[redacted] MSC-G-R-65-2

GEMINI PROGRAM MISSION REPORT

GT-3

GEMINI 3

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CLASSIFICATION CHANGE
To UNCLASSIFIED
By authority of EO11652, 6/1/72
Changed by Date APR [illegible]76

GROUP 4
DOWNGRADED
AT 3 YEAR INTERVALS;
DECLASSIFIED
AFTER 12 YEARS

[redacted]

APRIL 1965

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION ☐ MANNED SPACECRAFT CENTER

[redacted]

MSC-G-R-65-2

CHANGE SHEET

FOR

GEMINI PROGRAM MISSION REPORT GT-3

GEMINI 3

CHANGE 1

PREPARED BY: GT-3 Mission Evaluation Team

APPROVED BY: Charles W. Mathews
Charles W. Mathews
Manager, Gemini Program

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
July 1965

Page 1 of 4 pages

Make the following pen and ink changes in the Gemini Program Mission Report GT-3, MSC-G-R-65-2. Upon incorporation of the changes, insert this CHANGE sheet between the cover and the title page and write on the cover, "Change 1 inserted."

Signature of person incorporating changes

Date

1. Page viii: ✓ Change "5.1.11 Postlanding and Recovery System" to read "5.1.12 Postlanding and Recovery System." Change "5.2.1 Airframes" to read "5.2.1 Airframe".

2. Page xiii: ✓ Following "12.6.1 Spacecraft Systems", add "12.6.1.1 Structure", page "12-5".

3. Page xx: ✓ Change page numbers for figures 6-1 through 6-6, from 6-35 through 6-40 to 6-34 through 6-39, respectively.

4. Page 1-1: ✓ In first paragraph, eighth line, change "system" to read "systems".

5. Page 3-5: ✓ In paragraph 3.1.2.8 (a), change "doors" to read "door".

6. Page 3-12: ✓ In the Pyrotechnics system, line (a), change "doors" to read "door".

7. Page 3-20: In the column headed "Launch vehicle stations", change the first entry "X50.735" to read "X50.985". Delete the callout, "Spacecraft guidance computer coordinates", and the associated "X", "+", "-", and upper dimension designation.

8. Page 3-21: On upper left quarter of illustration, change "Positive toward viewer" to read "Indicated sign (+ or -) is toward viewer".

9. Page 4-2: In second line from bottom of page, change "three" to read "five".

10. Page 4-5: In eighth line from bottom of page, change "Reference 2" to read "Reference 4".

11. Page 4-7: In "Event" column, change 11th and 14th lines to read "IGS update sent (planned time is sample time)".

12. Page 5-2: In second line, change "vehicle postinspection" to read "vehicle postflight inspection". In 13th line, delete "(section 5.1.3)".

13. Page 5-6: In section 5.1.1.4, second paragraph, change the third sentence to read as follows: "The bottom part of the tab lies flat against the surface of the beryllium shingles; however, because of the shingle design, there is 0.125-inch gap between the surface of the shingle and the upper part of the tab which could be seen by the flight crew."

14. Page 5-7: In eighth line, change "2104" to read "Z104".

15. Page 5-13: In sixth line from bottom, change "00:24:46" to read "00:24:56".

16. Page 5-15: Change the last sentence of the first paragraph to be the first sentence of the second paragraph.

17. Page 5-18: In section 5.1.5.2.1, 11th line, change "duscussed" to read "discussed".

18. Page 5-19: In seventh line from bottom, change "aximuth" to read "azimuth".

19. Page 5-31: In first line, change "5.1.5.4.1" to read "5.1.5.4.2".

20. Page 5-42: In second paragraph of section 5.1.10.5.1, eighth line, change "tover" to read "towel".

21. Page 5-49: In "Aircraft 629" entry, second column, change "05:07:45.2" to read "05:07:01.2", and in third column change "11:01.2" to read "10:17.2".

22. Page 5-50: In heading for fifth column, change "hr:min:sec" to read "min:sec". In "First orbit" entry, first line, third column, change "01:34:03.20" to read "01:34:02.20", and in fourth column, change "01:32:59.55" to read "01:32:35.9". In "Second orbit" entry, first line, second column, change "00:42:23.75" to read "01:42:23.75".

23. Page 5-51: In heading of first major column, change "Ground elapsed time, min:sec" to read "Ground elapsed time, sec".

24. Page 5-52: In first column, first entry, change "361 sec" to read "358.4 sec". In second column, first entry, change "347.52 sec" to read "357.52 sec".

25. Page 5-53: In first column, fourth entry, change "02:17:00" to read "02:20:00". In last column, third entry, first and second lines, change "pertubations" to read "perturbations".

26. Page 5-54: In first column, first entry, change "04:22:00" to read "04:21:45" and in fourth entry, change "04:47:06" to read "04:47:10". In second column, first entry, change "04:21:22" to read "04:21:23".

27. Page 5-66: In figure 5.1.1, extend leader for "PDO4" to the lower curve.

28. Page 5-87: In third line of first paragraph, delete comma between "thrust" and "which".

29. Page 6-14: In sixth line, change "committment" to read "commitment".

30. Page 7-4: In paragraph 7.1.1.1 (b), Translational Systems Check (out-of-plane), revise fourth paragraph to read as follows: "Following the 10 fps translation, two 1 fps forward translations were accomplished, each followed by a 1 fps aft translation. A yaw attitude change was initiated at 02:19:18 g.e.t. in order to return the spacecraft to the SEF attitude".

31. Page 7-37: In "Action" column opposite elapsed time of 00:52, change "T_r" to read "T_R".

32. Page 7-40: In "Action" column opposite elapsed time of 04:29, change "T_r-5" to read "T_R-5". Opposite elapsed time of 04:32, change "R_r-1" to read "T_R-1".

33. Page 8-6: In Section 8.3.3, first paragraph, last line, change "RE-ANT ANT EXP." to read "RE-ENT ANT EXP".

34. Page 12-15: Add the following supplemental reports to TABLE 12-III:

Number	Report title	Responsible organization	Completion date	Text reference section and/or remarks
10	Evaluation of spacecraft on-board computer	International Business Machines Corporation	August 15, 1965	Section 5.1.5
11	Evaluation of HF communications system	Mission Evaluation Team	June 15, 1965	Section 5.1.

35. Page 12-17: In note (a), change "Table 12-III" to read "Table 12-VI".

MSC-G-R-65-2

GEMINI PROGRAM MISSION REPORT GT-3

Gemini 3

Prepared by: Gemini Mission Evaluation Team

Approved by:

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Authorized for Distribution:

George M. Low
George M. Low
Deputy Director

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

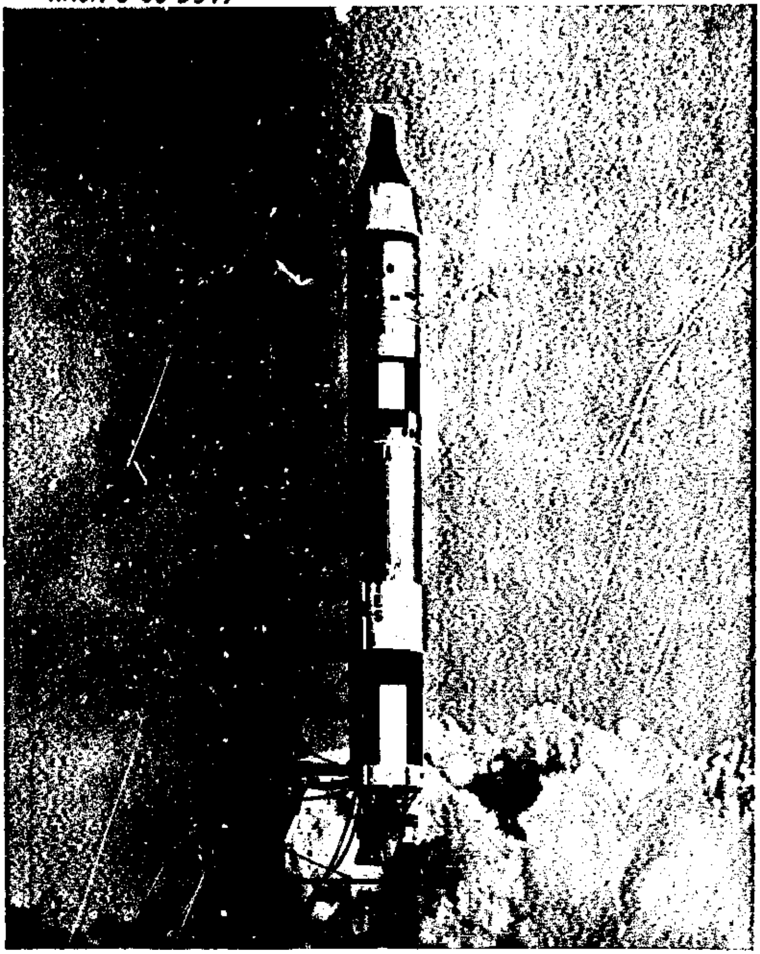
Manned Spacecraft Center

Houston, Texas

April 1965

[illegible] page C-13

NASA-S-65-3549



GT-3 space vehicle at lift-off

MAJOR SECTION LOCATOR

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FIGURES

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1.0 MISSION SUMMARY

The first manned mission of the Gemini Program (GT-3) was launched from Complex 19 at Cape Kennedy, Florida, at 9:24 a.m. e.s.t. on March 23, 1965. The mission was successfully concluded on the same day with recovery of the spacecraft by the prime recovery ship, the aircraft carrier U.S.S. Intrepid, at 22°26' N latitude, 70°51' W longitude at 5:03 p.m. e.s.t. (22:03 G.m.t.). This manned flight was accomplished after two unmanned missions – one which qualified spacecraft structure and the launch vehicle and its systems and one suborbital high-velocity reentry flight which qualified the spacecraft and systems for manned orbital flight and reentry heating. The spacecraft was manned by Astronaut Virgil I. Grissom, command pilot, and Astronaut John W. Young, pilot. The flight crew completed the three-orbit flight in excellent physical condition, having demonstrated full control of the spacecraft and competent management of the onboard systems.

The major objectives of the GT-3 mission were to demonstrate precise orbital maneuvering and evaluate manned orbital flight in the Gemini spacecraft, to qualify the spacecraft and spacecraft systems for long-duration manned missions, to demonstrate the maneuvering capability of the spacecraft orbital attitude and maneuver system, and to demonstrate the capability to control the spacecraft reentry flight path and ultimate landing point. In addition, it was desired to demonstrate satisfactory performance of the worldwide network, adequate prelaunch and launch procedures for the manned Gemini spacecraft, recovery systems and procedures, and the execution of three experiments in space.

All primary and secondary mission objectives were achieved with three exceptions. First, the accuracy of the controlled landing point was not as high as expected, primarily because during reentry the angle of attack was approximately 30 percent lower than expected. Second, the desired goals of the sea urchin egg experiment were not achieved as a result of a mechanical failure of the experimental apparatus. Finally, all desired inflight photographic coverage was not obtained because of an improper preflight lens setting on the 16-mm camera.

Some of the more important results of the flight, from the standpoint of future Gemini Program operations, were the successful demonstration of the operation of the orbital attitude and maneuver system, the successful operation of the spacecraft guidance and control system during orbit and reentry, and the evaluation of the spacecraft from the standpoint of manned operations in zero g. The flight crew found the design satisfactory for extended manned orbital operation with only minor revisions in equipment and procedure.

The launch vehicle performed satisfactorily in all respects. The flight to orbital insertion was well within allowable dispersions. All launch-vehicle systems operated satisfactorily in flight. A slightly high first-stage trajectory was realized because of a greater than predicted first-stage thrust. This condition was similar to the GT-2 flight, and new trajectory constants are being implemented for future Gemini missions.

All mission control, flight and network operations, and recovery operations were satisfactory. The GT-3 mission served to prepare the worldwide network and the operations personnel for missions of longer duration.

2.0 INTRODUCTION

The first-order mission objectives for the GT-3 mission were as follows:

(a) Demonstrate manned orbital flight in the Gemini spacecraft and further qualify the spacecraft and launch vehicle systems for future manned missions.

(b) Evaluate the two-man Gemini design and its effects on flight crew performance.

(c) Demonstrate and evaluate the operation of the worldwide tracking network with the spacecraft and flight crew.

(d) Demonstrate and evaluate the capability to maneuver the spacecraft in orbit using the orbital attitude and maneuver system (OAMS).

(e) Demonstrate OAMS capability to perform retro backup.

(f) Demonstrate the capability to control the reentry flight path and the ultimate landing point.

(g) Evaluate the performance of the following spacecraft systems:

  (1) Flight crew station controls and displays.

  (2) Environmental control system.

  (3) Gemini space suits.

  (4) Guidance and control system.

  (5) Electrical power and sequential systems.

  (6) Propulsion systems.

  (7) Communications and tracking systems.

  (8) Pyrotechnic systems.

  (9) Instrumentation systems.

  (10) Food, water, and waste management systems.

  (11) Landing and recovery systems,

(h) Demonstrate systems checkout, prelaunch, and launch procedures for a manned spacecraft with a two-man flight crew.

(i) Recover the spacecraft and evaluate the recovery system.

The second-order mission objectives for the GT-3 mission were as follows:

(a) Evaluate the following spacecraft systems:

(1) Flight crew equipment.

(2) Biomedical instrumentation system.

(3) Personal hygiene system (partial).

(b) Execute the following experiments:

(1) S-2, Effects of zero gravity on the growth of sea urchin eggs.

(2) S-4, Synergistic effect of zero gravity and radiation on white blood cells.

(3) T-1, Reentry communications.

(c) Evaluate the effects of the low-level longitudinal oscillations (POGO) of the launch vehicle on the flight crew.

(d) Obtain general photographic coverage in orbit.

All spacecraft and Gemini launch vehicle transmitted telemetry data, spacecraft onboard data, biomedical instrumentation data, ground-based radar data, and engineering photographic data obtained during the mission were used by the Mission Evaluation Team in evaluating the mission. The evaluation consisted of analyzing the flight data and comparing these data with data from all phases of ground tests. The results of these analyses are presented in this report.

More detailed analyses of these data are continuing as this report is being published. These analyses for the launch vehicle are overall performance, and performance of the radio guidance system.

Analyses of spacecraft performance are continuing in the areas of the guidance and control system, reentry aerodynamics, HF communications system, handling and stowage of cabin loose equipment, and cabin lighting.

A complete transcription of the air-to-ground voice and onboard tape recorder and the systems debriefing will be issued as supplemental reports. A list of all supplemental reports, including the responsible organizations, is contained in section 12.4 of this report.

The cooperation and contributions of the Space Systems Division of the Air Force in the preparation of the sections of this report concerning the performance of the launch vehicle are acknowledged.

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3.0 VEHICLE DESCRIPTION

The space vehicle for the GT-3 mission consisted of Gemini spacecraft 3 and Gemini launch vehicle 3 (GLV-3). Section 3.1 of this report describes the spacecraft configuration and section 3.2 describes the GLV configuration. Section 3.3 provides weight and balance data for GT-3. The major reference coordinates for the space vehicle are shown in figure 3-1.

3.1 GEMINI SPACECRAFT DESCRIPTION

This section describes the structure and major systems of spacecraft 3. Since spacecraft 2 contained production units of virtually all equipment to be used on manned missions, only the differences between spacecraft 3 and spacecraft 2 are considered. A description of spacecraft 2 is given in reference 1 and a detailed description of spacecraft 3 is given in reference 2. Table 3-I summarizes the differences between spacecraft 2 and 3.

3.1.1 Spacecraft Structure

The Gemini spacecraft lift-off configuration, shown in figure 3-2, was of the same basic design as the spacecraft flight-tested on the first two Gemini missions. Although the reentry assembly and the adapter assembly contained equipment that was slightly different from spacecraft 2, the basic structure, including the heat-shield ablative material (on spacecraft 2, the ablative material was half the normal thickness), was in accordance with the production design. The equipment differences are described in section 3.1.2 of this report.

3.1.2 Major Systems

3.1.2.1 Communications.- The communication equipment installed in spacecraft 3 and shown in figure 3-3 was similar to that installed in spacecraft 2. The major changes resulted from a flight crew being onboard for manual control of the equipment.

3.1.2.1.1 Voice communications: Two identical ultra-high-frequency (UHF) transmitter-receiver units, and a single high-frequency (HF) transmitter-receiver unit were installed in the reentry assembly equipment bay. A 1000-cps continuous-wave (CW) tone generated in the voice control center was transmitted through the HF transmitter-receiver during recovery operations. As a result of a constraint (a potential heating problem during reentry, if the antenna does not retract) on the extension of the HF antenna during flight, the HF transmitter-receiver was not used for voice communication except after landing; however, it was available as a standby unit if both UHF transmitter-receivers had failed.

A lightweight headset (microphone and single earphone) was available for use by a flight crew member when the space-suit helmet was removed. When not in use, the headset was stored in the right-hand food box. A tape recorder was supplied for recording the flight crew's voice transmissions and intercommunications.

3.1.2.1.2 Telemetry transmitter: In addition to the real-time telemetry transmitter located in the reentry assembly and the delayed-time telemetry transmitter located in the adapter assembly, a standby transmitter was installed in the reentry assembly equipment bay of spacecraft 3. Normally, when the spacecraft is within range of a ground station, the real-time transmitter is available for transmitting real-time data, the delayed-time transmitter is available for transmitting data stored on the onboard recorder, and the standby transmitter can be selected to transmit either real-time or delayed-time data, but not both, in case either of the primary transmitters fails. On spacecraft 3, the delayed-time transmitter failed during prelaunch activities and was not replaced. Consequently, the standby transmitter was used for all delayed-time transmissions for subsequent prelaunch activities and also during the flight.

3.1.2.1.3 Tracking subsystem and recovery subsystem: The transponders and beacons installed in the tracking subsystem and recovery subsystem were of the same configuration as those installed on spacecraft 2.

3.1.2.1.4 Command subsystem: The relays in the digital command system (DCS) were set up to receive real-time commands for the following functions:

(a) Select standby telemetry transmitter for real-time transmission.

(b) Select standby telemetry transmitter for delayed-time transmission.

(c) Select real-time telemetry and acquisition-aid beacon transmission.

(d) Select real-time and delayed-time telemetry transmission.

(e) Actuate C-band radar transponder.

(f) Actuate S-band radar transponder.

(g) Actuate abort indicators.

(h) Actuate playback tape recorder.

(i) Initiate calibration voltage for PCM programer.

3.1.2.1.5 Antenna subsystem: The antenna subsystem included eight antennas, the C-band helices, and multiplexing and switching networks similar to those installed on spacecraft 2. All antennas were operational; however, the HF whip antenna was extended only after landing and remained in the extended position until recovery. A dc-to-ac inverter (26 V ac, 453 cps) was installed in the C-band antenna system to supply power to the phase shifter. On spacecraft 2, the phase shifter obtained power from the dc-to-ac inverter (26 V ac, 400 cps) installed in the attitude control and maneuver electronics (ACME) system.

3.1.2.2 Instrumentation and recording.- In addition to the regular production instrumentation and recording equipment, spacecraft 3 required biomedical instrumentation and recording capabilities. The major instrumentation components are shown in figures 3-4 and 3-5. Table 3-II lists the measured spacecraft parameters mentioned in this report.

3.1.2.2.1 Biomedical instrumentation sensors and signal conditioners: The biomedical instrumentation sensors were attached to each flight crew member's body and the signal conditioners were contained within the flight crew member's undergarments. Selected physiological parameters were supplied by these sensors and signal conditioners to the biomedical tape recorder and to the PCM multiplexer/encoder system for transmission to a PCM ground station. Oral temperature, blood pressure, respiration rate and pattern, and two electrocardiograms were obtained for each flight crew member.

3.1.2.2.2 Biomedical instrumentation tape recorders: A tape recorder for recording selected physiological parameters was supplied for each flight crew member.

3.1.2.2.3 Biomedical instrumentation power: A power supply, located in the cabin section, converted the 28 V dc spacecraft power to the ±10 V dc required by the biomedical instrumentation system.

3.1.2.2.4 Special instrumentation: The special instrumentation associated with the crewman simulators on spacecraft 2 was not required for spacecraft 3.

3.1.2.3 Environmental control system.- The spacecraft 3 environmental control system (ECS), shown in figure 3-6, was essentially the same as the spacecraft 2 ECS. A more efficient launch-cooling heat exchanger was installed and the water management system was operative. Urine and water condensate from the suit-circuit heat exchanger were absorbed by a wick in the water boiler, or dumped overboard. The secondary-oxygen high-flow rate was 0.08 pound per minute for each flight crew member.

The water management system consisted of a water management panel, a water transfer line, and a transparent drinking-water tank located in the reentry assembly, which was replenished from a single water storage tank located in the adapter assembly. Because the fuel cell was not installed, the fuel-cell water collection tank was not required.

3.1.2.4 Guidance and control.- Except for minor changes, the guidance and control systems for spacecraft 3 were of the same configuration as those used on spacecraft 2.

3.1.2.4.1 Control system: All operating control system modes were demonstrated and evaluated. The radar range and range-rate indicator was not installed. The horizon-sensor system was modified to provide improved tracking stability and a more accurate means of alining the infrared (IR) axes of the horizon sensors with the body axes of the spacecraft. Two identical horizon sensors were installed for redundancy.

3.1.2.4.2 Inertial guidance system: All equipment in the inertial guidance system (IGS) was operative and the hand controls and switches were operated by the flight crew. The power supply for the IGS was modified to provide improved high-frequency impedance and also to provide increased isolation for protection against short circuits.

3.1.2.5 Time reference.- The time reference system for spacecraft 3 included an electronic timer, an event timer, and a Greenwich mean time (G.m.t.) clock, similar to those installed in spacecraft 2.

3.1.2.6 Electrical.- Except for the fuel-cell module having been replaced by a battery module, the electrical system for spacecraft 3, shown in figure 3-7, was similar to the electrical system for spacecraft 2. The additional battery module contained three 400-A-hr 16-cell silver-zinc batteries and was installed in the adapter assembly. The four pallet batteries and two pallet buses used for the special instrumentation on spacecraft 2 were not installed on spacecraft 3.

3.1.2.7 Propulsion.- The propulsion system for spacecraft 3 was basically the same as the one used for spacecraft 2.

3.1.2.7.1 Orbital attitude and maneuver system: In the orbital attitude and maneuver system (OAMS), shown in figure 3-8, the four radial-thrusting (translation) thrust chamber assemblies (TCA's) were replaced by dummies. Also, two burst diaphragms normally installed in component package B were deleted.

3.1.2.7.2 Reentry control system: In the reentry control system (RCS), one normally open motor-driven shutoff valve was installed in the fuel feed line of each ring and an identical valve was installed in the oxidizer feed line of each ring. These valves provide a positive shutoff of the systems when they are not required.

3.1.2.8 Pyrotechnics.- Spacecraft 3 required the regular pyrotechnic devices as installed in spacecraft 2 and also the following additional devices:

(a) An ejector to release the fresh air doors.

(b) Reefing-line cutters, guillotines, and mortar used with the drogue parachute.

(c) Lap-belt releases and shoulder-strap cutters, and the "Jetelox" releases associated with the backboard and egress kit jettisoning systems.

All pyrotechnic devices installed in spacecraft 3 were flight qualified.

3.1.2.9 Crew station furnishings and equipment.- Crew station furnishings and equipment required for manned flights, described in reference 1, were installed and operational in spacecraft 3. The following minor changes were incorporated on the display and switch panels (see fig. 3-9):

(a) The space-suit temperature gage indicated from 40° to 100° F.

(b) The cabin pressure gage indicated from 0 to 8 psig.

(c) A record position was added to each mode switch on the voice control center.

(d) The secondary guidance (GUID) switch was changed to permit selection of the inertial guidance system (IGS) and also to permit return to the radio guidance system (RGS) after staging.

(e) The switches for purging the fuel cell were replaced by toggle switches for controlling the motor-driven shutoff valves in the RCS system.

(f) A sensitivity control was added to the flight director indicator (FDI).

(g) The staging light was replaced by a duplicate stage I indicator light to indicate the status of thrust chamber pressure in each subassembly (SA 1 and SA 2) of the launch vehicle stage I engine.

(h) The fuel cell hydrogen and oxygen quantity gages were replaced with a device for displaying an abbreviated version of the flight plan that had been printed on a long film strip designed to fit over a roller.

3.1.2.9.1 Space suit: The G3C intravehicular space suit worn by each flight crew member was a custom-made, close-fitting, full-pressure garment. An internal ventilation system distributed oxygen to the extremities for cooling and respiration. The helmet and the gloves could be removed when desired for comfort and waste functions. The helmet visor could be opened readily for eating or convenience. The remainder of the suit was designed for continuous wear. Entrance to the suit was provided by a long zipper which extended down the middle of the back, through the crotch, and a short distance up the front of the suit. For redundancy, two microphones and two earphones were built into each helmet and were connected to the spacecraft communication system. Full length custom-fitted underwear and a custom-fitted neck dam for water egress were included as suit accessories.

3.1.2.9.2 Water and waste management system: A pistol-shaped water dispenser with a thumb-operated valve was attached to a coiled flexible water hose and mounted in a holster between the ejection seats. A urine-transport system consisting of a detachable relief tube with an adjustable receiver cone and a manually operated bellows accumulator was installed below the water dispenser between the ejection seats. Suitable controls for operating the water and waste system valves were located on an adjacent panel.

3.1.2.9.3 Stowage provisions: The crew station included containers for stowage of all flight crew equipment. A complete list of this equipment, and the location in which it was stowed, is included in table 3-III.

3.1.2.9.4 Ejection seat: Each ejection seat (see fig. 3-10) included the flight crew member's personalized contour, restraint systems, egress kit (containing oxygen), survival kit, ballute, personnel parachute, and associated pyrotechnics.

3.1.2.10 <u>Landing</u>.- In addition to the pilot parachute and the main parachute installed on spacecraft 2, the landing system for spacecraft 3 included a drogue parachute.

3.1.2.10.1 Drogue parachute: The drogue parachute was an 8.3-foot-diameter conical ribbon parachute contained in a mortar in the rendezvous and recovery (R and R) section. The functions of the drogue parachute were to provide high altitude stabilization of the reentry assembly within ±23° of its vertical axis and to deploy the pilot parachute upon command.

3.1.2.10.2 Landing system sequence: The planned GT-3 landing sequence was the same as the GT-2 landing sequence except for the action of the drogue parachute and the command pilot's control of the parachute deployment. The command pilot fired the mortar to deploy the drogue parachute after the reentry assembly had descended to approximately 50 000 feet above sea level. At approximately 10 600 feet, the command pilot fired pyrotechnic devices to sever the drogue parachute attachment cables. The drogue parachute apex cable pulled the pilot parachute from a mortar tube. Approximately 2.5 seconds after deployment of the pilot parachute, the action of a mild detonating fuse (MDF) separated the R and R section from the reentry assembly. As the pilot parachute pulled the R and R section away from the reentry assembly, the main parachute deployed from the open end of the R and R section.

As soon as the reentry assembly was stabilized in a one-point suspension, the pilot initiated pyrotechnic devices to shift to a two-point suspension. The two-point bridle placed the reentry assembly in the proper attitude (35°, nose up) for water landing. After touchdown, the command pilot jettisoned the main parachute.

3.1.2.11 Postlanding and recovery.- In addition to the postlanding and recovery equipment carried on spacecraft 2, flight crew survival equipment was required for spacecraft 3. This equipment included a liferaft pack and a survival kit for each flight crew member.

The survival kit contained food, water treatment materials, signaling devices, and medical supplies.

3.2 GEMINI LAUNCH VEHICLE DESCRIPTION

This section describes the GLV for the GT-3 mission. Since GLV-3 was of the same basic configuration as GLV-2, described in reference 1, only the significant differences will be described in this report. These differences are summarized in table 3-IV.

3.2.1 Structure

Two minor modifications were incorporated in the GLV-3 structure:

(a) In the stage I transportation-section skin splice, the riveting was changed from 0.5-inch to 1-inch pitch.

(b) On the stage I and stage II oxidizer and fuel tank , wider gaskets were installed for the manhole cover installations.

3.2.2 Major Systems

3.2.2.1 Propulsion.- Three modifications were incorporated in the GLV-3 propulsion system:

(a) The contamination shields were deleted from the fuel tank level sensors.

(b) A heat shield was added to the fuel accumulator piston.

(c) A redundant engine shutdown system (RESS) was added to stage II.

3.2.2.2 Flight control.- Three modifications were incorporated in the flight control system:

(a) A capability was added to permit the flight crew to switch back from secondary to primary guidance during stage II flight.

(b) The pitch program in the three-axis reference system (TARS) was changed to accommodate GT-3 mission requirements.

(c) All diodes susceptible to "gold-flaking" were replaced.

3.2.2.3 Radio guidance.- Because of the unique GT-3 mission requirements, the radio guidance system required revised guidance equations for insertion conditions and a change in the yaw velocity bias.

3.2.2.4 Hydraulics.- For GLV-3, the stage I secondary hydraulic system pressure was monitored as a shutdown parameter between engine ignition +2.0 seconds and TCPS make +1.0 seconds. The limit was set at 2000 ± 50 psig.

3.2.2.5 Electrical.- In the electrical system wiring, changes were required to accommodate the guidance - flight control switchback function (see section 3.2.2.2).

3.2.2.6 Malfunction detection.- In the malfunction detection system (MDS) the guidance - flight control switchback function was compatible with other affected systems. The provisions for spacecraft display of staging were deleted. Provisions were installed to permit separate spacecraft displays of thrust chamber underpressure for each stage I engine subassembly (SA 1 and SA 2).

3.2.2.7 Instrumentation.- In the instrumentation system, various FM-FM and PCM-FM measurements were changed.

3.2.2.8 Range safety.- A 3.5-second time delay between range safety shutdown and command destruct was incorporated in the command receivers.

3.2.2.9 Ordnance.- There were no modifications required in the ordnance system.

3.3 GT-3 WEIGHT AND BALANCE DATA

Weight data for the GT-3 space vehicle are shown in the following table:

Condition	Weight (including spacecraft), lb (a)
Ignition	344 519
Lift-off	340 949
BECO	83 401
Stage II, start of steady-state combustion	72 345
Stage II burnout	13 862

[a] Postflight trajectory weights obtained from Aerospace Corporation.

Spacecraft weight and balance data are as follows:

Condition	Weight, lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Launch, gross weight	7111.27	0.43	-0.22	109.80
Retrograde	5231.12	.11	-1.47	130.65
Reentry (0.05g)	4584.71	.12	-1.43	136.20
Main parachute deployment	4247.70	.04	-1.48	131.13
Touchdown (no parachute)	4137.21	.04	-1.53	129.04

[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.

TABLE 3-I.- SPACECRAFT 3 MODIFICATIONS

System	Significant changes incorporated in spacecraft 3 from spacecraft 2 configuration
Reentry assembly structure	Thickness of heat-shield ablative material in accordance with production design
Adapter assembly structure	No significant change
Communications	(a) Two UHF transceivers installed (b) Standby telemetry transmitter installed (used in place of delayed-time transmitter for spacecraft 3) (c) One lightweight headset supplied (d) Two UHF survival beacons supplied (e) One voice transmission tape recorder supplied
Instrumentation	(a) Special spacecraft 2 instrumentation deleted (b) Biomedical instrumentation system installed
Environmental control	(a) Secondary oxygen high-rate flow-rate 0.08 pound per minute to each flight crew member (b) Fuel-cell water-collection tank not installed
Guidance and control	Two horizon sensors installed
Time reference	No significant change
Electrical	(a) Fuel-cell module replaced by battery module containing three silver-zinc batteries (b) Four silver-zinc pallet batteries and two pallet buses deleted
Propulsion	(a) Four radial-thrusting TCA's inoperative in OAMS (b) Burst diaphragms removed from OAMS B-package (c) Two motor-operated propellant shut-off valves installed in RCS

TABLE 3-I.- SPACECRAFT 3 MODIFICATIONS - Concluded

System	Significant changes incorporated in spacecraft 3 from spacecraft 2 configuration
Pyrotechnics	(a) Ejector installed to release fresh air door (b) Reefing-line cutters, guillotines, and mortar provided for the drogue parachute (c) Lap-belt releases and shoulder-strap cutters, and "Jetelox" releases associated with the backboard and egress kit jettison system provided
Crew station furnishings and equipment	(a) Special instrumentation pallets and crewman simulators deleted (b) Flight crew displays and panels modified. (c) G3C spacesuit worn by each flight crew member (d) Stowage facilities provided for food, operational equipment, cameras, and waste
Landing	Drogue parachute installed
Postlanding and recovery	One set of survival equipment (including liferaft and survival pack) included for each flight crew member

TABLE 3-II.- SPACECRAFT INSTRUMENTATION MEASUREMENTS

Measurement	Description	Instrumentation range	Type of data
AA01	Time since lift-off	LSB = $\frac{1}{8}$ sec	Delayed time
AA02	Time since lift-off	LSB = $\frac{1}{8}$ sec	Delayed time
AA03	Time-to-go to retrofire	LSB = $\frac{1}{8}$ sec	Delayed time
AB01	Stage II cut-off (IGS command)	1 = cut-off	Delayed time
AB02	Spacecraft shaped-charge fire	1 = fire	Delayed time
AB03	Launch-vehicle from spacecraft separation	1 = separation	Delayed time
AD01	Adapter shaped-charge fire	1 = fire	Delayed time
AD02	Equipment section separation	1 = separation	Delayed time
AD03	Automatic retrofire initiation	1 = fire	Delayed time
AD05	Retrograde shaped-charge fire	1 = fire	Delayed time
AD06	Manual retrofire initiate	1 = fire	Delayed time
AD08	Retrorocket 3 fire	1 = fire	Delayed time
AD09	Retrorocket 2 fire	1 = fire	Delayed time
AD10	Retrorocket 4 fire	1 = fire	Delayed time
AE02	Pilot parachute deploy	1 = deploy	Delayed time
AE27	Drogue parachute deploy	1 = deploy	Delayed time
AE29	Drogue parachute release	1 = command	Delayed time
MA24	Reference junction temperature	-55 to 200° F	Delayed time
FC03	Outer skin	(a)	Delayed time
FD04	Outer skin	(a)	Delayed time
FD08	Outer skin	(a)	Delayed time
FE11	Ablation material backface	-55 to 1000° F	Delayed time

[a] 1300° F plus reference junction temperature MA24

TABLE 3-III.- CREW STATION STOWAGE LIST

Number	Item	Quantity	Remarks
1	Flight plan filmstrip	1	Installed on flight plan display located on center instrument console
2	Flight booklet	1	Stowed in plotboard
3	Star chart slider	1	Stowed in plotboard
4	Star chart	9	Stowed in plotboard
5	Tape recorder cartridges for voice tape recorder	4	Cartridges inserted in a belt and stowed in right-hand food box
6	Defecation bag	2	Stowed in right-hand food box
7	Food	2 man meals	Stowed in left-hand food box
8	Urine receiver and hose	1	Stowed in left side of modified center food box
9	Inflight medical kit	1	Stowed in rack on spacecraft sidewall, outboard, left-hand seat arm rest
10	Hose interconnects	2	Stowed in right-hand aft food box in a soft pouch with fitting removal wrench

TABLE 3-II. - SPACECRAFT INSTRUMENTATION MEASUREMENTS

Measurement	Description	Instrumentation range	Type of data
AA01	Time since lift-off	LSB = $\frac{1}{8}$ sec	Delayed time
AA02	Time since lift-off	LSB = $\frac{1}{8}$ sec	Delayed time
AA03	Time-to-go to retrofire	LSB = $\frac{1}{8}$ sec	Delayed time
AB01	Stage II cut-off (IGS command)	1 = cut-off	Delayed time
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AD02	Equipment section separation	1 = separation	Delayed time
AD03	Automatic retrofire initiation	1 = fire	Delayed time
AD05	Retrograde shaped-charge fire	1 = fire	Delayed time
AD06	Manual retrofire initiate	1 = fire	Delayed time
AD08	Retrorocket 3 fire	1 = fire	Delayed time
AD09	Retrorocket 2 fire	1 = fire	Delayed time
AD10	Retrorocket 4 fire	1 = fire	Delayed time
AE02	Pilot parachute deploy	1 = deploy	Delayed time
AE27	Drogue parachute deploy	1 = deploy	Delayed time
AE28	Drogue parachute release	1 = command	Delayed time
MA24	Reference junction temperature	-55 to 200° F	Delayed time
PC03	Outer skin	(a)	Delayed time
PD04	Outer skin	(a)	Delayed time
PD08	Outer skin	(a)	Delayed time
PE11	Ablation material backface	-55 to 1000° F	Delayed time

[a] 1300° F plus reference junction temperature MA24

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4	Star chart	9	Stowed in plotboard
5	Tape recorder cartridges for voice tape recorder	4	Cartridges inserted in a belt and stowed in right-hand food box
6	Defecation bag	2	Stowed in right-hand food box
7	Food	2 man meals	Stowed in left-hand food box
8	Urine receiver and hose	1	Stowed in left side of modified center food box
9	Inflight medical kit	1	Stowed in rack on spacecraft sidewall, outboard, left-hand seat arm rest
10	Hose interconnects	2	Stowed in right-hand aft food box in a soft pouch with fitting removal wrench

TABLE 3-III.- CREW STATION STOWAGE LIST - Continued

Number	Item	Quantity	Remarks
11	Blood pressure bulb and hose	1	Stowed with two adapters in the right-hand food box
12	Ejection seat safety pins	2 sets	Stowed on ejection seat backboard adjacent to associated pyro devices
13	Beacon coaxial cable	1	Stowed in right-hand food box
14	16-mm sequence camera body	1	Stowed in left side of modified center food box
15	16-mm film magazine	2	Stowed with item 14. One with camera
16	16-mm camera, 25-mm lens assembly	1	Stowed on camera body, item 14. Cine-ektar II
17	70-mm Hasselblad camera with lens	1	Stowed in upper right side of modified center food box
18	70-mm film magazine	2	One with camera in right side of modified center food box
19	Sight, ring camera assembly	1	Stowed with item 18
20	Swizzle stick	1	Stowed on overhead circuit breaker panel switch guard

TABLE 3-III.- CREW STATION STOWAGE LIST - Continued

Number	Item	Quantity	Remarks
21	Utility electrical cord	2	Stowed connected to the utility light. Cords could be removed from the light assembly when required for other use
22	Plotboards	1	Stowed in a collapsible cloth container on left side of pedestal in left-hand footwell
23	Launch day urine bag	2	Stowed with waste after use in right wing of center food box
24	Waste containers	4	Stowage same as item 6
25	Hygiene pads	2	Packed with food
26	Lightweight headset	1	Stowed same as item 6
27	Orbital path display	1	Stowed on item 22
28	Optical sight	1	Stowed under right-hand main instrument panel
29	CO_2 tapes	5	Stowed same as item 6
30	Urine collection device (UCD) adapter	1	Stowed in right-hand box
31	UCD clamp	2	Stowed with item 30

TABLE 3-III. - CREW STATION STOWAGE LIST - Concluded

Number	Item	Quantity	Remarks
32	Biomedical fitting removal wrench	1	Stowed with hose interconnects in soft pouch in right-hand aft food box
33	Personal hygiene towel	2	Stowed in left-hand food box
34	Knife	2	One per flight crew member. Stowed on space suit
35	Surgical scissors	2	One per flight crew member. Stowed on space suit
36	Sea-urchin egg unit, experiment S-2	1	Stowed on left-hand hatch torque box
37	Blood chromosomal unit, experiment S-4	1	Stowed on right-hand hatch torque box

TABLE 3-IV.- GLV-3 MODIFICATIONS

System	Significant changes incorporated in GLV-3 from GLV-2 configuration
Stage I structure	(a) Transportation-section skin splice riveting changed from $\frac{1}{2}$-inch pitch to 1-inch pitch (b) Manhole cover installation changed from narrow to wide gasket on oxidizer and fuel tanks
State II structure	Manhole cover installations changed from narrow to wide gaskets on oxidizer and fuel tanks
Propulsion	(a) Contamination shields deleted from the fuel tank level sensors (b) Redundant shutdown system provided for the stage II engine through the addition of a squib-operated valve in the oxidizer "boot-strap" line (c) Heat shield added to fuel accumulator piston assembly to protect the shaft and potentiometer
Flight control	(a) Added capability for flight crew initiated switchback from secondary to primary guidance/flight control system during stage II flight (b) Pitch program in TARS changed to accomodate GT-3 mission requirements (c) Diodes susceptible to "gold-flaking" replaced
Radio guidance	(a) Guidance equations revised for GT-3 insertion conditions (b) Yaw velocity bias in guidance equation changed (function of spacecraft c.g.)
Hydraulics	The stage I secondary system pressure monitored as a shutdown parameter between engine ignition +2.0 seconds and TCPS make +1.8 seconds. The limit was set at 2600 ± 50 psig
Electrical	Added circuitry for guidance/flight control switchback function

TABLE 3-IV.- GLV-3 MODIFICATIONS - Concluded

System	Significant changes incorporated in GLV-3 from GLV-2 configuration
Malfunction detection	(a) The malfunction detection package (MDP) has been modified as follows: a. Provisions for switchback added b. Wires shortened and bundle insulation improved c. Better diode insulation provided d. Three capacitors added to suppress rate switch package (RSP) noise NOTE: These changes were effective on GLV-2. However, on GLV-2 the switchback function was not incorporated in the other systems (b) Spacecraft display of staging removed. Separate spacecraft displays of stage I engine subassemblies 1 and 2 effective through staging provided
Instrumentation	Various FM-FM and PCM-FM measurements changed
Range safety	The 3.3-second time delay between range-safety engine shutdown and command destruct incorporated in the command receivers
Ordnance	None

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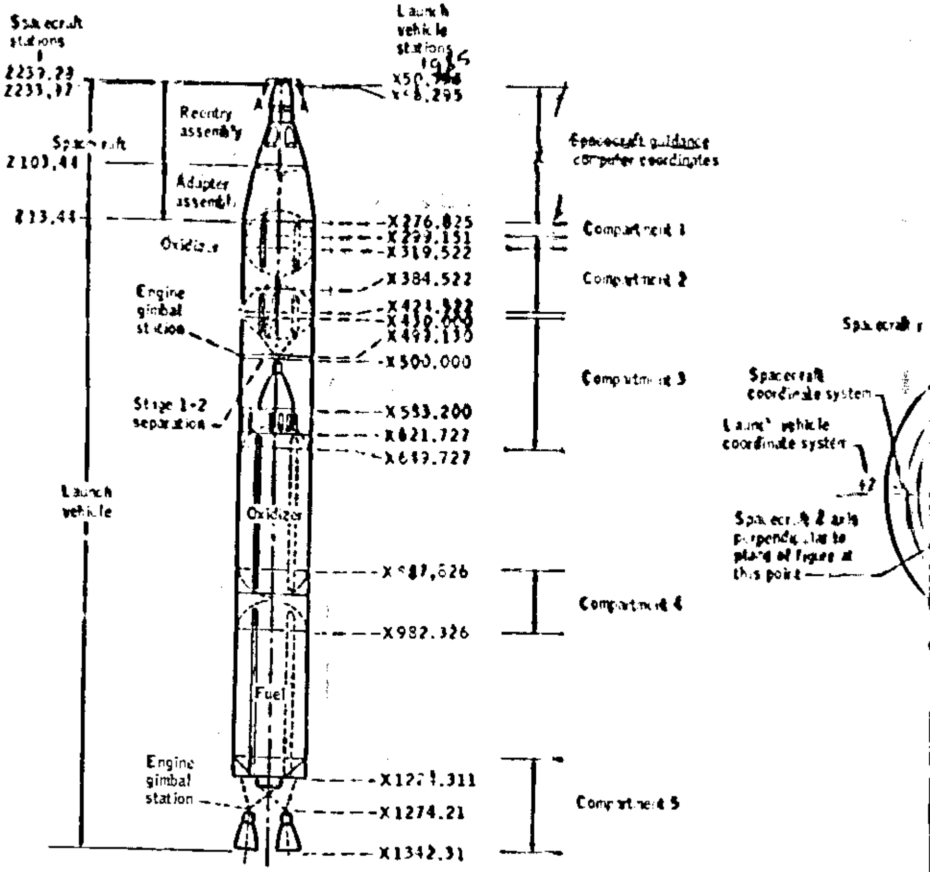
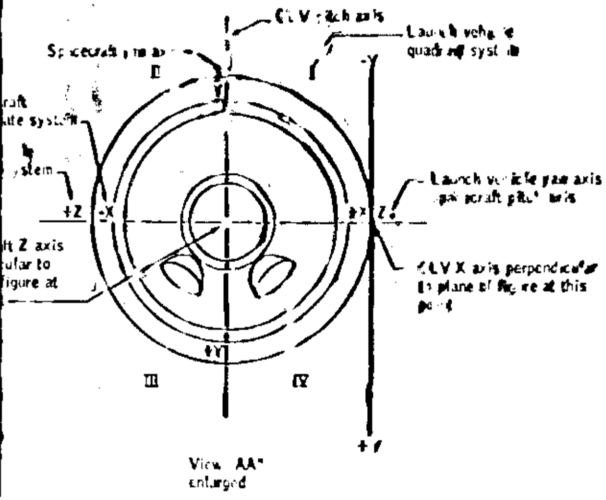
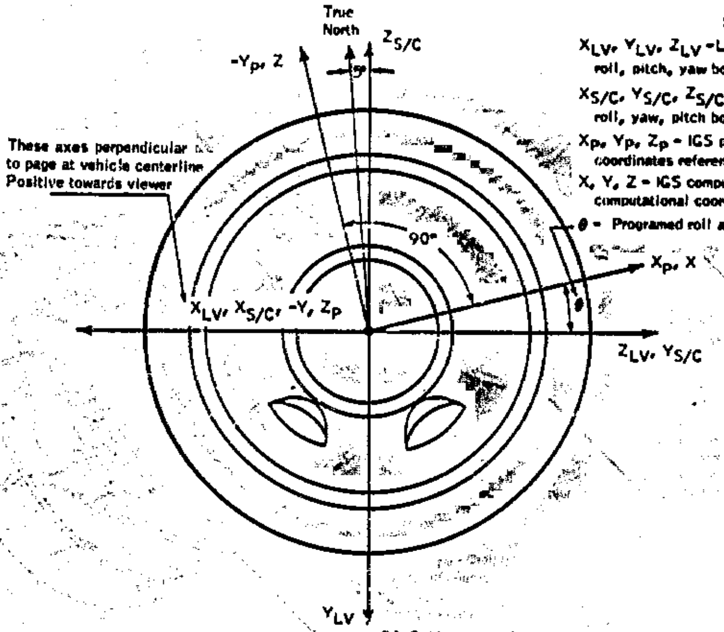




(a) Dimensional.

Figure 3-1. - Launch vehicle-spacecraft relationships.



Note

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 body axes, respectively

X_p, Y_p, Z_p = IGS platform inertial coordinates referenced to launch stand

X, Y, Z = IGS computer computational coordinates

θ = Programed roll angle

(b) Guidance coordinates.

Figure 3-1. Concluded.

UNCLASSIFIED

NASA-S-65-3481

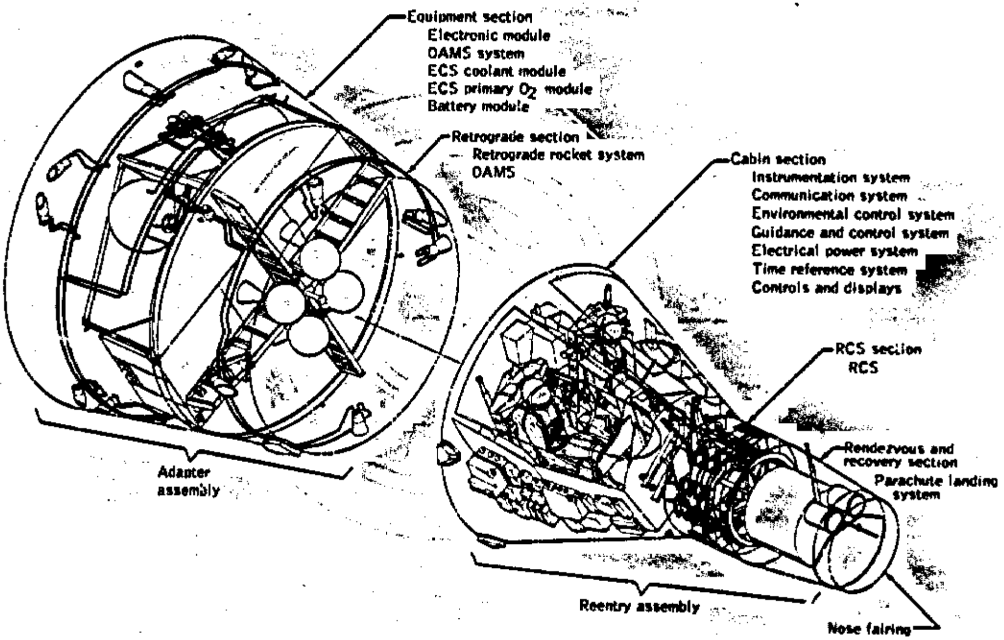


Figure 3-2. - Spacecraft arrangement and nomenclature.

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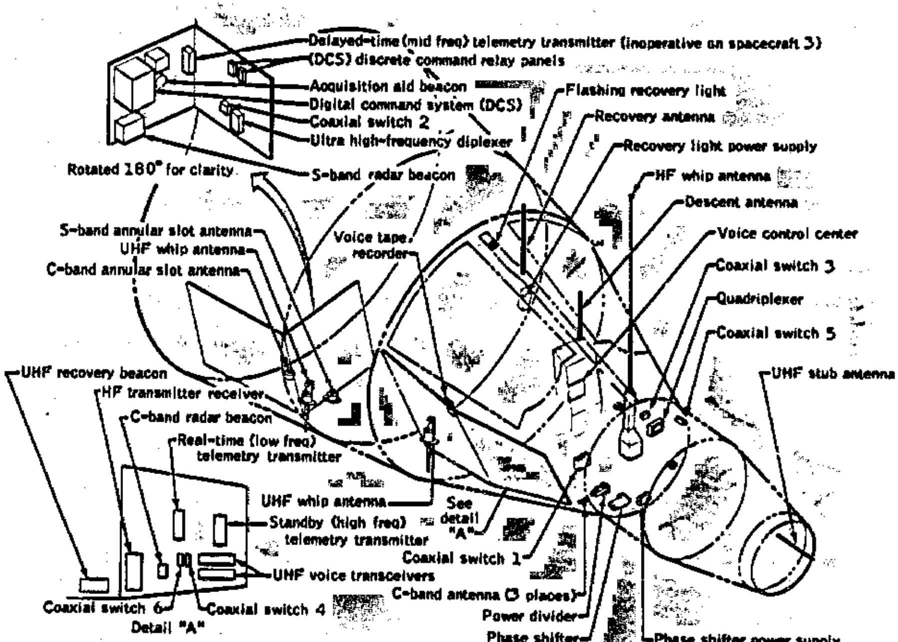

Figure 3-3. - Communication system.

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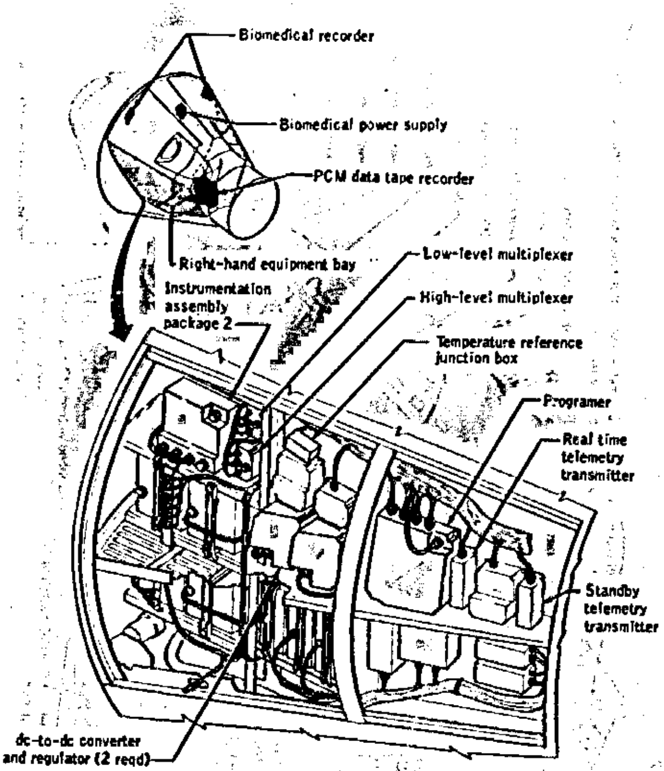

Figure 3-4. - Major instrumentation components (reentry section)

NASA-S-65-3484

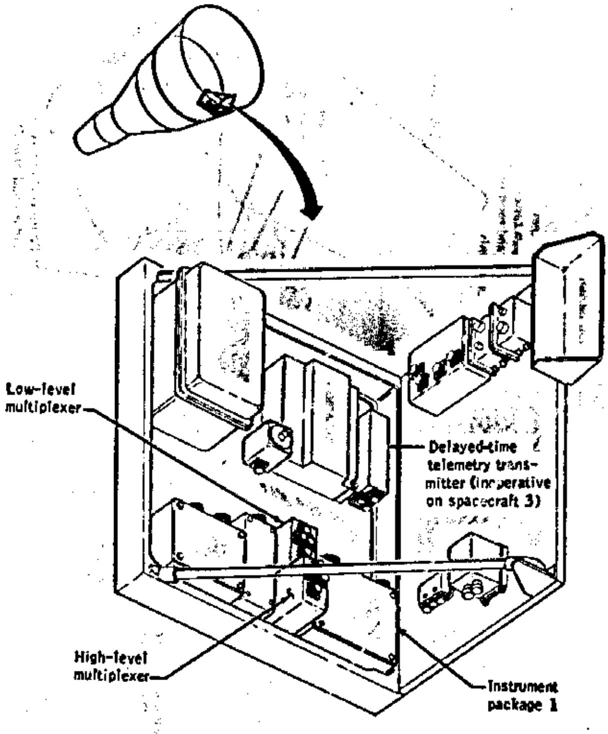

Figure 3-5. - Major instrumentation components (adapter section).

NASA-S-65-3250

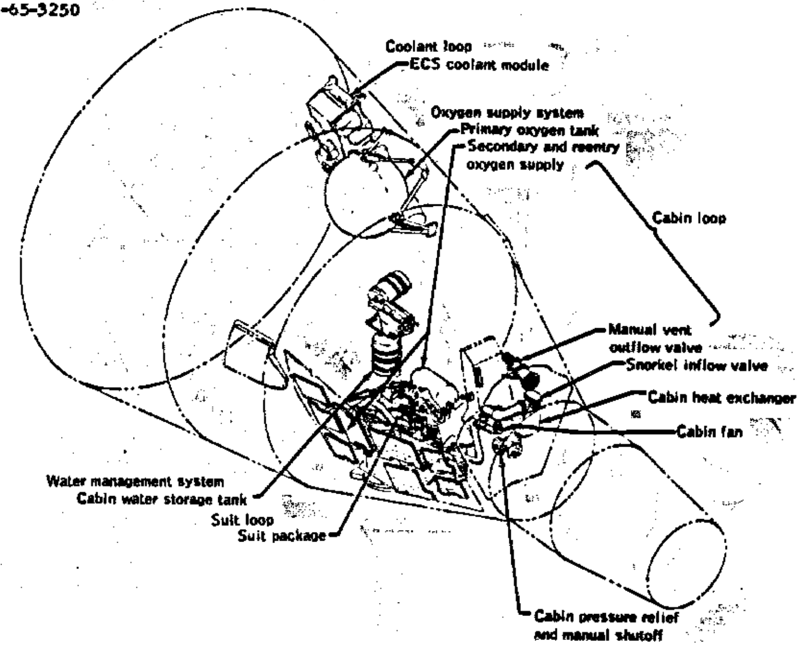

Figure 3-6. - Environmental control system.

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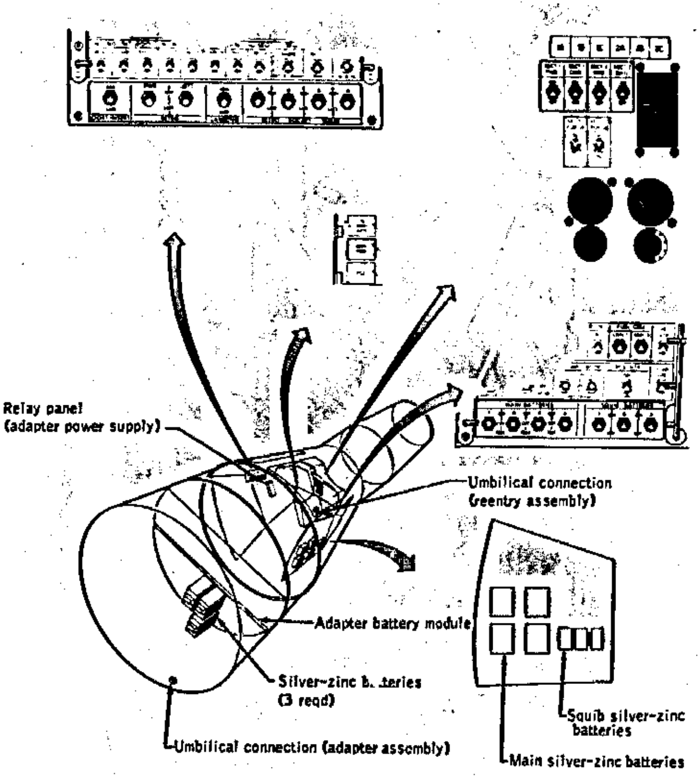

Figure 3-7. - Electrical power system.

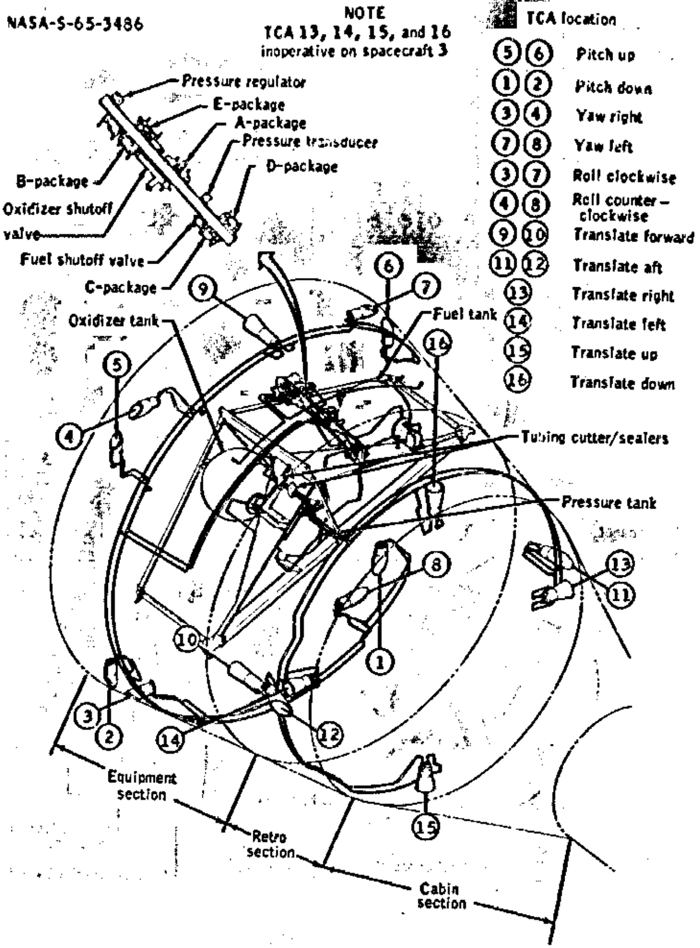

Figure 3-8. - Orbital attitude and maneuver system.

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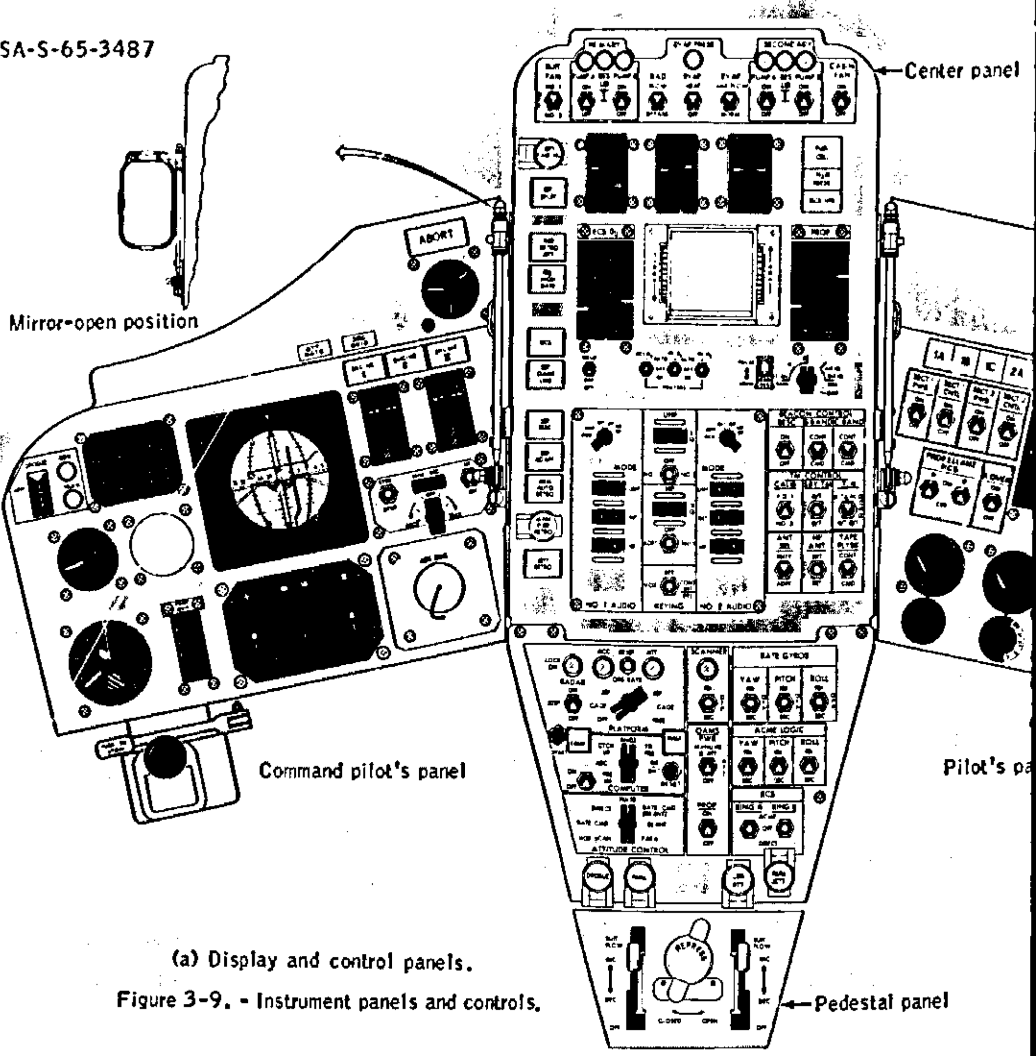
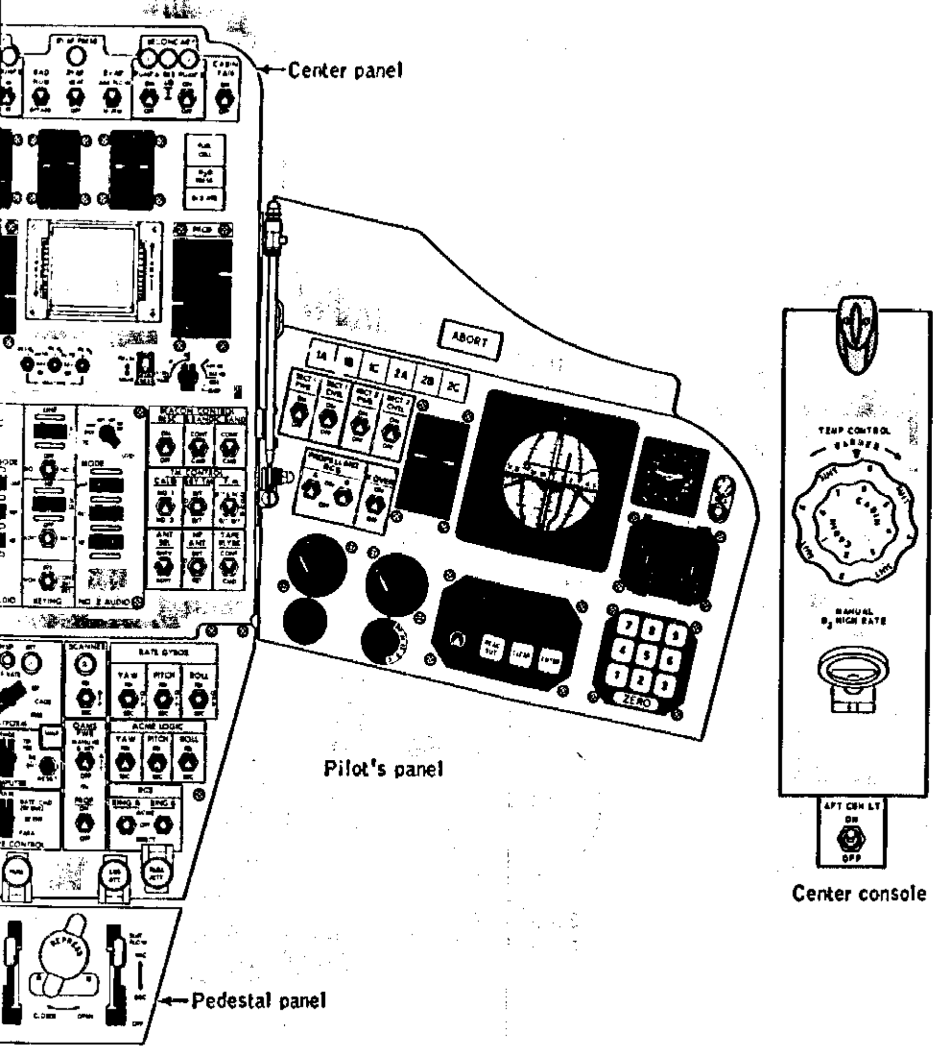
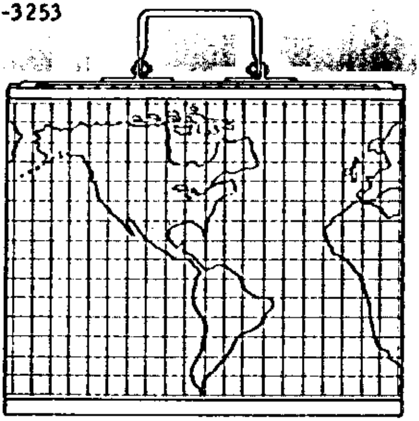


(a) Display and control panels.

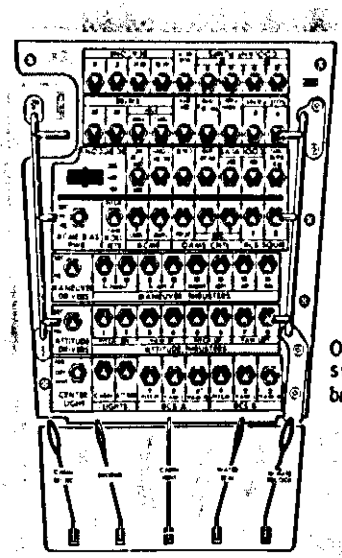
Figure 3-9. - Instrument panels and controls.


Center panel
ABORT
1A
1B
1C
2A
2B
2C
Pilot's panel
TEMP CONTROL
CABIN
ZERO
Center console
Pedestal panel

NASA-S-65-3253



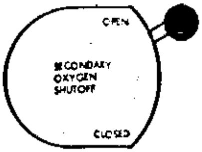
Plotting board



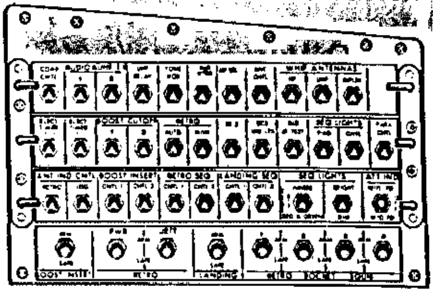
Overhead
switch/ci
breaker pa

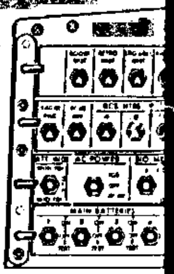
Left switch/circuit breaker panel

Right swi



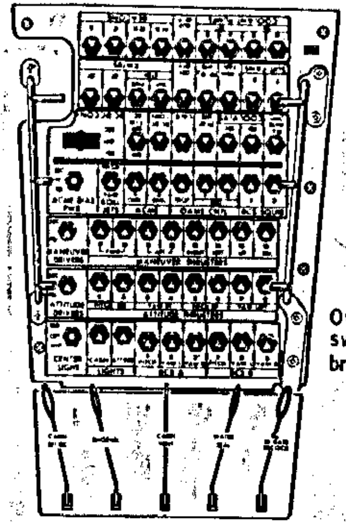
Secondary O_2 control handle





(b)

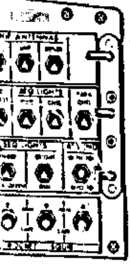
Abort control handle



Overhead switch/circuit breaker panel

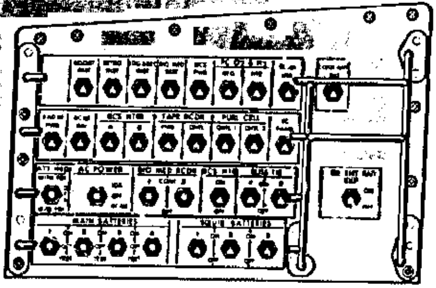
Water management panel

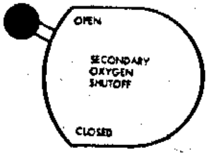
er panel



ntrol handle

Right switch/circuit breaker panel





Secondary O2 control handle

(b) Switch and circuit breaker panels.

Figure 3-9. - Concluded.

NASA-S-65-3488

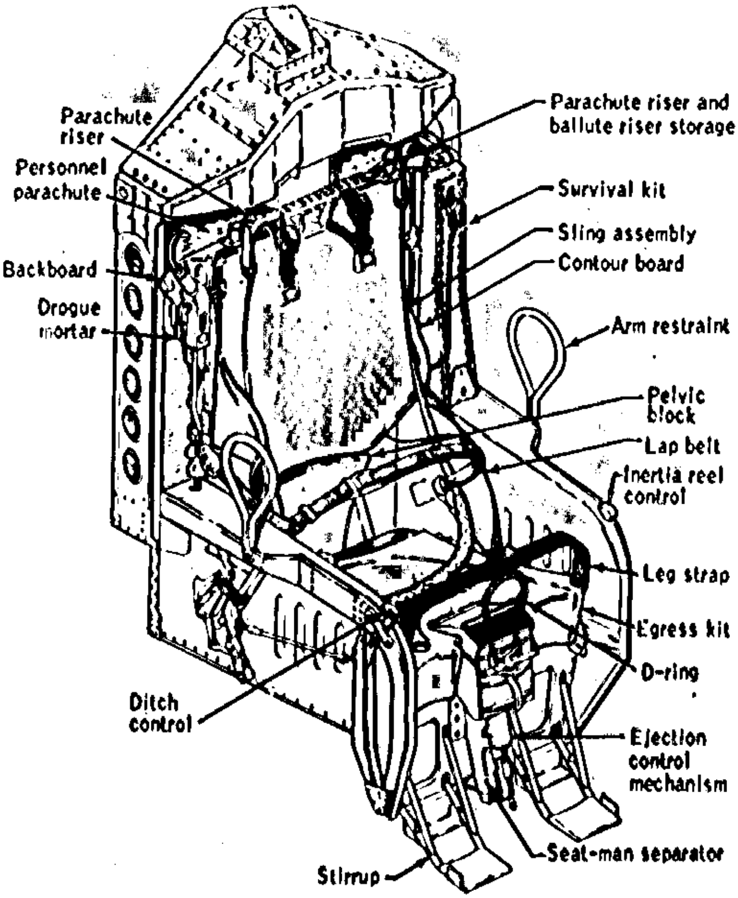


Figure 3-10. - Gemini ejection seat assembly.

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4.0 MISSION DESCRIPTION

4.1 PLANNED MISSION

The GT-3 mission was planned to demonstrate and evaluate manned orbital flight in the Gemini spacecraft, the spacecraft and launch vehicle systems, and the procedures necessary for the support of future manned long duration and rendezvous mission. A launch azimuth of 72° was chosen to take maximum advantage of existing tracking coverage.

The launch vehicle preset program in roll was to provide the required azimuth after launch from a launch-stand azimuth of 85° East of North; after termination of the roll program, the three-axis reference system (TARS) was to initiate a series of three preset pitch rates, the third of which was sequenced to end approximately 7 seconds after first stage cutoff. The radio-controlled portion of sustainer flight was to be initiated approximately 12 seconds after first stage cutoff. This trajectory was planned to inject the spacecraft into orbit at 87 nautical miles with a space-fixed velocity of 25 709 ft/sec and a space-fixed flight-path angle of 0.0°. The final 10 ft/sec of this velocity was to be provided by spacecraft aft-firing thrusters during the separation maneuver.

Just prior to the first perigee (over Texas), a retrograde maneuver of 66 ft/sec was to be performed using the forward-firing thrusters. The purpose of this maneuver was to produce an average spacecraft altitude of 90 nautical miles (87 to 93 nominally). This maneuver was planned to result in a perigee of not less than 75 nautical miles because of orbital lifetime considerations.

During the second orbit over the Indian Ocean (02:20:00 g.e.t.), a series of horizontal out-of-plane maneuvers totaling 14 ft/sec was programed with the spacecraft in a 90° yaw attitude. The series was to consist of a 10-ft/sec ΔV maneuver using the forward-firing thrusters, followed by four 1-ft/sec ΔV maneuvers alternating between using the aft- and forward-firing thrusters. These maneuvers were not specifically designed to modify the trajectory but as a spacecraft exercise; the overall effect, however, would move the nominal end-of-the-mission recovery area further from the downrange islands.

At the end of the third orbit, near Hawaii, the plan was to perform a retrograde translational maneuver of 92.5 ft/sec which was to be in-plane using the aft-firing thrusters. This maneuver was to occur 12 minutes prior to retrorocket initiation and place the spacecraft into a reentry elliptical orbit with a vacuum perigee of 45 nautical miles.

Nominal retrosequence would then provide reentry into a recovery area near Grand Turk Island, at 22.02° North latitude and 69.88° West longitude.

4.2 ACTUAL MISSION

Lift-off of the Gemini launch vehicle occurred at 14:24:00.064 G.m.t., approximately 24 minutes later than planned. Telemetry data indicate that the vehicle was rolled at the desired rate and to the desired azimuth. The preset pitch rates were also within their expected tolerance. All flight dynamic plotboards indicated that the first stage was high in altitude and high in velocity. However, the flight profile was still well within the expected 3σ high trajectory boundary. The lofted trajectory was caused mainly by high first-stage thrust. Stage I engine shutdown occurred 1.7 seconds earlier than predicted.

Staging was initiated at BECO (LO+152.45 seconds), and separation had begun by LO+153.09 seconds. The stage II thrust was higher than nominal, and, as in stage I, engine shutdown occurred early. The lofted trajectory was corrected during radio guidance system (RGS) steering. Second stage engine cutoff (SECO) was sent from the RGS at LO+333.75 seconds. Spacecraft separation did not occur until LO+359.02 seconds at SECO+25.27 seconds. This separation maneuver, using the orbital attitude and maneuver system (OAMS) aft-firing thrusters, ended at LO+371.32 seconds after adding 10.6 ft/sec to the spacecraft velocity.

The spacecraft was inserted into an acceptable elliptical orbit within the 3σ limits. The orbit had a perigee of 87.0 nautical miles and an apogee of 121.0 nautical miles. The apogee was 9 nautical miles lower than planned because the spacecraft insertion velocity was 17 ft/sec less than planned.

Near the end of the first orbit, at 01:33:00 g.e.t., it was planned to operate the forward-firing thrusters (small end forward) (SEF) to lower the apogee to 93 nautical miles maintaining the 87.0-nautical-mile perigee. In order to achieve this near-circular orbit, the planned 66 ft/sec ΔV was reduced to 48 ft/sec because of the first orbit 121.0-nautical-mile apogee. The actual ΔV, as applied during the thrust maneuver of 75 seconds duration was 49 ft/sec, resulting in a satisfactory 85.6-nautical-mile perigee and 91.2-nautical-mile apogee. This maneuver was performed with the spacecraft longitudinal axis parallel to the local horizontal, according to telemetry and flight crew reports.

