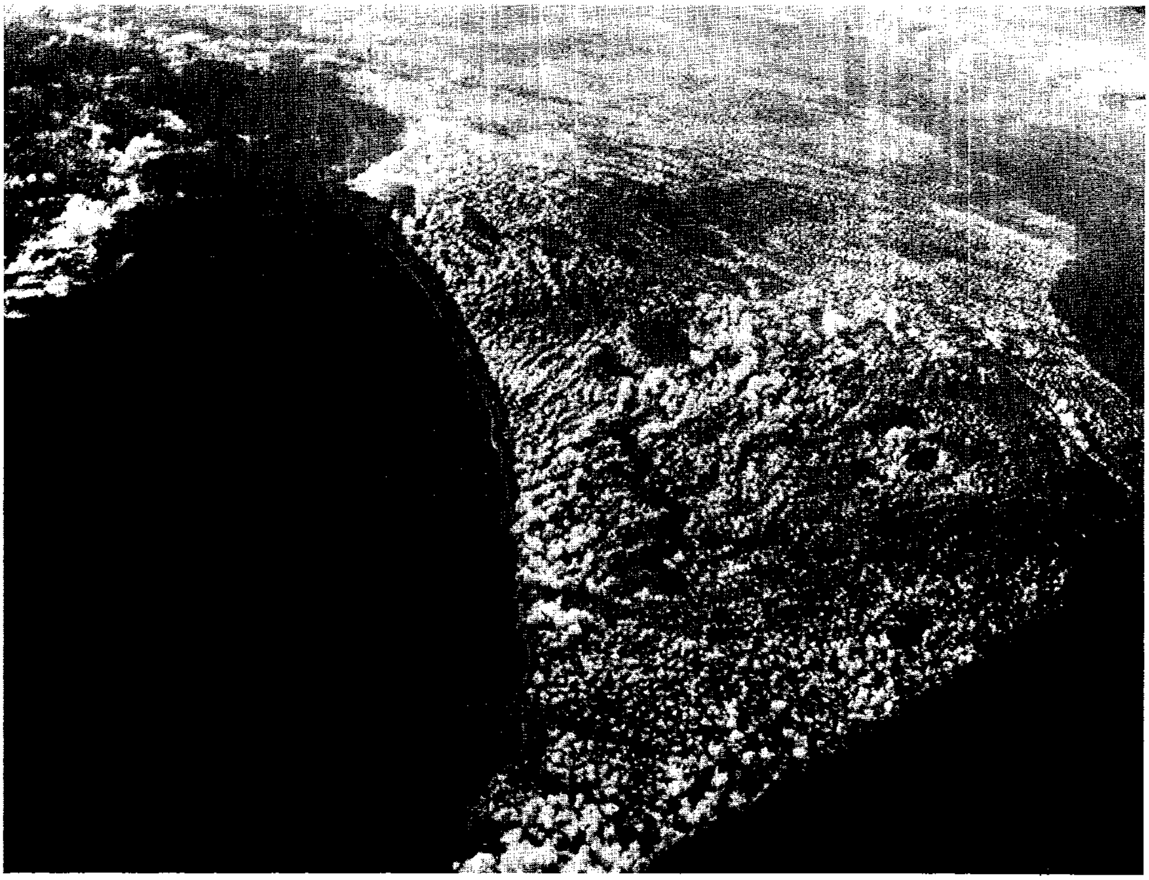

(a) View taken from the north at approximately 10:31 a.m. e.s.t. Cumulus clouds are numerous over the land with clear skies over the larger lakes.

Figure 8.13-1. - Experiment S-6, a series of three typical synoptic weather photographs taken on successive orbital revolutions over Florida on August 22, 1965.

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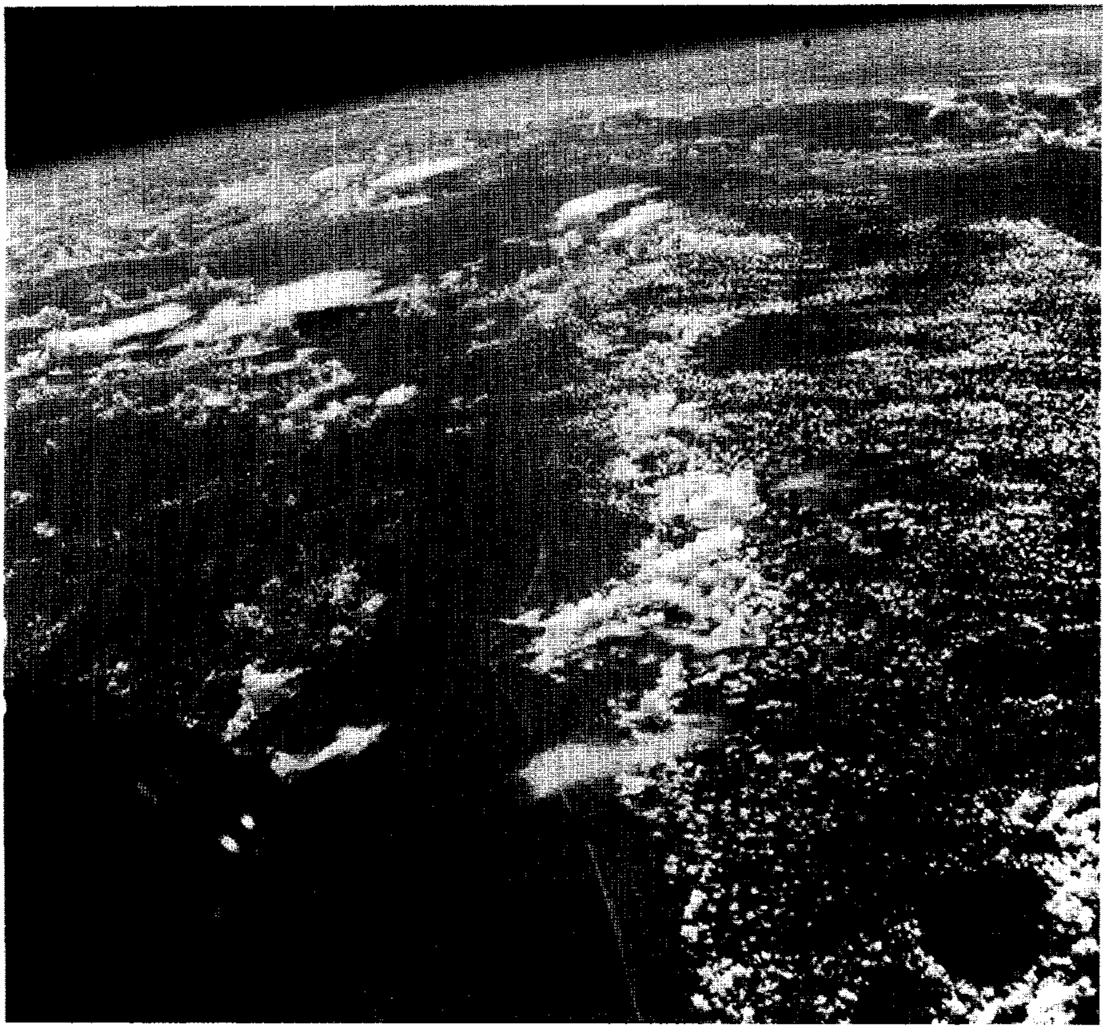

(b) View taken from the north at approximately 12:07 p.m. e.s.t. In the time since view (a) was taken (about 1 1/2 hours), many of the clouds over eastern Florida have reached the towering cumulus stage. Some show the anvil tops of mature thunderstorms as do several over the Bahama Islands.

Figure 8.13-1. - Continued.

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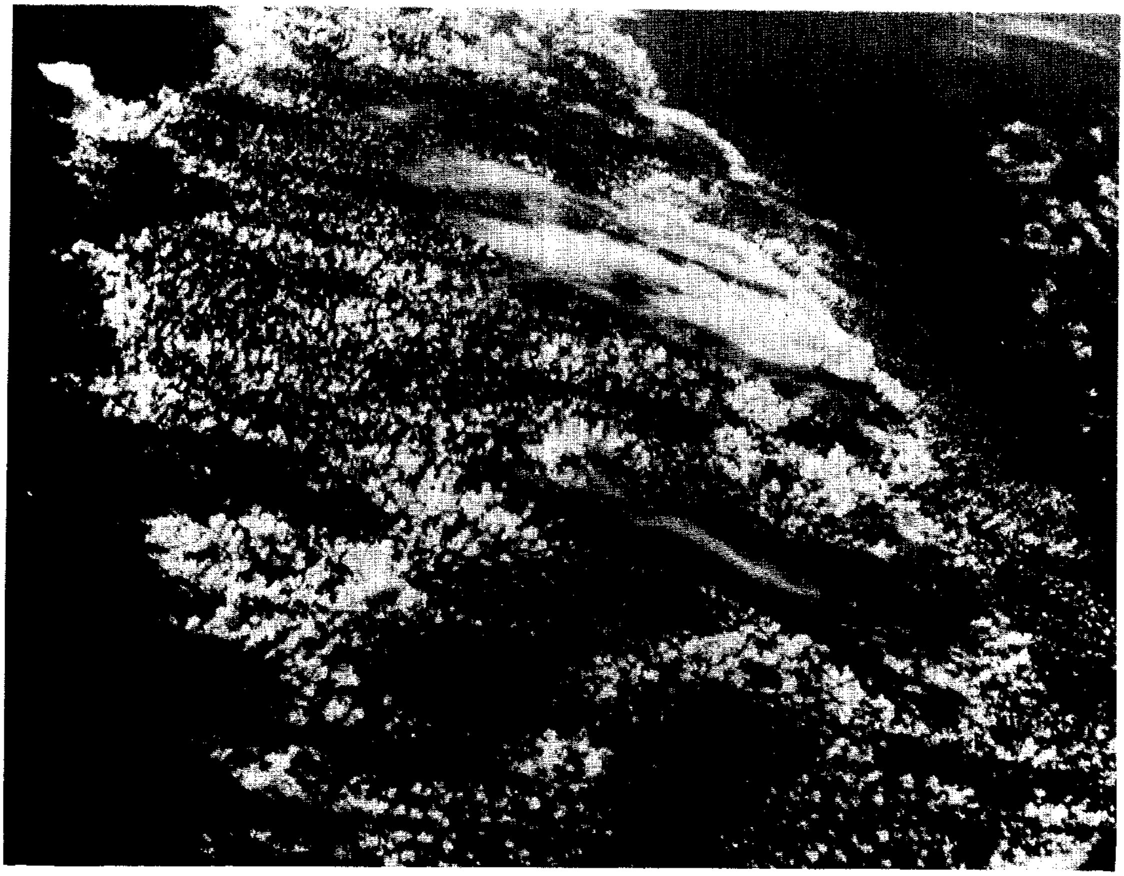

(c) View taken from the south at approximately 1:38 p.m. e.s.t. Large anvil tops of thunderstorms are prominent along the line of maximum cloudiness in eastern Florida.

Figure 8.13-1. - Concluded.

8.14 EXPERIMENT S-7, CLOUD TOP SPECTROMETER

8.14.1 Objective

The objective of experiment S-7 was to use a simple hand-held spectrograph to investigate the possibility of measuring cloud-top altitudes from satellites.

8.14.2 Equipment

The equipment consisted of a spectrograph fitted with a 35-mm camera body. The dispersing element in the spectrograph was a 3.0 by 3.2 cm replica grating with 1200 lines per millimeter blazed for 7500 Å in the first order. A filter with a short wavelength cutoff at 6700 Å was used to eliminate higher order spectra. The field lens (25-mm focal length, f/1.9) imaged the cloud on the 0.1 by 6.0 mm entrance slit. The 75-mm focal length, f/1.9 collimator lens collimated the beam of light that fell normally on the grating. The dispersed beam from the grating was focused on the lower part of the film frame by a 131-mm focal length lens. The instrument recorded the spectrum between 7500-7800 Å with a resolution of 5 Å. Hence, the oxygen A-band at 7600 Å was recorded on the film.

The instrument also photographed the cloud on the upper part of the film frame by means of a 75-mm focal length, f/4.5 secondary lens. The shutter of this lens was coupled to the camera body shutter so that the spectrum and the photograph of the cloud were obtained simultaneously.

High speed infrared film was used in the experiment. The film recorded the amount of absorption by oxygen in the path of the reflected solar ray from the cloud. The absorption of oxygen is related to the cloud-top altitude since oxygen has a constant mixing ratio in the atmosphere below approximately 100 Km.

8.14.3 Procedure

To keep the crew informed of the most promising cloud cover areas for the experiment, a 24-hour weather watch was begun at the National Meteorological Center, Suitland, Md., the day before the flight. In flight, the same information passed to the crew was transmitted also to the Air Weather Service, Fleet Weather Service, and the meteorologists at airports at appropriate locations. Aircraft on flights over areas of interest were provided with printed forms on which to log cloud-top data. Observations on clouds by pilots of civilian and military aircraft were planned to coincide in time and place, as nearly as possible, with scheduled photo-spectra exposures from the spacecraft. As areas in which aircraft support was available were limited, observations were scheduled in only three geographical areas, namely, the Caribbean, the Eastern Pacific, and the Philippine Islands. One of the Weather Bureau planes operating out of Miami, Florida, was on stand-by for the experiment, and made a special flight to measure the altitude of a thunderstorm over Key West.

