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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
MAY 1964

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GEMINI PROGRAM MISSION REPORT FOR GEMINI-TITAN 1

(GT-1) (U)

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Authorized for Distribution:

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George M Low
for Robert R. Gilruth
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Manned Spacecraft Center
Houston, Texas
May 1964

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1.0 SUMMARY

The Gemini-Titan 1 (GT-1) space vehicle was comprised of the Gemini spacecraft and the Gemini launch vehicle. The Gemini launch vehicle is a two-stage modified Titan II ICBM. The major modifications are the addition of a malfunction detection system and a secondary flight controls system. The Gemini spacecraft, designed to carry a crew of two men on earth orbital and rendezvous missions, was unmanned for the flight reported herein (GT-1). There were no complete Gemini flight systems on board; however, the C-band transponder and telemetry transmitters were Gemini flight subsystems. Dummy equipment, having a mass and moment of inertia equal to flight system equipment, was installed in the spacecraft. The spacecraft was instrumented to obtain data on spacecraft heating, structural loading, vibration, sound pressure levels, and temperature and pressure during the launch phase.

The GT-1 mission, designed to obtain flight verification of the Gemini launch vehicle and to demonstrate the flight compatibility of the spacecraft and its launch vehicle, was patterned after future Gemini manned missions. The launch trajectory was within the planned envelope. The spacecraft and second stage of the launch-vehicle were inserted into an orbit with a perigee of 86.6 nautical miles and an apogee of 173.0 nautical miles. The insertion parameters were within the design tolerance. The launch azimuth was selected to provide an optimum orbital ground track over the stations of the Manned Space Flight Network. Although the trajectory was designed for an orbital lifetime of several days, the mission was considered complete after three orbital passes.

On the second pass, radar detected small objects near the main vehicle. These objects reentered after a few orbital passes. Their size was believed small since their loss did not measurably decrease the size and weight of the spacecraft and second stage combination.

The last radar station which tracked the vehicle indicated that a reentry would occur in the South Atlantic area during the 64th orbital pass.

Vibration levels were generally well below predicted levels. The quasi-steady state longitudinal loading for the GT-1 flight was less than 25 percent of the structural capability of the spacecraft. Radial stress was lower than expected. Radial vibration on the pump package in the critical low frequency resonant modes was approximately one-tenth of the conservatively estimated maximum response. It is statistically probable that this value may increase on later flights but will stay well within design limitations.

The equipment dynamic environment in the 20 to 2,000 cps region was well within the equipment qualification launch spectrum except for vibrations in a narrow frequency band at 565 cps in which peak amplitudes were quite high; however, the energy contained in this narrow band was insignificant.

Heating characteristics of the GT-1 spacecraft were near predicted values for a nominal trajectory and were considerably less than values for the worst case design heating anticipated during reentry. Very little circumferential variation in temperature was noted.

Temperatures on the cabin section were generally lower than conservative design limits. Window temperatures were also lower than expected.

The adapter assembly, which is designed to withstand launch heating and structural loads, was subjected to temperatures lower than predicted; however, extrapolation of the data shows ample design margins for the more severe cases which may be encountered in future missions.

The acoustical environment as measured within the pressure vessel reached a peak near the time of maximum dynamic pressure as expected, but it was several decibels less than that used in the design requirements. The cabin pressure relief valve operated properly, and calculations using pressure measurements recorded at the Mission Control Center (MCC) during insertion and three orbital passes indicated a leak rate of approximately 700 cc/min, which is considerably less than the 1,000 cc/min permissible leak rate for spacecraft 1. Pressure measurements also indicated that all unpressurized compartments vented satisfactorily.

The C-band transponder and its associated radiating elements performed satisfactorily. Despite vehicle tumbling, all stations maintained track. Signal strength was adequate, code spacing remained nominal, and frequency drift was within tolerance.

Telemetry transmission during the mission was excellent. The two short losses of signal during the powered flight phase at all Cape Kennedy receiving stations resulted from flame attenuation and stage separation. However, these periods were covered by continuous reception at the Grand Bahama Island station.

The spacecraft instrumentation system performed satisfactorily. Loss of data was experienced from only one redundant temperature sensor which was discovered inoperative prior to launch and declared nonessential. A partial degradation of data quality from one vibration sensor was encountered and was due to amplitude excursions in excess of the full-scale range of the instrument.

An unexpected twisting of the cabin section electrical umbilical upon release at launch caused it to snag the adapter coolant umbilical release cable resulting in a 2.5 second premature release of the latter umbilical. No detrimental effect to the mission resulted.

The launch vehicle structure satisfactorily withstood the loads throughout the flight. Sustained longitudinal oscillations characteristic of earlier Titan II flight performance did not occur, thus demonstrating that the incorporation of a fuel accumulator and oxidizer standpipe was an effective solution to this problem. Flight loading was approximately 32 percent of design. Low frequency vibratory modes were less than predicted. Structural, fuel slosh, and engine mode vibrations were noted, but resulting loads were well below design levels.

Maximum skin temperatures along the launch vehicle forward skirt were 157° F as compared to predicted values of 325° F. Plans to add more insulation to this area should be reviewed, and, actually, a reduction of the material presently required should be considered.

Engine performance was satisfactory. Engine starts and engine shutdowns occurred well within design tolerances. One anomaly noted was an indicated 12-percent rise in the stage I oxidizer orifice inlet pressure approximately 25 seconds before staging.

The propellant feed system was satisfactory. A fuel-oxidizer temperature differential problem which occurred during preloading was corrected prior to launch.

The flight control system operation was acceptable. Programed flight control discretes occurred within time tolerance. A yaw offset due to aerodynamic factors required a 1° engine gimbal correction during stage I operation. A gradual change in the yaw and pitch attitude displacements during stage II flight, with respect to the gyro reference, was apparently the result of a slight misalinement of the engine thrust vector with the center of gravity.

The radio guidance system guided the second stage to a 2σ cut-off condition. The misalinement of the stage II thrust vector with respect to the vehicle center of gravity previously mentioned was also indicated by the performance of the flight control and guidance system. The MOD III radar data were of excellent quality and therefore did not contribute to the deviation from the planned mission trajectory.

The stage I and stage II hydraulic systems operation was nominal except for an excessive buildup of pressure in the secondary stage I hydraulic system during engine start. This pressure buildup, noted also during the preflight sequenced compatibility firing, is being investigated to determine if corrective measures are needed.

The electrical power system operated satisfactorily. Staging currents appeared normal. Remaining battery power at SECO was estimated to be 5.3 ampere-hours. Switchover to the secondary flight control system did not occur.