In the second orbit, while in darkness over the Indian Ocean, three small out-of-plane maneuvers were performed beginning at 02:16:59 g.e.t.

The maneuvers were completed at 02:17:25 g.e.t. The resultant out-of-plane ΔV was approximately 10 ft/sec toward North. Over Hawaii in the third orbit at 04:21:45 g.e.t., it was planned to operate the aft-firing OAMS thrusters (blunt end forward) (BEF) to achieve the planned reentry elliptical orbit with a predicted vacuum perigee of 45 nautical miles. In order to achieve this planned trajectory the planned 92.5 ft/sec ΔV had to be increased to 96 ft/sec because a precise average 90-nautical-mile orbit was not achieved with the first maneuver. The actual ΔV as applied during this thrust maneuver of 111 seconds duration was 98.4 ft/sec, resulting in a satisfactory reentry elliptical orbit with a vacuum perigee of 43.4 nautical miles. The spacecraft was at nominal attitudes of 0° pitch, 180° yaw, and 0° roll during this maneuver according to telemetry and flight crew reports.

At 04:32:28 g.e.t., the equipment section was separated. Automatic retrofire began at 04:33:23 g.e.t., and the retrorockets fired sequentially - 1, 3, 2, 4. From ignition of retrorocket number 1, the other rockets fired after time differences of 5.5, 5.2, and 6.0 seconds. The retrograde section was jettisoned at 04:34:08.6 g.e.t. At 04:40:26 g.e.t. the command pilot began the range needle control technique of reentry. At about 04:43:22 g.e.t., when the backup reverse bank-angle time was nearing, the command pilot rolled the spacecraft to a full lift attitude because the downrange error did not decrease to zero.

The spacecraft entered blackout at 04:39:59 g.e.t., and the end of blackout was 04:45:00 g.e.t. Maximum reentry deceleration was 4.3g at 04:45:05 g.e.t. A maximum dynamic pressure of 272 lb/in^2 occurred at 04:45:03 g.e.t. The drogue parachute was deployed manually at 04:46:51 g.e.t. Pilot parachute deployment occurred at 04:48:24 g.e.t., and landing occurred at 04:52:31 g.e.t. The landing point (retrieval point reported by the recovery ship) was at 22°26' N latitude, 70°51' W longitude.

4.3 SEQUENCE OF EVENTS

The times at which major events were planned and executed are presented in table 4-I. All events were completed as scheduled with the expected tolerance, indicating a satisfactory flight. The original plan was to jettison the radar and horizon sensor covers 45 seconds after BECO; however, as a result of qualification systems tests a decision was made to jettison the covers after spacecraft insertion into orbit.

[redacted]

4.4 FLIGHT TRAJECTORIES

The planned trajectories were preflight calculated nominal trajectories based on launch and insertion conditions in reference 3 and planned GT-3 mission attitudes and sequences. The actual trajectories are based on the Manned Space Flight Network tracking data and actual flight attitudes and sequences. The Patrick Air Force Base and 1959 ARDC model atmospheres were used below and above 25 nautical miles, respectively, for all trajectories except the actual launch phase, which used the atmosphere at the time of launch up to 25 nautical miles. The earth model for all trajectories was the Fischer Ellipsoid. A ground track of the GT-3 mission is presented in figure 4-1, and an altitude-longitude profile in figure 4-2. Launch, orbit, and reentry trajectories are presented in figures 4-3 to 4-5.

4.4.1 Launch

The launch trajectory data shown in figure 4-3 is based on the real-time output of the range safety impact predictor 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, seconds
IP-3600 (FPQ-6)	0 to 48
GMCF (GE Mod III)	48 to 404

The actual launch trajectory as compared with the planned launch trajectory in figure 4-3 was high in velocity, altitude, and flight-path angle during stage I powered flight. After BECO, the radio guidance system (RGS) corrected the trajectory error and guided the second stage to a near-nominal insertion. At BECO, the actual velocity, altitude, and flight-path angle were higher than planned by 96 ft/sec, 13 114 feet, and 2.23°, respectively. At SECO, the velocity and flight-path angle were low by 16 ft/sec and 0.02°, respectively, and the altitude was high by 338 feet. At spacecraft separation, the velocity was 17 ft/sec lower than planned, but the altitude and flight-path angle were high

[redacted]

[redacted]

by 376 feet and 0.04°, respectively. A comparison of the actual and planned trajectory parameters is presented in table 4-II. Conditions at spacecraft separation were obtained by integrating the spacecraft position and velocity vector from Guaymas tracking data backward through the orbital attitude and maneuver system (OAMS) separation sequence. The GE Mod III guidance and MISTRAM tracking radars had insertion velocities of 5 ft/sec and 12 ft/sec, respectively, higher than the actual velocity as obtained from the more accurate orbital ephemeris.

4.4.2 Orbit

The orbital trajectory is shown in figure 4-4. The actual trajectory was determined by three vector sources from Goddard Space Flight Center (GSFC). The trajectory before the Texas maneuver was determined by integrating the Guaymas vector from the first pass backward to insertion and forward to the initiation of the Texas maneuver. The trajectory between the Texas and lateral maneuvers was determined by integrating the Bermuda vector on the second pass backward to the end of the Texas maneuver, then forward to the initiation of the lateral maneuver. The trajectory from the end of the lateral maneuver to the preretro maneuver was determined by integrating the Bermuda vector from the third pass backward to the end of the lateral maneuver, then forward to the beginning of the preretro maneuver.

Comparison of the actual and planned orbital parameters and maneuvers is presented in table 4-II. The slight underspeed at insertion reduced the Texas maneuver requirements by the same velocity. The maneuver velocities calculated from the integrated trajectories are in agreement with the velocities determined from telemetry data of thruster activity and spacecraft attitudes.

4.4.3 Reentry

The actual and planned reentry phase of the trajectory is shown in figure 4-5. The planned reentry trajectory was based on manual closed-loop reentry guidance as described in reference 2. During the actual mission the flight crew flew a ground-computed bank angle until the on-board computer displayed zero cross-range error for landing. The flight crew then flew the spacecraft at full lift to correct the indicated downrange error, in an attempt to compensate for the unexpected low lift.

The actual trajectory was determined by integrating the Bermuda vector in the third orbit through the preretro maneuver and retrofire, then integrating the White Sands reentry vector to landing. Table 4-II

[redacted]

contains a comparison of the actual and planned reentry dynamic parameters and landing. The actual landing point was about 60 nautical miles short of the planned landing point. Integrating through the actual bank angles and using the preflight aerodynamics resulted in a landing point farther down range than planned. In order to simulate the actual reentry trajectory and hit the retrieval landing point, the lift coefficient was changed to reduce the lift-drag ratio by an average 32 percent. The spacecraft L/D aerodynamic computations during reentry are amplified in section 5.1.1 and further verify the computations used in the integrated trajectory. After the lift coefficient was changed, communications blackout from the integrated trajectory agreed with the recorded blackout times, the peak deceleration agreed within 0.1 g of the onboard telemetered deceleration, and the integrated trajectory touchdown time was within 2 seconds of actual touchdown. These sequences confirm the validity of both the integrated trajectory and the change in lift coefficient.

[redacted]

TABLE 4-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	-3.37	0.03
Stage I MDTCPS makes subassembly 1	-2.30	-2.45	-0.15
Stage I MDTCPS makes subassembly 2	-2.30	-2.36	-0.06
TCPS subassembly 1 and subassembly 2 make	-2.20	-2.25	-0.05
Shutdown lockout	-0.40	-0.45	-0.05
Lift-off (pad disconnect separation) (14:24:00.064 G.m.t.)	0	0	0
Roll program start	10.16	10.14	-0.02
Roll program end	20.48	20.42	-0.06
Pitch program rate no. 1 start	23.04	23.01	-0.03
Pitch program rate no. 1 end, no. 2 start	88.32	88.22	-0.10
IGS update sent	100.00	105.00	5.00
Control system gain change no. 1	104.96	104.82	-0.14
Pitch program rate no. 2 end, no. 3 start	119.04	118.84	-0.20
IGS update sent	140.00	145.00	5.00
Stage I engine shutdown circuitry armed	144.64	144.42	-0.22
Stage I MDTCPS unmake	154.01	152.38	-1.63
BECO (stage I engine shutdown (87FS2))	154.13	152.43	-1.70
Staging switches actuate	154.13	152.43	-1.70
Signals from stage I rate gyro package to flight control system discontinued	154.13	152.43	-1.70
Hydraulic switchover lockout	154.13	152.43	-1.70
Telemetry ceases, stage I	154.13	152.43	-1.70
Staging nuts detonated	154.13	152.43	-1.70
Stage II engine ignition signal (91FS1)	154.13	152.43	-1.70
Control system gain change	154.13	152.43	-1.70
Stage separation begin	154.83	153.09	-1.74
Stage II engine MDFJPS make	155.03	153.08	-1.95

[redacted]

TABLE 4-I.- SEQUENCE OF EVENTS - Concluded

Event	Planned time, g.e.t.	Actual time, g.e.t.	Difference, sec
Launch phase, sec			
Pitch program rate no. 3 end	162.56	162.32	-0.24
Radio guidance enable	162.56	162.31	-0.25
First guidance command signal received by TARS	169.00	168.50	-0.50
Spacecraft scanner cover jettisoned	NA	381.00	
Spacecraft radar cover jettisoned	NA	381.00	
Stage II engine shutdown circuitry armed	317.44	317.00	-0.44
SECO (stage II engine shutdown (91FS2))	338.42	333.75	-4.67
Redundant stage II shutdown initiated	338.42	333.77	-4.65
Stage II MDFJPS break	338.72	333.88	-4.84
Spacecraft separation (shape charge fired)	358.4	359.02	0.62
OAMS on	358.4	357.52	0.08
OAMS off	369.9	371.32	1.42
Orbital phase, hr:min:sec			
Texas maneuver initiate	01:33:00	01:32:59	-1.0
Texas maneuver complete	01:34:40	01:34:14	-26.0
Lateral maneuver initiate	02:20:00	02:16:59	-181.0
Lateral maneuver complete	02:20:35	02:17:25	-190.10
Reentry phase, hr:min:sec			
Preretro maneuver initiate	04:21:45	04:21:23.15	-21.85
Preretro maneuver complete	04:23:29	04:23:14.22	-14.78
Retrofire	04:33:45	04:33:22.74	-22.26
Communications blackout begins	04:40:42	04:39:59.0	-43.0
Communication blackout ends	04:45:21	04:45:00.1	-20.9
Drogue parachute deployment	04:47:10	04:46:51.18	-18.82
Pilot parachute deployed	04:48:49	04:48:24.34	-24.66
R and R section released, main parachute deployed	04:48:52	04:48:26.50	-25.5
Spacecraft landing	04:53:17	04:52:31	-46.0

TABLE 4-II.- COMPARISON OF PLANNED AND ACTUAL TRAJECTORY PARAMETERS

Condition	Planned	Actual	Difference
SECO			
Time from lift-off, sec.	358.42	353.75	-4.67
Time from lift-off, min:sec.	5:58.42	5:53.75	-00:04.67
Geodetic latitude, deg North	30.55	30.56	0.01
Longitude, deg West	72.05	72.21	0.16
Altitude, ft	530 848	531 186	338
Altitude, n. mi.	87.4	87.4	0.0
Range, n. mi.	461.5	453.6	-7.9
Space-fixed velocity, ft/sec	25 603	25 587	-16
Space-fixed flight-path angle, deg	0.01	-0.01	-0.02
Space-fixed heading angle, deg East of North	77.75	77.66	-0.09
Spacecraft separation			
Time from lift-off, sec	358.42	359.02	0.60
Time from lift-off, min:sec.	5:58.42	5:59.02	00:00.60
Geodetic latitude, deg North	30.83	30.92	0.09
Longitude, deg West	70.57	70.35	-0.22
Altitude, ft	531 129	531 505	376
Altitude, n. mi.	87.4	87.5	0.1
Range, n. mi.	539.7	552.9	13.2
Space-fixed velocity, ft/sec	25 699	25 682	-17
Space-fixed flight-path angle, deg	0.00	0.04	0.04
Space-fixed heading angle, deg East of North	78.54	78.66	0.12
Orbital parameters (a) Before orbit change			
Perigee altitude, statute mi.	100.2	100.0	-0.2
Perigee altitude, n. mi.	87.1	87.0	-0.1
Apogee altitude, statute mi.	149.7	139.3	-10.4
Apogee altitude, n. mi.	130.1	121.0	-9.1
Period, min:sec	88:29	88:18	-00:11
Inclination angle, deg	32.5	32.6	0.1

TABLE 4-II.- COMPARISON OF PLANNED AND ACTUAL TRAJECTORY - Concluded

Condition	Planned	Actual	Difference
Orbital parameters - Concluded (b) After orbit change			
Perigee altitude, statute mi.	100.1	98.4	-1.7
Perigee altitude, n. mi.	87.0	85.6	-1.4
Apogee altitude, statute mi.	107.0	104.8	-2.2
Apogee altitude, n. mi.	93.0	91.2	-1.8
Period, min:sec	87:47	87:47	00:00
Inclination angle, deg	32.5	32.6	0.1
Maneuvers			
Texas maneuver initiate, hr:min:sec, g.e.t.	01:33:00	01:32:59	-00:00:01
Maneuver complete, hr:min:sec, g.e.t.	01:34:40	01:34:14	-00:00:26
ΔV, ft/sec	66(48)[a]	49	-17(1)[a]
Lateral maneuver initiate, hr:min:sec., g.e.t.	02:22:00	02:16:59	-00:05:01
Maneuver complete, hr:min:sec, g.e.t.	02:22:35	02:17:25	-00:05:10
ΔV, ft/sec	19	19	0
Preretro maneuver initiate, hr:min:sec	04:21:45	04:21:23	-00:00:22
Maneuver complete, hr:min:sec, g.e.t.	04:23:29	04:23:14	-00:00:15
ΔV, ft/sec	92.5(96)[a]	98.4	5.9(2.4)[a]
Maximum conditions			
Altitude, statute mi.	149.7	139.1	-10.5
Altitude, n. mi.	130.1	121.0	-10.1
Space-fixed velocity, ft/sec	25 709	25 694	-15
Earth-fixed velocity, ft/sec	24 394	24 379	-14
Exit acceleration, g	7.2	7.6	0.4
Exit dynamic pressure, lb/sq ft	748	709	-41
Reentry deceleration, g	6.6	4.3	-2.3
Reentry dynamic pressure, lb/sq ft	413	272	-141
Landing point			
North latitude, deg:min	22:01	22:26[b]	00:25
West longitude, deg:min	69:52	70:51	00:59

[a] Updated maneuvers as determined in real time

[b] Retrieval point of carrier

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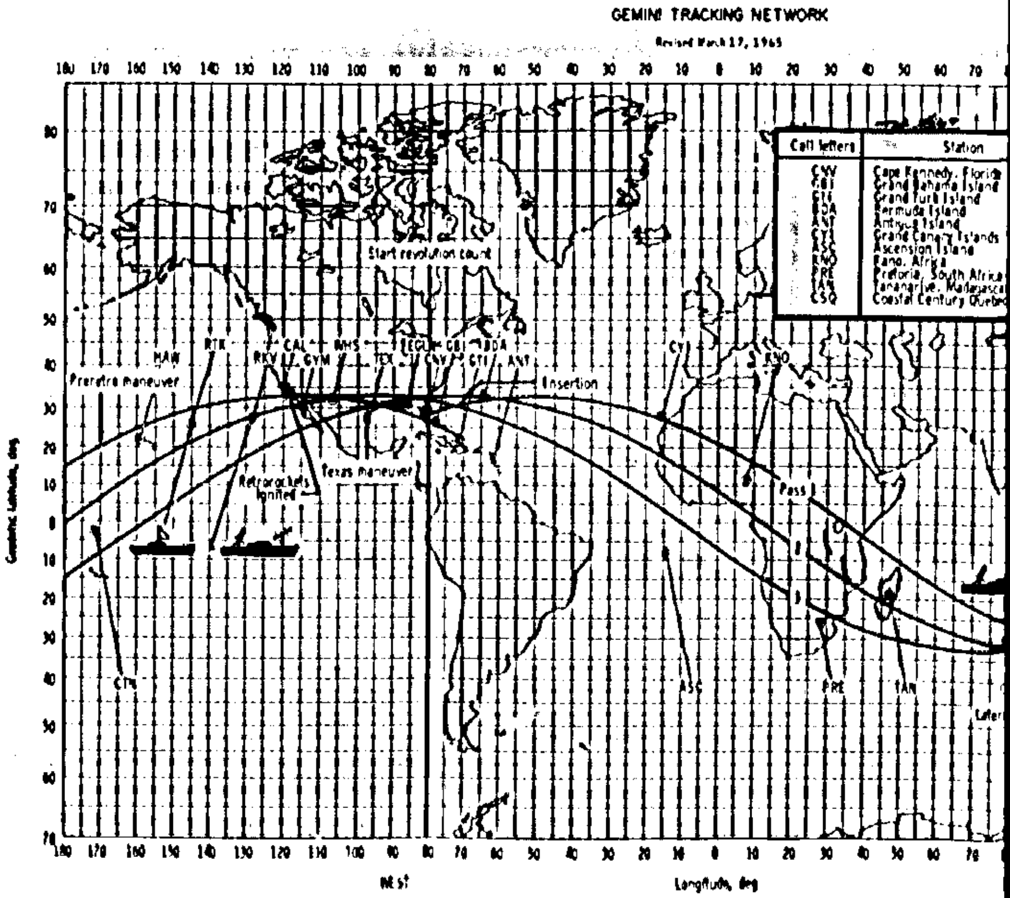
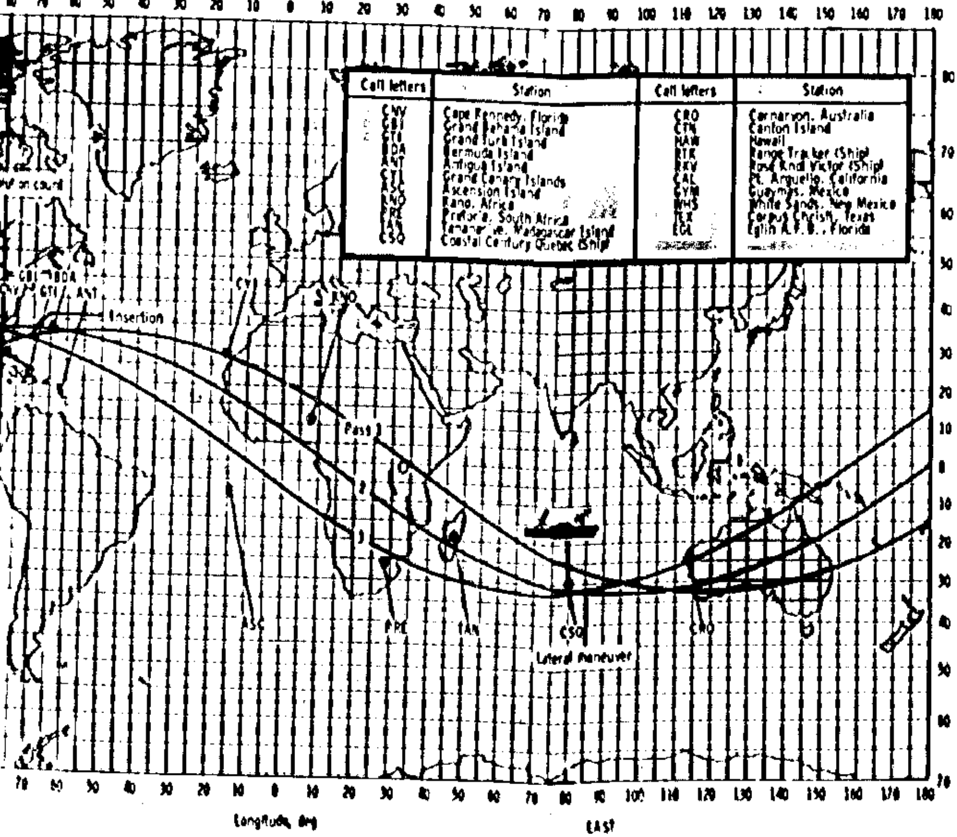


Call letters	Station
CNV	Cape Kennedy, Florida
GBI	Grand Bahama Island
GTI	Grand Turk Island
BDA	Bermuda Island
ANT	Antigua Island
CYI	Grand Canary Islands
ASC	Ascension Island
KNO	Kano, Africa
PRE	Pretoria, South Africa
TAN	Tananarive, Madagasca
CSQ	Coastal Century Quebe

Figure 4-1 - Ground track for the GT-3 orbital mission.

GEMINI TRACKING NETWORK

Revised March 17, 1965



Call letters	Station	Call letters	Station
CNV	Cape Kennedy, Florida	CRO	Carnarvon, Australia
GBI	Grand Bahama Island	CTN	Canton Island
GTK	Grand Turk Island	HAW	Hawaii
BDA	Bermuda Island	RTK	Range Tracker (Ship)
ANT	Antigua Island	RKV	Rose Knot Victor (Ship)
CYI	Grand Canary Islands	CAL	Pt. Arguello, California
ASC	Ascension Island	GYM	Guaymas, Mexico
KNO	Kano, Africa	WHS	White Sands, New Mexico
PRE	Pretoria, South Africa	TEX	Corpus Christi, Texas
TAN	Tananarive, Madagascar Island	EGL	Eglin A.F.B., Florida
CSQ	Coastal Century Quebec (Ship)	[illegible]	[illegible]

Figure 4-1 - Ground track for the GT-3 orbital mission.

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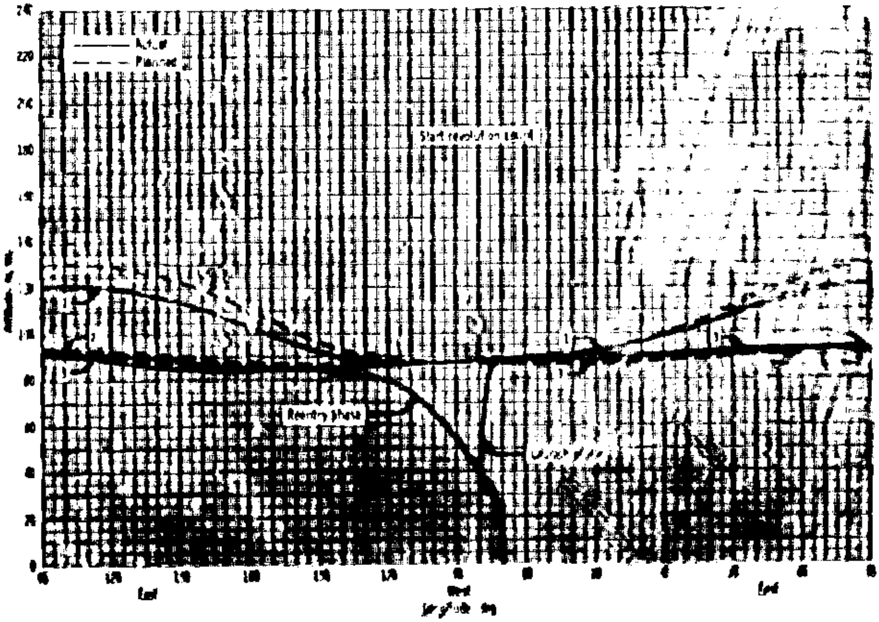
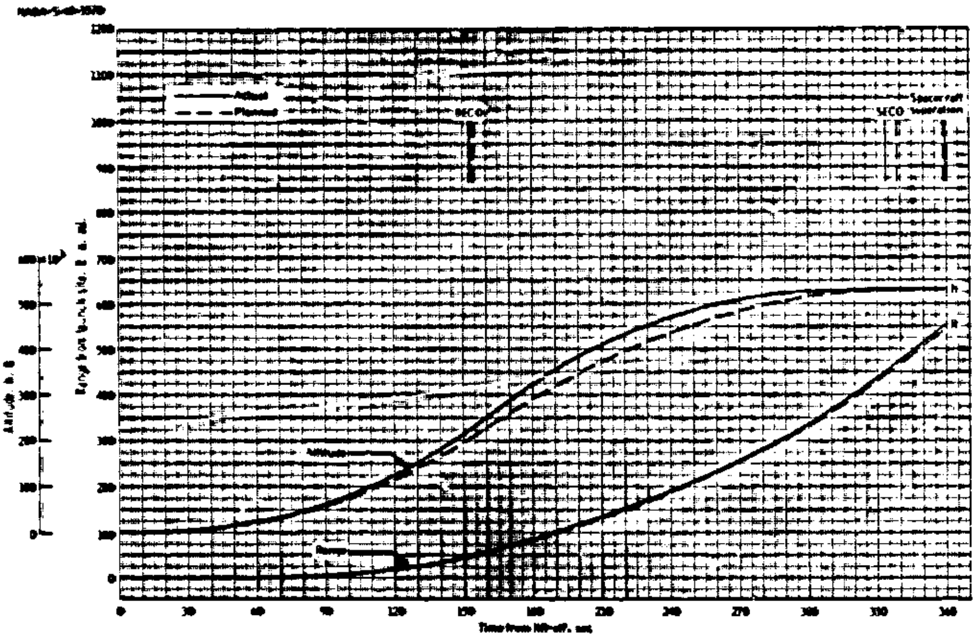
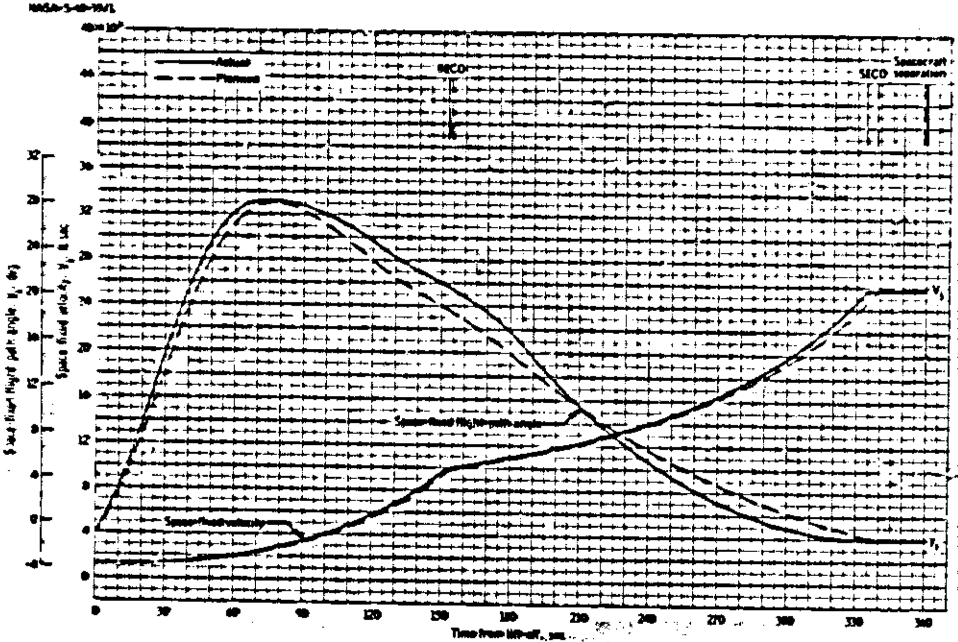


Figure 4-2.- Altitude - longitude profile.



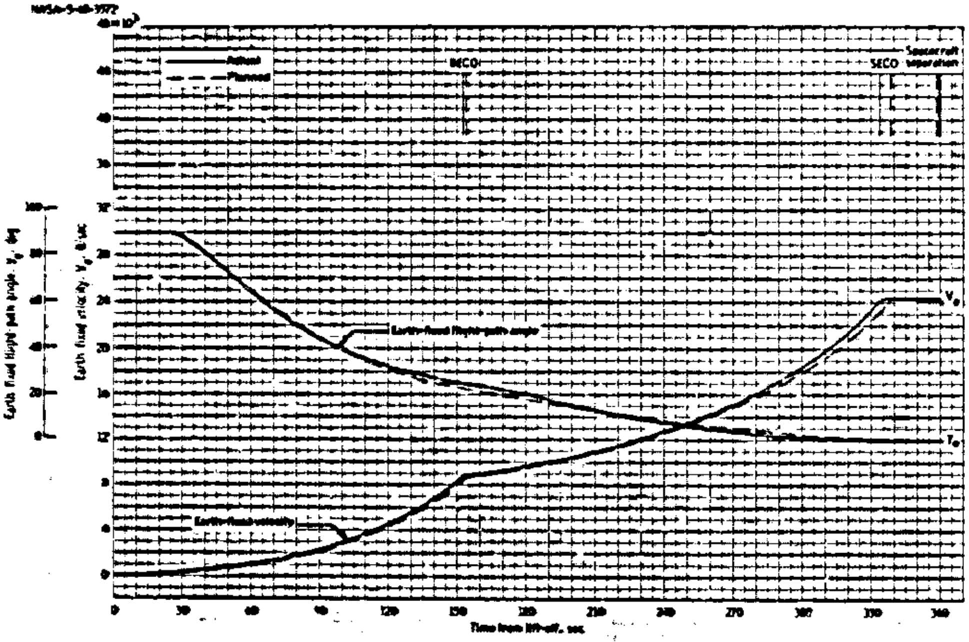
(a) Altitude and range.

Figure 4-3.- Time histories of trajectory parameters for GT-9 mission launch phase.



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

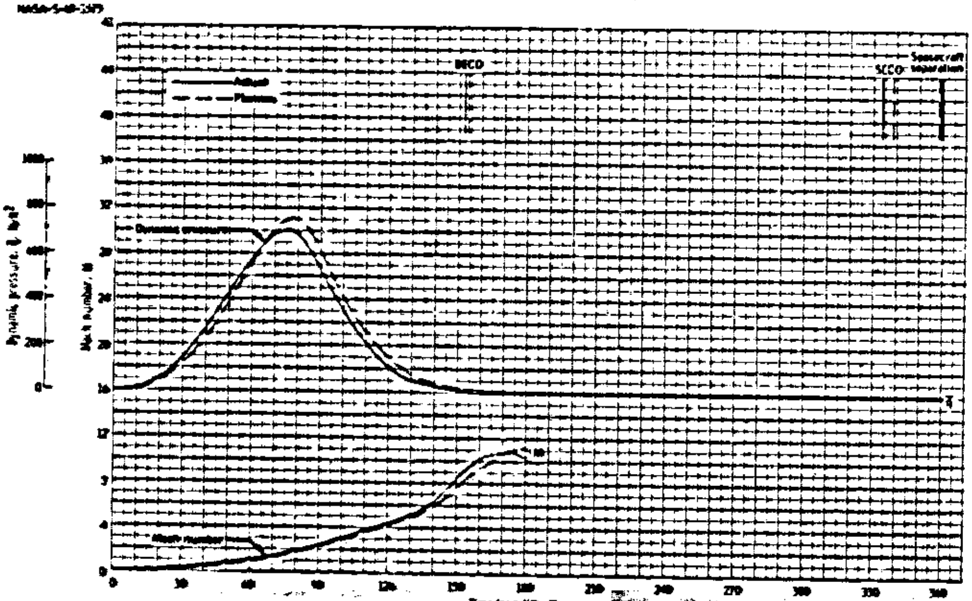
Figure 4-3 - Continued.



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

Figure 4-3 - Continued.

[redacted]



(d) Dynamic pressure and Mach number.

Figure 4-3.- Continued.

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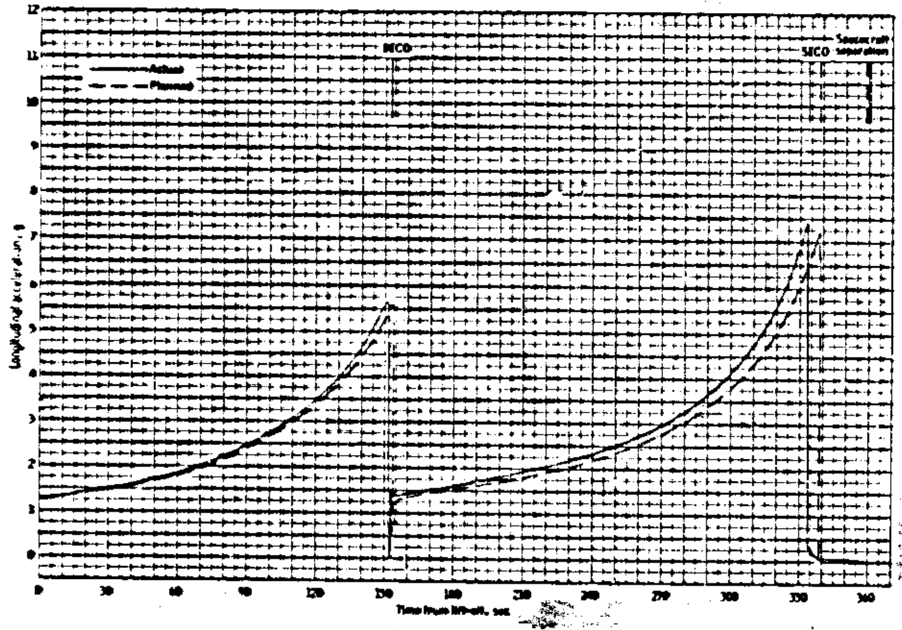

(a) Longitudinal acceleration.

Figure 4-3.- Concluded.

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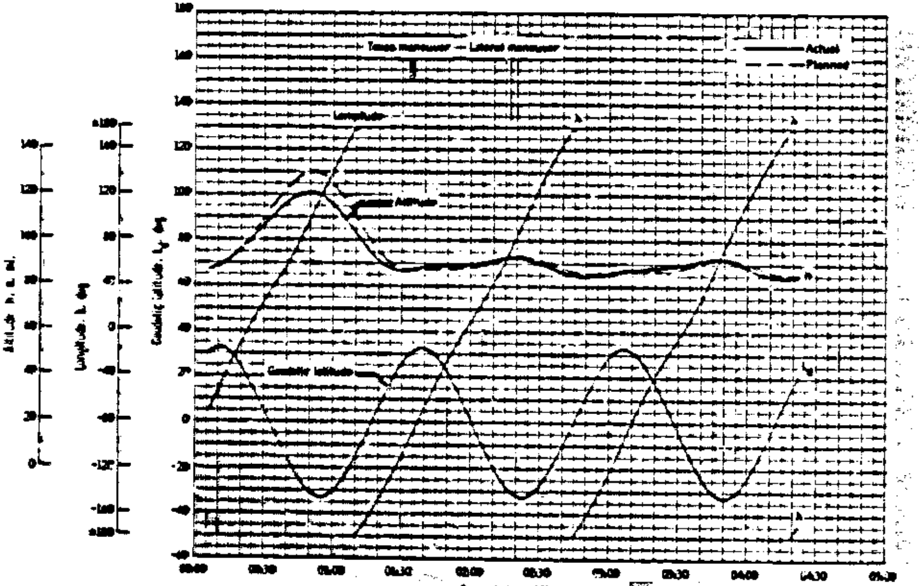

(a) Latitude, longitude, and altitude.

Figure 4-6 - Time histories of trajectory parameters for GT-3 mission orbital phase.

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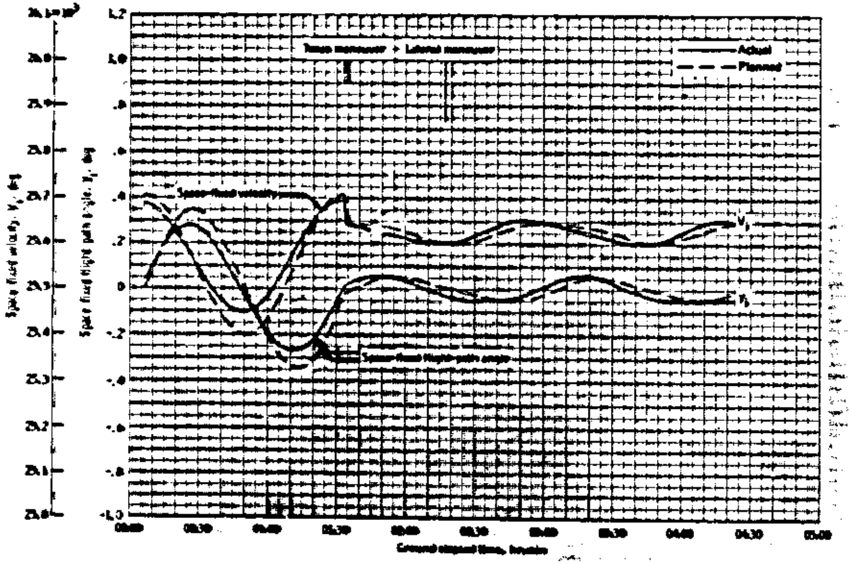
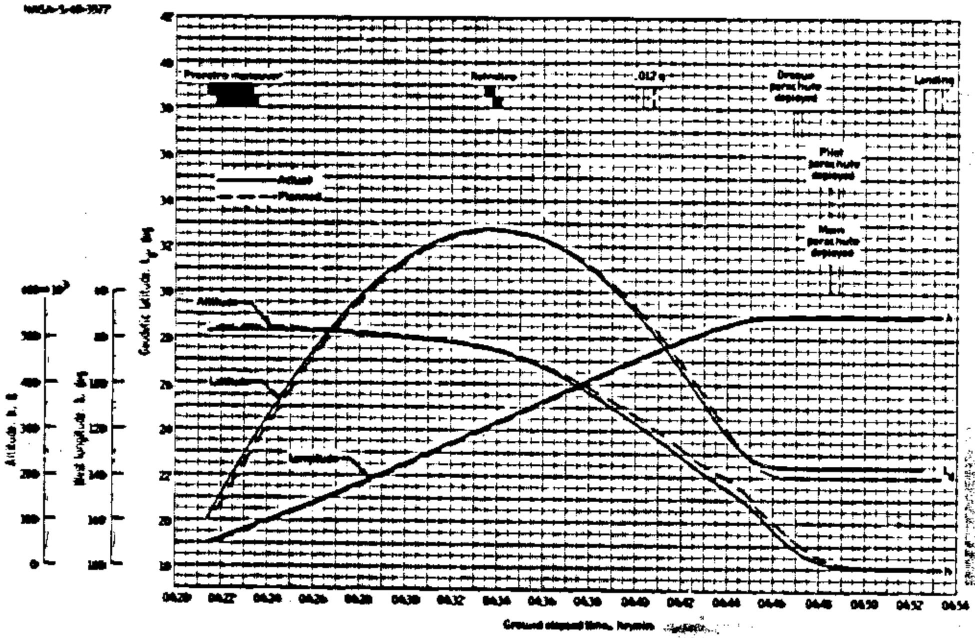


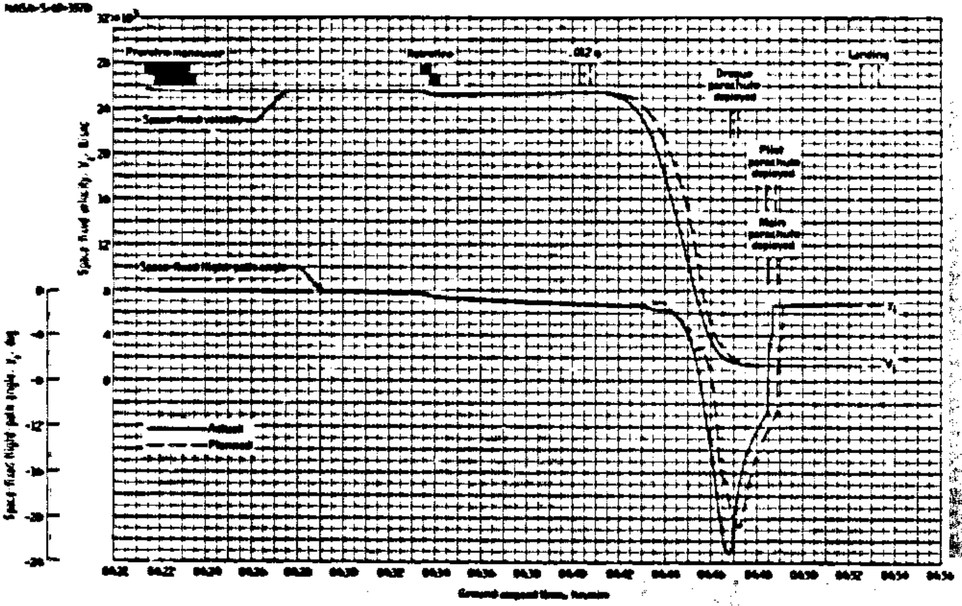
(b) Space-fixed velocity and flight-path angle.

Figure 4-4 - Concluded.



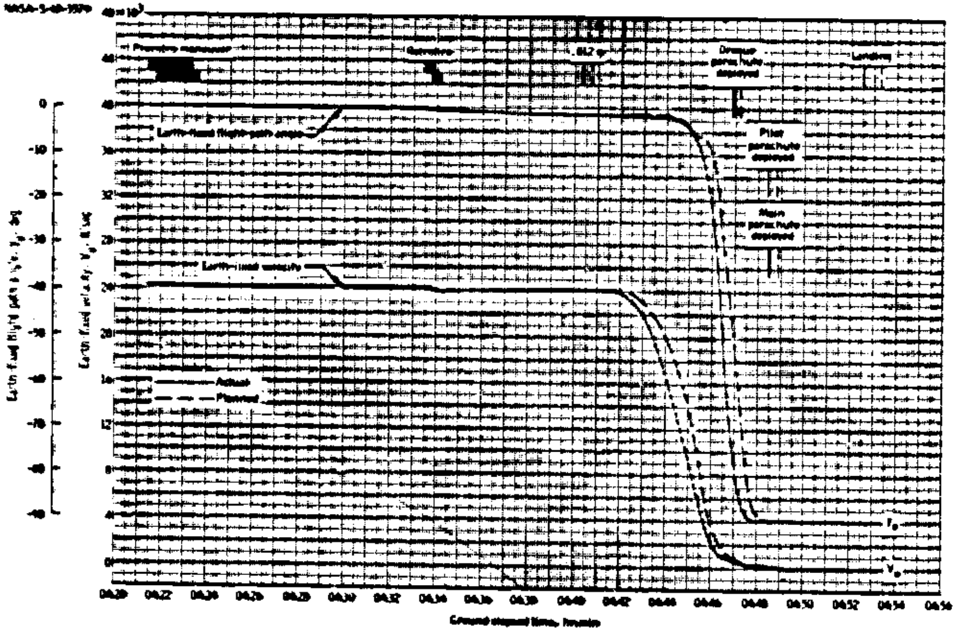
(b) Latitude, longitude, and altitude.

Figure 4-5 - Time histories of trajectory parameters for GT-5 mission reentry phase.



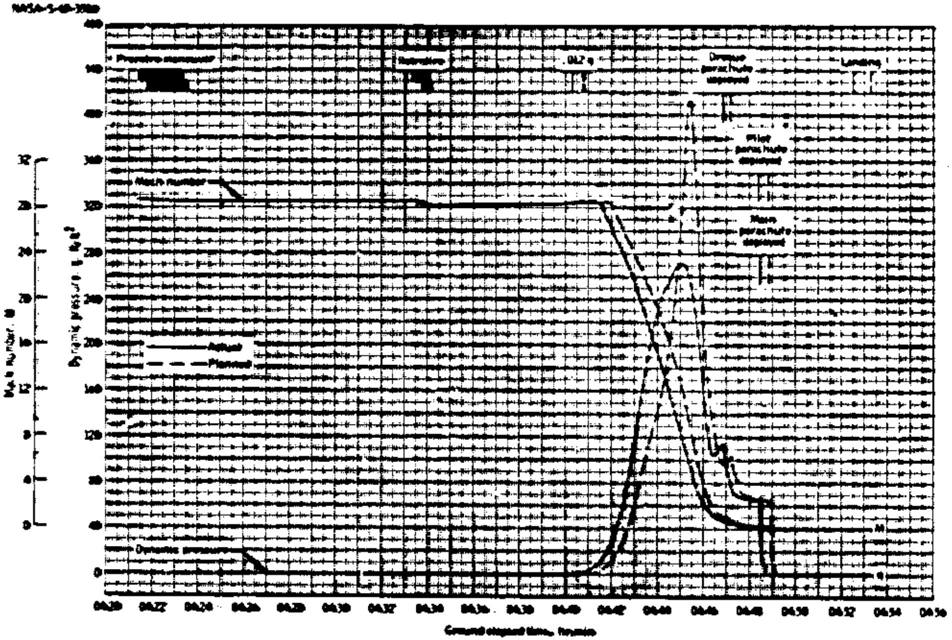
(b) Space-fixed velocity and flight-path angle.

Figure 4-5 - Continued.



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

Figure 4-9.- Continued.



(d) Dynamic pressure and Mach number.

Figure 4-5 – Continued.

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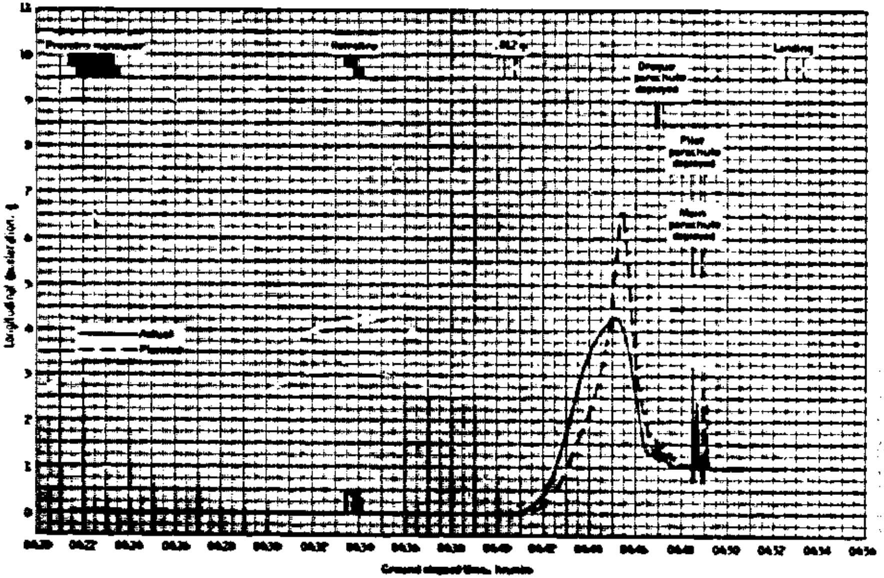


(b) Longitudinal deceleration.

Figure 6-5.- Concluded.

5.0 VEHICLE PERFORMANCE

GT-3 was the first Gemini mission to evaluate the man-machine combination of the spacecraft and its two-man crew. It proved to be a very successful test. The value of the man to a successful mission was demonstrated early in the flight when the primary dc-to-dc converter failed. Quick assessment of the situation located the difficulty, the secondary system was activated, and early termination of the mission or loss of valuable data was averted. A good evaluation of the compatibility of the crew and the crew compartment was made. Important information needed for missions to follow was derived from this test. This section contains an analysis of the data received.

The Gemini launch vehicle performed exceptionally well during the launch phase of the mission, placing the spacecraft into an orbital path that was near nominal.

Details of spacecraft performance are contained in section 5.1, and details of performance of the launch vehicle are contained in section 5.2 of this report.

5.1 SPACECRAFT PERFORMANCE

5.1.1 Spacecraft Structure

5.1.1.1 General.- Having been proven flightworthy by two unmanned flights, the spacecraft structure withstood the environment and loading during the first manned mission without an anomaly. By all indications, the launch loading for the GT-3 mission was approximately the same as for the two previous flights, which was a small fraction of the strength capability, and the reentry heating rate was considerably less severe than that experienced by GT-2, although the total amount of heat sustained was somewhat higher.

Structural instrumentation was considerably reduced for this flight, consisting of only seven thermocouples used to verify reentry heating predictions. These measurements are treated in detail in section 5.1.1.2.

An apparent reentry aerodynamic anomaly which contributed to the inability to achieve the target landing point is treated in section 5.1.1.3.

Finally, consideration of minor mechanical problems exposed by pilot debriefings and vehicle postinspection conclude the spacecraft structural section. (Section 5.1.1.4.)

5.1.1.2 Thermal environment.- The thermal environment encountered by GT-3 spacecraft has been examined for the launch, orbital, and reentry phases of the mission. This evaluation is based on calculated heating conditions derived from trajectory parameters and temperatures measured at the locations shown in figures 5.1-1 to 5.1-3.

5.1.1.2.1 Launch environment: The launch trajectory flown on GT-3 was similar to that of GT-2, and measured temperatures on the two flights were comparable. The maximum measured launch temperature of 480° F recorded by sensor PD04 located at Z163.4 on the bottom centerline occurred at 125 seconds after launch (section 5.1.3). Sensor PD08 recorded erratically, and the data are considered questionable. (See section 5.1.3.)

5.1.1.2.2 Orbit environment: The launch azimuth and time of lift-off resulted in an orbit which, for heating purposes, may be defined as having a geocentric angle between the sun and the spacecraft at the noon orbital position (β) of 29°. Noon orbital position is that point in orbit where the spacecraft is nearest the sun. This angle and the orbit altitude uniquely define conditions for radiant heating of the spacecraft. With the GT-3 spacecraft in a small-end-forward attitude, the β angle realized resulted in the upper right side being the hottest area on the spacecraft. The coldest region of the spacecraft was on the upper left side.

Because the GT-3 instrumentation was installed primarily for reentry, orbital maximum and minimum temperatures on the two sides of the spacecraft were not recorded. Sensor PD08 was again erratic during the orbital phase of the mission, hence the temperature measured on the top side is questionable. Peak temperatures of approximately 130° F, 90° F, and 90° F were recorded on the bottom of the cabin, reentry control system (RCS), and the rendezvous and recovery (R and R) sections, respectively.

5.1.1.2.3 Reentry environment: The GT-3 spacecraft reentry resulted in a maximum zero angle-of-attack stagnation-point heating rate of 49.8 Btu/ft^2-sec and total stagnation-point heating of 8650 Btu/ft^2 which compare closely with preflight predictions. As expected for the orbit altitude and eccentricity planned and flown by GT-3, the heating was considerably lower than the design values of 70 Btu/ft^2-sec maximum heating rate and 14 500 Btu/ft^2 maximum total heat. The design heating rate will only be achieved in certain abort reentries from launch. This was demonstrated by GT-2 which attained a peak rate of 71.8 Btu/ft^2-sec, although the total heat was only 6620 Btu/ft^2. The design total heat would be achieved in a reentry from within 60° of perigee in an 87 to 160-nautical mile elliptical orbit.

As a result of high local heating behind the most windward spacecraft-adapter interconnect fairing on GT-2, the center-of-gravity offset on the GT-3 spacecraft was reduced by 0.52 inch to decrease the trim angle of attack during peak heating. The angle of attack for GT-3 was 8° during maximum heating (Mach number of 20.0) as compared with a predicted value of 11.5° for this time. (See section 5.1.1.3.) This calculated angle of attack agrees somewhat with a visual clue provided by the apparent stagnation point which was displaced less than 12 inches from the center of the heat shield.

As expected, GT-3 afterbody temperatures were much lower than those on GT-2, and no areas of high local heating were encountered. The peak temperature recorded on the cabin along the bottom centerline was 910° F at sensor P006 as compared with a predicted value of 1090° F. Because the actual angle of attack was evidently lower than planned, the prediction would be expected to exceed the measured value. Reentry cabin section temperature histories are shown in figure 5.1-1.

The peak temperature of 740° F measured on the spacecraft beryllium sections by sensor P003 on the RCS section bottom centerline is also lower than the predicted value of 850° F because actual angle of attack was lower than planned. Reentry temperature histories on the beryllium sections are shown in figure 5.1-2.

Bondline temperatures on the heat shield were well below those of GT-2 because the total heat was only slightly higher, whereas the shield thickness was approximately doubled: 0.54 in. to 1.0 in. on the windward side, and 0.39 in. to 0.85 in. on the leeward side. The bondline experienced maximum temperature at touchdown on both flights. GT-2 measured 225° F and 185° F on the windward and leeward sides, respectively, while GT-3 measured 175° F and 155° F at the same locations. The ablation material of the GT-2 shield was reduced in thickness so that the low total heat anticipated would still produce bondline temperatures near the design value of 580° F. The shield performed better than expected in both flights which has resulted in a consideration to reduce the thickness for later flights. Transient heat-shield bondline temperatures are presented in figure 5.1-3.

Reentry heating rates on the spacecraft reentry assembly, computed from the outer skin temperatures, are low, as expected, with a maximum value of 5.65 Btu/ft^2-sec being obtained on the RCS section.

5.1.1.2.4 Overall performance of structure and heat protection: Postflight inspection of the spacecraft indicates that all structure and heat protection went through the mission in excellent condition. There is no evidence of inadequacies in this system. All afterbody shingles are clean and undamaged, whereas underlying insulation is only slightly discolored. No local heating problems were encountered behind the most windward spacecraft-adapter interconnect fairing. All three fairings remained securely attached to the spacecraft outer moldline.

A preliminary examination of the heat shield indicates that char depth (0.20 to 0.25 in.) was greater than on GT-2, although the surface condition reflects the lower heating rates encountered during the GT-3 reentry. The surface has a white appearance over the entire shield except on the most windward edge. This appearance resulted from a silicone dioxide coating deposited on the shield during the ablation process when the surface temperature is below approximately 3000° F. During the GT-2 mission, the combination of higher surface temperature and higher shear stress prevented this coating from being deposited except on the cooler leeward region. Where the coating is not present, the surface is grayish black in appearance. The postflight condition of the ablation shield is excellent.

A single unexpected occurrence during reentry heating was reported by the pilots. A thin foreign deposit, or coating, on the outer surface of both windows is reported to have burned off during the latter portion of the reentry heat pulse. This action occurred over an extremely short time interval and in no way interfered with overall visibility of the pilots. Evidence of such a coating was found after the flight. The investigation of this phenomenon is continuing.

5.1.1.3 Reentry aerodynamics.- The reentry trim angle of attack computed from GT-3 data was approximately 4° lower than that predicted throughout the high Mach number range (fig. 5.1-4). The trim angle varied from 3.5° lower than the predicted 12.5° at a Mach number of 24, to 4.0° lower than the predicted 11° at a Mach number of 16, to 4.0° lower than the predicted 9° at a Mach number of 10. Maneuverability was reduced approximately 33 percent by this lower angle of attack. The inability of spacecraft 3 to achieve its desired landing point may be partially attributed to the lower angle of attack. This implies that either the planned center-of-gravity (c.g.) offset was not attained during reentry, or there is a deviation in wind-tunnel data or the interpretation of it.

The planned reentry c.g. offset for spacecraft 3 was 1.44 inches below the axial axis. It was believed that this would produce a trim angle of attack of 11° at a Mach number of 16. Spacecraft 2 had an offset of 1.96 inches which produced a trim angle of 15° at a Mach number of 16. Spacecraft 3 flight data indicate the trim angle was 7° at a Mach number of 16, which implies that if the predictions from wind-tunnel data are correct, the reentry c.g. offset would have to have been about 1.0 inch.

To verify that there had not been a gross error in the preflight weight analysis, a postflight weight analysis was performed by weighing the spacecraft again and accounting for all missing equipment. This analysis resulted in an offset of 1.46 inches, confirming preflight records. It was thought that perhaps the flight crew were riding high in their seats; however, an analysis indicates both pilots would have had to have been an improbable 5 inches high in their seats to account for a 0.42-inch upward movement of the c.g. Another factor that contributed to decreasing the c.g. offset was the reentry communications experiment which ejected 15 pounds of water from the right-hand landing-gear compartment during reentry. However, the water was ejected at intervals throughout reentry, and the total effect would account for a reduction in trim angle of no more than about $1\frac{1}{2}^\circ$. Therefore, there appears to be some difference between the actual and wind-tunnel-derived aerodynamics of the reentry configuration.

The reentry aerodynamics for Gemini were produced in May 1962, in McDonnell Aircraft Corporation (MAC) Aerodynamic Impulse Tunnel, and is known as series I data. The state-of-the-art of instrumentation at that time was inadequate for defining test Reynolds numbers in that type of facility. Consequently, for prediction purposes, the data were considered to vary only with Mach number. However, the spectrum of Gemini design reentries involves varying Reynolds number conditions, Mach number of 15 occurring at Reynolds numbers between 0.5×10^6 and 2×10^6, so that some concern was given to possible Reynolds number effects.

Late in 1964, after improved data-acquisition techniques were initiated, MAC investigated the Gemini configuration at a Mach number of 15 for Reynolds number effects and concluded that the trim angle was definitely affected by Reynolds number, and that there were indications of two trim points about 5° apart for low Reynolds numbers. These data were extrapolated for higher Mach numbers and are known as series II data. This small uncertainty in the aerodynamics was not considered to be a problem for the unmanned GT-2; consequently, it was flown, and yielded the first flight data on reentry aerodynamics.

The GT-2 flight data tended to substantiate the series I tunnel data and corresponded to trim predictions to within 1° at a Mach number of 16. Therefore, for GT-3, predictions were based upon series I data modified by the GT-2 flight-data point.

These various data are compared in figure 5.1-4. From the afterbody heating standpoint, there is not much concern with the dispersion in the data because the series I data, confirmed by the GT-2 flight, gave the highest trim angles and were used as the design criteria. The problem is that the expected maneuverability was not achieved. However, even with the low angle and low orbit of GT-3, analysis shows a downrange capability of 195 nautical miles and crossrange capability of 37 nautical miles. Maneuverability will be greater for reentry from higher orbit altitudes. Although the maneuvering capability of the spacecraft was less than anticipated, the magnitude associated with existing aerodynamics should be adequate for future flights.

5.1.1.4 General structural and mechanical problems.- Two anomalies in the structural and mechanical area were mentioned by the pilots during their technical debriefing. They noted that during the maximum vibration period of launch, the nose fairing tabs over the docking fitting receptacles vibrated rather violently, and that the command pilot could not open the hatch after water landing.

It was mentioned that the tab was vibrating with about a $\frac{1}{4}$-inch amplitude when the vehicle approached sonic velocity. This tab is a fiber-glass part, 0.05-inch thick by 4 inches wide and 5 inches long, which is riveted at the forward end to the nose fairing. Although the bottom part of the tab lies flat against the beryllium shingle surface, because of the thin shingles on the upper surface, the two upper tabs, which the flight crew saw stand off of the shingle surface about $\frac{1}{8}$ inch. From a stress standpoint, there is no problem because the $\frac{1}{4}$-inch double-amplitude deflection represents only 25 percent of the strength of the material. With the tabs' natural frequency of 600 cycles per second and assuming vibration for the duration of the fluctuating pressure period, the number of cycles for the low stress level is insignificant from a fatigue standpoint. Therefore, it is concluded that, although these tabs do vibrate, their strength is adequate; however, an improvement change is being considered either to make the tabs lie flat against the shingles, or perhaps to eliminate them.

The command pilot indicated that he began to operate the hatch opening mechanism after landing. He terminated this attempt when he observed the pararescue personnel ready to open the hatch from the outside. This report has given rise to some concern that the mechanism would not function as required. Several conditions were explored during postflight inspection in an attempt to create a jammed mechanism without success. It was thought that perhaps a strong pull on the handle with the pawls improperly set might jam the mechanism so that it would not function even with correct pawl settings; however, this could not be done with reasonable forces applied. From this investigation it is apparent that the hatch operating mechanism is satisfactory.

[redacted]

Several items were reported in the postflight inspection (sec. 12.6) which appear to indicate structural discrepancies. The crushing of the phenolic ring at station 2104 behind the hoist loop resulted from the hoist loop fitting bearing against the ring during retrieval of the spacecraft from the water. This is normal, and some damage to the ring was anticipated. The washers and screws missing from the strip forward of the right-hand hatch window were removed by recovery personnel during RCS deactivation and inadvertently were not replaced. This has been confirmed by those personnel involved. The loose screws reported on the C-band antenna and elsewhere were actually not loose at all. This is a part of the afterbody shingle retention system where spacers under the head of the screws permit the large washers to rotate freely, thus giving the impression of loose screws. The damage to a filler strip on a landing-gear door either resulted from water impact at landing or impact sustained during late recovery operations. Either is considered possible and acceptable for this portion of the spacecraft. Later inspection of the C-band antenna, reported to be recessed, recorded it to be within acceptable installation tolerances. Local corrosion of components, such as the electrical plug reported in section 12.6, is to be expected after being exposed to salt water. Also the local hot spot on the umbilical is a normal indication of reentry heating.

5.1.2 Communications

Performance of the communications system was nominal and satisfactory with the possible exception of the high-frequency (HF) voice communications discussed in a succeeding paragraph. Data sources for this evaluation were the flight crew debriefing, logs and recorder charts from Air Force Eastern Test Range (AFETR) stations, recovery report, incomplete PCM telemetry tabulations, voice tape transcriptions, and other miscellaneous sources. Data from other Manned Space Flight Network stations were unavailable.

The time interval for communications blackout due to reentry plasma was determined from the following data:

Recorder charts

Station	Frequency, mc	Loss of signal, G.m.t.	Acquisition of signal, G.m.t.
GBI	230.4	19:04:00	19:08:58
GTI	230.4	--	19:09:00
CNV Tel II	230.4	19:03:45	--
CNV Tel III	230.4	19:03:52	--

Aircraft logs

Station	Frequency, mc	Loss of signal, G.m.t.	Acquisition of signal, G.m.t.
Silver 1	230.4	--	19:09:01
Silver 2	230.4	19:02:25	--
Silver 3	230.4	--	19:09:00
Silver 6	230.4	--	19:08:58

Therefore, the communications blackout time interval was approximately 19:04:00 G.m.t. to 19:09:00 G.m.t.

5.1.2.1 Voice communications.- The flight crew expressed satisfaction with UHF voice communications which were used during all phases of the mission. Postlanding exercise of HF equipment in both voice and

[redacted]

high frequency-direction finding (HF-DF) modes was a failure. The flight crew received no reply to their HF voice transmission, and it cannot be established with certainty that the signals received by supporting stations, as discussed in section 6.4.3, actually were originated by the spacecraft. Postflight tests of the HF system thus far have revealed that the power output of the transmitter was normal, the dc power cabling was normal, and the RF cabling from transmitter-receiver to antenna was normal. Remaining to be tested are the voice control center (VCC) and the cabling between the VCC and the transmitter-receiver. The antenna is discussed in the following section. [redacted]

No HF voice was transmitted in orbit from the spacecraft; however, there was considerable ground activity on this frequency based on charts showing signal strength of about the same magnitude as those recorded after the spacecraft landed and was supposedly transmitting HF. The same interference was present on the GT-2 mission charts. It may be deduced that spacecraft HF transmission cannot be positively identified on the signal strength charts unless some identification code is included. HF-DF stations reported bearings which indicate that they were receiving signals originating from locations other than the landing area. [redacted]

[redacted] Voice recordings from Mission Control Center (MCC)-Cape Kennedy were reviewed. UHF air-to-ground voice intelligibility was good even during the launch phase, and the flight crew reported good ground-to-air intelligibility. As a matter of preference, the push-to-talk mode was used by the flight crew throughout the mission. No squelch control and very few volume control adjustments were necessary. The flight crew reported that they could detect in their earphones when the recovery beacon and also the HF-DF stations were turned on. [redacted]

5.1.2.2 Antenna and multiplex systems.- Antenna and multiplex systems operated normally, as nearly as can be determined from available data, with the exception of the HF antenna. The flight crew reported that the HF antenna extended normally after landing but would not retract. Preliminary failure analysis indicates that the extend limit switch actuator failed. This resulted in a broken pin in the drive mechanism when the motor attempted to continue running in the extend direction. This drive pin failure would prevent retraction. Also, as in the GT-2 mission, transmission of HF in voice and in HF-DF mode was very poor, if at all. This discrepancy could have been caused by leaks in the antenna or failure of other components, as discussed in paragraph 5.1.2.1. As reported in section 6.4.3, pararescuemen stated that the HF antenna bent and fell over 2 hours after landing. These failures are under investigation at the present time.

Upon examination at Cape Kennedy, the HF antenna was found to contain 160 grams of salty water. It has not been positively identified as sea water. The antenna was bent over and tied down during recovery operations. This destroyed the extendable rubber boot seal and the case seal through which the radiating element extends, and caused other damage inside the case; therefore, it cannot be determined whether the seals leaked or the water entered after the damage occurred.

5.1.2.3 Radar transponders.- Performance of both C- and S-band radar transponders was normal insofar as can be determined. The available data reviewed consisted of the network controller report, station log sheets from Patrick Air Force Base (PAFB), Merritt Island Launch Area (MILA), Grand Bahama Island (GBI), and Antigua (ANT), and recorder charts from PAFB, GBI, and Grand Turk Island (GTI).

Low-frequency, low-amplitude signals were superimposed on the automatic gain control (AGC) and servo-loop error recordings during launch and reentry when the helix antenna system was used. This was expected, and was caused by beat notes between harmonics of the radar pulse repetition frequency (PRF), 160 cps, and the phase shifter excitation frequency, 453 cps. The frequency of the beat notes was high enough not to cause errors. The phase-shifter inverter frequency was previously chosen as a compromise between available equipment and harmonics of the 80 or 160 cps PRF or the other option of 71 or 142 cps PRF. Several unaccountable amplitude variations with a duration of 5 to 30 seconds appeared on recorder charts. Servo-error variations were 1 to 3 mils peak-to-peak. No tracking errors were evident from these variations.

Tracking station radar operators' logs contained notations concerning C-band radar beacon track at intervals during the normal communications reentry ion sheath blackout as follows:

Station	Acquisition of signal, G.m.t.	Loss of signal, G.m.t.
MILA	19:05:59	19:08:10
PAFB	19:04:53	19:07:33
GTI	19:08:45	19:12:20
GBI	19:07:20	19:08:49

Actual blackout time for communications was approximately 19:04:00 to 19:09:00 G.m.t. as reported in paragraph 5.1.2. The beacon track was verified on the GTI station recording. The skin track AGC, which was acquired at 19:07:36 G.m.t., was about 20 dB and tailed-off just before switch to beacon track. Beacon track AGC was initially about 35 dB and increased to 50 dB. There was a noise on the skin track and phase shifter interference on the beacon track AGC. MILA radar was programed to skin-track the spacecraft during the second orbit but failed to acquire. It then switched to beacon track and still did not acquire. [redacted]

5.1.2.4 Digital command system.- The performance of the digital command system (DCS) was evaluated by reviewing data consisting of (1) PCM tabulations of eight parameters for the ascent and the third orbit of the flight, (2) the network controller reports, (3) the flight crew debriefing, and (4) status reports from remote site flight controllers. Telemetry readings of system voltages, package temperature, and received signal strengths were nominal in value during the ascent phase of the mission. Remote site reports indicated, in general, satisfactory orbital DCS uplink performance. There were two instances of possible abnormal DCS operation. The uplink of C- and S-band real-time commands by the Coastal Sentry Quebec (CSQ) during the second orbit had no ground recording of a message acceptance pulse. CSQ also reported local interference and 14 telemetry dropouts during the second orbit. These problems may have been the result of tracking errors caused by the ship's roll or other reasons unknown at this time. DCS operation was successful at the next station, Hawaii. It also could not be determined if the spacecraft was properly acquired by CSQ at the time the commands were transmitted. The second possible abnormal DCS operation was evidenced by fluctuations in the telemetered signal strength of the DCS receiver connected to the diplexed antenna while the signal strength of the second receiver connected to the quadriplexed antenna remained nearly constant. The radiation patterns for the diplexed and quadriplexed antennas should be nearly the same, because they are similar elements both mounted on the lower centerline of the adapter assembly. [redacted]

These fluctuations did not appear to be of a magnitude that would have prevented proper operation of the receiver and, with the data taken on this flight, it is impossible to determine whether the redundant receivers both functioned or not. It is also significant to note that these two signal strengths were measured on a telemetry channel that was known to have a cycle variation throughout the flight. (See section 5.1.3.) These receivers are mounted in the adapter along with both of their antennas, therefore, there cannot be a postflight evaluation of this possible anomaly.

Performance of the spacecraft on-board digital command system appears to have been excellent, and it is very probable that these two instances were associated with the ground installations or telemetry. The flight crew verified computer and time reference system messages during the mission and expressed confidence in the proper reception of commands.

[redacted]

5.1.2.5 Telemetry transmitters and acquisition aid beacon.- Performance of the telemetry transmitters and acquisition-aid beacon appeared normal and satisfactory as nearly as could be determined from available data. The delayed-time telemetry transmitter failed during prelaunch testing, and it was decided to use the spare transmitter for flight on this 3-orbit mission instead of removing the adapter to replace the failed transmitter.

[redacted]

The data reviewed consisted of PCM tabulations for launch and reentry, aircraft operator's logs, and signal strength recorder charts from the Air Force Eastern Test Range (AFETR) stations. Data from other stations were unavailable at the time of publication of this report.

[redacted]

As indicated in paragraph 3.1.2, reentry blackout began about 04:40:00 g.e.t. and ended about 04:45:00 g.e.t. Signals were also lost from about 04:47:00 g.e.t. to 04:47:25 g.e.t. due to loss of a reentry antenna at R and R section separation until deployment of the descent antenna. Because the adapter delayed-time transmitter was not used for this mission, the only transmission at 246.3 Mc was the acquisition-aid beacon. The standby transmitter at 259.7 Mc was used for delayed-time (recorded) transmission. The performance of the acquisition-aid beacon was good. Acquisition was achieved with signal strengths of 4 to 6 microvolts, whereas, the normal signal was 10 to 20 microvolts.

5.1.2.6 Recovery aids.- Normal extension and operation of the recovery flashing light were reported by recovery forces. Normal operation of the UHF recovery beacon was reported by recovery aircraft which received in CW and pulse mode at distances up to 107 nautical miles.

5.1.3 Instrumentation and Recording System

An analysis of the data available reveals that the instrumentation system essentially performed within specifications. When the primary instrumentation dc-to-dc converter failed early in the flight, the pilots switched to the standby converter, and mission success was not jeopardized. The data losses caused by the primary dc-to-dc converter malfunction were obtained from the playback of the PCM tape recorded data for the first orbit and are shown in the following table:

Time from lift-off, hr:min:sec, g.e.t.		dc-to-dc converter inoperative time, min:sec
Instrumentation system		
Data loss	Data acquisition	
00:08:12	00:08:38	00:26
00:10:43	00:11:07	00:24
00:11:36	00:11:41	00:05
00:20:30	00:20:37	00:07
00:20:59	00:24:20	03:21
00:24:56	00:25:07	00:11
Total loss		04:34

For periods of 26, 24, 5, and 7 seconds, and for one longer period of 2 minutes 36 seconds, the instrumentation system was not operating properly before the pilots detected a malfunction at 00:23:35 g.e.t. The pilot required only 45 seconds to analyze the problem and switch to the standby dc-to-dc converter at 00:24:20 g.e.t. As a result, the instrumentation system was inoperative during this period for 3 minutes 21 seconds. After observing proper operation of the cabin displays when the standby dc-to-dc converter was turned on, the pilot switched back to the primary dc-to-dc converter at 00:24:46 g.e.t. Because there was a continued malfunction of the primary converter, the pilot returned to the standby converter at 00:25:07 g.e.t. and elected to remain in that configuration for the rest of the mission. Since the PCM tape recorder continued to run through the malfunction period, the +24 V dc output of the converter did not fail. The instrumentation system sensors and displays powered by the +5 V dc output were malfunctioning, as well as the PCM system (powered by the +24 V dc output). The PCM system failed to encode or to time the parameters whose excitation was external to the instrumentation system dc-to-dc converter. As a result of this failure, an inoperative time period totaling 4 minutes 34 seconds was realized during which the instrumentation system operated improperly. The primary converter was sent to the spacecraft contractor for failure analysis which revealed that the improper use of a locking device (externally serrated lockwasher) allowed a nut on the radio frequency interference (RFI) filter (condenser) to back off and short the +24 V dc line to the case ground. Because the +5 V dc output is derived from the +24 V dc circuitry, both outputs failed. The use of epoxy as a bonding agent to lock all of this type nuts in place on the dc-to-dc converters and the biomedical power supplies has been initiated and all flight units will be retrofitted.

A step function cyclic variation in parameters from the PCM low-level adapter multiplexer was noted during prelaunch checkout of the spacecraft, and it was determined that interference from the rate gyros was causing the problem. During prelaunch testing, the higher reading points were proven to be valid values and the low values were shown to be invalid due to the rate gyro interference. Because these parameters would be showing slowly varying temperatures and pressures and satisfactory data could be recovered, it was decided to fly with the problem. During this mission, this variation was as high as 10 percent of full scale. It is not known whether the trouble was caused by noise and ground loops in the spacecraft or by the adapter PCM low-level multiplexer. However, it is thought that this problem was peculiar to the spacecraft 3 combination of shielding and grounding existing at launch. A deliberate ground loop had to be made, involving this multiplexer, in order to obtain repeatable operation. Shielding and grounding on future spacecraft has been redesigned, therefore this problem should not be experienced on future spacecraft.

Real-time PCM biomedical data were considered excellent by the aeromedical monitors. The biomedical information recorded on the special biomedical recorders was also very good. Discussion and comparison of both the real-time PCM biomedical data and the onboard recorded biomedical data are found in section 7.2.

From the available data, it has not been possible to correlate signal strength with synchronization losses from the delayed-time PCM data. Therefore, proper operation of the PCM tape recorder during the transmit periods cannot be determined.

The outer skin temperature parameter, PD08, was intermittent prior to launch, and although it changed from a non-valid value up to a reasonable value at approximately +150 seconds after lift-off, the data obtained are considered to be questionable. All other parameters, 246 in number, operated well within specification, except the 32 on the adapter low-level multiplexer which had the cyclic variation. This variation did not affect the ability to recover the data and had been declared satisfactory for flight prior to launch. The data shown in table 5.1-I were obtained from an evaluation of the real-time data processed from Tel II, MCC, GBI, and aircraft 629 and 630.

It can be seen that the ratio of valid data to available data is over 90 percent for all stations except MCC during launch and the aircraft. The 63.3 percent value for MCC was due to antenna tracking problems.

A summary is presented in table 5.1-II of the delayed-time PCM data obtained by dumping of orbits 1 and 2 data at Tel II and GBI, and by recovery of the onboard PCM recorder. These data indicate that large losses of data occurred in the data dumps. From the processed data available, it was not possible to separate the RF link losses from those losses peculiar to a "bit jitter-PCM dump format" problem. Recovery of the third orbit and reentry data from the onboard recorder in the normal method resulted in 94.85 percent usable data recovery. The contractor is in the process of employing the internal diphase recorded signal to eliminate the bit jitter and provide positively synchronized data processing.

Because of turning on the real-time transmitter with the first dump already started, instead of simultaneously, a data gap of 2 minutes 1 second occurred where only a 20-second gap was expected because of tape reversal. Similarly, gaps of 1 minute 9 seconds at the start, and 2 minutes 3 seconds at the end of the second dump also occurred. Procedures to insure that the real-time transmission is started 20 seconds prior to beginning delayed-time (D-T) transmission and turning off the real-time transmitter no sooner than 20 seconds after D-T transmission is complete will hold data losses to a minimum.

5.1.4 Environmental Control System

The environmental control system (ECS) functioned normally throughout the mission, with all parameters indicating as expected, except for the radiator outlet temperature. The suit module maintained satisfactory comfort for the crew throughout launch, orbit, and reentry. The crew stated in section 7.1.2 that during periods of high work loads the flow rate through the pressure suits resulted in marginal cooling. Spacecraft 4 and all future spacecraft have been modified to allow simultaneous operation of both suit compressors to provide augmented cooling during these high work-load periods. Other variations from anticipated performance occurred in the primary O_2 pressure, a noticeable thrust caused by steam venting from the launch-cooling heat changer, and radiator outlet temperature.

5.1.4.1 Primary oxygen pressure.- The GT-3 ECS primary oxygen vessel was off-loaded to 70 percent of capacity in an effort to duplicate the decline in pressure experienced on GT-2 in the reactant supply system. During the prelaunch period, the vessel pressure was raised to 443 psia at T-164 minutes by heat transfer with the vessel in ambient conditions. Two short periods of manual heater operation raised the vessel pressure to approximately 900 psia at T-150 minutes. The ambient heat transfer then increased the vessel pressure to 1000 psia (indicator limit and relief-valve actuation level) just prior to lift-off. The indicated mass quantity was approximately 66 percent at lift-off. There was no indication of pressure decline during launch; both quantity and pressure indications remained stable at prelaunch values. The 1000-psia pressure continued throughout the mission, except during two brief periods of O_2 high-rate operation initiated by the flight crew at mission times of approximately 1 hour 14 minutes and 2 hours 41 minutes. During both periods of high-rate consumption the vessel pressure declined, as expected, but recovered and slowly rose to 1000 psia after high-rate valve re-cock.