As selected orbital points were approached, the crew was instructed to orient the spacecraft so that the right hatch window faced the earth's surface, and the spectrograph was operated when the spacecraft was over the desired cloud cover.

8.14.4 Results

Out of the 30 planned observations, 27 were obtained on different types of clouds. Observations were obtained of stratus off California, a tropical storm behind Doreen, stratus off the Philippine Islands, tropical storm Doreen, another tropical storm, and stratus off Guam and Florida. It is interesting to note the variation in the type of clouds photographed such as the stratus off Baja California, which was essentially a low cloud reaching 3000 ft, and tropical storm Doreen, which was essentially a very high cloud that might have reached 40 000 ft or more. Figure 8.14-1 shows the respective spectra of these two types of clouds. Figure 8.14-2 shows a print from the original film on which the spectrum and the cloud photograph are clearly visible.

Data from over 50 observations of direct measurements of cloud heights from civilian and military aircraft have been received with additional data being forwarded.

8.14.5 Conclusions

Experiment S-7 can be considered as an apparent success. The final results will not be known until all the relevant data are collected and analyzed.

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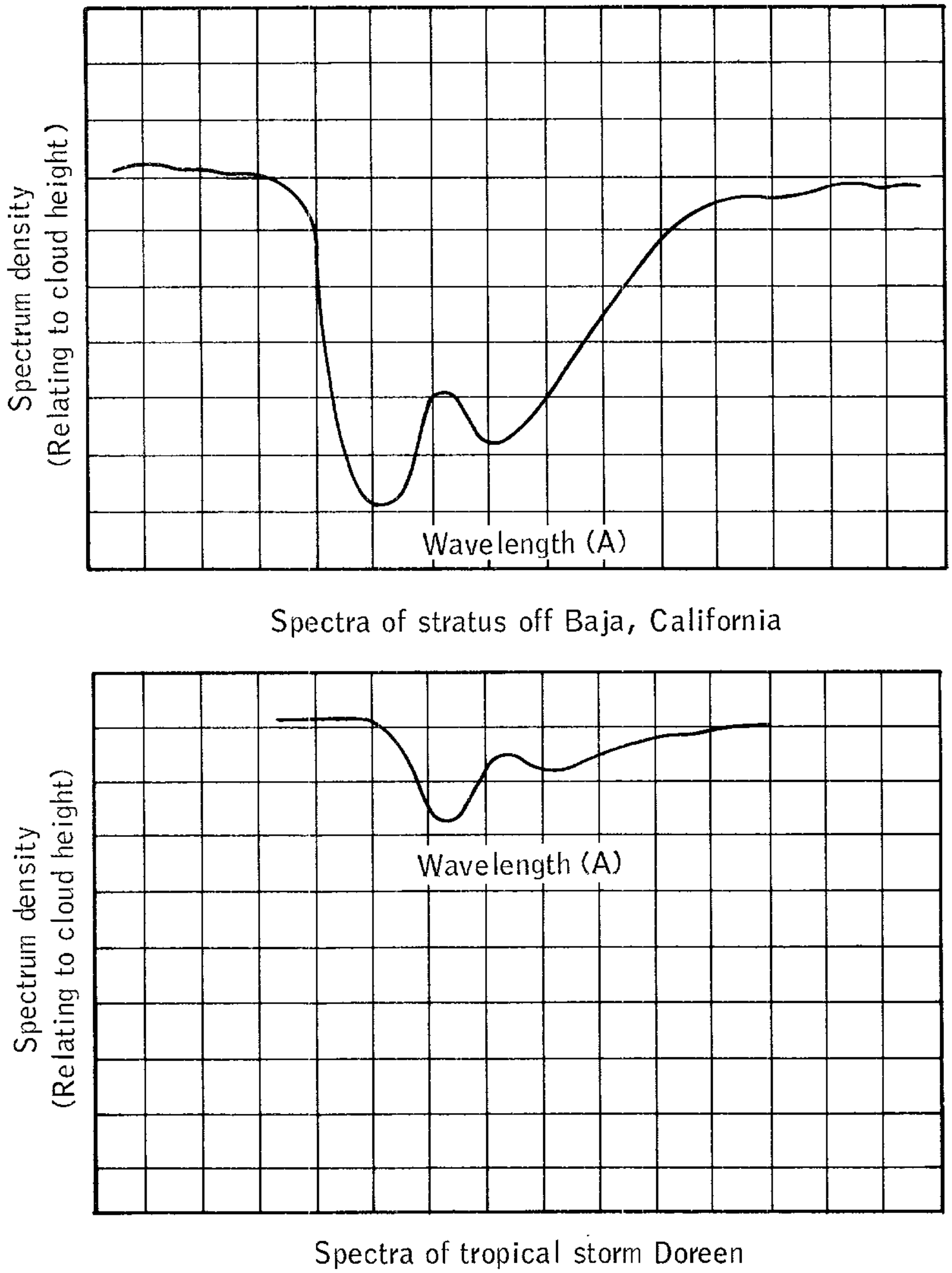

Figure 8.14-1. - Experiment S-7, spectra of two types of clouds.

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Figure 8.14-2. - Experiment S-7, print from original experiment film illustrating the spectrum and the cloud photograph.

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8.15 EXPERIMENT S-8/D-13, VISUAL ACUITY AND ASTRONAUT VISIBILITY

8.15.1 Objective

The first objective of experiment S-8/D-13 was to measure the visual acuity of the flight crew members before, during, and after long-duration space flights in order to ascertain the effects of a prolonged spacecraft environment. The second objective was to test the use of basic visual-acuity data combined with measured optical properties of ground objects and their natural lighting, the atmosphere, and the spacecraft window to predict the limiting naked-eye visual capability of the flight crew to discriminate small objects on the surface of the earth in daylight.

8.15.2 Equipment

The experimental equipment consisted of an in-flight vision tester for testing visual acuity, an in-flight photometer to monitor the spacecraft window, and test patterns at two ground observation sites.

8.15.2.1 In-flight vision tester.- The in-flight vision tester was a small, self-contained binocular optical device containing a trans-illuminated array of 36 high-contrast and low-contrast rectangles, half of which were oriented vertically in the field of view while the remainder were horizontal. Rectangle size, contrast, and orientation were randomized; the presentation was sequential, and the sequences were non-repetitive. Each rectangle was viewed singly at the center of a 30° adapting field, the apparent luminance of which was approximately 100 foot-lamberts. Both members of the flight crew made forced-choice judgments of the orientation of each rectangle and indicated their responses by punching holes in a record card. Optical alinement was accomplished by means of a bite-board equipped with the flight crew member's dental impression. Electrical power for illumination within the instrument was derived from the spacecraft.

8.15.2.2 In-flight photometer.- A photoelectric photometer was mounted near the lower right corner of the right hatch window to measure the amount of ambient light scattered by the window into the path of sight at the moment when observations of the ground test patterns were to be made. The photometer had a narrow (1.2°) circular field of view into the opening of a small black cavity a few inches away from the outside of the right hatch window. The photometric scale was linear and extended from 60 to 3000 foot-lamberts. Since the apparent luminance of the black cavity was always less than 60 foot-lamberts, any reading of the photometer was ascribable to ambient light scattered by the window. This information, combined with data on the beam transmittance of the window and on the apparent luminance of the background squares in the ground array, enabled the contrast transmittance of the window at the moment of observation to be calculated.

8.15.2.3 Ground observation sites.- Ground observation sites were provided on the Gates Ranch, 40 miles north of Laredo, Texas, and the Woodleigh Ranch, 90 miles south of Carnarvon, Australia. At the Texas site, twelve 2000 by 2000 feet squares of plowed, graded, and raked soil were arranged in a 4 by 3 matrix. White rectangles of styrofoam-coated wallboard were laid out in each square. Their length decreased in a uniform logarithmic progression from 610 feet in the northwest corner (square number 1) to 152 feet in the southwest corner (square number 12) of the array. Each of the 12 rectangles was oriented in 1 of 4 positions (i.e., north-south, east-west, or diagonal) and the orientations were random within the series of 12. Advance knowledge of the rectangle orientations were withheld from the flight crew since their task was to report the orientations. Provision was made for changing the rectangle orientations between passes and for adjusting their size in accordance with anticipated slant range, solar elevation, and the visual performance of the flight crew on preceding passes.

8.15.3 Procedure

Both of the flight crew members completed five or more preflight sessions in a laboratory training van during which they became experienced in psychophysical techniques and established physiological baselines descriptive of their individual visual performance. The statistical fluctuations in that performance were established, providing a means by which the ground pattern observations could be interpreted.

8.15.4 Results

In-flight vision tests were to be performed once each day by each crew member. Ground observations were to be made by the pilot with the command pilot orienting the spacecraft as prescribed in the flight plan. The results of these tests, together with preflight and postflight test results, are shown in figure 8.15-1. Unfavorable cloud conditions caused some scheduled observations of the ground markings to be deleted. In the latter part of the mission, lack of thruster control made observation of the ground patterns impossible.

Quantitative observation of ground marking was achieved only once. This occurred at the ground observation site near Laredo, Texas, during revolution 48. Despite the fleeting nature of the observation, there exists a reasonable probability that the pilot correctly discriminated the rectangles in the sixth and seventh squares. Since forced-choice responses to squares 8 through 12 were not given, presumably due to lack of viewing time, it can only be inferred that the threshold lay at square 6 or higher. Tentative values of the apparent contrast and angular size of the sixth and seventh rectangles at the Laredo site at the time of the observation are plotted in figure 8.15-2. The solid line in the illustration represents the preflight visual performance of the pilot as measured in the training van and the dashed lines represent the 1- and 2-sigma limits of his visual performance. The positions of the plotted points indicate that his visual performance at the time of revolution 48 was within the statistical range of his preflight visual performance.

8.15.5 Conclusions

Experiment S-8/D-13 appears to have achieved successfully both of its stated objectives. Data from the in-flight vision tester is complete and of high quality; preliminary evaluation indicates that the visual performance of the astronauts was not degraded during the 8-day mission. Results from observation of the ground site near Laredo, Texas, appear to confirm that the visual performance of the pilot during space flight was within the statistical range of his preflight visual performance and that laboratory visual acuity data can be combined with environmental optical data to predict correctly the limiting visual capability of astronauts to discriminate small objects on the surface of the earth in daylight.

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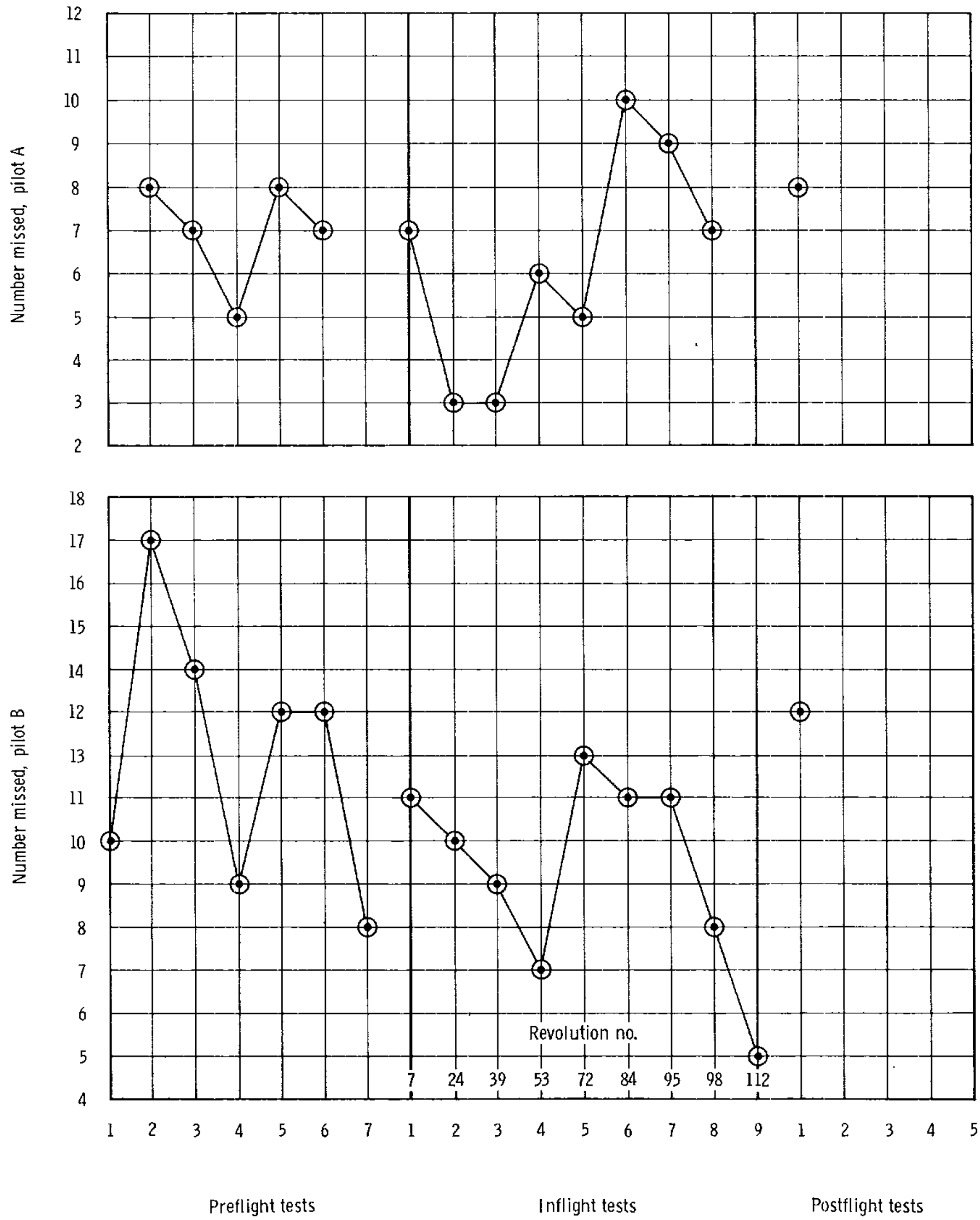

Figure 8.15-1. -Experiment S-8/D-13,vision tester results.