The malfunction detection system (MDS) functioned properly. No pitch or yaw rates approached MDS limits.

The airborne instrumentation system operated satisfactorily. Of a total of 240 parameters, none were completely lost. However, a partial loss of data from four sensors was experienced and one sensor reflected an unexplained increase in pressure (oxidizer pressurant orifice inlet pressure). Ground instrumentation was satisfactory with the following exceptions. A faulty rectifier tube in recorder no. 2 was replaced during countdown and one Sanborn recorder was unbalanced but corrected prior to countdown. Landline data acquisition was 100 percent. Eight minor PCM/FM and one FM/FM telemetry signal dropouts occurred in the reception at the Cape Kennedy Missile Test Annex, Telemetry building II.

The range safety system for GT-1 functioned as planned. No operational or equipment discrepancies occurred.

Ground complex performance was satisfactory and operated as planned except for the utilization of missile trajectory measurement (MISTRAM) system II, by the impact predictor. When the impact predictor attempted to switch to MISTRAM II, the data from MISTRAM II were poor and FPQ-6 data were used instead. The airborne MISTRAM transponder performance was nominal throughout the launch phase.

The Gemini launch vehicle (GLV) pyrotechnics for engine start and lift-off functioned normally. Because of the minimum start cartridge temperature requirements, an electric heater and polyethlene bag were used on the stage I start cartridges during the countdown. Monitoring was effected by thermocouples on the start cartridge and in the engine compartment.

The electrical and structural integrity of the Gemini launch vehicle and spacecraft interface was satisfactory throughout the flight. The loads experienced were a small percentage of its structural capability.

The operations within the Mission Control Center are presented in this report as they were observed during the mission.

Launch operations involved a 390-minute countdown with a scheduled $23\frac{1}{2}$ hour hold at T-330 minutes.

The launch was preceded by Mission Control Center and Atlantic Missile Range launch simulations. All facilities and mission operations personnel were in a "ready" condition for the launch operation. The countdown was smooth through lift-off. Manual fuel cutoff and destruct command signals received at the Gemini Launch Vehicle Systems Engineer Console in the MCC due to normal range operating procedures which were unknown to the MCC and the blockhouse personnel were discussed at length and will be coordinated for future exercises. Phase problems of the MISTRAM II at T-38 minutes were resolved by T-20 minutes. The launch-vehicle internal power transfer was normal, and all Mission Control Center instrumentation was acceptable at lift-off.

Spacecraft and launch vehicle parameters monitored were acceptable throughout the flight. Staging arm and shutdown signals were received. Staging was characterized by normal signals except for a momentary telemetry dropout due to RF flame attenuation. The stage II flight appeared normal, SECO lights operated properly, and thrust chamber pressure drop and engine under-pressure signals were received. Hydraulic pressure was constant for 10 seconds, dropped sharply, and rose to one-third the former value.

The conditions at insertion were acceptable and were so indicated by data from the guided missile computer facility. Parameter monitoring was effected for three orbital passes.

Voice exchanges between the Mission Control Center and remote stations consisted of contact times, signal strength, radar targets, and tumbling rates.

Network operations were characterized by the following:

(a) Slew test failure of Grand Bahama Island and San Salvador Island radars during countdown

(b) Two cases of frequency interference at Bermuda

(c) Phasing problem during countdown at MISTRAM II

(d) Teletype loss (due to lightning) with Carnarvon. Emergency contact was available.

(e) Noisy communications with Canary Island

(f) Good radar performance

(g) Continuous C-band transponder operation during passes

(h) Multiple objects noted by the several stations during the second and third passes

(i) Horizon-to-horizon telemetry except for dropouts due to spacecraft tumbling

(j) Normal timing system operation

(k) Unsatisfactory acquisition and performance of the automatically gimballed antenna vectoring equipment due to the unexpected but normal deviation of the TM transmitters

(l) Satisfactory command functions during launch

(m) Computer failure to execute switchover from the impact predictor to the guided missile computer facility no. 1 on command. (Later this switchover was executed).

Photographic coverage during the launch phase was generally good in quality but intermittant due to haze and cloud conditions. Quality of some film was also affected by underexposure, being out of focus, image bleeding, illegible timing, and film scratches and gouges. Aerial photography was not adequate to support data evaluation due to the following:

(a) Flight-path and position problems

(b) Buffeting and poor tracking

(c) Poor acquisition and inability to see the launch vehicle

(d) Camera power failure

(e) Obscuration by cloud cover.

2.0 INTRODUCTION

The first mission of the Gemini Program, the United States' second program of manned space exploration, was successfully conducted on April 8, 1964. This mission was unmanned and utilized the first production Gemini spacecraft and launch vehicle (modified Titan II). The mission was designated Gemini-Titan 1 (GT-1) and was launched at 11:00:01.692 a.m. e.s.t. from complex 19 at the Cape Kennedy Missile Test Annex, Cape Kennedy, Florida.

Spacecraft separation from the second stage of the launch vehicle was not planned for this mission, and consequently, the second stage of the launch vehicle and the Gemini spacecraft were inserted into orbit.

The mission was concluded successfully approximately 4 hours and 50 minutes after lift-off, which was the end of the third orbital pass over Cape Kennedy. Tracking, however, was continued by the Goddard Space Flight Center until the spacecraft reentered on the 64th orbital pass over the southern Atlantic Ocean.

The primary purpose of the GT-1 mission was to verify the structural integrity of the Gemini launch vehicle and Gemini spacecraft, as well as to demonstrate the capability of the Gemini launch vehicle and its systems to place the spacecraft into a prescribed earth orbit.

The first-order test objectives of the GT-1 mission were the following:

(a) To demonstrate the Gemini launch vehicle performance and to flight qualify the vehicle subsystems for future Gemini missions

(b) To determine the exit heating conditions on the spacecraft and launch vehicle

(c) To demonstrate the structural integrity and compatibility of the spacecraft and launch vehicle combination through orbital insertion

(d) To demonstrate the structural integrity of the Gemini spacecraft from launch through orbital insertion

(e) To demonstrate the ability of the Gemini launch vehicle and ground guidance systems to achieve the required orbital insertion conditions

(f) To monitor the switchover circuits as installed on the Gemini launch vehicle and to evaluate their sufficiency for mission requirements

(g) To demonstrate the switchover function, if anomalies occur within the primary autopilot or hydraulic systems that would require the use of the secondary autopilot or hydraulic systems

(h) To demonstrate the malfunction detection system

The second order test objectives of the GT-1 mission were the following:

(a) To evaluate the operational procedures used in establishing the Gemini launch vehicle trajectory and cut-off conditions

(b) To demonstrate the performance of the launch and tracking networks

(c) To verify orbital insertion conditions by tracking the C-band transponder system in the spacecraft

(d) To provide training for the flight dynamics, guidance switchover, and malfunction detection systems flight controllers

(e) To demonstrate the operational capability of the prelaunch and launch facilities.