During the flight, when vessel pressure rose to 1000 psia after the first O_2 high-rate usage, it was thought that the automatic heater control which maintains the vessel pressure between 800 and 910 psia had failed in the "on" position. The crew was then directed to turn the automatic heater off. The increase in vessel pressure after the second period of O_2 high-rate usage indicated that the combination of heat transfer into the vessel and usage from the vessel resulted in an increase in pressure. Heat transfer was near the maximum specification, and usage was low as a result of the low cabin-leakage rate. A computer analysis using the GT-3 vessel heat-transfer characteristics and estimated oxygen usage verified these conditions when compared with the flight data.

5.1.4.2 Launch-cooling heat-exchanger thrust.- The spacecraft experienced a continuous yaw-left condition for approximately the first 44 minutes of flight. This condition was apparently caused by overboard venting of steam from the launch-cooling heat exchanger. The launch-cooling heat exchanger is located in the adapter in such a position that it yields a yaw-left thrust vector as a result of any venting. The calculated thrust values of the overboard venting during the first 44 minutes of flight vary from 0.08 to 0.10 pound. These values are in agreement with the thrust required to produce the measured yaw-left acceleration.

The launch-cooling heat exchanger also operated for short periods of time during the daylight phase of each orbit. The calculated thrust value for these periods was 0.018 pound.

5.1.4.3 Radiator performance.- The thermal qualification test spacecraft (3A) testing showed that the radiator capacity was greater than required. Therefore, the adapter was striped for a resultant maximum radiator outlet temperature of 35° F. Radiator outlet temperature cycled to a maximum of 50° F during the hottest portion of the orbit, indicating a slight excess of striping.

5.1.5 Guidance and Control System

5.1.5.1 Summary.- The guidance and control (G and C) system functioned adequately throughout the flight except for two anomalies which should be corrected prior to the next flight. Table 5.1-III contains a time sequence of test events together with an indication of the function of each component during the tests. The two anomalies recorded, neither of which affected pilot safety or mission objectives, were:

(a) An excessive ascent navigational error (greater than planned 3σ limits) was evidenced in comparisons of the inertial guidance system (IGS) computations of position and velocity with those calculated postflight from ground tracking data. However, the IGS provided adequate backup to the primary stage I guidance. If the primary stage II guidance had exceeded acceptable limits and the crew had selected secondary guidance, an orbit within the planned operational "go-no-go" limits would have been achieved.

(b) An excessive number of horizon sensor loss-of-track indications reported by the pilots.

A series of circumstances prevented landing at the planned touchdown point; however, it has been determined that the IGS performance was not limiting and the G and C system demonstrated the capability of controlling to a planned touchdown point within the circular error probability (CEP) of 5 nautical miles.

5.1.5.2 IGS performance evaluation.-

5.1.5.2.1 Ascent phase: Figure 5.1-5 shows the comparison between the actual IGS steering commands (difference between measured attitude and computed desired attitude) and the preflight calculated upper and lower limit values. When flight values exceed these limits, detailed investigation is required to determine the effect on the mission; performance within the limits indicates acceptable performance. These comparisons show that the yaw IGS commands to the Gemini launch vehicle (GLV) were well within the limits of acceptable performance, however, the pitch commands exceeded the lower limit at approximately 250 seconds after lift-off and increased to a saturated value at 290 seconds. This is discussed in section 5.1.5.4. The initial roll command offset was caused by the GLV first stage engine offset.

Plots of the IGS roll, pitch, and yaw attitude errors for the ascent phase of the flight are shown in figure 5.1-6. Superimposed on these plots are the outputs of the primary guidance system (radio guidance system (RGS) and three-axis reference system (TARS)) adapter which are equivalent signals. A discussion of the difference between the attitude errors of the two guidance systems and possible explanations for their deviation follows.

(a) Programed flight (stage 1): Roll attitude error. Again, as observed on GT-2, the primary guidance system roll attitude error reflects a thrust misalinement in the launch vehicle engines. This misalinement requires a constant attitude of 0.8° to 1.0° to offset and is slightly larger than seen on GT-2. The most outstanding observation from the roll error is the difference between the IGS and the primary guidance system. Some of the error could be attributed to gimbal cross-coupling as a function of the yaw wind shear and/or of yaw drift that will be discussed in a subsequent section. However, this could account for only a small portion of the total error. Therefore, most of the difference is probably a function of roll drift of about 50 deg/hr between the two systems. From a velocity comparison it is believed that this drift is not induced by the IGS.

Yaw attitude error. The difference between the IGS and the primary guidance system yaw-attitude error reflects a yaw drift of about 20 deg/hr. Again, an analysis tends to indicate that this is not an IGS drift error. It is significant to note that the out-of-plane velocity at staging as measured by the IGS was about 150 fps in yaw off the predicted value considering launch day winds. (Section 5.2.3.1.)

Pitch attitude error. The primary guidance system pitch attitude error (in the interval before staging) indicates a thrust misalinement requiring an 0.8° offset to null. It is possible that this misalinement was present from lift-off but obscured by winds early in flight. The large deviation at 110 seconds is a function of autopilot gain change. This reaction was somewhat larger than expected, although the steering commands remain within acceptable bounds. Again, as noted in the other two axes, there is an apparent drift between the two systems on the order of 30 deg/hr. However, this difference appears to start at the initiation of the pitch program as if the programed pitch rates were slightly different. The analysis does not indicate any pitch drift for the IGS this early in flight.

Analysis of the inflight telemetry data indicates that both azimuth updates were correctly received. Platform misalinements have been computed to be as follows: (a) +0.010° at platform release, (b) -0.530° at 104.8 seconds, and (c) -0.524° at 145.9 seconds.

It is significant to note that, although the platform azimuth (roll gimbal) orientation relative to the spacecraft at release was established to within 0.01°, a misalinement of 0.525° was computed for launch as determined from the updates. This tends to indicate a buildup of mechanical alinement uncertainties from the launch-pad thrust mount, up through the structure to the stable member. These uncertainties are less than the 3σ values.

(b) Closed-loop steering (stage II): Pitch attitude error. The apparent IGS malfunction at about 200 seconds did not significantly degrade the total velocity magnitude as measured by the IGS; however, it degraded the IGS computation of the inertial flight-path angle and altitude. The degradation of the flight-path angle and altitude is the prime cause of the large pitch-down command as shown. This anomaly is discussed more completely in section 5.1.5.4.1.

Yaw attitude error. The transient at SECO is a normal reaction to the moment created by the roll nozzle on the second stage. Again, as seen on GT-2, the zero yaw error during the closed loop steering reflects that the azimuth updates received were properly executed.

Roll attitude error. Again the difference between the two systems is most probably a roll drift; however, it appears to be of a magnitude of about 30 deg/hr during the second stage of flight. Analysis does not indicate an IGS roll drift during this portion of flight.

If switchover to IGS had occurred on the flight prior to 200 seconds, the SECO conditions would have shown about the following differences from nominal: -19 ft/sec in velocity, -0.30° in flight-path angle, and -9602 ft in altitude.

With IGS steering, these injection conditions would result in lowering perigee approximately 45 000 feet and apogee approximately 10 000 feet (approximately 78 to 128 nautical mile orbit). The insertion velocity adjust routine (IVAR) correction displayed to the crew would indicate that an approximate 17 to 18 ft/sec thrust should be conducted to increase the spacecraft velocity because of the IGS measured difference between nominal "tail-off" and actual GLV "tail-off". Applying this correction would have raised perigee approximately 6000 feet and apogee by 50 000 feet, resulting in a final orbit with an apogee 40 000 feet higher and perigee 40 000 feet lower than that targeted (approximately 80-nautical mile to 137-nautical mile orbit).

Comparison of these conditions with the "go-no-go" criteria at insertion shows that the spacecraft orbit would have resulted in a "go" condition. The first orbit maneuver would have been adjusted to continue the mission as planned.

The incremental velocity indicator (IVI) display as computed by the onboard IVAR program was reconstructed using the IGS navigational and gimbal angle data and is presented in figure 5.1-7. The pilot reported -17 ft/sec for the first reading on the X window of the IVI (approximately LO+354 sec, or SECO + 20 sec). At LO+389 seconds, the reported readings were: X = -29 ft/sec (aft), Y = +7 ft/sec (right), and Z = +3 ft/sec (down). Since the pilot recordings confirm the computed results within reasonable limits, the orbit insertion equations and the computer-IVI interface are considered validated. Had the IVAR correction been executed when the IVI aft reading was 28 ft/sec (at approximately 372 sec) an orbital attitude and maneuver system (OAMS) maneuver to reduce the spacecraft orbital velocity by approximately 18 ft/sec would have been conducted (for the math flow used, the crew procedure was to thrust until the IVI indicated -10 ft/sec, or aft 10 ft/sec). This would have reduced apogee by approximately 9 nautical miles. If the correction at apogee to correct perigee had been executed (at LO+42 min 42 sec) applying +5 ft/sec (forward), perigee would have been raised 3 nautical miles, resulting in an approximate 90 nautical mile to 121 nautical mile orbit which is well within the "go-no-go" insertion conditions.

5.1.5.2.2 Orbital phase: After separation, the IGS provided an attitude reference by way of the attitude display group (ADG) and computed and displayed velocity changes through the IVI. The inertial maneuvering unit (IMU) was alined several times by using the horizon scanner for pitch and roll reference information. During the horizon-scanner test, an apparent excessive drift was reported when the "orbit rate" mode was utilized. This apparent "drift" was due to a misalinement of the platform remaining from the cage and alinement test, coupled with a normal transfer of yaw error to the roll axis over 90° of orbital travel. Table 5.1-IV lists gimbal angles and scanner outputs at four significant times during this period. The first data point, taken when platform caging began (switch to platform mode), indicates that an attempt was made to cage with spacecraft attitudes at the intended 10° off zero in each axis. By the start of alinement (select platform mode small-end forward (SEF)), however, the spacecraft had rolled an additional 15° while in the cage mode as shown by the change in horizon scanner angles. This resulted in the initial conditions at the start of alinement being approximately: -6.4° in pitch, +10.0° in yaw, and +16.4° in roll.

The crew switched from SEF mode to orbit rate mode which terminated alinement after an alinement period of 7 minutes 26 seconds (446 seconds).

Figure 5.1-8 shows the results of a computer simulation of platform alinement for three initial roll offsets. As can be seen, the yaw errors remain large long after the other two axes have approached null. For the alinement time utilized in the cage and aline checks and for the initial conditions listed when the crew switched the platform mode from SEF to orbit rate terminating the alinement torquing, the residual yaw error would have been approximately -12.5°.

With an initial alinement of -12.5° in yaw and pitch and roll alinement errors near zero when orbit rate torquing current is applied to the platform pitch gyro, the pitch gimbal will be rotated in a plane having an angle with respect to the orbit plane equal to the yaw misalinement. The pitch gyro sensitive axis (y-axis), the yaw gyro sensitive axis (z-axis), and the roll gyro sensitive axis (x-axis) form an orthogonal set. When this orthogonal set is rotated +90° about the y-axis (pitch rotation) with a negative yaw misalinement error between the orbit plane and the x-axis, the x-axis rotates into the orbit plane and the z-axis rotates out of the plane of the orbit and reaches an angle equal to the original yaw misalinement angle with the plane of the orbit. Since roll is measured about the x-axis, the z-axis out-of-plane angle appears as a negative roll angle error on the attitude display equal to the original yaw misalinement error. Further, if the orthogonal axes are rotated about the z-axis 90° (yaw 90°), the roll-attitude error returns to zero, and the original yaw misalinement angle appears as a negative pitch error with respect to the orbit plane. Because the pitch gyro torque rate is set approximately equal to the spacecraft orbit angle rate, the local vertical would rotate through 90° at the same time as the pitch gimbal rotates 90°. This indicates that the yaw misalinement error would appear as a pure roll error after traveling 90° around the orbit from the point at which alinement was terminated. At this time the out-of-plane thrust maneuver

[redacted]

was being conducted, and the scanner indicated a pitch-down of 3°. Further, the IGS velocities during the planned 10 fps translation test were 2.7 ft/sec down and 9.3 ft/sec horizontal. These components indicate a platform roll with respect to the spacecraft of 16.2° or a 13.2° misalinement, considering the 3° spacecraft pitch down. This agrees reasonably well with the 12.5° which would have existed at that time, thus substantiating that the system behaved normally. Because of the approximations involved here, however, this problem will be further investigated and a reconstruction attempted. Figure 5.1-8 shows yaw errors remain several minutes after roll and pitch have apparently nulled or are within the ADG readout capability, indicating that it is necessary that the mode continue for the full time allotted.

The slight IVI drift which occurred in flight prior to the maneuver periods was not due to an equipment malfunction. This drift resulted because the computer constants were not changed to the best available preflight test values. This change was not made because it was thought that the error was small and that the use of existing constants would not jeopardize equipment evaluation or mission objectives.

The velocities accrued during the three translation maneuvers, as well as other velocity changes, are shown in table 5.1-V. The table contains the velocity increments actually calculated in the computer along with values corrected for the uncompensated x-accelerometer bias of -0.1226 pulses/second (-3.71×10^{-4}g). This bias error caused the erroneous IVI countup reported by the flight crew prior to both orbit translation maneuvers.

For translation no. 1 the elapsed time from COMP depress to thrust initiate was 230 seconds. The velocity computed due to bias error over this time interval is 2.8 ft/sec which accounts for the change in the X IVI reading from -48 to -51 ft/sec over the same interval and as recorded by the flight crew. Similar values for the OAMS retromaneuver are 136 seconds, 1.7 ft/sec, and an IVI drift from 96 to 98 ft/sec.

5.1.5.2.3 Reentry phase: For the GT-3 mission, the pilot received backup commands from the ground to fly a constant 45° roll left (lift vector at 50° toward the south), reverse the bank angle at 19 hours 8 minutes 17 seconds Greenwich mean time reverse bank angle (GMTRB) (10 minutes 54 seconds from retrofire), and then to hold a constant 55° roll right (lift vector at 50° toward the north). Prior to GMTRB the crossrange error approached null; and, in accordance with the steering procedure, a maximum lift orientation (0° roll angle) was established to reduce the downrange error. This roll attitude was maintained until guidance termination at drogue deployment, and the downrange error remained with the flight director indicator showing a requirement for +60 nautical miles. Figure 5.1-9 shows the computer-commanded bank

[redacted]

[redacted]

angle, crossrange error, and downrange error time histories during the guided portion of the reentry. Figure 5.1-10 shows the roll angle time history.

Retrograde IVI readout: The retrograde velocity components displayed to the pilots on the incremental velocity indicator (IVI) were 331 ft/sec in the aft direction, 105 ft/sec in the right direction (the downward component), and 4 ft/sec in the up direction (right component during reentry). These components displayed in the windows of the IVI are identical to the components generated by taking the difference in the sums of the accelerometer output (telemetry data) before and after retrofire and applying the accelerometer bias, scale factor, and misalinement matrix. (Allowance is made for 1 or 2 pulses between last telemetry frame prior to retrograde and retrograde initiation.) This demonstrates that the IVI's, as well as the interface between the computer and the IVI's, were operating properly during this phase of the mission.

Entry into the bank angle logic: Four minutes 12.3 seconds after retrofire initiation, the computer discontinues navigation with an integration interval of 16 seconds (vacuum portion of the reentry trajectory). The telemetry data indicate that at this time the navigated altitude had decreased to a value below 400 000 feet. The computer began to generate a predicted zero-lift range and a non-zero commanded bank angle 8 minutes 8.6 seconds after retrofire initiation. The value of the density-altitude parameter (D) at this time was 7.99193, which is the value of D associated with an acceleration of 5.7 ft/sec^2, which indicates that proper entry was made into the guidance logic.

Roll attitude during reentry: Between a navigated altitude of 400 000 feet and the initiation of reentry steering, the roll gimbal angle indicates that a full-lift (heads down) attitude was held within 2.5°. Five minutes 59.8 seconds after retrograde, the roll attitude was changed to a near 45° roll left. This attitude was maintained with excursions up to 15°, although the average deviation from the 45° roll angle was only about 2° to 3° until 10 minutes 12.5 seconds after retrograde. At this time, the spacecraft was rolled toward a full lift attitude which was maintained with occasional excursions to 20°. The average value of the roll angle during this period of time was approximately 5° roll left, whereas an attitude of 5° roll right (due to lateral c.g. offset) should have been maintained. It should be noted that the 45° roll left attitude was maintained until the crossrange error was nearly zero, indicating that the reentry guidance procedure outlined in the flight plan was being executed.

Guidance commands during reentry: The bank angle command generated by the computer at the time that the bank angle logic was entered was

[redacted]

approximately 33°. This increased to about 36°, and then decreased to near zero prior to the time that the pilot went to the full-lift attitude. This indicates that the spacecraft was falling short of the target, and the computer was commanding more lift in the downrange direction than was applied by the open-loop 45° bank angle program.

Navigation accuracy: Table 5.1-VI indicates a comparison of the actual telemetry parameters with those generated after the flight using gimbal angles, spacecraft body rates, and accelerometer outputs from the flight at two points during reentry. This table indicates close agreement between the sets of data which demonstrates the proper functioning of the reentry mode of the onboard computer.

Table 5.1-VII(a) gives a summary of terminal coordinates from several sources. The IGS-computed position at drogue deployment is approximately $4\frac{1}{2}$ nautical miles from the touchdown coordinates obtained from preliminary STL ground-tracking information, and 2 nautical miles from the touchdown coordinates based on spacecraft retrieval. In addition, the IGS-computed radius (altitude) is 5.21 nautical miles (31 676 ft) lower at drogue deployment than that obtained from the ground-tracking data. These navigation errors are within the variarion expected because of initial condition uncertainty, IMU misalinement, and IMU component errors. Suspected errors revealed by analysis of ascent performance were evaluated during reentry and would have resulted in errors at drogue deployment as shown in table 5.1-VII(b). From these calculations, the y-gyro drift of -18 deg/hr is excluded as a possible error during reentry. The other error sources could have existed during reentry, but cannot be isolated because of masking by other error sources of comparable magnitude.

Touchdown error evalution: The spacecraft was approximately 60 nautical miles short of the target at drogue deployment, with the target 2.3 nautical miles to the right of the spacecraft heading (based on the ground-tracking termination point).

The OAMS retro error as measured by the onboard computer was approximately 4 ft/sec. This error would shift the footprint approximately 24 nautical miles uprange. The retrorocket impulse error was measured as approximately 3 ft/sec and was near +2° in error in combined pitch and yaw, as seen by the inertial platform. These errors would shift the footprint an additional 25 nautical miles uprange. After the completion of the retroburn, ground tracking indicated a difference from the planned velocity, and corrections were implemented to change the bank angle program 10° and change the Greenwich mean time to reverse bank angle (GMTRB) approximately 11 seconds. With the assumed atmospheric density and spacecraft aerodynamics, this change in bank angle would have shifted the touchdown point for the open-loop programed bank angle reentry (OLPBAR) for approximately 48 nautical miles downrange, effectively cancelling the retro errors. However, the effects of density, Reynolds number, and trim angle of attack combined to produce an effective reduction in L/D of 28 percent to 35 percent. This reduction in L/D reduced the footprint downrange capability from approximately 355 nautical miles to approximately 195 nautical miles. As a result, the OLPBAR touchdown point shifted 65 nautical miles uprange. In addition, the actual L/D lowered the effectiveness of the OLPBAR 10° correction from 48 nautical miles to approximately 20 nautical miles, which would have produced a total miss of approximately 93 nautical miles for the OLPBAR technique (65 +48 -20). However, the actual technique flown changed the bank angle to command full-lift at the time the ADO display to the crew showed zero cross-range error. Figure 5.1-9 shows the resultant change in predicted downrange error of approximately 40 nautical miles toward the target which, if subtracted from the estimated 93 nautical mile OLPBAR error, would show an approximate 53 nautical-mile miss, which agrees reasonably well with preliminary tracking-measured drogue deployment point. Additional shifts between the retrieval point and the drogue display coordinates could occur as a result of drifts on the parachute trajectory due to winds, and to drifting of the spacecraft due to ocean currents and surface winds, for a total effect of perhaps 5 nautical miles.

Since the target was located approximately 146 nautical miles downrange from the heel of the original footprint, and assuming the 49 nautical mile shift of the footprint uprange due to the OAMS and retrorocket errors, then the target was located approximately 195 nautical miles from the heel of the actual footprint. This indicates that if a full lift or near full lift trajectory had been flown, as indicated by the computer commands and downrange flight director indications, the spacecraft would probably have reached the target. This analysis is preliminary and indicates that a more detailed analysis and additional flight data are required before a final conclusion may be reached.

Figure 5-1-11 illustrates the shift in the footprint and the reduction in area resulting from the non-nominal aerodynamics discussed in the previous paragraphs. The controllable downrange travel of the reduced footprint is about 195 nautical miles, as compared with 355 nautical miles for the preflight nominal. This reduced control capability, and the actual flight plan to execute a fixed bank angle determined from tracking after retrofire until reaching zero cross-range error, combined with the 49 nautical mile shift in footprint, resulted in the observed miss distance.

In order to evaluate the performance capability of the IGS-generated roll commands and the backup roll commands, simulations of these techniques were run with the initial conditions and aerodynamics obtained from the GT-3 flight. These results are given in table 5.1-VIII.

5.1.5.3 Control system performance evaluation.-

5.1.5.3.1 Separation: The control system was activated 357 seconds after lift-off with the firing of the two aft translation thrusters. Actual separation occurred 1.7 seconds later with the firing of the shaped-charge releasing the spacecraft from the GLV. The spacecraft body rates and attitude-control commands during this period are shown in figure 5.1-12. It is indicated that a roll was commanded and roll rate was achieved prior to separation. The relative positions of the launch vehicle and spacecraft as they moved apart are shown in figure 5.1-13. It is evident that despite the roll, no recontact was experienced and a clean separation was achieved. The combined translation-roll maneuver continued (with increased acceleration) for 12.2 seconds when translation was stopped. The control mode was switched from direct to rate command 4.5 seconds after shaped-charge fire. This point is coincident in figure 5.1-12 with the end of divergence of the roll rate. The roll maneuver continued to an attitude of 0°. The rate excursions appear to be more in response to hand-controller commands than to disturbance torques due to separation.

5.1.5.3.2 Translation maneuvers: The first OAMS translation maneuver was executed by using maneuver thrusters 11 and 12 (forward firing), and lasted for 75 seconds. Considerable firing activity was seen in the OAMS attitude solenoid commands (thrusters 1 to 8) during the translation maneuver. The roll and yaw gyros indicated rates of +0.2°/sec after the initiation of the translation maneuver, and the pilot attempted to null these rates by use of the control handle in roll and in yaw. Each time, the rates returned to ±0.2 deg/sec, indicating a possible null error or a distrubance torque. The pitch rate gyro indicated an acceleration of +0.07 deg/sec^2 which the pilot attempted to null out throughout the burn time. The pitch unbalance torque was approximately +5.0 ft-lb.

During a period of no yaw thrusting, the yaw rate gyro indicated an acceleration of +0.35 deg/sec^2 which was caused by an unbalance torque of 25 ft-lb, indicating a possible thruster misalinement.

The second translation maneuver was initiated by using thrusters 11 and 12 (forward-firing) and 9 and 10 (aft-firing). Telemetry data showed an aft translation of 10 ft/sec followed by four translations of 1 ft/sec, 2 forward and 2 aft. Ignition of thrusters 11 and 12 imparted accelerations of +0.09, +0.14, and +0.31 deg/sec^2 in pitch, roll, and yaw, respectively, resulting in unbalance torques of 7, 3, and 22 ft-lb in pitch, roll, and yaw, respectively. These values agree well with those obtained in the first translation maneuver, tending to verify the unbalance torques. During both periods when thrusters 9 and 10 were firing, no evidence of unbalance torques was noted.

5.1.5.3.3 Control mode test: This test was scheduled to verify that the control system would generate proper firing commands in response to input signals during direct and rate command (reentry) modes. It was planned to command input signals sequentially in ± pitch, roll, and yaw for each mode. Table 5.1-IX(a) is a summary of direct mode operation. The control-stick deflection was determined when the rate gyros responded to the stick deflection in an attempt to determine thresholds of the control stick. The results are tabulated in the table but are inconclusive with the data presently available. Torques were calculated from the measured attitude accelerations during periods of thrust and are compared to torques calculated from nominal thrusts and moment arms. Since no better agreement between computed and measured torques than shown in table 5.1-IX(a) was obtained, the only conclusion that can be made using available data is that the proper thrusters did fire when commanded. Table 5.1-IX(b) summarizes the rate command mode planned test. The torque comparisons yielded better agreement than in direct; however, it appears that the commanded rates were not maintained for a long enough period to allow achieving the commanded rates and, consequently, the evaluation is inconclusive.

5.1.5.3.4 Horizon scanner check: This test was scheduled to verify that when the sun was within 4° of the horizon and in the horizon scanner's field of view, the scanner would function normally without losing track, and the scanner pitch and roll signals would experience no perturbations. It was planned to control the spacecraft attitude to orient the longitudinal axis 5° to the left of the sun and execute a 5 deg/sec yaw-right maneuver to expose the scanner field of view to the sun. The test was initiated at 02:07:49.5 g.e.t. with gimbal angles of 180.68°, 178.60°, and 351.18° in pitch, roll, and yaw, respectively, with the sun approximately 3° above the horizon. As the spacecraft yawed from left to right past the sun, no loss of track was observed and the control system generated proper thruster firing commands. Figure 5.1-14 presents the results of the test which shows the effect of crew inputs during horizon scan mode operation. These curves indicate that the horizon sensors—ACME combination maintained the attitude within the intended limits.

5.1.5.3.5 Reentry: Rates during the retro maneuver were held to below 2 deg/sec in pitch and yaw while in rate command mode. The roll gyro was turned off as planned. The reentry was flown in direct mode two-ring operation. Figure 5.1-15 shows time histories of the resulting rates, attitudes, and wind-vector. Reentry oscillations became apparent just after a ground elapsed time of 4 hours 42 minutes. Roll commands created the major yaw and pitch rate disturbances during reentry. At 4 hours 44.5 minutes, the period in pitch was approximately 2 seconds, and in yaw approximately 1.8 seconds with rates between 7 to 10 deg/sec. At 4 hours 46 minutes, the pitch and yaw rates reached 20 deg/sec for short intervals. Although a bank angle reentry was flown, the fuel consumption on reentry and the reentry attitude rates were higher than those using rate command mode on GT-2. During the reentry period shown in the figure, solenoid command activity was low; however, activity was greater than on GT-2. Rate command was reinitiated after drogue deployment to stabilize spacecraft motions. The major attitude control activity was associated with maintaining roll attitude during the atmospheric portion of reentry. Although the control activity prior to atmospheric reentry was slightly greater than expected, there was not much indication of overcontrol characteristics during these periods.

5.1.5.4 Anomalies.-

5.1.5.4.1 Inertial guidance system navigation anomaly: In order to determine the source of the pitch steering command saturation during launch, comparison plots between GE-Burroughs and IGS and MISTRAM and IGS velocity were examined. Table 5.1-X shows preliminary orbit injection parameters at SECO + 20 seconds as measured by various tracking systems. Two sets of velocity comparisons were made. One set in guidance thrust coordinates compares tracking data with telemetered accumulated accelerometer count. (See fig. 5.1-16.) This comparison contains no errors caused by airborne computer limitations, and was used to estimate inertial sensor errors. The other velocity comparison set (not shown) is in guidance inertial coordinates and represents total error of the IGS system. Comparisons made between the IGS output and external tracking data indicate large inertial-guidance-system errors on this flight. There was, however, no evidence of the kind of accelerometer malfunction which occurred on GT-2 during the high acceleration intervals. (The GT-2 error was evidenced by the x-accelerometer gaining excess velocity during high acceleration.)

GE Mod III, Burroughs, and MISTRAM tracking data were used to analyze guidance performance on GT-3. The GE Mod III was of good quality for the ascent, but degraded in the elevation measurement near SECO because of the low-look angle. The quick-look MISTRAM data for the velocity comparison was of excellent quality and very smooth and continuous throughout powered flight after acquisition at 8.0 seconds. There was generally good agreement between MISTRAM and GE data over most of ascent and their quality was excellent for guidance accuracy analysis. The MISTRAM data, because of its superior accuracy, were used as the reference for this analysis.

The indicated guidance system errors at BECO and SECO, which were obtained from the position and velocity comparisons in guidance inertial coordinates are shown in table 5.1-X(a) and represent the total IGS error. The values in brackets represent errors contributed by the accelerometer and gyro sources. The total guidance-system errors at orbit insertion (SECO+20 sec) are shown in table 5.1-X(b). These values were estimated using MISTRAM I (100 000-ft base leg) as a reference.

Preliminary analysis of the indicated guidance-system errors is as follows: An explanation of the error along each computer coordinate is given since the accelerometer and gyros are alined along these axes. The major component of the velocity error in the x-direction is due to large x-accelerometer scale factor and bias error. A good explanation of the data is obtained by assuming error-source magnitudes of:

x - scale factor error	850 ppm g/g
x - bias error	520 ppm g

These error sources contribute 27 ft/sec of the x-velocity error (as shown in fig. 5.1-16). The remaining 16 ft/sec of x-error was contributed by the pitch-attitude error discussed in the next paragraph. An additional error of .4 ft/sec is caused by approximations in the onboard navigation equations.

The dominant error on this flight appeared in the y-computer axis and is caused by a pitch-axis attitude-reference error in the platform as shown in figure 5.1-16. The indicated error can be matched approximately by assuming a constant pitchdown gyro drift (y-gyro drift) of 18 deg/hr starting at approximately 190 seconds. This error model fits to within 7 ft/sec throughout the flight. Because the tracking accuracy is considered to be on the order of 0.1 ft/sec during most of the flight, improvement in fit could be obtained by considering additional small error sources as well as modification to the constant drift model. The azimuth update amounted to approximately 0.5° and limited the out-of-plane (z) velocity error to less than 2 ft/sec at SECO as shown in figure 5.1-16. There is not at this time any evidence of significant x or z gyro drift errors. An estimate of indicated accelerometer and gyro error sources to account for velocity errors are given in table 5.1-XI. Estimates of the orbit injection parameters, total inertial velocity, inertial velocity components, and flight-path angle are shown in table 5.1-XII.

The flight has been reconstructed by using telemetered IMU accelerometer and gimbal angle quantities, and by simulating the computations and operations performed by the onboard computer. Reconstructed values of position and velocity agree to within 75 feet and 0.3 ft/sec in all axes at LO+560 seconds, thus indicating that the errors were introduced prior to or at the IMU-computer interface. Further evidence tends to rule out the interface since the error appears in only two axes while common computer hardware is used to accumulate inputs from all three axes. At this time then, it is thought that the anomaly occurred in the IMU.

The MISTRAM comparison is smooth, exhibiting no sudden deviations, and the error does not propagate as one which is a function of g loading. Similarly, vibration is not a likely cause since the error first became evident during a time in second-stage operation which should be very smooth (less than $0.1g_{rms}$ was measured at this time on GT-2). In addition, there was no detectable accelerometer error during the maximum portion of the flight. Correlation of the errors along the two axes reduces the probability that an intermittent accelerometer scale factor, or bias change, in either axis was the major cause. Also discounted, after examination of the telemetered gyro torquing current, was the possibility of a temporary closure of the IMU orbit rate torquing relay.

At this time, the most likely causes are an excessive pitch drift (18 deg/hr) or a sudden pitch-axis reference change (approximately 0.5° down). Figure 5.1-16 shows an error beginning at lift-off but changing in characteristics after about LO+220 seconds, when the anomaly became evident. The errors prior to LO+220 seconds, as indicated in the previous discussion, can be accounted for by assuming an accelerometer bias of 520 ppm g and a scale factor of 850 ppm g/g. Given these bias and scale-factor errors, the remaining error, after LO+220 seconds, can be accounted for by coupling the effects of malfunction assumed in the vertical axis into the downrange axis. This correlation between the two axes substantiates the premise that the malfunction was singular and about the pitch axis.

Further analysis tends to make the sudden reference shift less likely as a cause. Detailed inspection of the pitch gimbal angle data shows no such shift during this time interval. Also, the IMU attitude malfunction circuitry would have detected the presence of significant orientation errors between the gyro and stable element. The pilots indicated that no malfunction lights came on at any time during the flight. Introduction of a step reference shift late in the trajectory (which is necessary to match the curve) does not reproduce the error in the earlier part. Finally, a detailed comparison of launch-vehicle thrust attitude as computed by the GE-Burroughs system and the IGS pitch gimbal history indicates an increasing pitch misorientation between 250 and 320 seconds. These factors, coupled with the excellent fit produced by assuming a gyro drift, tend to rule out the reference shift in favor of the gyro drift-type malfunction. Detailed inspection and test of the platforms will be performed.

5.1.5.4.1 Horizon sensor loss of track anomaly: Although the horizon sensors performed successfully in all modes of operation, an unexpectedly high number of "loss of track" indications occurred during the flight. A listing of all the "loss of track" indications was compiled together with time of occurrence and correlated activity; approximately 60 major occurrences were indicated. Thirty-five of these occurrences could be correlated to times when OAMS attitude solenoids 1 to 8 were commanded. Twelve correlated to exceeding attitude limits, and two to other switching functions. Twelve could not be correlated to any specific function. The total elapsed time when "loss of track" signal was energized, was probably less than 4 to 6 minutes for the entire orbital period. Since the horizon sensor is intended to operate in modes to provide pilot relief, these occurrences are undesirable, and the number of circumstances clearly indicates performance different from planned. Further investigation and special testing are required to isolate the causes of this undesirable operation and to implement corrective action.

The flight crew also indicated that in the horizon scan mode, pulses in the incorrect direction were observed following scanner loss-of-track. (See section 7.1.2.1.4). The problem has been investigated and first analysis shows that no pulsing occurred in the horizon scan mode after loss of track, but some pulsing did occur after re-acquisition. This pulsing appeared to be random in direction and not consistently in a direction to drive away from the horizon. Should pulses occur to drive away from the horizon, the system would pulse in the correct direction in order to regain and maintain control of the spacecraft, if sufficient time were allowed. When in this mode, the system may be operating properly even when the loss of track light is illuminated due to a four-second delay in the IMU acquisition circuit. Special testing during the GT-4 mission will be conducted to establish and confirm detailed system operational characteristics. Ground testing will also be conducted to confirm these results.

5.1.6 Time Reference System

Operation of the time reference system installed in spacecraft 3 was within specifications as nearly as can be determined from available data. The G.m.t. clock was running and was approximately correct when checked on board the recovery ship, although its reading was not verified by comparison with accurate G.m.t. The event timer was used frequently by the flight crew during the mission for a visual display of time-to-go to retrofire $\left(T_R\right)$ countdown. This was verified by comparison with the electronic timer.

The electronic timer was evaluated by comparing the available ascent and reentry PCM telemetry data (AA01, AA02, and AA03) with the ground station G.m.t. The timer started counting at LO+0.071 second. During the first 400 seconds of flight, the mean system time was slow with respect to the range time by approximately 18 msec. Available preliminary data indicate that the total drift of the electronic timer from lift-off to landing was approximately 130 msec. This is within the procurement specification of 10 parts per million (ppm) at 25 ± 10° C. Examination of the AD03 (auto-retrofire) parameter indicates that the electronic timer successfully initiated the auto-retrofire command at 04:33:23.126 g.e.t.

5.1.7 Electrical System

A review of available data from telemetry, onboard PCM recorder, and voice tape recorder indicates that the electrical power and sequential system performance was satisfactory. No anomalies have been discernible from the data. Main bus voltages and currents were within specification and prediction. Observed voltage transients or depressions on the common control and squib buses were found to be normal. It is noted that during spacecraft equipment adapter section separation, squib bus 1 voltage decreased from 26.61 to 18.92 V in 3 seconds and returned to 24.18 V in 4.75 seconds. The major electrical sequential spacecraft events and time of occurrence are tabulated in table 5.1-XIII.

The postflight inspection revealed that six fusistors (identified in table 5.1-XI) were blown. All of the blown fusistors were in circuits related to the equipment adapter separation sequence. Blown fusistors caused by bridgewire-to-case shorts were common in the Mercury program. This is the first observed case for Gemini. The fuse blocks on spacecraft 3, upon examination, were found to contain water. Two of the fuse blocks have been removed from the spacecraft and will be examined in the Malfunction Analysis Laboratory at Cape Kennedy.

A diode in the onboard lights test circuitry failed, and when the crew placed the sequential lights test switch to AMBER, the telemetry indication of attitude malfunction was changed to the ON state. Normally, telemetry readouts of cockpit displayed events should not respond to light test functions; however, since this problem was known and verified prior to flight, it was of little further consequence.

The analysis and corrective action for the primary dc-to-dc converter failure are discussed in section 5.1.3.

5.1.8 Propulsion System

5.1.8.1 Orbital attitude and maneuver system.-

5.1.8.1.1 Preflight: The orbital attitude and maneuver system (OAMS) was activated on the launch pad and all measured system parameters were verified to be normal. Static test firings of thrust chamber assemblies (TCA's) were satisfactorily accomplished prior to lift-off.

5.1.8.1.2 Mission performance: The OAMS performance as reported by the crew was satisfactory throughout the mission. Flight data indicate nominal performance of the pressure regulator. The temperature data were within anticipated limits and well within the operational capability of the system. Based on temperature-pressure computations, the propellant used during the entire mission was calculated at 201.5 pounds which compares very favorably with the preflight estimated value of 195.5 pounds.

The maneuvers planned for this mission were successfully executed in accordance with preflight and flight-updated requirements. Thrust from the 25-pound attitude TCA's adequately controlled the spacecraft throughout the mission, as did the 79- and 95-pound TCA's used during orbital maneuvers.

No malfunctions, failures, or anomalies are known to have occurred during the flight. The slow yaw reported by the flight crew (see section 5.1.4) was determined not to have been caused by TCA leakage, but to have originated from the environmental control system.

5.1.8.2 Reentry control system.-

5.1.8.2.1 Preflight: The B ring was activated on the pad and the measured data appeared normal. Static firings of each RCS thruster were satisfactorily accomplished with two bursts of 0.5-second duration each.

5.1.8.2.2 Performance: The crew reported satisfactory system performance over the entire mission. Flight regulated pressure data were nominal throughout the mission. Other than the RCS plume observation experiment for which only the B-ring was used, the demands on each ring were essentially identical with two-ring control used for retrofire, reentry, and drogue parachute stabilization. Computations from pressure and temperature data show that propellant consumption was 24 pounds from ring A and 25 pounds from ring B. This compares with an anticipated utilization of 16.8 pounds from both rings combined. The difference is attributed to the two-ring reentry which was planned late and therefore not included in the total estimate and the drogue parachute stabilization which was an unknown requirement prior to flight and was not considered in the preflight estimate. Thirty pounds of propellant were consumed by both rings prior to drogue parachute deployment.

5.1.8.2.3 System deactivation: The RCS was drained, purged, and filled with flush fluids under a pad pressure of 25 psig at Mayport, Florida. After pressurization, it was noted that A-ring TCA 2 was leaking a large quantity of one of the flush fluids. There was no reported indication of leakage during flight, or after landing while in the water, and this TCA is in the direct field of view of the command pilot. The most suspect possibilities for the leak are:

(a) Damage to the poppet or teflon seats of the solenoid valve. In the postflight system evaluation testing of spacecraft 2, the teflon seats in two fuel valves in the A-ring appeared very soft, almost liquid, and very large gas leakage rates were recorded. Possible causes of this damage are system and AGE flushing fluids.

(b) Damage to the valves from the extended time in salt water.

(c) A spurious electrical signal to the solenoid.

Peripheral cracks were reported in several of the RCS thrust chambers. (See section 12.6.) This is the normal result of ablation of the expansion chambers augmented by thermal shocks at water landing. It cannot be considered an anomaly nor does it reflect on the performance of the propulsion system.

5.1.8.3 Retrograde rockets:- Retrorocket performance appeared nominal. Retrofire was performed sequentially with time delays of 5.5, 5.2, and 6.0 seconds between firing signals to retrorockets 1 and 2, 2 and 3, and 3 and 4, respectively. The total velocity change obtained from the rockets was 347.9 ft/sec compared with an anticipated value of 346.9 ft/sec.

5.1.9 Pyrotechnics System

All functions required of the pyrotechnics system during the GT-3 mission were performed satisfactorily. There was no instrumentation on any of the individual pyrotechnic devices and, except for the devices recovered, it cannot be determined whether all redundant elements functioned.

Although the fresh-air door actuator functioned, the redundant cartridge did not fire. Investigation has shown that the reason for only one fired cartridge was very likely a peculiarity of the drogue deploy switch. Except for this switch, both the electrical circuits and the cartridge have performed satisfactorily in postflight testing. The malfunction has been repeated, using actual spacecraft wiring circuits, when the switch was depressed at certain angles. A failure analysis of the switch will be performed.

Resistance checks of other recovered cartridges showed that consistent opening of electrical circuits within the cartridges did not occur after firing. The devices showing this characteristic and the respective postflight resistance values follow:

Device	Function	Serial no.	Resistance, ohms
52-72724-11-105	Hoist loop door	049	4.83
52-72724-3-123	Pyroswitch B (circuit 1)	90	4.52
52-72724-3-123	Pyroswitch B (circuit 2)	90	5.4+
52-72724-3-123	Pyroswitch E (circuit 1)	955	0.59
52-41703-229-137	Single point disconnect	139	5.4+
52-41703-245-121	Forward bridle disconnect (circuit 1)	0556	4.97
52-41703-245-121	Forward bridle disconnect (circuit 2)	0556	5.4+
52-41703-245-121	Aft bridle disconnect (circuit 1)	0567	5.4+

A detailed postflight examination of these cartridges has not been performed; however, the resistance values are consistent with the deposition of slag across the terminals, as was previously experienced during development. Fusistors were found to have open-circuited in one or two circuits leading to pyroswitches B, C, J, and E, and to both circuits of pyroswitch D, indicating temporary hard shorts in these lines.

The pyrotechnic breech on one of the hatch actuators could not be removed using normal wrench torque. It will be X-rayed, and if no thread damage is evident it will be forcibly removed. Welding distortion of this part normally tends to make it difficult to remove; therefore, this may not be a discrepancy.

Shearing of the bayonet pins on one of the connectors to the rendezvous and recovery (R and R) section MDF ring cartridges was observed. No blow-by was evident, and the failure appeared similar to shock-induced occurrences observed in developmental tests. The successful function of the cartridge in firing the MDF ring is not affected by the failure and no adjacent areas are endangered; consequently, no action is required.

5.1.10 Crew Station Furnishings and Equipment

5.1.10.1 Crew station design and layout.-

5.1.10.1.1 Overall design: In spite of the limited cockpit volume available in the Gemini spacecraft, the flight crew found the design satisfactory for orbital flight. The most significant problem resulted from the close proximity of the spacecraft windows to the normal head positions of the flight crew members. The distance between the space suit faceplates and the windows was approximately 8 inches. When the spacecraft parachute suspension changed from single-point to dual-point, the heads of both flight crew members were thrown forward sharply. The command pilot's faceplate was broken, but there was no injury to either flight crew member. Further details of this problem are included in section 5.1.11.

[illegible]

In the weightless condition, the flight crew members tended to ride up 1 or 2 inches from their seats. There was a tendency for their bodies to straighten out and for their heads to contact the underside of the hatches. The flight crew's ability to move around in the cockpit during flight was slightly better than at sea-level conditions. The effects of zero gravity on crew station suitability were small, but they were generally favorable rather than adverse.

Visibility during launch was limited to the sky above or in front of the spacecraft. In orbit, the visibility was good within the limited field of view of the windows. During reentry, the horizon was visible until approximately 100 000 feet. After parachute deployment and inversion to the dual-point suspension, the flight crew members could not see the ground, even when they moved as close to the window as possible.

The only problem area was that associated with parachute inversion as indicated previously. No other problems in overall crew station design were noted in this flight.

5.1.10.1.2 Equipment layout: Accessibility to controls and equipment was easier in flight than on the ground. No difficulties were experienced in accessibility. The side and center food boxes were readily accessible and were used throughout the mission. The remaining stowage locations were not used during the orbital phase of the mission because of the short duration of the flight and because there was no requirement to do so. The only problem with equipment layout was the location of the side food-box extensions. These containers, intended for stowage of dry waste, were inaccessible to the pilot while he was wearing the space-suit gloves. They were located outboard of each ejection seat, and the opening between the seat structure and the cabin walls was too narrow to allow the space-suit glove and wrist ring to pass. Further evaluation using a flight-configured spacecraft will be required to determine whether or not these containers will be usable when the gloves are removed.

5.1.1.10.1.3 Crew furnishings: The lap belt, shoulder straps, and parachute harness were satisfactory for this flight. No significant difficulties were encountered in removing, stowing, unstowing, or attaching the restraint straps and harness. The space suits were removed after the spacecraft was on the water, and the parachute harnesses were donned again without difficulty. The ejection seat contours were satisfactorily fitted for the GT-3 flight crew. No significant discomfort was experienced during the prelaunch, ascent, or reentry phases of the mission. The command pilot's elbow support was raised prior to launch and remained in place for the entire flight. The seat headrests provided adequate support during launch and reentry. No problems were encountered with the elbow support or the seat headrests at any time during the mission.

5.1.10.1.4 Cabin lighting: The cabin lighting was satisfactory for daylight operation. The illumination on the bright setting was adequate except for the center panel and, in particular, the flight-plan roller. Nearly full bright intensity was required on the center light to illuminate the flight-plan roller. Lighting on the side instrument panels was adequate.

The cabin lighting was poor for orbital operations on the dark side of the earth. In order to illuminate the center panel, particularly the flight-plan roller, it was necessary to set the center light to medium on white light. This intensity destroyed the flight crew member's night adaptation and interfered with outside vision. When the interior lights were turned down to permit seeing outside, the illumination of the center panel and the flight-plan roller was inadequate.

5.1.10.2 Controls and displays.-

5.1.10.2.1 Controls: The maneuver hand control operated satisfactorily during spacecraft separation and the subsequent translation maneuvers. Access to this control was slightly easier in zero gravity because the pilots tended to ride up and forward in the seats. The attitude hand control operated satisfactorily in all control modes throughout the flight. The command pilot indicated that the control forces were very close to those experienced in the mission simulator. The abort hand control was not used in flight. No difficulty was encountered in reaching or holding this control during the launch phase.

The telelight switches and the switch locks on the sequence panel were operated in flight without difficulty. The sensor switches for the green light on the retro-adapter separation switch were deactivated before the flight. The secondary O_2 high-rate switch was not used until after landing. All other switch and circuit-breaker controls operated satisfactorily. Only one switch was actuated inadvertently in flight. This was a control system circuit breaker, which was displaced several times while stowing and unstowing the swizzle stick. The manual data-insertion unit (MDIU) keyboard was operated repeatedly without difficulty. The pilot indicated that insertion of parameters was no more difficult than reading them out, as far as the keyboard unit was concerned.

The ejection seat controls were found to be satisfactory although no actual use of the D-ring handle occurred. The shoulder harness release and the ditching release controls were satisfactory. The water and waste panel controls were operated satisfactorily during the prelaunch, orbit, and postlanding phases of the mission. The controls were satisfactory in all cases.

The voice tape-recorder controls on the voice control center (VCC) caused considerable operational inconvenience during the flight. When either pilot selected "Record" on the VCC, he could not transmit to the ground stations. On several orbital passes, transmissions from the pilot to some of the ground stations were lost because the VCC mode selector switch was in "Record" rather than "UHF."

5.1.10.2.2 Displays: The flight crew indicated that the displays provided in the cockpit were satisfactory for operational use except as follows:

(a) The flight crew found the need for direct reference to ground elapsed time (g.e.t.) and Greenwich mean time (G.m.t.). The use of a wrist watch alleviated the G.m.t. requirement, and the pilot used the elapsed time feature of the G.m.t. clock as a backup reference for time since lift-off. Poor readability of the G.m.t. clock hampered its use for any purpose. The flight crew indicated a preference for digital display of G.m.t. rather than by dial and pointers.

(b) The indicator on the center panel was the only means provided for warning the flight crew of a loss in cabin pressure. Since the indicator was not readily visible to either member of the flight crew, there was some concern that a drop in cabin pressure might not be noticed until it was dangerously low. The flight crew has expressed a preference for a warning light or horn to signal loss of cabin pressure.

(c) The flight-plan roller did not display sufficient information at one time to allow the pilots to anticipate future events. This difficulty was attributed primarily to the short length of the flight-plan roller limiting the amount of information which could be displayed. For future missions, a possible solution would be to supplement the flight-plan roller with other information to help the pilots anticipate future events.

The locations were considered satisfactory for all the displays except that of the cabin-pressure indicator. Since this indicator must be monitored closely by the pilot during launch, a position nearer the right side of the center panel would make it more readily visible. The visibility of the instruments on the center panel was poor under dark-side lighting conditions. The problem is discussed in detail in section 5.1.10.1.4. The accuracy of the displays was satisfactory. Prior to the flight, the cabin pressure indicator had shown errors up to 0.4 psi, primarily caused by hysteresis in the indicating system. In flight, this instrument read 5.6 psia when the telemetered cabin pressure read 5.57 psia. Because of qualification problems, the accuracy of the G.m.t. clock was questioned prior to the flight. There was no check in flight which would establish the magnitude of any error in the G.m.t. clock.

The view of the horizon during reentry was better than had been anticipated. Both flight crew members had an adequate view of the horizon down to an altitude of about 100 000 feet. The reference marks on the window frames were satisfactory for judging bank angles at or near 60°. The view of the horizon was confirmed to be adequate for backup attitude reference for daylight reentries.

5.1.10.3 <u>Space suits and accessories</u>.- The general comfort and operational suitability of the Gemini G3C space suits were good. There were no significant discomfort or mobility problems at any time during the flight. The inner cloth liners had been removed from the suits prior to the flight. The flight crew indicated that the mobility and comfort of the suits were substantially increased by removal of these liners and that no difficulties resulted. All aspects of the space suit were compatible with the spacecraft except for the incidents described in this section.

The fingertip lighting system was used extensively in flight and was found to be particularly helpful in augmenting the cabin lights. The location of these lights on the first joint of the second and third finger of each hand made them vulnerable to damage during operation of various cockpit switches, particularly the recessed push button switches. Relocation of these lights to the second joint of the fingers would reduce the susceptibility to damage. In addition, it was found that the fingertip light switches were susceptible to inadvertent operation. Increased actuation force or protection of these switches is required to avoid unintentional operation.

As described in section 5.1.10, the command pilot's visor was broken during the transition of the spacecraft to two-point suspension on the recovery parachute. The force to which the visor was subjected cannot be given a quantitative value; but the flight crew described the helmet impacts as severe, although their heads remained within the helmet liners. This incident was a good indication of the impact protection provided by the space suit helmets. In the postflight review of the pilot's helmet, it was found that the visor and the top of the helmet had been badly marked by a painted surface. The marks were in a critical viewing area of the visor and would have interfered with visibility. The pilot had gone through most of the flight with his helmet touching the underside of the hatch; consequently, a possible solution would be to provide better hatch padding.

During the waste evaluation in flight, the pilot was unable to position the urine receiver properly because of the location of the structural tab on the forward end of the main suit zipper. In order to eliminate this interference, the pilot cut the structural tab. Subsequent use of the urine receiver was satisfactory. During the evaluation of the defecation device the pilot discovered that the opening in the back of the underwear was not large enough to permit normal positioning of the device. He eliminated the interference by tearing the back of the underwear.

[redacted]

5.1.10.4 Pilots' operational equipment.- The flight crew reported that the flight-plan strip was satisfactory for use on the flight-plan roller, except for the limitation in the length displayed, as discussed in paragraph 5.1.10.2. The flight booklet was not used because of the short duration of the flight. The primary reference for the inflight checklist and trajectory information was the set of flight cards carried by the pilot in the leg pocket of his space suit. The star charts and star-chart slider were not used in flight because no dark-side navigational tasks were required. [redacted]

The plotboard was evaluated and used in flight, and several discrepancies were noted. The orbital path display could not be rotated around the plotboard without opening the cover. When the cover was opened, the flight booklet and the other items stowed inside tended to float out. The elastic tiedown cords within the plotboards were not used, apparently because of handling difficulties. The command pilot indicated that the plotboard did not fit in a position suitable for writing. The flight crew suggested the use of simple kneeboards to hold cards and paper for writing and displaying inflight data. [redacted]

The voice-tape recorder cartridges were satisfactory for handling and insertion. The flight crew reported that there was no way of determining if the recorder were operating properly after a tape cartridge was inserted. Difficulty was encountered at one time with the recorder door latching mechanism. Because the recorder cannot be seen, the pilot stated that he had difficulty recognizing with a gloved hand that the door was properly closed. After the third reload he apparently did not succeed in closing the door which resulted in the loss of recording on one complete tape cartridge. The lightweight headset was not used during any phase of the mission. [redacted]

The 16-mm sequence camera apparently operated satisfactorily; however, the proper lens settings for the low-light-level photography involved in the thruster plume evaluation had not been given to the flight crew prior to flight. As a result, the 16-mm photography was a complete failure. The 70-mm Hasselblad camera operated satisfactorily, and 26 pictures were obtained during the last orbital pass. Picture resolution and color resolution were excellent. The optical ring sight was not used to aim the Hasselblad camera; instead, the pilot alined the camera by eye while holding it against the spacecraft window. The optical sight was not considered satisfactory. The command pilot commented that the optical sight, because of its size, interfered with vision outside the window. In addition, difficulty was encountered in seeing the reticle pattern while tracking an earth target, because the reticle pattern was not bright enough.

The swizzle stick was satisfactory. The pilot used the swizzle stick during prelaunch to gain access to the controls on the water management panel between the seats, and in orbit, to gain access to some of the loose equipment stowed in the aft end of the right-side food box.

5.1.10.5 Personal equipment.-

5.1.10.5.1 Urine disposal system: Both flight crew members wore urine collection devices (UCD) during the prelaunch and ascent phases of the mission. The pilot removed his UCD soon after insertion of the spacecraft into orbit. At that time, ballooning of the UCD bag, resulting from the cabin-pressure decrease during ascent, caused difficulty in removal and stowage. The pilot eventually cut the bag to release the entrapped air. During the second orbital pass, the UCD worn by the command pilot became detached and some leakage occurred. The UCD check valve prevented any gross leakage from the bag.

The pilot used and evaluated the spacecraft urine transport system (UTS) in orbit. Upon completion of the initial use, leakage from the receiver cone occurred. This leakage was caused by the bellows being extended to the full length before the micturition was completed. Urine then backed up in the receiver line and leaked out into the cockpit. During the second use, the pilot extended the bellows slower and maintained a suction on the bellows necessary to drain the receiver line. Only minor leakage occurred during this use, and a utility towel was used to collect the urine released into the cabin. The results of the evaluation indicated that the UTS was satisfactory, providing care was exercised in the rate of extension of the bellows.

5.1.10.5.2 Defecation device: The defecation device was used satisfactorily by the pilot during the waste evaluation phase of the flight. The basic bag design and adhesive characteristics were acceptable; however, the pilot was unable to burst the disinfectant bag properly. The heat seal of the bag was stronger than the specification value and it was extremely difficult to burst the bag by hand pressure. The pilot also experienced difficulty in handling and disposing of the paper packed with the defecation bag. The paper tended to float away since there was no retention method after removal from its stowage pocket, and after use it was difficult to insert in the defecation bag. A separate disposal bag for the paper would simplify the handling problems.

5.1.10.5.3 Food and water systems: Two rehydratable food packages containing apple sauce and grapefruit juice were used and evaluated in flight. Although the rehydration process was accomplished satisfactorily without leakage, the pilot stated that a considerable amount of manual dexterity was required to accomplish this task while wearing the space-suit gloves. Both flight crew members ate from the rehydrated food packages without difficulty and also sampled one package of bite-size chicken. These items were free from crumbs; however, the pilot indicated that the food was too salty. The bite-size brownies were not used because of concern about possible oil leakage from the package.

After the rehydratable food packages were used, the pilot, while wearing the space-suit gloves, had difficulty in removing the germicide pill and inserting it into the food bags. The small tab containing the pill became detached and drifted away before it could be inserted into one of the bags. This is attributed to a poor heat seal on the tab. The pilot successfully inserted the pill into the other bag. After rolling up the feeding end of one food bag, the pilot noted that grapefruit juice was seeping out of the end of the bag. The use of tape to seal the ends of these bags would prevent fluid leakage of this type.

Both flight crew members used the water dispenser with satisfactory results during orbit and after landing. The water pressure throughout the orbital phase was approximately 6 psi above cabin pressure. This pressure was satisfactory for drinking and rehydration. All drinking was accomplished with the space-suit faceplate open. The drinking ports on the helmets were not used. After landing, the pilot attempted to use the blood-pressure bulb to pressurize the water tank, but he had difficulty in inserting the blood-pressure bulb into the water-tank pressurizing adapter. This problem is still being investigated.

5.1.10.5.4 Carbon dioxide detection tapes: One carbon dioxide detection tape was used by the command pilot. The tape indicated a concentration of CO_2 equivalent to a partial pressure of 2 mm Hg. The detection circle for a CO_2 concentration of 4 mm Hg was partially saturated. No indications were noted in the detection circles for 6 or 8 mm Hg. Operation of the tapes was apparently successful with exposure times of approximately 40 seconds. The spacecraft system for sensing CO_2 concentration indicated a partial pressure of approximately 0.5 mm Hg throughout the flight. The difference between the tape readings and the spacecraft data appears to be the result of excessive exposure times. For subsequent flights the nominal exposure time will be reduced to 30 seconds. The basic operation of the sensing tapes and the resulting accuracy of measurement were satisfactory for flight use.

5.1.10.5.5 Inflight medical kit: The inflight medical kit was not used during the mission. The flight crew indicated that proper indoctrination in the use of the kit had been provided, that the kit was readily accessible, and that they would have been able to use the kit had it been necessary.

5.1.10.5.6 Personal hygiene system: The principal items of personal hygiene used during the mission were the 12 by 24-inch towels which served as moisture absorbers. They were used extensively throughout the flight for utility purposes. The flight results indicate that an ample and readily accessible supply of towels is needed in future missions. The personal hygiene wet pads and the oral hygiene chewing-gum packet in the food packages were not used during the mission.

5.1.10.5.7 Life vests: Before opening the hatches after landing, the flight crew removed their parachute harnesses with the life vests attached. After removing their space suits, they again donned the harnesses before egressing from the spacecraft. After egress and while waiting to be picked up by the helicopter, they inflated their life vests as a safety precaution. No difficulties were noted in use or inflation of the life vests.

5.1.10.5.8 Survival equipment: After landing, the command pilot used his ditching handle to release his ejection seat backboard in preparation for removing the survival kit. Before any further steps were taken, he decided to egress without the survival kit, because the recovery crew had ample equipment on hand. The command pilot indicated that he could have satisfactorily removed the survival kit, if necessary.

5.1.10.6 Equipment stowage.-

5.1.10.6.1 Equipment location: The operational and experimental equipment carried in the crew station and the location for each item are listed in table 5-II. No difficulty was experienced in access to any of the stowed equipment.

5.1.10.6.2 Equipment removal and replacement: The flight crew reported that the 70-mm Hasselblad and the 16-mm sequence camera were hard to unstow and stow in their respective compartments in the centerline stowage box located overhead between the seats. This difficulty was a fitting problem rather than an accessibility problem. The compartments within the centerline stowage box were lined with a cushioning material to protect the equipment against launch vibrations. Insufficient clearance was allowed between the equipment and this cushioning material, and the resulting tight fit made it difficult to stow and unstow the equipment. This problem is peculiar to spacecraft 3 only, because this stowage container is not used on subsequent spacecraft.

5.1.10.6.3 Waste disposal: Both pilots had difficulty in disposing of dry waste such as overwraps for food and hygiene items. Since neither could reach the side food-box extensions, they stowed their dry waste in the side boxes along with other articles. This trash tended to float back into the cabin every time the side boxes were opened. Because he was pressed for time, the pilot elected to stow his UCD and used defecation device in the right-side box also. After landing, these items had to be removed in order to reach the emesis bags. In future flights most of the waste disposal problems can be avoided by using the stowage pouches as waste containers. The lack of these stowage pouches in spacecraft 3 contributed to the difficulties of dry waste disposal.

5.1.10.7 Bioinstrumentation system.-

5.1.10.7.1 System description: The medical monitoring instrumentation flow on the GT-3 mission consisted of a flight instrumentation package for each pilot and two biomedical magnetic tape recorders. The flight package consisted of an oral-temperature measuring system, blood-pressure measuring system, respiration-rate and pattern device, and two electrocardiogram systems. The signal conditioners for each of these parameters were worn inside the spacesuit in pockets in the underwear. The package was identical for each flight crew member. Both pilots reported that the bioinstrumentation system was worn throughout the mission without interference or discomfort. Neither pilot received any skin abrasions or raw spots from the bioinstrumentation equipment. The equipment was confirmed to be suitable for flight use.

5.1.10.7.2 Real-time measurements: The two biomedical recorders ran simultaneously throughout the mission and recorded the electrocardiogram and respiration parameters on the redundant basis with the real-time transmission of the PCM telemetry. The data received from the real-time transmission were of high quality. The magnetic-tape recordings were reduced, and satisfactory performance of the recorders was verified. The problems encountered with medical monitoring during the mission were confined to those of an operation nature.

5.1.10.7.3 Blood-pressure-measuring system problems: During the first orbital pass the command pilot reported that he was unable to insert the blood-pressure bulb adapter into the space suit fitting and, therefore, could not inflate his blood-pressure cuff. He made this attempt at his first scheduled blood-pressure reading in the flight plan. The command pilot made no further attempt to take the blood-pressure measurement during the mission. During the preflight bioinstrumentation system checkout, no problem was detected. After recovery, the flight surgeon on the carrier was able to mate the blood-pressure bulb carried on the spacecraft with the spacesuit fitting, with no indication of what the trouble might have been. The command pilot's space suit was inspected upon its arrival at Cape Kennedy, and no damage could be detected at the blood-pressure fitting nor was there difficulty in inserting the blood-pressure bulb. The problem in flight is attributed to difficulty in seeing the blood-pressure fitting during dark-side operation and the pressure of other inflight activities. It should also be noted that this bulb was a new double-ended bulb that was supplied to the flight crew on launch day. Both ends fit the receptacle, but only one end could be used to perform the blood-pressure function. The other end was for water transfer. The pilot successfully completed all planned blood-pressure measurements with excellent readouts on telemetry, except on the first pass over Cape Kennedy when he attempted the task at 01:40:49 g.e.t. and again at 01:42:08 g.e.t. Unfortunately, both MCC (Tel III) and CNV (Tel II) sustained losses of signal (LOS) at 01:40:13 g.e.t. and 01:40:28 g.e.t., respectively, and the data were not recovered.

5.1.11 Landing System

The parachute landing system on GT-3 performed as designed with no known failures. All functions occurred when commanded by the command pilot, and the timing of the sequential events was within established tolerances. Figure 5.1-17 depicts the events of the landing system as they occurred on the GT-3 mission. Based on altimeter readings by the command pilot, the drogue parachute was deployed at 50 000 feet, the pilot parachute and the main parachute at 10 600 feet. The terminal rate of descent was 30 ft/sec as given by the cockpit indicator. The flight crew reported that the landing shock was well within acceptable limits. The main parachute remained inflated and dragged the spacecraft through the water before the bridle disconnects were actuated. The surface winds in the recovery area were about 20 knots.

None of the parachutes or components were available for postflight inspection. The rendezvous and recovery (R and R) section had no flotation capability, and no attempt was made to recover the main parachute. However, all evidence indicates that none of these components had been damaged. The flight crew's ability to assess the condition of the parachutes was excellent. Even though the reentry control system (RCS) was activated while the drogue parachute was deployed, the flight crew did not detect any damage to the fabric materials as a result of the fuel and oxidizer fumes.

After the drogue parachute was deployed, spacecraft oscillations began to build up, reaching a magnitude of ±20°, as estimated by the flight crew, before the RCS was turned on and the oscillation damped out. The drogue parachute was in the reefed condition at the time the RCS was turned on. The RCS was turned off at about 30 000 feet, indicating that the drogue parachute disreefed during the period of RCS activation. (For a 50 000-foot deployment, disreef nominally occurs when the spacecraft descends to 40 000 feet.) The spacecraft remained stable under the influence of the drogue parachute until it was cut loose at pilot parachute deployment. To reposition the spacecraft to the two-point bridle landing attitude, the single-point disconnect was released, and the spacecraft rotated freely through an angle of 55° from the vertical before the forward bridle leg stretched tight. The attendant forces probably caused the flight crew members' heads to rotate forward so that their helmets contacted the window area. The visor on the command pilot's helmet was damaged. There are insufficient spacecraft data available for a meaningful analysis of this event. A suitable resolution of the problem will require a ground-test program.

5.1.12 Postlanding and Recovery System

Recovery photographs indicate that all of the recovery aids were deployed as planned during the descent and after landing. The descent and UHF rescue beacon antennas were released and extended as the bridle was stripped from its stowage trough during the two-point repositioning maneuver. The sea dye marker apparently activated upon touchdown. The hoist loop and flashing recovery light were deployed when the main parachute was jettisoned. Approximately 5 minutes after touchdown, the HF antenna was extended, but would not retract for the spacecraft recovery operation. The operation and effectiveness of these aids are discussed elsewhere in this report.

The flotation attitude, prior to installation of the flotation collar, was approximately as predicted. The flight crew indicated that stability was greatly improved after the installation of the flotation collar. After landing, the command pilot released the seat backboard with the intention of pulling out his survival kit and raft. During this activity the ballute deployment and the release pyrotechnic device was actuated, resulting in cutting of the ballute bridle. This is the normal action of the system and does not constitute any malfunction. The recovery crews subsequently replaced safety pins in the fired devices during the spacecraft safing operations, as required in the recovery procedures.

TABLE 5.1-I.- REAL-TIME DATA SUMMARY FOR TEL II, MCC-CAPE KENNEDY, GBI, AND AIRCRAFT 629 AND 630

Acquisition period (available data)			Usable data received					Valid data, percent of available data
AOS, g.e.t., hr:min:sec	LOS, g.e.t., hr:min:sec	Δ time, min:sec	AOS, g.e.t., hr:min:sec	LOS, g.e.t., hr:min:sec	Δ time, min:sec	Sync losses, min:sec	Valid data, min:sec	
				Tel II				
00:00:00.0	00:06:59.9	06:59.9	00:00:00.0	00:06:33.5	06:33.5	00:00.0	06:33.5	93.7
01:33:36.0	01:40:04.7	06:28.7	01:33:38.2	01:40:04.6	06:26.4	00:07.7	06:18.7	97.4
03:09:18.6	03:12:49.8	03:31.2	03:09:18.4	03:12:49.5	03:31.1	00:00.0	03:31.1	100.0
04:39:08.4	04:39:27.6	00:19.2	04:39:08.1	04:39:27.3	00:19.2	00:00.0	00:19.2	100.0
				MCC-Cape Kennedy				
00:00:00.0	00:06:59.9	06:59.9	00:00:00.0	00:04:28.3	04:28.3	00:02.7	04:25.6	63.3
03:08:57.1	03:12:49.8	03:52.7	03:08:56.7	03:12:49.5	03:52.8	00:00.0	03:52.8	100.0
				GBI				
00:00:48.0	00:07:21.6	06:33.6	00:00:47.8	00:07:21.5	06:33.7	00:27.6	06:06.1	93.0
01:34:12.0	01:40:16.8	06:04.8	01:34:11.8	01:40:14.2	06:02.4	00:00.0	06:02.4	99.3
03:08:54.6	03:13:11.3	04:16.7	03:08:54.4	03:13:11.1	04:16.7	00:00.0	04:16.7	100.0
				Aircraft 629				
04:56:44.0	05:07:45.2	11:01.2	04:58:08.0	04:58:51.1	00:43.1	00:00.0	00:43.1	6.5
				Aircraft 630				
04:45:13.1	04:56:17.8	11:04.7	04:46:32.1	04:55:24.7	08:52.6	02:18.5	06:34.1	59.3

TABLE 5.1-II.- DELAYED TIME PCM DATA SUMMARY

Data source	Usable data period			Sync losses, hr:min:sec	Valid data, hr:min:sec	Valid data, percent
	AOS, g.e.t., hr:min:sec	LOS, g.e.t., hr:min:sec	ΔT, hr:min:sec			
First orbit						
Tel II[a]	00:01:26.25	01:34:03.20	01:32:59.55	03:53.655	01:26:42.25	93.6
GBI[b]	00:04:15.05	01:35:22.3	01:31:07.25	07:40.91	01:23:26:34	91.6
Second orbit						
Tel II	00:42:23.75	03:07.41.6	01:25:17.85	18:29.05	01:06:48.82	78.3
GBI	01:42:33.4	03:07:39.2	01:25:05.8	09:39.4	01:15:26.4	88.6
Third orbit and reentry						
Onboard recorder	03:15:18.4	04:59:24.7	01:44:06.3	05:21.6	01:38:44.7	94.8

[a] Sync losses include 233.655 seconds loss incurred by dc-to-dc converter malfunction.

[b] Sync losses include 233.655 seconds loss incurred by dc-to-dc converter malfunction.

TABLE 5.1-III.- TEST EVENTS

(a) Launch Phase

Ground elapsed time, minutes			Event	System				Test number
Planned	RCS	IGS		Acme	Comp	IMU	HS	
10.16	10.14	10.18	Roll program start	IGS backup	Ascent	Free	Search (Pri)	2
20.48	20.42	20.47	Roll program end	IGS backup	Ascent	Free	Search (Pri)	2
25.04	25.01	22.971	No. 1 pitch rate start	IGS backup	Ascent	Free	Search (Pri)	2
88.52	88.22	88.458	No. 2 pitch rate start	IGS backup	Ascent	Free	Search (Pri)	2
100.00	105.00	105.39	No. 1 IGS update	IGS backup	Ascent	Free	Search (Pri)	2
104.96	104.82	105.20	No. 1 gain change	IGS backup	Ascent	Free	Search (Pri)	2
119.04	118.84	119.29	No. 3 pitch rate start	IGS backup	Ascent	Free	Search (Pri)	2
140.0	145.00	146.52	No. 2 IGS update	IGS backup	Ascent	Free	Search (Pri)	2
162.56	162.52	167.996	Termination of pitch program	IGS backup	Ascent	Free	Search (Pri)	2
338.43	333.75	333.65	SECO	IGS backup	Ascent	Free	Search (Pri)	2
		333.80	Initiation of IVAR					

TABLE 5.1-III.- TEST EVENTS

(b) Orbit and Reentry

Ground elapsed time, hr:min:sec		Event	System				Test no.	Remarks
Planned	Actual		ACME	Computed	IMU	Horizon scanner		
361 sec	347.52 sec	Forward separation thrust (10 ft/sec)	Direct, rate command	Ascent	Free	Search (primary)	5	Clean separation is shown. The burn lasted 13.9 sec and resulted in a ΔV of 12.16 ft/sec.
363 sec	358.32 sec	90° roll	Direct, rate command	Ascent	Free	Search (primary)	5	A negative roll was initiated in the direct mode followed 5 sec later by rate command control. The roll rate peaked at -20°/sec.
00:16:00	00:19:30.7 00:20:20	Direct control check rate command (reentry) check	Direct, rate command	Prelaunch	Orbit rate	[illegible]ng [illegible]ary)	4	The pilots' stick commands resulted in the expected jets burning. Torques were generally close to those expected. Unaccountable disturbance torques of 0.1, 0.7, 1.5 ft-lb in roll, pitch, and yaw, respectively, were measured from 0:08:40 to 0:09:00. This anomaly appeared to be intermittent. No data were available past 0:10:00 to define these characteristics accurately.
01:00:00	00:59:05	RCS ring-B plume observed	Direct	Prelaunch	Orbit rate	Tracking (primary)		Plumes were observed by pilots. No loss of track occurred with secondary scanner operating.
	01:08:57	Catchup mode check	Pulse	Catchup	Orbit rate	Tracking (primary)	5	Catchup mode functioned as expected. Initial IVI's were set at 50, -30, 40, at gimbal angles of 369.78°, 0.648°, and 359.42° for θ, ϕ, and ψ, respectively. When the gimbal angles indicated 320.69°, 359.60°, and 334.98°, the IVI's indicated 69.0, -1.0, and +0.0. The small error in these readings is attributed to an uncompensated x accelerometer bias of 0.01226 ft/sec^2.

[redacted]

[redacted]

TABLE 5.1-III.- TEST EVENTS

(b) Orbit and reentry - Continued

Ground elapsed time, hr:min:sec		Event	System				Test no.	Remarks
Planned	Actual		ACME	Computed	IMU	Horizon scanner		
01:33:00	01:32:59	Translate aft	Rate command	Catchup	Orbit rate	Tracking[1] (secondary)	6	Aft burn lasted 77 seconds. Pitch and yaw angles deviated from zero by 0.34° to -0.39° and 0.75° to -0.35°, respectively.
01:40:00	01:46:16	Platform aline and caging check	Pulse	Prelaunch	Cage-SEF	Tracking (secondary)	7	When caging was initiated, pitch, yaw, and roll angles were -7.5°, 11°, and 12°, respectively. Approximately 7 min of SEF alinement followed. Data indicate 12.5° of yaw misalinement remained because of short alinement time.
02:10:00	02:07:49	Horizon scanner check	Direct, horizon scanner	Prelaunch	Orbit rate	Tracking (secondary)	8	No loss of track occurred, or perturbations in pitch and roll output, when secondary scanner was exposed to the sun.
02:17:00	02:16:59	Fore-aft translational system check (out of plane)	Direct	Catchup	Orbit rate	Tracking (secondary)	9	Fore and aft thrusters responded to the maneuver stick commands.
02:30:00	02:34:16 02:52:30 not complete	Horizon scanner control characteristic check	Direct, horizon scanner	Prelaunch	Orbit rate	Tracking (secondary); primary at 02:40:00	10	From available data, it appears that two acquires occurred; however, these tests were not identified. Secondary scanner was utilized until 02:40:21 when primary was switched on-line.
02:57:00	Not complete	Tracking task	Pulse	Prelaunch	Orbit rate	Tracking (primary)	11	The initiation of tracking task was not identified.
03:30	Not accomplished	Control mode characteristic check 1.0 pulse mode 2.0 rate command	Pulse, rate command	Prelaunch	Orbit rate	Tracking (primary)	12	Test was canceled during flight.

TABLE 5.1-III.- TEST EVENTS

(b) Orbit and reentry - Concluded

Ground elapsed time, hr:min:sec		Event	System				Test no.	Remarks
Planned	Actual		ACME	Computed	IMU	Horizon scanner		
04:22:00	04:21:22	Translate forward	Rate command	Catchup	BEF	Tracking (secondary)	14	Jets 9 and 10 burned for 112.9 sec. Pitch and yaw angles deviated from zero by 1.50° to -1.84° and 0.68° and -0.72° in pitch and yaw, respectively, during this time.
04:33:45	04:33:22	Retrofire	Rate command	Reentry	Free	Tracking (secondary)	15	During retro, average pitch and yaw angles were 17.55° and 0.605°, respectively.
04:37:52	04:37:37	400 000 ft	Direct	Reentry	Free	Jettisoned		
04:47:06	04:46:51	Drogue parachute deployment	Direct, rate command	Reentry	Free	Jettisoned		

TABLE 5.1-IV.- HORIZON SENSOR ALINE TEST SIGNIFICANT EVENTS

Event	Time from lift-off, hr:min:sec	Pitch gimbal angle, deg	Yaw gimbal angle, deg	Roll gimbal angle, deg	Horizon scanner pitch, deg	Horizon scanner roll, deg	Remarks
Begin caging	01:46:16	-7.4	10.9	11.6	-7.8	5.8	Pitch torquing current jumped to 16 volts at this time implying switch to cage.
Begin aline SXF	01:46:55	-0.1	0.1	2.6	-6.4	19.0	Roll and yaw torquing current saturates implying switching to aline SXF.
Begin ORB rate	01:54:02	0.4	0.8	-0.5	0.8	-1.6	[illegible] for alinement was ≈ 7 minutes 26 seconds.
Begin translation maneuver	02:16:59	271.9	70.9	92.3	-3.0	6.0	Translation begins ≈ 22 minutes 57 seconds after ORB rate.