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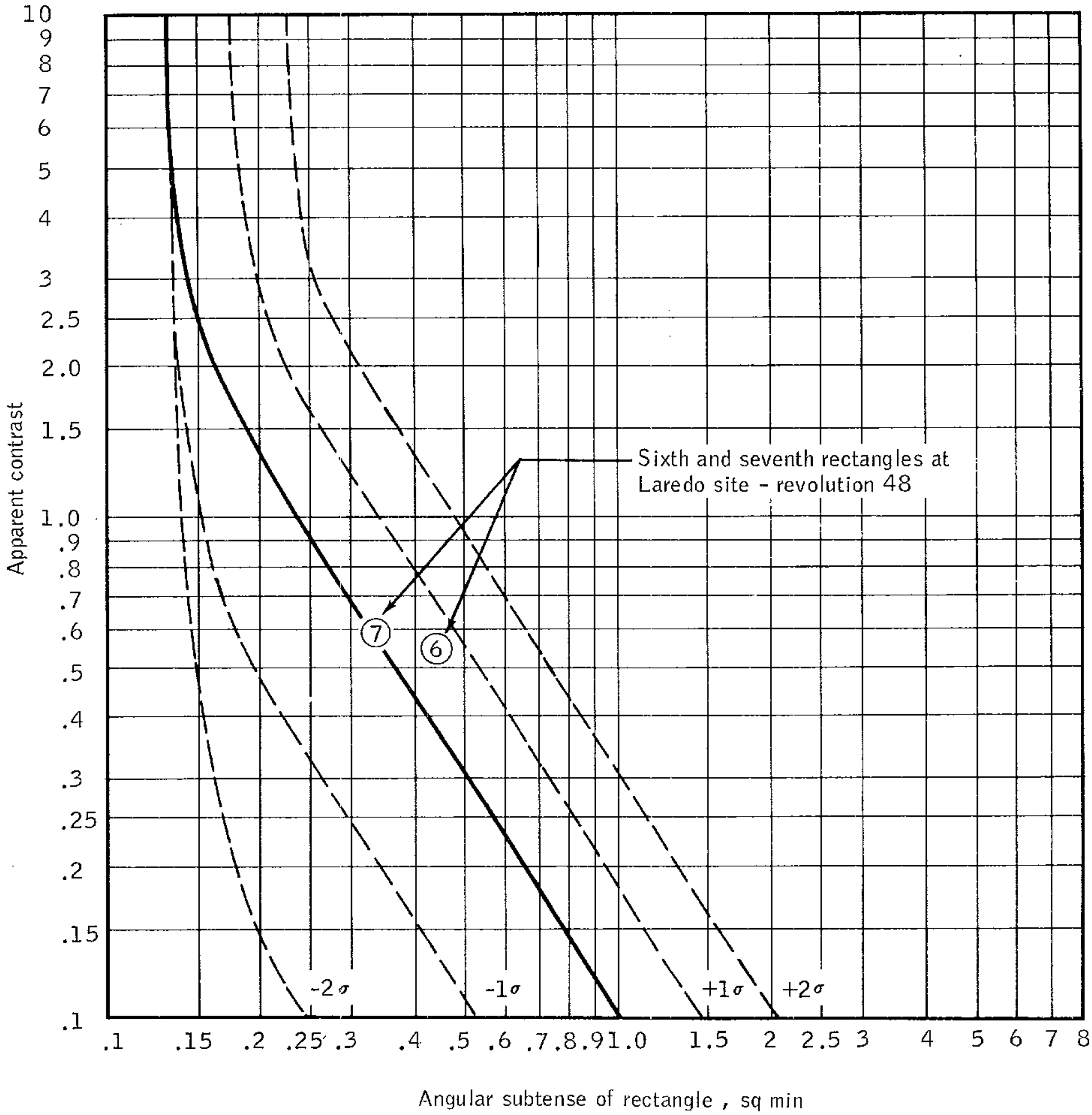

Figure 8.15-2. - Experiment S-8/ D-13, preflight visual thresholds - Pilot.

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9.0 CONCLUSIONS

The performance of the spacecraft, launch vehicle, flight crew, and mission support was satisfactory for the Gemini V mission. The objectives of the mission were met with three exceptions: the evaluations of the rendezvous guidance and navigation system in conjunction with the rendezvous evaluation pod and the capability of either pilot to maneuver the spacecraft in orbit to a close proximity with another object were not conducted because of the decision to power down the spacecraft. Also, the attempted controlled reentry resulted in a landing 19.7 miles off track and 89.3 miles short of the planned landing point.

The flight contributed significantly to the knowledge of manned space flight, especially in the area of long-duration flight and man's ability to operate for this period of time in space.

The following conclusions were obtained from the evaluation of the Gemini V mission:

1. The satisfactory performance of the Gemini launch vehicle and its associated ground guidance equipment resulted in the most nominal insertion conditions yet achieved in the Gemini Program.

2. An unsatisfactory method was used to determine whether a proper charge had been attained on the Gemini launch-vehicle oxidizer standpipe. This resulted in an improper charge of the system, which allowed noticeable build-up of a longitudinal oscillation (POGO) for a short period during stage I flight.

3. The overall voice communications of the Gemini V mission were excellent. It was demonstrated that there is a sufficient design performance margin for communications between Mission Control Center-Houston and the flight crew.

4. The environmental control system of the spacecraft provided a satisfactory cabin environment, free from excessive humidity and objectionable odors, and maintained the crew at a comfortable temperature during the entire mission.

5. The overall operation of the rendezvous radar system was satisfactory, and the system demonstrated the necessary range and angle tracking functions to be considered acceptable for future rendezvous missions.

6. The fuel-cell power system operated very satisfactorily. The system provided the necessary voltage and amperage levels required throughout the mission, and the lower pressure realized from the oxygen supply tank had no effect on the system operation. This flight demonstrated the capability to store cryogenic fluids for long periods of time in space.

7. The orbital attitude and maneuver system exhibited a significant failure after 5 days, and showed progressive degradation from that time to the end of the flight.

8. The basic crew station design was found satisfactory for long-duration flight with only minor discrepancies which are the subject of recommendation 6 in section 10.0.

9. Stowage of waste material, experiments, and operational equipment required a greater amount of time than expected. The crew stated they were overly burdened with these tasks.

10. Cabin lighting was adequate for dayside operations. However, when the crew was required to perform operations that required reading instruments as well as looking out the window, it was necessary to use the light-polarized filters in the windows. The cabin lighting was marginal during nightside operations for reading panel instruments, because the center cabin light had to be reduced considerably to distinguish the horizon or stars.

11. Rehydratable foods were highly palatable, whereas bite-size foods became flavorless and were too dry and rich.

12. The new urine disposal system was satisfactory except for deterioration of the latex receivers, which reacted chemically to the urine.

13. The Gemini V flight proved that man can stay in earth orbit for extended periods of time with no inflight or postflight medical effects which might affect his ability to perform normal tasks.

10.0 RECOMMENDATIONS

The following recommendations are made as a result of the evaluation of the Gemini V mission:

1. The manual charging equipment and/or procedures should be modified to insure that a proper charge has been obtained on the Gemini launch vehicle standpipe, in case of a malfunction of the automatic charging equipment.

2. A more reliable system to insure against frozen propellants should be provided for the OAMS thrusters and supply lines. This system should use minimum ampere-hours over the mission profile.

3. Activities associated with experiments and operational checks should be minimized to the maximum extent possible during the first two to three revolutions, which should be devoted to checkout of spacecraft systems.

4. A firm freeze date should be established on operational equipment, experiments, and the flight plan. This freeze date should be formal and honored by all concerned.

5. Continued emphasis should be applied toward securing optimum quality control during all phases of preparation for the mission.

6. The following recommendations are made to correct the minor discrepancies in the crew station:

(a) The computer start light should have dimming capability.

(b) The utility lights should be strengthened to prevent breakage under normal handling. Stowage should be corrected to prevent high stress in the light assembly upon removal.

(c) The reported high temperatures of the cabin lights and associated dimming controls should be investigated and corrective action taken if required.

(d) The light polarized window filters should be redesigned to prevent surface scratching and control knob breakage under normal use.

(e) The fuel-cell oxygen purge switch configuration should be modified to operate without being held manually for 2 minutes.

(f) The attitude reference marks on the windows should be identified to avoid confusion, and marks should be added for the 0°-0°-0°, horizon scan, and 60°-60° bank horizons.

(g) Shielding should be provided for the optical sight to eliminate the reflections from the sun and the center cabin light.

(h) Additional velcro should be installed overhead for stowage of loose items during flight.

(i) The green stowage bags should be modified to provide better access and should be fabricated from more durable material.

7. A low residue diet should be used a few days prior to flight in order to cut fecal bulk to a minimum during flight. Medication should be carried in the medicine kit in case inflight looseness develops.

8. The T-2 day physical should be moved further back from flight day to prevent undue stress on the crew this close to the flight.

9. Emphasis should be placed on the use of rehydratable food rather than bite-size food, and greater attention should be given to providing a more appetizing array of foods.

10. The fabrication materials, methods, and sequences used for manufacturing rehydratable food bags should be reviewed. Corrective action should be taken to prevent the blocking of the feeder ports which led to the inflight failures in Gemini V.

11. Sufficient time should be allocated in the flight plan each day to enable the crew to review, plan, and accomplish stowage activities.

12. A minimum of 4 hours should be allowed for stowage purposes for reentry.

13. More out-the-window training should be made available to the flight crew for attitude reference.

14. The M-1 experiment equipment should be redesigned to reduce the noise level.

11.0 REFERENCES

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12.0 APPENDIX

12.1 VEHICLE HISTORIES

12.1.1 Spacecraft Histories

Spacecraft histories at the contractor's facilities at St. Louis, Missouri, are shown in figures 12.1-1 and 12.1-2, and at Cape Kennedy, Florida, in figures 12.1-3 and 12.1-4. Figures 12.1-1 and 12.1-3 are summaries of activities with emphasis on spacecraft systems testing and prelaunch preparation. Figures 12.1-2 and 12.1-4 are summaries of significant, concurrent problem areas.

12.1.2 Gemini Launch Vehicle Histories

Gemini launch vehicle (GLV) histories at the contractor's facilities at Denver, Colorado, and Baltimore, Maryland, and at Cape Kennedy, Florida, are shown in figures 12.1-5 and 12.1-6. Concurrent problem areas and significant manufacturing activities are shown with the GLV test and prelaunch preparation activities.