Each of the first- and second-order test objectives was satisfactorily fulfilled.

An evaluation has been made of all available data, and the results of this evaluation are presented in this report.

Supplements to this report will be prepared as necessary and will be published under separate cover to augment this report.

3.0 VEHICLE DESCRIPTION

The Gemini launch vehicle (GLV) and the Gemini spacecraft comprise the Gemini space vehicle. The GLV is a Titan II intercontinental ballistic missile (ICBM) which has been modified to launch a manned spacecraft. The Gemini spacecraft is designed to carry a two-man crew on earth orbital and rendezvous missions. The space vehicle for the first Gemini-Titan (GT-1) mission is shown in the lift-off configuration in figure 3-1.

3.1 GEMINI SPACECRAFT DESCRIPTION

Gemini spacecraft 1, shown in figure 3-2, is the first production Gemini spacecraft. The spacecraft was unmanned for the GT-1 mission, and was equipped with instrumentation designed to obtain data on exit heating, structural loads, temperatures, vibrations, and pressures to be telemetered to ground tracking stations. For the most part, the normal Gemini spacecraft systems carried on manned missions were not installed in the GT-1 spacecraft. Instead, dummy equipment or ballast was installed. In cases where dummy equipment was used (retrorockets, and reentry control system thrusters), it simulated the weight, center of gravity, and moment of inertia of the actual equipment. Where ballast was used, it simulated only the weight and center of gravity of the actual equipment. Spacecraft separation from the second stage of the GLV was not planned for this mission, and consequently, the combined second stage and spacecraft were inserted into orbit. See reference 1 for a more complete description of the Gemini spacecraft.

3.1.1 Spacecraft Structure

Structurally, the spacecraft consists basically of the reentry and the adapter major assemblies. The spacecraft is of semimonocoque construction; titanium, magnesium, and aluminum are the primary structural materials. The overall dimensions of the spacecraft are as follows: length, 226.67 inches (18.89 feet); diameter at the heat shield, 90.0 inches (7.5 feet); and diameter at launch vehicle-adapter interface, 120.00 inches (10.00 feet).

3.1.1.1 Reentry assembly.- The reentry assembly (fig. 3-2) is composed of three primary structural sections: the cabin section, the reentry control system (RCS) section, and the rendezvous and recovery (R and R) section. Heat-protection materials cover the entire external surface of the reentry assembly.

3.1.1.1.1 Cabin section: The cabin section (fig. 3-2) is a truncated cone: the forward end is attached to the RCS section, and the aft end is attached to the adapter assembly. The cabin has an internal pressure vessel which is the crew station for two people on a manned mission; however, for the GT-1 mission, electrical and electronic instrumentation equipment was mounted on pallets in this pressure vessel. The shape of the pressure vessel also allows space between it and the outer conical shell for the installation of most of the systems plus insulation material. The structural design criteria for the pressure vessel requires it to withstand an ultimate burst pressure of 12.0 psi and an ultimate collapsing pressure of 3.0 psi. Outlining the cabin section and equipment bays is a heliarc-welded frame section of ring segments, stringers, and longerons. The cabin section has a double-wall titanium skin to contain the cabin atmosphere. The unbeaded inner wall is seam welded to the outer wall. The inner wall is reinforced by vertical and longitudinal stringers on its outside surface. Pressure bulkheads are attached to each end of the cabin section, and two hatches are provided on the side of the cabin section for spacecraft and instrumentation checkout, and astronaut ingress and egress.

3.1.1.1.2 Reentry control system section: The reentry control system (RCS) section (fig. 3-2) is mounted between the cabin section and the R and R section. This section is cylindrical and has a height of 18.00 inches. The section has a center titanium cylinder with a bulkhead on each end. Around the periphery of the cylinder are eight stringers to which are attached eight shingles which form the outer skin of the RCS section. Since reentry and recovery of the spacecraft were not planned on the GT-1 mission, this section contained dummy thrusters and ballast.

3.1.1.1.3 Rendezvous and recovery section: The rendezvous and recovery (R and R) section (fig. 3-2) is the forward end of the spacecraft and is attached at its aft end to the RCS section. The rearward portion of the R and R section is cylindrical in shape and the forward portion is a truncated cone. The R and R section has a height of 48.14 inches, including the fiber-glass nose fairing. The section's primary structure consists of rings, stringers, and titanium bulkheads with an outer cover of shingles. Since reentry of the spacecraft was not planned for this mission, the R and R section contained ballast to provide the correct weight and center of gravity.

3.1.1.1.4 Heat protection structure: The heat protection materials (fig. 3-2) for spacecraft 1 consist of a heat shield, fiber-glass nose fairing, fiber-glass horizon scanner fairing, Rene' 41 and beryllium shingles, and Min-K insulation material.

The heat shield is a dish-shaped structure of honeycomb filled with silicone elastomer. It is attached to the aft end of the cabin section by eighteen 0.25-inch-diameter bolts. The shield's diameter is 90 inches and its spherical radius is 144 inches. The shield is an ablative device whose purpose is to protect the reentry assembly from the extreme thermal conditions which are encountered during reentry into the earth's atmosphere. However, since separation of the spacecraft from the launch-vehicle second stage, as well as reentry and recovery, was not planned for this mission, the main purpose of the shield on this flight was to maintain structural completeness. Four 1-foot-diameter counter bores were made in the heat shield ablative material on this spacecraft to insure that the spacecraft would be totally destroyed during reentry.

The nose fairing is a short truncated cone of fiber-glass construction. It forms the forward 6.14 inches of the spacecraft and is attached to the R and R section. The fairing protects the rendezvous radar and its antennas from the heat experienced during the first stage of powered flight and is jettisoned 45 seconds after second-stage ignition; however, the radar was not installed on spacecraft 1, and, therefore, the fairing was not jettisoned.

The horizon-scanner fairing is constructed of fiber glass and is attached at the intersection of the cabin section and the RCS section. It protects the horizon-scanner equipment from the heat experienced during the first stage of powered flight. The fairing is jettisoned 45 seconds after second-stage ignition; however, no horizon-scanner equipment was installed on spacecraft 1, and the fairing was not jettisoned.