TABLE 5.1-V.- INERTIAL GUIDANCE SYSTEM MEASURED VELOCITY CHANGES[a]

Event	DCS		IGS computed ΔV, fps				IGC computed ΔV corrected accelerometer bias error, fps			
	Preflight planned, fps	Ground command, fps	V_R	X	Y	Z	V_R	X	Y	Z
Second stage tail-off	96	-	79.8	78.5	15.2	0.8	79.6	78.1	15.2	0.8
Spacecraft separation	10	-	12.1	12.0	1.7	1.1	11.9	11.8	1.7	1.1
Translation no. 1	66	48.5	48.8	-48.8	0.9	-0.9	49.7	-49.7	0.9	-0.9
Translation systems check	10	-	9.7	0.6	-2.7	9.3	9.7	0.2	-2.7	9.3
OAMS retro	98	98.1	98.9	-98.9	1.5	-0.9	99.9	-99.9	1.5	-0.9
Retrograde	346	-	347.9	-331.4	103.9	-3.5	348.2	-331.7	103.9	-3.5

[a] These revised velocity changes were determined by using the accelerometer outputs, biasing them, and transforming through the misalignment and scale factor matrix, as determined from preflight data. These velocity changes are those resulting from the applied thrusts only and do not include the velocity changes due to gravity.

TABLE 5.1-VI.- GUIDANCE AND NAVIGATION PARAMETERS FROM FLIGHT RECONSTRUCTION

LAS quantity	T = 4:37:37.15 hr:min:sec, g.e.t. Altitude = 400 000 feet			T = 4:46:10.65 hr:min:sec, g.e.t. Guidance termination, $D = D_0$		
	Actual	IBM	NAC	Actual	IBM	NAC
Time in mode, t, sec	826.000	826.000	826.000	1 339.125	1 339.125	1 339.125
Distance to earth center, V_e, ft	21 300 572	21 300 558	21 300 527	20 961 420	20 961 555	20 960 545
Spacecraft earth reference velocity, V_E, ft/sec	24 071.574	24 071.555	24 070.905	1 722.090	1 719.587	-
Flight-path angle (earth reference), γ, deg	-1.055	-1.0542	-1.055	-41.906	-41.8938	-40.252
Downrange error, $R_n - R_p$, n. mi.	NA	NA	NA	61.999	62.460	-
Crossrange error, R_s, n. mi.	-39.089	-	-	0.096	0.108	-
Commanded bank angle, B_c, radians	NA	NA	NA	0	0.9548	-
Inertial latitude, φ, deg	31.170	31.1728	31.181	22.384	22.3889	22.38489
Inertial longitude, θ, deg	-173.588	-173.6079	-173.518	-142.571	-142.5827	-142.569
Predicted zero lift range, R_p, n. mi.	NA	NA	NA	2.297	2.297	-
Computed density altitude, D, nondimensional	NA	NA	NA	4.648	4.6551	-
Heading to target, ψ_T, deg	101.604	-	-	117.429	117.4025	-
Range to target, R_T, n. mi.	1 691.126	-	-	64.149	64.548	-
Spacecraft earth reference heading, ψ_E, deg	100.543	100.5515	100.498	117.525	117.4988	117.285

TABLE 5.1-VII.- REENTRY TRAJECTORY ERROR INFORMATION

(a) Terminal coordinates determined from various tracking stations

	Latitude (geodetic), deg North	Longitude, deg West	Target miss distance, n. mi.
Touchdown (STL preliminary tracking)	22.49	70.758	56.2
Section 6 - Integrated trajectory	22.43	70.85	58.4
Touchdown (based on retrieval coordinates)	22.45	70.83	58.5
IGS position at drogue deployment	22.483	70.845	60.6
Target coordinates	22.02	69.88	-

(b) Position error at drogue deployment computed from postulated ascent component errors

	Downrange, ft	Altitude, ft
x accelerometer bias error = 0.01226 fps^2	1 600	1 520
y gyro drift error = -15 deg/hr	-19 000	-143 000
y gyro reference shift of -0.05 deg	-	-20 900
x accelerometer scale factor of 0.085 percent	1 410	1 340

TABLE 5.1-VIII.- EFFECT OF STEERING AND AERODYNAMICS ON MISS DISTANCE

	Latitude (geodetic), deg N	Longitude, deg W	Miss distance, n. mi.	Difference from tracking, n. mi.
1. Target coordinates	22.02	69.88		
2. GT-3 flight (STL preliminary tracking)	22.49	70.758	-56.2	--
3. Simulated GT-3				
(a) Measured bank angle profile (MAC Aerodynamics)	22.48	71.7	-75.9	-19.7
(b) Fixed bank angle profile 50° - 0° (MAC Aerodynamics)	22.27	70.64	-48.1	+8.1
(c) IGS steering				
1. MAC Aerodynamics	22.07	70.25	-25.2	--
2. NASA Aerodynamics	22.124	70.304	-27	--
(d) Back-up roll commands				
1. NASA Aerodynamics	22.714	71.297	-95	--

NOTE: 1. NASA Aerodynamics revised by reiterating an aerodynamic coefficient assuming an aerodynamic total pressure profile.

2. MAC Aerodynamics revised using IGS measured L/D and angle of attack

Mach number	Mission L/D, assumed	Preflight L/D, assumed
16	0.111	0.175
20	0.13	0.200

TABLE 5.1-IX.- CONTROL SYSTEMS OPERATION

(a) Direct mode

Axis	Thruster on time, sec	Stick position when gyro senses acceleration, deg	Stick position when gyro senses deceleration, deg	Acceleration, deg/sec^2	Measured torque, ft/lb	Computed torque, ft/lb
+ Pitch	1.00	+5	+5	+1.7	250	+350
- Pitch	2.08	-2	-1	-4.0	-340	-350
+ Roll	1.04	+2	+5	+2.6	+120	+170
-Roll	1.25	-3	-1	-4.0	-150	-170
+ Yaw	1.04	+1	+1	+4.9	+350	+350
- Yaw	1.25	-3	-1	-4.0	-300	-350

(b) Rate command mode

Axis	Control stick commanded rate, deg/sec	Final spacecraft rate, deg/sec (a)	Measured torque, ft-lb	Computed torque, ft-lb
+ Pitch	+9	+3	+357	+352
- Pitch	-9	-2.2	-337	-352
+ Roll	+9	+2	+179	+172
- Roll	-	-	-	-172
+ Yaw	+9	-0.2	+325	+352
- Yaw	-9	-4.4	-330	-352

[a] When control stick returned to zero.

TABLE 5.1-X.- NAVIGATION AND GUIDANCE ERRORS

(a) IGS navigation errors

	Position error, ft			Velocity error, ft/sec					
	ΔX	ΔY	ΔZ	$\Delta\dot{X}$		$\Delta\dot{Y}$		$\Delta\dot{Z}$	
GE Mod III (at BECO)	1950	-430	-620	8	(12.5)[a]	-1.5	(-1)	-1	(-0.5)
MISTRAM (at BECO)	1900	-400	-600	8	(10.0)	-0.5	(0.5)	-0.5	(-0.2)
GE Mod III (at SECO)	5100	-5300	-900	36	(40)	-39	(-140)	-1	(-2.0)
MISTRAM (at SECO)	5000	-4800	-825	39	(43.0)	-120	(-124)	-1.5	(-1.3)

[a] Numbers in parenthesis are IMU error contribution.

(b) Total guidance errors at SECO + 20 sec

Computer axis	Total guidance position errors, ft	Errors at SECO + 20 velocity errors, ft/sec
X	5500 ± 300	39 ± 2
Y	-6130 ± 100	-125 ± 6
Z	-820 ± 100	-1 ± 1

TABLE 5.1-XI - SUMMARY OF ASCENT GUIDANCE SYSTEM ERRORS

	Computed	3σ spec values
x accelerometer bias	520 ppm g/g	300 ppm g/g
x accelerometer scale factor	850 ppm	360 ppm
Platform misalinement about y (vertical)	-13 sec	1 min (inflight)
y gyro drift (initiated at 190 sec)	18.4°/hr	0.5°/hr
X navigation equation approximation errors	4 ft/sec	

[redacted]

TABLE 5.1-XII.- PRELIMINARY ORBIT INJECTION PARAMETERS AT SECO +20 SECONDS

System	Inertial velocity, fps	Inertial flight-path angle, deg	Inertial velocity components (computer coordinates)		
			$\dot{X}$	$\dot{Y}$	$\dot{Z}$
Nominal	25 697	+0.0098	25 288	4 566	17.6
IGS	25 697	+.32	25 325 ±1	4 355 ±1	-216 ±1
STL preliminary BET	25 682	+.01	25 285 ±1	4 487 ±1	-214 ±0.5
STL MISTRAM 100 000	25 682	+.01	25 285 ±1	4 487 ±5	-214 ±0.5
STL GE Mod III	25 687	-.06	25 286 ±5	4 518 ±30	-215 ±0.2
Goddard GE Mod III	25 688	-.69			
MISTRAM (IP)	25 694	-.144			
Reconstructed from Guaymas first orbital pass	25 682	+.036			

[redacted]

TABLE 5.1-XIII.- GT-3 EVENTS

Parameter	Time, hr:min:sec (a)
AB01 (2nd stage cutoff - IGS)	00:05:33.6
AB02 (spacecraft shaped charge fire)	00:05:59
AB03 (launch vehicle spacecraft separate)	00:05:59.3
Fairing jettison	00:06:20.7
AD01 (adapter shaped charge fire)	04:32:27.6
AD02 (equipment section separate)	04:32:27.7
AD03 (auto retrofire)	[b]04:33:23.1
AD05 (retroshaped charge fire)	04:34:08.6
AD06 (manual retrofire initiate)	04:33:24.2
AD08 (retro no. 3 fire)	04:33:28.2
AD09 (retro no. 2 fire)	04:33:33.4
AD10 (retro no. 4 fire)	04:33:39.4
AE02 (pilot parachute deploy)	04:48:24.4
AE27 (drogue parachute deploy)	04:46:51.4
AE28 (drogue release)	04:48:24.2

[a] Accuracy: -0.1 to +0.2 sec

[b] Accuracy of AD03 is -0.4 to +0.2 sec due to data dropout

TABLE 5.1-XIV.- BLOWN FUSISTORS

Fuse block	Fuse no.	Application
XF-G	(4-22)	Adapter equipment wire pyro switch E-1
XF-M	(4-22)	Wiring pyro switch B-1
XF-M	(4-27)	Wiring pyro switch C-1
XF-M	(4-53)	Adapter equipment wire pyro switch J-1
XF-N	(4-20)	Power wire guillotine D-1
XF-W	(4-46)	Adapter equipment wire pyro switch D-2

NASA-S-65-3593

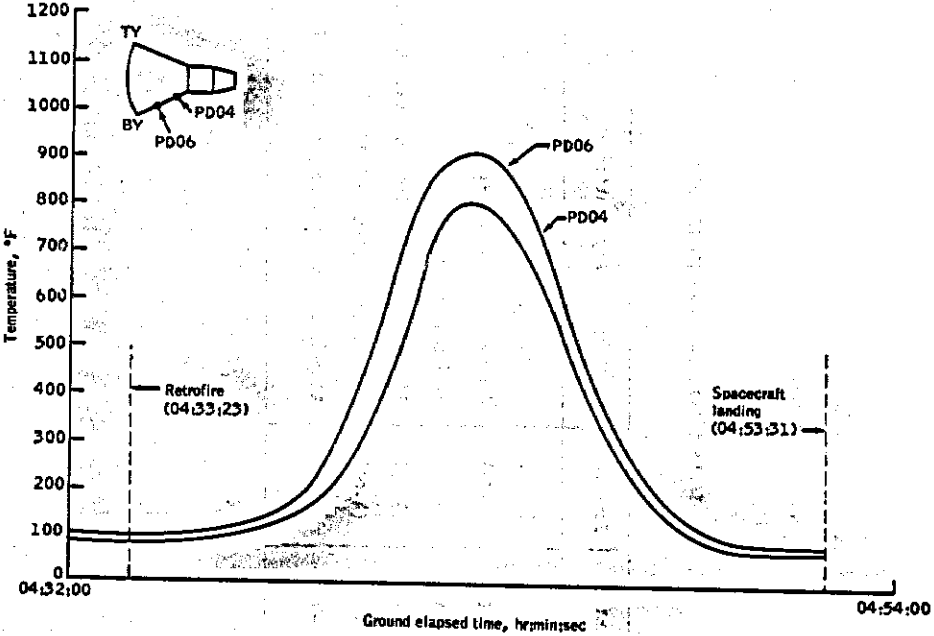


Figure 5.1-1. - Cabin section - reentry shingle temperatures

NASA-S-65-3592

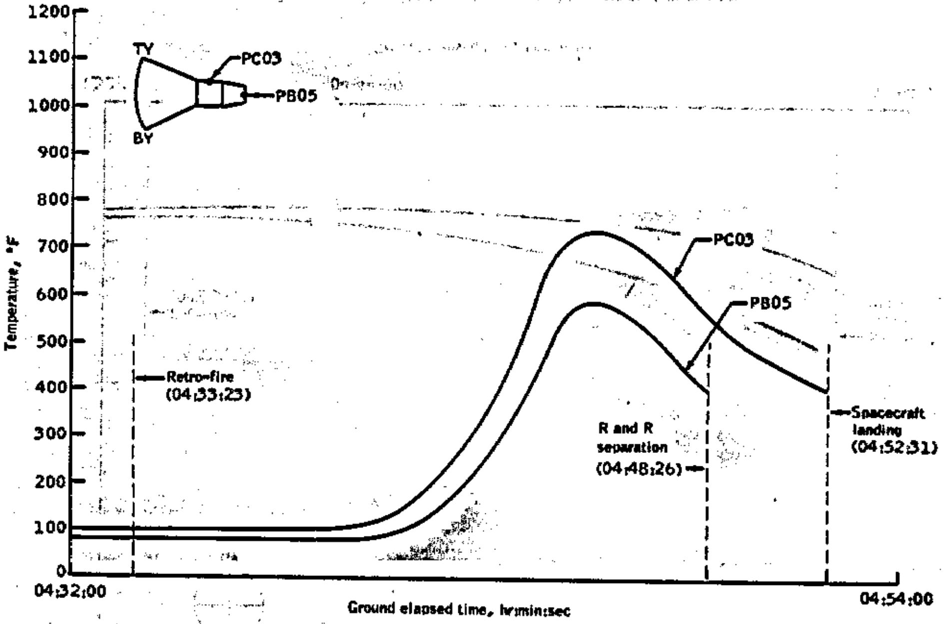


Figure 5.1-2. - Reentry control system and rendezvous and recovery section reentry shingle temperatures

NASA-S-65-3594

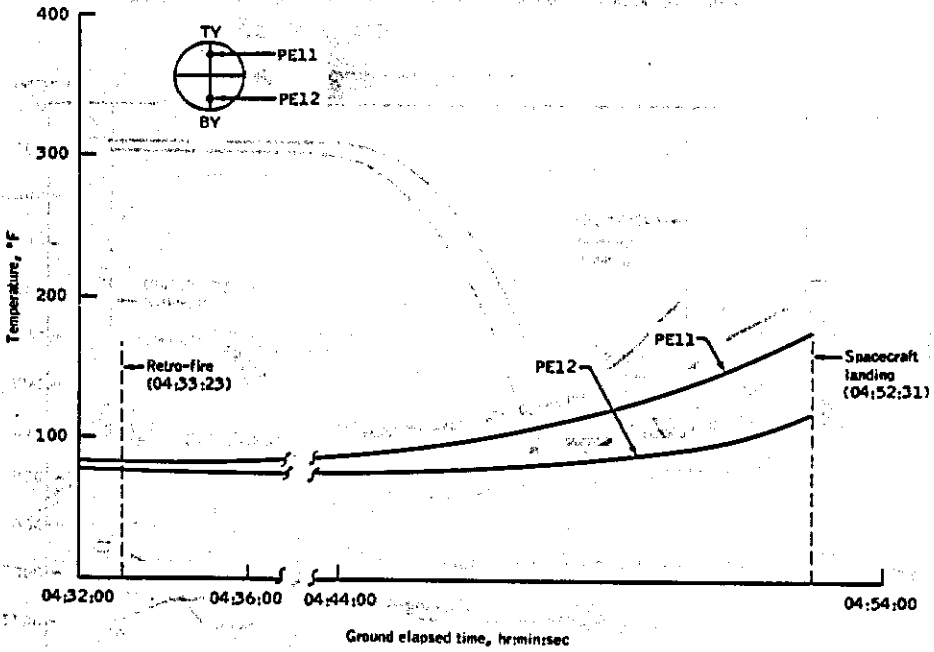


Figure 5.1-3. - Ablation material bond line - reentry temperatures

NASA-S-65-3604

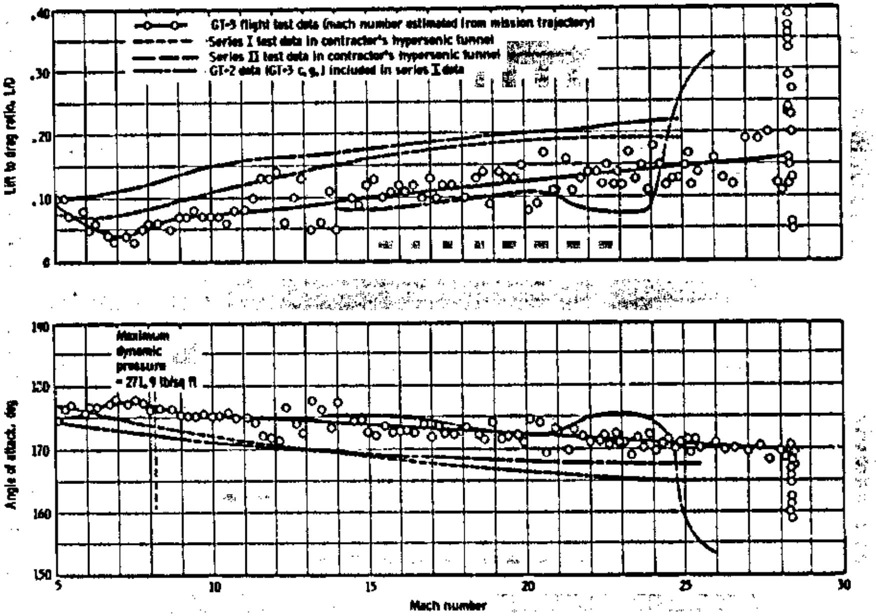


Figure 5.1-4. - Reentry aerodynamic parameters

NASA-S-65-3622

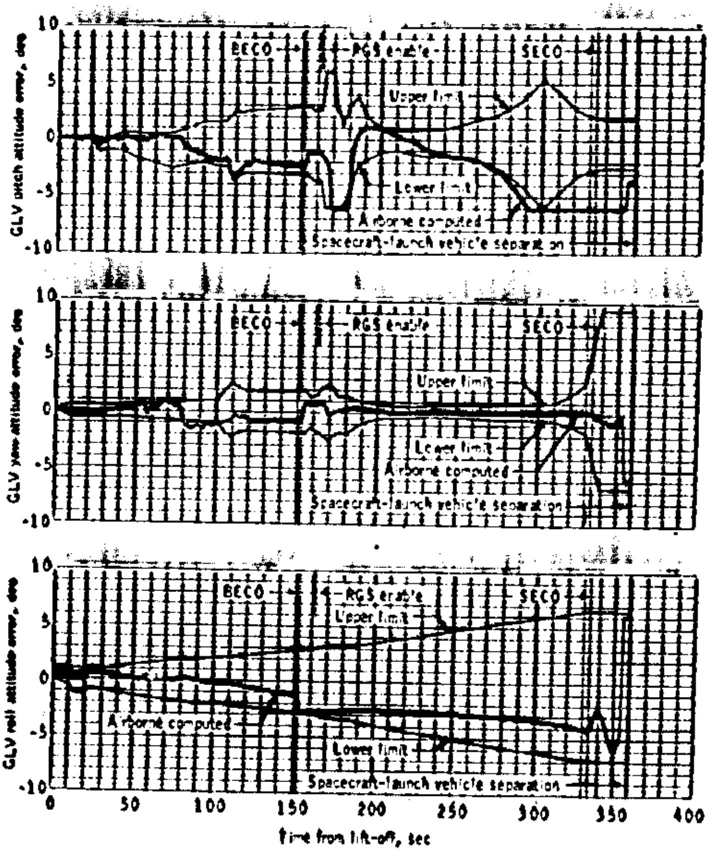
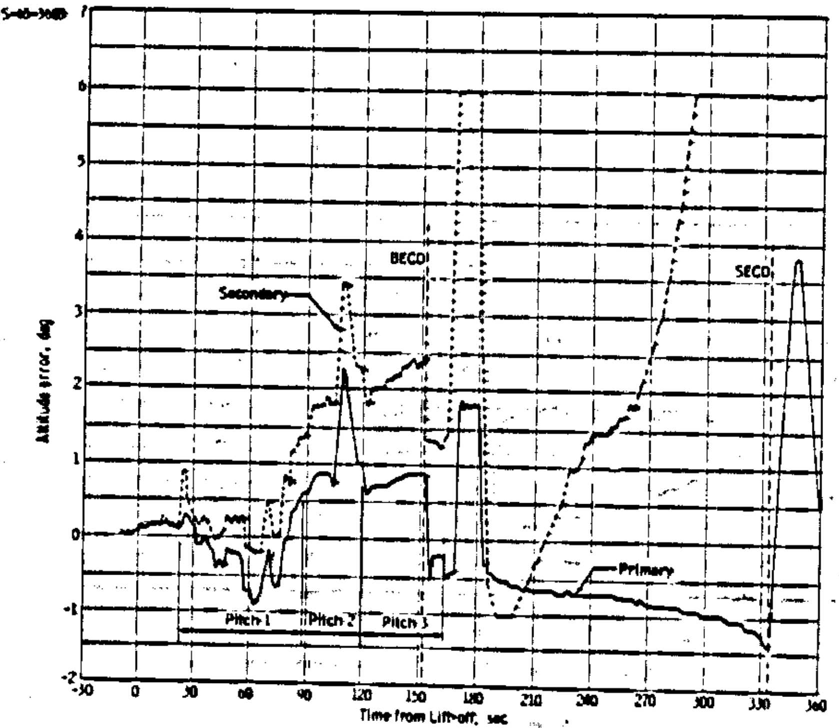


Figure 5.1-5. - Comparison of steering commands during launch with preflight value

NASA-S-65-3680

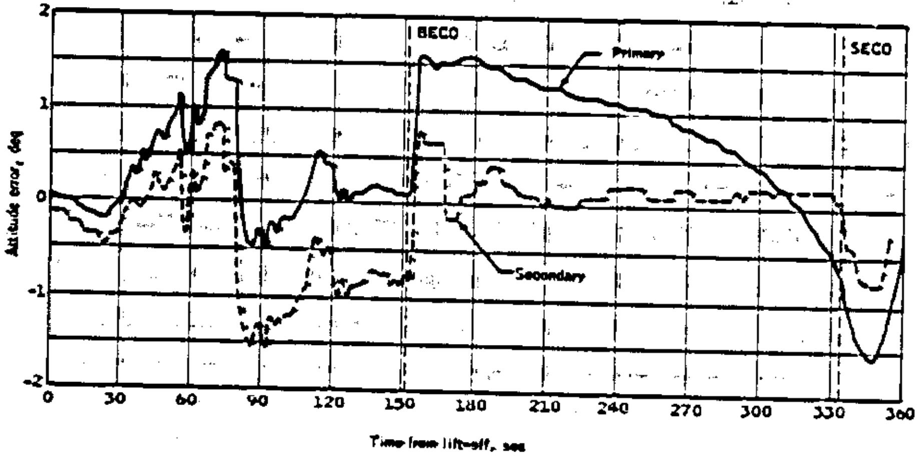


Time from Lift-off, sec

(a) Pitch attitude error

Figure 5.1-6.- Attitude error comparison

NASA-S-65-3601

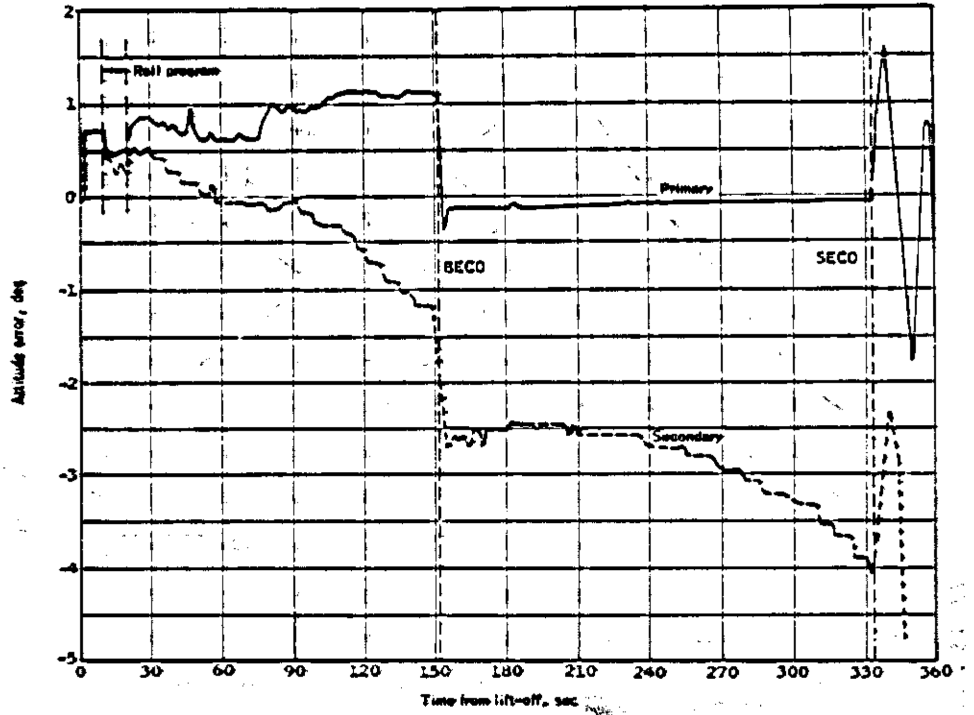


(b) Yaw attitude error

Figure 5.1-6.- Continued

NASA-S-65-3602

[redacted]



(c) Roll attitude error

Figure 5.1-6. - Concluded

[redacted]

NASA-S-65-3582

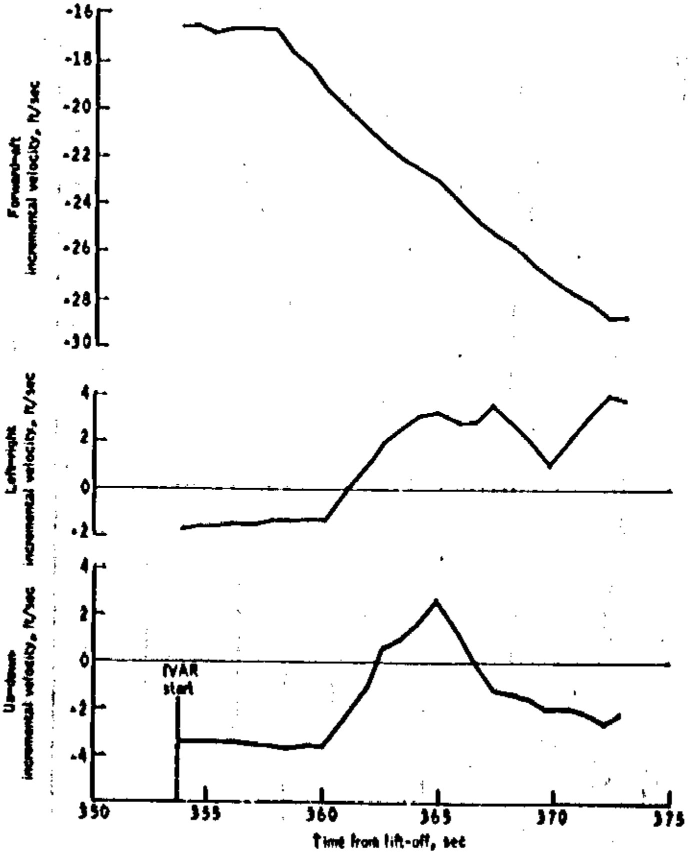


Figure 5.1-7. - Reconstructed IVAR computations.

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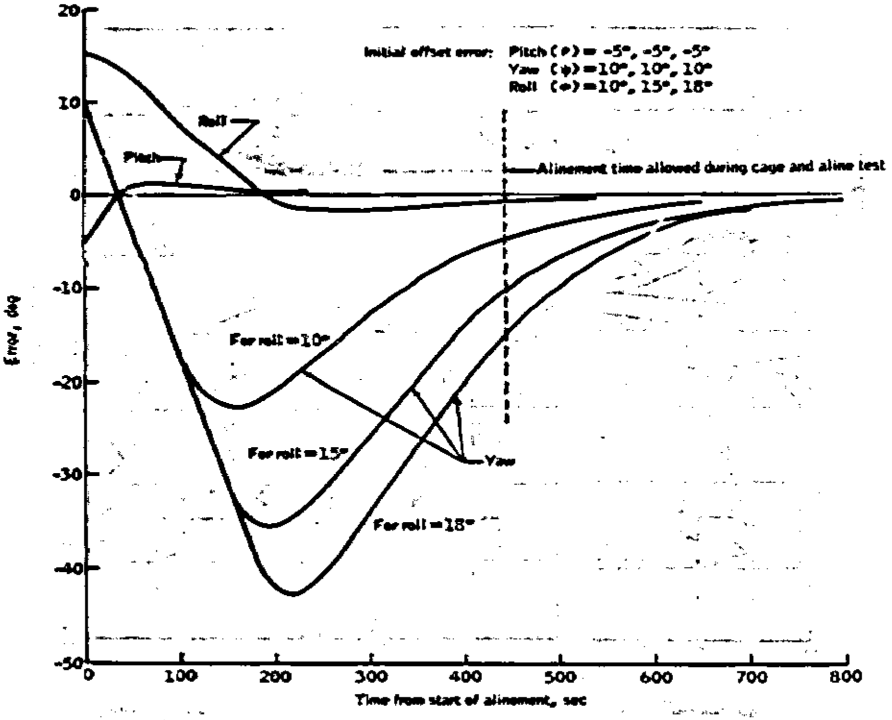

Figure 5.1-8.- Typical platform yaw alinement response from initial offset errors

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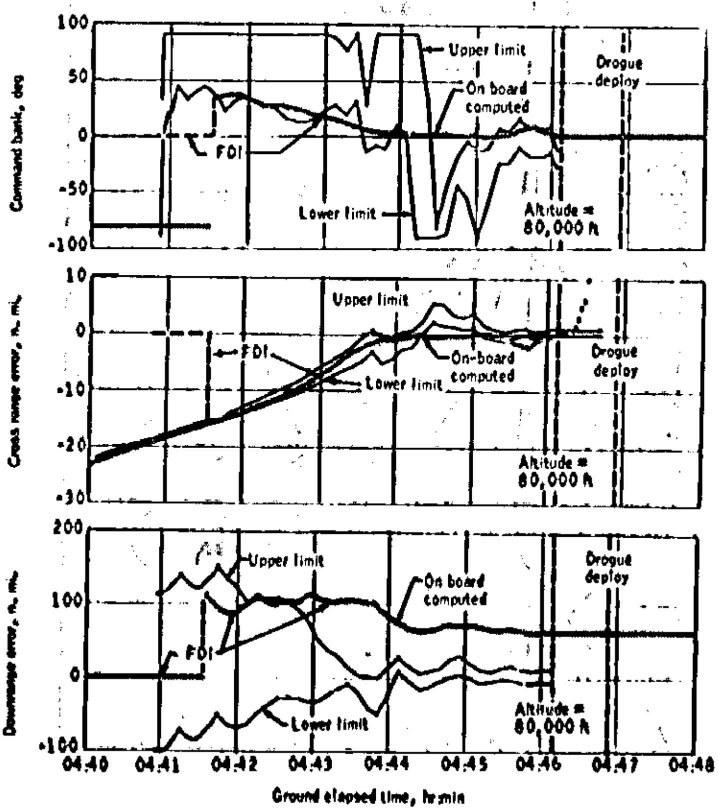

Figure 5.1-9. - Reentry comparisons

NASA-S-65-3630

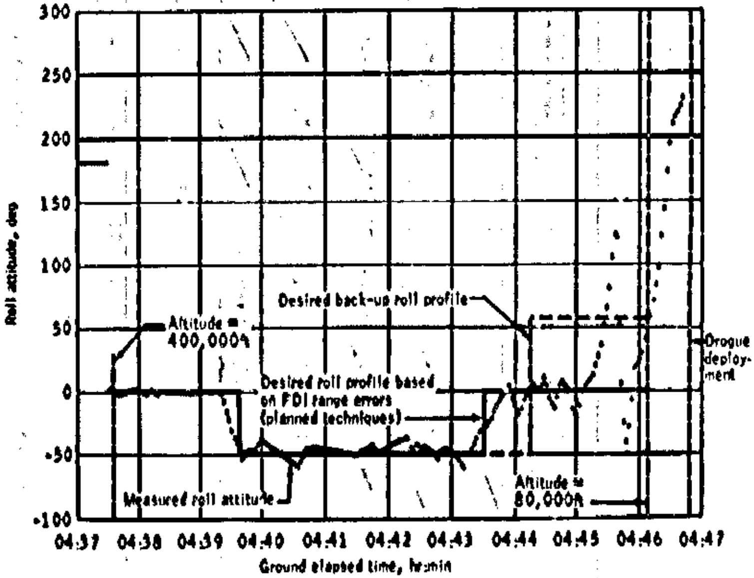

Figure 5.1-10. - Reentry roll angle time history

NASA-S-65-3615

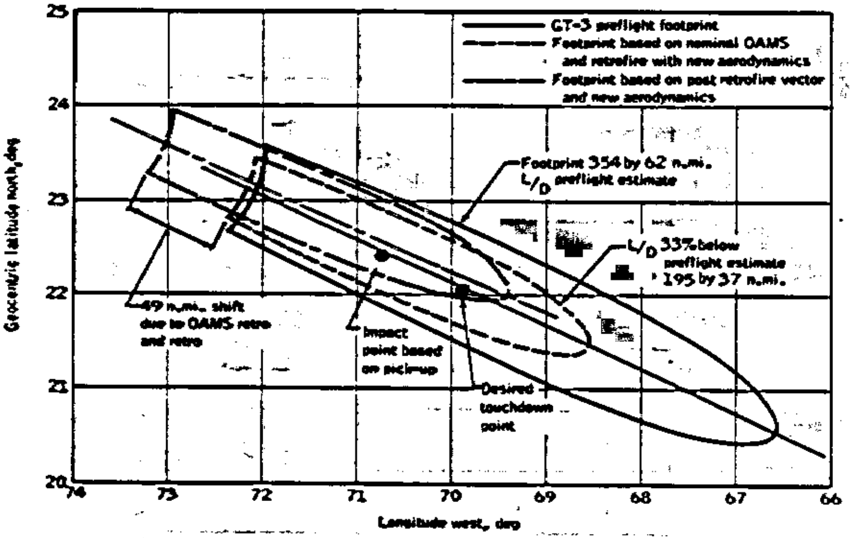

Figure 5.1-11.- Effect of retro and aerodynamics on footprint capability

NASA-S-65-3632

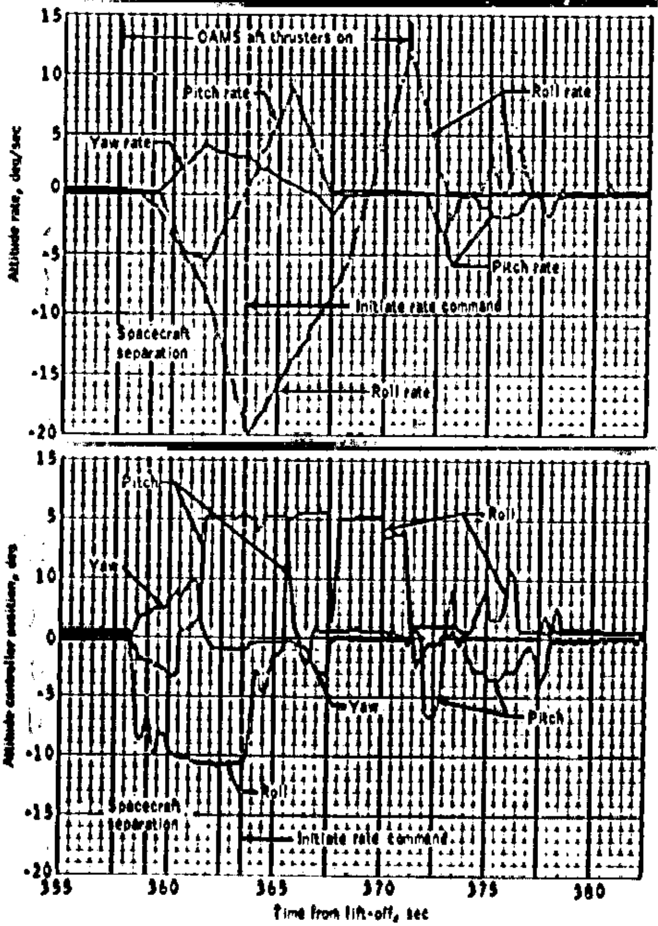

Figure 5.1-12.- Spacecraft separation transients

NASA-S-65-3556

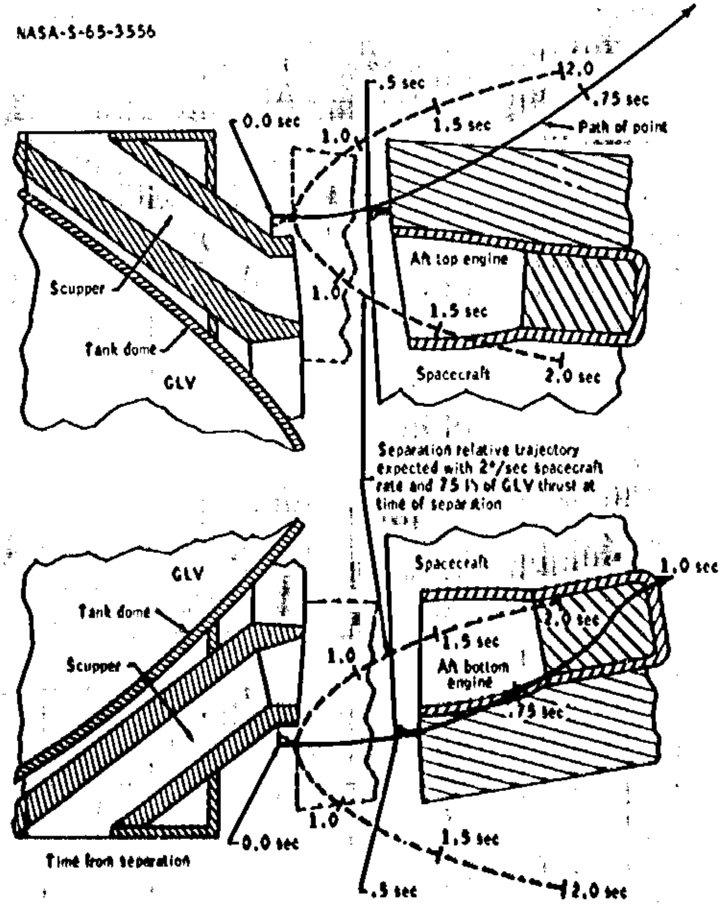

Figure 5.1-13.- Relative positions of GLV and spacecraft after separation

NASA-S-65-3632

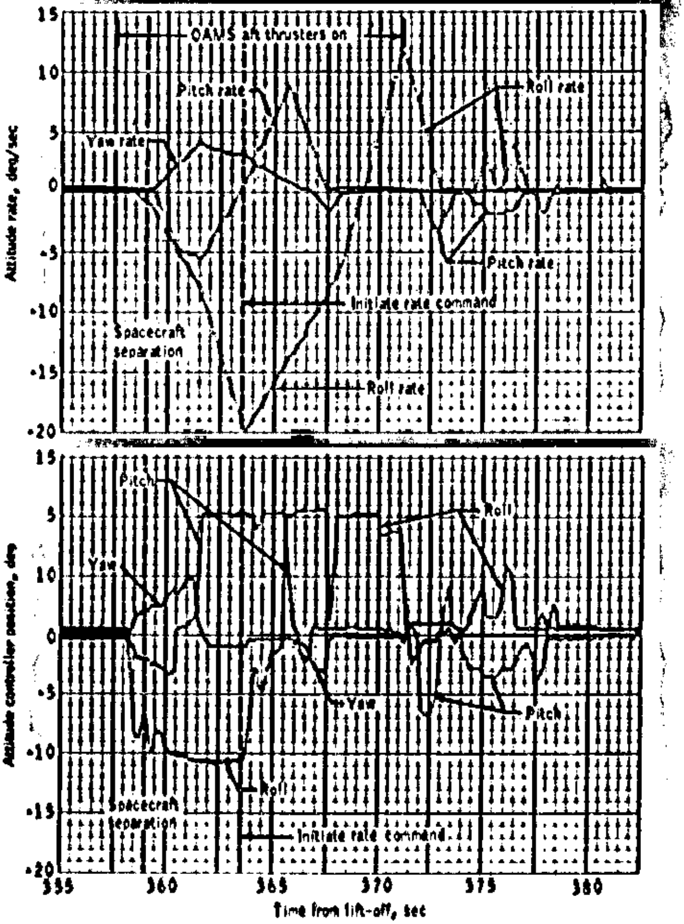

Figure 5.1-12.- Spacecraft separation transients

NASA-S-65-3556

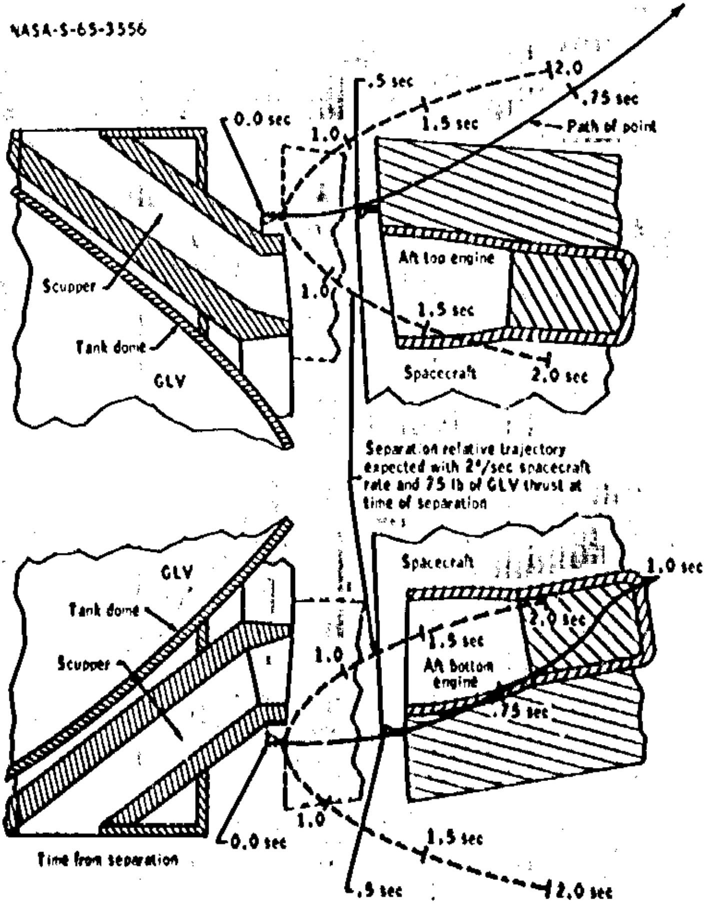

Figure 5.1-13 - Relative positions of GLV and spacecraft after separation

NASA-S-65-3631

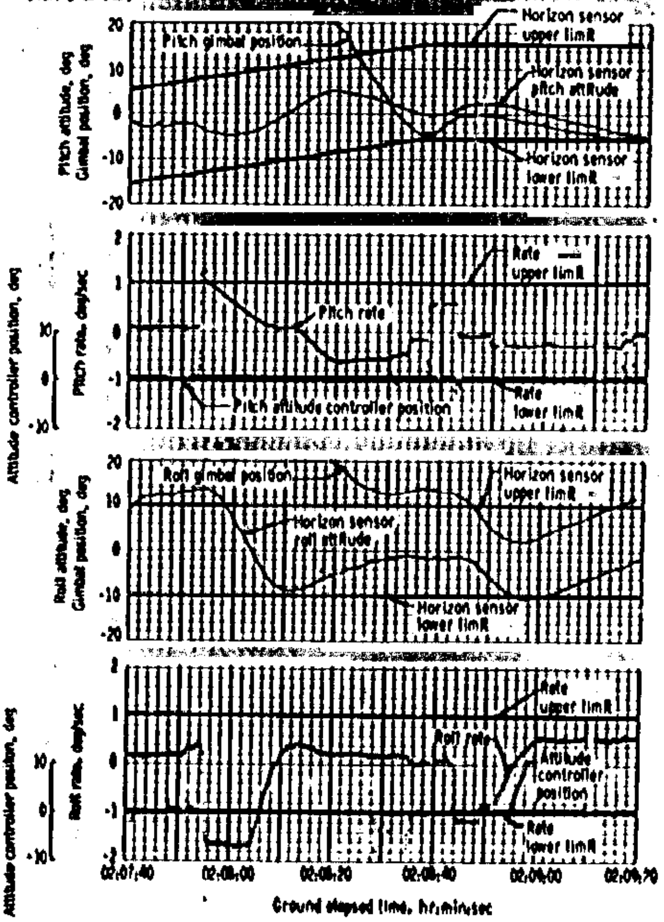

Figure 5.1-14. - Comparison of horizon scanner and spacecraft attitude

[redacted]

NASA-S-65-3646

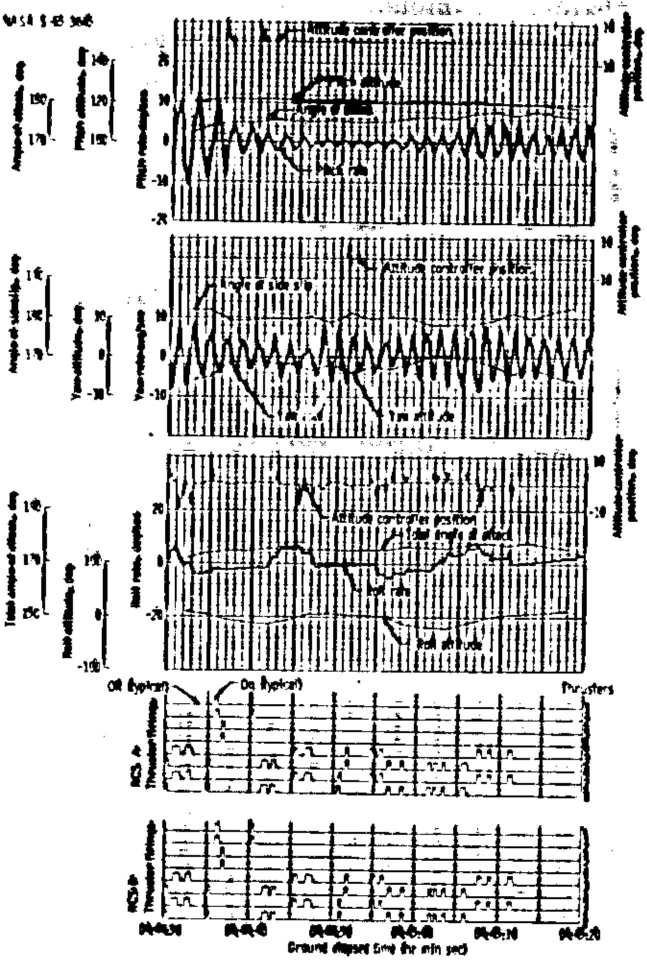

Figure 5.1-21.- Reentry control comparison

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[redacted]

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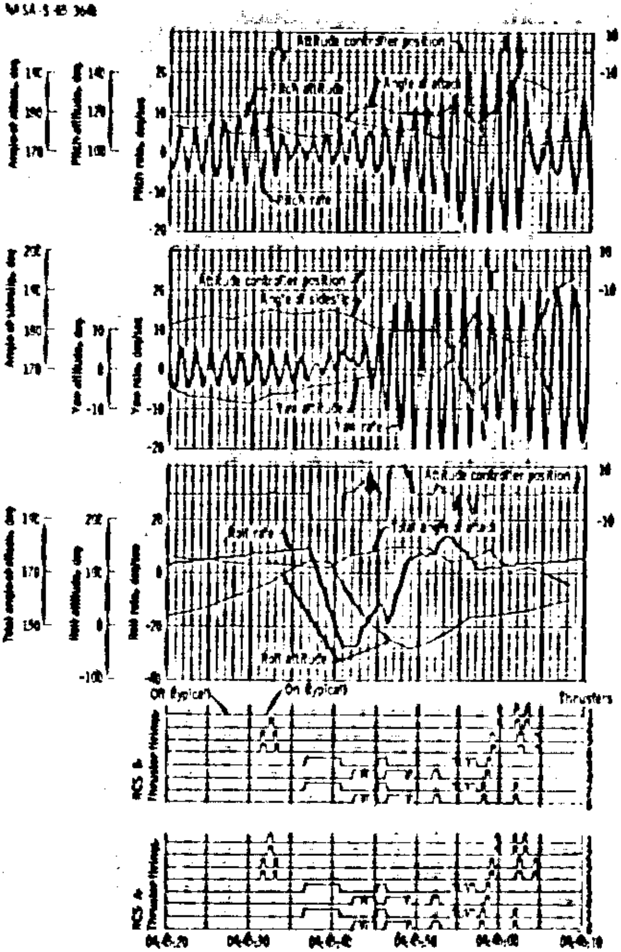

Figure 5.1-15 - Concluded

[redacted]

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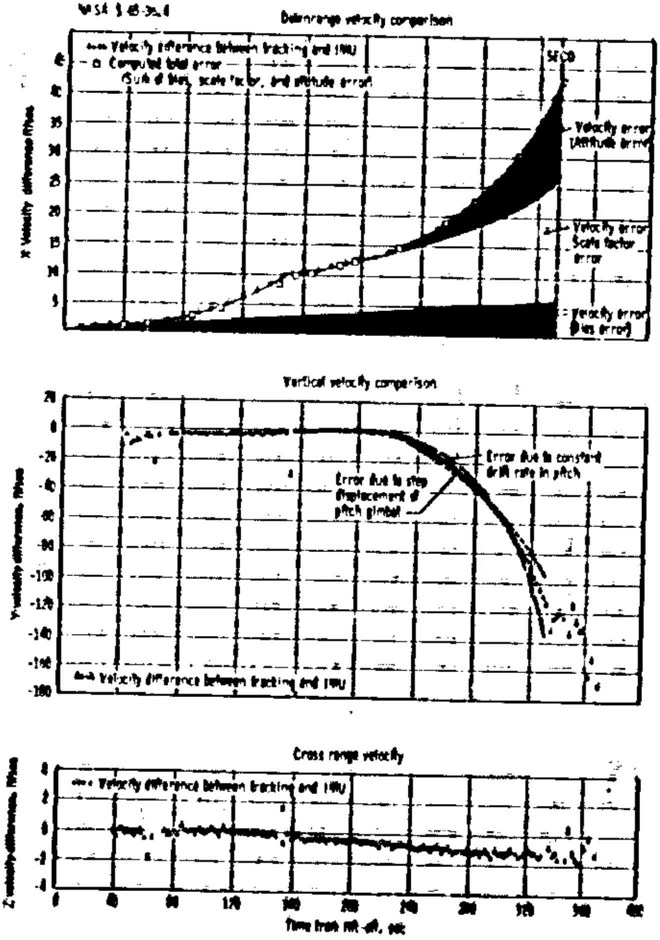

Figure 5.1-16 - Velocity comparisons

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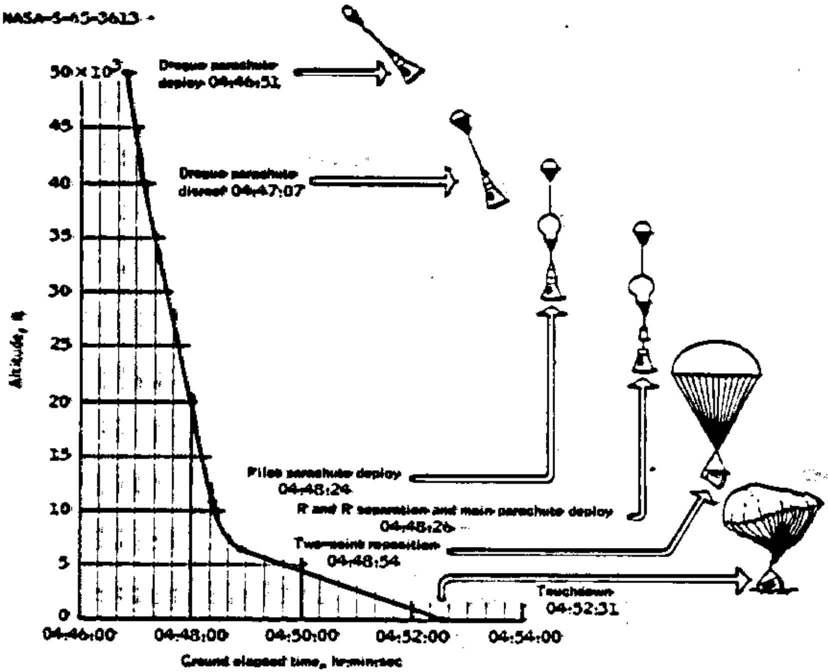

Figure 5.1-17 - Landing system performance

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5.2 GEMINI LAUNCH VEHICLE PERFORMANCE

The performance of the Gemini launch vehicle (GLV) was satisfactory. The lofted first-stage trajectory reported in section 4.0 was caused primarily by thrust, which was higher than predicted and, to a lesser degree, by flight-control errors. The performance of individual systems is discussed in the following subsections.

5.2.1 Airframe

Analysis of GT-3 data indicates that the flight environment was within vehicle design requirements, and that flight loading was well within the structural capability of the launch vehicle. Specific areas of interest are discussed in the following paragraphs.

5.2.1.1 Skin temperature.- Skin temperatures compare favorably with those noted on GLV-2. The GLV-3 levels were approximately 20° F to 25° F higher than GLV-2, primarily because of the higher ambient temperatures at lift-off. The heating effect of air flow across the stage I oxidizer dome was negligible since the temperature rise was only 27° F between 45 seconds g.e.t. and staging at 152.43 seconds g.e.t.

5.2.1.2 Longitudinal oscillation.- A number of well-defined periods of longitudinal oscillation developed after the launch vehicle passed through the transonic region. The frequency of the oscillation ranged from 9.2 cycles per second at 67 seconds g.e.t. to 18 cycles per second at BECO. The peak response amplitudes indicate that the oscillation was maximum at 131.5 seconds g.e.t., and reached a magnitude of ±0.146g at a frequency of 12 cycles per second.

5.2.1.3 Vibration environment.- The vibration measurements during stage I flight indicate that the environment was well within the qualification requirements ($14.3g_{rms}$). The lateral vibration level of the radio guidance system (RGS) equipment mount was $0.96g_{rms}$ at lift-off and $0.95g_{rms}$ during the transonic period. The vertical vibration level was $2.09g_{rms}$ at lift-off and $2.20g_{rms}$ during the transonic period.

5.2.1.4 Structural loads.- Ground winds were approximately 14 mph during the countdown of GT-3, resulting in a combined static and dynamic bending moment of 435 000 in.-lb at launch vehicle station 1224. This value is approximately 3.6 percent of the vehicle bending capability. Structural responses to ignition transients were approximately the same as observed on previous flights. However, transverse bending loads at release were approximately twice those on GLV-2 but were well within the design loads envelope.

Flight loads were computed using filtered responses of launch-vehicle vibration measurements. These measurements show good correlation with calculated frequencies in figure 5.2-1 and result in the bending moments shown in table 5.2-I.

The total computed load for the maximum αq flight condition (69 sec g.e.t.) was 440 600 pounds equivalent axial load, or 55.8 percent of the launch-vehicle compressive strength at station 935 (critical launch-vehicle station at this time of flight).

Structural loading for the BECO flight condition was determined directly from axial acceleration, longitudinal oscillations, and lateral vehicle responses. Axial acceleration was 5.65g, which is about 2.6 percent below the maximum dispersed value of 5.8g. Longitudinal oscillations were ±0.146g at launch-vehicle station 280. Lateral responses were as shown in figure 5.2-1. The combined effects of these loads produced 269 800 pounds of equivalent axial compression load at station 320, or 77 percent of ultimate design load.

5.2.1.5 Staging.- During the debriefing, the flight crew reported that they saw particles during staging of the launch vehicle. The motion of debris, which normally originates from the area of the interstage structure during the staging function, is radially out from the vehicle, and, since some forward velocity could be imparted to the particles, a momentary sighting is possible. There has been no evidence of damage to the vehicle second stage in past Titan II and GLV flights; therefore, this debris is not considered a problem.

5.2.2 Propulsion System

Performance of the propulsion system was satisfactory.

5.2.2.1 Stage I engine performance.- Table 5.2-II presents a summary of pertinent performance parameters and indicates general agreement with preflight prediction. The thrust level is about 1.5 percent high. This is discussed in section 5.2.2.4. The thrust build-up traces on both subassemblies were well within the range of previous Titan II and Gemini experience. Both subassemblies exhibited a start transient in excess of the specification value of 75 percent of rated thrust. The value recorded includes an instrumentation overshoot that has been noted on all previous Gemini launches. The engine contractor is presently attempting to determine the true value of the transient.

Steady-state engine performance was normal. Analysis of flight data indicates excessive cyclic pressure oscillations in several of the stage I turbopump and gas generator parameters. These oscillations were present in the gas generator chamber pressures of both subassemblies (120 to 135 psi, peak-to-peak), fuel discharge pressure of subassembly 1 (220 psi, peak-to-peak), and oxidizer discharge pressure of subassembly 2 (200 psi, peak-to-peak). Oscillations of a similar nature were present in GLV-2, and investigation to date indicates an instrumentation phenomenon.

Engine shutdown was initiated by oxidizer exhaustion with fuel exhaustion occurring during the shutdown transient.

5.2.2.2 Stage II engine performance.- Table 5.2-III presents a summary of pertinent performance parameters during stage II engine operation and indicates generally good agreement between postflight and preflight calculations. Stage II start transient was normal with a chamber pressure overshoot of approximately 10 psi. Step pressure was approximately 540 psia. The 10-psi overshoot is insignificant, while the step pressure is well above specification minimum. Stage II steady-state performance was normal with the exception of low-frequency pressure oscillation during the period from approximately 210 seconds to 260 seconds, g.e.t. This oscillation occurred during the time of minimum oxidizer pump suction pressure. The pressure fluctuations were most severe in the oxidizer circuit but were present to a lesser degree in the fuel and gas generator circuits. Thrust chamber pressure oscillations were approximately 35 psi, peak-to-peak, at 5.5 cps. This phenomenon has been noted on a number of Titan II flights, with the magnitude encountered on this flight being about 50 percent of the maximum previously seen. This oscillation appears to have no detrimental effect on system performance, and discussion with the flight crew indicates that they could not sense the oscillation.

Stage II shutdown was accomplished by simultaneous operation of the pressure sequencing valve and the redundent engine shutdown valve. The shutdown transient was generally of the expected form with a sharp drop-off, but had a slightly lower total impulse than was predicted (38 000 lb-sec, actual, 41 000 ± 10 000 lb-sec, predicted).

5.2.2.3 Propellant and autogenous system performance.- The following tables show good agreement of predicted data with loaded propellant weights and predicted data with flight average propellant temperature.

Propellant Loading

Component	Stage I		Stage II	
	Predicted, lb	Actual, lb	Predicted, lb	Actual, lb
Oxidizer	171 676	171 689	38 165	38 161
Fuel	90 440	90 458	21 959	21 957

Average In-flight Propellant Temperatures

Component	Stage I		Stage II	
	Predicted, °F	Actual, °F	Predicted, °F	Actual, °F
Oxidizer	43.6	44.0	46.4	44.8
Fuel	42.8	44.0	40.9	41.3

A comparison of predicted and actual tank pressures shows excellent agreement and is indicative of proper tank pressurization system performance.

5.2.2.4 High staging altitude.- An investigation of the cause for the high BECO altitudes experienced on the GT-3 flight and the previous Gemini flights indicates thrust levels higher than predicted. Trajectory matching of GT-1 and GT-2 in conjunction with seven previous Titan II flights has indicated that a stand-to-flight thrust bias must be added to the existing engine simulation. These biases, approximately 8200 and 900 pounds thrust for stage I and II, respectively, and a correction of 1.7 seconds to stage I specific impulse (I_{sp}) will be added to the existing engine simulation prior to the calculation of the final GT-4 trajectories. A revised pitch program will be utilized on GT-4 to account for the higher thrust.

5.2.2.5 Performance margin.- The achieved GLV-3 payload capability was 8510 pounds. This represents an excess of 1399 pounds beyond the 7111-pound spacecraft which was placed into orbit. For comparison, the nominal payload capability for GLV-3, assuming launch at the end of planned countdown on a hot spring day at Cape Kennedy, was 8313 pounds. Thus, the difference between achieved and predicted payload capabilities was 197 pounds.

Additional burning time remaining at SECO was 3.96 seconds.

5.2.3 Flight Control System

GT-3 was controlled in the primary mode throughout stage I and stage II flight. All system parameter values were nominal. The flight control system's probable contribution to the high BECO altitude was caused by the three-axis reference system (TARS) pitch program which was 1.2 percent low, and a positive pitch gyro drift of 3.6 deg/hr (based on preflight laboratory data). These two items could account for approximately 4000 feet of the observed altitude error.

5.2.3.1 Stage I flight.- Maximum gimbal displacements and rates at ignition were normal. The values are listed in table 5.2-IV. Lift-off motion resulted in a peak engine roll motion of 0.13°. The maximum roll transient was ±1.55 deg/sec and occurred at 0.3 second g.e.t. This transient was damped out in 1.5 seconds. The flight control system response was proper, resulting in a stage I lift-off bias of 0.7°, clockwise.

The roll and pitch programs were properly executed and were close to nominal in rates and duration. All TARS-initiated discretes were executed and occurred close to the nominal times. Values for respective rates and times are given in section 5.2.5, (table 5.2-VII.)

The largest vehicle attitude errors and rates during the maximum dynamic pressure are listed in table 5.2-V.

A comparison between primary and secondary attitude errors, as shown in figures 5.1-6 (a) to (c), revealed satisfactory system performance during stage I. The differences noted were caused primarily by respective drifts and inter-axis cross-coupling effects. In addition, attitude errors were established to counteract aerodynamic forces because of a predominantly westerly wind distribution aloft.

The staging sequence was normal. As soon as the stage II hydraulic system pressurized and thrust-vector control was attained, damping of staging effects was effective. Maximum rate indications during staging were:

Attitude	Peak rate, deg/sec	Time, BECO + sec
Pitch	-1.5	1.2
Yaw	+1.5	1.6
Roll	-2.0	0.7

The maximum staging attitude errors were as follows:

Attitude	Error, deg	Time, BECO + sec
Pitch	-0.59	2.2
Yaw	+1.8	2.4
Roll	-0.48	1.2

5.2.3.2 Stage II flight.- Pitch and roll oscillations were damped approximately 1 second after BECO, and yaw oscillations within 3 seconds after BECO. The initiation of stage II powered flight caused the disappearance of stage I roll bias and revealed the biases existing for the stage II configuration. Because of normal structural deformation, pitch and yaw thrust biases were developed and maintained throughout sustainer flight. The magnitudes were -0.6° vehicle pitch-down and +1.4° yaw right. Roll was negligible.

The three-axis reference system (TARS) properly issued the radio guidance system (RGS) enable discrete at 162.31 seconds g.e.t. At L0+168.28 seconds, a small steering command of 0.5 deg/sec pitch-down preceded a maximum pitch-down command of 2.0 deg/sec. These commands were received and properly executed. Following this maximum steering-rate command and throughout the remainder of the flight, the primary flight-control system responded satisfactorily to the small radio guidance system pitch and yaw commands. For the remainder of stage II, flight, the primary system sensed perturbations of the vehicle center-of-gravity shift as expected, and issued proper commands to maintain attitude stabilization.

The differences between primary and secondary attitude errors, as shown in figures 5.1-6 (a) to (c) indicate normal performance with the exception of the abnormally high positive inertial guidance system (IGS) secondary system pitch error. (For further information on the IGS performance during ascent see section 5.1.5). Excluding this anomaly, combinations of normal drift rates, vehicle center-of-gravity shift, and thrust biases produced the indicated differences.

5.2.3.3 Post-SECO.- The vehicle turning rates for the 20-second flight period after SECO were quite low as shown in table 5.2-VI.

5.2.4 Hydraulic System

The launch vehicle hydraulic system operated satisfactorily. As a result of suspected hydraulic pump-compensator sticking during the GT-2 flight (in which the stage I primary hydraulic-system pressure dropped to near the switchover pressure limit), newly cleaned pumps were installed in both the primary and secondary hydraulic systems of stage I and in the hydraulic system of stage II of GLV-3. Special precautionary procedures were used to prevent contamination from entering the hydraulic system during test or sampling of the system. Prior to launch, a gauss-meter was used to observe that the pressure compensators on the pumps were operable.

5.2.4.1 Stage I primary system.- The data obtained indicate that the use of the standard pump compensator in the GLV-3 primary hydraulic system reduced the drop in pressure and the pressure oscillation that occurred on GLV-1 which had a short differential pump compensator installed. A comparison of the stage I primary hydraulic pressures obtained during the launch of GLV-1, GLV-2, and GLV-3 is shown in figure 5.2-2.

At 110 seconds prior to ignition, the electric motor pump pressurized the primary system to a normal system operating pressure of 3265 psia. As shown in figure 5.2-2, the engine start transient, beginning 0.77 second after engine ignition, produced flow demands that dropped primary pressure to 2596 psia. Pressure recovery occurred immediately, indicating proper engine-driven pump-compensator response. On recovery, the pressure peaked at 3314 psia with a steady-state pressure of 3065 psia being reached within 1.7 seconds after engine ignition. The pressure decayed normally during flight to 2900 psia at staging.

5.2.4.2 Stage I secondary system.- The use of the standard pump compensator on the GLV-3 secondary hydraulic system also eliminated the high pressure peak obtained on GLV-1 when the short differential compensator was used and verifies the results obtained on GLV-2. A comparison of the stage I secondary hydraulic pressures obtained during the launch of GLV-1, GLV-2, and GLV-3 is shown in figure 5.2-3.

Hydraulic pressure began to develop immediately after engine ignition. The pressure overshoot reached a maximum of 3137 psia, indicating very good pump compensator response. A steady-state pressure of 2980 psia was reached within 1.2 seconds after engine ignition. The secondary system was capable of fulfilling the hydraulic requirements if a switchover had occurred.

5.2.4.3 Stage II system.- Because the stage II hydraulic pump with the short differential compensator had given satisfactory performance on Titan II and previous Gemini flights, the use of this pump assembly on stage II was continued on GLV-3.

The hydraulic pressure reached a peak of 3680 psia within 1.7 seconds after engine ignition. A steady-state pressure of 3000 psia was reached within about 8 seconds after engine ignition, but decreased to 2900 psia at SECO. No significant pressure perturbations occurred during the flight.

5.2.5 Guidance System

The vehicle was guided by the primary M [illegible] radio guidance system (RGS) which performed satisfactorily throughout the countdown and flight.

5.2.5.1 Programed guidance.- The programed guidance system for the first 162.31 seconds after lift-off consisted of sequenced events in the roll and pitch channels provided by the primary flight-control system. The sequenced events, as shown in table 5.2-VII, occurred within the acceptable limits.

As discussed in section 4.0, a lofted (dispersed) first-stage trajectory was flown. The errors at BECO (see section 4.4.1) were approximately 2σ and amounted to 96.00 ft/sec high in velocity, 13 114.0 feet high in altitude, and 2.23° high in flight-path angle.

5.2.5.2 Closed-loop 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 LO+394.6 seconds. At this time, track went into a period of intermittent lock until final loss of signal at LO+403.9 seconds (70.2 seconds after SECO). Track was maintained to an elevation angle of 2.0° above the horizon. The average received signal strength at the central station during second stage operation was satisfactory, and azimuth tracking errors were small.

Rate lock was continuous, except for a momentary interruption at staging, from LO+45.2 seconds to LO+394.4 seconds (60.7 seconds after SECO). Lateral rate noise during the latter portion of flight was less than 0.022 ft/sec.

Normal steering commands were transmitted, as planned, from the ground computer at LO+168.0 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 12.8 seconds. The steering gradually returned to a relatively small and constant pitch-up command of 2.5 percent 5.5 seconds later. This produced a continuous pitch rate of 0.05 deg/sec until SECO minus 2.5 seconds.

Yaw steering started at LO+168.0 seconds. The yaw commands were of very small magnitude, with the commands over the closed-loop portion of flight amounting to positive and negative yaw rates of from 0.04 to 0.06 deg/sec (2 to 3 percent).

SECO occurred at LO+333.747 seconds which was 4.67 seconds earlier than planned, and at an elevation angle of 6.92° as compared with a planned elevation angle of 6.62°. The auxiliary sustainer engine cutoff (ASCO) signal was sent at LO+333.789 seconds via the range safety command transmitter.

The resultant SECO+20 second conditions, although dispersed, were well within 3σ limits. The flight-path angle was 0.04°, the velocity was 25 682 ft/sec, and the altitude was 87.5 nautical miles. As shown in table 4-II, the flight-path angle was 0.04° high, the velocity was 17 ft/sec low, and the altitude 376 feet high. An investigation of the cause for the lower-than-planned velocity is in progress. Approximately 8 to 10 ft/sec has been attributed to the lower-than-predicted tail-off impulse of the redundant engine shutdown system discussed in section 5.2.2. The remainder is attributed to guidance and network tracking errors, non-linearity in specific impulse, gyro drifts, scale factor errors, thrust misalinements, and shifting center of gravity. At the end of tail-off, tumbling velocities were -0.40 deg/sec pitch down, 0.20 deg/sec yaw-right, and 0.3 deg/sec roll-clockwise (CW).

The computing system in conjunction with the RGS track, rate, and airborne systems, completed all prelaunch and launch operations in a normal and satisfactory manner.

The inertial guidance system (IGS) updates were sent from the computer and verified by the buffer as follows:

Update sent, time from lift-off, seconds	Update verified, time from lift-off, seconds	Value, ft/sec
100.0	105.01	-348.5
140.0	145.01	-199.0

These transmission times provided the 5-second delay required by the spacecraft inertial guidance system and the digital command system verification by telemetry.

In figures 5.2-4 and 5.2-5, the velocity and flight-path angle are shown in the regions of SECO and tail-off. The launch-vehicle radio guidance system data and the range safety computer (IP 3600) data are shown to illustrate the quality of the post-SECO data used for the orbital determination (go-no-go). It is noticed that the range safety velocity and flight-path angle data experienced an anomaly during tail-off. This was caused by the logic of the range safety real-time program, which, because of internal data quality checks, switched from MISTRAM I long-base-leg data to MISTRAM I short-base-leg data.

5.2.6 Electrical System

5.2.6.1 Ground.- Both the accessory power supply (APS) and the instrument power supply (IPS) power transfers were normal. Voltage deviations of approximately 1 V occurred. The umbilical-drop sequence was as planned and the respective separation times are shown below:

Umbilical	Disconnect, LO + sec	Umbilical	Disconnect, LO + sec
3 D1M/3 D2M	Lift-off	3 B1E	0.657
3 D1E	0.174	2 B1E	.797
3 D2E	.416	2 B2E	.806

The presence of an apparent 4.2-A drop in the IPS at lift-off is attributable to the temporary bypassing of the IPS shunt caused by the umbilical pull sequence in conjunction with the AGE-IPS ground.

5.2.6.2 Airborne.- All programed events, such as staging, SECO, and spacecraft separation, were accomplished without incident and were normal. The APS and IPS functioned normally throughout the flight. No high current transients at staging were experienced on this flight as they were on the GT-2 mission. This indicated that no squib wires or separation nut wires were shorted to the structure. As a result of the spacecraft pyro-cutter separating the interface wiring, a small transient was imposed on the APS (12 A) and the IPS (9 A) at spacecraft separation.

Throughout the launch sequence, no anomalies or disturbances existed on the 115-V, 400-cps static inverter. The 25-V dc supply, the 5-V dc instrumentation supply, and the 26-V, 800-cps power source maintained a relatively constant level during the flight.

5.2.7 Instrumentation System

5.2.7.1 Ground.- During propellant loading and launch countdown, 107 measurements were monitored and recorded by the Complex 19 ground instrumentation equipment. Parameter assignments were 44 landline measurements on chart recorders, 22 landline measurements on magnetic tape, and 41 airborne (PCM) real-time measurements on chart recorders. All ground measurements performed as expected with no anomalies.

5.2.7.2 Airborne.- There were 242 measurements programed (184 PCM analog, 46 PCM bilevel, and 12 FM-FM measurements). Erratic tracking from Telemetry Building II (Tel II) at Cape Kennedy Missile Annex has been verified by review of signal strength records. This same problem existed during the GT-2 launch.

Data acquisition was 100 percent, and the only signal perturbations were on measurements 0029 and 0517.

(a) Measurement 0029 (bootstrap fuel venturi inlet pressure SA-2) - At LO+92.585 seconds, the signal level dropped from 4.2 to 1.1 V-telemetry for 1 second. A review of the signal characteristics indicates a short-duration controlled change. Further investigation is in progress.

(b) Measurement 0517 (bootstrap fuel venturi inlet pressure SA-3) - During the entire GLV flight, this measurement displayed periods of oscillation and/or dropout. The dropout was especially discernable during stage II powered flight when the transducer was pressurized.

There was a total of 41 periods of dropout, some of which lasted 0.5 seconds. The remaining data are considered accurate. A review of signal characteristics indicates an intermittent transducer problem.

5.2.8 Malfunction Detection System

Performance of the malfunction detection system (MDS) during preflight checkout and flight was satisfactory. The analysis of flight data indicated that all MDS hardware functioned properly. The values of the MDS parameters are shown in table 5.2-VIII.

5.2.8.1 Engine MDS.- The malfunction detection thrust chamber pressure switch (MDTCPS) actuation times have been evaluated and are listed in the following table. The stage I engine subassembly 1 and subassembly 2 switches actuated at 573 psia and 580 psia, respectively. The stage II malfunction detection fuel injector pressure switch (MDFJPS) pressure cannot be determined, because there is no analog telemetry channel of injector pressure.

The switch actuation times and corresponding pressures were as follows:

Switch	Condition	Actuation time from lift-off, sec	Pressure, psia
Subassembly 1 MDTCPS	Make Break	-2.45 +152.40	573 568
Subassembly 2 MDTCPS	Make Break	-2.36 +152.38	580 575
Subassembly 3 MDFJPS	Make Break	+153.08 +333.89	

5.2.8.2 Airframe MDS.- The MDS rate switch package (RSP) performed properly throughout the flight. No vehicle overrates occurred from lift-off through SECO + 20 seconds.

The tank pressure transducers performed satisfactorily throughout countdown and flight. All fuel and oxidizer transducer A and B outputs were within 70 mV of each other. (A maximum of 150 mV is allowed by specification.) The command pilot reported nominal pressure readings throughout the launch, and good agreement between all A and B indicator pairs.

5.2.9 Range Safety and Ordnance

The range safety system and all ordnance performed satisfactorily.

5.2.9.1 Range safety.- Preflight checkout of both command receivers with no. 1 and no. 2 range transmitters (FRW-2) was normal. Telemetry data from the receiver were obtained until loss of signal (LOS) at approximately 96 seconds after spacecraft separation. Telemetry recordings and range logs agreed on the following events:

Time from lift-off, sec	Event
67.1	Station 1 transfer to high power using quad high-power antenna
76.8	Malfunction of rotary joint of quad antenna caused poor quality of data for spacecraft DCS system. DRUL monitor auto switch to failsafe transmitter 2 (FRW-2)
87.4	Manual switchback to master transmitter 1 (FRW-2)
115.8	Station 1 transfer to station 3 using 10 kW ESCO antenna
153.2	Staging event telemetry dropout. (Signal level dropout for 250 msec)
333.8	Auxiliary Second Stage Cut-off (ASCO) received (0.062 second after guidance SECO)

MISTRAM I was used as the primary source for impact prediction (IP) and provided accurate information. First motion of the IP plotting board at central control occurred at LO+0.036 second, and MISTRAM tracking functions occurred as follows:

Time from lift-off, sec	Event
1.5	Calibrate channel first lock
4.8	Range channel first lock
5.5	Calibrate channel unlock AGC measurement 785 signal shift down to threshold unlock level for 0.8 second
20.5	Calibrate channel first sweep
22.1	Range channel unlock AGC measurement 784 signal shift down to threshold unlock level for 0.2 second
153.2	Staging event telemetry dropout 250 msec
353.8	Transponder performance normal through completion of range safety requirement
389.4	Station I handover to station I signal shift down unlock level measurement 789 for 0.1 second. Transponder performance poor to unlock for 10.8 seconds
401	Transponder locked to station II signal until 2 hours 23 minutes 16.2 seconds after lift-off
430	Loss of telemetry data

5.2.9.2 Ordnance.- Preflight checkout, installation, and inflight operations of the ordnance items were satisfactory. Postlaunch inspection verified that all hold-down nuts functioned normally.