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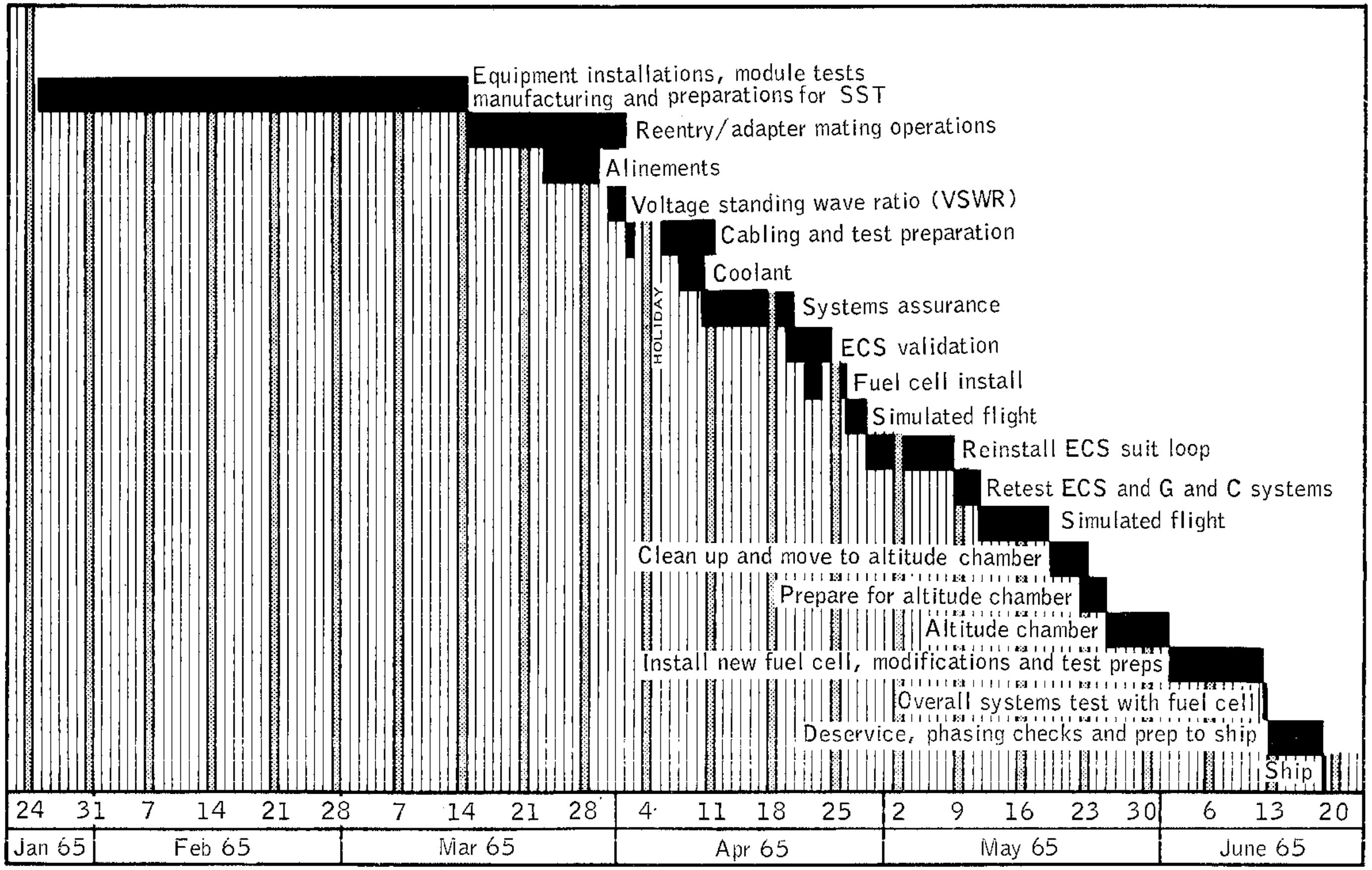

Figure 12.1-1.- Spacecraft 5 test history at contractor facility.

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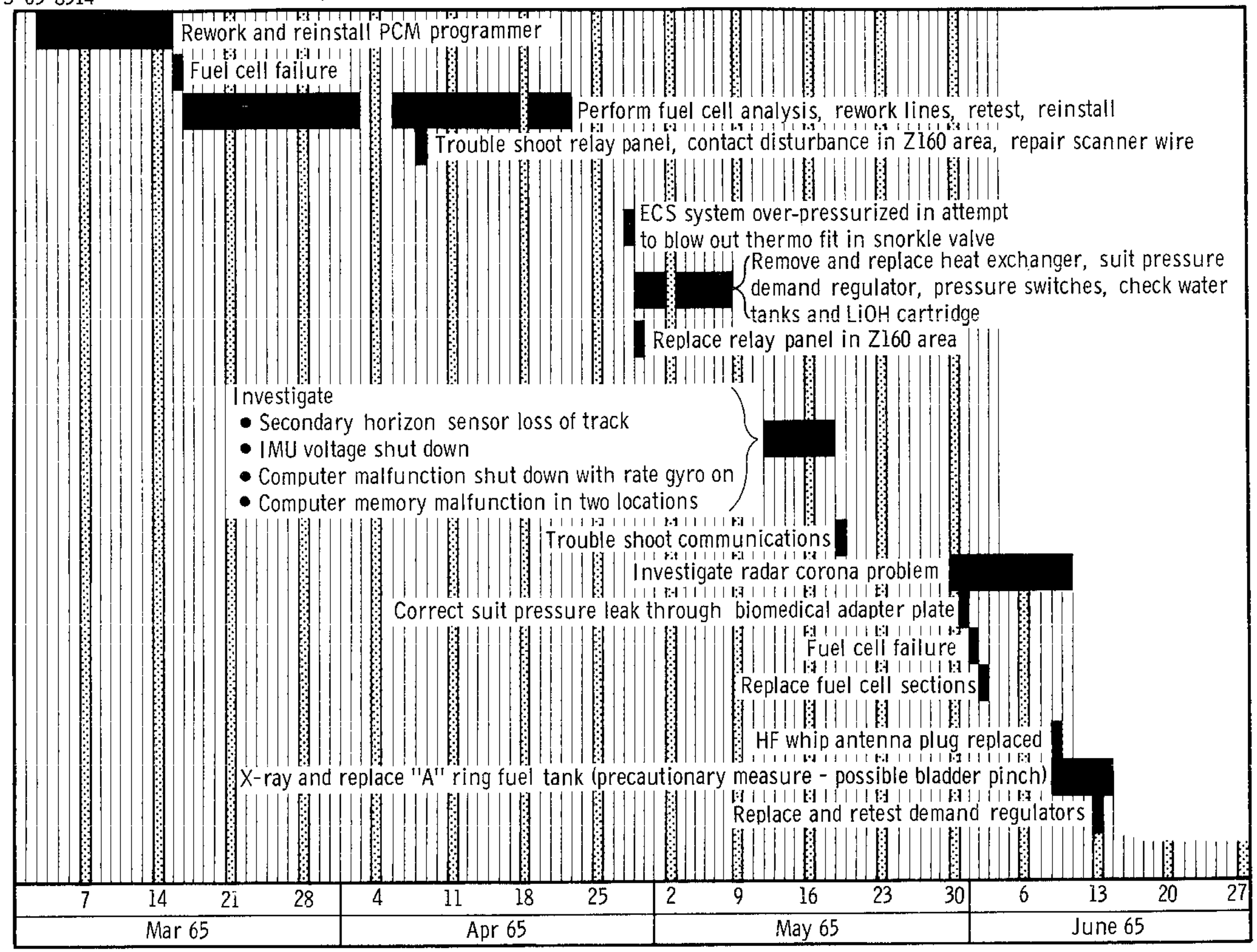

Figure 12. 1-2. - Spacecraft 5-significant problem areas at contractor facility.

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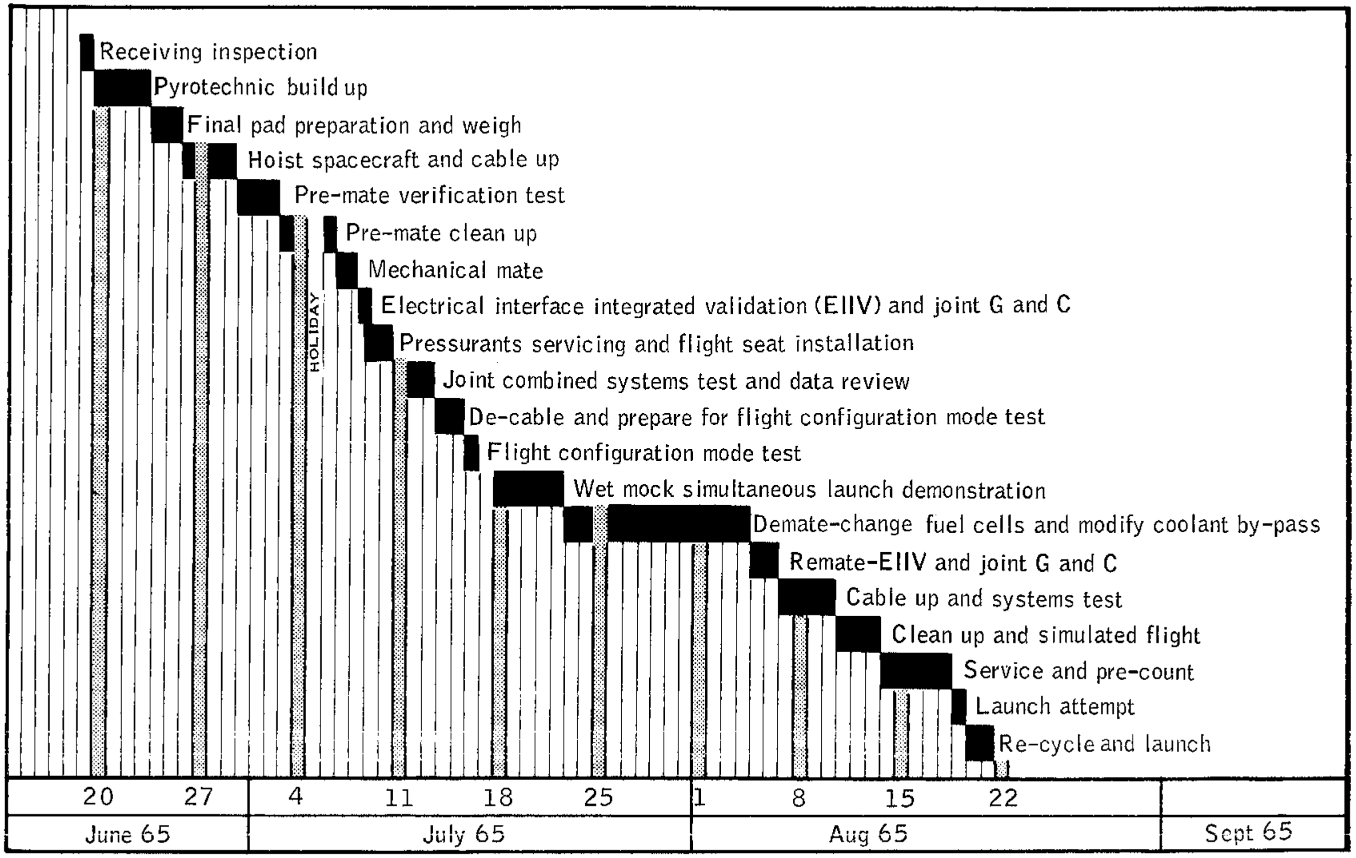

Figure 12.1-3. - Spacecraft 5 test history at Cape Kennedy.

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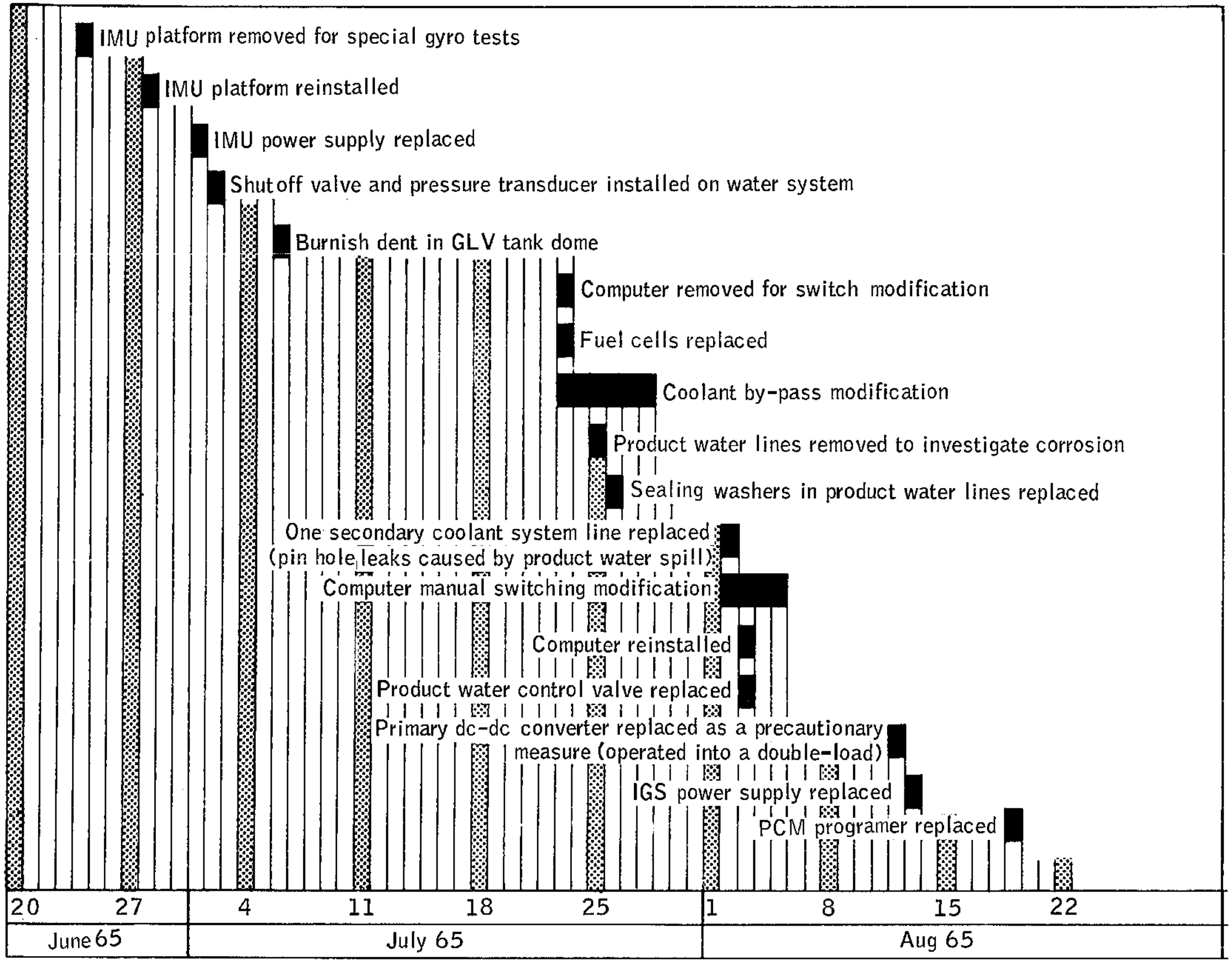

Figure 12.1-4. - Spacecraft 5 significant problems at Cape Kennedy.