The external surface of the cabin section is constructed of beaded Rene' 41 shingles. The external surfaces of the RCS and R and R sections are constructed of beryllium shingles for heat protection during launch and reentry.

The insulation blankets used on the Gemini spacecraft provide the crew and equipment with a great degree of protection from the heat of launch and reentry by effectively preventing the transfer of retained heat from shingles and heat shield to the pressure vessel and equipment. The insulation also aids in temperature stabilization of the pressure vessel and spacecraft equipment during orbital flight. A secondary benefit of the insulation is its inherent ability to reduce the sound power level in the pressure vessel during the launch phase. The blankets are outside the large bulkhead and inside the shingles of the cabin section, the reentry control system (RCS) section, and the rendezvous and recovery (R and R) sections.

3.1.1.2 Adapter assembly.- The adapter assembly (fig. 3-2) has two primary structural sections: the retrograde section and the equipment section. The forward end of the assembly is attached to the aft end of the reentry assembly by three titanium retaining straps, and the aft end of the adapter assembly provides the interface for mating the complete spacecraft with the launch vehicle. The adapter is a truncated cone of semi-monocoque construction having a height of 90.00 inches, a diameter at the forward end of 88.30 inches, and a diameter at the aft end of 120.00 inches. The adapter's basic structure consists of circumferential aluminum rings spliced together with extruded magnesium alloy stringers. The outer skin of the adapter is formed from magnesium sheets which are attached to the magnesium stringers. The function of the adapter for the GT-1 mission was to mate the spacecraft with the Gemini launch vehicle and to provide data for determining the structural effects of flight on the adapter and simulated equipment.

3.1.1.2.1 Retrograde section: The retrograde section (fig. 3-2) forms the forward one-third of the adapter assembly and provides the mating point for the adapter assembly with the reentry assembly. An aluminum I-beam X-frame is a major part of this section; it was used for the mounting of the four dummy retrorockets and for providing structural rigidity. Measurements were also made on the I-beam frame to determine its structural rigidity.

3.1.1.2.2 Equipment section: The equipment section (fig. 3-2) forms the aft two-thirds of the adapter assembly. Aluminum-alloy tubing in a truss-beam arrangement is bolted to the primary structure and was used for mounting of dummy equipment. A blast shield was also installed on the forward end of this section which normally will insure that the retrorocket blast during abort or retrofire maneuvers does not cause explosions or other events which may have catastrophic results on the reentry assembly.

3.1.1.3 Spacecraft-launch-vehicle mating.- The spacecraft is mated to the Gemini launch vehicle through a continuous, machined, aluminum-alloy ring (fig. 3-3). This ring is 120.00 inches in diameter and is attached by 20 bolts to the mating ring of the Gemini launch vehicle. The launch-vehicle mating ring has three alinement pins and four index marks to insure proper spacecraft-launch-vehicle alinement.

3.1.1.4 Cabin equipment arrangement.- The cabin (pressure vessel) shown in figure 3-4 contained the instrumentation pallets, a pressure transducer, some temperature sensors, vibration sensors, accelerometers, and two microphones for sound pressure level measurements. The two pallets (left and right) were platforms on which the instrumentation and communications equipment were mounted (fig. 3-5). These pallets, along with ballast weights, were installed on the ejection-seat rails attached to the large pressure bulkhead.

3.1.2 Major Systems

Because spacecraft 1 was used for a structural test, many of the Gemini spacecraft systems were not required and therefore not installed.

Some of the systems used in spacecraft 1 were production equipment, especially adapted for use in this spacecraft; the other systems were especially designed for use in this spacecraft only.

The following paragraphs contain descriptions of the systems included in spacecraft 1.

3.1.2.1 Communications.- The communications system facilitates the ground tracking of the spacecraft during the course of the entire mission or as limited by battery life. This system for GT-1 consisted of a production C-band radar transponder and associated equipment (fig. 3-5). The radiating elements provide the radiation and reception coverage for the C-band transponder in all directions from the spacecraft, except forward and aft. They are spaced 120° apart around the cabin section of the reentry assembly. A block diagram of the communications system is shown on figure 3-6.

3.1.2.2 Telemetry and instrumentation system.-

3.1.2.2.1 Telemetry: A PAM-FM-FM telemetry system was used to send data from the instrumentation system to ground receiving stations. The three telemetry transmitters were standard Gemini spacecraft equipment. The integrated telemetry system and instrumentation system is shown in block diagram form in figure 3-6.

3.1.2.2.2 Instrumentation: The PAM-FM instrumentation system in spacecraft 1 monitored the following parameters: the environment and flight characteristics of the spacecraft, structural temperature measurements at various points on the spacecraft, and structural dynamic measurements at selected points on the spacecraft. See reference 2 for a more detailed description of the telemetry and instrumentation system.

3.1.2.2.1 Sensors: Two types of temperature sensors were used: thermocouples, and resistance elements. Sixty-three thermocouples were used for temperatures over 900° F, and eighteen resistance-element temperature sensors were used for temperatures below 900° F. For a listing of the temperature sensors, see the measurement list in table 3-I.

A total of nine pressure transducers were located throughout the spacecraft. All sensors read absolute values (0 to 15 psia) with the exception of the sensor which measured the cabin differential pressure (0 to 6 psid) referenced to the cavity between the cabin pressure vessel and the RCS section. Refer to table 3-I for a list of pressure measurements.

The low-frequency vibrations (1 to 30 cps) were measured by force balance servo accelerometers. The mid-frequency (20 to 600 cps) and high-frequency (20 cps to 2 kc) vibrations were monitored by piezoelectric accelerometers.

Three force balance servo accelerometers with a frequency response of 0 to 1 cps were used to measure spacecraft accelerations along the X-, Y-, and Z-axes. See table 3-I for accelerometer ranges.

The noise in the cabin was measured over the frequency range of 37.5 to 9,600 cps. The overall range of the system was 90 to 155 db (ref. 0.0002 dynes/cm^2).

Two break wires were used to monitor for an inadvertent separation of the horizon-scanner fairing.

3.1.2.3 Miscellaneous equipment.- The miscellaneous equipment for the GT-1 spacecraft were the electrical power source, cabin pressure relief valve, prelaunch cooling system, and the umbilicals.

3.1.2.3.1 Electrical power source: The power source consisted of a 24 volt d-c, 45 ampere-hour, silver-zinc battery which was located on the left pallet as shown in figure 3-5(a).