TABLE 5.2-I.- SUMMARY OF LAUNCH-VEHICLE STRUCTURAL DYNAMIC BENDING LOADS

Flight time	Applicable bending mode	Bending moment, in.-lb, at launch vehicle station –		
		276 (Interface)	320	935
Max qα, (69 sec g.e.t.)	1st structure	0.022×10^6	Not a critical condition	0.173×10^6
	2nd structure	.072		.092
	3rd structure	.012		.013
	Stage I engine	.077		.318
Pre-BECO	Stage II fuel slosh	0.005×10^6	0.011×10^6	Not a critical condition
	1st structure	.009	.011	
	Stage I engine	.050	.015	

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

Parameter	Preflight prediction	Postflight	Difference, percent
Thrust (engine, standard inlet conditions), lb	435 475[a]	442 987	1.72
Thrust (engine, flight average), lb	459 884	466 986	1.54
Specific impulse (standard inlet conditions), $\frac{\text{lb-sec}}{\text{lb}}$	258.72	259.62	0.35
Specific impulse (flight average), $\frac{\text{lb-sec}}{\text{lb}}$	275.85	276.81	0.36
Engine mixture ratio (average between sensors)	1.9142	1.902	-0.64
Engine mixture ratio (standard inlet conditions)	1.9285	1.911	-0.91
Oxidizer flow rate (average between sensors), lb/sec . . .	1097.48	1107.1	0.87
Oxidizer flow rate (standard inlet conditions), lb/sec	1110.26	1122.0	1.05
Fuel flow rate (average between sensors), lb/sec	573.33	581.0	1.05
Fuel flow rate (standard inlet conditions), lb/sec	575.71	587.1	2.09
Burn time (87FS1 to 87FS2), sec .	157.73	155.80	-1.12

[a] Includes 2450-lb bias added to acceptance tests

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

Parameter	Preflight prediction	Postflight	Difference, percent
Thrust (chamber, at standard inlet conditions), lb	98 457	99 917	1.48
Thrust (chamber, flight average), lb	98 736	100 614	1.87
Specific impulse (standard inlet conditions), sec	309.49	311.3	0.54
Specific impulse (flight average), sec	310.09	311.6	0.42
Engine mixture ratio (standard inlet conditions)	1.7832	1.774	-0.46
Engine mixture ratio (average between sensors)	1.7524	1.755	0.15
Oxidizer flow rate (standard inlet conditions), lb/sec . . .	204.01	205.4	0.68
Oxidizer flow rate (average between sensors), lb-sec . . .	203.57	206.8	1.59
Fuel flow rate (standard inlet conditions), lb/sec . . .	114.41	115.5	0.95
Fuel flow rate (average between sensors), lb/sec . . .	116.16	117.8	1.41
Burn time (91FS1 to 92FS2), sec	184.29	181.31	-1.61

TABLE 5.2-IV.- STAGE I IGNITION DISPLACEMENTS

Actuator	Displacement, in.
Pitch 1_1	-0.14
Yaw/roll 2_1	0.27
Yaw/roll 3_1	0.09
Pitch 4_1	-0.06
Axis	**Rate, deg/sec**
Pitch (stage II)	±0.9
Yaw (stage II)	±0.2
Roll (stage II)	±1.0

TABLE 5.2-V.- MAXIMUM ATTITUDE ERRORS AND RATES

Axis	Attitude error, deg	Time from lift-off, sec
Pitch	2.26	109
Yaw	1.68	71
Roll	1.44	107
Axis	**Rate, deg/sec**	**Time from lift-off, sec**
Pitch	-0.90	71.5
Yaw	-0.95	95
Roll	-0.55	58.5

TABLE 5.2-VI.- POST-SECO RATES

Axis	Parameter	Rate, deg/sec
Pitch	Maximum positive rate at SECO + 2.4 sec	0.9
	Maximum negative rate at SECO + 17.2 sec	-.4
	Rate at SECO + 20 sec	-.4
Yaw	Maximum positive rate at SECO + 13.3 sec	0.2
	Maximum negative rate at SECO + 3.4 sec	-.2
	Rate at SECO + 20 sec	.2
Roll	Maximum positive rate at SECO + 2.1 sec	0.4
	Maximum negative rate at SECO + 10.4 sec	-.6
	Rate at SECO + 20 sec	.3

TABLE 5.2-VII.- PLANNED AND ACTUAL EVENT TIMES AND VEHICLE 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	10.14	-0.02	1.25	1.14	-0.11
Roll program end	20.48	20.42	-.06	1.25	1.14	-.11
Pitch program 1 start	23.04	23.01	-.03	-.697	-.70	+.003
Pitch program 1 end	88.32	88.22	-.10	-.697	-.70	+.003
Pitch program 2 start	88.32	88.22	-.10	-.495	-.49	-.005
Pitch program 2 end	119.04	118.84	-.20	-.495	-.49	-.005
Pitch program 3 start	119.04	118.84	-.20	-.246	-.20	-.046
Pitch program 3 end	162.56	162.32	-.24	-.246	-.20	-.046

[redacted]

[redacted]

TABLE 5.2-VIII.- GT-3 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 (1300 psia equiv)	3300	-2.17	2540	-2.39
Stage II secondary hydraulics	Pressure switch (400 to 1000 psia S/D signal to MDS)	3170	-2.78	2770	+150
Stage I tandem actuators					
No. 1 subassembly 2 pitch	±4.0°	+0.85°	+61.5	-0.40°	+106.7
No. 2 subassembly 2 yaw/roll	±4.0°	+1.00°	+70.0	-0.68°	+84.5
No. 3 subassembly 1 yaw/roll	±4.0°	+0.30°	+92.0	-1.20°	+70.3
No. 4 subassembly 1 pitch	±4.0°	+0.35°	+108	-1.00°	+61.0
Stage I pitch rate (+ = up, - = down)	+2.5 deg/sec -3.0 deg/sec	+0.15 deg/sec	+105.8	-0.95 deg/sec	+25.4
Stage I yaw rate (+ = right, - = left)	±2.5 deg/sec	+0.35 deg/sec	+93.5	-0.9 deg/sec	+79.0
Stage I roll rate (+ = clockwise, - = counterclockwise)	±20.0 deg/sec	+2.20 deg/sec	+0.30	-1.7 deg/sec	+152.4
Stage II pitch rate (+ = up, - = down)	±10.0 deg/sec	+0.2 deg/sec	+164.5	-1.9 deg/sec	+171 to 181.5
Stage II yaw rate (+ = right, - = left)	±10.0 deg/sec	+1.30 deg/sec	+154.1	-0.30 deg/sec	+156.1
Stage II roll rate (+ = clockwise, - = counterclockwise)	±20.0 deg/sec	+0.60 deg/sec	+154.0	-0.1 deg/sec	+335.0

NASA-S-65-3619

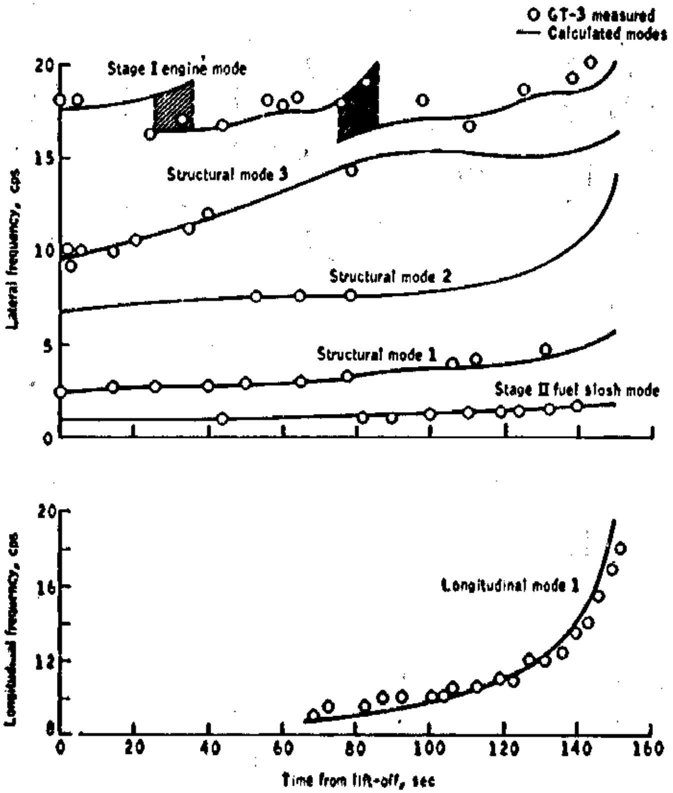


Figure 5.2-1. - Vibration modes for stage I flight

NASA-S-65-3599

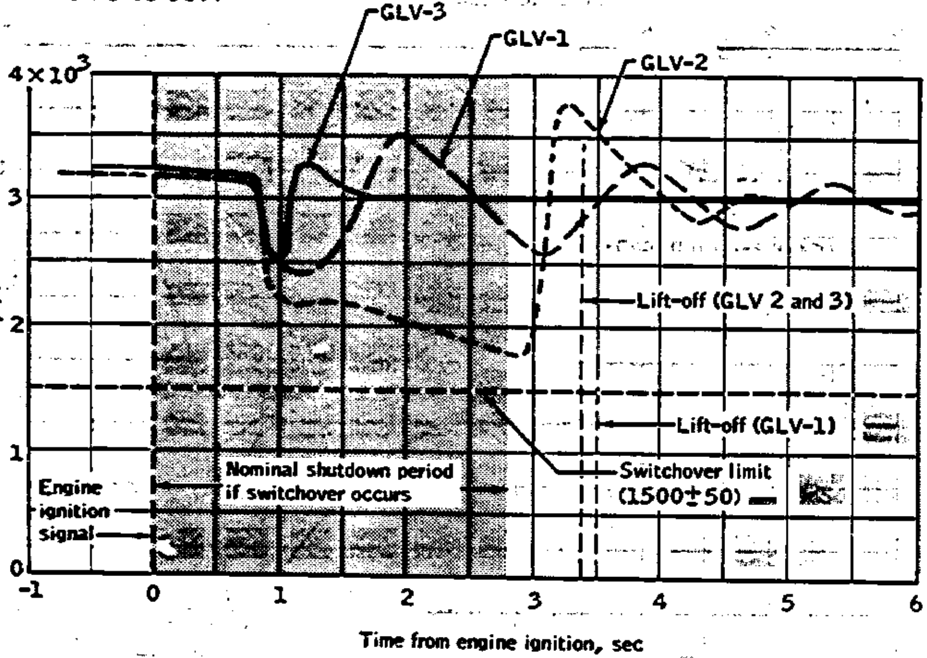


Figure 5.2-2. - Comparison of GLV stage I primary hydraulic pressures.

NASA-S-65-3600

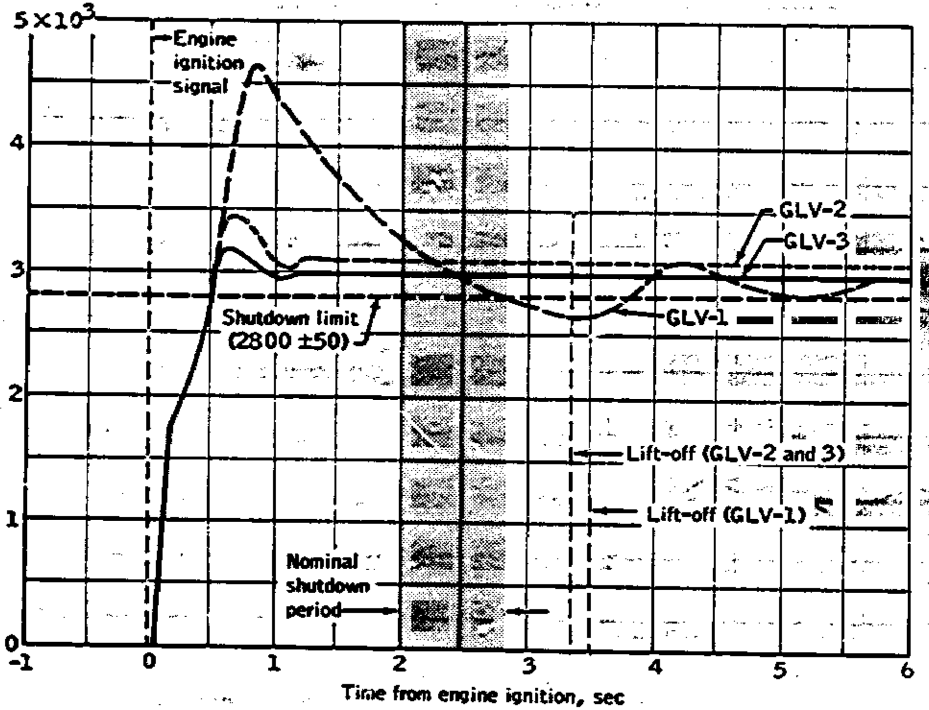
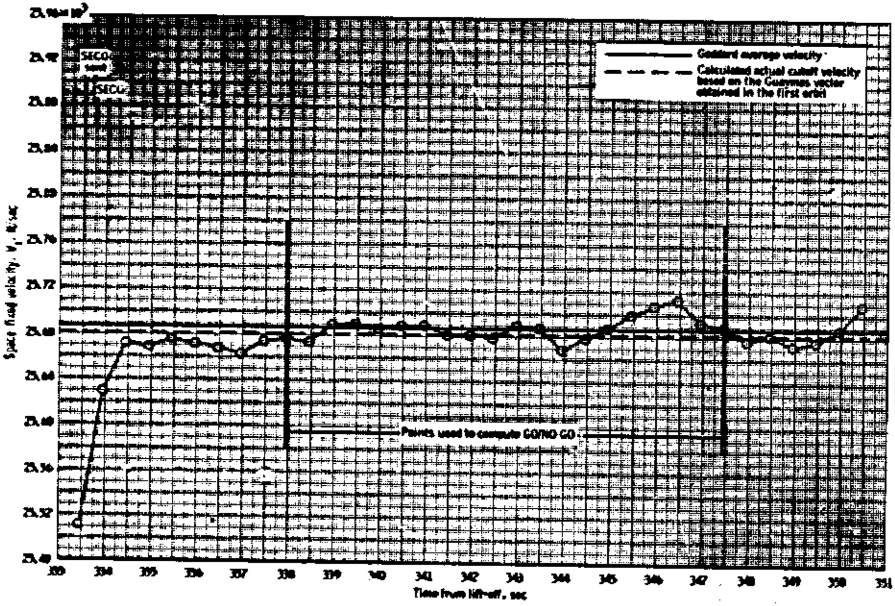


Figure 5.2-3. - Comparison of GLV stage I secondary hydraulic pressures

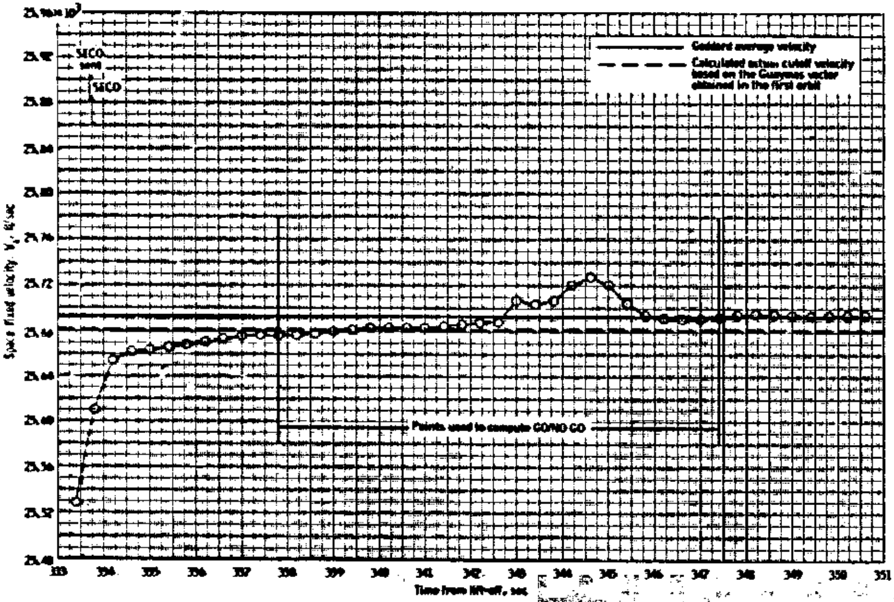
NASA-S-69-3689



(a) Launch vehicle guidance data.

Figure 5.2-4.- Space-fixed velocity in the region of cutoff.

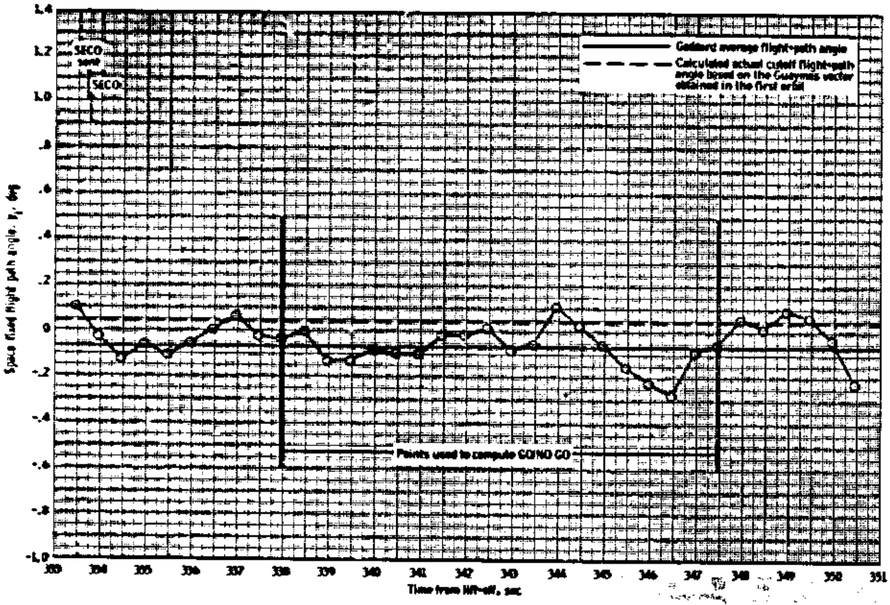
NASA-S-68-3612



(b) MISTRAM I Range Safety computer (IP-3600) data.

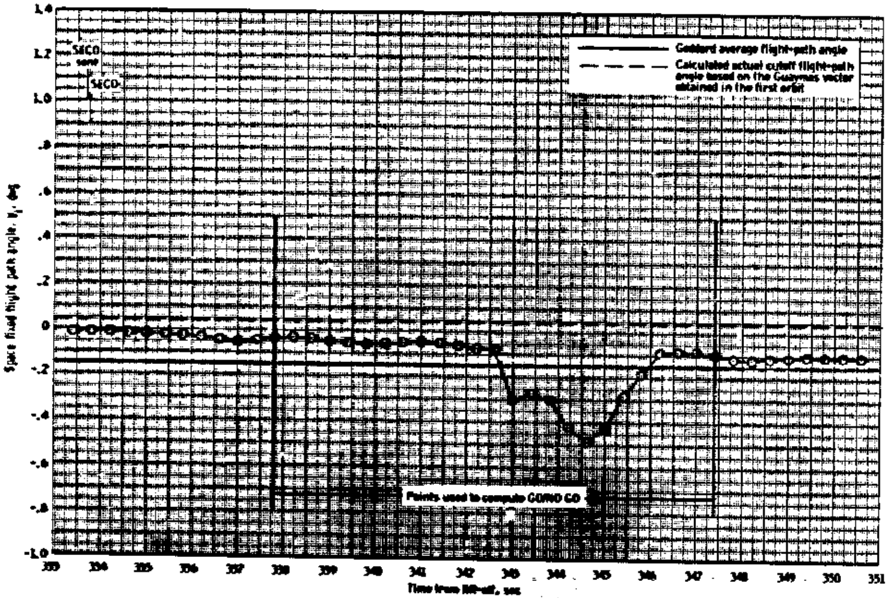
Figure 5.2-4. - Concluded.

NASA-S-65-3620



(a) Launch vehicle guidance data.

Figure 5.2-5.- Space-fixed flight-path angle in the region of cutoff.



(b) MISTRAM I Range Safety computer IP-3600 data.

Figure 5.2-5. - Concluded.

5.3 GEMINI LAUNCH-VEHICLE-SPACECRAFT INTERFACE PERFORMANCE

The various aspects of the Gemini launch-vehicle-spacecraft interface, as defined in reference 5, 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 telemetry measurements made on both sides of the interface, and by observation of the crew.

An inspection of the mechanical interface prior to and after the mating of the launch vehicle and the spacecraft showed no discrepancies from the specification drawings. The sealing and venting requirements were visually established instead of using a pressure test as was done on previous flights.

Structural loading of the launch-vehicle-spacecraft interface was calculated for the maximum $q\alpha$ flight condition. The combined effects of quasi-steady loads, modal response, and compartment pressure produced approximately 31 380 pounds equivalent axial compression load, or 17 percent of the tested interface compressive strength. The equivalent axial tension strength was approximately 35 110 pounds, or 26.5 percent of the tested tension strength.

The electrical interfacing circuitry on both sides of the interface functioned without anomalies.

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6.0 MISSION SUPPORT PERFORMANCE

6.1 PRELAUNCH OPERATIONS

6.1.1 Gemini Spacecraft

Spacecraft system functional tests, and servicing and mechanical work were completed satisfactorily during the precount period. The final countdown was started at 07:00:00 G.m.t. on March 23, 1965, and continued uninterrupted except for a launch-vehicle hold. Count progression was ahead of schedule by 10 to 20 minutes up until the launch vehicle hold at T-35 minutes.

Only one problem occurred in connection with the spacecraft. During the thermocouple checks, it was noticed that sensor PD08 did not register. No correction was attempted.

Static firing of the prescribed thrusters was completed within the scheduled time allotted in the count, and the countdown proceeded satisfactorily to lift-off.

6.1.2 Gemini Launch Vehicle

In general, the GT-3 propellant loading and launch countdown proceeded satisfactorily except for holds caused by two small launch-vehicle propellant leaks as noted in detail in the following paragraphs. Prior to discovery of the stage I engine oxidizer leak, both the launch vehicle and the spacecraft countdowns were ahead of schedule by 30 minutes or more. In addition, propellant-loading problems, such as those encountered on previous flights, with flowmeters, totalizers, et cetera, were not encountered, and the aerospace ground equipment (AGE) performed properly. The significant events during the prelaunch operations are discussed in the following paragraphs.

Oxidizer loading was started at 6:21 p.m. e.s.t., March 22, 1965, with a temperature of 21.6° F in the ready storage vessel (RSV). Oxidizer loading was completed at 7:58 p.m. e.s.t. without incident. After completion of oxidizer loading, a leak was detected at the stage II oxidizer fill and drain disconnect pressure cap on the launch vehicle. The pressure cap was replaced and the leak stopped. Subsequent failure analysis of the leaking cap revealed that small metallic particles, lodged under the seal, caused improper seating.

Fuel loading was started at 10:35 p.m. e.s.t. on March 22, 1965, with 26° F temperature in the RSV, and was completed at 11:35 p.m. e.s.t. without incident.

Final countdown was initiated at 2:00 a.m. e.s.t. (07:00:00 G.m.t.) on March 23, 1965, and proceeded without incident until approximately T-70 minutes, when an oxidizer leak was noted on engine subassembly 1. The oxidizer was leaking between the jam nut and bulkhead elbow of the oxidizer discharge pressure transducer (the instrumentation boss on the oxidizer pump discharge flange). The filling uses a stainless steel seal. The leak was stopped by backing off the nut and retorquing to 350 inch-pounds. (Maximum torque allowable for this fitting is 360 in.-lb.) It is planned to eliminate this seal for subsequent flights by welding the elbow to the pump discharge fitting.

A countdown hold at T-35 minutes lasted 24 minutes to permit observation time to insure that the leakage had stopped. No further problems were encountered and the count proceeded smoothly to launch.

The damage to the pad resulting from the GT-3 lift-off blast was minimum, and less than that associated with the GT-2 launch.

6.2 FLIGHT CONTROL

The events discussed in this section are reported in real time as they appeared to the flight controllers; consequently, there may be certain inconsistencies with other sections of the report which are based on extensive postflight analyses.

6.2.1 Premission Operations

6.2.1.1 <u>Premission schedule</u>.- The flight control teams assigned to the Mission Control Center (MCC) at Cape Kennedy, to the tracking ships, and to the remote sites were at their respective stations in time for the prelaunch simulations and tests conducted between March 5 and March 22, 1965. The MCC-Houston received data from the MCC-Cape Kennedy during the systems tests but was unable to provide the planned support because of several data interface problems.

6.2.1.2 <u>Documentation</u>.- The documentation provided for this mission was satisfactory except that numerous late changes in equipment configuration caused some problems. Reference 6 lists the changes that were incorporated. The network required 63 Instrumentation Support Instructions (ISI's); however, this number is considered nominal for support of the first manned mission of a new spacecraft.

6.2.1.3 MCC network flight control operations.- The MCC flight control team supported and participated in all of the launch-complex simulations and tests. These prelaunch operations included the launch abort and orbital flight simulations required to formalize the ground and flight crew procedures and data flow.

6.2.1.4 Countdown.- During the precount, the inertial measuring unit (IMU) was removed from the spacecraft to permit the cover hold-down screws to be replaced with special hardware. When the IMU was reinstalled, the X-axis accelerometer measurement had changed 5×10^{-4} radians (in terms of null bias offset) while the maximum allowable change limit is only 3×10^{-4} radians. If no adjustment were made in the bias correction factor (KX_4) in the onboard computer, the change of 5×10^{-4} radians would result in an error of approximately 5 to 10 ft/sec at inertial guidance system (IGS) SECO. The decision was made to leave the bias correction at the value measured before removal of the inertial measuring unit (IMU) and to accept the possibility of a 5 to 10 ft/sec error at SECO. The IMU contractor is investigating this problem which has also previously occurred in the Centaur vehicle which uses the same accelerometer. The effect of the error in bias correction is discussed in section 5.1.5.2.2.

[redacted]

When the flight crew set the sequential lights test switch to amber, the telemetry indicated an attitude malfunction. The reason for the discrepancy was determined to be a diode failure in the onboard lights test circuitry and there was no further concern with the problem. The failure of transducer PD08 is discussed in sections 5.1.1.2 and 6.1.1 of this report. The only hold in the countdown occurred at T-35 minutes when a small leak in a transducer fitting in the GLV stage I engine stopped the count for 24 minutes. This hold is discussed in section 6.1.2 of this report.

During the spacecraft propulsion system check, the flight crew had to fire the OAMS pitch-down thrusters four or five times to get a good burn. After the OAMS system had been activated, the first test of propellant-remaining computations based on actual spacecraft data was conducted at the Houston Auxiliary Computer Room (ACR). The parameters for helium source pressure and temperature, helium temperature at the fuel tank and at the oxidizer tank, and helium regulated pressure were relayed via the spacecraft analysis (SPAN) support room to Houston. Using these parameters and a mixture ratio of 1.55, the ACR computed that 295.7 pounds of usable propellant remained. This computation, when compared with a known usable propellant at loading of 295 pounds, validated the ACR for use.

[redacted]

6.2.2 Mission Operations Summary

6.2.2.1 Powered flight.- The GLV engine performance was close to nominal except for a slightly high stage I thrust and a slightly low stage II thrust chamber pressure. The engine performance is discussed in detail in section 5.2.2 of this report. The pitch program was initiated on time, but appeared to be slightly low resulting in a lofted trajectory that was aggravated by the excessive stage I thrust. However, at no time was the trajectory considered to be out of bounds.

The inertial guidance system (IGS) updates at LO+105 seconds and a LO+145 seconds were transmitted on time, and verified. Burroughs guidance was initiated at LO+168 seconds, and carried full pitch-down steering of 2 deg/sec for 12.5 seconds. The IGS attitude error converged and both systems seemed to be nominal. The particular velocity and altitude at which staging occurred permitted the radio guidance system (RGS) to assume a fixed attitude until SECO, with only minor steering commands in pitch required. At approximately LO+ 270 seconds, the IGS developed an attitude error trend which saturated at 6° nose down at LO+293 seconds. When the error signal was still saturated at LO+310 seconds, the IGS was considered unsatisfactory for switchover except in the event of a catastrophic malfunction in the primary system. In the last 30 seconds of powered flight, the IGS yaw attitude error (launch vehicle pitch axis) slowly increased in a negative direction and at SECO was off the low end of the scale. A complete discussion of the RGS and IGS performance is contained in sections 5.1.5 and 5.2.3 of this report.

Although the plotboards indicated a slightly underspeed condition, SECO was obviously GO. The cut-off conditions and insertion vectors were as follows:

Data source	Velocity, ft/sec	Flight-path angle, deg
GE-Burroughs	25 678	-0.07
IP	25 698	-0.147
Bermuda (BDA)[a]	25 676	-0.01

[a] The large difference between the BDA values and the other two values resulted from BDA transferring to orbital phase before the data had time to build a good short-arc solution.

[redacted]

6.2.2.2 Orbital.- It was first indicated that SECO occurred at LO+335 seconds (determined later to be LO+333.75 seconds) followed by a 13-second orbital attitude and maneuver system (OAMS) separation burn. The OAMS burn imparted 12.6 ft/sec velocity to the spacecraft and, based on BDA data, resulted in an orbit of 86.9 to 120.8 nautical miles. The orbital ephemeris was later updated by a Canary Island (CYI) differential correction to 87.1 and 121.8 nautical miles. The MCC-Cape Kennedy confirmed that the main batteries had been taken off-line and that insertion checklist had been completed.

CYI acquired the spacecraft at 00:15:00 g.e.t. and relayed the 2-1 retrofire times. With the radiator in the flow position, a radiator outlet temperature of 80° F was recorded. The radiator was then returned to the bypass condition. Blood-pressure data were received from the flight crew. Several unsuccessful attempts were made to correct a slight left-yaw condition and the pulse mode was used to compensate for it. No excessive propellant usage was evident. The sea urchin egg experiment was started at 00:20:00 g.e.t. At loss of signal (LOS), CYI reported that an adapter shaped-charge bilevel event light came on.

By way of voice remoting, Kano (KNO) informed Cape Kennedy that the spacecraft dc-to-dc converter had failed. The correct orbital parameters were relayed to the flight crew at this time. Unsuccessful attempts were made to correct the left-yaw condition by using the secondary yaw and ACME logic and the secondary attitude drivers.

The flight crew could not be contacted via Tananarive (TAN) air-to-ground remote communication. The radiator was placed in flow condition before acquisition of signal (AOS) by the Coastal Sentry Quebec (CSQ) at 00:43:58 g.e.t. A radiator inlet temperature of 76° F and an outlet temperature of 42° F was recorded by the CSQ. Onboard readouts concerning the radiator status were relayed back to Cape Kennedy.

Carnarvon (CRO) gave the flight crew a GO for a second orbit. All spacecraft clocks were synchronized. The preretrofire update and its associated maneuver load were uplinked to the spacecraft computer. Data for the Texas maneuver were passed to the flight crew. At CRO LOS, the primary O_2 pressure was reading off the scale on the high side.

CSQ and CRO summary message data, when computed by the ACR, showed that there was the possibility of an abnormally high OAMS propellant usage of approximately 15 pounds per hour up to that time. Of concern was the possibility that one of the two propellants was leaking through a solenoid valve which was partially stuck open. A leak in OAMS fuel (MMH) of 15 pounds per hour would seriously restrict later portions of the flight profile. The possibility of turning the OAMS propellant motor valves off was considered; however, it was decided to wait until after the Texas maneuver before taking any action, since considerably more data on the OAMS system would be available at that time. The spacecraft contractor support personnel reported that a thruster fuel-valve leak would result in negligible thrust; however, an oxidizer valve leak could produce up to 5 pounds of thrust.

The Rose Knot Victor (RKV) received and validated their command loads for the Texas maneuver and for the 3-1 T_R time. The flight crew was advised via Canton (CTN) air-to-ground voice remoting to leave their OAMS propellant switch on until after the Texas maneuver so that the OAMS could be fully evaluated at MCC-Cape Kennedy. The flight crew was also advised of the high primary O_2 pressure reading which is discussed in section 5.1.4.1 of this report.

The RKV was briefed and asked to relay certain onboard and ground parameters back to the MCC. Upon AOS, the RKV gave the flight crew a GO and uplinked the 3-1 T_R time. The RKV was required to transmit the maneuver load twice because the onboard computer was not in the proper mode. The RKV noted an indication of aft-firing thrusters during this orbital pass, and this was confirmed by Guaymas (GYM).

GYM handed over to Texas (TEX) at 01:30:33 g.e.t. and TEX counted down to the maneuver burn. Attitudes held well, and it was confirmed from several sources that the OAMS functioned properly. Readouts were obtained on the propellant following the maneuver although there was some confusion as to the correct incremental velocity indicator (IVI) readings.

As data became available from RKV and GYM, it was apparent that there had been no abnormal OAMS usage. The OAMS data from the Texas maneuver agreed well with the ΔV and time of burn. After this time, the yaw-left drift did not cause any concern relative to the OAMS propellant quantities.

During the last half of the first orbit and until CSQ on the second orbit, the water boiler outlet temperature was about 2° F lower than the inlet and then rose to about 2° F higher. This was reported by CRO as an indication that the water boiler was out of water. From information available at MCC-Cape Kennedy, this did not seem to be the case.

The radiator continued to perform as predicted, except during the second orbital pass over MCC-Cape Kennedy when the radiator outlet temperature rose to 45° F, but it was back to 34.6° F at CSQ.

The primary oxygen pressure continued to rise to approximately 1000 psi during the flight, and the flight crew would periodically use O_2 high rate to bring the pressure back down. This is believed to be the result of the pressure switch actuating at a setting above specifications, but did not present a significant problem, and the oxygen usage appeared to be compatible with the flight plan. (Section 5.1.4.1.)

The standby transmitter was powered-up to transmit the tape dump over the MCC. The preretrofire updates for 3-1 were transmitted to the spacecraft and verified. A second blood-pressure readout was requested of the pilot and information on the launch-vehicle sighting was given to the crew before BDA LOS.

The 2-B times were passed to CYI and relayed to the spacecraft upon acquisition. CYI reported that all systems looked good from the ground and commenced to send the calibration command. The new orbit perigee of 85.6 nautical miles and apogee of 92.6 nautical miles was confirmed by Eglin (EGL) and relayed to the flight crew. Following each CYI pass debriefing, the aeromedical data were transmitted to the MCC-Cape Kennedy via the voice-data line. After each orbital pass, CRO data were also transmitted to MCC-Cape Kennedy via the voice-data line.

After CYI LOS, RETRO decided not to relay a new 3-1 update to CRO since the load from the MCC-Cape Kennedy was good. Four minutes later at 02:10:00 g.e.t., voice communications to CRO and to the CSQ were reported out, and preparations were made to contact the CSQ via SYNCOM. This mode of communication was successful although marginal.

At 02:15:00 g.e.t., CSQ acquired and was requested to receive information preceding and following the burn. They successfully received the IVI and attitude readouts, transmitted a time-to-reset (T_X) ACQ-AID ON COMMAND. The T_X received a spacecraft reject and was retransmitted several times, but no confirmation was ever received of its acceptance during the pass.

The CSQ received an oral temperature reading from the flight crew. They also received some onboard propellant readings before and after the burn. CSQ gave the crew a GO status and received a good blood-pressure measurement from the pilot before LOS.

Over CRO on the second orbital pass, the command pilot reviewed the yaw-left problem stating that all electrical controls on the OAMS attitude circuitry had been tried without any effect and that there was no cross-coupling into roll. Since the problem could not be controlled electrically, it was obvious that if the drift were the result of an OAMS problem, it had to be that one of the two OAMS yaw-left

TCA oxidizer valves was stuck open. However, there was no roll cross-coupling or abnormal propellant depletion, and it became extremely doubtful that the yaw-left drift was caused by the OAMS. Stability of the nominal data from the RCS ring B throughout the flight also ruled this system out as a possible cause of the drift.

During the analysis of the yaw-drift problem, it was suggested that the drift might be caused by the water boiler boil-off. Discussions between GNC and contractor support personnel indicated that the water boiler exhaust could, in fact, give the spacecraft an acceleration of 0.025 deg/sec in a yaw-left direction. On the final orbital pass over Cape Kennedy, the CAP COM attempted to relate this acceleration to the rate buildup the spacecraft was experiencing. This was never fully accomplished because of other communication requirements.

Further analysis of the abnormally high OAMS usage computed by the Houston ACR using first orbit data from the CSQ and CRO showed it to be the result of about 10° increase in OAMS source helium temperature. This increase was noted again during the third orbital pass but not on the second pass. At this time, it is not known whether this parameter was erratic or whether the increase was caused by a heat pulse from the day portion of the orbit. If it were a correct reading due to a heat pulse, the source pressure should have also been affected and the ACR computation should have yielded a valid propellant-remaining solution. Also, the OAMS regulated pressure temperatures at the fuel and oxidizer tanks did not reflect a pulse of this magnitude. Hence, it is suspected that the source temperature readout was erratic.

It was found after RKV AOS that the 8-ball drifted only when the platform was in the ORBIT-RATE mode. The small end forward (SEF) mode alined the 8-balls properly. The RKV also received the oral temperature and blood-pressure readouts before handing over to GYM. GYM asked the pilot to turn on the real-time telemetry transmitter (RT-TM) and the acquisition aid beacon (ACQ AID). The coolant pump checks were completed, and pump A was removed from the primary coolant loop.

Over Cape Kennedy, the status of the platform modes other than ORBIT RATE was determined to be satisfactory. It was left to the discretion of the flight crew as to which mode to use, although it was suggested that reentry be in the blunt end forward (BEF) mode. The spacecraft standby transmitter was commanded off following the completion of the tape dump at the MCC-Cape Kennedy. The flight crew switched from their secondary scanners back to the primary system.

The command load containing the 4-1 preretro update was properly uplinked to the onboard computer and verified by the ground. Subsequent control mode checks were deleted since they had been accomplished earlier while checking the effectiveness of the platform in the various modes.

CSQ passed the 3-B times and the reentry mode data to the flight crew. They received the time of the activation of the sea urchin experiment and were standing by for completion of the preretro checklist at LOS.

CRO obtained the onboard OAMS source pressure temperature and propellant quantity values. CRO loaded the new 4-1 data into the spacecraft computer and confirmed that the preretrofire checklist was complete and that the spacecraft clocks were synchronized with the ground clocks. They also transmitted the 4-1 times and the event times for the reentry phase. It was determined at this time that the RKV was to be the prime station for retrofire and GYM was to be backup.

6.2.2.3 Reentry. - At 04:21:23 g.e.t. the preretro burn was initiated for 111 seconds and resulted in a change in velocity of 99 ft/sec aft. The attitudes held good for the duration of the burn. Hawaii (HAW) relayed the final IVI readouts following the burn. The RKV confirmed that the T_R clock was synchronized, that the spacecraft was in the proper reentry mode, and that the T_R -5 minute checklist was completed. RKV then counted down to retrofire with the T_R clock leading by $\frac{1}{2}$ second. The adapter separation was good and the automatic retrofire of all four retrorockets was successful. The RKV handed over to GYM at that time.

GYM reported the IVI readouts as 331 aft, 105 right, 4 down, and also confirmed that retrojettison was on time. These IVI's indicated that retro attitude was 18° pitch (nose down, BEF) instead of 16°. The voice transfer via TEX was set up so that the reentry data could be relayed to the spacecraft before blackout. A bank angle of 45° left, 55° right, and a time to reverse bank of 19:08:17 G.m.t., was passed to the flight crew. The MCC received approximately 30 seconds of data before blackout indicating that O_2 high rate was not on. At 04:40:00 g.e.t., the first impact prediction (IP) was determined. The flight crew was instructed to use nominal times for their main and drogue parachute deployment.

During reentry the cabin temperature rose to approximately 100° F, well below the predicted value, and suit temperature rose to 72° F. No confirmation was received to indicate the flight crew ever used O_2 high rate. Both secondary oxygen bottles were at 5100 psi at the end of blackout, which was the same pressure that was present at lift-off.

During blackout, Merritt Island Launch Area (MILA) reported receiving C-band track, although the UHF from the spacecraft was not readable. The CAP COM at the MCC transmitted on UHF during blackout, but received no confirmation of communication reception onboard. After coming out of blackout, the flight crew reported that their onboard computers indicated they would be 25 miles short of nominal impact prediction (IP). (Editors note: This first reading by the flight crew was an error resulting from the 2 to 1 conversion scale factor on the FDI. The correct reading was subsequently determined to be 50 miles short of the nominal IP.) The UHF telemetry contact was good down to 18 000 feet. The drogue parachute was confirmed and through intermittent telemetry, GNC indicated a good main parachute at 5500 feet.

The final GSFC IP, based on GTI data, was 22°29' N latitude, 70°48' W longitude. The recovery aircraft, Big Box 14, reported 22°23' N latitude and 70°50' W longitude. The USS Intrepid reported the pickup position as 22°27' N latitude and 70°50' W longitude.

6.3 NETWORK PERFORMANCE

6.3.1 MCC and Remote Facilities

The network for the GT-3 mission consisted of the Mission Control Centers at Cape Kennedy and in Houston (MCC-Cape Kennedy and MCC-Houston), Merritt Island Launch Area (MILA), Grand Bahama Island (GBI), Grand Turk Island (GTI), Antigua (ANT), Ascension (ASC), Pretoria (PRE), Bermuda (BDA), Grand Canary Island (CYI), Kano (KNO), Tananarive (TAN), instrumentation ship Coastal Sentry Quebec (CSQ), Carnarvon (CRO), Woomera (WOM), Canton (CTN), Hawaii (HAW), instrumentation ship Rose Knot Victor (RKV), tracking ship Range Tracker (RTK), California (CAL), Guaymas (GYM), White Sands (WHS), Texas (TEX), Eglin (EGL), and two telemetry aircraft and two voice-relay aircraft. The network is shown in figure 4-1.

The network countdown was picked up at T-420 minutes on March 23, 1965. The digital command system (DCS) and the PCM telemetry checkouts were included in the computer and data flow integrated subsystem (CADFISS) tests for the first time.

6.3.2 Network Facilities

6.3.2.1 Remote sites.-

6.3.2.1.1 Telemetry: Telemetry data were recorded by all network stations except WOM which was scheduled as a backup radar-support station.

Network telemetry support was good with few problems and these did not jeopardize support of the mission. The biomedical data were successfully received by most of the stations.

Minor errors were encountered in the Gemini real-time patching tape 3 (GRP-3). Documentation for the cardiotachometer and respiration rate converters was not available. The accumulation of the X-axis acceleration could not be displayed on the Gemini systems monitor console (GSMC) Sanborn unit because the documentation called for patching the wrong word gate. A standby telemetry transmitter exercise over GYM caused approximately 60 seconds of real-time data to be lost at TEX and MCC-Houston. On the third orbital pass, real-time data retransmission from ANT contained invalid subframe data.

Table 6-I lists the percentage of normal horizon-to-horizon coverage received at each station. The blank columns in the table indicate that the spacecraft did not radiate that frequency. Passes with low percentages were caused by low signal-strength dropouts and not by equipment failures.

6.3.2.1.2 Radar: C- and S-band radar coverage was provided by the stations listed in table 6-II. C-band coverage was provided also by ASC, PRE, and the RTK. The C- and S-band transponders performed well throughout the mission, and all sites reported good tracking on most of the favorable passes.

In those instances where actual coverage was less than nominal (see table 6-II), the loss of radar track was caused by scale changes during the pass, reacquisition, and transponder sharing. Some minor phasing problems were noted during the first and second orbital passes over the United States. During the second pass, the CAL C-band tracking was lost when WHS used automatic phasing to acquire initial track. Manual phasing was used at WHS for the remainder of the mission.

BDA reported a sudden increase in C-band transponder signal strength at 00:09:49 g.e.t. This increase was apparently due to the C-band antenna switchover; however, it occurred about 3 minutes later than the expected switchover time. BDA also reported seeing wobbulator modulation on the C-band transponder returns, prior to antenna switchover.

The Radar Controller's plotting board in the room for the network support team (NST) failed just after the first pass over Australia and was not repaired until almost one complete orbit later. The lack of a real-time display at the Radar Controller's console was a handicap during radar handover on the first pass over the United States. Additional confusion on this pass resulted when the Radar Controller and GYM understood that the California Verlort was actively tracking the transponder (CAL S-band radar was to track passively). The CAL C-band radar reported interference at the same time, and the Radar Controller inadvertently commanded them off instead of the S-band radar at CAL. The confusion was quickly cleared and the C-band radar at CAL reacquired.

All stations scheduled to track during reentry were successful in obtaining track. The EGL FPS-16 was tracking at the time blackout started. Neither of the mainland radars at Cape Kennedy (PAT and MILA) could see the transponder returns (even though EGL was interrogating the transponder) and acquired a skin track. Then the MILA TPQ-18 switched to transponder track at the point of closest approach (PCA). The PAT FPQ-6 was scheduled to remain on skin track. The GBI TPQ-18 acquired skin track initially and switched to the transponder at PCA.

From the available data, it appears that the ion sheath on the Gemini spacecraft has more attenuation from certain aspect angles than the attenuation measured on the Mercury spacecraft. The GTI TPQ-18 acquired skin track prior to the end of blackout and switched to transponder track at 19:08:45 G.m.t. (approximate end of blackout was 19:08:30 G.m.t.), and lost transponder track at 19:12:28 G.m.t. GTI reacquired transponder track at 19:15:55 G.m.t., and lost all returns at 19:16:30 G.m.t.

To determine the landing area, radar skin track of the expended second stage of the GLV was programed for the PAT FPQ-6 radar and the Wallops Island SPANDAR radar during the launch phase and the second and third orbital passes. The NORAD SPADATS radars at Moorestown, Turkey, and Trinidad were requested to skin track the GLV after insertion and to supply metric data to Goddard Space Flight Center (GSFC).

The GLV trajectory was established at spacecraft separation by using tracking data transmitted by the GE-Burroughs guidance system. The orbital elements of the GLV as obtained from the launch source were as follows:

Epoch, G.m.t.	14:30:00
Semi-major axis, n. mi	3545.9
Eccentricity	0.005106
Inclination, deg	32.59
Argument of perigee, deg	85.02
Longitude of assigned node, deg	79.16
True anomaly, deg	346.76

During launch, the PAT radar switched to skin track at LO+250 seconds and tracked the GLV until LO+380 seconds at a slant range of approximately 620 nautical miles at LOS. GSFC generated and transmitted acquisition data to Wallops Island and to PAT for skin track and transmitted the preceding orbital elements to NORAD immediately after insertion. During orbits 2 and 3, the PAT and Wallops Island radars did not acquire the GLV. The NORAD Moorestown Site furnished the following information during orbits 2 and 3:

	Orbital pass 2	Orbital pass 3
Number of observations . . .	1	10
Acquisition time, G.m.t. . .	16:00:50	17:34:51
LOS, G.m.t.	16:01:08	17:36:20
Elevation angle, deg	4.8	0.7 to 4.5

The single observation obtained during the second pass did not match the GLV or spacecraft trajectories. Data from the third pass were not of sufficient quantity or quality, as a result of the low elevation angle, to establish a definitive trajectory, although it was established as GLV data. The GLV trajectory used for generating radar pointing data was passed through the observed data to verify the accuracy of the pointing data. The pointing data on the third orbit was 10 seconds (time) behind the radar observations. The inaccuracy is well within the proven skin-track acquisition techniques as proven during the Saturn series missions and on the GT-1 mission.

Classical orbital elements were furnished to the NORAD Trinidad site by GSFC. Despite good look angles on the fifth orbit, the expended GLV was not acquired.

On March 24, 1965, Turkey NORAD site tracked the GLV from 06:33:17 to 06:36:28 G.m.t., with a maximum elevation of 5.7°. A routing error prevented these data from being received by the GSFC computer section until more than 6 hours later. The 21 data points were used to update the GSFC computer. On the basis of these data, the best possible estimate is that the GLV second stage reentered during orbit 13 at approximately 08:30:00 G.m.t., on March 24, 1965. The following actions are believed to have been the major reasons for failure of the MSFN radar network to acquire the orbiting GLV:

(a) Lack of updating of the ephemeris early in the lifetime.

(b) Failure of the Trinidad site to acquire and supply update information on orbit 5.

(c) Phasing of the GLV off the major part of the range at the time MSFN radar became available to support GLV track.

(d) Lack of real-time trajectory data from Turkey SPADATS.

For the GT-4 mission, a committment for additional radar sites to skin track the GLV during the launch phase and first orbit will be requested. In addition, closer coordination with NORAD should insure delivery of better SPADATS tracking data in a more timely manner. It is anticipated that this increased effort will provide the necessary data for a more accurate impact prediction for the GT-4 launch vehicle.

6.3.2.1.3 Acquisition aids: All acquisition systems were checked out and ready to support the mission at lift-off, and performed satisfactorily throughout the mission. Performance of these systems is considered excellent and it appears that system coverage approached 100 percent of that predicted. CRO experienced some difficulty in obtaining a lock-on at acquisition of signal (AOS) on each orbital pass because of apparent acquisition beacon frequency instability and multipath fading. The CSQ experienced some receiver lock problems in the phase-lock mode on the first pass because of command 1 interference. The interference was present on ensuing passes but receiver lock was achieved. Ship rolling caused elevation errors. On the third pass, WHS experienced a loss of auto-track for a short time because of a loss of real-time telemetry. This would not have occurred if WHS had been tracking acquisition beacon as prime. A loss of the real-time telemetry signal occurred over the western United States, presumably a result of switching by the flight crew. KNO tracked manually using the telemetry signal strength indicator at 230.4 Mc/sec. BDA and MCC-Cape Kennedy experienced excellent auto-track percentages although MCC-Cape Kennedy had no acquisition beacon on the third pass because the adapter section had separated.

6.3.2.1.4 Command control system: The command control system experienced minor hardware problems during the prelaunch phase but was checked out and ready to support the mission at launch. The IGS updates were uplinked successfully and on schedule.

During the orbital phase, only one discrepancy with the DCS was reported. The ANT carrier was turned on at 17:39:46 G.m.t. and off at 17:45:49 G.m.t. for a total time of 6 minutes 3 seconds. There were no sync pattern or simultaneous errors noted, but 42 transmit errors were counted. This resulted in six transmitter fail-overs. The transmit errors were attributed to a defective coil in the terminal timing unit at AFETR. The mission AOS and LOS times were within 30 seconds of nominal and no updating of the command handover plan was required. Some of the activities of the command control sites during each orbital pass included the following:

Station	First orbit
CYI	None
CSQ	None
CRO	Received PRE-RETRO load with MANEUVER, MANEUVER load, and PRE-RETRO load without MANEUVER; no retransmits of teletype (TTY) loads were required; time-to-retrofire (T_R) uplink was non-valid on first and second attempts; third attempt was valid; non-valid attempts may have been because of the low-look angle at time of transmission; PRE-RETRO was uplinked validly on first transmission.
HAW	Not used during first orbit.
RKV	Requested retransmit of TTY load; first T_R uplink was non-valid; second T_R uplink and computer load uplinked validly.
TEX	None
	Second orbit
CNV	Received PRE-RETRO load with MANEUVER, MANEUVER load, and PRE-RETRO load without MANEUVER; no retransmits of TTY load were required; T_R and computer load were uplinked validly.
CYI	Received PRE-RETRO load without MANEUVER, PRE-RETRO load with MANEUVER, and MANEUVER load. No retransmits of TTY loads were required; CAL ON and CAL OFF were uplinked validly.

Station	Second orbit-continued
CSQ	Uplinked four time-to-reset (T_X) commands, but no message acceptance pulses (MAPS) were received.
CRO	None
HAW	T_R and T_X uplinked validly.
RKV	None
TEX	None
	Third orbit
MCC-Cape Kennedy	PRE-RETRO load data time group (DTG) 23/1707 received twice, PRE-RETRO load DTG 23/1708 received once, and the MANEUVER load DTG 23/1718 was not received; MANEUVER load translation was received twice and DCS was loaded manually from the translation; T_R and computer load were uplinked validly; STBY TM R/T MOD and TAPE DUMP ON could not be validly uplinked; possibly caused by a loss of real-time telemetry during the pass.
CYI	Not used third orbit.
CSQ	None
CRO	Received PRE-RETRO load with MANEUVER, MANEUVER load, and PRE-RETRO load without MANEUVER; no retransmits of TTY loads were required; T_R and a computer load were uplinked validly.
HAW	No activity monitored over GSFC conference loop and HAW postlaunch instrumentation message (PLIM) not received at this time.
RKV	None

6.3.2.1.5 MISTRAM: The Valkaria (VAL) MISTRAM system received data in the active mode, and the Eleuthera (ELU) MISTRAM system was operated in the passive mode during the mission. Table 6-III shows the tracking periods of both systems.

6.3.2.2 Computing.- No problems with the 7094 computers at GSFC were noted during the mission. The CDC-3600 real-time computer facility at Cape Kennedy supported the DOD inter-range radar computer acquisition system which was operated on an engineering-test basis only. The 1218 computer – remote site data processor (RSDP) performed 100 percent successfully.

Some stations did not receive acquisition at the normal station horizon time (H) -25 minutes and H -5 minutes but did receive acquisition messages while the pass was in progress. These messages were initiated by the retrofire maneuver just prior to or during the track. This is the normal mode of operation and will occur during any maneuver performed by the spacecraft.

6.3.2.3 Communications.-

6.3.2.3.1 Voice and teletype: Voice and teletype communications were provided by two separate networks. The mission network consisted of the NASA communications network (NASCOM) augmented by facilities of the Department of Defense. The recovery network was provided by DOD with augmentation by NASCOM through direct leased circuits.

During the mission, most of the problems associated with the communication networks were of a relatively short duration, due to the fast restoration of the carriers under critical coverage. One of the problems was with the teletype sending circuits to the CSQ. Teletype communication was maintained with the CSQ by shifting back and forth between the Bassendean and the Syncom paths. There was a 20-minute failure of all voice communication into Australia; however, the teletype circuit remained open. HAW had a low-level voice signal during the third pass. This problem at HAW had occurred previously during simulations and is being investigated.

During the second orbit, software problems arose with the communication processor (CP) which necessitated a recycle of the on-line systems. The CP was off the air from 17:27:00 G.m.t. to 17:29:00 G.m.t. During this period, radar and telemetry designated (JJ) data tapes were held and delivered to the computer personnel as an after-the-fact move. Computer output messages were retained and reentered immediately after the CP was returned to use. Approximately 8300 messages, excluding data, were handled by the CP between 07:00:00 G.m.t. and 20:00:00 G.m.t.

WHS had a problem resulting from the numerous drops placed on the radar handover loop and the switching, conference, and monitoring arrangement (SCAMA) loop. This problem is being investigated by the DOD.

6.3.2.3.2 Air-to-ground communications: The UHF communications systems functioned very well. Communications from the MCC-Cape Kennedy to the spacecraft were remoted successfully through the network stations. Minor problems included poor voice intelligibility with TAN caused by interference, and low audio-level on the CSQ CAP COM console caused by the air-to-ground recorder which had to be disconnected during the first two passes. The HF system was not planned for use except during the recovery phase.

6.3.2.3.3 Relay aircraft: Two C-130 aircraft equipped with direct-voice relay equipment provided spacecraft-UHF to MCC-Cape Kennedy-HF duplex communications during the descent and recovery operations. The HF link from the aircraft to the MCC-Cape Kennedy was inoperative and is being investigated.

6.3.2.3.4 Frequency interference: HAW was the only network station to report frequency interference. This was on the C-band frequency of 5766 Mc/sec. The interfering signal was traced to a U.S. Navy Task Group and was secured. EGL and the station at Fort Huachuca, Arizona, reported interfering radio-teletype signals on the HF frequency of 15 016 kc/sec, but all interfering signals were located and secured, including one in a foreign country.

6.4 RECOVERY OPERATIONS

6.4.1 Recovery Force Deployment

The areas along the ground track where recovery ships and aircraft were located are shown in figure 6-1. The recovery forces were assigned positions in these areas in order to reach any point in their particular area within specified access times. The ship and aircraft access times, which varied for the different areas, were based on the probability of the spacecraft landing within a given area and the amount of recovery support provided in that area. Ship access time is defined as the elapsed time between the preliminary establishment of the approximate spacecraft landing point and the positioning of a recovery ship alongside the spacecraft. Aircraft access time is defined as the elapsed time between the preliminary establishment of the approximate spacecraft landing point and the installation of the flotation collar around the spacecraft by pararescuemen. It should be emphasized that access time is primarily a planning parameter and is based upon favorable operating conditions.

Nineteen ships, 12 aircraft, 11 helicopters, and several small special vehicles were used for recovery support in the planned landing areas. In addition, 30 aircraft were deployed around the world on strip alert for contingency recovery support. Table 6-IV summarizes the type of support available and the access times for the various areas. Department of Defense (DOD) routine operational ships and aircraft were used for the recovery support. NASA provided the DOD with special equipment, such as retrieval cranes for use aboard destroyers, airborne UHF electronic receivers, and spacecraft flotation collars. All recovery aircraft were equipped with the UHF receivers which gave the aircraft the capability to "home" on the spacecraft UHF location aids. These aircraft carried three-man pararescue teams equipped to parachute to the spacecraft and flight crew and render assistance. The destroyers along the ground track, with the exception of DD-2, DD-9, and DD-10, were equipped with spacecraft retrieval cranes. Twin turbine helicopters (SH-3A) were provided onboard the carrier to carry two three-man pararescue teams, flotation collars, and photographers to the spacecraft landing point within the primary landing area. Carrier fixed-wind aircraft were also available, if required, to assist the "on-station" aircraft in locating the spacecraft.

As indicated in table 6-IV, the launch-site recovery force consisted of helicopters, amphibious surface vehicles, special land vehicles, Navy craft, and small boats. This force, in addition to its launch-site recovery role in case of an early abort, was also capable of providing on-scene salvage support for launch vehicle and spacecraft components.

Before the flight, the recovery forces underwent extensive training, in addition to individual unit training in specific phases of the recovery operation at home bases and enroute to "on-station" positions, recovery simulations were conducted jointly with the aircraft, helicopters, ships, and the recovery control room (RCR) in the Mission Control Center (MCC), Cape Kennedy. These simulations were conducted for both the downrange and launch-site recovery forces.

6.4.2. Location and Retrieval

With the exception of aircraft assigned to area 4-1, all recovery forces were on station at launch time and were in communication with the RCR at MCC-Cape Kennedy. Weather conditions were favorable for spacecraft location and retrieval in all the planned landing areas.

During the countdown, recovery forces were periodically informed of the count status. The recovery forces were informed of lift-off at 14:24:00 G.m.t. Throughout the early minutes of flight, recovery forces were continually informed of the flight progress. WINREP 1 (summary of flight progress and verification of lift-off time) was sent to recovery forces at 00:06:00 g.e.t. Shortly after transmission of the WINREP, the following ships were released by the recovery force commander to normal operational control: all launch-site recovery forces; ATF, DD-1, DD-2, DD-4, DD-5, DD-6, AO, DD-7, and DD-8.

Destroyer DD-3 and two aircraft from area A were redeployed to positions in area 2-1. After the spacecraft passed the retrofire point for area 2-1, DD-3 and DD-9 were released by the recovery commander. The search aircraft assigned to area 2-1 proceeded to their new positions in area 3-1. After the spacecraft passed the retrofire point for area 3-1, DD-10 and DD-11 were released, and the search aircraft returned to Bermuda. Forty-five minutes before the spacecraft landed in area 4-1, three search and rescue aircraft (14, 15, and 16) arrived on their respective stations in area 4-1.

At 04:38:00 g.e.t. (19:02 G.m.t.), the recovery forces were notified that retrofire for a landing in area 4-1 had occurred. At 04:41 g.e.t. (19:05 G.m.t.) CALREP 1 (calculated spacecraft landing position and time) was sent to the recovery forces as being "nominal," meaning that the estimated landing point was as planned. Recovery forces in area 4-1 and spacecraft location and retrieval information are shown in figure 6-2. Upon receipt of CALREP 1, the carrier helicopters were launched. (By plan, the aircraft carrier was positioned

5 miles downrange and steaming toward the planned landing point.) At 19:08 G.m.t., the air search radar aboard the carrier picked up and tracked the spacecraft at a slant range of 197 nautical miles at 307° and continued to track the spacecraft until loss of signal (LOS). The final radar fix placed the spacecraft 60 nautical miles from the nominal landing point. At 19:16 G.m.t., the helicopters were vectored to this position, and a radar fix was transmitted to the RCR. At 19:17 G.m.t., aircraft 16 transmitted a BARREP (report by a unit obtaining an electronic DF on signals radiating from the spacecraft. Spacecraft landing occurred at 19:16 G.m.t. and a BARREP was received from aircraft 15 at 19:23 G.m.t. At 19:29 G.m.t. (13 minutes after spacecraft landing), aircraft 14 transmitted a JIGREP (positive visual contact with the spacecraft after landing). Pararescuemen from aircraft 14 then parachuted to the spacecraft. Aircraft 14 at 22° 22' N latitude, 70° 50' W longitude reported the first sighting of the spacecraft. At 19:32 G.m.t. the Coast Guard cutter Diligence (WTC-14) acquired the spacecraft on radar at 9 miles and vectored her helicopter to the scene. Three minutes later, the Coast Guard helicopter had the spacecraft in sight. At 19:41 G.m.t., the prime recovery helicopter put the swimmers and flotation collar in the water and reported that the spacecraft was floating normally with both hatches closed (fig. 6-3). At 19:46 G.m.t., the flotation collar in the water and reported that the spacecraft was floating was opened by the swimmers upon request of the command pilot.

At 20:05 G.m.t., the flight crew began to egress, and within 4 minutes both crew members were reported to be aboard the helicopter (fig. 6-4) and enroute to the aircraft carrier 35 miles away. At 20:28 G.m.t., the recovery helicopter landed aboard the aircraft carrier, 1 hour 12 minutes after spacecraft landing.

The aircraft carrier had a line attached to the spacecraft (fig. 6-5) by 21:51 G.m.t., and at 22:03 G.m.t., the spacecraft was on deck in its dolly (fig. 6-6). The position of spacecraft retrieval was 22° 27' N latitude, 70° 50' W longitude. At this time, members of the recovery team began the examination of the spacecraft exterior and started the postlanding procedures.

6.4.3 Recovery Aids

Approximately 2 hours after landing, the pararescuemen observed that the HF antenna bent and fell over. The pilot commented that he made contact with the antenna as he was egressing; however, the "bend" first occurred at a point about 4 feet above the spacecraft and it is not believed to have resulted from crew contact. Both the worldwide Federal Communications Commission (FCC) and DOD HF/DF networks were alerted to listen for spacecraft HF signals. Several stations ranging from as close as Puerto Rico to as far away as Midway received HF signals. Signal strength was generally fair to strong and signal quality poor. Bearings from stations were poor. At this time it is not known whether spacecraft HF voice was received by any HF stations. The signals were received from 19:24 to 19:33 G.m.t., with one station reporting signals at 20:00 and 20:05 G.m.t.

The spacecraft recovery flashing light operated properly throughout the recovery operation.

The sea dye marker was observed by all recovery aircraft in the area at ranges from 4 to 10 nautical miles. Dye was still being emitted in large quantities as the spacecraft was brought aboard the carrier.

Signals from spacecraft UHF equipment were received by aircraft as follows:

Aircraft	G.m.t.	Range, n. mi.	Type	Receiver	Frequency, Mc
14	19:16	27	Pulse	SPP	243
	19:15	30	Voice	ARA-25	296.8
15	19:13	57	CW	SPP	243
	19:18	55	Pulse	SPP	243
	-	60	Voice	ARA-25	296.8
16	19:15	107	CW	SPP	243
	19:38	34	Pulse	SPP	243
	19:41	25	Voice	ARA-25	296.8
Coast Guard helicopter	19:20	12	Voice	ARC-52	296.8

6.4.4 Postretrieval Procedures

The spacecraft postretrieval procedures aboard the U.S.S. Intrepid proceeded as specified in the GT-3 Recovery Operations Manual. The spacecraft exterior was examined for apparent damage, and detailed photographs were taken. The swimmers reported that they caused the heat-shield center-top pitted areas. Pitted areas at the top left edge of the shield were not caused during recovery operations. The windows were not completely transparent as a result of condensation behind the outer glass. An area about 1 inch wide around the periphery of each of the windows was clear.

All major pyrotechnics except the harness release actuator and tailute release mechanism were safetied and recorded before spacecraft shutdown. All required equipment was then removed and packaged.

At no time after landing did recovery personnel notice any reentry control system (RCS) propellant fumes being emitted.

The spacecraft onboard cameras, film PCM tape recorder, and voice tapes were flown from the carrier at about 01:30 G.m.t. to Patrick Air Force Base, Florida.

The rendezvous and recovery (R and R) section and the main parachute were never sighted.

At 13:30 G.m.t. on the second day after recovery (March 25), the flight crew left the aircraft carrier via an S2E aircraft. At 18:30 G.m.t. the same day, the spacecraft was unloaded at Mayport Naval Station, Mayport, Florida, where the RCS was deactivated by the Landing and Safing Team. The following morning, 3 days after recovery (March 26), the spacecraft was flown to the Cape Kennedy skid strip. The airplane arrived about 12:30 G.m.t. and the spacecraft was delivered to NASA Cape Operations representatives.

6.4.5 Spacecraft RCS Deactivation

The RCS was deactivated at the Mayport Naval Station to safe the system prior to flying the spacecraft to Cape Kennedy aboard the C-130 aircraft.

The Landing and Safing Team, consisting of NASA and spacecraft contractor engineers and technicians, was responsible for deactivating the RCS according to procedures of reference 5.

After the spacecraft was unloaded from the carrier at Mayport, it was transported by dolly to a previously selected, fairly isolated area where deactivation was begun at 18:35 G.m.t. Normal safety procedures were observed throughout the operation. There was no indication of toxic vapors from any of the 16 RCS thrust chamber assemblies (TCA's) when checked with a portable propellant vapor detector. The RCS shingles were then removed and packed in polyethylene bags.

Before the pressurant in each ring was relieved to atmospheric pressure, the team obtained pressure readings of source pressure from test point TP 1 (A package) of both rings and regulated lock-up pressure from test point 4 (B package) of both rings. A flexible $\frac{1}{4}$-inch inside-diameter flexible hose, 4 feet long, from TP 1 to a calibrated 3000-psi pressure gage, was utilized for this operation. Source pressure readings of 1600 psig and 1490 psig (ambient dry bulb temperature of 76° F) were obtained from A-ring and B-ring, respectively. Regulator lock-up pressure readings of 291 psig and 3000 psig were obtained for the A-ring and B-ring, respectively. The pressures in each ring were then relieved to atmospheric pressure. Immediately after the source pressurant draining operation upstream of the system check valves, the pressurant upstream of the propellant bladders and downstream of the system check valves was relieved through test points TP 4 and TP 6 by venting through separate scrubber units.

Following these operations, nitrogen pressure of 50 psig was utilized through TP 6 to push the remaining oxidizer in A-ring out TP 11 into a propellant holding container. The next step allowed the opening of the motorized valves upstream of the TCA's to provide a direct flow path out the TCA's for the flushing operation.

An external 28 V dc auxiliary power supply was used to open the motorized valves. Before the spacecraft propellant switch was placed in the "on" position, circuit breakers 1 and 3 of RCS A and B on the spacecraft overhead switch and circuit breaker panel were switched to the "off" position for safety purposes. Circuit breaker 2 of the RCS A and B were in the "off" position when the team received the spacecraft.

Following this operation, the proper plumbing connections were made to the oxidizer side of ring A to flush with Freon-MF. To perform this operation, an auxiliary power supply and an electrical control box were required to open the 8 RCS TCA oxidizer solenoid valves. After the equipment was properly connected, a switch on the control box (designed to open only the oxidizer or the fuel solenoid valves in one ring) was positioned from the "off" position to the "oxidizer" position and back to the "off" position almost instantaneously because of a fairly strong static firing in the A-ring TCA's. Contractor and NASA personnel are in the process of checking the electrical control box to determine whether the problem was in the spacecraft electrical system or in the control box electrical system. The flushing of the oxidizer A-ring continued normally after the fuel in the A-ring had been drained into the holding container.

Before the flushing operation began on B-ring, both the oxidizer and the fuel propellants were drained into separate holding containers. The remainder of the deactivation continued normally, and the last operation of the deactivation was to provide a positive 25-psig nitrogen pad on the upstream side of the tank bladders and the pressurant system. However, once the nitrogen padding was completed, flush fluids could be seen leaking from thruster 2 of ring A. This leaking was stopped by closing the motorized valves in the RCS.

All torque values of the caps and valves of test points were recorded and conformed to instructions.

The propellants in the holding containers were brought back to Cape Kennedy and measured by weight. The weights were as follows:

Ring	Oxidizer, lb	Fuel, lb
A	3.68	1.65
B	5.55	4.15

The RCS deactivation required approximately 10 hours to complete. Three hours of this period were used for spacecraft preparation and post-deactivation procedures.

TABLE 6-I.- TELEMETRY COVERAGE

Station	Orbit number	Link 230.4 (real-time), percent	Link 246.3 (acquisition-aid), percent (a)	Link 259.7 (stand-by delayed), percent
MCC	Launch	99.9	20	
	1	100	100	75
	2	80	100	95
	3	100		
Tel II	Launch	100	20	
	2	50		75
	3	100		
Tel III	Launch	100	20	
	1	100	100	100
	2	50		75
	3	100		
BDA	1	100		
	2	100		100
	3	100		100
GBI	Launch	100	30	
	1	100	100	100
	2	68	100	100
	3	80		
GTI	Launch	99	45	
	1	91	85	100
	2	93	100	100
	3	80		
CAL	1	100		
	2	96		
	3	95		

[a] Acquisition aid beacon was turned on shortly after insertion.

TABLE 6-I.- TELEMETRY COVERAGE - Continued

Station	Orbit number	Link 230.4 (real-time), percent	Link 246.3 (acquisition-aid), percent (a)	Link 259.7 (stand-by delayed), percent
ANT	3	100	75	90
ASC	3	99	99	
PRE	2	100	100	
	3	100	100	
CYI	1	99		
	2	99		
KNO	1	100	100	
	2	95	100	
TAN	1	100		
	2	100		
	3	100		
CRO	1	100		
	2	100		
	3	100		
CTN	1	80		
	2			
HAW	2	95	100	
	3	95	100	
GYM	1	100		
	2	100		100
	3	100		

[a] Acquisition aid beacon was turned on shortly after insertion.

TABLE 6-I. - TELEMETRY COVERAGE - Concluded

Station	Orbit number	Link 230.4 (real-time), percent	Link 246.3 (acquisition-aid), percent (a)	Link 259.7 stand-by delayed, percent
TEX	1	100		100
	2	100		100
	3	100		
CSQ	1	98		
	2	99		
	3	97		
RKV	1	100	100	
	2	100	100	
	3	100	100	
RTK	2	100	100	
	3	99	99	

[a] Acquisition-aid beacon was turned on shortly after insertion.