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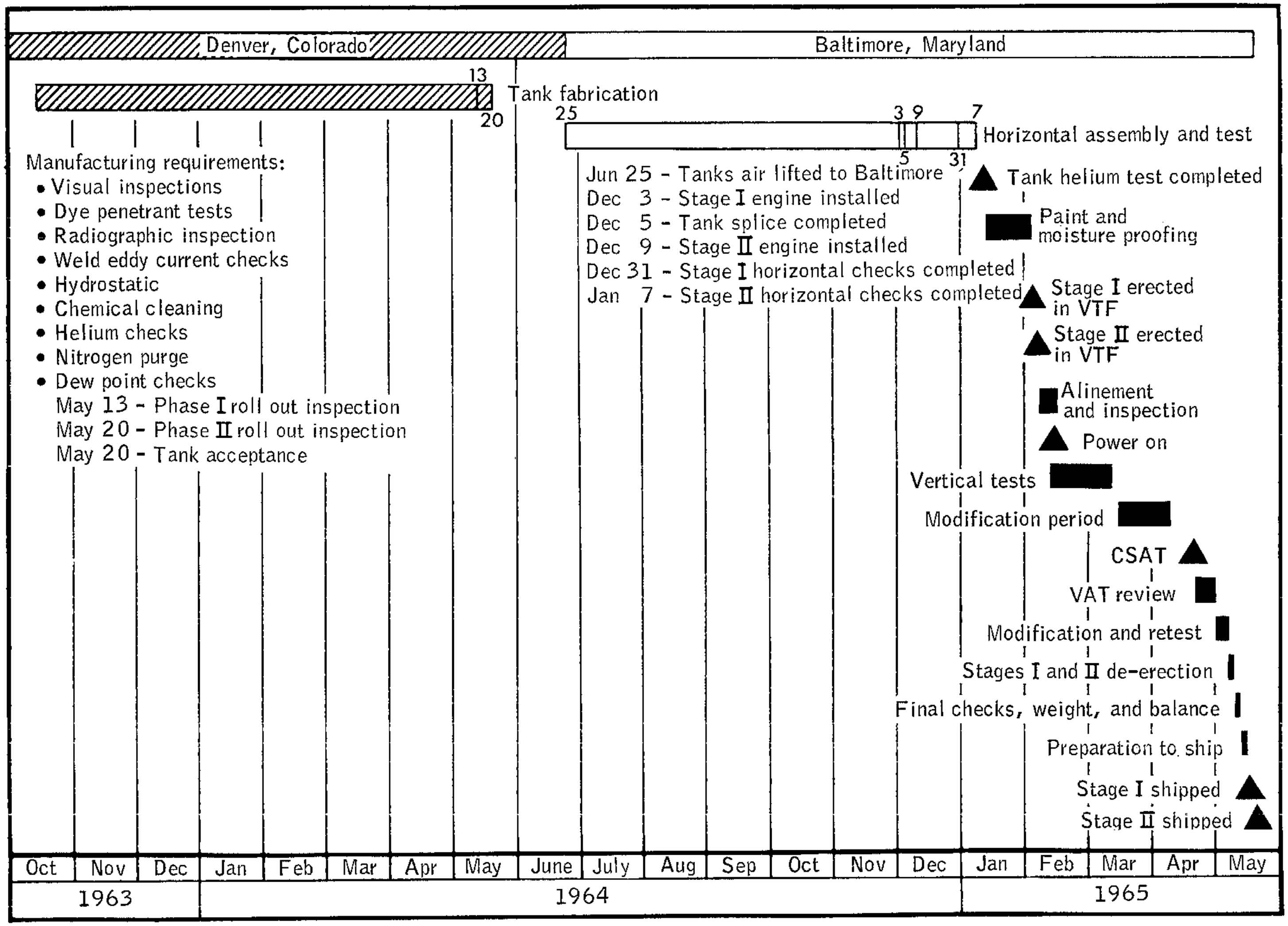

Figure 12.1-5. - GLV-5 history at Denver and Baltimore.

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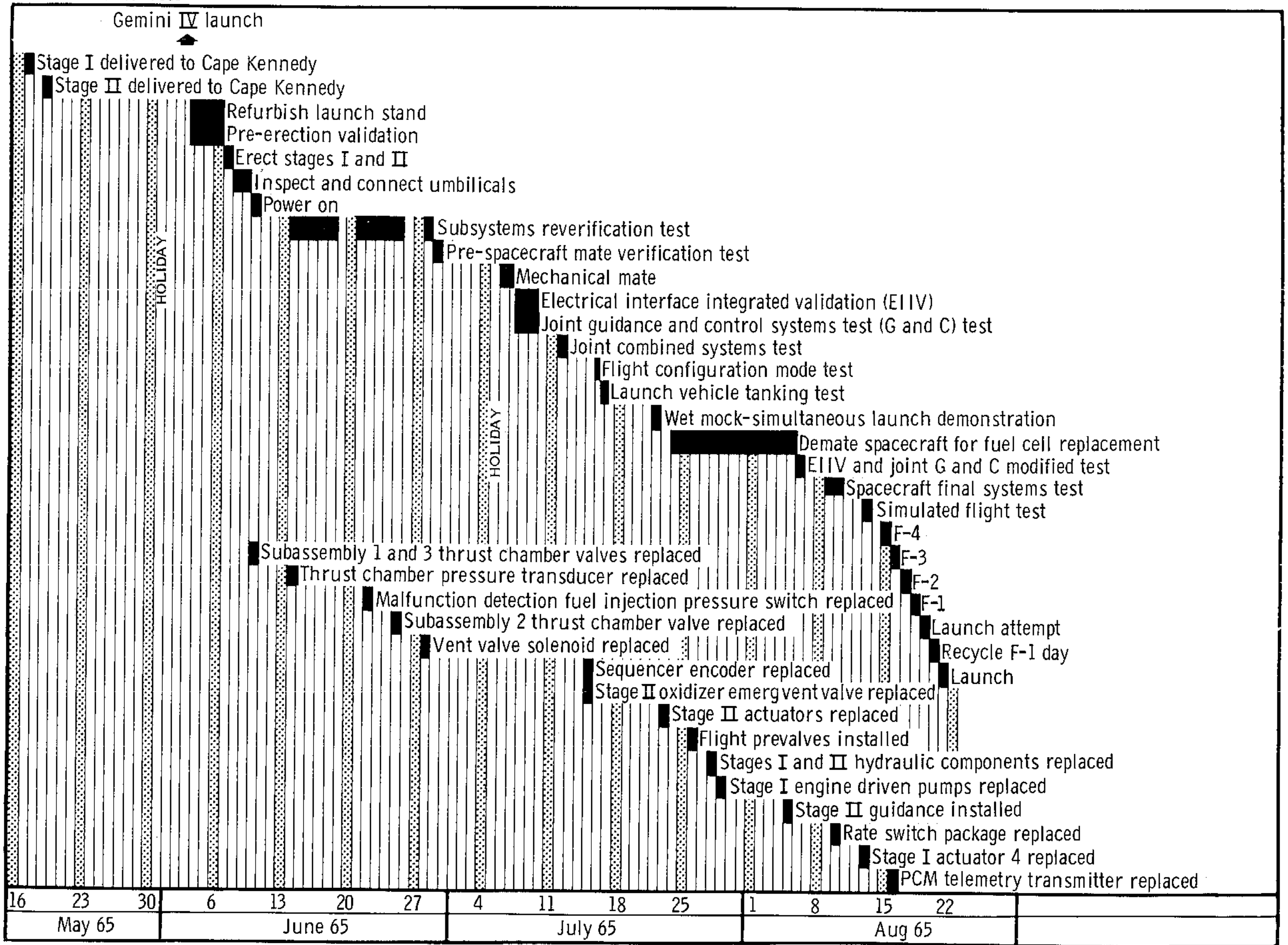

Figure 12.1-6. - GLV history at Cape Kennedy.

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12.2 WEATHER CONDITIONS

The weather conditions in the launch area at Cape Kennedy were satisfactory for all operations on the day of the launch, August 21, 1965. Surface weather observations in the launch area taken at 09:00 e.s.t. (14:00 G.m.t.) were as follows:

Cloud coverage	$\frac{2}{10}$ covered, scattered at 8000 feet
Wind direction	calm
Wind velocity	calm
Visibility, miles	10
Pressure, in. Hg	30.09
Temperature, °F	87
Dew point, °F	74
Relative humidity, percent	66

Weather observations taken at 12:55 G.m.t., August 29, 1965, onboard the U.S.S. Lake Champlain located at latitude 29°48' N., longitude 69°45' W. were as follows:

Cloud coverage	$\frac{4}{10}$ covered, cumulus at 2000 feet
Wind direction, deg	235
Wind velocity, knots	8
Visibility, miles	10
Temperature, °F	83
Dew point, °F	76
Relative humidity, percent	73
Sea temperature, °F	86
Sea state	3 ft waves

Table 12.2-I presents the launch area atmospheric conditions at the time of lift-off (14:00 G.m.t.). Table 12.2-II provides weather data in the vicinity of Cape Kennedy at the time of reentry. Figures 12.2-1 and 12.2-2 present the launch area and reentry area wind direction and velocity plotted against altitude.

TABLE 12.2-I.- LAUNCH AREA ATMOSPHERIC CONDITIONS

AT 14:00 G.m.t., AUGUST 21, 1965

Altitude, ft[a,b]	Temperature, °F[b]	Pressure, lb/sq ft[b]	Density, slugs/cu ft[b]
0×10^3	86	2127	2247×10^{-6}
5	62	1786	1984
10	46	1490	1709
15	56	1236	1460
20	17	1019	1245
25	0	835.0	1059
30	-20	678.1	900.3
35	-42	545.5	762.7
40	-65	433.4	640.7
45	-85	339.8	529.7
50	-96	263.4	422.2
55	-89	204.3	322.1
60	-81	159.4	245.8
65	-76	124.7	189.4
70	-69	98.1	146.3
75	-67	77.3	114.9
80	-62	61.2	89.6
85	-53	48.5	69.5
90	-49	38.6	54.9
95	-51	30.9	43.8
100	-47	24.6	34.7
105	-31	19.6	26.8
110	-30	15.7	21.4

[a]See note [a] at end of table.

[b]See note [b] at end of table.

TABLE 12.2-I.- LAUNCH AREA ATMOSPHERIC CONDITIONS AT 14:00 G.m.t., AUGUST 21, 1965 – Continued

Altitude, ft[a,b]	Temperature, °F[b]	Pressure, lb/sq ft[b]	Density, slugs/cu ft[b]
115×10^3	-29	12.7	17.2×10^{-6}
120	-19	10.3	13.6
125	-21	8.3	11.1
130	-12	6.8	8.8
135	2	5.5	7.0
140	26	4.5	5.5
145	26	3.8	4.5
150	23	3.0	3.7
155	18	2.6	3.1
160	16	2.1	2.6
165	16	1.7	2.1
170	19	1.4	1.8
175	15	1.2	1.5
180	15	1.0	1.2
185	8	0.8	1.0
190	4	.7	0.8
195	0	.5	.7
200	-6	.4	.6
205	-11	.4	.5
210	-18	.3	.4
215	-25	.2	.3
220	-33	.2	.3
225	-42	.2	.2

[a]See note [a] at end of table.

[b]See note [b] at end of table.

TABLE 12.2-I.- LAUNCH AREA ATMOSPHERIC CONDITIONS AT 14:00 G.m.t., AUGUST 21, 1965 – Concluded

Altitude, ft[a,b]	Temperature, °F[b]	Pressure, lb/sq ft[b]	Density, slugs/cu ft[b]
230 × 10^3	-51	.1	.2 × 10^{-6}
235	-61	.1	.1
240	-72	.1	.1
245	-83	.1	.1
250	-95	0	.1
255	-108	0	.1
260	-121	0	0
265 through 300	-127	0	0

[a]Above 180 000 feet, the data were extrapolated to a standard atmosphere at 300 000 feet.

[b]Accuracy of readings is as follows:

Altitude, ft	Temperature error, °F	Pressure rms error, percent	Density rms error, percent
0 to 60 × 10^3	1	1	0.5
60 to 120	1	1	.8
120 to 165	4	1.5	1.0
165 to 200	6	1.5	1.5
200 to 300	9	1.5	2.5

TABLE 12.2-II.- REENTRY ATMOSPHERIC CONDITIONS AS MEASURED AT CAPE KENNEDY AT 12:56 G.m.t., AUGUST 29, 1965

Altitude, ft[a,b]	Temperature, °F[b]	Pressure, lb/sq ft[b]	Density, slugs/cu ft[b]
0×10^3	53	2125	2337×10^{-6}
5	64	1784	1974
10	48	1490	1701
15	30	1235	1468
20	16	1018	1247
25	-3	833.7	1062
30	-21	677.1	900.1
35	-45	544.0	764.7
40	-67	431.9	640.7
45	-81	339.2	522.1
50	-89	264.4	415.0
55	-87	205.7	321.7
60	-83	160.2	248.0
65	-75	125.5	190.1
70	-68	99.0	147.0
75	-63	78.1	114.7
80	-64	61.8	91.0
85	-61	48.9	71.4
90	-56	38.9	55.9
95	-47	30.1	43.5
100	-40	24.6	34.3

[a]See note a at end of table.