3.1.2.3.2 Cabin pressure relief valve: A cabin pressure relief valve was installed to maintain cabin pressure at the required level during the launch phase.

3.1.2.3.3 Prelaunch cooling system: The cooling system for spacecraft 1 was peculiar to that spacecraft, in that the system was operational only prior to lift-off. The system consisted of equipment coldplates connected to the coolant umbilical.

3.1.2.3.4 Umbilicals: All three spacecraft-to-ground umbilicals were installed and designed to be ejected during the launch sequence. The electrical umbilical to the reentry assembly and the coolant umbilical to the adapter assembly were operational for the spacecraft. The electrical umbilical to the adapter assembly was also connected; however, it was used only to qualify the umbilical ejection system.

3.2 GEMINI LAUNCH VEHICLE DESCRIPTION

The Gemini launch vehicle (GLV) is a two-stage intercontinental ballistic missile (ICBM) which has been modified and "man-rated" for use in the Gemini Program. The propulsion system uses storable hypergolic propellants which are contained in individual propellant tanks in each stage. The modifications which were made to the basic Titan II to achieve the "man-rated" GLV are as follows:

(a) The addition of a secondary flight control system and a switchover capability

(b) The addition of a fully redundant manfunction detection system

(c) The provision of an engine shutdown capability from the spacecraft

(d) The addition of redundant electrical signals to the stage II engine start cartridge

(e) The provision of a 120-inch-diameter cylindrical skirt forward of the stage II oxidizer skirt for mating the spacecraft to the launch vehicle

(f) The addition of a redundant electrical system

(g) The removal of the retrorockets, vernier rockets, and allied equipment

(h) The replacement of the Titan II inertial guidance system with the MOD III-G radio guidance system (RGS)

(i) The addition of a three-axis reference system (TARS) to provide attitude reference and open-loop programing to the autopilot system during the period of flight before the radio guidance system is activated

(j) The replacement of certain Titan II support trusses with light-weight trusses

(k) Necessary modifications to the range safety system.

For a more complete description of the Gemini launch vehicle, see reference 2.

3.2.1 Gemini Launch Vehicle Structure

The launch vehicle's diameter is 10 feet, its first stage is 70.675 feet long, and its second stage is 28.27 feet long. (See fig. 3-7.) The overall length of the spacecraft-GLV combination is 108.067 feet. The aft 9 feet of the GLV second stage is inserted into the interstage structure of the first stage. The two stages are joined together by four studs and eight explosive nuts which are used for staging. Blast ports are provided at the aft end of the interstage area for venting the stage II engine combustion products at staging.

Aluminum alloy is the primary structural material used in the GLV. The semimonocoque shell uses integral stringers for primary load carrying capability and internal ring frames for stringer stability. Fuel and oxidizer tanks are integral parts of the structure in both stages providing local pressure stabilization.

3.2.1.1 Stage I.- Structurally, stage I of the GLV consists of a forward interstage structure, an oxidizer-tank forward skirt, an oxidizer tank, a between-tank structure, and a fuel tank. These components are assembled from fore to aft in the order listed. The interstage structures (forward skirt and between-tank structure) are fabricated assemblies employing a riveted skin, stringers, and frame. The oxidizer tank is a welded structure consisting of a forward dome, a tank barrel, an aft dome, and an oxidizer propellant outlet. The fuel tank, also a welded structure, consists of a forward dome, a tank barrel, an aft dome, fuel propellant outlets to the prevalves, and an internal conduit. A propellant line directs the oxidizer from the oxidizer tank to the oxidizer prevalves through the internal conduit in the fuel tank. The aft section of the stage I airframe, consisting of four longerons, provides the attachment provisions for the welded steel truss which supports the stage I engine assembly. These longerons also serve to attach the launch vehicle to the thrust mount.

3.2.1.2 Stage II.- Structurally, stage II of the GLV consists of the forward oxidizer tank skirt which serves as the mating plane between the spacecraft and launch vehicle, an oxidizer tank, a between-tank structure, a fuel tank, and an aft skirt. These components are assembled from fore to aft in the order mentioned. The type of construction in this stage is the same as that used in stage I. The aft section of the fuel tank provides the attachment point for the monocoque cone which supports the stage II engine assembly.

3.2.2 Major Systems

3.2.2.1 Propulsion system.- The two-stage propulsion system for GLV-1 is adapted from the system used on the Titan II missile. Minor changes have been made to the system to "man-rate" it for use as a spacecraft launch vehicle. The most significant of the changes to this system was the addition of a malfunction detection system.

3.2.2.1.1 Stage I: The first-stage portion of the propulsion system consists of two independently operating, gimballed engine subassemblies mounted on a single frame. The subassemblies are designed to operate simultaneously to provide the required total thrust. Each subassembly consists of a thrust chamber, a turbo pump, and a gas generator. The engine ignition is initiated by solid-propellant cartridges which receive an electrical discrete signal from the launch control equipment. These cartridges provide the hot gas which starts the turbine. The turbine then drives the turbopump and supplies propellants to the engine subassemblies. During the powered phase of the flight, the gas generators drive the turbopump and operate from the propellants discharged by the turbopumps.

The inflight pressurization of the propellant tank is provided by an autogenous (self-pressurizing) system that functions as long as the engine subassemblies are operating.

3.2.2.1.2 Stage II: The second-stage portion of the propulsion system consists of one chamber unit that is similar in construction and operation to one stage I engine subassembly. The engine is designed specifically for operation at high altitudes. It, therefore, contains an ablative skirt attached to the regeneratively cooled thrust chamber. This ablative skirt provides an expansion ratio of 49.2:1 for stage II as compared with the 8.1 expansion ratio of stage I.

The thrust chamber is gimbal mounted; however, the gimballing action of the single chamber provides flight control only about the pitch and yaw axes. A roll-control nozzle has been added to the second stage for flight control about the roll axis. This nozzle ducts the gas-generator exhaust gas overboard, and roll control is obtained through the swiveling action of the nozzle. The second-stage engine is started in the same manner as the stage I engine subassemblies, except the start signal is received by virtue of stage I thrust termination.

An autogenous tank pressurization system is used in the stage II fuel tank. This system is not used in the oxidizer tank, as the tank is pressurized to a high level before flight which, when augmented by inflight acceleration forces, produces the required pump net positive suction-head pressure.

3.2.2.2 Flight control system.- The flight control system consists of a primary and a secondary system.

The primary system consists of a three-axis reference system (TARS) package, an autopilot package, an adapter package, a set of first-stage rate gyros, tandem actuators and redundant hydraulics on the first stage, and single actuators on the second stage.