TABLE 6-II. - RADAR COVERAGE TIMES

Station	Orbit	Radar	Actual, min:sec	Nominal, min:sec
PAT	1	C	5:49	6:57
	2	C	None (GLV skin track)	-
	3	C	None (GLV skin track)	-
	4	C	None (GLV skin track)	-
MILA	1	C	6:57	6:57
	2	C	5:01	6:14
	3	C	None (spacecraft skin track)	6:20
	4	C	3:46	4:40
GBI	1	C	6:24	6:13
	2	C	4:20	5:48
	3	C	4:01	6:05
	4	C	1:29	4:41
GTI	1	C	4:35	4:21
	2	C	2:48	3:04
	3	C	4:53	4:55
	4	C	8:54	7:30
ANT	1	C		-
	2	C		-
	3	C	3:58	4:07

TABLE 6-II.- RADAR COVERAGE TIMES - Continued

Station	Orbit	Radar	Actual min:sec	Nominal, min:sec
BDA	1	S	5:38	6:44
		C	6:22	
	2	S	5:53	6:35
		C	6:43	
	3	S	4:08	6:18
		C	5:51	
CYI	1	S	6:36	6:52
		C[a]	4:18	
	2	S	4:36	4:50
		C	4:42	
WOM	1	C	Passive only	7:57
	2	C	Passive only	6:11
CRO	1	S	6:15	7:02
		C	6:26	
	2	S[b]	5:54	5:56
		C	5:14	
	3	S	6:42	6:41
		C	5:48	
HAW	1	-	-	-
	2	S	4:36	5:13
		C	4:42	
	3	S	5:06	6:23
		C	6:20	

[a] C-band on engineering-test basis

[b] S-band on engineering-test basis

TABLE 6-II.- RADAR COVERAGE TIMES - Concluded

Station	Orbit	Radar	Actual, min:sec	Nominal, min:sec
CAL	1	S	Passive only	3:18
		C	1:59	
	2	S	3:00	6:05
		C	4:15	
	3	S	2:37	6:09
		C	5:40	
GYM	1	S	3:54	6:31
	2	S	Passive only	6:15
	3	S	Passive only	5:34
WHS	1	C	2:00	6:07
	2	C	3:00	6:31
	3	C	3:36	5:53
TEX	1	S	6:12	6:26
	2	S	6:01	6:01
	3	S	Passive only	5:17
EGL	1	C	3:58	6:31
	2	C	2:43	6:26
	3	C	4:02	4:56

TABLE 6-III.- MISTRAM TRACKING PERIODS

(Antenna pattern not favorable during first 100 seconds after lift-off)

Acquisition, LO+sec	Loss, LO+sec	Comment
Valkaria MISTRAM		
25.35	62.75	On track on base line
62.75	79.55	On track short only base line
84.35	153.35	On track on base line
153.8	389.45	On track on base line
Data used for impact predictor (IP) by AFETR computer		
41.4	62.75	Long and short base line
72.9	79.55	Short base line
94.45	153.35	Long and short base line
164.10	389.45	Long and short base line
Eleuthera MISTRAM		
168.95	181.70	Passive
219.5	337.65	

TABLE 6-IV.- RECOVERY SUPPORT

Landing area	Access time, hr.		Support
	Aircraft	Ship	
Launch site:			4 LARC (amphibious vehicle) 2 LVTR (amphibious vehicle with spacecraft retrieval capability) 2 boats (40 and 50 feet long) with underwater salvage team
Pad	5 min		
Land	10 min		
Water (ejected)	2 min		
Water (spacecraft	15 min		1 LCU (large landing craft with spacecraft retrieval capability) 4 HH-3C helicopters 2 MSO (minesweeper with salvage capability)
Launch abort:			9 aircraft on station (1 HU-16, 5 HC-54, 2 HC-97, 1 P-3)
A	4	9 to 10	
B	3	3	7 aircraft on standby at Patrick Air Force Base, Bermuda, and Las Palmas
C	3	3	
D	3	3	1 ATF (spacecraft retrieval and deep water salvage capability) 7 destroyers, 1 oiler, 1 cruiser
Primary (4-1)	4	1	3 aircraft on station (HC-54) 1 destroyer 2 Coast Guard cutters with 1 helicopter each 1 aircraft carrier with SH-3A helicopters (5 used)
Secondary 2-1 and 3-1	3	1	2 destroyers per area (1 from launch abort area) 2 aircraft (1 HC-97, 1 P-3 from launch abort area)
Contingency	18		30 aircraft on strip alert - worldwide at staging bases.
Total			19 ships, 11 helicopters, 42 aircraft

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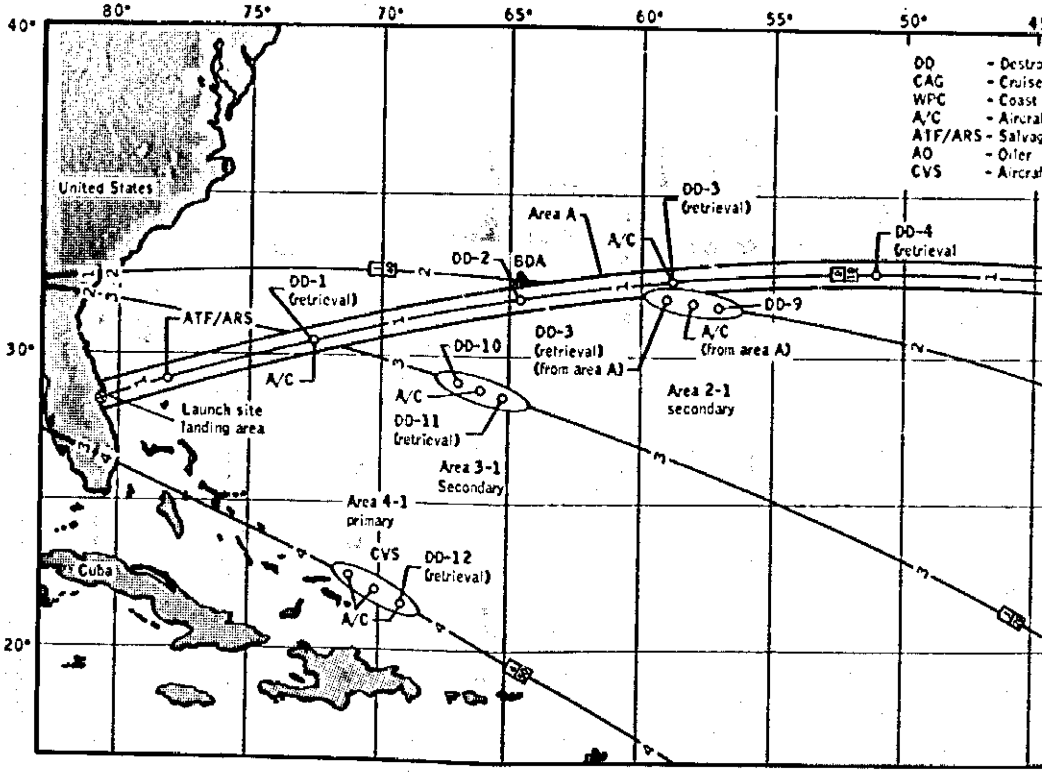

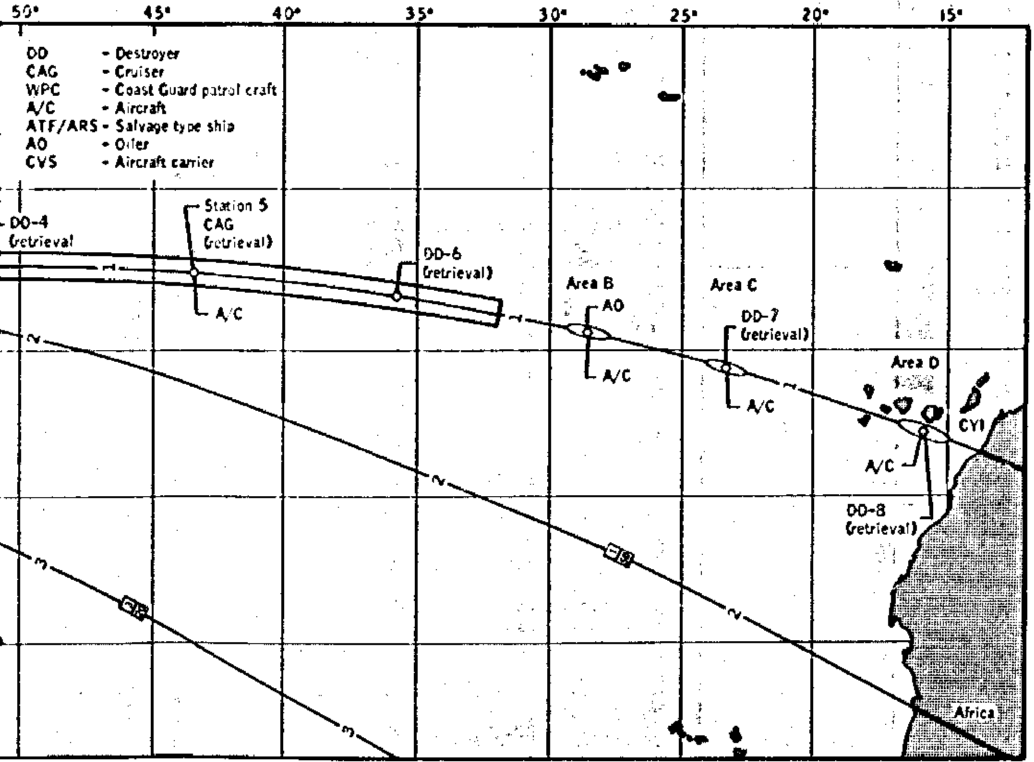

Figure 6-1. - GT-3 planned landing areas and downrange recovery force support.

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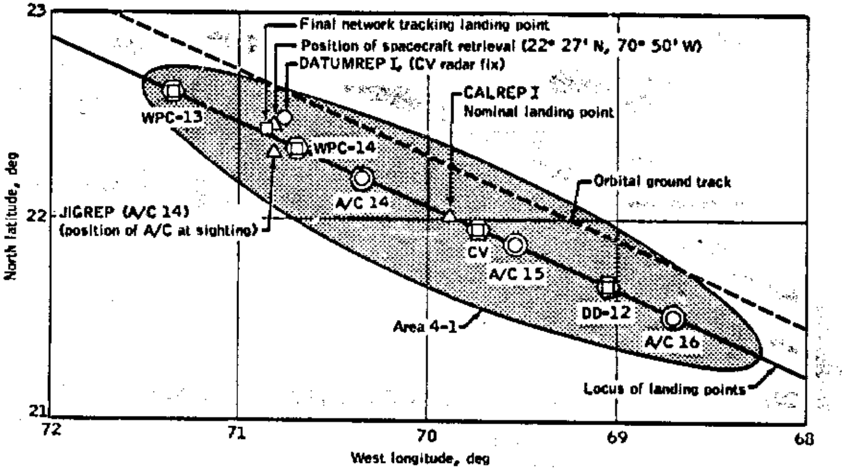

WPC - Coast Guard patrol craft
A/C - Search and location aircraft
CV - Aircraft carrier Intrepid
DD - Destroyer
CALREP - Calculated landing position based on data available shortly after retrofire
DATUMREP - Best position estimate of spacecraft after landing
JIGREP - Visual sighting of spacecraft after landing

Figure 6-2. - Details of primary landing area.

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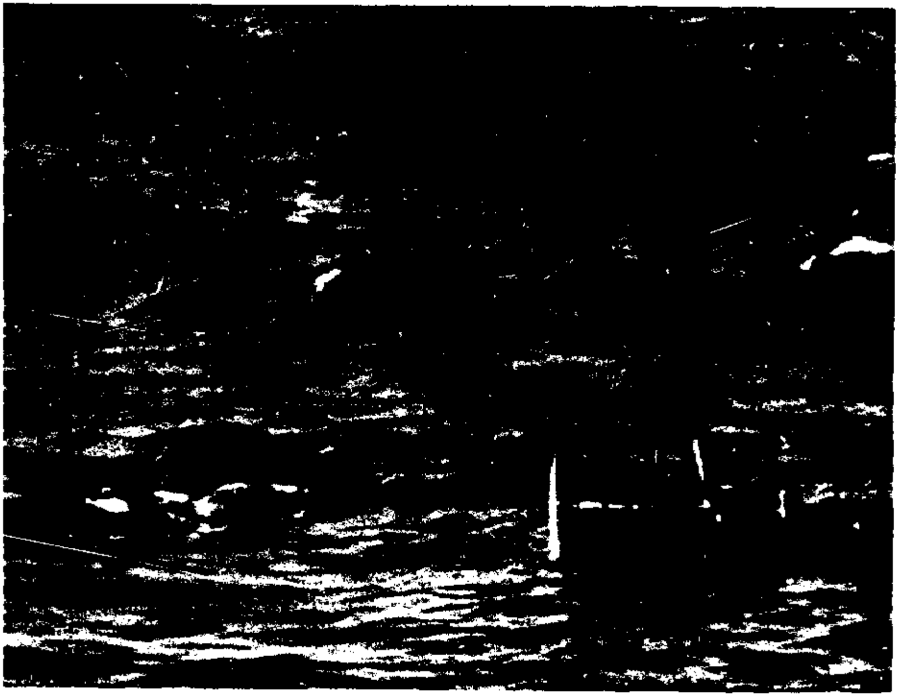

Figure 6-3. - GT-3 spacecraft shortly after landing.

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Figure 6-4. - Helicopter retrieval of GT-3 crew.

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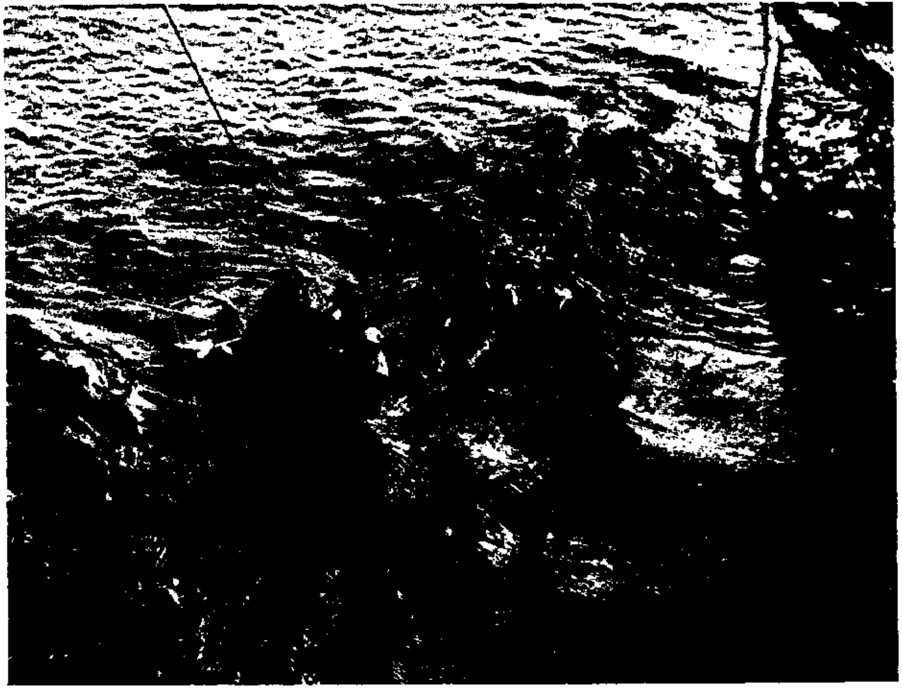

Figure 6-5. - In-haul line attached to GT-3 spacecraft.

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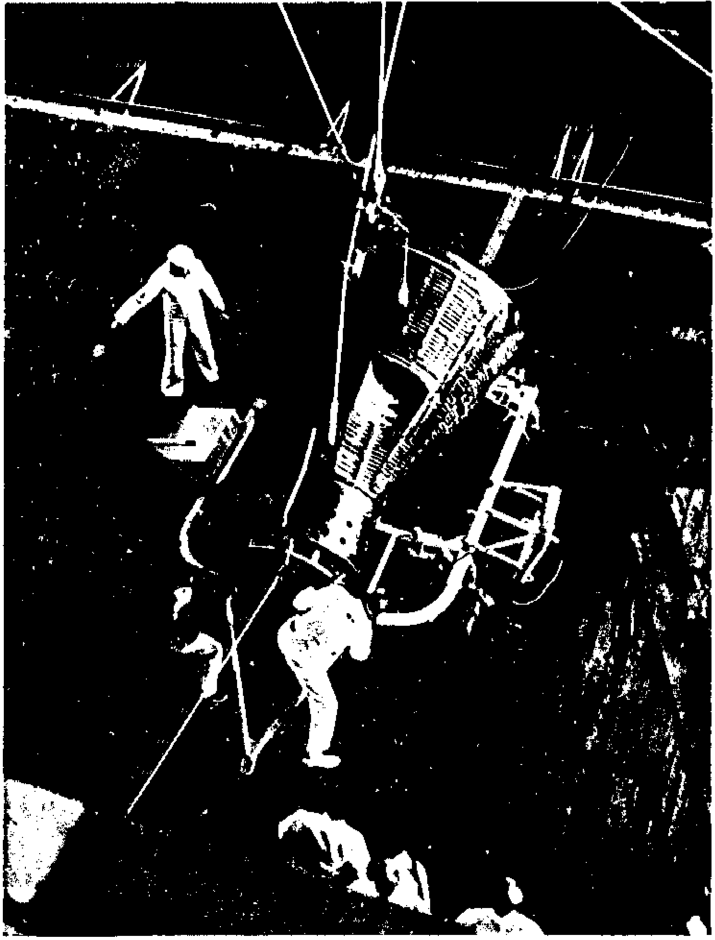

Figure 6-6. - GT-3 spacecraft positioned in dolly aboard aircraft carrier Intrepid.

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NASA-S-65-3633

Astronaut Virgil I. Grissom, Command Pilot

NASA-S-65-3606

Astronaut John W. Young, Pilot

7.0 FLIGHT CREW

7.1 FLIGHT CREW PERFORMANCE

7.1.1 In-Flight Activities and Training

The prescribed flight crew activities during the GT-3 mission are documented in the flight plan dated March 4, 1965, with revisions dated March 16 and March 22, 1965. A summary of the flight plan is presented in figure 7-1. Crew training in preparation for the GT-3 mission started in April 1964 and continued through March 22, 1965.

7.1.1.1 In-flight activities.- The flight crew performed the required flight activities at approximately the planned times with some minor exceptions. A summary of the crew activities performed is presented in table 7-I, and the pilots' reports on their activities are in section 7.1.2.

7.1.1.1.1 Translation maneuvers: Three spacecraft translation maneuvers were to be accomplished on this mission. These were an orbit adjustment in the first orbit, an out-of-plane translation system check in the second orbit, and an orbit attitude and maneuver system (OAMS) preretro translation prior to retrofire.

(a) Orbit adjust translation burn - The orbit adjustment burn scheduled for 01:33:00 g.e.t. was to be accomplished with the spacecraft in small-end-forward (SEF) attitude using the forward-firing maneuver thrusters. The burn was to be made with the platform in orbit rate mode, the computer in the catchup mode, and the OAMS attitude control mode in rate command. Duration for the 48-fps burn was given to the crew as 74 seconds. The 48-fps ΔV was to be inserted into the incremental velocity indicator (IVI) after computer catchup mode initiation, then driven to zero during the burn.

The pilot initiated catchup mode and inserted the 48-fps ΔV into the IVI. However, prior to the start of the burn, ΔV had increased to 51 fps (this behavior is discussed in section 5.1.5.2.2). Thrusting was initiated at 01:33:00 g.e.t. (with an IVI readout of 51 fps forward) and was continued for 75 seconds until 01:34:15 g.e.t. at which time the IVI read 2 fps forward, indicating a ΔV of 49 fps. Rates were controlled within the ±0.2 deg/sec deadband automatically by the attitude-control system during the burn. However, interaction rates in pitch, roll, and yaw due to the translational thrusting about an offset spacecraft center of gravity caused the spacecraft to change pitch, roll, and yaw attitudes at a rate equal to the deadband rate of +0.2 deg/sec. The resulting attitude change was noticed by the command pilot, and short-duration stick inputs (approximately $\frac{1}{2}$-sec bursts) were commanded to hold attitude. Inasmuch as the rates commanded by the pilot when overriding rate command are 20 to 30 times greater than the deadband rate, instantaneous overdamping occurred. The highest instantaneous rates were commanded by stick inputs and did not exceed -2 deg/sec in roll and yaw. The highest rate due to pitch stick input was -1.3 deg/sec.

(b) Translational systems check (out-of-plane) – The computer mode was to be catchup; the platform mode, orbit rate; and attitude control mode, direct. The spacecraft was to be yawed to +90° with pitch and roll at 0°. The procedure was to thrust aft until the forward-aft IVI read 10 fps, then thrust forward for 1 fps, then aft for 1 fps, then forward for 1 fps, then aft for 1 fps. The attitude indicator (8-ball) was to be used for attitude reference during the translations. The check was to be terminated by yawing the spacecraft back to small-end-forward (SEF) attitude and then switching the attitude control mode to horizon scan.

The initiation time of the check was nominal. Thrusting started at 02:18:40 g.e.t., and 10 fps was obtained in 16 seconds. The command pilot elected to use out-the-window reference rather than the 8-ball because of an orbit rate or a platform misalinement problem which had not been isolated at that time.

Telemetry data showed that the platform yaw gimbal during this 10-fps translation varied from +70° to +72° to +69° at end of thrust. Interaction rates in pitch, roll, and yaw due to translational thrusting about an offset c.g. were present during the 10-fps change in velocity. The command pilot damped these rates throughout the burn. However, a slight amount of overdamping was induced as a result of the much greater rates of the OAMS attitude thrusters.

Following the 10-fps translation, a 1-fps forward translation was accomplished followed by a 1-fps aft translation. The other two translations (1-fps forward and 1-fps aft) were not accomplished, and a yaw attitude change was initiated at 02:19:18 g.e.t. in order to return the spacecraft to SEF attitude.

There were no pitch and roll perturbations recorded on the voice tape as planned, probably because of the extremely small perturbations.

(c) OAMS preretro translation – This translation was scheduled to begin 12 minutes prior to retrofire, with T_R being 04:33:23 g.e.t. The flight plan called for the platform to be in orbit rate, the computer in catchup, and the attitude control mode in rate command. The aft-firing thrusters were to be used with the spacecraft in the blunt-end-forward (BEF) position. The ΔV to be subtracted was to be inserted into the IVI prior to T_R - 12 minutes and following initiation of the computer catchup mode. The ΔV to be subtracted was transmitted via voice to the spacecraft crew as 96 fps with a burn time of 109 seconds. Further, as a result of a possible problem with the orbit-rate platform mode, it was decided to make the translation using the platform BEF mode.

The 96-fps ΔV was inserted into the IVI at approximately 04:19:00 g.e.t. When the translation was initiated approximately 2 minutes later at 04:21:22 g.e.t., the reading had increased to 98 fps. The translation was initiated and lasted until 04:23:14 g.e.t. at which time the IVI readout had decreased to 1 fps. Because the IVI was used as the prime indication for the translation, the translation time was 3 seconds longer than estimated. Telemetry gimbal-angle data during the translation indicate that attitudes were held very close to nominal. Pitch gimbal readouts indicate maximum pitch angles of +0.684° and -2.16°, maximum roll angles of +0.144° and -1.666°, and maximum yaw angles of +0.612° and -0.54°.

7.1.1.1.2 Tracking: The tracking task was initiated at 03:02:00 g.e.t. as the spacecraft was approaching the west coast of the United States. The spacecraft was pointed at the planned target site which encompasses the area from El Centro, California, south to Mexicali, Mexico, as shown in the photograph taken by the pilot, figure 7-2. Yuma, Arizona, the Colorado River, the southern tip of the Salton Sea, and the northern tip of the Gulf of California are also identifiable in the photograph. During this task, the optical sight was located in the command pilot's window, and the pilot was to photograph the site for target identification. As a result of considerable cloud cover in the area, it was difficult to select a target point to be tracked, and acquisition occurred too late in the pass to establish the required tracking rate with the pulse mode. Direct control mode was selected, and the required tracking was accomplished with an accuracy of about $\pm\frac{1}{2}°$ as reported by the pilot. The tracking task was further complicated by the illumination of the sight reticle pattern which was too dim against the bright earth background, causing the pilot to lose the sighting reference intermittently during the tracking pass.

7.1.1.1.3 Reentry control: Reentry control procedure as planned preflight and as accomplished in flight was essentially identical. The procedure called for attitude control and maneuver electronics (ACME) direct control mode using two rings of RCS. After retrojettison, the spacecraft was to be controlled to 180° roll, 0° pitch, and 180° yaw. This attitude was to be held until approximately 2 minutes after the time of 400 000-foot altitude as transmitted from the ground. The spacecraft was then to be rolled to the ground-computed bank angle and held in this attitude until the crossrange error displayed on the flight director indicator (FDI) was nulled. After nulling the crossrange error, the spacecraft was to be rolled to 180° to obtain full lift until the downrange error indicators were nulled. After correcting for crossrange and downrange, the spacecraft was to be rolled at a constant rate until approximately 80 000-foot altitude at which time the roll rate was to be stopped in preparation for drogue parachute deployment. Throughout the period from initiation of bank until drogue parachute deployment, the pitch and yaw rates were to be damped. During the actual reentry, the pilot selected ACME direct control mode with two rings of RCS. The crossrange error was nulled by using a 45° bank, and the spacecraft was then rolled to a 180° roll angle (full lift). The downrange error did not null, however, and constant roll rate was not initiated. The sequence of control events that occurred during the reentry period is shown in table 7-II. During the lifting portion of the reentry, the spacecraft oscillatory pitch and yaw rates were generally controlled loosely and allowed to build up to ±9 deg/sec; however, when the rates reached these magnitudes, the pilot damped to ±2 deg/sec.

7.1.1.2 Crew-related mission objectives performance.- Review of the entire mission in light of the mission objectives indicates that all of the crew-related mission objectives were met. The crew was able to perform well in the two-man Gemini vehicle. The presence of two men enabled computer operation and systems control in parallel with manual attitude and maneuver control of the spacecraft and the performance of tasks required by the experiments. Also, a greater number of systems evaluations can be conducted during a given time span. Spacecraft maneuvering capability with the OAMS system was demonstrated, and the crew considered the spacecraft easily maneuverable. The crew station controls and displays were evaluated along with the environmental control system (ECS), suits, guidance and control (G and C) systems, communication systems, and food and waste management system. Some problems did occur in these systems; however, valuable information was obtained on required fixes and, in general, the systems were adequate. Considerable difficulty was experienced with the pilots' stowed equipment; however, all the equipment was evaluated and sufficient information was obtained to evaluate the problems. The pilot took several photographs from orbit with a 70-mm camera, and the results were judged excellent. The crew was able to operate the experimental equipment, and the experimental objectives were met on two experiments. Detailed discussion of the systems and crew equipment from a hardware standpoint is covered in other sections of this report. (See sections 5.1.10 and 8.0.)

7.1.1.3 Flight plan variations.- The horizon-scan control mode characteristics check scheduled for an elapsed time of 00:02:30 g.e.t. was deleted. During this time, the command pilot was analyzing the orbit-rate-mode platform problem and was also assisting the pilot with the food and waste evaluation which was scheduled concurrently.

Minor shifts occurred in the actual time that various activities were performed. However, the flight crew was able to reschedule and complete these activities without compromising the mission.

The food and waste evaluation was completed; however, the pilot noted in the postflight debriefing that sufficient time was not available for a complete evaluation, especially during the latter phases of this evaluation.

The command pilot was not able to obtain any blood-pressure measurements because of his inability to connect the blood-pressure bulb. (See section 5.1.10.7.5.)

The spacecraft drifted in yaw during the first and second orbits as a result of the launch heat-exchanger exhaust; however, the drift was easily controlled by the crew and did not seriously interfere with the completion of the planned activities. (See section 5.1.4.)

The suit integrity checks scheduled during the first and second orbits were not performed. The flight crew, however, felt confident of suit integrity.

7.1.1.4 Crew training.- Crew training has been accomplished by crew participation in spacecraft tests in order to gain familiarization with spacecraft 3, and on various simulators and training devices as described in the following paragraphs.

7.1.1.4.1 Spacecraft tests: The GT-3 prime flight crew spent approximately 35 hours in spacecraft 3 and several times this amount as outside monitors and participants during the spacecraft tests at McDonnell Aircraft Corp. and at Cape Kennedy. Table 7-III represents the cockpit hours accumulated by each crew member, and table 7-IV is a summary of all the major preflight training activities.

7.1.1.4.2 Gemini mission simulations: The GT-3 crew started training on the Gemini Mission Simulator (GMS 2) at St. Louis, Missouri, in April 1964. During an approximate 2-month period, each crew member obtained 35 hours of general familiarization and part-task training at this location. They resumed training on the Gemini Mission Simulator at Cape Kennedy on November 19, 1964. During the 4 months preceding the GT-3 launch, each crew member accumulated an additional 80 hours of simulation training, 40 hours of which were in the Gemini pressure suits. Particular training emphasis throughout this period was placed on practicing procedures during the launch phase of the mission. The majority of the in-orbit training activities consisted of practicing critical orbital tasks and rehearsing parts of all of the three orbits. A summary of the GT-3 mission simulator training is included in table 7-V. Generally, the Gemini Mission Simulator training was satisfactory and provided an excellent simulation of the spacecraft as indicated by flight performance and postflight crew comments. There did exist, however, a few training equipment inadequacies which reflected upon crew in-flight activities. These areas are elaborated in succeeding sections of this report.

7.1.1.4.3 Special and part-task training activities: The GT-3 flight crew completed several training programs which were conducted to give them experience with as many of the space-flight conditions as possible. Some of the more important training activities of this nature are described in the following paragraphs.

Parachute - Each crew member completed several parachute tows with attendant release and drop onto land and in the water to prepare them for a possible mode I abort situation.

Egress training - The flight crew received spacecraft water egress practice in the flotation tank and in the Gulf of Mexico using boilerplate 201 and static article 5. Training consisted of briefings, films, demonstrations on use of egress and survival equipment, and practice in "shirt sleeves" as well as with full egress equipment.

Centrifuge - The GT-3 crew participated in two Gemini centrifuge programs conducted at the Naval Air Development Center, Johnsville, Pennsylvania. During these programs, launch and reentry acceleration profiles were experienced, and the spacecraft was controlled during normal and selected abort simulations.

Launch abort training - The GT-3 crew members each participated in two launch abort simulation programs on the moving-base simulator. This simulator permitted the crew to experience some of the vibration cues in conjunction with various abort situations and definition of optimum abort procedures for a wide variety of launch vehicle or spacecraft systems malfunctions.

Planetarium - The crew made two trips to the Morehead Planetarium, Chapel Hill, North Carolina, for reviewing the entire celestial sphere, and, in particular, those portions near their launch date orbital track. The primary purpose of this training was for backup spacecraft orientation and navigation in case of inertial platform or communications failure.

Briefings - The crew received formal systems briefings of two or more days each at Houston, Texas, St. Louis, Missouri, and Cape Kennedy. In addition, they participated in many informal systems briefings in conjunction with various training activities. Flight-plan reviews were conducted on a periodic but continuous basis throughout their preflight training program. Several informal briefings on medical aspects, experiments, procedures, and other operational reviews were conducted.

7.1.1.5 Evaluation of training.- Overall, the training prepared the GT-3 flight crew adequately for the mission. The flight crew was particularly proficient in the launch phase and the more critical activities associated with the orbit and reentry phases. Because of the early and extensive participation in the development of the flight hardware, the crew members had an excellent systems background prior to commencement of their preflight training program.

From the experience gained in this flight, it appears that some change in emphasis in the crew training program would be beneficial.

7.1.1.5.1 Out-the-window reference: Very early in their training program, the crew received some practice in controlling the spacecraft attitude and rates by using an external visual display at McDonnell Aircraft Corp. and on the Gemini part-task trainer. Additional practice using an out-the-window reference during the reentry phase of the mission, as well as experience in using out-the-window references during the launch and orbital phases of the mission, would have been beneficial.

7.1.1.5.2 Operational and experimental crew equipment: Additional training in the handling and operation of the stowed equipment would have been desirable primarily to save crew time in performing the assigned tasks associated with this equipment. This problem can be alleviated by earlier availability of training equipment and an early freeze on equipment and location configuration.

7.1.1.5.3 Reentry simulation: Additional reentry simulation flights, both with instruments and out-the-window display, and particularly in conjunction with ground-support personnel, to refine air-to-ground communications procedures further would have been desirable.

7.1.1.6 Summary.- During the GT-3 mission there were no major discrepancies in flight crew performance, and crew-related mission objectives were met. The crew responded correctly to the few minor spacecraft anomalies. The minor deviations in the flight plan were expected because flight plans are written for nominal missions and any anomalies require alterations to the planned activities.

7.1.2 Flight Reports

The following report is the command pilot's and pilot's observations during the GT-3 mission. The material is presented as remembered by the flight crew using their notes and the onboard tapes for reference.

7.1.2.1 Command pilot's report.-

7.1.2.1.1 Prelaunch: The prelaunch operations went very smoothly starting with wake-up, physical examination, breakfast, suiting up, and insertion. The countdown went smoothly with the exception of a 24-minute delay caused by a fuel leak at the base of the launch vehicle. We were aware of the spacecraft status at all times; however, we would have liked to have had more information on the Gemini launch vehicle (GLV). When we were inserted into the spacecraft (fig. 7-3), the spacecraft count was 20 minutes ahead of the GLV which meant normally that we would spend 20 extra minutes in the spacecraft prior to launch.

From the crew standpoint, it would have been better to wait until the count called for crew ingress. The less time the crew has to spend in the spacecraft prior to launch, the more efficient they are going to be. This is particularly important for the longer duration flights. Although I was uncomfortable throughout this period, it did not reach the point of being painful or in any way degrade my performance subsequent to launch. My feet felt as though they were going to sleep at times, but I did not have any severe pressure points. The only thing that was irritating to me was something on the headrest which stuck to my helmet so that I could not raise my head.

Prelaunch communications were good. We had good hardline from Mission Control Center (MCC) and the blockhouse, and all RF communications were clean. The only difficulty occurred when the pilot and I were on different missile operations interphone systems (MOPIS) channels; then our transmissions were subject to crosstalk.

Prelaunch procedures were generally good. The positions of the controls and switches were well established prior to crew insertion. Verification of proper switch and control positions for launch required only a few minutes. The procedure for firing the orbital attitude and maneuver system (OAMS) thrusters was satisfactory; however, I am not positive that the crew can determine that the OAMS thrusters are operating properly. I could see the reentry control system (RCS) pitch-down thruster firings (a reddish-orange flame) with which there is a very definite sharp sound. After seeing and hearing the RCS pitch-down thrusters fire, it was easy to relate the sound to the remaining RCS thruster firings. The OAMS thrusters were farther from the cockpit, and the sound was slightly muffled.

The preflight training was quite adequate for prelaunch procedures. We had spent many hours in the spacecraft and the Gemini mission simulator during preflight training. The only new situation was the occurrence of noise and vibration when the prevalves were opened, otherwise it was very much like being in the Gemini mission simulator. The only additional prelaunch training that I would have liked would have been slide-wire practice wearing the Gemini pressure suit. This would have increased my confidence in my ability to get away from the white room in case of an emergency.

7.1.2.1.2 Powered flight: The CAP COM gave a countdown to lift-off. I could hear the engines light off, and the clock started running at the same time CAP COM called the bolt firing. There was not a distinct feeling when lift-off occurred. It was a gentle, smooth lift-off with no jolt or disturbance and with no cue as to when the bolts blew. The booster instruments functioned exactly as expected. The two engine lights went out at the time CAP COM called ignition. A few seconds after lift-off, vertical motion could also be detected out the window. The first-stage fuel and oxidizer pressure indicators dropped slowly and leveled off within normal limits. The roll program started and stopped right on time, and this could also be detected out the window. The pitch program commenced on time, and it was very smooth and steady. CAP COM informed us that we were "slightly high on flight path but no problem" just prior to 2 minutes elapsed time, which was reassuring and gave advance warning that the booster would probably pitch down at BECO similar to GT-2.

The noise and vibration during first stage was quite low, less than what we had experienced on the moving-base simulator. We picked up some aerodynamic noise prior to approaching sonic velocity. At this point, I noticed some shock waves and the antenna fairing fluttering. It gave me some concern because if it had come loose it might have hit the window. After reaching sonic velocity, it became quiet, and the flight was smooth throughout the rest of the launch.

Prior to BECO, we were pitched down sufficiently to see the horizon. At staging, the sudden drop from approximately 6g to 1g or less was, of course, very apparent but quite smooth. At this time, we saw quite a bit of outside debris and a flash of fire from the "fire in the hole" second-stage start. I expected this, but it surprised the pilot. I do not recall the engine 1 lights blinking on, but engine 2 light went out immediately. Second-stage fuel and oxidizer indicators operated properly.

At radio guidance system (RGS) initiation, the booster pitched over very smoothly to about 10° below the horizon, and stayed there quite some time. It then gradually steered back up above the horizon, and then back down to it. At about 4 minutes elapsed time, we got a GO from

Cape CAP COM. Second stage was smooth; however, it had a more pulsating sound than the first stage. I did not notice any effects of POGO during powered flight.

7.1.2.1.5 Insertion: SECO was clean and sharp. I could feel just a little bit of tail-off. At 29 seconds after SECO, the incremental velocity indicators (IVI's) counted up. I then started thrusting, using rate command for attitude control, and the pilot separated the spacecraft from the booster. I could not hear the aft-firing thrusters, but we could tell we were accelerating. At this time, there was quite a bit of debris moving by outside the spacecraft. The IVI's indicated a 17-fps overspeed before separation burn. I forgot to check the time I started the seperation OAMS burn and thrusted until an elapsed time of 6 minutes 9 seconds, which I had used during training. I could not hear the aft-firing translation thrusters but could hear the attitude thrusters. I think this was the reason that the pilot was a few seconds slow in switching to rate command at separation. Upon separation and translating forward, the spacecraft had a tendency to pitch down just as it did on the Gemini mission simulator. The spacecraft had built up a small pitch-down rate while performing the separation burn; however, the pilot switched to rate command, and I was able to aline the spacecraft quickly to the horizon in all three axes. Upon conclusion of the separation maneuver, my IVI's read 29-fps overspeed. We punched off the horizon-scanner covers and antenna fairing, which also made quite a bit of noise and debris. I was a little concerned about some of this debris getting into the horizon scanners.

We started through the insertion checklist at this time, but the pilot did not get the last item (battery check) completed until we were over Africa. The critical items - boost insert "safe," equibs off, D-ring stowed, and drogue pins in - were done immediately. I did not test the telelights at this time. We stowed the arm restraints, but I did not stow the elbow support. It was up the entire flight, and I did not notice it until we were reentering.

Cape CAP COM called up our perigee and apogee altitudes which sounded to me like 87 to 145 nautical miles. These altitudes would have coincided quite well with an approximate 30-fps overspeed condition, which is just about what was displayed on the IVI's (29 aft). We became suspicious about this situation during the first OAMS retro-maneuver which called for a 48-fps velocity reduction rather than the nominal 66 fps. Examination of the ground-to-air communications, postflight, verified that Cape CAP COM did relay, at approximately 7 minutes 30 seconds elapsed time, that we were inserted into an 87 to 125-nautical-mile orbit. This situation had no effect upon the flight except that the IVI's were apparently not giving correct information at insertion. (However, a satisfactory orbit would have resulted from the application of the IVI information.)

The entire launch phase was nearly perfect. Communications were excellent throughout this period. I felt that I was well prepared for this phase of the mission. The training simulators duplicated the launch quite well except the flight was much smoother than what we had experienced on any of the simulators, particularly the moving-base simulator. The moving-base simulation provided very good training, but the vibrations, POGO, and noise we experienced on this trainer were much too extreme. The Gemini mission simulator is very similar except that it simulates greater rates at maximum dynamic pressure (max q) and at RGS initiate than we experienced on this flight.

7.1.2.1.4 Orbit:

Platform alinement - Shortly after insertion, I alined the platform with the spacecraft using the horizon as a reference. Acquiring pitch and roll by using the horizon was very simple. Yaw alinement was a little more difficult, but by pitching down approximately 10° to 15°, the zero yaw attitude became apparent and agreed with the zero yaw position on the attitude indicator (8-ball). Alinement at night by using external references can be accomplished accurately, but requires more time. The alinement mode worked fine throughout the flight. At the completion of each alinement, the platform and horizon agreed very well. Either method of alining the platform (instruments or window) is quite satisfactory. In either case, alinement requires only a few minutes, and I believe a somewhat better job can be done by using the flight director indicators. The alinement torquing rate was similar to what we expected. During one alinement check, we displaced the spacecraft 10° in each axis, and the alinement was accomplished within 5 minutes.

An out-the-window display on the Gemini mission simulator would have increased my confidence in alining the platform visually. During the first two or three yaw alinement attempts, the pilot and I disagreed as to the exact yaw heading by about 3° or 4°. Later in the mission, we agreed perfectly. Determining yaw alinement was a learning process, and we would have had more confidence in our abilities in this area if we had had a good out-the-window display system during training.

Shortly after the start of the second orbit, I noticed that the 8-ball was not in agreement with my view of the horizon. I was conducting a check of the horizon scanners during which the entire horizon scanner band is exposed to the sun. I alined the platform with the horizon. However, upon going to orbit rate, the 8-ball immediately started to drift off in roll. I believe the error was much greater than what could have occurred as a result of an error in alinement in the yaw axis. In a couple of instances, the 8-ball was in error in roll by as much as 25°. This error, together with undetermined yaw drift and the apparent inadvertent thrusting in the wrong direction that occurred in the horizon-scan mode each time the scanner-ignore light illuminated, resulted in skepticism as to proper platform operation throughout most of the flight. (For discussion of platform alinement see section 5.1.5.)

Thruster and control mode operation - I checked the direct and reentry rate command-control modes shortly after the initial platform alinement. The direct-control mode worked almost exactly as it did on the Gemini mission simulator. The reentry rate-command deadband appeared to be too large, on the order of ±5° or 6° in pitch and yaw rather than the nominal ±4° in these axes. I do not recall what the deadband was in the roll axis. This definitely ruled out the reentry rate command as a preferred control mode during reentry. The pulse mode and rate-command mode functioned as expected. The rate-command mode definitely is the best mode for maintaining spacecraft attitude during translations and retrofire; however, the direct mode is certainly also satisfactory for these operations. The horizon-scan mode is the preferred pilot-assist mode, and the pulse mode is preferred for the attitude changes when performing experiments.

The pulse mode is excellent for controlling and making fine spacecraft attitude or rate adjustments. I did have some difficulty tracking a town just north of the Gulf of California using the pulse mode; however, there were several mitigating circumstances which increased the difficulty of the tracking task. Perhaps the biggest problem was the absence of good tracking targets as a result of cloud coverage. We finally located a small area that was clear, and we had to hurry to get the bore-sight reticle on the target. By this time the target was almost below me and thus the rates were changing rapidly. I tried to track the target by using the pulse mode, but it was moving too fast so I switched to direct. Using this control mode, I was able to hold within $\frac{1}{2}^{\circ}$ of the selected target point. My biggest difficulty at this time was seeing the "pipper" on the target. It was not sufficiently bright to stand out against the earth. It would appear and disappear, depending upon the brightness of the earth's terrain. The pulse mode would probably work satisfactorily if the target could be acquired early and the reticle were sufficiently bright. Normally, the weather will probably not permit early acquisition, and the use of the direct-control mode will be required.

The direct mode worked beautifully throughout the flight. This is perhaps the best mode for any large or rapid control maneuvers. Both the direct- and rate-command modes are excellent for tight control of the spacecraft RCS rings. There is a big difference between using one ring and two RCS rings. Two-ring authority gives the spacecraft a sharp kick, whereas, one-ring control is felt more as a nice soft push. One-ring control is the better method except during the retrofire event.

There is a tendency to over-control using two rings and, of course, fuel consumption is much higher. The direct mode worked extremely well during reentry. Control response was rapid, and it was very simple to damp out spacecraft oscillations with just a few controller "blips."

Throughout the first two orbits, the spacecraft continually drifted to the left in yaw while in the horizon-scan or pulse-control mode of operation. I would bring it back to zero yaw, and it would drift again to the left, building up to as much as 3 deg/sec. We thought that it was a thruster leak and, consequently, went through several procedures to try to stop it, all with negative results. As the flight progressed, this yaw-left drift steadily decreased. I did not notice any drift during the third orbit.

The horizon-scan control mode appeared to operate properly; however, as a result of the yaw drift and other anomalies, an analysis as called for in the flight plan was not accomplished. The horizon-scan control mode maintained pitch and roll attitudes within the deadband (±5°) if yaw drift was controlled and the scanner-ignore light did not illuminate. It appeared as though we were getting a scanner-ignore frequently within the nominal scanner operation limits. When this occurred, which was quite frequently, thrusting in the wrong direction would result. We did a lot of switching between the primary and secondary scanners trying to keep them operating. It appeared that they had a duty cycle time of about 1 hour. The thrusting that occurred upon getting the scanner-ignore signal was opposite to that which should normally occur to maintain the proper horizon-scan mode limits. On a few occasions, the resultant rate would build up to approximately $2\frac{1}{2}$ to 3 deg/sec in a pitch-down direction. The ignore light also came on within the nominal horizon-scanner limits during the night side of the orbit.

As a result of these anomalies, I did not have too much confidence in the platform and began to hold the spacecraft quite close to 0°, 0°, 0° or 0°, 0°, 180° positions as appropriate. I wanted to make sure that I had good alinement for the retrofire maneuver and the final OAMS retromaneuver; therefore, I flew much of the last orbit in the pulse mode and monitored 8-ball and horizon alinement quite closely utilizing the small-end-forward (SEF) or blunt-end-forward (BEF) reference modes.

<u>Translation maneuvers</u> – Our first OAMS burn called for a 48-fps ΔV reduction, instead of the nominal 66-fps reduction. This caused us to wonder about our IVI readout accuracy at insertion. We inserted the velocity into the computer and started the computer approximately 3 minutes before the time to burn. The IVI's started to count up immediately, and 48 came up in the proper window. Prior to the burn time, however, the IVI was reading 50. (For an explanation of this behavior,

see section 5.1.5.2.2.) I used rate command for attitude control, and the spacecraft had a tendency to pitch down which is what we had experienced on the Gemini mission simulator. I burned 48 ΔV leaving 2 ΔV in the window. This first burn required 1 minute 14 seconds using the forward-firing thrusters which was close to the expected time. I could not hear the forward-firing thrusters but could hear the attitude thrusters. A good cue that I was still translating was by the debris moving toward and sticking to the instrument panel. Except for the change in IVI readout prior to thrusting, the first burn was accomplished without incident.

We made the out-of-plane translation check controlling the spacecraft attitude with the direct-control mode. Direct control was certainly satisfactory for maintaining good attitude, but the rate-command mode is without a doubt the easiest method. The translation thrusters, both forward and aft, appeared to fire very well.

The final OAMS retro was accomplished quite smoothly. We waited until closer to burn time to start the computer, thinking that the IVI readout would be less likely to change in the shorter time interval by decreasing the time interval between START COMP and thrusting. We pushed START COMP approximately 2 minutes prior to burn, but we did not get a readout. The pilot reentered the velocity, pushed START COMP again and a ΔV of 96 appeared in the IVI window, which increased to 97 prior to burn. I translated down to zero using the aft-firing thrusters which required 2 minutes 27 seconds. I had no difficulty whatsoever in performing the OAMS retro maneuvers, and the procedures that we practiced on the Gemini mission simulator were perfect.

We conducted a reentry control system (RCS) plume observation test to determine the degree of occultation of the horizon which resulted when using the RCS thrusters during night conditions. I could see the horizon fairly well through the pitch thrusters, but the yaw thrusters completely obliterated the horizon. A night-side retrofire would be virtually impossible using the horizon or stars as a reference. The RCS thruster plume appears as a light, reddish-orange color with little sparks, and subtends an angle of approximately 25° out to a distance of 10 feet or more. The forward-firing thrusters subtend a slightly larger angle (30° or more) and appear to project to a distance of 35 or 40 feet. The forward-firing translation thruster plumes did not appear to impinge upon the spacecraft.

7.1.2.1.5 Retrofire: As mentioned previously, I was comparing the 8-ball with the horizon very closely during the third orbit to make sure that they were in good alinement for retrofire. We completed our preretro checklist on schedule and stowed all loose items. The preretrofire procedures were completed according to the checklist, exactly the way we performed them on the Gemini mission simulator. We double-checked most of the procedures and switch positions to be absolutely positive they were right. The SEP OAMS event was a medium loud report, SEP ELEC was somewhat quieter but very audible, and SEP ADAPT was very loud and is felt as a jolt. There was no doubt when the the adapter separation pyros fired; the firing caused a slight attitude disturbance. At T_R - 30 seconds, we armed AUTO RETRO and retrorocket squib no. 1. I waited 3 or 4 seconds to make sure it did not fire and then armed the other three retrorocket squibs.

The ground countdown to retrofire matched perfectly with the event timer and computer. The first retrorocket fired right at zero count. The ground also called each retrorocket as it fired by the actual crossfire order which is 1, 3, 2, 4. This gave momentary concern because I wondered what had happened to the number 2 retrorocket. I used the rate-command mode and both RCS rings for control. I used the 8-ball for attitude, with rates displayed on the flight director indicator (FDI), which was the method I had used during training. The retrofire disturbance was easier than those usually experienced on the Gemini mission simulator. The only significant disturbance I noticed was in yaw when the number 4 retro fired. I believe we used the right retrofire practice philosophy during training (i.e., a majority of more severe retrorocket disturbances than expected in flight interspersed with several relatively easy retrofire cases).

I maintained spacecraft attitude very close to nominal throughout the firing of the retrorockets. I do not believe pitch and yaw attitudes deviated as much as 1° from the nominal retroattitude. I held roll quite close to zero, although it has no direct effect upon the resultant ΔV. The IVI's after completion of retrofire displayed ΔV's of 331 aft, 105 right, and 4 up, which is actually 331 aft, 105 down, and 4 right. The 105 ΔV down was somewhat more than I had ever seen on the Gemini mission simulator.

Retroadapter jettison was quite audible and can definitely be felt. We again noticed quite a bit of debris around the spacecraft.

7.1.2.1.6 Reentry: After the retro section was jettisoned, I selected pulse mode. Attitudes were easy to control, and the system was working as I expected. Since reentry was one of the bigger unknowns, I elected to return to two-ring RCS DIRECT after only a brief check of the pulse mode.

I rolled to the 180° roll, maximum lift attitude, about 2 minutes after retrofire. At about 4 minutes 30 seconds after retrofire, I started receiving initial steering commands from the computer to hold maximum lift. Prior to retrofire, we had received backup reentry guidance quantities of time at 400 000 feet of 4 minutes 6 seconds after retrofire, reverse bank time of 11 minutes after retrofire, and bank angles of 55° left and 65° right. Just prior to communications blackout, we received updated bank angles of 45° left and 55° right, but the new reverse time did not get through.

I rolled to a left bank of 45° at 6 minutes after retrofire. Shortly after this the FDI needle went full-scale, indicating that we were short and to the north of the landing point. The crossrange error started to steer out almost immediately and went to zero about 10 minutes after retrofire. The initial downrange error was about 60 to 70 miles short. This decreased to about 30 to 40 miles momentarily and then returned to 50 to 60 miles and remained there until the computer stopped guiding at 80 000 feet.

I rolled to heads-down, or maximum lift, attitude at the time the crossrange error went to zero. The accelerometer was indicating about 1 g at this time. I was having some difficulty holding a constant bank angle. The spacecraft seemed to want to drift in roll. Each time I got engrossed in watching the fireball or other phenomena out the window, the bank angle would deviate slightly. The pitch and yaw rates never did exceed ±4° on our rate needles. This agreed very well with the out-the-window view and the rates as shown on the 8-ball.

The spacecraft was very stable during the reentry. The direct-control mode worked very well; control response was quite rapid and accurate. My flight director indicator (FDI) was displaying crossrange and downrange information, whereas the pilot's FDI was displaying rates. I flew instruments as my primary reference throughout reentry, occasionally cross-checking out the window. Therefore, I did not make many out-the-window observations. I did note that the pitch and yaw attitude is apparent by the material ablating from the spacecraft and that this in itself provides a fairly good spacecraft control reference.

The reentry rate oscillations were almost exactly like those that we had experienced on the Gemini mission simulator. The frequency increased and amplitude decreased as g increased and just the reverse as g decreased. The oscillation frequency seemed to be less than 1 cycle per second. I had no difficulty at all in damping the oscillations. I would check the pilot's FDI occasionally, or he would inform me that the rates were building up. Usually, one or two controller inputs would damp the pitch and/or yaw rates to zero.

The g-level did not exceed approximately 4g; however, it stayed at 4g for quite a long time. Although I held the spacecraft at the maximum lift attitude (±5° in order to keep crossrange zeroed) during most of the reentry, the initial uprange error indication (approximately half-scale up needle deflection) did not diminish. The computer still indicated that I was 50 to 60 miles short at drogue parachute deployment.

The reentries that we had flown on the Gemini mission simulator were not very realistic compared to what we experienced on the flight. We had flown several closed-loop reentries on the simulator but had always been able to alter our touchdown location significantly. Although the reentry was easy to control, it would have been beneficial to have had a good out-the-window display and to have practiced more reentry simulations in conjunction with the ground-support personnel.

We deployed the drogue parachute at approximately 50 000 feet and turned off the RCS propellant valves. The drogue parachute looked good and initially stabilized the spacecraft. However, the spacecraft oscillations began to increase, the pilot turned the RCS propellant valves back on, and I damped the spacecraft oscillations using the rate-command control mode. The propellant valves were turned off at 30 000 feet, and rate command was left on to deplete the fuel and oxidizer from the lines.

The 40 000-foot light illuminated very close to 40 000 feet on the altimeter. Prior to drogue deployment, the altimeter was very erratic and unreliable. At about 10 600 feet on the altimeter, the 10 600-foot light illuminated, and I deployed the main parachute. It remained in a reefed condition for approximately 6 seconds and then disreefed. There were no holes or torn panels, and it looked very good. The rate of descent was 30 ft/sec. After satisfying myself that I had a good main parachute, I went to landing attitude and got the biggest surprise of the entire flight: the spacecraft literally dropped from a vertical position to a 35° position! My head snapped forward with sufficient force to break my faceplate on the reticle mounting bracket. The pilot was also thrown forward, and he scratched his faceplate. I thought at first that we had lost our main parachute. My shoulder harness was not locked. We could not determine just when we would hit the water or at what attitude, but we were as ready for landing as we could be under these circumstances.

7.1.2.1.7 Postlanding: Landing was quite soft compared to single-point release. We remained submerged for several seconds until I jettisoned the parachute. As soon as we jettisoned the parachute, the spacecraft popped to the surface. Apparently, the wind caused the parachute to drag us in an almost completely submerged position. This did have an advantage because it positioned us blunt end into the wind, and consequently, the yellow smoke or stream from the thrusters was blown away from the spacecraft. There was considerable sputtering and yellow smoke from the thrusters. We closed the snorkel valve for a brief period because we noticed some grayish smoke and fumes in the cockpit, which we deduced as emanating from the hot shingles.

The spacecraft did not leak at all as far as we could tell. It floated nicely with very little right roll. Water was not washing over either window. We received word that the pararescuemen would be with us shortly. I did see one of them drop into the water some 20 yards in front of the nose, but I did not see the other pararescuemen until the collar was attached, and he walked past my window.

An earlier communication informed us that the USS Intrepid was only 5 miles from our position. Based on this information, we decided to remain in the spacecraft until ship pickup. Sometime later (approximately 20 minutes), we were informed that the Intrepid was 55 miles from our position; therefore, we could not expect spacecraft pickup for almost 2 hours. We were getting extremely uncomfortable in our suits; so we elected to take the suits off, egress, and be picked up by helicopter. I attempted unsuccessfully to open my hatch about the same time that one of the pararescuemen started to open it from the outside. I am not positive whether or not he had inserted his hatch-opening wrench prior to my attempt to open it from the inside; nevertheless, he did open it from the outside. I had previously removed the Gemini pressure suit with some difficulty; therefore, I put on my lifevest, got in one of the available three-man rafts, and was subsequently picked up by helicopter. The pilot did likewise.

7.1.2.2 Pilot's report.-

7.1.2.2.1 Countdown: The countdown proceeded smoothly, with the only delay being in relation to the engine fuel leak problem that was corrected after a short hold at T-35 minutes. All spacecraft systems checked satisfactorily, with indications similar to those experienced during the Wet Mock Simulated Launch. The opening of the launch vehicle prevalves provided the only unexperienced vibrations noticed in the count.

I could hear both the orbital attitude and maneuver system (OAMS) and reentry control system (RCS) isolation squib valves fire very faintly.

7.1.2.2.2 Powered flight: Engine ignition was normal, and lift-off was very smooth with no cue from acceleration. The first-stage engine noise and vibration at ignition were much less than I had expected or experienced in any simulations. After 3 seconds, I started the elapsed time indicator of the 8-day clock while Cape CAP COM was calling release and lift-off. The first few seconds after lift-off, I could sense vertical launch-vehicle motion but practically no airframe vibration. I could feel the roll and pitch program initiate by smooth launch-vehicle motions. The motions correlated perfectly with the operation of the inertial guidance system (IGS) attitude director indicator (ADI). Ambient noise increased until the vehicle reached supersonic speeds, at which time the ambient noise reduced somewhat; however, noise levels were never as loud as expected. The noise increased again as maximum dynamic pressure (max q) was reached and then reduced to approximately the levels experienced in the Gemini mission simulator for the remainder of powered flight.

At approximately 50 seconds after lift-off, the cabin pressure held momentarily at 5.8 psid for 1 or 2 seconds, after which it slowly increased to 6.7 psid. After SEP spacecraft, I noted that the cabin pressure was around 5.7 psid. The attitude error needle was indicating between 2° and 3° launch vehicle high in pitch, late in the first-stage operation. There was a vibration that I estimate at 20 cps of very low amplitude immediately before staging. This was the only vibration that I noticed that appeared to be mechanical. Aerodynamic buffet vibrations were not noted, and I do not believe there was any significant vibration associated with the launch. At staging, I was surprised at the separation noise and debris. The vacuum start of the second-stage engine produced a momentary yellow-orange flame around the spacecraft which also surprised me. Consideration of the event, however, made me realize that this is normal for fire-in-the-hole staging.

At BECO, the launch vehicle pitch IGS flight director indicator (FDI) attitude error went full-scale, down, on the low scale for approximately 15 seconds before we noted that the RGS was steering out the launch-vehicle high indication. After steering commenced, the launch vehicle pitched down. The horizon out the window was a beautiful sight. For the early part of second-stage flight, significant motion over the ground was not apparent. However, as the second stage began to accelerate, the velocity over the ground became very apparent. As we went to launch phase III, I was fascinated by the view out the window. I was, therefore, only periodically monitoring the IGS pitch-attitude error which indicated we were 6° high (full-scale saturated signal). It held this indication to SECO.

7.1.2.2.3 Orbital flight:

Insertion - At SECO, launch vehicle acceleration dropped rapidly to zero with no noticeable launch vehicle thrust tail-off. I felt no vertigo or disorientation of any sort. At this time, I selected direct attitude control and maneuver and attitude on OAMS power and waited for SECO plus 20 seconds to separate the spacecraft from the launch vehicle. I was waiting to hear the thrusters firing. As the command pilot depressed the maneuver handle, I could just barely hear them. After the maneuver handle was operated for 2 seconds, I separated the spacecraft, and it pitched forward with a noticeable acceleration. The aft-thruster noise is a very poor cue for spacecraft separation. I was a little slow getting to rate command because I was watching an unusual display of white flakes which shot ahead of the spacecraft at separation. MCC said that we were GO and gave us our orbital elements. I did not note V_{gp} or T_{ap}. After separation, the horizon-scanner fairings were jettisoned. I could see them go out laterally and to the rear of the spacecraft through the command pilot's window. I did not see the antenna fairing jettison. The complete insertion checklist was not accomplished at this time. The D-rings were stowed right away and so were the ejection seat drogue parachute pins. The drogue safety pin stowage which requires us to turn completely around in the seat is very easy to do in zero g. I also stowed the inboard arm restraint, but did not stow the outboard one which I later used to hold equipment. I shut off the right-hand secondary O_2 bottle, changed the waste valve to normal, and left the water valve in pressure off. I did not test the main batteries until we were past Kano. Prior to the Grand Canary Islands, I began to unstow the blood-pressure bulb, the Hasselblad camera, the 16-mm sequence camera, and the film magazines.

<u>Systems operations</u> - On the first pass over Grand Canary, we lost the primary dc-to-dc converter. The first indication that I saw was a reading of 250 psia on the O_2 pressure gage. Initially, I read this to be a possible contingency. Preflight briefings had indicated we might have a cryogenic pressure drop, so I went to the manual oxygen heater. I immediately noticed that in addition neither the cabin pressure nor other consumables gages were registering, which indicated an obvious primary dc-to-dc converter failure. Selection of secondary dc-to-dc converter brought all instrumentation back on the line. I rechecked the primary converter, and it was inoperative; therefore, we remained on the secondary converter for the rest of the flight.

The environmental control system (ECS) operation was satisfactory during the flight. The suit temperature was fairly comfortable, although the suit oxygen flow was marginal during work-load periods. Suit inlet temperature was 54° to 55° on the night side and 58° to 59° on the day side. Cabin temperature varied from about 90° to 93°, which is too warm with sunlight in the cabin. On the night side, I was fairly comfortable when I opened the suit over Carnarvon for a waste evaluation.

During the first orbit over the Canaries with the radiator in flow position, we were advised the outlet temperature was too high, so I went back to radiator bypass. Later, over CSQ, we had a radiator outlet temperature of 42° F and left it in the flow position.

The oxygen high-rate flow check was accomplished. Oxygen high rate flow shuts off the suit compressors, and its temperature is the temperature of the primary oxygen. The O_2 high-rate actual suit gas-flow rate is very low. Therefore, we were not as cool during O_2 high rate flow as in normal oxygen flow. We recommended that O_2 high rate be used only for maintaining a positive differential pressure during cabin purge, for cleaning out CO_2 from the suit, and for conducting suit integrity checks.

The best mode of operation for the ECS in zero-g is with faceplates open and the recirculating valve open. With the faceplates open in zero-g, there is very little noticeable decrease in suit pressure because the suit balloons out from the body, and, therefore, there is no noticeable decrease in cooling with the faceplates open. The need for single- or double-coolant loop operation can be determined by crew comfort and suit-inlet temperature, assisted by recommendations furnished from the ground.

The quality of communications during the flight was most satisfactory on UHF. We received no readable communications over HF after we landed. The main procedural error which I repeatedly committed during the flight was attempting to transmit to the ground while still in the record position for the onboard voice tape recorder. The design of the hand-held push-to-talk switch has always caused difficulty in determining proper operation for transmissions.

The control mode characteristic check of the pulse mode and the rate command mode duplicated the design characteristics which we had used in the Cape mission simulator. The rate command mode is very tight and easy to operate with exceptionally fine "flying" qualities.

I started the food and waste evaluation just past the Canaries on the second orbit and continued until we were past Carnarvon. I reconstituted some applesauce and grapefruit juice, and opened a package of chicken bites. The chicken bites were not very tasty and were rather difficult to get out of the package while wearing pressure-suit gloves. The applesauce and grapefruit juice reconstituted rapidly with no leakage from the water valve. Eating the applesauce and drinking the juice were normal in all respects. I noted that under zero-g, juice droplets crept out of the food port in the food package even though it was folded. I recommend that food ports be taped closed to prevent food and juice leakage into the spacecraft. Also, the crew should attempt to eat as much of every meal as possible to prevent possible putrefaction and to minimize the stowage problems.

The waste-disposal system evaluation was completed. I believe that some of the problems of waste disposal in zero-g will have to be endured in Gemini. I did not have enough time to evaluate all aspects of the system properly. Adequate time in future flights must be allowed for meal preparation, eating, and waste disposal, in order to keep the crew healthy in their semi-closed environmental control system. The drinking water facility was satisfactory. We drank water, reconstituted food, and operated within the pressure-off drinking valve configuration during the flight. The blood-pressure bulb was used to pressurize the drinking tank after we were in the Atlantic Ocean in the spacecraft. It proved adequate for water pressurization.

I gave blood pressure readings throughout the flight since the command pilot was unable to install his blood-pressure bulb plug into the suit blood-pressure fitting. The respiratory maneuvers and oral-temperature measurements were accomplished as planned for evaluation by ground medical monitors.

Visual sightings - I was surprised during launch by the lights that came up around the spacecraft when the second stage ignited. During the latter portion of the launch phase, I was absorbed by the out-the-window view of the earth. At spacecraft separation, I noticed what appeared to be white flakes fly past us and there were many pieces of small debris floating around. I saw the horizon-scanner fairings through the pilot's window go out laterally from the spacecraft as they were jettisoned at insertion.

During the first pass, I took 16-mm pictures of the RCS plumes of ring B. During the second orbit, I photographed the forward-firing thrusters during the translational systems check. The plume of the forward-firing maneuver thruster is a large yellow-orange glow with a lot of sparks and extends for 30 or 40 feet at about a 34° or 40° subtended angle. It is a very spectacular view at night. During the first night pass, the first stars that I recognized were in the Southern Cross and Alpha and Beta Centauri. On the last night pass, I picked out the northern sky constellations very well. Taurus, Auriga, Orion, and the Pleiades appeared the same as they appeared from an airplane at 40 000 feet.

Most of the southern part of the United States was covered by clouds. We did track a town inland from the northern tip of the Gulf of California. Photographic identification later showed this town to be Mexicali. Some street divisions were evident.

The out-the-window view of the reentry was exactly like the color pictures of the OT-2 reentry. I could not detect that the reentry communication experiment noticeably changed the colors. Throughout reentry, I could see the horizon clearly.

Experiments - I activated the human-blood irradiation experiment over Carnarvon at 50 minutes, approximately 30 or 40 seconds elapsed time. The time called to Carnarvon was 50 minutes 18 seconds. The experiment was deactivated after 20 minutes. The clearance between the hatch and the experiment was much smaller than on the mission simulator, causing some difficulty in operating the experiment.

I activated the reentry communications experiment using Greenwich mean time called up after retrofire. I held the push-to-talk switch for 4 minutes from the activation time.

7.1.2.2.4 Retrofire: I received and inserted our OAMS retroburn information into the computer 12 minutes prior to retrofire, and the command pilot started his retroburn for a velocity change of 96 ft/sec with a burn time of 1 minute 49 seconds. We both completed the T_R - 5 checklist at least five times. I checked the main batteries and got the adapter batteries off the line. The command pilot checked the RCS A- and B-ring thrusters. The T_R -1 checklist was then completed. The OAMS lines separated with a rather loud click. The electronics separated with a softer click as the little guillotines fired. However, adapter separation was a loud bang, and I felt a little retrograde motion. It felt as if it really kicked off. After adapter separation, an object flew out laterally from the spacecraft. I could see the object through the command pilot's window. It looked like the pump package, but it could have been the thermal curtain rolled up or some other similar silver adapter package. I armed autoretro at T_R - 30, and the command pilot turned on the retrorocket squibs. We checked computer T_R at 22 seconds against the event timer to validate auto retrofire time. Retrofire gave us a $\frac{1}{2}$g to 1 g retrograde boot; I pushed the manual retrofire at T_R + 2. The command pilot's retrofire control was excellent. The attitude stayed right in the center of the circle on the IGS ADI, within 1°. The last retrorocket had a noticeable yaw effect similar to, but of less magnitude than, the yaw torque of the nominal retrofire pattern on the Cape simulator during training. The retrojettison and post-retrojettison checklists were checked and rechecked.

We had considerable discussion with the ground at this time on our IVI readings, backup reentry bank angles, reverse bank-angle times, and reentry-experiment activation times. I wrote this information on the spacecraft walls, having no other convenient place to write. We must have an accessible plate at the pilots' fingertips on which to record this essential information. It is recommended that all times of events after retrofire be based on elapsed time from retrofire instead of

G.m.t., because both the event timer and the elapsed time clock are counting up from retrofire. We did not receive the last backup reverse bank-angle time before we entered blackout.

7.1.2.2.5 Reentry: The reentry was exactly like the technicolor pictures out the window of GT-2. The gases are the same color gases, and the pattern had the same spiral flow. The spacecraft appeared to be dynamically very stable. Spacecraft oscillations on the IGS FDI correlated exactly with spacecraft motions out-the-window. The 45° bank angle never varied during my cross-checks between the window and the IGS FDI until we went to maximum lift. The fine particles coming off the heat shield in the vicinity of the right adapter interconnect fairing were going up over the rendezvous and recovery (R and R) section as they did during GT-2. They were coming up almost vertically from the heat shield as if our trim angle were reduced. The ionized particles were dimly persistent, and I could see our reentry path for a long way behind the spacecraft.

The spacecraft pitch and yaw rates were exactly the same rates that we had seen during our training on the Cape mission simulator. The rates built up somewhat just before reentry g increased. Then as we started up the reentry g profile, the rate frequency went up but its amplitude came down. The rates were never out of the realm of control. When the rates built up to 4 deg/sec, the command pilot immediately damped them. I do not think we need to damp these rates anyway. Subjectively, I did not think that we had over 4.5g (5 maximum) during the reentry profile.

During the reentry, I could see the retroadapter following us, and we watched it burn up as it entered the atmosphere. When the plasma nearly dissipated, the outer window pane got a series of wrinkled etched-like lines over its surface; however, they did not excessively degrade visibility during reentry. After we landed, I could no longer see them.

It seemed to me that we held reentry g for a long time, even after we were pitching fairly well vertically. At 50 000 feet, I was watching the differential cabin pressure indicator, altimeter, and the absolute pressure gage. I shut off the RCS propellant valves as the command pilot deployed the drogue parachute. The spacecraft oscillations increased at this, so I reopened the RCS propellant valves and the rate-command control mode damped the oscillation immediately. I again shut off the RCS propellant valves at approximately 30 000 feet to open the inlet snorkel and the cabin vent at 28 000 feet. At 10 600 feet, the command pilot deployed the main parachute which came right out and disreefed in about 10 seconds. We waited at least another 10 seconds, and then the command pilot went to landing attitude, and the spacecraft experienced a severe jolt. It pitched both of us into the windshield.

Even though my shoulder harness was locked, I do not think it was very tight because I hit my visor on the windshield. After going to the landing attitude, I spent the rest of the descent going through the post-main checklist.

7.1.2.2.6 Landing and recovery: As soon as we hit the water, we were dragged by the parachute. It felt like the spacecraft was trying to nose into the water. The command pilot jettisoned the parachute when he saw that we had water over both windows. The impact of the water landing was not nearly as severe as the jolt of the single-point release. After we landed, we were in contact with the ship on UHF radio. We received conflicting reports on our distance from the USS Intrepid. At first, we were advised that we were 5 miles away and then 55 miles away. This led to conflicting times as to how soon we would be recovered. It is essential to get out of the pressure suits as soon as you land if there is to be any delay in recovery. Further, it is important to position all spacecraft controls, switches, and safety pins as soon as possible because I noted that seasickness reduces crew efficiency.

We went to O_2 high rate in the water and closed our visors for a short time because of what appeared to be oxidizer fumes leaking from the RCS. There was also an odor of burnt metal. There was no evidence of leaks or any water in the ECS well.

The spacecraft in the water did not appear to roll as far to the right as static article 5 did in the Gulf of Mexico.

7.2 AEROMEDICAL

The prime aeromedical objectives of this flight were to provide two additional data points in the store of scientific information on man's response to space flight and to give additional assurance that man can be committed to longer duration missions aboard the Gemini spacecraft. Experience gained during Project Mercury indicated few physiological changes in a flight of this duration, and few were found. The programed medical measurements (with one exception) were performed, and high quality, reliable data were obtained. These data are presented chronologically by phase of flight; that is, preflight, inflight, and postflight. The preflight section contains background data on the crew and carries these observations through the prelaunch phase to lift-off. The inflight section contains data which were received by telemetry at MCC and the Gemini network tracking sites. This information is supplemented by the record from the onboard biomedical tape recorder. The postflight section includes recovery information obtained during the postflight medical examinations and debriefings. Physiological findings during all phases of this flight were as expected. The tilt-table studies, designed to determine subclinical levels of physiological deconditioning, revealed no significant changes. Other matters of aeromedical concern were the problems encountered during evaluation of the food, water, and waste systems, as well as the general housekeeping. These are discussed as part of the postflight debriefing. The specific system analyses are found in section 5.1.10. An analysis of data from this flight indicates that there are no medical contraindications for the GT-4 mission as planned.

7.2.1 Preflight

7.2.1.1 Medical histories.- The background medical information on the flight crew includes extensive information collected since their selection as astronauts and was augmented by data collected during various tests specific to GT-3. A summary of the aeromedical evaluations performed on the GT-3 crew is listed in table 7-VI. During spacecraft systems tests, centrifuge tests, and flight simulations, the crewmen were required to wear Gemini space suits and be fully sensored for biomedical measurements. The bioinstrumentation system consisted of two leads of electrocardiogram, impedance pneumogram, oral temperature (using an oral thermistor), and a blood-pressure measuring system. These systems were carefully calibrated prior to use. The blood-pressure measuring system (BPMS) was calibrated prior to suiting by comparing a simultaneous clinical blood-pressure reading with that observed from the BPMS. During all major tests involving a practice crew countdown, as well as on launch morning, biomedical data were collected for approximately 1 hour prior to crew ingress into the spacecraft. This served the dual purpose of verifying the performance of the bioinstrumentation and collecting prelaunch medical data.

During the final 2-week period prior to launch, three tilt-table studies were conducted on both crewmen to serve as baseline data for postflight tilt studies conducted aboard the recovery carrier. For these studies, a modified Stokes litter was used (ref. 8). During this procedure, the electrocardiogram and impedance pneumogram were recorded continuously, and blood pressures were taken periodically. Both crewmen tolerated the preflight tilt procedure well, and no abnormal findings were noted. These data were presented for both preflight and postflight examinations in figures 7-4 and 7-5.

7.2.1.2 Preflight activities.- The preflight activity of the crew was not under strict medical supervision, and no medical quarantine was imposed.

7.2.1.2.1 Diet: For this flight it was deemed undesirable to limit the crewmen's preflight diet to low residue foods because one of the planned objectives of the mission was to evaluate the waste disposal system which has been developed for missions of longer duration. The preflight diet was a normal one and tended to be high in protein and moderate in carbohydrate content.

7.2.1.2.2 Drug evaluation: Three weeks prior to flight, both prime and backup crewmen were tested for adverse effects or sensitivity to each medication which is included in the onboard medical kit. All drugs tested were taken orally, and no adverse effects were noted.

7.2.1.3 Preflight medical examinations.- Comprehensive medical evaluations were conducted on the backup and prime crewmen on F-10 and F-2 days in order to determine the preflight baseline and their fitness for flight. The F-2 day physical examination was conducted by a team of medical specialists including a radiologist, neuropsychiatrist, dentist, opthalmologist, otolaryngologist, internist-cardiologist, and flight surgeon. The results of these evaluations are recorded as a permanent entry in the individual's medical record. The pilot of the prime flight crew exhibited an elevation in systolic and diastolic blood pressure which had been noted on previous examinations. This is not considered to be clinically significant at this time and in no way disqualified him for this flight. There were no other significant findings on either the prime or backup crew. A brief preflight physical was conducted by NASA flight surgeons on launch morning and the crew were considered ready for flight.

7.2.1.4 Prelaunch medical data.- Preflight aeromedical procedures and examinations are listed in tables 7-VII and 7-VIII. Hematology studies are included in tables 7-IX and 7-X. Table 7-XI includes the preflight urine studies. At no time during any prelaunch test were abnormalities noted in the electrocardiogram of either crewman. An increased respiration rate from 18 to 20 breaths/min to 25 to 30 breaths/min was noted on the pilot whenever he was in the spacecraft. This phenomenon was present during all spacecraft tests. On launch morning the pilot's respiratory rate response was in no way different from these previous tests. No abnormal blood-pressure measurements, except as noted in section 7.2.1.3, were recorded during any of the preflight tests, physical examinations, or on launch morning. Oral temperatures were normal at all times during these preflight and launch day activities. The heart rate was elevated in both crewmen approximately 2 minutes prior to lift-off. These data, included in figures 7-6 and 7-7, are considered in no way unexpected or abnormal.

7.2.2 Inflight

The inflight portion of the aeromedical report begins with lift-off and ends with spacecraft landing, an elapsed time of 4 hours 52 minutes 31 seconds.

7.2.2.1 Physiological measurements.- Physiological measurements obtained from the Gemini bioinstrumentation system, which is described in section 7.2.1.1, as well as certain environmental parameters, were monitored by physicians at the Mission Control Center (MCC) and at various remote (network tracking) sites. Analog biomedical data from Bermuda (BDA), the Canary Islands (CYI), Carnarvon (CRO), Hawaii (HAW), and Corpus Christi (TEX) were also transmitted to MCC by means of voice-data lines. This served the purpose of allowing the MCC surgeon to observe, in real time, the same biomedical data which were available to the remote-site surgeon. The quality of these analog data was satisfactory for clinical analysis. The electrocardiograms and pneumogram from each crewman were recorded on an onboard biomedical tape recorder. The inflight physiological measurements are shown in figures 7-6 and 7-7.

7.2.2.1.1 Electrocardiograms: There were no abnormalities of rate, rhythm, or pattern seen during review of the electrocardiogram (ECG's) recorded on the onboard biomedical tape recorder. Increases in the heart rate of both crew members were present as expected during dynamic portions of flight. There were also periods of increased rate associated with various crew activities such as the food and waste evaluation and control of the spacecraft during reentry.

7.2.2.1.2 Respiration: The respiratory rates, as measured by the impedence pneumograph, were within the expected range of normals for this crew. A respiratory maneuver, which consisted of three consecutive deep breaths, afforded a satisfactory means for correlating time on the spacecraft biomedical tape recorder.

7.2.2.1.3 Blood pressure: Three blood-pressure measurements on each crewman were scheduled in the flight plan. However, none was obtained on the command pilot. Although no mechanical failure has been found in the system, he was unable, in the time allotted, to mate the blood-pressure bulb properly to the fitting on his pressure suit (see section 5.1.10.7.5). The pilot was able to mate this bulb to his suit, and three inflight blood-pressure measurements were received as planned. The pilot's blood pressure, as measured from data transmitted to the ground shortly after orbital insertion, was 175/125. At a peak cuff pressure of 175 mm Hg, the Korotkoff sounds were still present. Thus, the systolic pressure may be even higher than 175 mm Hg, and this value may not be reliable. A valid blood-pressure value (146/95) was recorded at 2 hours 25 minutes on the second revolution over Australia, and another (126/90) was recorded at 2 hours 57 minutes over the tracking ship (RKV) in the Pacific Ocean. It is apparent that increased systolic and diastolic pressures were observed in the early phases of orbital flight with a gradual decrease as the flight progressed. There were, however, too few measurements to draw any significant conclusions.

7.2.2.1.4 Oral temperature: The oral temperature of both crew members was measured at specific intervals which were programed on the flight plan, and are recorded in figures 7-6 and 7-7. The inflight values correlate well with the normal preflight levels.

7.2.2.2 Medical observations.- At no time during the flight did the crew have need of medical care or advice. The environmental control system functioned as planned, and the crew remained quite comfortable. There were no abnormal sensations such as vertigo, disorientation, dizziness, pain, fatigue, elation, somnolence, nausea, hunger, or thirst reported during any portion of the flight.

7.2.3 Postflight

Postflight medical information was gathered on the crew from the time of spacecraft landing until approximately 48 hours thereafter. These data were obtained primarily by clinical and laboratory examinations, although a single lead of electrocardiogram was recorded during the postflight tilt studies. Postflight deviations from the normal were limited to mild dehydration of both crew members and minimal, asymptomatic decrease in pulse pressure with an elevation of heart rate during the initial postflight tilt studies. These findings will be discussed in the following paragraphs.

7.2.3.1 Recovery activities.- Medical recovery activities were planned before the mission, and no difficulty was encountered in adhering to the basic plan.

7.2.3.1.1 Planned recovery activities: In areas of the greatest probability of landing, a surgical team composed of a surgeon, anesthesiologist, and surgical technician was placed aboard the recovery vessel. In areas of lesser probability for recovery, a surgeon was placed on one vessel and an anesthesiologist was placed on the next vessel. In this manner, these two physicians could easily be brought together if needed to provide medical care. In the prime landing area, in addition to the above coverage, a flight surgeon from the Center Medical Office of the Manned Spacecraft Center was positioned to coordinate the medical debriefing and other postflight medical activities. The general examination schedule for handling the flight crew was outlined in advance. Initially, emphasis was placed upon obtaining a blood specimen and starting the operational tilt procedure as soon as possible. A total of 3 hours was set aside for the initial medical evaluation. Another blood specimen was to have been obtained 6 hours after the first sample, and a repeat of the tilt studies was scheduled at that same time. A brief (45 min) medical debriefing of the crew was to be conducted the morning following the flight. A final evaluation by the Medical Specialty Team who had examined the crew 2 days before the flight was scheduled as soon as possible after the crew returned to the launch site on the second day after the flight.