[b]See note b at end of table.

TABLE 12.2-II.- REENTRY ATMOSPHERIC CONDITIONS AS MEASURED AT CAPE KENNEDY AT 12:56 G.m.t., AUGUST 29, 1965 — Continued

Altitude, ft[a,b]	Temperature, °F[b]	Pressure, lb/sq ft[b]	Density, slugs/cu ft[b]
105×10^3	-40.2	19.8	27.4×10^{-6}
110	-38.0	18.0	24.7
115	-33.5	14.6	19.6
120	-25.6	11.8	15.6
125	-18.3	9.5	12.7
130	-20.7	7.8	10.0
135	-8.9	6.3	8.2
140	-9.7	5.2	6.5
145	2.0	4.2	5.2
150	15.6	3.5	4.3
155	20.0	2.9	3.4
160	16.9	2.4	2.9
165	21.5	2.0	2.4
170	17.2	1.6	2.0
175	24.4	1.3	1.6
180	22.7	1.1	1.3
185	24.5	0.9	1.1
190	25.7	.8	1.0
195	22.9	.6	.8
200	22.5	.5	.6

[a]See note a at end of table.

[b]See note b at end of table.

TABLE 12.2-II.- REENTRY ATMOSPHERIC CONDITIONS AS MEASURED AT CAPE KENNEDY AT 12:56 G.m.t., AUGUST 29, 1965 - Concluded

Altitude, ft[a,b]	Temperature, °F[b]	Pressure, lb/sq ft[b]	Density, slugs/cu ft[b]
205×10^3	1.1	0.4	0.5×10^{-6}
210	-9.7	.4	.5
215	-13.0	.3	.4
220	-20.2	.2	.3
225	-28.3	.2	.3
230	-37.0	.2	.2
235	-46.5	.1	.2
240	-56.7	.1	.1
245	-67.5	.1	.1
250	-79.1	.1	.1
255	-91.4	0	.1
260	-104.4	0	.1
265 through 300	-109.9	0	0

[a]Above 180 000 feet, the data were extrapolated to a standard atmosphere at 300 000 feet.

[b]Accuracy of readings is as follows:

Altitude, ft	Temperature error, °F	Pressure rms error, percent	Density rms error, percent
0 to 60×10^3	1	1	0.5
60 to 120	1	1	.8
120 to 165	4	1.5	1.0
165 to 200	6	1.5	1.5
200 to 300	9	1.5	2.5

NASA-S-65-8544

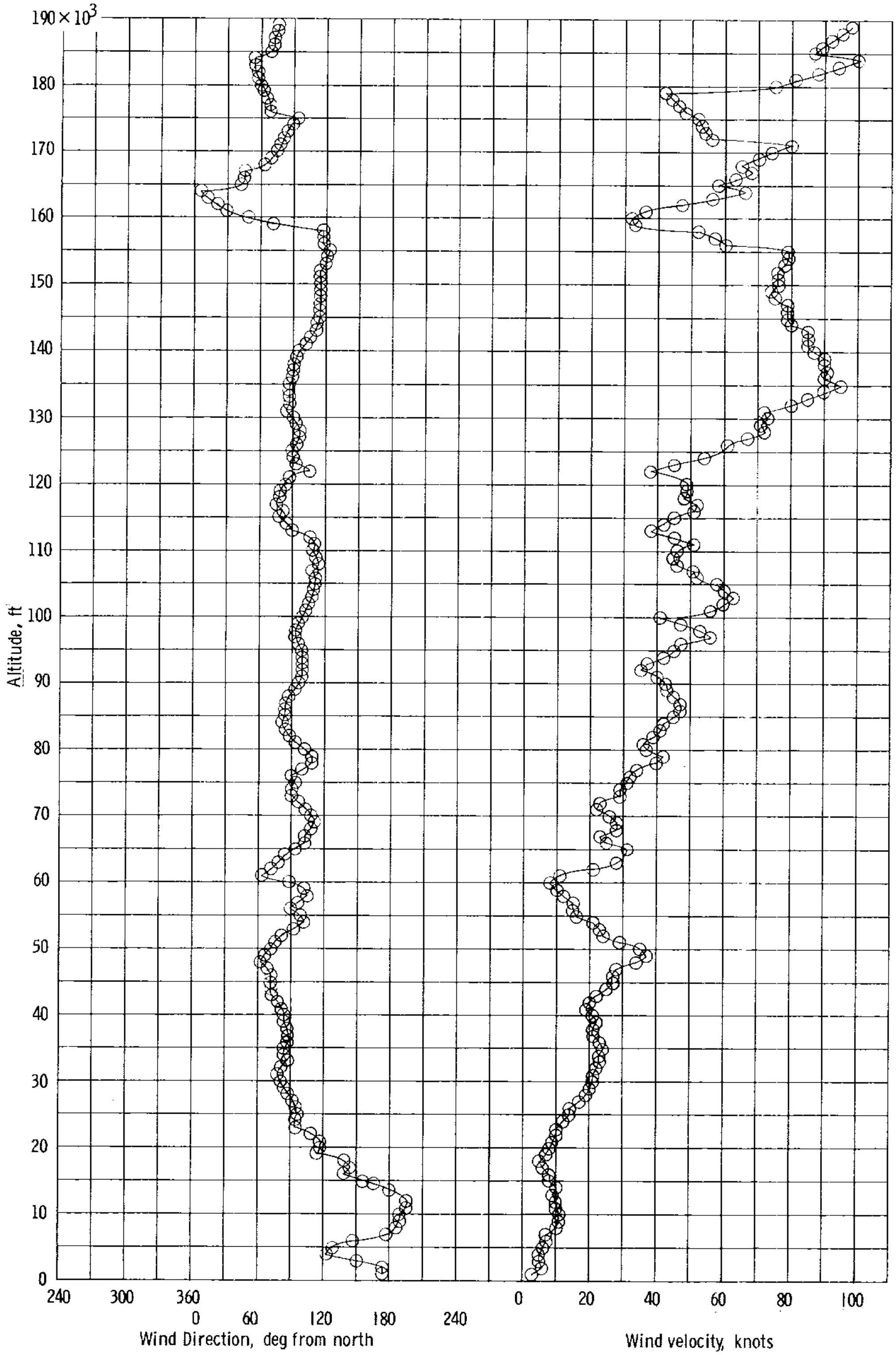

Figure 12.2-1. - Variation of wind direction and velocity with altitude for the launch area at 14:00 G.m.t., August 21, 1965.

NASA-S-65-8542

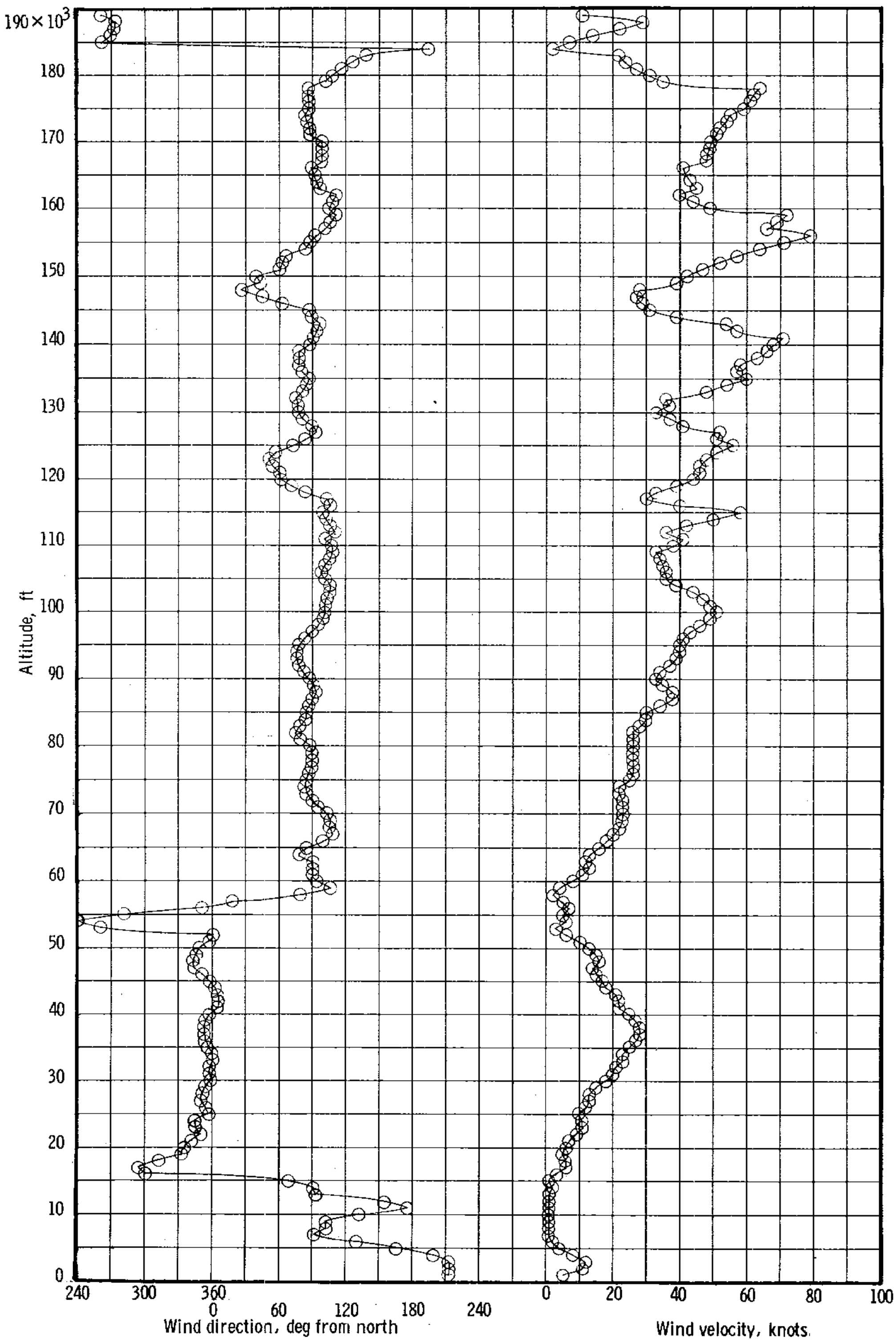

Figure 12.2-2. - Variation of wind direction and velocity with altitude for the reentry area at 12:56 G. m. t., August 29, 1965.

12.3 FLIGHT SAFETY REVIEWS

The flight readiness of the spacecraft and launch vehicle for the Gemini V mission, as well as the readiness of all supporting elements, was determined at the Flight Safety and Mission Review meetings noted in the following paragraphs.

12.3.1 Flight Readiness Review

12.3.1.1 Spacecraft.- The Flight Readiness Review for spacecraft 5 was held July 29, 1965. All systems were found ready for flight except for the following:

(a) Computer required rework to incorporate manual power-sequencing capability.

(b) REP batteries were to be replaced when spacecraft was demated.

(c) Voice control center was to be replaced with new unit containing a separate on-off voice tape recorder switch and a silence (sleep) switch.

(d) Fuel-cell sections were to be replaced and new product water valves installed.

(e) Ejection seats were to be removed so that new hose clamps could be installed in the environmental control system.

(f) A screen was to be installed across the sensor head of the MSC-1 experiment.

(g) Flight blood-pressure reprogramer adapters had not been received, installed, or tested in the spacecraft.

After these items were accomplished and the remainder of the testing satisfactorily completed, the spacecraft was found ready for flight.

12.3.1.2 Launch vehicle.- A technical review of GLV-5 was held in Los Angeles, California, on July 21, 1965. The Air Force Space Systems Division, assisted by personnel from the Aerospace Corporation, presented the status of the launch vehicle to members of the Flight Safety Review Board. Open items to be resolved prior to flight included the following:

(a) Coordination of timing of the remote tuning and opening of prevalves with the Range Safety organization.

(b) Review of erector checkout.

(c) Review of gearbox flushing tests.

(d) Review of accumulator reservoir replacement and resultant hydraulic system testing.

(e) Reports on results of command receiver investigation and failure analysis of the three-axes reference system (TARS) timer.

(f) Review of stage I actuator-switch problem.

(g) Review of rate gyro high-torquing-gain out-of-specification conditions.

On August 16, 1965, a Readiness Review was held. All open items from the Technical Review were discussed and resolved satisfactorily. All ground and airborne systems were found ready for flight.

12.3.2 Mission Review

The Mission Review Board was convened on August 17, 1965. All elements reviewed their status and were found in readiness to support the launch and mission.

12.3.3 Flight Safety Review Board

The Air Force Flight Safety Review Board met on August 18, 1965, and recommended that the launch vehicle be committed to flight.

12.4 SUPPLEMENTAL REPORTS

Supplemental reports for the Gemini V mission are listed in table 12.4-I. The format will conform to the external distribution format of the NASA or external organization preparing the report. Each report will be identified on the title page as being a Gemini V supplemental report. Before publication, the supplemental reports will be reviewed by the cognizant Mission Evaluation Team (MET) Senior Editor, the Chief Editor, and the MET Manager, and will be approved by the Gemini Program Manager.