The TARS package contains the programer and the pitch, yaw, and roll displacement gyros. The pitch and yaw guidance inputs to the TARS package during stage II operation are obtained from the radio guidance system (RGS) decoder after guidance enable. The autopilot package contains most of the autopilot circuitry and three rate gyros. The adapter package is used to adapt the TARS output signal to the input of the autopilot package since this autopilot was designed to accept a d-c input from an inertial guidance system. The tandem actuator receives inputs from the primary and secondary flight control systems.

The secondary system consists of an autopilot package, a set of rate gyros on the first stage, and includes the secondary coils of the tandem actuators on the first stage. The autopilot and rate-gyro packages are identical to those used for the primary system. In the event of a switchover, the displacement reference and guidance steering signals of the primary system are connected to the secondary system at the secondary autopilot package input for GT-1 only.

3.2.2.3 Guidance.- The guidance system consists of an airborne subsystem and a ground system. The airborne subsystem consists of a pulse beacon, a rate beacon, and a decoder. The beacons are interrogated by the ground station. The decoder accepts steering and discrete commands from the ground station and transmits these commands to the flight control system.

The ground station is the Mod III A ground station consisting of the tracking subsystem, the rate subsystem, and a computer.

The tracking subsystem is a monopulse radar used for position measurements and as a data link between the computer and the launch vehicle.

The rate subsystem is a continuous wave (CW) Doppler radar used for velocity measurements.

The computer is a digital computer that is capable of providing guidance-evaluation displays and solving guidance equations to determine guidance commands. Range, elevation, and azimuth; and $\dot{R}$, $\dot{P}$, and $\dot{Q}$ velocities are received from the radar. Steering commands and discrete signals are sent to the tracking subsystem encoder for transmission to the launch vehicle. The computer generates a backup shutdown (ASCO) command which is transmitted to the launch vehicle by the range safety command radio link.

Two hydraulic systems are used in the GLV to position the engine thrust chambers and roll nozzle so that the GLV can be controlled in its correct flight path. The stage I system consists of a primary and a secondary system.

The stage I tandem hydraulic actuators are positioned proportionately by signals from the flight control system during the powered flight. The actuators cause the engine subassemblies to be repositioned and thereby control the GLV flight path in the pitch, yaw, and roll axes. A self-sustaining secondary hydraulic system operates on a standby basis during first-stage powered flight and assumes control if a malfunction requiring a switchover occurs.

The stage II hydraulic system provides power to the flight control system for control of the flight path of the combination of the stage II launch vehicle and spacecraft during the stage II powered flight phase. The engine is gimbaled in pitch and yaw planes. Control about the roll axis is accomplished by proper positioning of the off-center roll control nozzle.

3.2.2.4 Electrical system.- The electrical system provides and distributes nominal 28 v d-c and 115 v a-c, three-phase, 400-cycle power to the GLV equipment. Ground power is supplied for prelaunch test checkout and operation until power is transferred to the launch vehicle during the countdown. The airborne electrical system is composed of two major subsystems: the power distribution subsystem and the electrical sequencing subsystem.

3.2.2.4.1 Power distribution subsystem: The power distribution subsystem is divided into the accessory power system (APS), the instrument power system (IPS), and the stage I destruct battery.

The APS and IPS buses receive airborne power from separate 28 v d-c silver-zinc rechargeable batteries rated at 12 ampere-hours each. These batteries will supply GLV electrical power for a minimum of 10 minutes after stage II engine shutdown.

The APS provides power to the static inverter, the malfunction detection system (MDS), the APS command receiver, the APS shutdown and destruct circuitry, the RGS, the primary flight control system, the sequencing system, and the stage II engine start circuitry.

The IPS provides power to the MDS, the missile trajectory measurement (MISTRAM) system, the IPS command receiver, the IPS shutdown and destruct circuitry, the flight control system, the sequencing system, and the airborne instrumentation system.

The stage I destruct battery provides power, in case of inadvertent separation, for stage I engine shutdown and the stage I destruct system.

3.2.2.4.2 Sequencing system: The sequencing system provides power for the correct sequencing of major GLV events from first-stage engine start to stage II engine shutdown. Some of the major functions of the sequencing system are resetting of the stage I prevalves switch, lockout of engine shutdown from the spacecraft for the first 40 seconds of flight, actuation of the APS and IPS staging switches, shutdown of stage I engine, firing of the staging nuts, starting of the stage II engine, and arming of the stage II engine shutdown relays.

3.2.2.5 Malfunction detection system.- The malfunction detection system (MDS) monitors critical GLV parameters and supplies indications to alert the astronauts and ground personnel of any potentially catastrophic malfunctions of the GLV. Analog circuits and gages provide redundant indications of fuel and oxidizer tank pressures for both stages in the spacecraft and to the Mission Control Center (MCC). Indications of electrical power bus voltage exist in the spacecraft and are provided in analog to MCC via telemetry. Bilevel signal circuits provide indications of overrate, low engine-chamber pressure, switchover to the secondary flight control system, and staging. These indications are provided to allow interpretation and initiation of abort procedures, if required. The system also accomplishes automatic switchover to the secondary flight control system in the event of engine hardover, attitude overrates, or loss of hydraulic pressure.

For the GT-1 mission, the MDS parameters were not sent to the spacecraft since no monitoring systems were installed in the spacecraft, nor was a crew onboard. The parameters were displayed via telemetry to the ground for monitoring by MCC personnel.

3.2.2.6 Instrumentation system.- The GLV instrumentation system provides malfunction and performance data for use in verifying the vehicle's ability to place the spacecraft into the required earth orbit. Evaluation of the data will also provide for the localization of malfunctions and the verification of subsystems operation during preflight checkout.

The airborne instrumentation system consists of two radio telemetry links. The FM-FM link transmits on 237.0 megacycles, and the PCM-FM link transmits on 244.3 megacycles. Each system provides for sensing, signal conditioning, and multiplexing of the data.

Most of the GLV measurements are carried on the PCM-FM link which has 196 analog channels and 48 bilevel channels. All of the channels are time-multiplexed variously at rates of 20, 40, 100, 200, and 400 samples per second. For a list of the GLV measurements for GT-1, see table 3-II and figure 3-8.

The FM-FM link is a seven-channel frequency multiplex and a 100-kilocycle reference signal, with no commutation. The link includes an onboard record-reproduce tape recorder, which is used to acquire data during the RF flame-attenuation period experienced during staging. These recorded data are transmitted to ground tracking stations after the attenuation period ceases.