7.2.3.1.2 Postflight medical activities: The postflight medical activities of the crew are outlined in table 7-XII. Initially, the crew elected to remain in the spacecraft pending pickup by the recovery aircraft carrier. Since the carrier retrieval was delayed, the crew elected to doff their pressure suits and leave the spacecraft. They were recovered by a helicopter and flown to the aircraft carrier. Shortly before doffing the suits, both crewmen became nauseated and the command pilot vomited a moderate quantity of almost clear, greenish fluid. His nausea was rapid in onset and was almost entirely relieved by vomiting. No antimotion sickness medication was used as no nausea was experienced before landing. It is felt that this malady is attributable to the motion of the spacecraft on the water and was probably enhanced by the warm environmental conditions in the spacecraft after landing. Both crewmen reported subjective discomfort as a result of the uncomfortably warm temperatures prior to doffing the pressure suits. There were no subjective symptoms suggestive of hypotension at any time during the recovery phase of the mission. A physician aboard the retrieving helicopter reported that a brief and superficial examination revealed no medical abnormalities in the crew.

7.2.3.2 Examinations.- Initial detailed examination was accomplished in the ship's sick bay as soon as the crew came aboard the recovery carrier. After a blood specimen for laboratory examination and the S-4 experiment was obtained and the tilt studies were completed, a general examination was conducted and included the following components: vital signs, eyes, nose, throat, ears including audiogram, neck, thorax, spine, neurological examination including position sense, lungs, heart, abdomen, extremities with maximum measurements, PA and lateral chest X-rays, electrocardiogram, complete blood count, and urinalysis. With the exception of the mild dehydration manifested by subjective thirst and the blood and urine findings, no significant abnormalities were noted during this examination which is summarized in tables 7-IX to 7-XI.

7.2.3.3 Tilt studies.- An operational tilt procedure similar to that used after the Mercury-Atlas 9 mission was employed again on GT-3. (See ref. 8.) A Stokes litter was tilted to 70° head-up position after a 5-minute period for heart rate and blood pressure stabilization. The tilt phase of the procedure lasted 15 minutes, and no flack test was performed. Thereafter, the litter was returned to the horizontal position, and restabilization occurred within 5 minutes. Continuous ECG and respiratory recordings, as well as intermittent blood-pressure measurements, were obtained during the three tilt studies accomplished at the launch site before the mission. Two tilt studies were performed on each crewman following the mission with continuous ECG monitoring. The initial postflight tilts on both crewmen revealed an elevation in heart rate and decrease in pulse pressure as shown in figures 7-6 and 7-7. The second postflight tilts, however, were essentially the same as preflight observations. No other ECG changes were present on either subject. The variables in tilt studies were controlled as much as possible. However, operational requirements necessitated the performance of the preflight tilt studies in different places and by different observers. Likewise, the postflight tilt studies were done in the relatively unfamiliar environment of the ship's sick bay, where the motion of the ship was mild, but some extraneous distractions could not be excluded. Although these variables tend to discredit the significance of the postflight tilts, such a response was not expected after a mission of such a short duration. The significance of these findings is not known at this time and must await further evaluation.

7.2.3.4 Medical debriefing.- Medical debriefing of the crew was initially conducted with each crewman individually on the morning after the mission. Additional medical debriefing was conducted simultaneously with both crew members shortly after return to the launch site on the second day after the mission. Although these debriefings were extensive and detailed, only the information considered pertinent to this evaluation is presented here.

7.2.3.4.1 Premission comments: Before the mission, both crewmen ate a normally balanced diet and maintained a satisfactory level of muscular tone by regular exercises. These consisted of running, working in the gym, and performing calisthenic exercises. They slept soundly for $6\frac{1}{2}$ hours on the night before the mission, ate a good breakfast of steak and eggs, and felt no discomfort or undue tension at any time prior to launch.

7.2.3.4.2 Powered flight: The noise, acceleration, and vibrations of powered flight were the same or less than anticipated and were no problem for the crew.

7.2.3.4.3 Weightlessness: At no time did either crewman experience disorientation, nausea, vertigo, hunger, breakoff phenomenon, somnolence, nor any untoward physiological or psychological reaction.

7.2.3.4.4 Vision: Vision and color perception were normal, although specific ground sightings were limited by cloud cover.

7.2.3.4.5 Waste evaluation: The pilot performed scheduled food and waste evaluations during the orbital phase. No evidence of any remarkable physiological differences attributable to space flight were observed during urination or defecation. There was some difficulty in completing the food and waste evaluation in the limited time available. Further mechanical difficulty due to weightlessness was experienced in collecting the stool in the defecation bag. There was some difficulty in fit and comfort of the launch urine bag, and mechanical difficulty in operating the uriceptícle and urine dump system. System analyses of these items are included in section 5.1.10.

7.2.3.4.6 Food evaluation: Samples of the various types of in-flight foods were carried on this flight for evaluation purposes. Although there were no physiological differences during eating and drinking which can be related to space flight, some mechanical difficulties in preparing food and resealing containers were experienced.

7.2.3.4.7 Reentry: The g forces of retrofire caused no difficulty, and reentry was subjectively similar to the acceleration profile previously experienced on the centrifuge, although the actual g forces were considerably less in the actual mission. After main parachute deployment and when bridling occurred and the spacecraft assumed the 45° pitch-up landing attitude, both crew members were thrown forward by the sudden pitch of the spacecraft. Their helmet visors impacted against the lower rim of their respective spacecraft windows. The pilot's visor was scratched and the command pilot's visor was ruptured by a sighting reticle positioned on the lower edge of the left window rim. Neither pilot was injured by this incident. The remainder of the landing sequence was normal, and no difficulty was experienced as a result of return to the earth's gravity. Significant parts of the medical debriefing concerned with the postlanding phase of the mission are presented in section 7.2.3.1.2.

TABLE 7-I.- FLIGHT CREW SEQUENCE OF EVENTS

Elapsed time, hr:min (a)	Action
00:00	Lift-off
00:06	Separation from GLV and insertion Started insertion checklist
00:10	Completed insertion checklist (except for main battery check) Started platform alinement (used caging technique)
00:12	Completed platform alinement
00:13	First voice report of yaw drift problem
00:15	Plotboard unstowed Switched to UHF no. 2
00:16	Switched the radiator to flow
00:18	Blood pressure (pilot) Returned the radiator switch to bypass
00:20	Recovered g.e.t. time mark to set elapse time watch Activated sea urchin egg experiment
00:21	Control mode check complete
00:24	Noted loss of primary dc-to-dc converter and switched to secondary
00:30	Pilot removed and stowed launch day urine bag
00:44	Radiator switched to flow
00:47	Respiratory maneuver (command pilot)
00:48	Completion of oral temperature (pilot)

[a] Time to nearest minute

TABLE 7-I.- FLIGHT CREW SEQUENCE OF EVENTS - Continued

Elapsed time, hr:min (a)	Action
00:50	Started human blood irradiation experiment
00:51	Received GO from ground for second orbit Received 2-1 DCS preretro command load Received reentry quantities and preretro update quantities by voice
00:52	Checked T_r update on time mark from ground
00:55	Switched the secondary coolant loop OFF and the evaporator to NORMAL
00:59	Started the RCS plume evaluation
01:03	Main battery checked
01:06	Closed both faceplates for ECS check 1
01:07	Started the catch-up mode check
01:10	Deactivated the human blood irradiation experiment
01:12	Completed the catch-up mode check
01:14	Activated O_2 high rate using manual handle (Attempt to lower primary O_2 pressure)
01:15	Started platform alinement Actuated sea urchin egg experiment Recocked O_2 high rate
01:24	Recovered DCS maneuver update for translation no. 2
01:33	Started translation no. 1
01:34	Completed translation no. 1 (duration of translation controlled in time, Δt = 75 sec)

[a] Time to nearest minute

TABLE 7-I.- FLIGHT CREW SEQUENCE OF EVENTS - Continued

Elapsed time, hr:min (a)	Action
01:35	Initiated manually controlled dump of delayed-time tape recorder data
01:37	Recovered 3-1 DCS preretro command load
01:40	Received update time for horizon scanner check Blood pressure (pilot)
01:41	Tape dump complete
01:46	Start of platform alinement and caging check (command pilot) Start of food and waste evaluation (pilot)
01:49	Command pilot attempted to obtain a blood pressure
01:54	Completion of platform alinement and caging check Spacecraft yaw 180° in preparation for horizon scanner check
01:58	Onboard report that 8-ball and horizon did not agree. (First indication of platform orbit rate problem)
02:06	Addition report of horizon scanner and/or platform problem
02:08	Horizon scanner check started
02:16	Report of 8-ball and horizon disagreement
02:17	Translation system check
02:24	Started platform alinement

[a] Time to nearest minute

TABLE 7-I.- FLIGHT CREW SEQUENCE OF EVENTS - Continued

Elapsed time, hr:min (a)	Action
02:30	Horizon scan control mode characteristics check deleted (command pilot evaluating platform problem and assisting with the food and waste evaluation)
02:35	Continued with platform alinement (stars used for attitude reference)
02:41	Start of O_2 high rate check
02:43	Main batteries on
02:45	Manual ECS O_2 heaters on
02:46	O_2 high rate recocked
	Manual ECS O_2 heaters - OFF
02:52	Gage correlation report started
02:55	Platform SEF mode checked and reported operating correctly
02:58	Gage correlation report completed
03:02	Start of tracking task — tracked Mexicali, Mexico
03:02	Coolant pump checks complete
03:06	Standby telemetry transmitter used to transmit real-time telemetry data
03:11	Checked platform orbit rate mode
03:13	Cabin fan switched on
03:15	Cabin fan switched off
03:19	Started platform stabilization check

[a] Time to nearest minute

TABLE 7-I.- FLIGHT CREW SEQUENCE OF EVENTS - Continued

Elapsed time, hr:min (a)	Action
03:24	Completed platform stabilization check and started control mode characteristics check
03:24	Completed control mode characteristics check
03:51	Activated sea urchin egg experiment
03:56	Preretro checklist complete
03:59	Started event timer counting down to retrofire
04:00	Switched computer to catch-up
04:12	Main battery check
04:21	Started OAMS retro translation (96 ft/sec)
04:23	Completed OAMS retro translation
04:25	Computer switched to reentry mode
04:29	T_r-5 min checklist complete
04:32	R_r-1 min checklist complete
04:33	Auto retrofire
04:33	Manual retrofire pushed
04:37	400 000 ft calculated by computer
04:41	Down and cross range errors displayed by the computer computer
04:41	Reentry communications experiment started
04:46	Reentry communications experiment completed

[a] Time to nearest minute

TABLE 7-I.- FLIGHT CREW SEQUENCE OF EVENTS - Concluded

Elapsed time, hr:min (a)	Action
04:47	Drogue parachute deployed
04:46	Main parachute deployed
04:49	Post main checklist
04:53	Landing

[a] Time to nearest minute

TABLE 7-II.- SEQUENCE OF CONTROL EVENTS DURING REENTRY

Time, hr:min:sec g.e.t.	Event
04:34:08	Retrojettison (The spacecraft attitude at this time was approximately 0° roll, -16° pitch, 180° yaw)
Unknown	Spacecraft rolled to 180° (0° bank)
04:37:36	400 000-ft altitude (no spacecraft action)
04:39:29	Spacecraft rolled to a 45° bank (45° roll left)
04:43:51	Crossrange FDI indication nulled, spacecraft rolled to a 0° bank (180° roll, full lift)
04:48:24	Drogue parachute deployed

TABLE 7-III.- SPACECRAFT TEST PARTICIPATION (COCKPIT HOURS)

Test	Location	Crew	Time, hrs:min
Systems assurance	St. Louis	Prime	3:00
	St. Louis	Backup	3:00
Simulated flight	St. Louis	Prime	2:00
Altitude chamber	St. Louis	Prime	9:30
	St. Louis	Backup	5:00
Premate simulated flight	Cape Kennedy	Prime	7:30
Joint combined systems	Cape Kennedy	Prime	6:00
	Cape Kennedy	Backup	2:00
Pilot ingress dry run	Cape Kennedy	Backup	1:30
Flight configuration mode	Cape Kennedy	Backup	1:15
Wet mock simulated flight	Cape Kennedy	Prime	2:45
Final simulated flight	Cape Kennedy	Prime	2:30
	Cape Kennedy	Backup	4:00
Countdown	Cape Kennedy	Backup	3:30

TABLE 7-IV.- SUMMARY OF MAJOR GT-5 CREW PREFLIGHT TRAINING ACTIVITIES

Training Activity	Crew Member[a]											
	1-1			1-2			1-3			1-4		
	Hours	Sessions	Runs	Hours	Sessions	Runs	Hours	Sessions	Runs	Hours	Sessions	Runs
Spacecraft testing (cockpit)	55	11	—	55	11	—	22	12	—	22	12	
Gemini mission simulator	113.5	59		121.5	62		77	39		88	44	
Parachute training			7			7			9			7
Egress exercises	16	4		16	4		12	3		12	3	
Launch abort training (LTV)	19		225	15		154	15		152	14		142
Centrifuge training (AMAL)	9.5	3	15	11	3	14	5	2	8	8.5	3	15
Planetarium (Morehead)	16	4		16	4		16	4		16	4	
Space flight readiness flying		Approximately 25 hours per month each crew member through training period										
Briefing		Over 200 hours per crew member				—	—	—	—	—		

[a] 1-1 Command pilot prime crew
1-2 Pilot prime crew
1-3 Command pilot backup crew
1-4 Pilot backup crew

[b] Both command pilot and pilot received credit for each flight phase and each systems failure.

TABLE 7-V.- SUMMARY OF GEMINI MISSION SIMULATOR TRAINING

Training Activity	Crew Member[a]			
	1-1	1-2	1-3	1-4
	Hours (to nearest half hour)			
General familiarization (St. Louis)	36	36	34	34
Specific mission training (Cape Kennedy)	77.5	85.5	43	54
Total hours	113.5	121.5	77	88
Suited hours – portion of total hours	38.5	44.0	15	18
	Number of flight phases practices[b]			
<u>Launch</u>				
Normal	20	21	11	12
Mode I	7	7	3	4
Mode II	32	35	29	37
Mode III	7	9	3	5
<u>Insertion</u>				
Normal	13	14	10	10
Overspeed	5	5	4	4
Underspeed	4	4	2	3
<u>Orbit</u>				
Platform alinement	8	8	9	9
Flight plan practice	9	9	5	5
<u>Retrograde (manual control)</u>				
Retrofire	107	131	48	112
Reentry	64	71	35	43
	Number of systems failures[b]			
Booster	51	53	23	28
Sequential	57	58	24	26
Electrical and communications	34	35	30	37
ACME	17	15	16	17
OAMS	30	33	21	26
RCS	16	16	9	10
G and N	36	39	43	45
ECS	21	22	35	37
Total systems failures	262	271	201	226

[a] 1-1 Command pilot prime crew 1-3 Command pilot backup crew
1-2 Pilot prime crew 1-4 Pilot backup crew

[b] Both command pilot and pilot received credit for each flight phase and each systems failure.

TABLE 7-VI.- SUMMARY OF AEROMEDICAL EVALUATIONS

Date	Activity	Medical study or support
Nov. 16, 1964	Altitude-chamber spacecraft checkout	Physical examination before and after altitude chamber tests. Biosensors used during tests.
Dec. 5	Final systems test	Physical examination before and after test. Biosensors used during tests.
Dec. 7 Dec. 10	Centrifuge tests (AMAL)	Physical examination before and after centrifuge runs. Biosensors used during tests.
Feb. 26, 1965	Flight simulation no. 1	Biosensors used during simulation.
March 8	Flight simulation no. 2	Physical examination before simulations. Biosensors used during simulation.
March 17	Flight simulation no. 3	Biosensors used during simulation.
March 13	T-10 day physical examination	Complete physical examination. Blood drawn for typing and cross matching.
March 21	T-2 day physical examination	Comprehensive physical examination by medical evaluation team. Complete blood and urine studies.
March 23	Physical examination, launch day	Final physical examination prior to launch.
March 23	Physical examination on the U.S.S. Intrepid at sea	First physical examination postflight. Blood and urine studies. Two tilt procedures on each crew member.
March 25	Postflight physical examination	Final postflight physical examination by medical evaluation team. Complete blood and urine studies.

TABLE 7-VII.- CLINICAL EVALUATION - COMMAND PILOT

[See tilt studies for blood-pressure values]

	Preflight (launch site), March 21, 1965, 8:00 to 12:00 a.m., e.s.t.		Postflight (shipboard), March 23, 1965, 4:00 to 6:00 p.m., e.s.t.		Postflight (launch site), March 25, 1965 10:30 to 11:30 a.m., e.s.t.	
Weight, lb	158		155¾		158	
Temperature, °F	98.4		97.8		98.4	
Respiration rate, breaths/min	16		16		16	
Heart rate, beats/min.	66		96		65	
Heart, lungs	Normal		Normal		Normal	
Skin	Normal		Minimal reaction to colostomy tape of biosensors		Normal	
Comments	Excellent health		Thirsty, alert, cooperative		Excellent health	
Extremity measurements, (a)	Left	Right	Left	Right	Left	Right
Forearm, max, in.	12½	11¾	12¾	11¼	12½	11¾
Forearm, min, in.	7½	6⅜	--	--	7½	6⅜
Calf, max, in.	15¼	16	15⅛	15¼	15¼	16
Calf, min, in.	8½	8⅜	--	--	8½	8⅜

[a] Shipboard extremity measurements not made by same individual who made the other two sets of measurements.

TABLE 7-VIII.- CLINICAL EVALUATION - PILOT

[See tilt studies for blood pressure values]

	Preflight (launch site), March 21, 1965, 8:00 to 12:00 a.m., e.s.t.	Postflight (shipboard), March 23, 1956, 4:00 to 6:00 p.m., e.s.t.	Postflight (launch site) March 25, 1965, 10:30 to 11:30 a.m., e.s.t.
Weight, lb	165	161½	164
Temperature	97.9	98.8	98.6
Respirations rate, breaths/min	16	18	20
Heart rate, beats/min. . .	60	72	62
Heart, lungs	Normal	Normal	Normal
Skin	Clear	Minimal reaction micropore tape at all biosensor sites	Clear
Comments	Excellent health	Thirsty, mildly euphoric, alert, cooperative	Excellent health, rested

Extremity measurements, (a)	Left	Right	Left	Right	Left	Right
Forearm, max, in.	10¼	10¾	10	10¾	10¼	10¾
Forearm, min, in.	6¾	7⅞	--	--	6¾	7⅞
Calf, max, in.	14⅞	15⅛	14¾	14¾	14⅞	15⅛
Calf, min, in.	8½	8⅜	--	--	8½	8⅜

[a] Shipboard extremity measurements not made by same individual who made the other two sets of measurements.

TABLE 7-IX.- HEMATOLOGY - COMMAND PILOT

Determination	Preflight	Postflight		
Date, 1965 Time, e.s.t.	March 13, 8:00 a.m.	March 23, 8:00 a.m.	March 23, 9:30 p.m.	March 25, 10:30 a.m.
White blood cells, /mm^3	5 900	9 600	7 300	6 400
Neutrophiles, percent	51	78	72	40
Lymphocytes, percent	45	21	21	53
Monocytes, percent	3	1	1	3
Eosinophiles, percent	1	0	0	1
Basophiles, percent	0	0	0	3
Platelets/mm^3	380 000	—	—	200 000
Red blood cells, millions/mm^3	4.53	4.67	4.33	4.9
Hemotocrit, percent	44	47	43	45
Hemoglobin, g/100 ml	15.0	14.8	13.2	14.8
Corrected sedimentation rate, mm/hr	8	—	—	8
Blood morphology	Normal	Normal	Normal	Normal
Sodium, mEq/l	141	145	140	140
Potassium, mEq/l	4.85	5.1	4.5	4.9
Chloride, mEq/l	103	103	102	102
Calcium, mEq/l	4.95	5.3	5.1	5.1
Phosphorus, mg/100 ml	3.6	4.6	5.4	3.6
CO_2, mEq/l	28	—	—	24.5
Glucose, mgm/100 ml (non-fasting)	100	—	—	100
Total protein, g/100 ml	7.15	7.85	7.40	7.9
Albumen	4.25	4.95	4.95	5.0
Globulin	2.90	2.90	2.45	2.9

TABLE 7-X.- HEMATOLOGY - PILOT

Determination	Preflight	Postflight		
Date, 1965 Time, e.s.t.	March 13, 8:00 a.m.	March 23, 4:00 p.m.	March 23, 9:50 p.m.	March 25, 10:30 a.m.
White blood cells, /mm^3	7 475	8 800	—	9 100
Neutrophiles, percent	54	80	81	64
Lymphocytes, percent	38	19	17	27
Monocytes, percent	5	1	0	4
Eosinophiles, percent	3	0	1	3
Basophiles, percent	0	0	0	2
Platelets/mm^3	292 000	—	—	225 000
Red blood cells, millions/mm^3	5.28	4.68	—	5.3
Hematocrit, percent	45	47	—	45
Hemoglobin, g/100 ml	15.2	14.4	—	15.1
Corrected sedimentation rate, mm/hr	8	—	—	5
Blood morphology	Normal	Normal	Normal	Normal
Sodium, mEq/l	141	144	142	140
Potassium, mEq/l	5.2	5.0	4.5	5.5
Chloride, mEq/l	107	107	106	105
Calcium, mEq/l	4.7	5	5	5.15
Phosphorus, mg/100 ml	3.5	4.5	5.25	4.0
CO_2 mEq/l	25.5	—	—	26.7
Glucose, mg/100 ml	98	—	—	75
Total protein, g/100 ml	6.70	7.15	7.40	7.15
Albumen	4.25	4.95	4.70	4.60
Globulin	2.45	2.20	2.70	2.55

TABLE 7-XI.- URINALYSIS

[Two postflight urine specimens on each subject were inadvertently not collected]

	Preflight	Postflight				
Command pilot						
Date, 1965 Time, e.s.t.	March 21 8:30 a.m.	March 23 5:50 p.m.	March 24 7:15 a.m.	March 24 12:45 p.m.	March 24 11:00 p.m.	March 25 10:30 a.m.
Color, appearance . . .	Yellow, clear	Straw, clear	Straw, slightly hazy	Straw, clear	--	Yellow, clear
Specific gravity . . .	1.025	1.019	1.025	1.018	1.017	1.016
pH	5	5	5	5	6	8
Albumen, sugar, acetone, bile	Negative	Negative	Negative	Negative	Negative	Negative
Microscopic	Rare white blood cells	1 to 3 white blood cells, rare gran cast, mucous threads	4 to 6 white blood cells, many mucous threads, amorphous sediment	2 to 3 white blood cells, 0 to 1 red bblood cell, mucous threads	2 to 4 white blood cells, mucous threads	Rare white blood cells
Volume, cc	38	190	210	280	290	154
Pilot						
Date, 1965 Time, e.s.t.	March 21 8:30 a.m.	March 23 4:00 p.m.	March 24 7:15 a.m.	March 24 12:45 p.m.	March 24 11:20 p.m.	March 25 10:30 a.m.
Color, appearance . . .	Yellow, clear	Yellow, clear	Dark, hazy	Straw, clear	--	Yellow, clear
Specific gravity . . .	1.022	1.025	1.027	1.025	1.005	1.024
pH	5	6	5	6	5	6
Albumen, sugar, acetone, bile	Negative	Negative	Negative	Negative	Negative	Negative
Microscopic	Rare white blood cells, epithelial cells	0 to 1 white blood cell	1 to 2 white blood cells	0 to 1 white blood cell, rare amorphous sediment	0 to 1 white blood cell, rare amorphous sediment	Rare white blood cells
Volume, cc	46	116	180	175	650	150

TABLE 7-XII.- POSTFLIGHT MEDICAL ACTIVITIES

Date, 1965	Time, e.s.t.	Activity
March 23	2:17 p.m.	Landing
	2:42 p.m.	Doff pressure suits
	3:05 to 3:09 p.m.	Egress from spacecraft
	3:28 p.m.	Arrive aboard recovery ship
	3:50 p.m.	Blood specimens (begin medical exam)
	4:01 p.m.	Tilt study, command pilot
	4:40 p.m.	Tilt study, pilot
	5:10 p.m.	Complete initial examination
	6:50 p.m.	First postflight meal (unrestricted)
	9:50 p.m.	Second blood specimens
	10:00 p.m.	To bed (command pilot)
	11:00 p.m.	To bed (pilot)
March 24	8:30 to 9:45 a.m.	Medical debriefing and second tilt study
March 25	8:45 a.m.	Depart recovery ship
	10:30 a.m.	Medical specialists examination, launch site

NASA-S-65-3642

SUMMARY FLIGHT PLAN

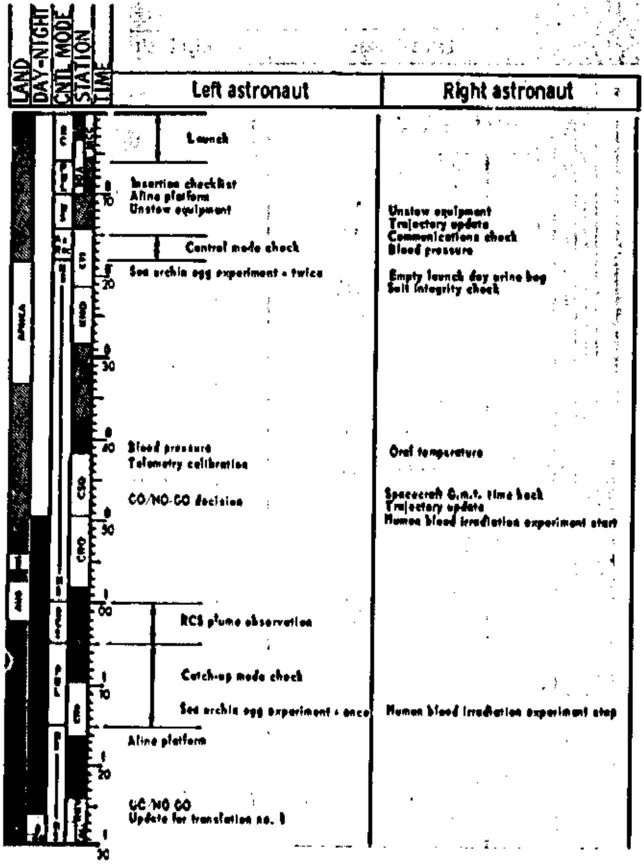

Figure 7.-1. - Summary flight plan.

NASA-S-65-3643

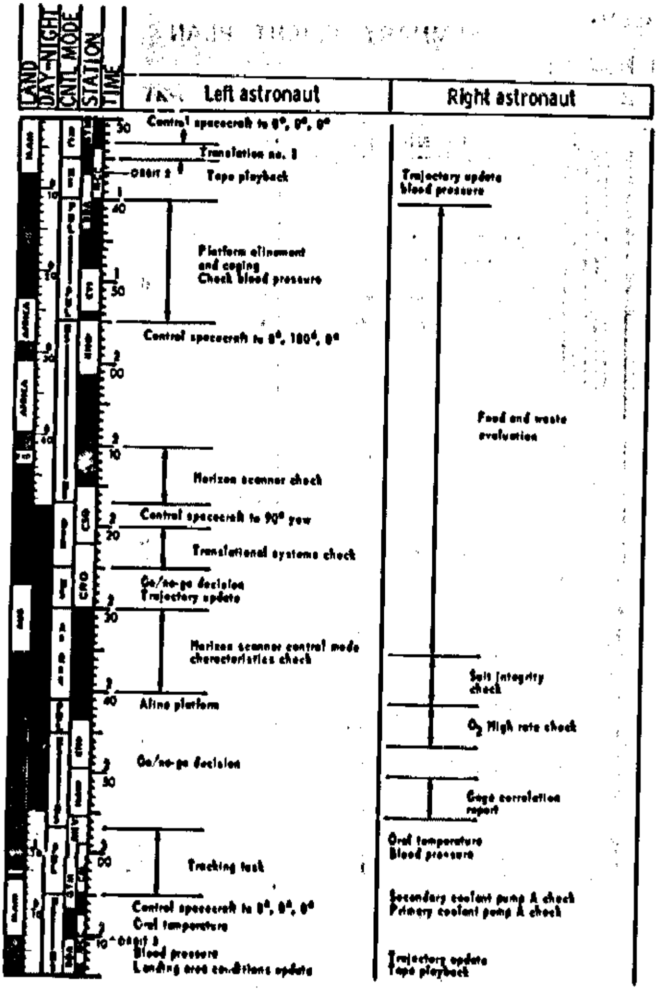

Figure 7.-1. - Summary flight plan (Continued)

NASA-S-65-3644

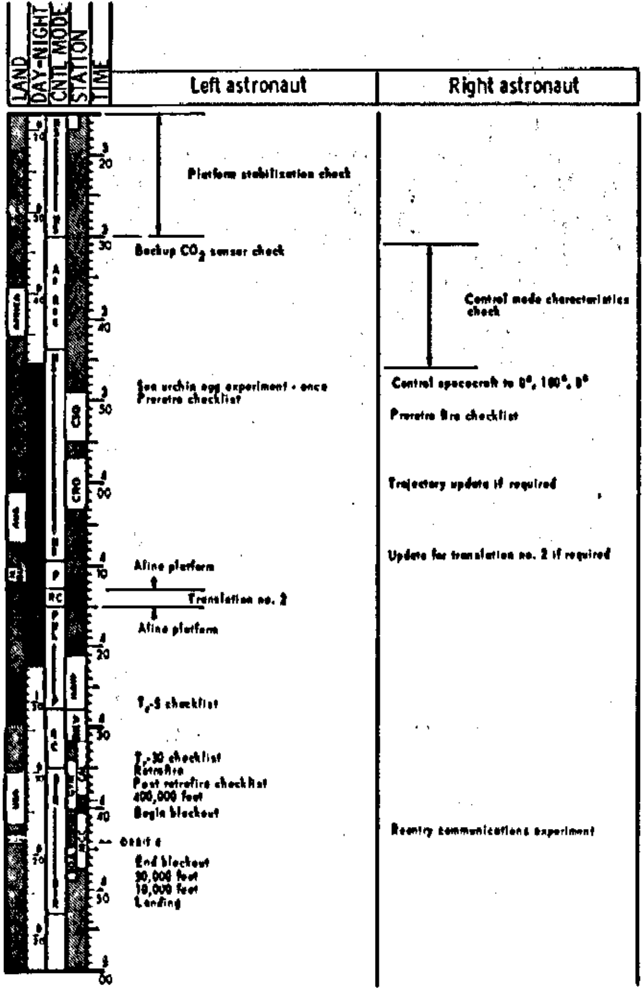

Figure 7.-1. - Summary flight plan (Concluded)

NASA-S-65-3595

Figure 7-2. - Target area for tracking task

NASA-S-65-3634

Figure 7-3. - Flight crew in spacecraft prior to launch (viewed through window of open hatch)

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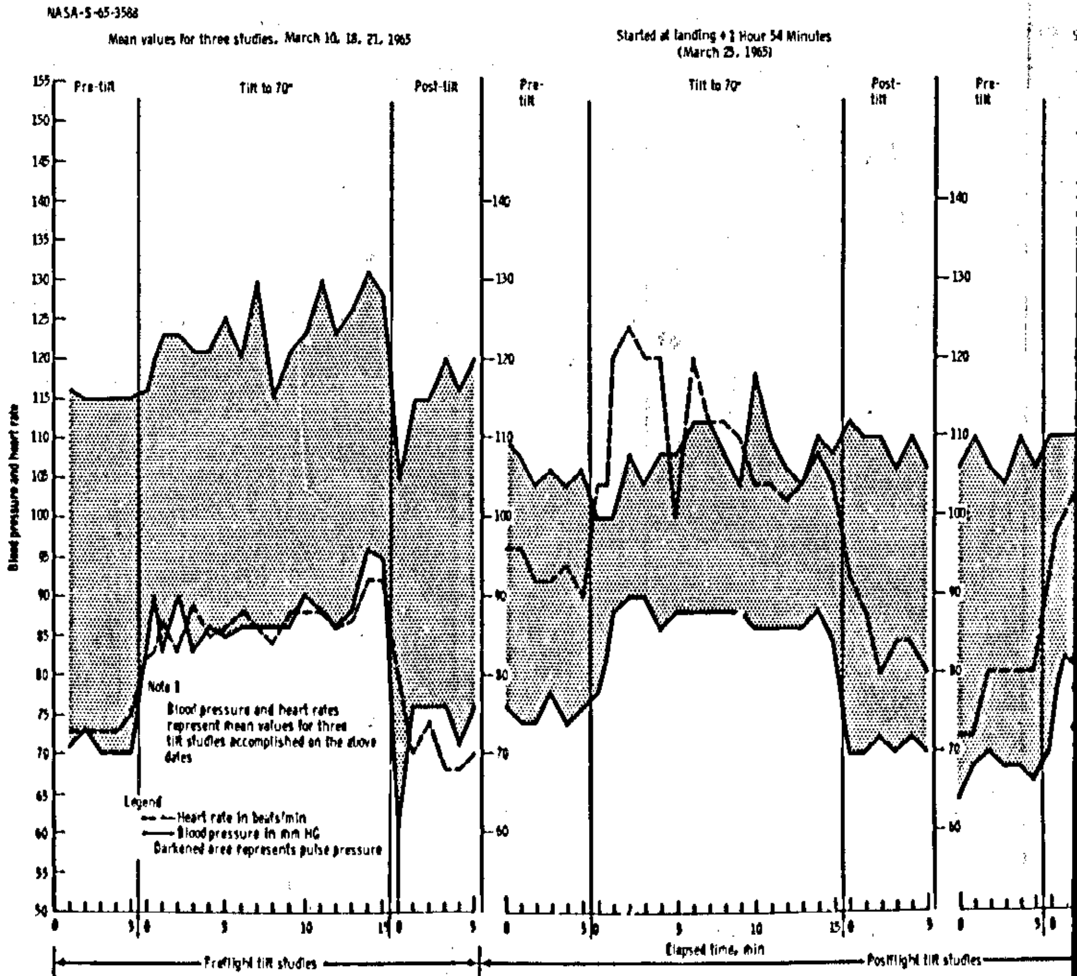


Mean values for three studies, March 10, 18, 21, 1965

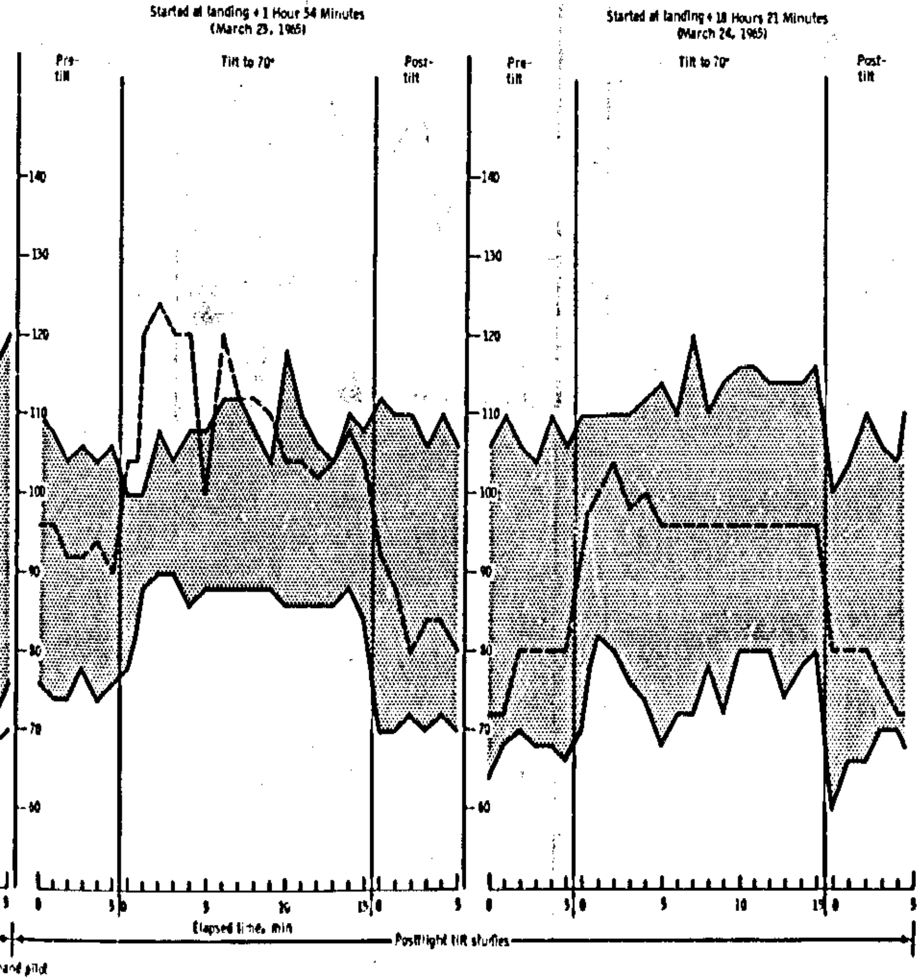
Started at landing + 1 Hour 54 Minutes
(March 25, 1965)

Note 1

Blood pressure and heart rates represent mean values for three tilt studies accomplished on the above dates

Legend
— — Heart rate in beats/min
—— Blood pressure in mm HG
Darkened area represents pulse pressure

Figure 7-4. - Tilt table studies, command pilot.



Started at landing + 1 Hour 54 Minutes (March 23, 1965)

Started at landing + 18 Hours 21 Minutes (March 24, 1965)

[redacted]

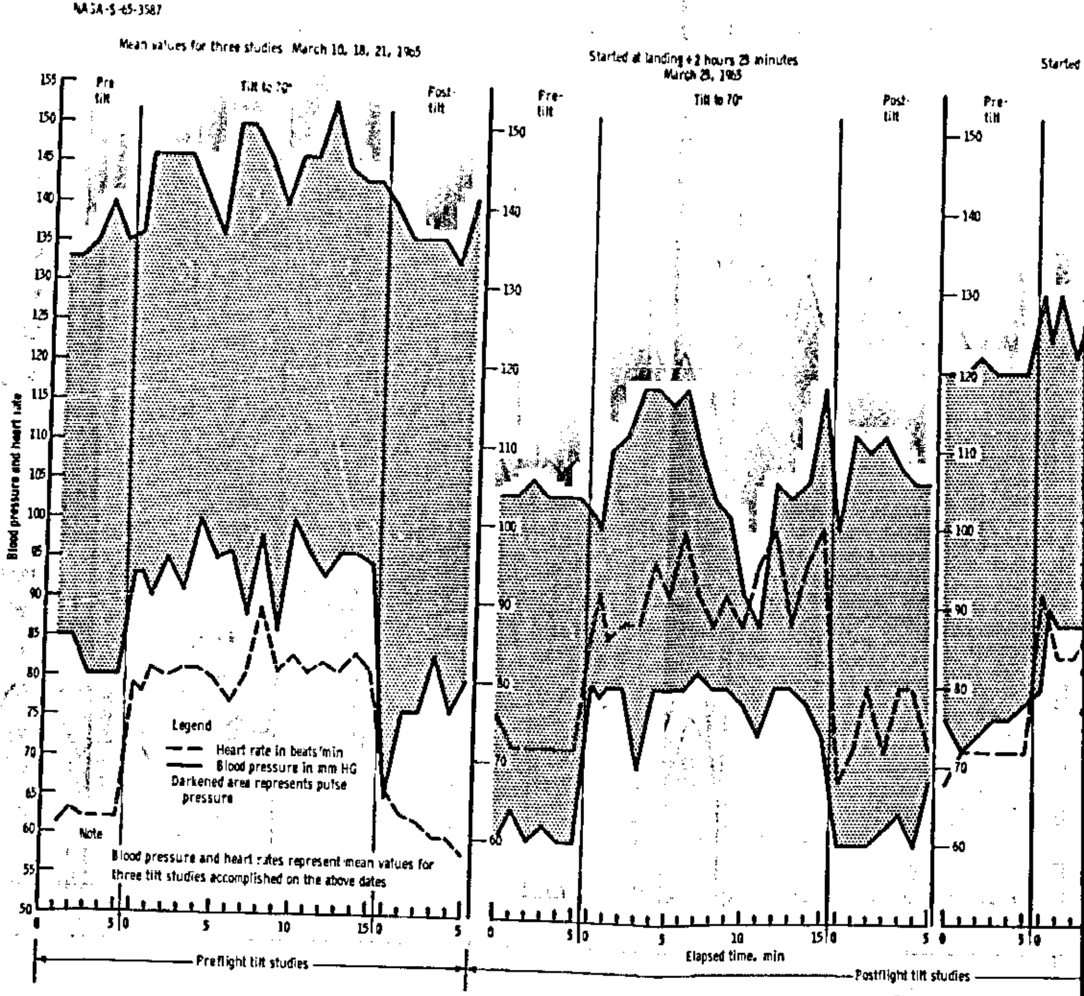


Figure 7-5. - Tilt table studies, p

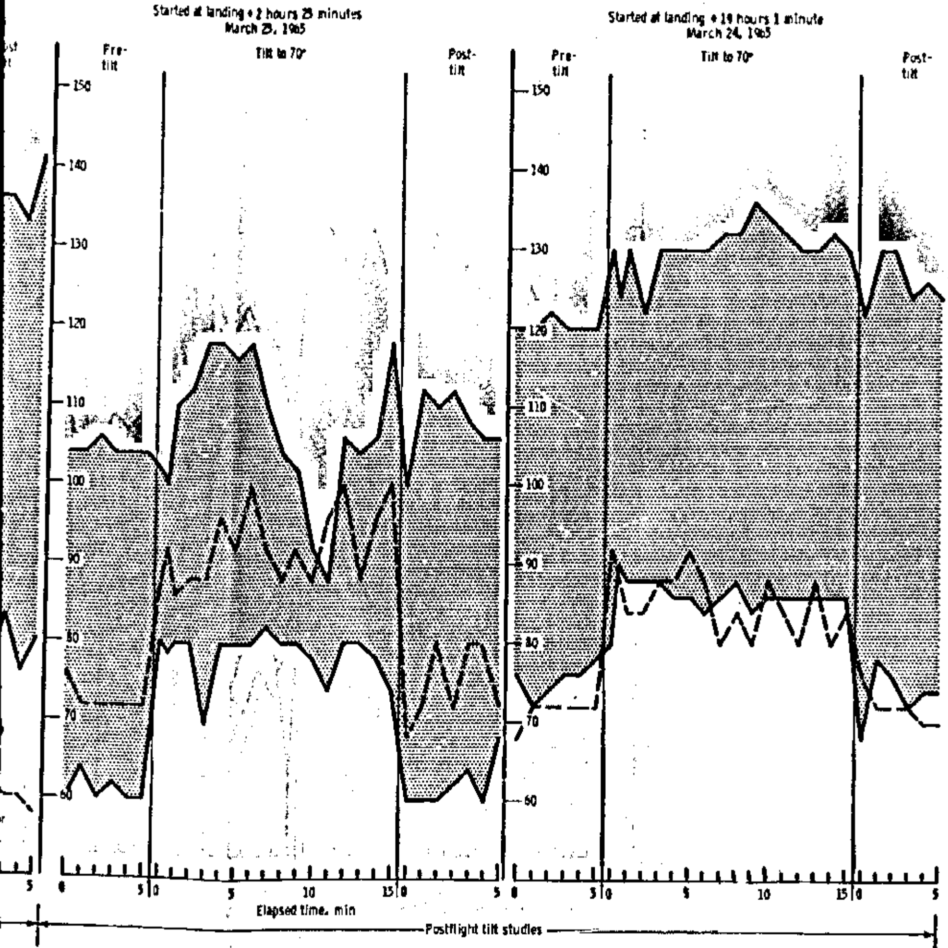
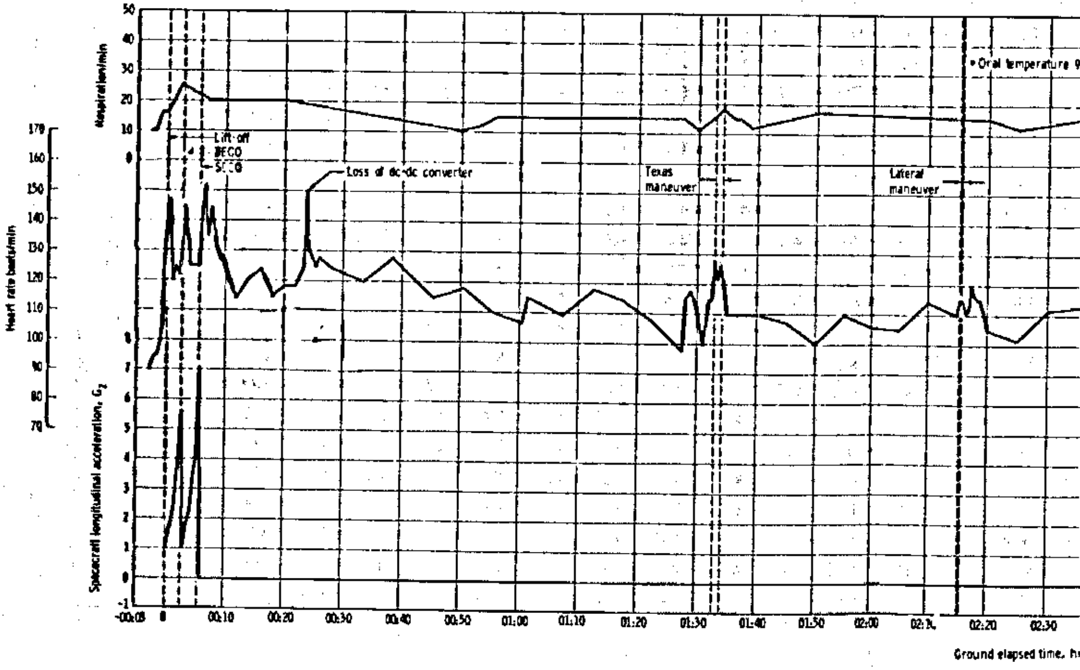


Figure 7-5. - Tilt table studies, pilot

NASA-S-65-3598



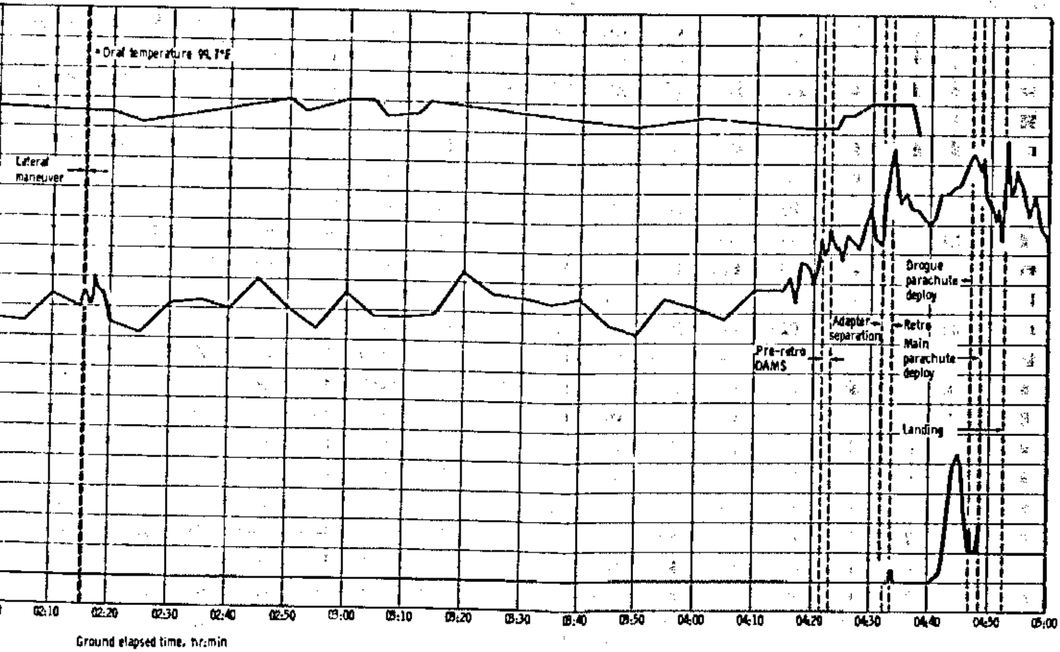
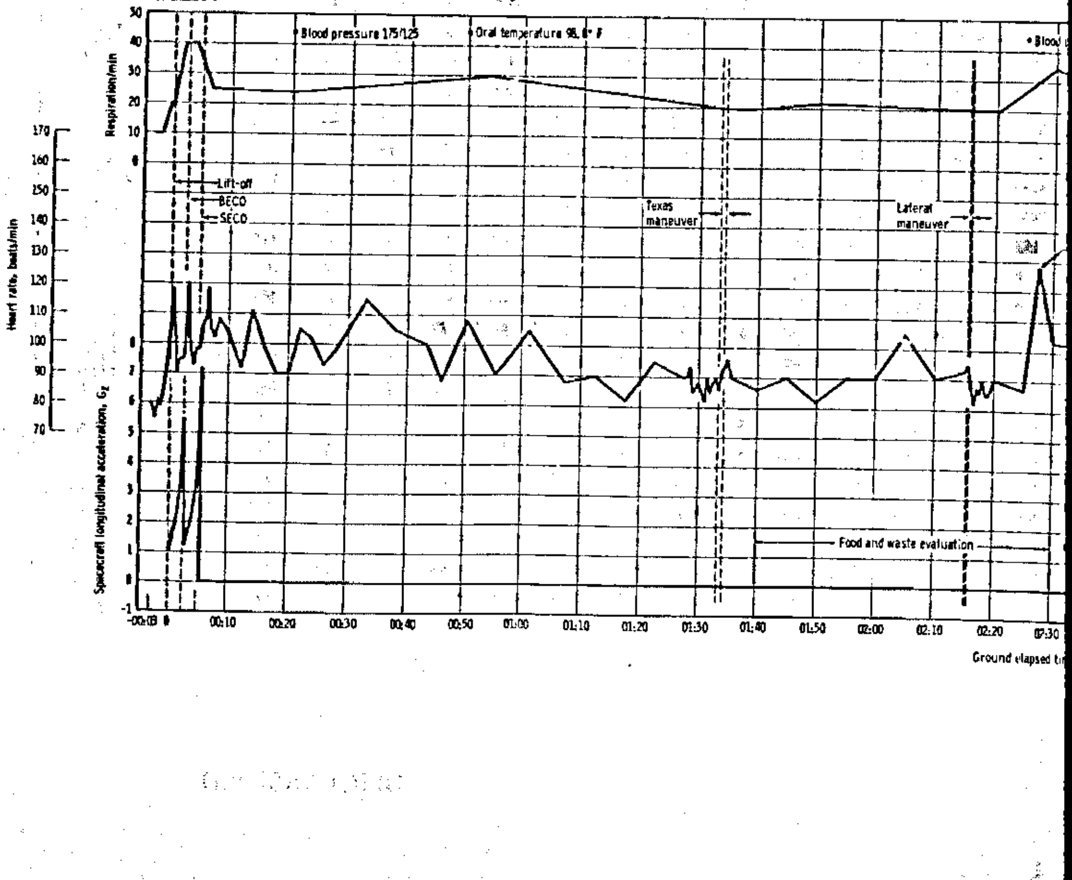


Figure 7-8. - Physiological measurements - command pilot

NASA-S-65-3597



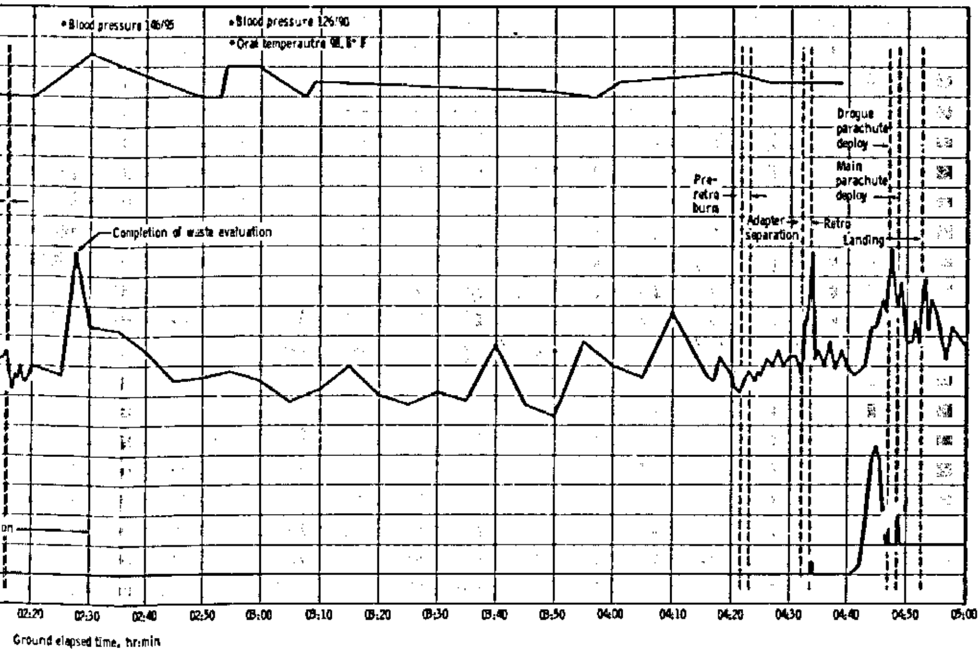


Figure 7-7. - Physiological measurements, pilot

8.0 EXPERIMENTS

The GT-3 mission was the first in the Gemini series to have inflight experiments incorporated into the planned mission objectives. Two of the three GT-3 mission experiments were sponsored by the NASA Office of Space Science and Applications: Experiment S-2, Effects of Sub-Gravity on the Fertilization and Development of Sea Urchin Eggs; and Experiment S-4, Synergistic Effects of Zero Gravity and Radiation on White Blood Cells. The remaining GT-3 mission experiment was sponsored by the NASA Office of Advanced Research and Technology: Experiment T-1, Reentry Communications. Experiments S-4 and T-1 are considered to have been completely successful; however, data from Experiment S-2 reveal that all objectives were not achieved. The S-2 experimenter is Dr. R. S. Young of NASA Ames Research Center. The S-4 experimenter is Dr. M. A. Bender of Oak Ridge National Laboratory, and the T-1 experimenters are W. F. Cuddihy, L. C. Schroeder, and T. E. Sims of NASA Langley Research Center.

8.1 EXPERIMENT S-2, THE EFFECTS OF SUB-GRAVITY ON THE FERTILIZATION AND DEVELOPMENT OF SEA URCHIN EGGS

8.1.1 General

The objective of this experiment was to evaluate the effects of sub-gravity (fields substantially less than 1) on fertilization, cell division, differentiation, and growth of a relatively simple biological system.

The experimental apparatus was cylindrical in shape and consisted of eight specimen chambers as shown in figure 8-1. Each chamber was divided into three compartments so that the sperm, ova, and fixative solution were separate. Rotation of a handle actuated either the fertilization or the fixation process within the various chambers. Florida sea urchin eggs (Arbacia punctulata) were used in this experiment. Four chambers were to be fertilized shortly before launch, while the remaining four chambers were to be fertilized shortly after orbital insertion. Growth in both groups of fertilized eggs was to be inhibited at various stages of development by the addition of the fixative solution.

After recovery, slight changes or abnormalities in mitotic figures, growth, and differentiation of the cells exposed to space flight were to be compared with results obtained from an identical experiment performed simultaneously on the ground.

8.1.2 Procedure

The experiment required five scheduled operations (that is, rotation of the handle 12° to the right and release, after which it returned to the start position for the next operation) as follows:

Time	Operation	Description
30 minutes before launch	1	Fertilization of chambers 1, 2, 3, and 4
00:20:00 g.e.t.	2 and 3	Fertilization of chambers 5, 6, 7, and 8 Fix chambers 2 and 4
01:10:00 g.e.t.	4	Fix chambers 1, 3, 5, and 6
03:50:00 g.e.t.	5	Fix chambers 7 and 8

Verification of each operation was given so that the operation of the ground control could be synchronized with the flight experiment. Cabin temperature was noted and recorded along with time of each operation. After recovery of the spacecraft, the experimental devices were removed immediately and returned to experiment personnel at Cape Kennedy for analysis.

8.1.3 Results

The experiment was flown and recovered as scheduled. Subsequent analysis shows that the stated scientific objectives of the experiment were not achieved.

The reasons for failure are primarily mechanical. There may have been leakage in the formalin chambers sufficient to damage the eggs. In addition, the operating mechanism failed. As a result, subsequent operation of the handle did not properly actuate the device. These problems resulted in an incomplete experiment and in conditions which prevent accurate interpretation of that part of the experiment which was completed.

8.2 EXPERIMENT S-4, SYNERGISTIC EFFECTS OF ZERO-G AND RADIATION ON WHITE BLOOD CELLS

8.2.1 General

Experiment S-4 was designed to determine if there is a synergistic effect of zero-g or other space-flight parameters and radiation on the induction of chromosomal aberrations in human leukocytes. A sealed experimental device, shown in figure 8-2, was located on the right hatch of spacecraft 3, and a duplicate device was located in a hangar at Cape Kennedy during the flight. The flight device was actuated at 00:50:18 g.e.t. and deactuated at 01:10:18 g.e.t., thus starting and stopping the irradiation of a series of human blood samples by phosphorous 32 beta rays. The corresponding actuation and deactuation of the ground control device occurred at 00:52:00 g.e.t. and 01:12:00 g.e.t., respectively. Blood samples were put in tissue culture by 6 hours after the spacecraft landed. At approximately 77:00:00 g.e.t., all cultures were fixed, and chromosome preparations were made.

The function of the flight and the ground-control experimental devices, the enclosed time, temperature, and radiation measuring instruments, and all blood cultures were completely successful. Although scoring of the chromosome preparations and analysis of the data have not yet been completed, preliminary results do not show any effect of weightlessness or other space-flight parameters on human chromosomes, nor any synergism between these factors and the production of chromosomal aberrations by radiation.

8.2.2 Experiment

The assembled flight device and the ground control were successfully actuated and deactuated at the required times and all internal apparatus functioned perfectly. The flight device was removed from the spacecraft a few minutes after it was recovered. Color film records were successfully recovered from the experiment equipment and were developed. These film records agree with the actuation and temperature information provided by the flight and ground-control records.

The film records indicate that the flight device reached a temperature of 90° F during assembly and that it did not exceed 100° F nor drop below 58° F at any time during the preflight, flight, and recovery periods. The ground-control device did not become as warm, remaining between 34° F and 84° F during these periods. Further confirmation of the proper functioning of the equipment temperature recorders is provided by postflight X-rays, which show that neither the high (110° F) nor the low (10° F) limit coulometers in the experiment equipment registered any movement.

On both the flight and the ground-control instrument package film records, the actuation marker appears in the proper position, and in that position only. Although the time resolution of these instruments was not designed to be high enough for accurate time determination, the time markers appear to be of the proper width. That the exposure was of the required duration is confirmed by fluoroglass dosimeters incorporated into the blood sample chamber screws.

8.2.3 Dosimeters

Each experimental device contained twenty 1- by 6-mm cylindrical fluoroglass dosimeters, two within the volume of each of the ten blood samples. In addition, two 8- by 8- by 4.7-mm fluoroglass block dosimeters were incorporated in each instrument package to measure the non-electron background radiation. All 44 dosimeters were recovered and read on a fluorimeter calibrated against fluoroglass standards that had been given known cobalt 60 gamma ray exposures at the National Bureau of Standards. The results, together with preliminary dose estimates, are presented in table 8-I. The values are averages of two readings for each dosimeter for each pair of blood samples. The readings and preliminary dose estimates agree with both the theoretical expectations and the results of previous control experiments. A large part of the approximately 2-rad dose to the control blood samples was a result of Bremsstrahlung with a peak energy of about 65 KeV. The large-volume block dosimeters located in the ground-control instrument package registered an exposure of approximately 2 rads of Bremsstrahlung also. Those dosimeters from the flight instrument package registered somewhat higher exposures, because of the ambient radiation encountered by spacecraft 3 during its flight.

8.2.4 Blood Samples

All blood samples from both the flight and the ground-control experimental devices were successfully cultured and yielded satisfactory chromosome preparations. Very little haemolysis was noted after the blood samples had been centrifuged, and no other evidence of gross cell damage was seen. Chromosome preparations from both preflight and postflight blood samples from the flight crew have been scored. Of the experimental samples, only two of the five dose groups have been scored as of this reporting. The results are shown in table 8-II.

The chromosome analyses of each flight crew member's blood sample show no increase in aberration frequency due to the space flight. Two dicentric chromosomes seen in the samples from the pilot appear to be identical, and were without acentric fragments. They, therefore, cannot be attributed to the space flight. The deletion frequencies seen are typical of normal individuals.

The aberration frequencies seen in the control blood samples are what is expected in cells exposed to such a low radiation dose and agree well with the results of previous control experiments. The irradiated samples have yielded the expected types of aberrations at approximately the rates measured in previous control experiments. The frequency of chromosome-type deletions in the irradiated flight samples is somewhat higher than that for the irradiated ground-control samples, but the difference does not appear to be significant. In any case, there is no great difference between the flight and ground-control frequencies of dicentric chromosomes.

Although a final conclusion must wait completion of scoring of the chromosome preparations from the blood samples in the three remaining exposure groups and subsequent statistical analysis of the data, the preliminary results of the S-4 experiment fail to support the hypothesis that there is a synergistic effect of any space-flight parameter and radiation, at least on the production of chromosomal aberrations.

8.3 EXPERIMENT T-1, REENTRY COMMUNICATIONS

8.3.1 General

A method of overcoming reentry radio blackout by injecting water into the flow field was demonstrated during the GT-3 mission. Preliminary data show that significant levels of signal-strength increase were observed corresponding to portions of the water injection sequence on UHF telemetry (TM) (230.4 mc) and on UHF voice (296.8 mc) frequencies. Detailed analysis of the experiment results will be reported by NASA after the many mission records necessary for such an analysis are made available.

8.3.2 Introduction

Cut-off of communications with spacecraft during reentry is caused by the ionized sheath which envelops the spacecraft as a result of the high temperature generated while entering the atmosphere. In a research program called Project RAM (Radio Attenuation Measurements), Langley Research Center has been investigating the communications blackout problem for several years.

One of the most promising techniques for reducing reentry communications effects to come out of this work is the spraying of materials into the air stream to reduce ionization levels sufficiently for standard communications frequencies to pass to and from the spacecraft. This technique was previously tested on a small spacecraft launched from Wallops Island, Virginia. The material used was water.

The GT-3 mission offered an opportunity to test the technique under practical conditions (that is, on a large blunt spacecraft with an ablative heat shield). Various amounts of water for a number of brief periods were sprayed into a sector of the flow field during the reentry period. During the data period, this sector faced generally south and slightly upward. The data of interest are UHF and C-band signal amplitudes as monitored at the many ground receiving stations located near the reentry corridor. Preliminary data from several ground receiving stations show that signals from the spacecraft which were blocked out were reestablished during portions of the water injection sequence.

No attempt was made to provide operational communications for the GT-3 mission.

8.3.3 Experiment Description

The onboard experiment equipment consisted of a water-injection system which was designed into the spacecraft right main landing-gear well. Figure 8-3 illustrates the location of the experiment on the spacecraft, and figure 8-4 illustrates the installation of the experiment onto the inside of the door. The system was completely independent of other spacecraft interfaces except for the actuation switch and electrical wiring to the crew station. This actuation switch was located on the spacecraft right switch panel and was labeled RE-ANT ANT EXP.

The system was actuated by the pilot. Actuation simultaneously supplied water expulsion pressure through the regulator to the water tank and started the mechanical timer as indicated in figure 8-5. The timer opened and closed contacts to each of three solenoids once every 15 seconds, allowing water to be ejected. This system was designed to produce pulses alternately of low, medium, and high flow rates with durations of 0.1 second, 0.25 second, and 0.5 second. Shown in figure 8-6 is flow rate plotted against time for one injection cycle. The water supplied was depleted after 10 complete cycles.

The experiment actuation time was to be computed after retrofire. On a nominal mission, the pilot was instructed to actuate the experiment 45 seconds after the onset of blackout, which occurs at an altitude of about 308 000 feet. The pilot noted the time of actuation, and deactivated the experiment at least 4 minutes after actuation as required.

During the experiment operation period, ground stations located near the reentry corridor monitored and recorded signal strength on UHF TM (230.4 mc), UHF voice (296.8 mc), and C-band (5610 mc) frequencies. The following stations were designated to monitor signal strength:

Station	Frequencies monitored		C-band
	UHF, telemetry	UHF, voice	
Corpus Christi	X	X	
Eglin AFB	X	X	X
MCC (Cape Kennedy)	X	X	
Tel II (Cape Kennedy)	X	X	
MILA			X
Patrick AFB			X
GBI	X		X
GTI	X		X
Anclote Point	X	X	X
Key West (LRC)	X	X	
Homestead (LRC)	X	X	
Aircraft	X	X	

The locations of stations with respect to the blackout and experiment period are shown in figure 8-7.

[redacted]

8.3.4 Results and Discussion

Retrofire occurred at 04:33:23 g.e.t. The onset of the UHF blackout occurred at 04:39:59 g.e.t., and the experiment was activated at 04:41:14 g.e.t. The experiment lasted approximately 150 seconds. The variation of altitude with time is shown with reference to the experiment times in figure 8-8. Event times are also given in figure 8-8.

The flow rate of the water injection system was determined by preflight calibration and was not monitored in real time. Ten cycles of three different flow rates ranging from 0.3 lb/sec to 7.5 lb/sec were employed during the 150-second data period noted in figure 8-8.

At this reporting, only preliminary signal strength records are available for study, and these are available from only three stations (Key West, Homestead, and Grand Bahama Island). The data from these stations show that significant levels of signal recovery were observed corresponding to portions of the water injection sequence on UHF telemetry and voice frequencies. UHF signal strength plotted against mission time from the Key West station shows two well-defined recovery pulses, and, in addition, a third less definitive one. Each corresponds to the high water-flow rate. (See fig. 8-9.) GBI and Homestead stations observed only the first two signal recovery pulses. No UHF signal recovery was observed from the low and medium flow rates. C-band records are not yet available for study.

8.3.5 Concluding Remarks

The water addition technique for permitting communications during reentry was demonstrated for a large blunt-shaped spacecraft with an ablative heat shield. The mass flow required to produce significant levels of signal strength increase was shown to be between 1.5 pounds per second and 7.5 pounds per second for the particular experiment configuration used for spacecraft 3 reentry. The fact that the voice and telemetry links were reestablished is significant inasmuch as the orientation of the experiment (facing slightly upward) and the location of the injector nozzles relative to the antenna were far from optimum.

[redacted]

TABLE 8-I.- PRELIMINARY DOSE ESTIMATES FROM FLUOROGLASS DOSIMETERS INCORPORATED IN EXPERIMENT S-4 BLOOD SAMPLE CHAMBERS

Device	Nominal dose	Average reading, $^a\mu A$	Net β-ray dose, $^a\mu A$	Estimate of β-ray dose, rad	Estimated total dose, rad
Ground control	0	4.5	-	-	2
	50	18.1	13.6	50	52
	100	27.7	23.2	86	88
	150	41.9	37.4	138	140
	200	51.0	46.5	172	174
Flight device	0	4.2	-	-	2
	50	18.0	13.8	51	53
	100	29.5	25.3	94	96
	150	38.3	34.1	126	128
	200	46.8	42.5	157	159

[a] One μA equals two Co^{60} γ-ray rad, as the instrument was set up.

TABLE 8-II.- PARTIAL RESULTS OF EXPERIMENT S-4 CHROMOSOME ABERRATION ANALYSIS

Subject	Sample	Cells scored	Estimated dose	Chromatid deletions	Chromosome deletions	Dicentric chromosomes
Pilots						
Command pilot	Preflight	100	–	3	0	0
	Postflight	200	–	2	0	0
Pilot	Preflight	100	–	0	1	[a]1
	Postflight	200	–	0	1	[a]1
Control						
Male	Ground	200	2	3	3	1
	Flight	200	2	0	0	0
Female	Ground	200	2	0	0	0
	Flight	200	2	1	3	0
Irradiated						
Male	Ground	200	88	4	6	4
	Flight	200	96	4	14	7
Female	Ground	200	88	1	2	9
	Flight	200	96	2	7	9

[a] These two dicentric chromosomes appear identical; both lacked acentric fragments.