The same distribution will be made on the supplemental reports as that made on the Mission Report.

TABLE 12.4-I.- SUPPLEMENTAL REPORTS

Number	Report title	Responsible organization	Completion date	Text reference section and remarks
1	Launch Vehicle Flight Evaluation Report - NASA Mission GT-5	Aerospace Corporation	October 21, 1965	Section 5.2 standing requirement
2	Launch Vehicle No. 5 Flight Evaluation	Martin Company	October 5, 1965	Section 5.2 standing requirement
3	Manned Space Flight Network Performance for the Gemini V Mission	Goddard Space Flight Center	October 21, 1965	Section 6.3 standing requirement
4	Gemini V Spacecraft Inertial Guidance System Evaluation	TRW Systems	October 5, 1965	Section 5.1.5 standing requirement
5	GLV-5 Oxidizer Tank Inspection and Analysis	Martin Company	November 1, 1965	Section 5.2
6	Gemini V Spacecraft Guidance System Evaluation	International Business Machines Corporation	October 5, 1965	Section 5.1.5 standing requirement
7	Part I - Special High Frequency (HF) Voice Communications Test	Engineering and Development Directorate - MSC	November 1, 1965	Section 5.1.2
	Part II - Special Ultra-High Frequency (UHF) Voice and Telemetry Test	Engineering and Development Directorate - MSC	October 12, 1965	Section 5.1.2
8	Postflight Orbit and Reentry Trajectory Reconstruction	TRW Systems	October 5, 1965	Section 4.1 standing requirement

UNCLASSIFIED

12.5 DATA AVAILABILITY

Tables 12.5-I, 12.5-II, and 12.5-III list the mission data which are available for evaluation. The trajectory and telemetry data will be on file at the Manned Spacecraft Center (MSC), Computation and Analysis Division, Central Metric Data File. The photographic data will be on file at the MSC Photographic Technology Laboratory.

TABLE 12.5-I.- INSTRUMENTATION DATA AVAILABILITY

Data description

Paper recordings

- Spacecraft telemetry measurements
  - (Revolutions 1, 2, 3, 4, 14, 15, 16, 17, 18, 29, 30, 31, 32, 33, 43, 44, 45, 46, 47, 48, 58, 59, 60, 61, 62, 73, 74, 75, 76, 77, 87, 88, 89, 90, 91, 92, 102, 103, 104, 105, 106, 107, 111, 117, 118, 119, 120, reentry)
- Experiment D-4/D-7 parameters for selected times
  - (Revolutions 1, 14, 16, 17, 30, 31, 32, 33, 45, 47, 51, 61, and 103)
- GLV telemetry measurements (launch)
- Telemetry signal-strength recordings
- MCC-H plotboards (Confidential)
- Range safety plotboards (Confidential)

Magnetic tapes

- Experiment D-4/D-7 parameters
  - (Revolutions 1, 2, 8, 14, 16, 17, 30, 31, 32, 33, 45, 47, 51, 61, 62, 103, 105, and onboard tape)

Radar data (Confidential)

- IP-3600 trajectory data
- MISTRAM
  - Natural coordinate system
  - Final reduced
- C-band
  - Natural coordinate system
  - Final reduced
- Trajectory data processed at MSC and GSFC (launch and orbital)

Voice transcripts (Confidential)

- Air-to-ground and onboard recorder
- Technical debriefing

GLV reduced telemetry data (Confidential)

- Engineering units versus time plots

Spacecraft reduced telemetry data

- Engineering units versus time
  - Ascent phase
    - Plots and tabulations of all system parameters including G and C
  - Orbital phase
    - Time histories of all system parameters excluding high sample rate parameters for revolutions 1 and 2
    - Parameter tabulations (statistical) for revolutions 1, 2, 3, 4, 5, 7, 8, 9, 14, 15, 16, 17, 18, 19, 20, 21, 22, 30, 32, 33, 34, 37, 47, 75, 115, 116, 117, 118, 119, and 120.
    - Parameter plots (statistical) for revolutions 1, 2, 3, 4, 7, 14, 15, 17, 21, 32, 34, 37, 47, 75, and 120
    - Time history tabulations of selected parameters for selected times for revolutions 1, 2, 3, 7, 14, 17, 19, 32, 47, 54, 55, 59, 75, 81, and 120
    - Plots of selected parameters for revolutions 1, 2, 3, 4, 7, 14, 17, 19, 32, 34, 37, 47, 75, and 120
    - Tabulations of selected G and C parameters for selected times for revolutions 1, 2, 14, 17, 32, 106, and 111
    - Plots and tabulations of radar system parameters for selected times for revolutions 2, 17-18, 47-48, 74-75, and 106-107

TABLE 12.5-I.- INSTRUMENTATION DATA AVAILABILITY - Concluded

Data description	
Reentry phase Plots and tabulations of all systems parameters Event tabulations Sequence of event tabulations versus time (including thruster firings) for ascent, reentry, and revolutions 1, 2, 3, 4, 5, 7, 8, 14, 15, 16, 17, 18, 19, 20, 21, 22, 30, 32, 33, 34, 47, 54, 55, 59, 73, 74, 75, 76, 79, and 81 Special computations Ascent phase IGS computer word flow tag correction (Confidential) Special aerodynamic and guidance parameter calculations (Confidential) Steering derivation calculation (Confidential) Angle of attack computation (Confidential) MISTRAM versus IGS velocity comparison (Confidential) Mod III radar versus IGS velocity comparison (Confidential)	Ampere-hour calculations for revolutions 1, 2, 3, 4, 5, 6, 7, 8, 9, 11, 14, 15, 16, 17, 18, 20, 21, 22, 115, 116, 117, 118, 119, and 120 Orbital phase OAMS propellant-remaining computations for revolutions 1, 73, 74, 75, 76, and 79 OAMS thruster activity computations for revolutions 1, 75, and 79 Sunrise and sunset time calculation Experiment MSC-1 tabulations for revolutions 1, 2, 3, 5, 8, 21, 32, 33, 34, and 115 Onboard radar coordinate transformations for selected times for revolutions 2, 17-18, 47-48, 74-75, and 106-107 Reentry phase Lift-to-drag ratio and angle of attack Calculations RCS propellant remaining and thruster activity computations

TABLE 12.5-II.- SUMMARY OF PHOTOGRAPHIC DATA AVAILABILITY

Mission phase	Number of still photographs	Motion picture film, footage
Launch and prelaunch	55	8068[a]
Recovery		900
Swimmer deployment and installation of collar	81	
Egress of flight crew	115	
Aircraft carrier		1075
Loading of spacecraft and arrival of flight crew	157	
Inspection of spacecraft	85	
Mayport, Florida		450
General activities	20	
RCS deactivation	15	
Cape Kennedy postflight inspection		
Exterior views of spacecraft	6	1500
Detail inspection views	77	
Onboard spacecraft		258
70-mm camera	244	
35-mm camera	127	
Experiment S-1, Zodiacal Lights	10	
Experiment S-5, Synoptic Terrain Photography	172	
Experiment S-6, Synoptic Weather Photography	175	
Experiment S-7, Cloud Top Spectrometer	35	
Apollo Landmark Experiment	35	

[a]Engineering sequential film only.

TABLE 12.5-III.- LAUNCH PHASE ENGINEERING SEQUENTIAL CAMERA DATA AVAILABILITY

Sequential film coverage, item	Size, mm	Location	Presentation
1.2-9	16	50-foot tower	GLV launch
1.2-10	16	50-foot tower	GLV launch
1.2-11	16	50-foot tower	GLV launch
1.2-12	16	50-foot tower	Spacecraft launch
1.2-13	16	50-foot tower	Spacecraft launch
1.2-14	16	50-foot tower	GLV, explosive bolt action
1.2-15	16	50-foot tower	GLV, explosive bolt action
1.2-16	16	East launcher	GLV, possible fuel leakage
1.2-17	16	West launcher	GLV, possible fuel leakage
1.2-18	16	North launcher	GLV, engine observation
1.2-19	16	South launcher	GLV, engine observation
1.2-20	16	Base umbilical tower	GLV, umbilical disconnect
1.2-21	16	First level umbilical tower	GLV, umbilical disconnect
1.2-22	16	Second level umbilical tower	GLV, umbilical disconnect
1.2-23	16	Fourth level umbilical tower	GLV, umbilical disconnect
1.2-24	16	Fifth level umbilical tower	GLV, umbilical disconnect
1.2-25	16	Sixth level umbilical tower	GLV, umbilical disconnect
1.2-26	16	Sixth level umbilical tower	Cable cutters
1.2-27	16	Sixth level umbilical tower	J-bars and lanyard observation
1.2-28	16	Top umbilical tower	Spacecraft umbilical disconnect
1.2-30	16	50-foot tower	Umbilical booms 3 and 4
1.2-33	16	Cape Kennedy	Tracking
1.2-35	16	Cape Kennedy	Tracking
1.2-37	35	Cape Kennedy	Tracking
1.2-38	70	False Cape	Tracking, IGOR
1.2-39	70	Cocoa Beach	Tracking, ROTI
1.2-40	70	Melbourne Beach	Tracking, ROTI
1.2-41	35	Patrick Air Force Base	Tracking, IGOR
None	35	Aircraft no. 461	Tracking

12.6 POSTFLIGHT INSPECTION

The postflight inspection of the spacecraft 5 reentry assembly was conducted in accordance with reference 11 at the John F. Kennedy Space Center (KSC) from August 31 to September 23, 1965. The R and R section, with attached drogue and pilot parachute, and the main parachute were not recovered.

Certain items of equipment were removed from the spacecraft on-board the prime recovery ship and disposed of in accordance with spacecraft test requests (STR's) 5000B through 5009.

The reentry assembly was received at KSC in good condition on August 31, 1965. The external appearance was similar to spacecraft 3 and 4.

The following is a list of the discrepancies noted during the detailed inspection of the reentry assembly:

(a) Styrofoam flotation blocks located in the fuel section of the RCS were returned as loose pieces. A small piece of the Eccofoam flotation material was missing.

(b) Three structural stiffeners located in the equipment bay forward of the environmental control system (ECS) well showed excessive corrosion.

(c) Three battery straps showed excessive corrosion.

(d) A residue was on the exterior surface of both hatch windows.

(e) Water marks, residue, and corrosion indicated sea water had possibly seeped into the ECS well.

(f) A foreign material of the consistency of cup grease was found on the exterior of RCS thruster 3B.

(g) Two aerospace ground equipment (AGE) test point connectors contained sea water.

(h) Shorts between pins were located in the wire bundle between the primary horizon sensor and the sensor electronics.

(i) A terminal board in the instrumentation system contained loose ground wires due to the nut being tight against the wire terminals.

(j) The voice tape recorder would not advance the tape in the cartridge.

12.6.1 Spacecraft Systems

12.6.1.1 Structure.- The overall appearance of the spacecraft structure was good. The external appearance of the reentry assembly was similar to spacecraft 3 and 4. The appearance of the heat shield was normal, and the stagnation point was located 12.6 inches below the horizontal center line and 2.3 inches to the right of the vertical center line. The heat shield was removed and dried. The dry weight of the heat shield was 319.90 pounds without the insulation blankets. Two plugs were removed from the heat shield for analysis in accordance with STR 5033.

Three of the battery straps and three stiffeners in the equipment bay forward of the ECS well showed excessive corrosion. These items will be analyzed in accordance with STR 5507.

Water marks, residue, and corrosion on the ECS door indicated sea water in the ECS well. STR 5510 was written to investigate the ECS door seal and relief valve for possible leakage, and none could be identified.

Residue similar to that on the windows of spacecraft 3 and 4 was noted, and STR 5011 was written to determine the constituents.

12.6.1.2 Environmental control system.- Drinking water samples were taken and prepared for analysis in accordance with STR 5508. The total water removed was 13.62 pounds.

The secondary oxygen system was removed and disposed of in accordance with STR 5013.

The lithium hydroxide cartridge was removed from the ECS package and weighed. The cartridge weighed 122.53 pounds with a center-of-gravity 8.18 inches from the bottom of the cartridge. The carbon dioxide sensor was removed from the ECS package and sent to the vendor for testing in accordance with STR 5036. The sensor was found to be in good working order and within specified calibration units.

12.6.1.3 Communications system.- The external appearance of all the communications equipment located in the equipment bays was good, and little evidence of corrosion was exhibited.

The HF whip antenna was extended and retracted using an external power source applied at the battery connector and using spacecraft circuitry and switches in accordance with STR 5021A. No anomalies were found.

The check of the operation of the voice tape recorder was completed in accordance with STR 5026. When a tape cartridge was inserted in the recorder, the motor would run, but it would not advance the tape in the cartridge. Confirmation of the proper operation of the end-of-tape indicating light was made. The recorder was removed for shipment to the vendor.

12.6.1.4 Guidance and control system.- The inertial measurement unit (IMU), attitude control maneuver electronics (ACME), computer, auxiliary computer power unit (ACPU), and horizon sensor electronics were removed onboard the prime recovery ship, and were cleaned and packaged in accordance with reference 10. Upon receipt at Cape Kennedy with the spacecraft, these items were disposed of in accordance with STR's 5004 through 5008.