The outputs of these two links are fed to an antenna diplexer and then through a power divider to the four flush-mounted antennas on the outer skin of compartment 2. (See fig. 3-7.)

3.2.2.7 Range safety and pyrotechnics systems.- The range safety and pyrotechnics system provides the Range Safety Officer (RSO) with a means of accurately determining range, range rate, velocity, and trajectory information, as well as a means for destroying the launch vehicle in case the impact prediction point moves into an undesirable area. These tasks are normally accomplished using a MISTRAM system, a command control system, an ordnance system, and an inadvertent-separation stage I destruct system. The systems equipment includes four command control antennas, a six-port junction, and two command receivers. The engine shutdown and vehicle destruct commands which are iniated by the RSO are received by the onboard command receivers.

The MISTRAM transponder, in conjunction with ground stations at Valkeria, Florida, and Eleuthera Island in the Bahamas, provides continuous trajectory data for use by the RSO in evaluating, in real time, the performance of the GLV's guidance and control system.

The ordnance system, operating in conjunction with the command receivers, receives a destruct command from the receivers, which through a motor-driven switch, energizes the destruct initiators.

3.3 GT-1 WEIGHT AND BALANCE DATA

The weight and balance data for the GT-1 space vehicle are shown in the following table:

Condition	Weight (including spacecraft), lb (b)	Center-of-gravity location, in. (a)		
		Y	Z	X
Ignition	344,565	0	59.9	776.4
Lift-off	341,000	0	59.9	776.7
Stage I burnout	85,021	-0.1	59.8	443.2
Stage II start of steady-state combustion	73,044	-0.26	59.95	345.6
Stage II burnout	13,785	-1.46	59.73	290.9

[a]X-axis reference is GLV station 0.000 (located 51.295 in. forward of the spacecraft nose fairing). Y-axis is referenced to the centerline of the vehicle. Z-axis is referenced to the waterline (60 in. below centerline) of the vehicle.

[b]Postflight trajectory weights obtained from Aerospace Corporation, as measured during the flight.

The spacecraft weight and center-of-gravity location are as follows:

Condition	Weight, lb	Center-of-gravity location, in. (a)		
		X	Y	Z
Entire mission	7,026	-0.54	-0.73	110.73

[a]Z-axis reference is located 13.44 inches aft of the Gemini launch vehicle-spacecraft mating plane (GLV station 263.385). The X- and Y-axes are referenced to the centerline of the vehicle.

TABLE 3-I.- SPACECRAFT INSTRUMENTATION MEASUREMENTS FOR GT-1

(a) Low-level commutator (link 1 and link 2)

Measurement	Description	Range
MA26	Low-level full-scale	20.0 mv d-c
MA20	Low-level zero reference	0.0 mv d-c
MA28	Reference junction temperature no. 1	-55 to + 200° F
MA29	Reference junction temperature no. 2	-55 to + 200° F
PA01	Outer skin temperature	[a]Ambient to ambient + 895° F
PA02	Outer skin temperature	[a]Ambient to ambient + 895° F
PA03	Outer skin temperature	[a]Ambient to ambient + 900° F
PD06	Outer skin temperature	[a]Ambient to ambient + 1000° F
PB03	Outer skin temperature	[a]Ambient to ambient + 873° F
PB04	Outer skin temperature	[a]Ambient to ambient + 873° F
PB05	Outer skin temperature	[a]Ambient to ambient + 873° F
PB07	Separation joint magnesium strip	0 to + 600° F
PB14	Outer skin temperature	[a]Ambient to ambient + 895° F
PB15	Outer skin temperature	[a]Ambient to ambient + 885° F
PB16	Outer skin temperature	[a]Ambient to ambient + 885° F
PC03	Outer skin temperature	[a]Ambient to ambient + 885° F
PC06	Outer skin temperature	[a]Ambient to ambient + 885° F
PC07	Outer skin temperature	[a]Ambient to ambient + 885° F
PC15	Outer skin temperature	[a]Ambient to ambient + 873° F
PD03	Outer skin temperature	[a]Ambient to ambient + 1000° F
PD04	Outer skin temperature	[a]Ambient to ambient + 1000° F
PD07	Outer skin temperature	[a]Ambient to ambient + 1010° F
PD08	Outer skin temperature	[a]Ambient to ambient + 1005° F
PD09	Outer skin temperature	[a]Ambient to ambient + 1020° F
PD10	Outer skin temperature	[a]Ambient to ambient + 1030° F
PD11	Outer skin temperature	[a]Ambient to ambient + 1020° F
PD56	Cabin wall temperature	0 to + 300° F
PD57	Cabin wall temperature	0 to + 300° F
PF03	Outer skin temperature (inside)	[a]Ambient to ambient + 873° F
PD19	Horizon sensor bump, center	[a]Ambient to ambient + 873° F
PD20	Horizon sensor bump, inflection	[a]Ambient to ambient + 1190° F
PD21	Horizon sensor bump, inflection	[a]Ambient to ambient + 1190° F
PD28	Window, RH-in-outer pane temperature	[a]Ambient to ambient + 880° F
PD29	Window, RH-inner pane temperature	0 to + 300° F
PD30	Window, LH-in-outer pane temperature	[a]Ambient to ambient + 890° F
PD58	Window, RH-in-outer pane temperature	[a]Ambient to ambient + 873° F

[a]Ambient temperature determined either by reference junction temperature MA28 or MA29