NASA-S-65-3551

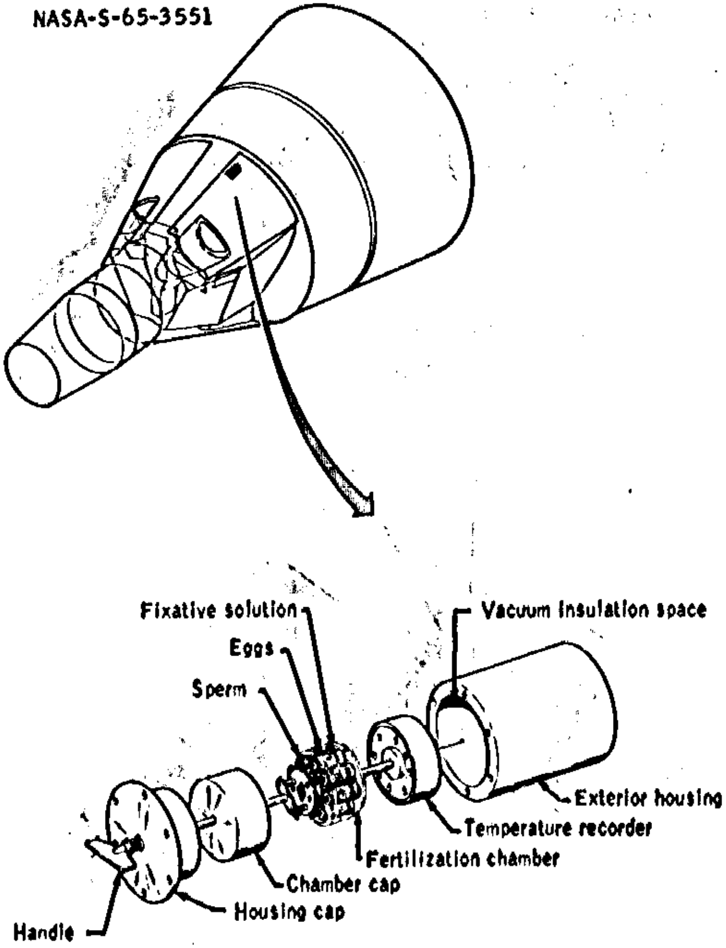


Figure 8-1. - S-2 sea urchin egg experiment equipment

NASA-S-65-3550

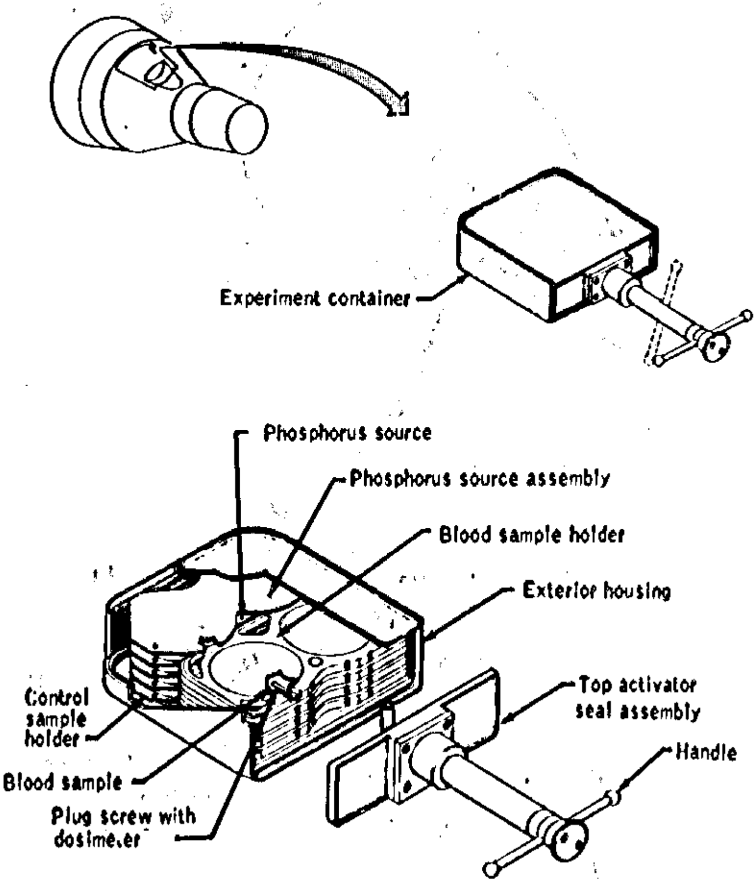


Figure 8-2. - S-4 blood cell experiment equipment

NASA-S-65-3553

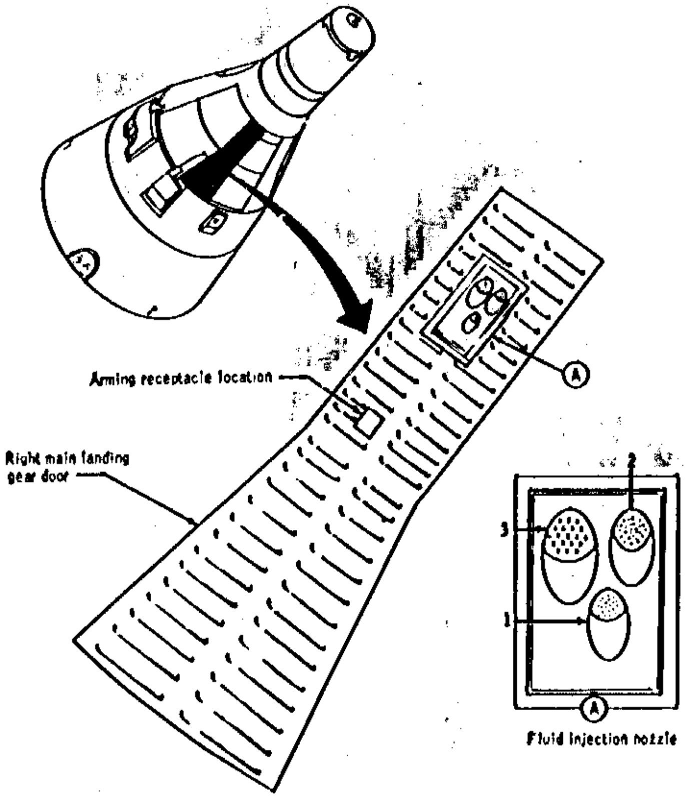


Figure 8-3. -Gemini spacecraft location of reentry communications experiment

NASA-S-65-3555

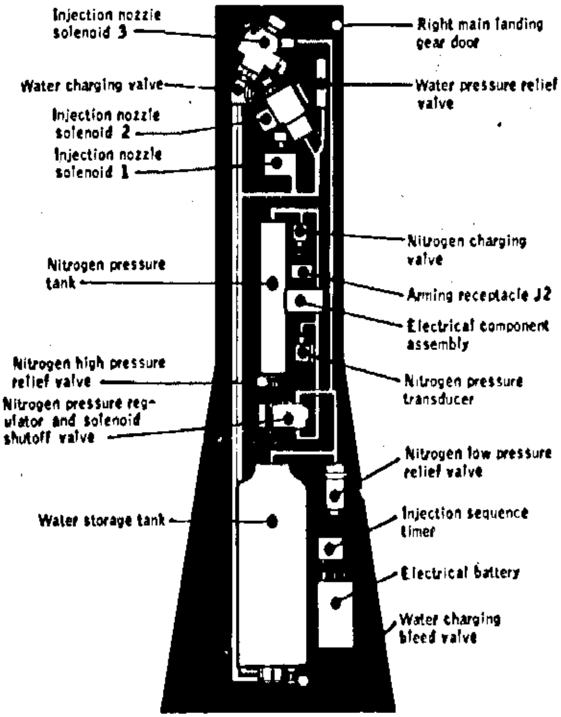


Figure 8-4. - Reentry communications experiment equipment installation on the inside of the right main landing gear door

NASA-S-65-3554

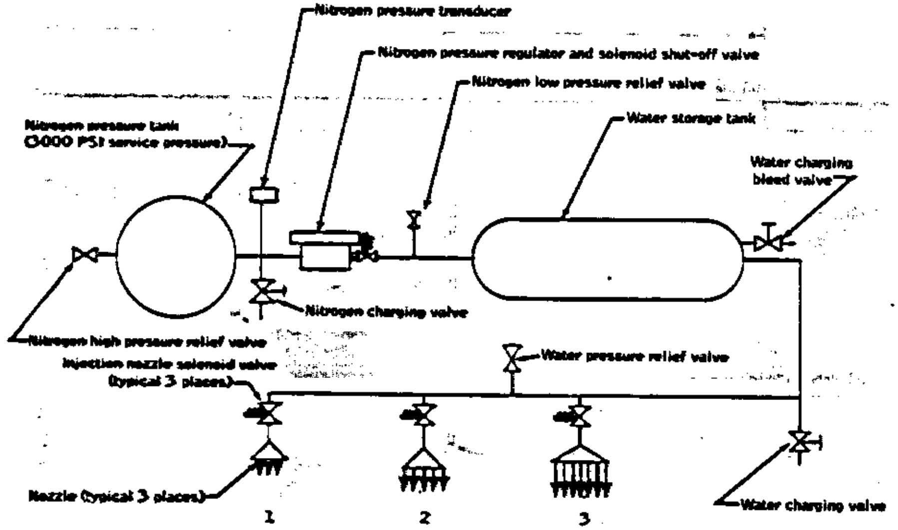
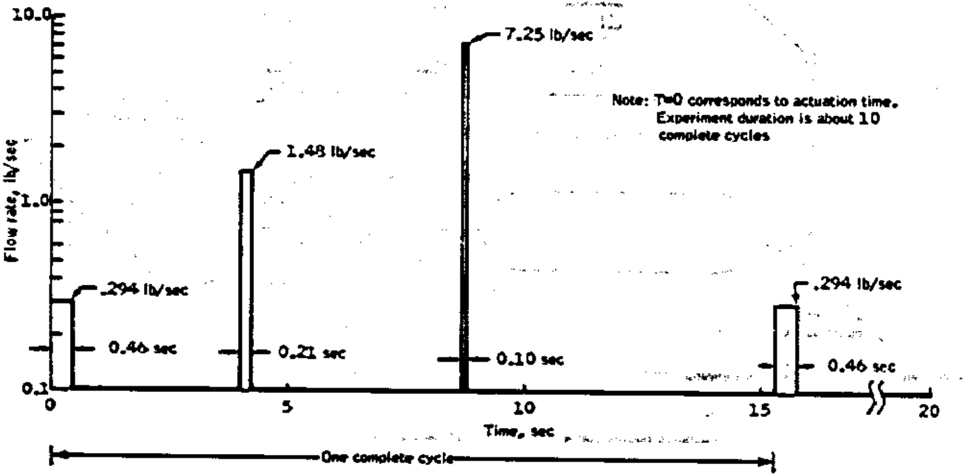


Figure 8-5. - Reentry communications experiment flow diagram

NASA-S-65-3566



Note: T=0 corresponds to actuation time. Experiment duration is about 10 complete cycles

Figure 8-6. - Flow rate cycle for T-1 Gemini reentry communications experiment

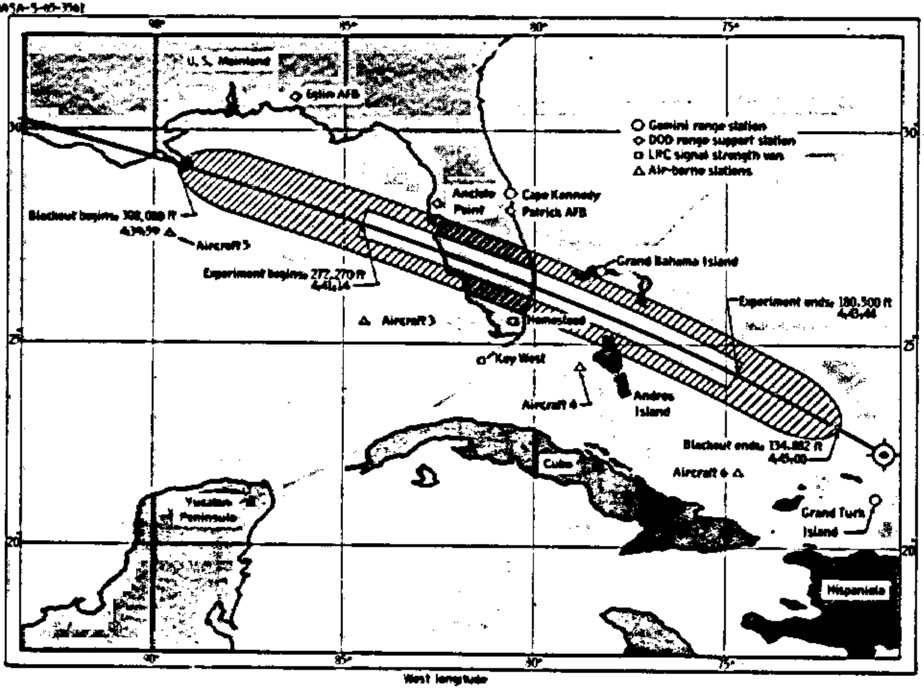


Figure 8-7. - T-1 experiment ground station locations

NASA-S-65-3552

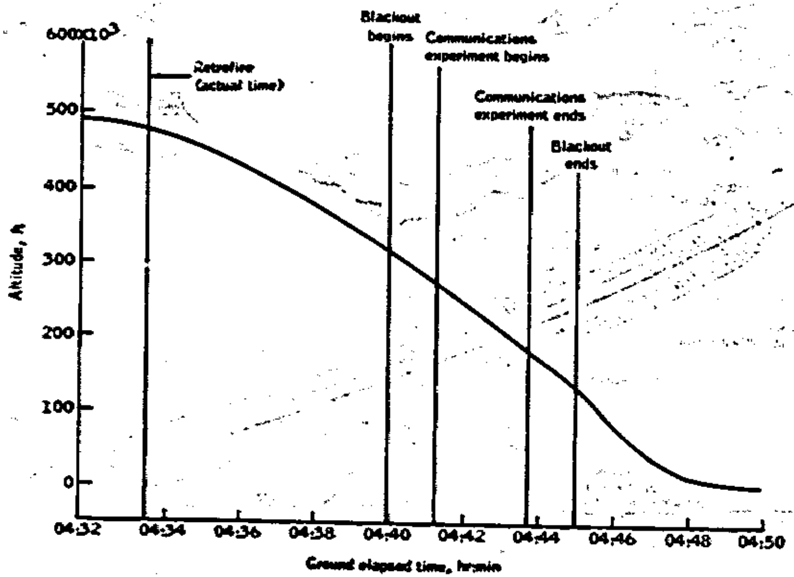


Figure 8-8. -Reentry altitude plotted against time

NASA-S-65-3568

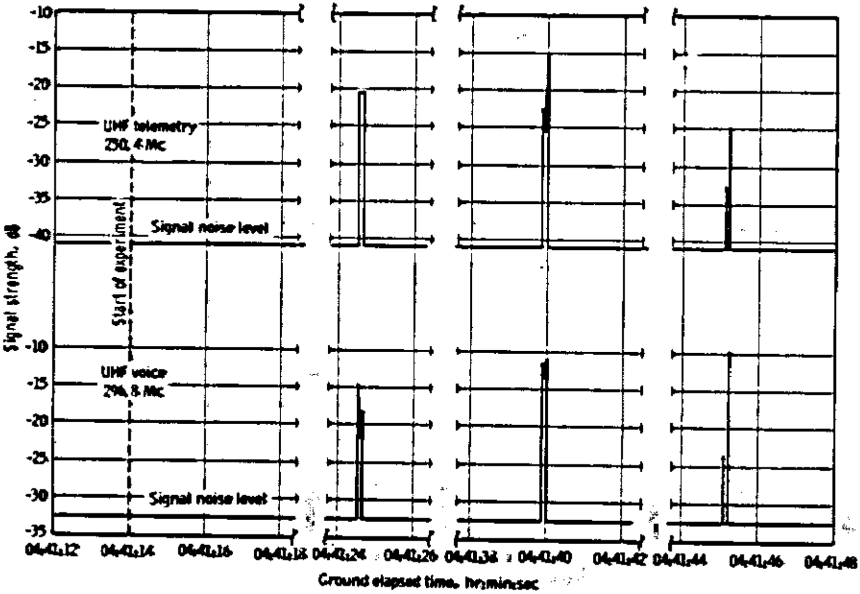


Figure 8-9. - Signal strength plotted against time, LRC Key West Station, Florida for 7.25 lb/sec flow rate

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9.0 CONCLUSIONS

The performance of the spacecraft, launch vehicle, flight crew, and mission support was satisfactory for the GT-3 mission. The objectives of the mission were met with minor exceptions. Operational characteristics in launch, orbit, and reentry space-flight environments were identified for incorporation into future mission planning.

The flight added significant knowledge to the store of information on manned space flight, particularly in the area of precise maneuvering in orbit under the control of the flight crew. A large amount of information was obtained concerning the response of trained flight crew members in a space environment, and assurance was provided that man can be committed to longer duration missions in the Gemini spacecraft.

Evaluation of the data derived from the mission yielded the following specific conclusions:

1. The Gemini launch vehicle, as on the previous flight, flew a trajectory which was higher than that programed for the first stage, primarily because the engine thrust was higher than predicted.

2. The spacecraft yaw-left drift, experienced during the first 44 minutes of flight, was caused by thrust from venting of the launch-cooling heat exchanger.

3. The inertial guidance system (IGS) accumulated velocity errors during the 90 seconds prior to second-stage cutoff which were greater than expected, but not large enough to have prevented a safe orbit had switchover taken place at any time during launch.

4. Proper systems management and adequate training were demonstrated by the crew when they analyzed the primary dc-to-dc converter failure and switched to secondary converter, restoring proper operation of the panel instruments within 45 seconds of detecting this failure.

5. The horizon-sensor system exhibited a large number of "loss of track" indications when the spacecraft was within the intended operating attitudes for use of the horizon sensor. The losses were short and no cause for concern on this flight; however, this did show that a decided problem will exist if it is not resolved prior to the longer duration flights.

6. Although the crew observed "roll drift" on the attitude display group (ADG) during the flight, analysis of the results indicates that

the cause of the drift was incomplete alinement during the platform alinement procedure.

7. The relatively small erroneous count-up of the incremental velocity indicators (about 1 ft/sec per minute of computing time) was caused by a constant in the computer which was known to be wrong and which normally nulls out errors due to zero bias of the platform accelerometer.

8. Blood-pressure measurements were not obtained from the command pilot during this flight because late changes in the equipment resulted in a lack of familiarity with the equipment.

9. The operation of the primary oxygen system at 1000 psia, instead of the designed 800 to 910 psia, was a result of the balance between the heat transfer into the vessel and the low oxygen usage rate.

10. The flight crew's operational equipment requires minor improvements, and increased emphasis should be placed on flight-crew training in this area prior to subsequent missions.

11. Both flight crew members were mildly dehydrated after this flight as evidenced by the slight loss of weight, minimal hemoconcentration and subjective thirst. This dehydration may have been caused by fluid redistribution under weightless conditions, fluid loss after landing, an insufficient fluid intake during flight, or a combination of all three.

12. The space suit causes crew discomfort if left on in the spacecraft for 15 or 20 minutes after landing.

13. HF voice and direction-finding communications were not established during several attempts after landing.

14. Strength records indicate that the Tel II antenna tracking was erratic.

15. The spacecraft angle of attack during reentry was considerably less than predicted from wind-tunnel test data. The actual landing point was short of the planned landing point partially because of the reduced lift capability of the spacecraft during reentry. Although the intended landing point was not achieved, analysis has indicated that the guidance and control systems performed as expected, and would probably have controlled the vehicle to the intended landing point within the accuracies expected.

16. The forces exerted on the flight crew during the change from single-point to two-point suspension were in excess of those anticipated by the flight crew.

17. The reentry trajectory utilized was too mild from a heating standpoint to provide a critical evaluation of either the heat shield or the afterbody protection; however, there is evidence that the heat shield has excessive protection for even the most severe reentry which any planned Gemini mission will produce.

18. One of a pair of redundant cartridges which actuate the cabin forward compartment fresh-air door did not fire because of a faulty switch.

19. The crew compartment is acceptable except for minor discrepancies which are listed in the Recommendations (section 10) of this report.

As a result of analysis of data derived from the GT-3 mission and reports by the flight crew, it is concluded that the Gemini spacecraft is suitable for manned space flight of extended duration.

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10.0 RECOMMENDATIONS

The following recommendations are made as a result of the GT-3 mission evaluation:

1. Corrective action should be taken to assure proper operation of the spacecraft inertial measuring unit. Further analysis and testing should be conducted on the inertial measuring unit flown on GT-3 to determine the cause of the excessive velocity error prior to SECO.

2. Tests and analysis should be performed on the horizon sensor system to determine the cause of the numerous unlocks and to provide corrective action, if necessary, prior to GT-4.

3. Procedures for inertial platform alinement should include maintaining the alinement attitude for a minimum of 8 minutes, after the pitch and roll attitude appears alined on the horizon within 1°, for initial offsets no greater than 5°, 10°, and 10° in pitch, yaw, and roll, respectively. This minimum time should be increased to 10 minutes, if the initial offsets are as large as 5°, 10°, and 20° in pitch, yaw, and roll, respectively. In addition, a check of alinement should be made after 90° of orbital flight.

4. The lift-to-drag ratio associated with the GT-3 mission should be used for orbital reentry trajectory calculations until more data are available from subsequent Gemini mission.

5. A review of factors contributing to footprint size and shift should be made to assure an adequate footprint. Error reduction in the determination of footprint size may be required. Three sigma tolerances resulting from present procedures of measuring and controlling applied velocity impulses may be too large.

6. The venting port for the launch-cooling heat exchanger should incorporate means to eliminate the torquing moment on the spacecraft.

7. A study should be made of the HF communications system, including all effects known, to determine if it is satisfactory for long distance communications. In support of this study, special tests should be performed of the ground network prior to GT-4 to determine the adequacy of the receiving and recording capabilities and to standardize the readiness procedures, operational checks and methods, and postflight reporting. In addition, a review should be made of the preflight test procedures for checkout of the spacecraft HF system, including the

antennas. These tests should establish flight readiness of the system as late as possible prior to launch.

8. The flight crew or ground command should turn on the real-time telemetry transmitter at least 20 seconds prior to start of tape playback through the delayed time transmitter to avoid unnecessary data losses.

9. An analysis should be made to determine possible reduction in heat shield thickness.

10. A test program should be conducted to define and evaluate the forces imposed on the flight crew during the repositioning of the spacecraft from the single-point disconnect to the two-point landing attitude. Results of this program should provide the information needed to make any modifications or initiate any flight crew procedures that may be necessary for future flights.

11. Correction of the following problems relative to the crew compartment of the GT-4 vehicle is required:

a. The inadequacy of the flight plan roller display in providing advance information on flight events.

b. The need for interim stowage of bits and pieces of waste such as used food packages and clipped off ends, used note paper, used towels, et cetera.

12. The following problems relative to the crew compartment should be corrected as soon as schedule permits:

a. Inadequate lighting of the flight plan roller.

b. Poor readability of the G.m.t. clock.

c. Lack of visual or aural warning for cabin pressure loss.

d. Poor visibility of the cabin pressure gage.

e. Marking of the space-suit visors by the underside of the hatches.

13. The details of stowage locations and procedures for handling loose equipment should be given greater emphasis prior to flight.

14. Correction of the incompatability between the space suit structural zipper tab and the urine disposal equipment is required prior to the GT-4 mission.

15. Correction of the following food problems is required prior to the GT-4 mission:

a. The tendency of rehydrated juices to seep from the ends of the food containers after they have been used and folded for stowage.

b. The difficulty in heat sealing the food container joints which apparently caused the loss of a germicide pill.

16. Correction of the following problems relative to waste management are required prior to the GT-4 mission:

a. The inadequate method of stowing and disposing of the paper associated with the defecation device.

b. The difficulty in breaking the disinfectant bag in the defecation device.

17. The following procedures should be adopted for subsequent missions:

a. The requirement to drink water should be called out at regular intervals in the flight plan and reports of water consumption should be made to the aeromedical flight controllers because subsequent missions will last over 1 day.

b. Upon landing, the flight crew members should remove their spacesuits immediately after determining a safe condition of the spacecraft, unless the touchdown point is within 20 minutes of a recovery ship.

c. The establishment of retrograde maneuvers in future mission planning must take into consideration the reports submitted by the flight crew members concerning the effects of glare from the rocket plumes on eye adaptation and horizon visibility.

18. Gemini mission simulator exercises should include realistic problems such as velocity error during launch, accelerometer bias errors during orbital maneuvers, and variations in lift-to-drag ratio of the spacecraft.

19. The launch vehicle engine model should be updated to account for the known biases as determined from the GT-2 and GT-3 flights.

20. The launch vehicle pitch program should be corrected as a result of the higher than expected engine thrust on GT-2 and GT-3.

21. An investigation should be conducted by the ETR to determine the cause of and to correct the erratic tracking of Tel II radar as noted on both the GT-2 and GT-3 missions.

11.0 REFERENCES

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2. McDonnell Aircraft Corp.: NASA Project Gemini Familiarization Manual. SEDR 300, Mar. 15, 1964 (revised).

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4. International Business Machines: Gemini Reentry Math Flow 3 and Math Flow Low Mod I and Mod II Description. IBM, 64-554-0044, Sept. 25, 1964.

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7. McDonnell Aircraft Corp.: Gemini Spacecraft Post-Flight RCS Deactivation. SEDR F-399, Mar. 18, 1965.

8. Catterson, A. D.; McCutcheon, E. P.; Minners, H. A.; and Pollard, R. A.: Aeromedical Observations. Mercury Project Summary Including Results of the Fourth Manned Orbital Flight May 15 and 16, 1963. NASA SP-45, Supt. Doc., U.S. Government Printing Office (Washington, D.C.), pp. 299-326.

9. McDonnell Aircraft Corp.: Project Gemini Postflight Inspection and Cleanup, Spacecraft 3. SEDR F498-3, 1965.

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12.0 APPENDIX

12.1 VEHICLE HISTORIES

12.1.1 Spacecraft Histories

Spacecraft histories at McDonnell Aircraft Corporation, St. Louis, Missouri, are shown in figures 12-1 and 12-2, and at Cape Kennedy in figures 12-3 and 12-4. Figures 12-1 and 12-3 are summaries of activities with emphasis on spacecraft systems testing and prelaunch preparation. Figures 12-2 and 12-4 are summaries of significant, concurrent problem areas.

12.1.2 Gemini Launch Vehicle Histories

Gemini launch vehicle (GLV) histories at Martin Company Denver Division, Denver, Colorado, Martin Company Baltimore Division, Baltimore, Maryland, and Cape Kennedy, Florida are shown in figures 12-5 and 12-6. Concurrent problem areas and significant manufacturing activities are shown with the GLV test and prelaunch preparation activities.

12.2 WEATHER CONDITIONS

Weather conditions at dawn on launch day did not appear encouraging with overcast-to-broken clouds over Cape Kennedy. As the count progressed, the clouds became broken-to-scattered, and at T-0 the Cape area was almost clear with scattered cumulus clouds to the east and west. Weather observations in the launch area taken at 14:50 G.m.t. were as follows:

Cloud coverage	Scattered at 11 000 feet
Wind direction, deg	150
Wind velocity, knots	14
Visibility, miles	10
Pressure, in. Hg	30.03
Temperature, °F	73
Dew point, °F	64
Relative humidity, percent	74

Weather observations taken on board the U.S. Coast Guard cutter Diligence located at 22.4° N, 70.6° W at 16:00 G.m.t. were:

Cloud coverage	3/8 covered, cumulus; cloud base 2000 feet; middle layer of alto-cumulus clouds at 10 000 feet.
Wind direction, deg	90
Wind velocity, knots	20
Visibility, miles	10
Pressure, in. Hg	30.17
Temperature, °F	78
Dew point, °F	72
Relative humidity, percent	83
Wave height, feet	6 to 7

The variation of launch-area and recovery-area wind direction and velocity with altitude is presented in figures 12-7 and 12-8. Tables 12-I and 12-II list the atmospheric conditions measured in the launch area and recovery area for various altitudes.

12.3 FLIGHT SAFETY REVIEWS

The flight readiness of the spacecraft and launch vehicle for the GT-3 mission, as well as the readiness of all supporting elements, was determined at Flight Safety and Mission Review meetings.

12.3.1 Spacecraft

12.3.1.1 Flight Readiness Review.- The spacecraft Flight Readiness Review was held on March 10, 1965. During the review of the guidance and control system, the Board asked that a failure analysis be performed on the H-15 platform, and that the vibration qualification testing be completed on the platform end-cap modification prior to the Mission Review. The contractor was asked to locate all the dummy pyrotechnic cartridges that had been used on spacecraft 3, and to investigate the moisture problem experienced in the mild detonating fuse (MDF) used to release the survival kit. These requests were complied with and satisfactory answers were received. All other systems and the spacecraft were found ready for flight, pending completion of the final simulated flight test. This test was successfully conducted on March 18, 1965.

12.3.2 Launch Vehicle

12.3.2.1 Flight Readiness Review.- The launch vehicle Flight Readiness Review was held on March 20, 1965. The Air Force Space Systems Division 6555th Aerospace Test Wing personnel summarized the testing and operations accomplished at the Air Force Eastern Test Range on this launch vehicle, and all systems were found ready for flight.

12.3.3 Mission Review

The GT-3 Mission Director convened the Mission Review Board on March 20, 1965. The status of all elements were reviewed and found in readiness for support of the launch.

12.3.4 Flight Safety Review Board

The Air Force Flight Safety Review Board met on March 22, 1965, and since all systems were ready, recommended to the Mission Director that the launch vehicle be committed to flight.

12.4 SUPPLEMENTAL REPORTS

Supplemental reports for the GT-3 mission are listed in table 12-III. The format will conform to the external distribution format of the NASA or contractor organization preparing the report. Each report will be identified on the title page as being a GT-3 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.

12.5 DATA AVAILABILITY

Tables 12-IV to 12-VI list the mission data which are available for evaluation. The instrumentation and trajectory 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 Division.

12.6 POSTFLIGHT INSPECTION

The postflight inspection of the spacecraft 3 reentry assembly was scheduled to be conducted in accordance with reference 9 at the Kennedy Space Center (KSC) from March 26 to April 16, 1965. The reentry assembly was received in good condition. Neither the rendezvous and recovery (R and R) section nor the parachutes were recovered. Exterior views of the reentry assembly are shown in figures 12-9 and 12-10. The following list contains the discrepancies noted during the detailed inspection of the reentry assembly:

(a) All electrical fuse blocks contained water. (See section 5.1.7.)

(b) Several RCS thrusters had peripheral cracks. (See section 5.1.8.2.)

(c) The horizon-scanner electrical-receptacle housings were corroded. (See section 5.1.1.4.)

(d) The phenolic ring aft of the hoist loop at Z104 was crushed. (See section 5.1.1.4.)

(e) The HF antenna was severed under the sealing boot. (See section 5.1.2.2.)

(f) Six of the fuses in the fuse block holders had been blown. (See section 5.1.7.)

(g) One cartridge of the fresh-air door had not detonated. (See section 5.1.9.)

(h) The electrical connector to the left-hand MDF detonator at Z192 had the bayonet pins sheared off and was hanging loose from the cartridge. (See section 5.1.9.)

(i) The HF antenna contained approximately 160 gm of water. (See section 5.1.2.2.)

(j) Various shingle washers and screws were loose. (See section 5.1.1.4.)

(k) Washers and screws were missing from a strip forward of the right-hand hatch window. (See section 5.1.1.4.)

[redacted]

(l) Two screws were loose and the right-hand C-band antenna was loose and recessed. (See section 5.1.1.4.)

(m) The rubber umbilical seal had evidence of a local hot-spot in the upper left-hand corner. (See section 5.1.1.4.)

(n) The shingle filler strip was broken and torn loose at the aft edge of the right-hand landing-gear door. (See section 5.1.1.4.)

12.6.1 Spacecraft Systems

12.6.1.1 Structure.- The structure of spacecraft 3 exhibited no unexpected condition. The effects of reentry heating were noted by a slight discoloration behind the lower right spacecraft-to-adapter tie. (See fig 12-11.) The heating effect was much less severe than was noted on spacecraft 2, and less heating was further evidenced by less discoloration of the insulation blankets under the external shingles (see fig. 12-12). (This condition was as expected.)

The heat shield was in good condition, although it appeared to have a whiter cast than did the spacecraft 2 heat shield, probably a result of less heating. The stagnation point was determined to be between 9 and 10 inches from the centerline as compared to 24.5 inches for spacecraft 2.

The heat shield was removed, weighed, and dried in the altitude chamber. The weight prior to drying was 336.90 pounds and after drying was 327.15 pounds. After drying, there were six 4-inch diameter cores removed for analysis. The two heat-shield thermocouples were contained in two of the removed core samples. Two of the core samples were sent to McDonnell Aircraft Corp., St. Louis, Missouri, and the remaining four were sent to NASA-MSC, Houston.

The left-hand hatch was cycled several times to check the latching mechanism postflight. The hatch operated normally.

The right-hand hatch window assembly was removed and sent to NASA-MSC, Houston, for investigation of optical properties and meteoroid impact.

All spacecraft insulation blankets were removed and the spacecraft was weighed. The pitch and yaw center-of-gravity locations were determined also.

The phenolic ring aft of the hoist loop at Z104 was crushed, probably during the recovery operation. (See section 5.1.1.4.)

[redacted]

Various shingle washers and screws were loose, and washers and screws were missing from a strip forward of the right-hand hatch window. The shingle filler strip at the aft edge of the right-hand landing-gear door was broken and torn loose. (See section 5.1.1.4.)

The equipment bays sustained corrosion much the same as was reported for spacecraft 2, although the corrosion on spacecraft 3 perhaps was more severe.

12.6.1.2 Environmental control system (ECS).- After removal from the ECS package, the lithium-hydroxide canister was weighed (34.2 pounds) and sent to NASA-MSC, Houston, for analysis. The demand regulators were sent to the contractor. The ECS system was serviced in accordance with reference 9. After a sample of the drinking water had been obtained and sent to NASA-MSC, Houston, the remaining water was drained from the tank. No corrosion was noted on the ECS door seal as had been evident on spacecraft 2. There was no water in the interior of the ECS cavity.

12.6.1.3 Communications.- The external appearance of all the communication equipment located in the equipment bays was good and exhibited little evidence of corrosion. The HF antenna had been severed in the boot area and approximately 160 gm of salt water was in the HF antenna package. The C-band antennas located near the small pressure bulkhead appeared to be depressed below the external surface and contained two loose screws.

The HF antenna was removed for further analysis. The spacecraft circuitry for the HF antenna extend-retract function was checked and found to be satisfactory.

12.6.1.4 Guidance and control system.- Although the external appearance of this equipment was good, the computer apparently suffered damage from corrosion. This was determined by noting corrosion around the elapsed time meter. The damage is believed to be extensive since the vent holes would allow free entry of sea water. This equipment, as well as the horizon-sensor electronic packages, was removed and cleaned.

The platform, electronics package, and computer were returned to the contractor for further analysis.

The two horizon sensor electronic packages contained 1 to 2 ounces of salt water each. These units were returned to the contractor for further analysis.

12.6.1.5 Pyrotechnics.- Pyrotechnic resistance checks were performed on all actuated pyrotechnic cartridges, and eight pyrotechnic cartridges registered resistance values. The remaining cartridges indicated open circuits. One of the redundant fresh-air door cartridges was not detonated and was sent to the malfunction analysis laboratory of Kennedy Space Center (KSC). The spacecraft circuitry from the crew station to the fresh-air door was checked by firing high energy simulated squib (HESS) and found to be satisfactory.

The hatch actuators, rocket catapults, and seat pyrotechnic devices were removed and sent to the KSC pyrotechnics group for storage.

The electrical connector on the R and R section jettison left-hand detonator at 2192 was hanging loose. The three bayonet pins on the detonator had been sheared, allowing the electrical connector to become disengaged. The detonator, detonator block, and electrical connector were removed and sent to the malfunction analysis laboratory at KSC. The postflight visual inspection of the wire bundle guillotines, bridle release mechanisms, and other pyrotechnic devices disclosed that all appeared to have functioned normally.

12.6.1.6 Instrumentation and recording system.- The shingle with the FD08 thermocouple was removed and visually inspected. An electrical continuity check was conducted on the thermocouple and spacecraft wiring to the first connector. The thermocouple and spacecraft wiring were found to be satisfactory. The primary dc-to-dc converter was removed and sent to the contractor for failure analysis. The spacecraft circuitry to the primary dc-to-dc converter was rechecked and found to be satisfactory. The CO_2 partial pressure transducer was removed and sent to the contractor for calibration checks. The PCM tape recorder was removed and sent to the contractor for data recovery. The two telemetry transmitters and programer did not appear to be adversely affected by the salt water immersion.

The voice tape recorder was removed and sent to the contractor for refurbishment and reuse.

12.6.1.7 Electrical system.- All electrical fuse blocks were opened and found to contain water. Six of the fuses in the fuse block holders had been blown (fig. 12-13). The primary and secondary horizon-sensor electrical-receptacle housings were corroded. The rubber umbilical seal had a local hot spot in the upper left-hand corner (fig. 12-14). The main and squib batteries were removed and the discharge was controlled from 20 V to 20.5 V. The current leakage due to salt water immersion was checked and recorded in reference 9. The batteries and their terminals appeared unaffected by corrosion, the battery retainer straps were not corroded like the straps on spacecraft 2, and the terminal straps appeared to be in good shape with no excessive corrosion evident.

12.6.1.8 Crew station furnishings and equipment.- The crew station appeared as expected and switch positions were photographed, checked, and recorded in reference 9.

The five ECS controls located on the overhead panel were cycled, and they all operated normally.

12.6.1.9 Propulsion system. - The reentry control system (RCS) section appeared normal. The only notable fact was that the thrust chamber assemblies on the bottom side of the spacecraft (A and B rings) were charred on all external surfaces with some delamination, and the potting that restrains the assembly in the steel cans had been subjected to temperatures that caused the material to flow. The assemblies on the top side were not as severely affected.

12.6.1.10 Landing system. - Neither the R and R section with the attached drogue and pilot parachutes nor the main parachute was recovered. The single-point bridle-release mechanism and the main parachute forward and aft bridle-release mechanisms appeared to have functioned normally.

12.6.1.11 Postlanding recovery aids. - The phenolic ring aft of the hoist loop at Z104 was crushed. The flashing recovery-light bulb was received as a loose piece.

12.6.1.12 Experiments. - The T-1 Reentry Communications experiment, mounted on the right-hand landing-gear door (fig. 12-15) was removed and deserviced. The backup experiment door was deserviced, and the flight shingles were mounted on the backup door. The flight experiment door was sent to the experimenter at the Langley Research Center. The backup experiment door was installed on the reentry section.

The S-2 sea urchin egg growth and the S-4 radiation and zero g effects on blood experiment containers were removed from the hatches and sent to NASA-MSC, Houston.

12.6.2 Continuing Evaluation

The following is a list of the approved Spacecraft Test Requests (STR's) for the postflight evaluation:

Number	System	Purpose
3000	Communications - HF antenna	Determine if the HF antenna configuration is acceptable for subsequent flights. (See section 5.1.2.2.)
3001	Structure - heat shield	Determine reentry effects on the heat shield. (See section 5.1.1.2.4.)

Number	System	Purpose
3002	ECS - lithium-hydroxide canister	Analyze the lithium hydroxide and charcoal after the mission.
3003	Structure - right-hand window assembly	Examine optically the right-hand window. (See section 5.1.1.2.4.)
3005	Propulsion system - RCS	Conduct postflight evaluation of the RCS A and B rings.
3006	Crew station	Remove the flight crew equipment for evaluation and assessment for reuse.
3007	Voice tape recorder and shock absorber mount	Refurbish the voice tape recorder for use on subsequent spacecraft.
3008	Voice tape recorder cartridges	Refurbish the five voice-recorder cartridges for reuse on subsequent spacecraft.
3011	Communications - HF voice	Perform a failure analysis. (See section 5.1.2.2.)
3012	Structure - left-hand window	Determine if ultraviolet optical coating is on inside of outer window pane. (See section 5.1.1.2.4.)
3013	Guidance and control	Investigate ascent and orbit IGS anomalies. (See section 5.1.5.4.1.)
3014	Pyrotechnics	Dispose of the normally unfired pyrotechnics.
3015	Structures	Determine the cause of loose and missing shingle bolts. (See section 5.1.1.4.)
3016	Structure - left-hand and right-hand window assemblies	Further investigate the window fogging and moisture anomaly. (See section 5.1.1.2.4.)
3500A	ECS - demand regulators	Perform a failure analysis of the demand regulator.
3502	ECS - drinking water	Determine adequacy of drinking water system sterilization.

Number	System	Purpose
3503	Instrumentation - temperature thermocouple	Troubleshoot thermocouple malfunction. (See section 5.1.1.2.1.)
3504	Instrumentation - primary dc-to-dc converter	Perform a failure analysis. (See section 5.1.3.)
3506	Instrumentation - CO_2 partial pressure transducer	Check the calibration of the CO_2 partial pressure transducer.
3507	Guidance and control	Remove guidance and control equipment for evaluation.
3510	Propulsion (RCS)	Return the RCS section to the contractor for utilization as a relief-valve test bed.
3511	Propulsion	Investigate the cause of oxidizer vapor reported by flight crew after landing.
3512	Ejection seats ballute deployment and release	Investigate the ballute deployment and release mechanism on the left-hand seat.
3514	Structure	Obtain weight and c.g. at reentry. (See section 5.1.1.3.)
3515	Structure	Remove the Rene' shingles and associated blankets.
3516	Pyrotechnic - Z192 MDF ring	Investigate failure mode of the left-hand MDF detonator at Z192. (See section 5.1.9.)
3517	Structure - heat-shield surface	Locate the region of stagnation-point heating as defined by the streaked flow pattern on the ablative surface. (See section 5.1.1.2.3.)
3519	Communications - HF antenna	Determine why the HF antenna did not retract prior to flight crew egress. (See section 5.1.2.2.)

Number	System	Purpose
3520	Pyrotechnic fresh-air door release	Investigate postflight resistance readings. (See section 5.1.9.)
3521	Ejection seats	Provide temporary assignment of spacecraft 3 ejection seats to spacecraft 4 through WMSL and static fire (approximately May 15, 1965).
3522	Guidance and control - horizon sensor electronics	Investigate the reason for intermittent track loss. (See section 5.1.5.4.2.)
3523	Electrical fuse blocks	Determine the cause of leakage of water into electrical fuse blocks. (See section 5.1.7.)
3524	Pyrotechnics - fresh-air door	Investigate failure of fresh-air door cartridge to fire. (See section 5.1.9.)
3525	Electrical - fresh-air door	Determine if spacecraft wiring caused the cabin-air inlet squib not to detonate in flight. (See section 5.1.9.)
3526	Instrumentation - PCM tape recorder	Recover data lost from the onboard PCM tape recorder due to vibration on re-entry.
3527	ECS	Conduct manned sea-level test on ECS and G4C suits in order to simulate and investigate suit flutter phenomenon experienced on spacecraft 4 in St. Louis.
3528	Structure - left-hand cabin window	Determine if space environment causes film to appear between glass panes of inner window. (See section 5.1.1.2.4.)
3529	Propulsion system RCS A ring	Determine postflight cause of RCS TCA no. 2 oxidizer valve sticking in open position, as determined from postflight data. (See section 5.1.8.2.3.)
3530	Communications - HF antenna	Return HF antenna to contractor. (See section 5.1.2.2.)
3531	Pyrotechnic - fresh-air door cartridge	Investigate fresh-air door cartridge circuitry. (See section 5.1.9.)

Number	System	Purpose
3533	Pyrotechnics - recovery	Remove drogue parachute and ballute aneroids.
3534	Lighting system (52-79378)	Perform a failure analysis of diode which resulted in a press-to-test malfunction indication. (See section 5.1.7.)
3535	Control box-elect; propulsion system and RCS decontamination	Determine cause of inadvertent static fire of the RCS during post-launch recovery operations at Mayport, Florida. (See section 5.1.8.2.3.)

TABLE 12-I.- LAUNCH-AREA ATMOSPHERIC CONDITIONS
AT 14:39 G.m.t., MARCH 23, 1965

Altitude, ft	Temperature, °F	Pressure, lb/sq ft	Density, slugs/cu ft
0×10^3	73.2	2137.8	2318.7×10^{-6}
5	52.5	1789.7	2025.9
10	36.3	1488.1	1742.6
15	25.5	1231.2	1477.3
20	7.7	1011.7	1261.2
25	-8.0	825.8	1065.2
30	-28.8	668.3	903.8
35	-51.2	534.9	763.1
40	-71.1	423.3	634.9
45	-88.1	331.0	519.0
50	-91.3	257.7	407.7
55	-102.5	199.2	325.2
60	-98.5	153.5	247.6
65	-85.5	119.3	185.9
70	-70.1	93.6	140.1
75	-60.5	73.9	107.9
80	-62.3	58.5	85.8
85	-60.2	46.4	67.5
90	-52.8	36.8	52.8
95	-49.4	29.2	41.7
100	-46.5	23.4	33.0
105	-45.0	18.6	26.2

TABLE 12-II.- RECOVERY-AREA ATMOSPHERIC CONDITIONS AT 19:24 G.m.t.
AT GRAND TURK ISLAND, MARCH 23, 1965

Altitude, ft	Temperature, °F	Pressure, lb/sq ft	Density, slugs/cu ft
0×10^3	81.5	2128.2	2270.5×10^{-6}
5	56.3	1785.1	2007.2
10	46.0	1487.0	1711.3
15	32.7	1233.7	1458.9
20	18.9	1018.4	1239.7
25	-1.3	834.2	1060.0
30	-25.8	676.7	908.6
35	-45.6	542.8	763.7
40	-67.2	430.7	639.1
45	-81.9	338.1	521.6
50	-98.0	262.5	423.0
55	-108.9	276.0	335.7
60	-103.9	155.2	254.2
65	-92.2	119.9	190.0
70	-76.5	93.4	142.0
75	-67.7	73.5	109.4
80	-62.7	58.1	85.2
85	-55.3	46.2	66.4
90	-57.3	36.5	53.0
95	-44.5	29.0	40.9
100	-40.4	23.2	32.4
105	-34.6	18.6	25.6

TABLE 12-III.- SUPPLEMENTAL REPORTS

Number	Report title	Responsible organization	Completion date	Text reference section and/or remarks
1	GLV Engineering Evaluation Report (GT-3)	SSD and contractor (Aerospace)	May 22, 1965	Section 5.2 Standing requirement
2	Launch Vehicle No. 3 Flight Evaluation	SSD and contractor (Martin)	May 7, 1965	Section 5.2 Standing requirement
3	Manned Space Flight Network Performance for the Third Gemini Mission	Goddard Space Flight Center	May 22, 1965	Section 6.3 Standing requirement
4	GT-3 Spacecraft Inertial Guidance System Evaluation	Space Technology Laboratory	May 7, 1965	Section 5.1.5 Standing requirement
5	Transcription of GT-3 Air-to-Ground and Onboard Voice Recorder	Flight Crew Operations Directorate	April 23, 1965	NA
6	Transcription of GT-3 Systems Debriefing	Mission Evaluation Team	May 3, 1965	NA
7	Evaluation of Velocity Error in IMU During Ascent Phase	McDonnell and Honeywell	May 21, 1965	Section 5.1.5
8	Evaluation of Horizon Sensor Unlock Anomaly	McDonnell and ATL	May 21, 1965	Section 5.1.5
9	Evaluation and Test Report of Changing from Single-Point Suspension to Two-Point Suspension	McDonnell	May 21, 1965	Section 5.1.11

TABLE 12-IV.- INSTRUMENTATION DATA AVAILABILITY

Data description

Paper recordings

- Telemetry signal-strength recordings
- Air-to-ground voice signal-strength recordings
- MCC plotboards (Confidential)
- Range safety plotboards (Confidential)

Radar data (Confidential)

- IP-3600 trajectory data
- MISTRAM
  - Natural coordinate system
  - Final reduced
- C-band
  - Natural coordinate system
  - Final reduced
- Trajectory data processed at GSFC (launch and orbital)

Voice transcripts (Confidential)

- Air-to-ground and onboard recorder
- Technical debriefing (on recovery ship)
- Systems debriefing
- Scientific debriefing

GLV reduced telemetry data (Confidential)

- Engineering units versus time plots
- Vibration
  - Power spectrum density plots
  - g_{rms} plots

Spacecraft reduced telemetry data

- Engineering units versus time
- Ascent phase (plots and tabulations)
  - System parameters (excluding G and C)
  - G and C system parameters (Confidential)
- Orbital phase (plots)
  - System parameters (excluding G and C)
  - G and C system parameters (Confidential)
- Reentry (plots and tabulations)
  - System parameters (excluding G and C)
  - G and C system parameters (Confidential)

Special computation

- Computed values versus time (plots and tabulations)
- Heat transfer rates
- Ascent angle of attack
- Reentry lift-to-drag ratios and angle of attack
- Propellant weight and reentry control system pressurant leakage
- Spacecraft positions and special parameters from IGS (Confidential)
- Transformation of body accelerations to IGS coordinates (reentry)
- IGS simulation (ascent and reentry)
- G and C analysis data (comparison of IGS telemetry data with tracking data - ascent and reentry)

TABLE 12-V.- SUMMARY OF PHOTOGRAPHIC DATA AVAILABILITY

Mission phase	Number of still photographs	Motion picture film, footage
Launch and prelaunch	18	11800[a]
Recovery		800
Swimmer deployment and installation of collar	56	
Egress of flight crew	60	
Aircraft carrier inspection		2000
Loading of spacecraft on Intrepid	35	
General inspection of spacecraft	55	
Hatch inspection [redacted]	6	
RCS inspection [redacted]	15	
Interior of spacecraft	15	
Mayport, Florida [redacted]		1100
Loading and unloading activities	48	
RCS deactivation	60	
Cape Kennedy postflight inspection		
Exterior views of spacecraft		
Heat shield		
Outer and inner skin [redacted]		
RCS inspection [redacted]		
Interior inspection [redacted]		
R and R inspection [redacted]		
ECS compartment [redacted]		
Miscellaneous [redacted]		
Onboard spacecraft	25	

[a] Table 12-III contains a detailed description of launch engineering sequential and metric photographic coverage.

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TABLE 12-VI.- LAUNCH PHASE ENGINEERING SEQUENTIAL CAMERA DATA AVAILABILITY

Operations Requirements Document - 3600		Sequential film coverage item (a)	Camera					
Page	Item		Type	Size	Lens	Speed, frames/sec	Location	Presentation
71	1	L.2-30 L.2-31 L.2-32	Tracking	16-mm	40 in.	96	Cape Kennedy	Spacecraft centered in frame from lift-off to loss of vehicle.
71	2	L.2-27 L.2-28	Fixed	16-mm	40-mm	400	Complex 19	Spacecraft upper and lower umbilical plugs showing disconnect.
71	3	L.2-12 L.2-13	Fixed	16-mm	100-mm	200	Complex 19	Spacecraft centered in bottom of frame.
72	4	L.2-30 L.2-32	Tracking	16-mm	40 in.	96	Cape Kennedy	Spacecraft centered in frame from lift-off to loss of vehicle.
72	1	L.2-7 L.2-8	Fixed	16-mm	15-mm	24	Complex 19	Fuel-storage tanks to show possible leakage or spillage in the area.
73	2	L.2-4 L.2-5 L.2-6	Fixed	16-mm	15-mm 15-mm 25-mm	24	Complex 19	General surveillance of space vehicle, launcher, and launcher stand.
73	3	L.2-1 L.2-2 L.2-3	Fixed	16-mm	15-mm 15-mm 25-mm	24	Complex 19	Space vehicle, launcher, and launcher stand centered in frame. Cameras remotely operated by Test Conductor in case of an emergency.
74	4	L.2-20	Fixed	16-mm	10-mm	400	Complex 19	3D1E and 3D2E umbilical plugs.
		L.2-21						2DFVT umbilical plug.
		L.2-22						1DOVT umbilical plug.
		L.2-23						3B1E and associated umbilical plugs.
		L.2-24						3B2E and associated umbilical plugs.
		L.2-25						Cable cutters.

[a] Data listed by item number in AFETR Operations Directive 3600 for GT-3.

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TABLE 12-VI.- LAUNCH PHASE ENGINEERING SEQUENTIAL CAMERA DATA AVAILABILITY - Continued

Operations Requirements Document - 3600		Sequential film coverage	Camera					
Page	Item	Item (a)	Type	Size	Lens	Speed, frames/sec	Location	Presentation
74	5	1.2-28	Fixed	16-mm	10-mm	400	Complex 19	End of umbilical boom 3 to observe J-Bars and lanyards during launch.
		1.2-29			152-mm	400		Umbilical booms 3 and 4 to show umbilical and lanyard action following umbilical release.
74	6	1.2-14 1.2-15	Fixed	16-mm	13-mm 152-mm	400	Complex 19	Lower portion of space vehicle and A-frames to observe explosive bolt action and space vehicle first motion.
75	7	1.2-18 1.2-19	Fixed	16-mm	13-mm	400	Complex 19	Engine bells centered laterally.
75	8	1.2-16 1.2-17	Fixed	16-mm	10-mm	400	Complex 19	Engine area.
75	9	1.2-9 1.2-10 1.2-11	Fixed	16-mm	25-mm	400	Complex 19	Space vehicle centered in frame to show movement and vibration at launch.
75	10	1.2-33	Tracking	16-mm	20 in.	64	Cape Kennedy	Track from lift-off to loss of vehicle with space vehicle centered in frame throughout track.
		1.2-34	Tracking	16-mm	40 in.	64	Cape Kennedy	Track from lift-off to loss of vehicle with space vehicle centered in frame throughout track; if any components fall from the vehicle during powered flight, track the falling debris.

[a]Data listed by item number in AFETR Operations Directive 3600 for GT-3.

TABLE 12-VI.- LAUNCH PHASE ENGINEERING SEQUENTIAL CAMERA DATA AVAILABILITY - Concluded

Operations Requirements Document - 3600		Sequential film coverage - item (a)	Camera					
Page	Item		Type	Size	Lens	Speed, frames/sec	Location	Presentation
		1.2-35	Tracking	35-mm	120 in.	64	Cape Kennedy	Track from first acquisition to loss of vehicle; engine section centered until I/F ratio allows full space vehicle to be centered.
		1.2-36	Tracking		80 in.	64		
77	12	1.2-37	Tracking (IGOR)	70-mm	180 in.	30	False Cape	Track from first acquisition to loss of vehicle. Engine section centered in frame until I/F ratio allows full space vehicle to be centered to show staging if event is recordable.
		1.2-38	Tracking (ROTI)	70-mm	400 in.	30	Cocoa Beach	
		1.2-39	Tracking (IGOR)	35-mm	360 in.	32	Patrick Air Force Base	
		1.2-40	Tracking (ROTI)	70-mm	500 in.	20	Melbourne Beach	
90	5		Tracking (Airborne)	16-mm 35-mm	24 in. 32 in.	180 80	Cape Kennedy Area	Airborne photographic coverage of the launch sequence from specially configured aircraft for surveillance during the maximum aerodynamic pressure region.
		1.2-42 1.2-43		35-mm 35-mm	40 in. 60 in.	96 96		
135	3	1.2-41	Tracking (EN-7 Image Enhancement System)		500-mm		Patrick Air Force Base	Track first acquisition to loss of vehicle to provide high altitude video coverage for transmission to MCC-Cape.

[a] Data listed by item number in AFETR Operations Directive 3600 for GT-3.

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NASA-S-65-3994

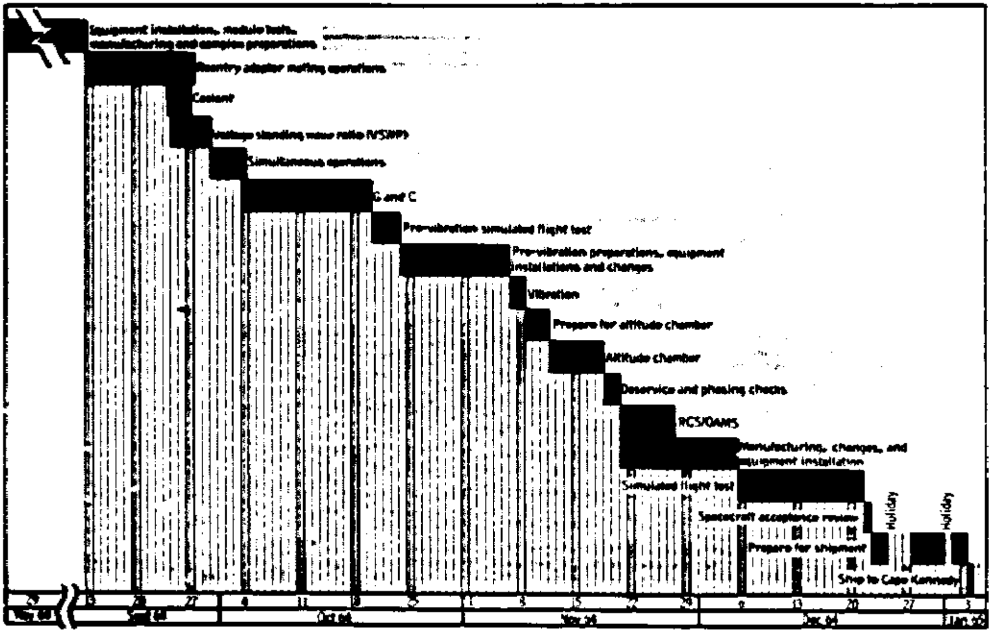

Figure 12-1. - Spacecraft 3 test history at contractor facility.

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NASA-S-65-3760

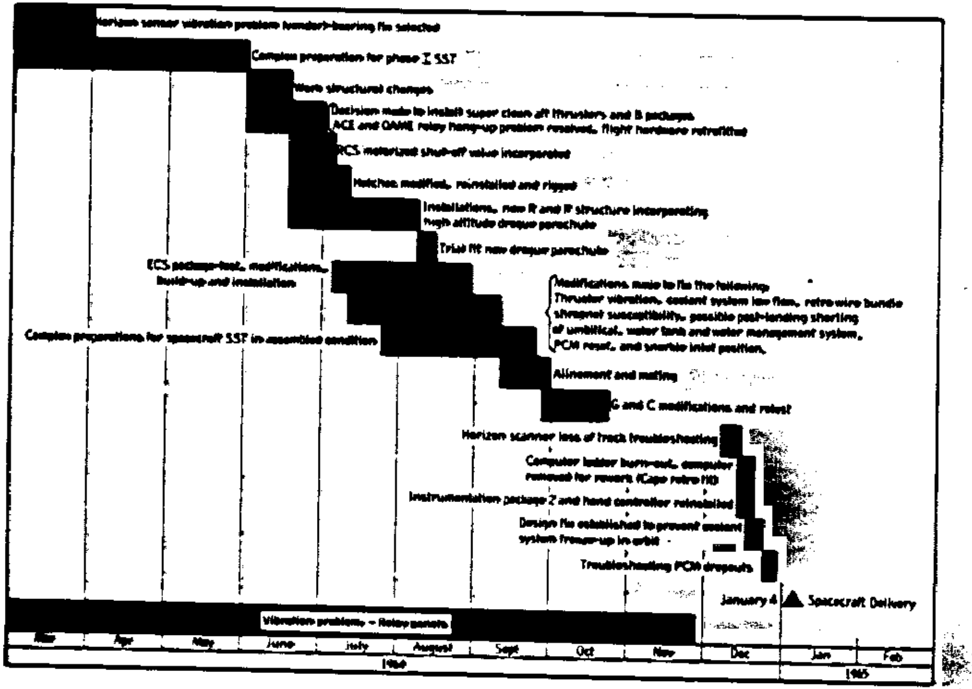

Figure 12-2. - Summary of significant problem areas, spacecraft 3 at contractor facility.

NASA-S-65-[illegible]

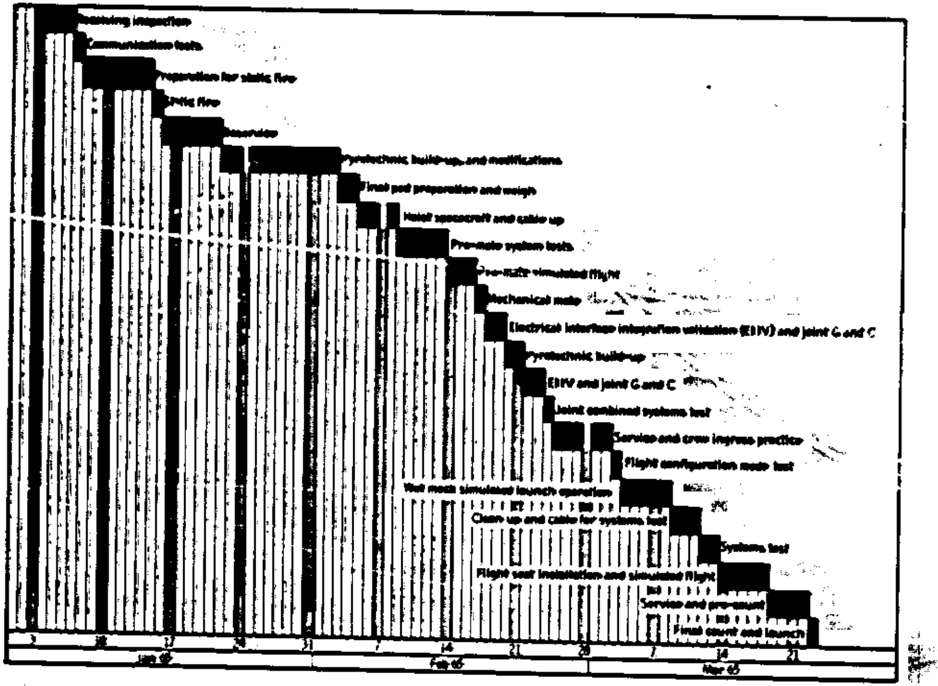

Figure 12-3. - Spacecraft 3 test history at Cape Kennedy [redacted]

NASA-S-65-399

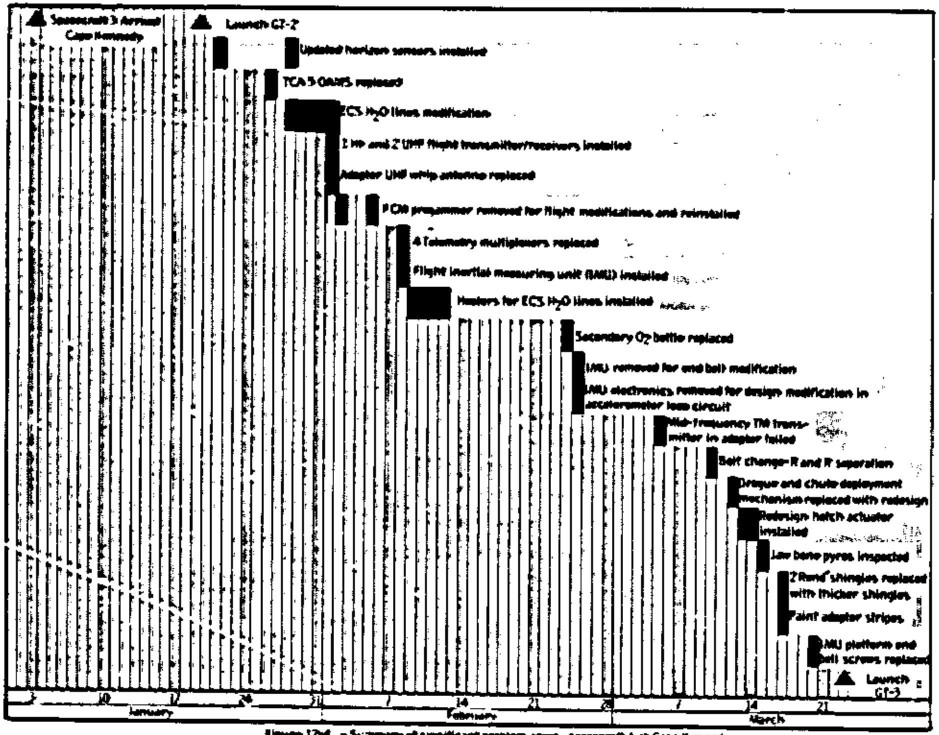

Figure 12-6. - Summary of significant problem areas, Spacecraft 5 at Cape Kennedy.

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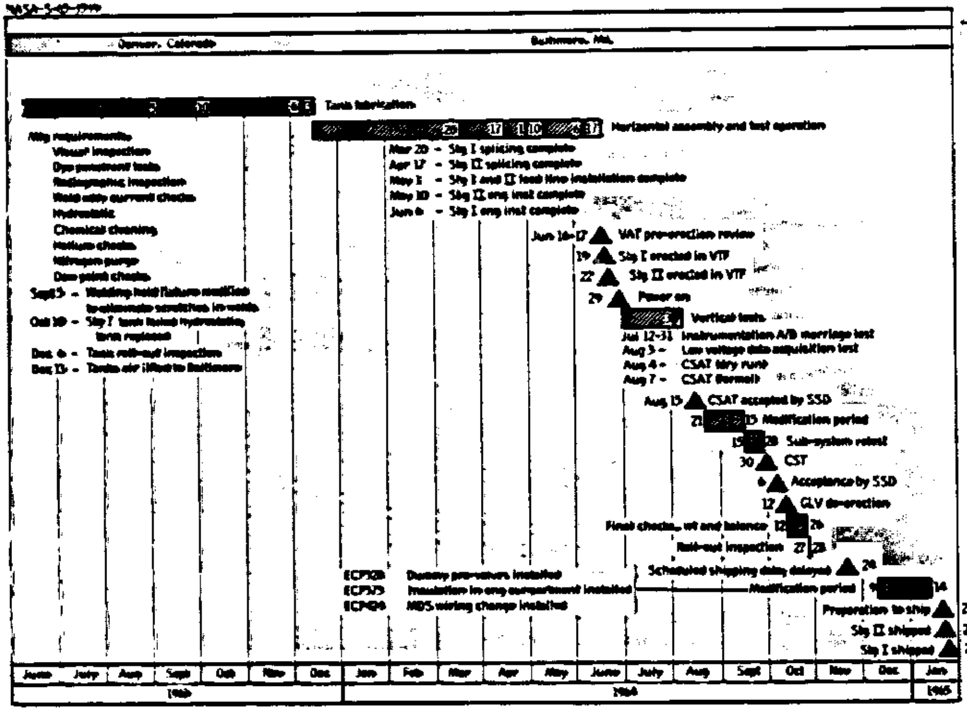

Figure 12-5. - Gemini launch vehicle 3 history at Denver and Baltimore.

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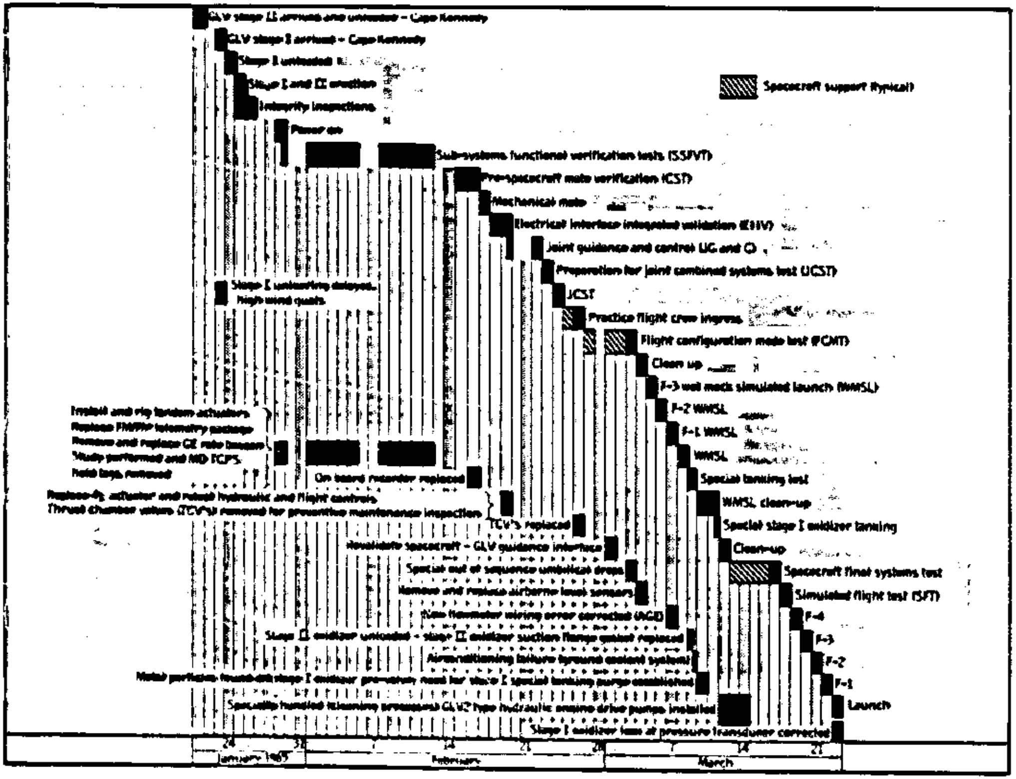

Figure 12-6. - Gemini launch vehicle 3 history at Cape Kennedy.

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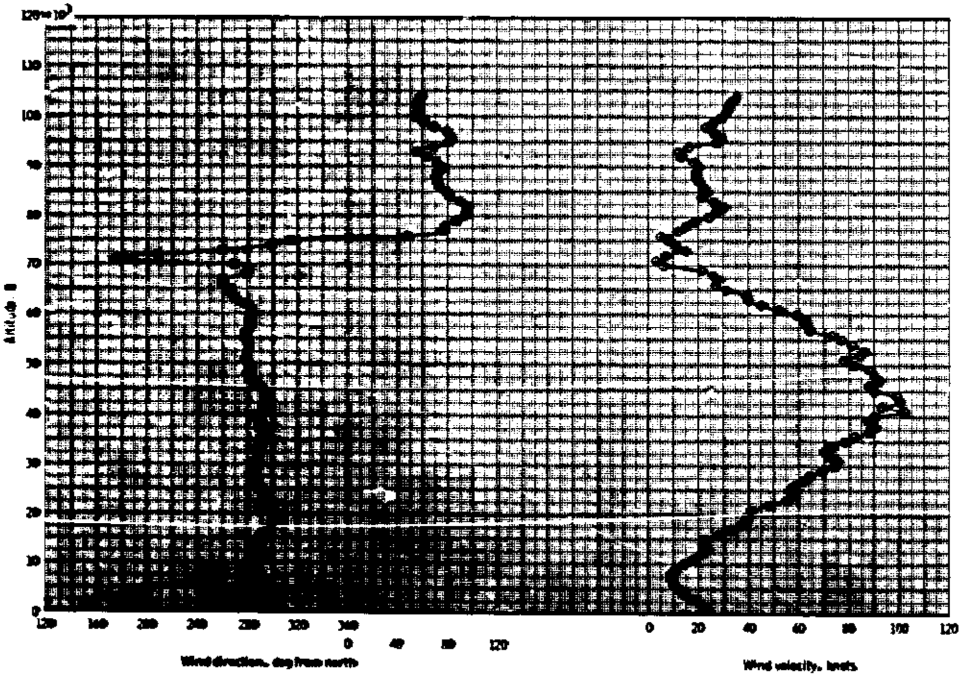

Figure 12-7.- Variation of wind direction and velocity with altitude for launch area.

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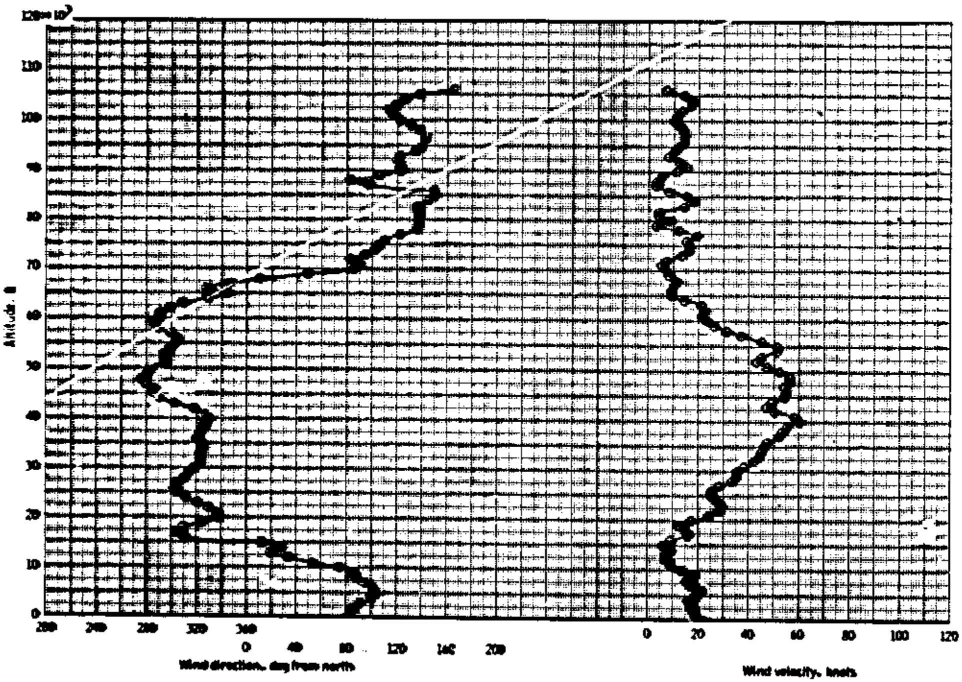

Figure 12-8.- Variation of wind direction and velocity with altitude for recovery area.

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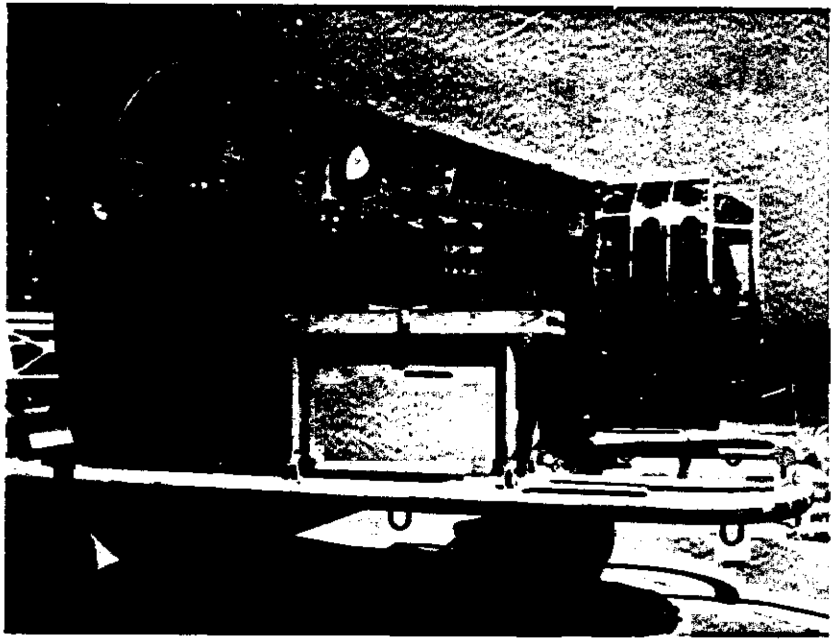

Figure 12-9. - Reentry assembly - after flight.

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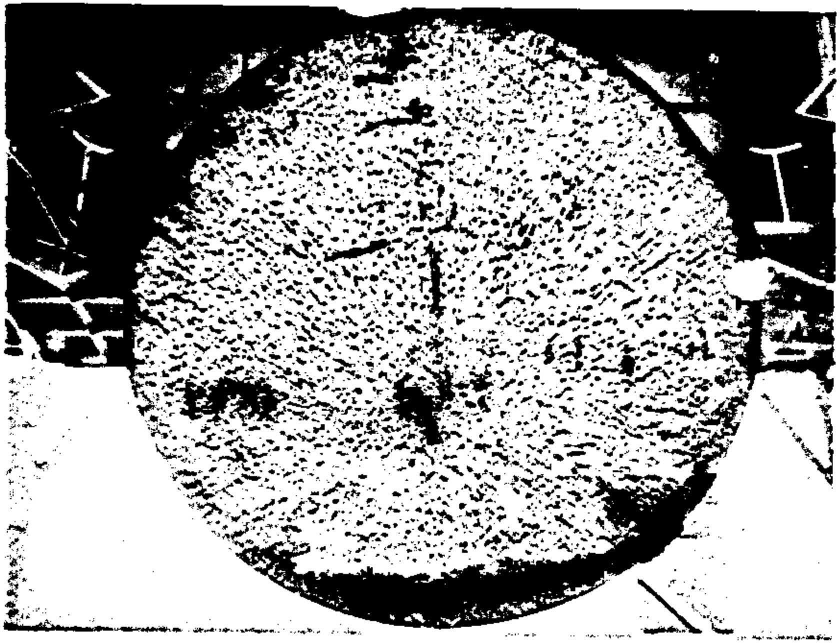

Figure 12-10. - Heat shield - after flight.

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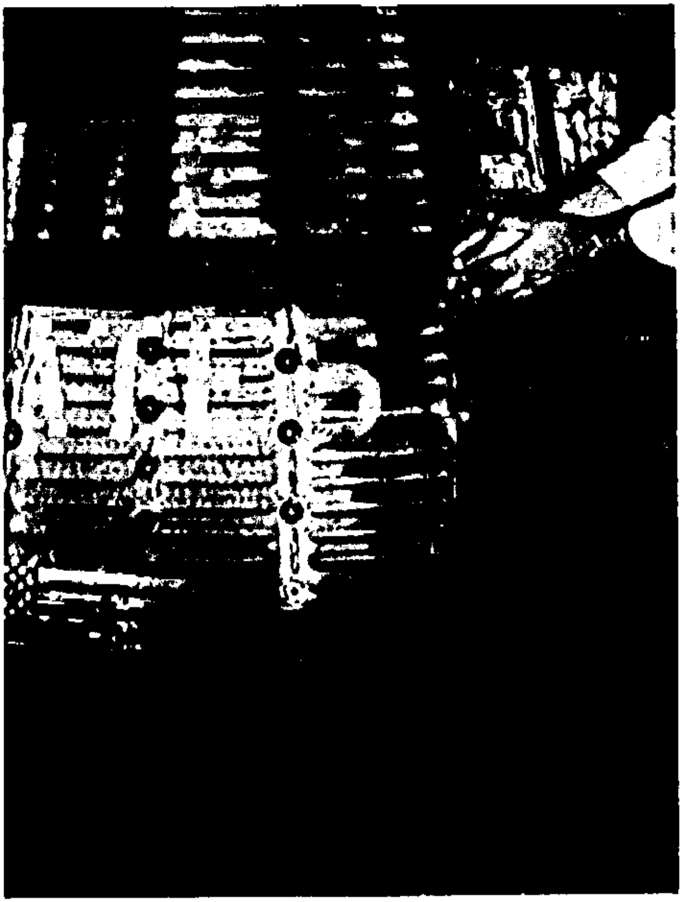

Figure 12-11. - Heat discoloration of lower right spacecraft to adapter tie.

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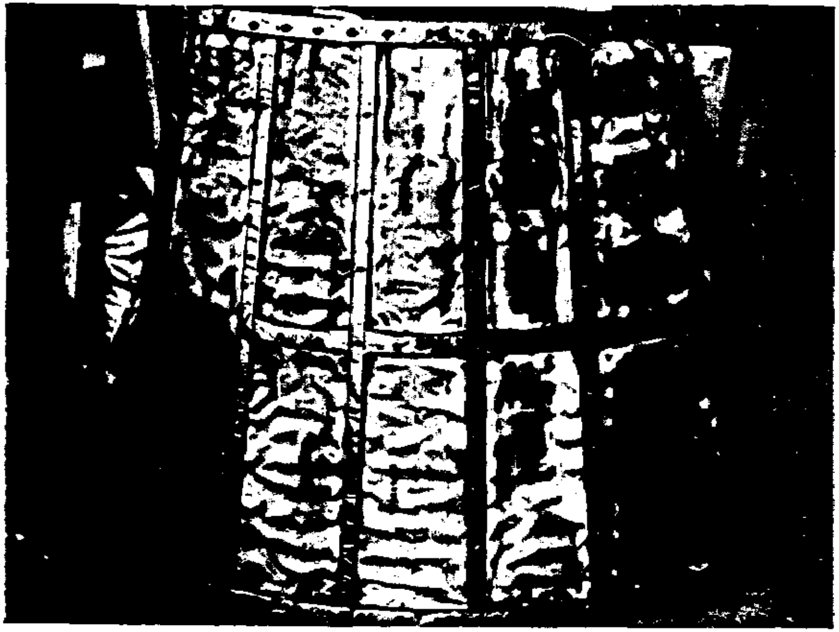

Figure 12-12. - Heat discoloration of insulating blankets.

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Figure 12-13. - Results of water in fuse blocks.

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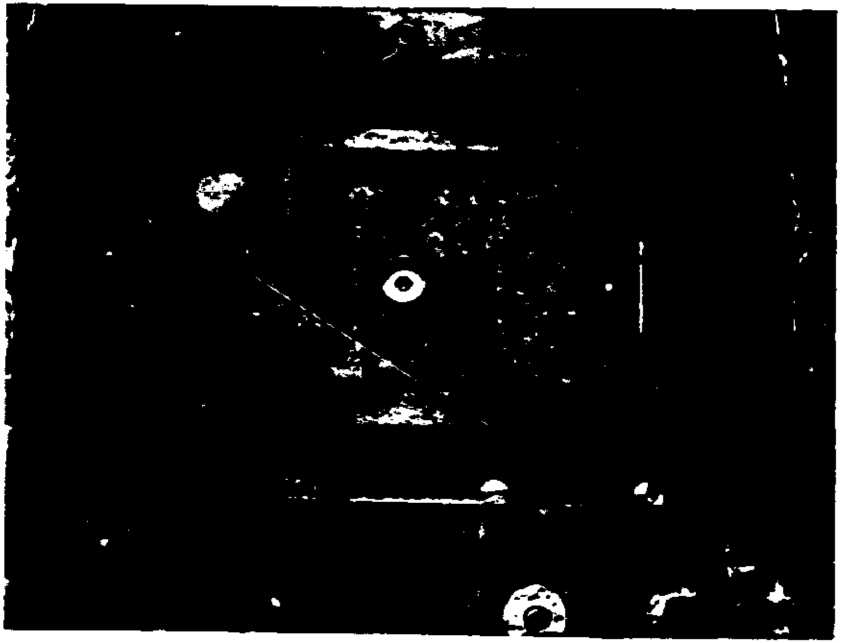

Figure 12-14. - Hot spot on umbilical seal.

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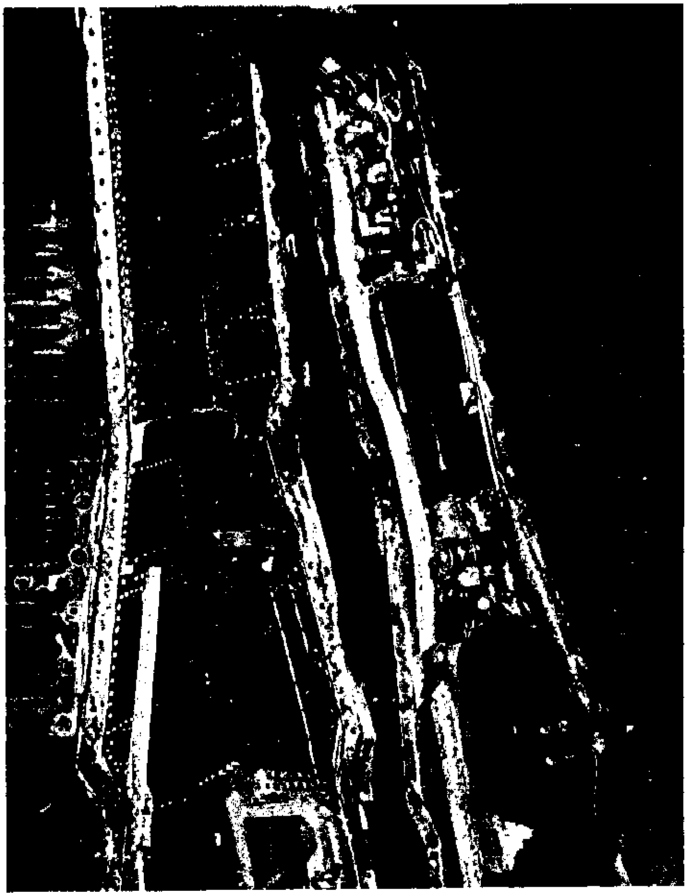

Figure 12-15. - Reentry communications experiment installation.

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