A continuity check of the wire bundle from the primary horizon sensor head to the electronics package in accordance with STR 5504 was completed. Shorts existed between pins 1, 3, 27, 28, and 29 in wire bundle 221A. No shorts existed in wire bundle 221B. These shorts were determined to be normal for the postlanding configuration.

12.6.1.5 Pyrotechnics.- Pyrotechnic resistance checks were performed on all actuated pyrotechnic cartridges.

The hatch actuator breeches, rocket catapults, and seat pyrotechnic devices were removed and sent to storage.

The pyrotechnic switch "G" cartridge (52-72724-3-123) was removed for inspection in accordance with STR 5503. Visual inspection revealed that residue from the detonation of the explosive mix, by coincidence, had caused the resistance across the bridgewire connections to be very close to the same as the preflight measurement.

The electrical connectors to the mild detonating fuse (MDF) detonators at station Z192 had the bayonet pins sheared off and were hanging loose from the cartridges. This condition has been noted on previous missions and is considered acceptable. Both detonators had high detonation.

The ejection seat drogue and ballute aneroids were removed onboard the prime recovery ship and returned to Cape Kennedy for testing in accordance with STR 5002. All units were found to be in tolerance.

The postflight visual inspection of the wire bundle guillotines, parachute bridle release mechanisms, detonators, and other pyrotechnics disclosed that all appeared to have functioned normally.

12.6.1.6 Instrumentation and recording system.- The PCM tape recorder was removed onboard the primary recovery ship and prepared for failure analysis in accordance with STR 5019.

The programer, instrumentation package no. 2, high-level multiplexer, and low-level multiplexer were removed, cleaned, and packaged onboard the prime recovery ship in accordance with reference 12. Upon receipt at Cape Kennedy with the spacecraft, these items were disposed of in accordance with STR's 5001A and 5003.

A continuity check of the spacecraft instrumentation wiring revealed that the voltage into the programer and tape recorder could be dropped by wiggling the loose ground wires on terminal board 8, pin 4. The wires were loose because the nut on the terminal board was not tight against the wire terminals. The check also revealed that the dc-dc regulated voltage outputs were not functioning properly; however, this is probably a result of the wiring and converter having been submerged in salt water.

12.6.1.7 Electrical system.- The main and squib batteries were removed and discharged in accordance with reference 12. The following table lists the ampere-hours remaining in the battery after flight when discharged to the specified level of 20 volts with the battery still delivering the currents specified in the reference.

Main	Discharge, A-hr	Squib	Discharge, A-hr
1	40.65	1	7.52
2	41.00	2	4.86
3	42.33	3	6.00
4	42.90		

The main and squib batteries were placed in storage for ground test use.

The current leakage due to salt water immersion was checked and recorded in reference 12. No anomalies were found.

A check of the circuitry to the stage II fuel pressure gage in accordance with STR 5018 was completed. No anomalies were found.

A check of the reentry assembly circuitry for the OAMS heaters in accordance with STR 5027 revealed no anomalies.

The fuse block status check in accordance with reference 12 was completed.

A check of the reentry assembly circuitry to the RSS oxygen tank heater in accordance with STR 5016 revealed no anomalies.

STR 5064 was written to investigate the cause of the high current drain during postlanding. This work was continuing as this report was being published.

12.6.1.8 Crew station furnishings and equipment.- The appearance of the cabin interior was good. The cabin absorbent material samples were removed on the prime recovery ship and disposed of in accordance with STR 5009. The flight crew equipment removed on the prime recovery ship was dispositioned in accordance with STR 5000B.

The determination of hatch closing forces in accordance with STR 5023 was completed. Three tests of each hatch utilizing the canvas strap handle gave the following loads:

Left hatch, lb	Right hatch, lb
32	32
31	32
31	32

The investigation of the failure of the right utility light in accordance with STR 5040 is being conducted. The investigation of the single utility cord and voltage regulator in accordance with STR 5022 and 5041 was conducted. A broken wire was found at the connector.

The cause of the intermittent operation of cabin temperature indicator was investigated in accordance with STR 5505 and found to be a bent pin in the connector to the temperature sensor.

A wiring check of the mass quantity indicator used to monitor fuel-cell oxygen was conducted in accordance with STR 5506 and no anomalies were found.

The opening torque of the hatches, applied at the external hatch socket, was measured as 225 in-lb required for the right hatch, and 300 in-lb for the left hatch.

Functional checks of the following mechanical linkages were performed in accordance with reference 12 and no anomalies noted.

(a) Five ECS loop handle controls

(b) Two secondary oxygen controls

(c) Three cabin and suit air flow controls

(d) Cabin and suit temperature controls

(e) Manual oxygen high rate control

12.6.1.9 Propulsion system.- The RCS thrust chamber assemblies (TCA's) appeared normal. Peripheral cracks were noted in three of the TCA's. A residue was noted in the throat of some thrusters. Neither of these are considered to be a problem and have been noted on previous flights.

The following amounts of RCS propellant were returned in separate containers with the reentry assembly from the Mayport, Florida, deactivation area and sent to Patrick Air Force Base for analysis in accordance with STR 5025.

	A-ring propellant weight, lb	B-ring propellant weight, lb
Oxidizer	0.08	4.90
Fuel	0	4.63

TCA 3B was returned with a foreign material resembling grease on the exterior of the thruster. (See fig. 12.6-9.) A sample of the material was removed and was analyzed in accordance with STR 5509. The material was found to be foreign to the spacecraft and is believed to have attached to the thruster after landing.

The char depth of TCA 6A and 7A was determined in accordance with STR 5512.

12.6.1.10 Landing system.- The single-point bridle release mechanism and the main parachute forward and aft bridle release mechanism appeared to have functioned normally.

The pilot parachute deployment circuitry and latching relay were investigated in accordance with STR 5031 and found to have functioned normally. The data were reprocessed, and the signal was received on time.

12.6.1.11 Postlanding recovery aids.- The flashing recovery light and the hoist loop doors appeared to have functioned normally but were not recovered.

Two blocks of styrofoam flotation material from the forward portion of the RCS section were returned as loose pieces. These blocks could have been knocked off during landing impact or attachment of the flotation collar by the swimmers. A small piece of the Eccofoam flotation material was missing from the lower forward part of the RCS section and could have been damaged during recovery.

Satisfactory operation of the HF whip antenna extend-retract mechanism was determined during the postflight inspection in accordance with STR 5021A.

12.6.1.12 Experiments.- Most experiment equipment was removed on the prime recovery ship and disposed of in accordance with STR 5000B.

The M-1 experiment equipment was tested in accordance with STR 5020A. A digital readout, through the equipment transducer, of the pressure remaining in the oxygen bottle revealed that a pressure of 100 psi at a temperature of 75° remained. This reading was taken on September 7, 1965, at 10:00 a.m. e.s.t.

12.6.2 Continuing Evaluation

The following is a list of the spacecraft test requests (STR's) that have been approved for the postflight evaluation of reported spacecraft anomalies and problems:

Number	System	Purpose
5001A	Instrumentation and recording	Conduct failure analysis on PCM programer. (See section 5.1.3.1.)
5002	Pyrotechnic	Evaluate ejection seat drogue and ballute aneroids. (See section 5.1.9.)
5009	Crew station	Analyze samples of spacecraft cabin absorbent material

Number	System	Purpose
5010	Pyrotechnic	Provide verified documentation of any anomalies uncovered during postflight disassembly of the Gemini ejection seat system.
5011	Structure	Determine composition and origins of residue on hatch windows. (See section 5.1.1.2.)
5015	Crew station (Aeromedical)	Permit MSC microbiology laboratory examination of passive radiation dosimeter packets before any cleaning process is effected on them. (See table 7.2-X.)
5016	Electrical	Determine cause of RSS oxygen tank heater failure. (See sections 5.1.7.2.7 and 5.1.7.3.1.)
5017	Environmental control system	Determine if lithium hydroxide entered the suit loop and cabin. (See section 5.1.4.)
5018A	Electrical	Determine cause of the stage II fuel-tank pressure gage malfunction during launch. (See section 5.1.10.2.2.)
5019	Instrumentation and recording	Analyze PCM tape recorder failure and determine cause of poor quality delayed-time data dumps during mission. (See section 5.1.3.2.)
5020A 5020B	Experiment	Determine the postflight status of the experiment M-1 equipment prior to and after removal from the spacecraft. (See section 8.5.)
5021A	Communication	Test of spacecraft 5 HF antenna. (See section 5.1.2.6.)
5022	Crew station	Determine the cause of failure of the optical sight. (See section 5.1.10.4.4.)

Number	System	Purpose
5023	Crew station	Determine the hatch closing force on a representative flight spacecraft.
5024	Crew station	Determine the cause of the difficulty in obtaining blood pressure readings during the mission. (See section 5.1.10.3.)
5025	Propulsion	Determine composition of the RCS fuel and oxidizer and examine system for evidence of particulate contamination. (See section 5.1.8.2.3.)
5026	Communications	Check the voice tape recorder to determine if this unit malfunctioned during flight. (See section 5.1.2.)
5027	Propulsion and electrical	Investigate OAMS heater circuits. (See section 5.1.8.1.2.)
5031	Electrical and landing	Determine if the pilot parachute deploy circuitry and components are in a normal condition. (See section 5.1.7.4.)
5033	Structure	Provide information of the thermal performance of the heat shield during reentry in lieu of thermal instrumentation. (See section 5.1.1.2.)
5034	Guidance and control	Confirm computer correct operation and verify status of the radar memory locations. (See section 5.1.5.4.2.)
5035	Guidance and control	Determine cause of primary horizon sensor failure during Gemini V flight. (See section 5.1.5.4.1.)
5036	Environmental control system	Evaluation of CO_2 sensor to determine any degradation of performance. (See section 5.1.4.)
5037	Crew station	Determine sun reflection characteristics of the optical sight. (See section 5.1.10.4.4.)

Number	System	Purpose
5038	Guidance and control, and electrical	Check the spacecraft wiring associated with the radar and the horizon sensor primary system. (See section 5.1.5.4.)
5039	Guidance and control	Conduct test to repeat failure effects observed on spacecraft 5 flight radar on the second, third, and fourth passes over KSC. (See section 5.1.5.4.2.)
5040	Crew station	Determine cause of failure of right utility light. (See section 5.1.10.1.4.)
5041	Crew station	Determine cause of failure of voltage regulator for the optical sight. (See section 5.1.10.4.4.)
5042	Crew station	Determine extent and manner of failure of the dry stowage bags. (See section 5.1.10.1.1.)
5044A	Guidance and control	Determine travel of FDI range needle in response to simulated computer inputs. (See section 5.1.5.2.3.)
5056	Crew station	Determine the cause and extent of the overheating of the cabin lights. (See section 5.1.10.1.4.)
5057	Crew station	Determine the cause of the discrepancies on the left and right auxiliary window shades. (See section 5.1.10.1.4.)
5058	Crew station	Determine the cause of the malfunction of the photoevent indicator. (See section 5.1.10.4.2.)
5059	Crew station	Determine the cause of the problem in operating the left-hatch lock-open release lever reported by the crew. (See section 5.1.10.2.1.)

Number	System	Purpose
5060	Crew station	Determine the cause of leakage from the dual pressurization port manifold block mounted in the blood pressure port location on the pilot's G4C space suit. (See section 5.1.10.3.2.)
5061	Crew station	Determine the cause of the post-landing malfunction of the blood pressure inflator. (See section 5.1.10.6.2.)
5063	Electrical	Determine if the urine tube heater circuitry and components are in a normal condition. (See section 5.1.7.1.)
5064	Electrical	Determine the cause of the high current drain which started 25 seconds after splashdown. (See section 5.1.7.1.)
5065	Crew station	Check operation of DCS light. (See section 5.1.10.2.2.)
5066	Structure	Investigate possible micrometeorite impact reported by crew (determined to be "oil-canning" of structure).
5501	Structure	Determine heat shield stagnation point location. (See section 5.1.1.1.)
5503	Pyrotechnic	Determine if pyro switch G-1 igniter bridge wire is blown. (See section 12.6.1.5.)
5504	Guidance and control, and electrical	Check continuity of primary spacecraft wire bundle from sensor head to electronics package. (See section 5.1.5.4.1.)
5505	Crew station	Pinpoint cause of cabin temperature indicator failure. (See section 5.1.10.2.2.)

Number	System	Purpose
5506	Crew station and electrical	Investigate remaining spacecraft wiring as possible cause of high reading noted on the mass quantity indicator when monitoring fuel-cell oxygen quantity during mission (determined to be caused by ungrounded shield).
5509	Propulsion	Determine composition of green viscous material found in a large deposit on RCS TCA 3B nozzle. (See section 5.1.8.2.3.)
5510	Structure	Determine cause of water collection in ECS well. (See section 5.1.1.3.)
5512	Propulsion	Determine char depth on RCS TCA's 6A and 7A.
5514A	Crew station	Examine urine dump solenoid valve for signs of contamination which would cause flow restriction. (See section 5.1.10.5.4.)
5517A	Environmental control system	Investigate flow of spacecraft 5 space suit and cabin temperature control valve assembly. (See section 5.1.4.)

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