TABLE 3-I.- SPACECRAFT INSTRUMENTATION MEASUREMENTS FOR GT-1 - Continued

(a) Low-level commutator (link 1 and link 2) - Continued

Measurement	Description	Range
PD42	Inner skin temperature	0 to + 400° F
PD43	Outer skin temperature	[a]Ambient to ambient + 1020° F
PD44	Outer skin temperature	[a]Ambient to ambient + 1020° F
PD45	Outer skin temperature	[a]Ambient to ambient + 1200° F
PD46	Outer skin temperature	[a]Ambient to ambient + 1200° F
PD47	Outer skin temperature	[a]Ambient to ambient + 1205° F
PD48	Outer skin temperature	[a]Ambient to ambient + 1155° F
PD49	Outer skin temperature	[a]Ambient to ambient + 1000° F
PD50	Outer skin temperature	[a]Ambient to ambient + 1010° F
PD51	Outer skin temperature	[a]Ambient to ambient + 1045° F
PD52	Outer skin temperature	[a]Ambient to ambient + 1045° F
PF01	Outer skin temperature (inside)	[a]Ambient to ambient + 873° F
PF02	Outer skin temperature (inside)	[a]Ambient to ambient + 873° F
PD59	Window, RH-in-inner pane temperature	0 to + 300° F
PF04	Outer skin temperature (inside)	[a]Ambient to ambient + 873° F
PF06	Shape charge, (retrograde-adapter)	0 to + 300° F
PF07	Shape charge temperature (retrograde-adapter)	0 to + 300° F
PF08	Shape charge temperature	0 to + 300° F
PF09	Outer skin temperature	[a]Ambient to ambient + 890° F
PF10	Outer skin temperature	[a]Ambient to ambient + 890° F
PF11	Outer skin temperature	[a]Ambient to ambient + 885° F
BH03	Main bus total current	0 to 7.6 amps
PF13	Outer skin temperature	[a]Ambient to ambient + 890° F
PF14	Stringer temperature between tube-skin	[a]Ambient to ambient + 890° F
[b]PF15	Stringer 58 bottom of tube temperature	[a]Ambient to ambient + 890° F
PF16	Stringer 16 between tube-skin temperature	[a]Ambient to ambient + 890° F
PF17	Stringer 16 bottom of tube temperature	[a]Ambient to ambient + 895° F
PG01	Outer skin (inside) temperature	[a]Ambient to ambient + 873° F
PG02	Outer skin (inside) temperature	[a]Ambient to ambient + 873° F
PG03	Outer skin (inside) temperature	[a]Ambient to ambient + 873° F

[a]Ambient temperature determined either by reference junction temperature MA28 or MA29

[b]Sensor disabled due to improper operation

TABLE 3-I.- SPACECRAFT INSTRUMENTATION MEASUREMENTS FOR GT-1 - Continued

(a) Low-level commutator (link 1 and link 2) - Concluded

Measurement	Description	Range
PG04	Outer skin (inside) temperature	[a]Ambient to ambient + 873° F
PG05	Equipment to retrograde fairing temperature	[a]Ambient to ambient + 873° F
PG06	Outer skin (inside) temperature	[a]Ambient to ambient + 873° F
PG07	Inter ring temperature	0 to + 300° F
PG08	Inter ring temperature	0 to + 300° F
PG09	Lower ring (launch vehicle attachment) temperature	0 to + 300° F
PG10	Lower ring (launch vehicle attachment) temperature	0 to + 300° F
PG14	Inter ring temperature	0 to + 300° F
PG15	Inter ring temperature	0 to + 300° F
PG16	Outer skin temperature	[a]Ambient to ambient + 873° F
PG17	Outer skin temperature	[a]Ambient to ambient + 873° F
PG18	Outer skin temperature	[a]Ambient to ambient + 873° F
PG19	Outer skin temperature	[a]Ambient to ambient + 890° F
PG20	Pyrotechnic launch vehicle temperature	0 to + 300° F
PG21	Outer skin (ring) temperature	[a]Ambient to ambient + 890° F
PG22	Equipment to retrograde fairing temperature	[a]Ambient to ambient + 873° F
PG23	Equipment to retrograde fairing temperature	[a]Ambient to ambient + 873° F
PG24	Equipment to retrograde fairing temperature	[a]Ambient to ambient + 873° F

[a]Ambient temperature determined either by reference junction temperature MA28 or MA29

TABLE 3-I.- SPACECRAFT INSTRUMENT MEASUREMENTS FOR GT-1 - Continued

(b) High-level commutator (link 2 and link 3)

Measurement	Description	Range
MA18	High-level full-scale	+5.0 v d-c
MA17	High-level, zero reference	0.0 v d-c
BG01	Main bus voltages	+15 to +35 v
CB01	Cabin Press (referenced to forward compartment)	0 to 6 psid
CB07	Forward compartment absolute pressure (ref.)	0 to 15 psia
KB02	Static pressure	0 to 15 psia
QB04	Equipment compartment absolute pressure (RH)	0 to 15 psia
QB09	Landing gear compartment absolute pressure	0 to 15 psia
QB10	Equipment compartment absolute pressure (LH)	0 to 15 psia
QC14	Retrograde compartment absolute pressure	0 to 15 psia
QC15	Equipment compartment absolute pressure	0 to 15 psia
QD06	Cover cavity absolute pressure	0 to 15 psia
QG00	Acoustic noise microphones	------------
QG01	Sound pressure level 37.5 cps to 9.6 kc	105 to 131 db
QG02	Sound pressure level 37.5 cps to 9.6 kc	129 to 155 db
QG03	Sound pressure level 37.5 to 75 cps	90 to 116 db
QG04	Sound pressure level 37.5 to 75 cps	114 to 140 db
QG05	Sound pressure level 75.0 to 150 cps	95 to 121 db
QG06	Sound pressure level 75.0 to 150 cps	119 to 145 db
QG07	Sound pressure level 150 to 300 cps	95 to 121 db
QG08	Sound pressure level 150 to 300 cps	119 to 145 db
QG09	Sound pressure level 300 to 600 cps	100 to 126 db
QG10	Sound pressure level 300 to 600 cps	124 to 150 db
QG11	Sound pressure level 600 to 1200 cps	100 to 126 db
QG12	Sound pressure level 600 to 1200 cps	124 to 150 db
QG13	Sound pressure level 1.2 to 2.4 kc	95 to 121 db
QG14	Sound pressure level 1.2 to 2.4 kc	119 to 145 db
QG15	Sound pressure level 2.4 to 4.8 kc	90 to 116 db
QG16	Sound pressure level 2.4 to 4.8 kc	114 to 140 db
QG17	Sound pressure level 4.8 to 9.6 kc	90 to 116 db
QG18	Sound pressure level 4.8 to 9.6 kc	114 to 140 db
AB09	Scanner-fairing break wire 1	Bilevel
AB10	Scanner-fairing break wire 2	Bilevel

TABLE 3-I.- SPACECRAFT INSTRUMENTATION MEASUREMENTS FOR GT-1 - Concluded

(c) Transmitter 246.3 (link 1)

Measurement	VCO, kc	Description	Range
QA09	3.0	X-axis vibration	+2g to -2g
QA10	3.9	Y-axis vibration	+2g to -2g
QA11	5.4	Z-axis vibration	+4g to -4g
	10.5	Commutated (low-level)	
QC17	40.0	Z-axis vibration	+16g to -16g
QC22	52.5	Radial vibration	+100g to -100g
QB12	22.0	X-axis vibration	+16g to -16g

(d) Transmitter 259.7 (link 2)

Measurement	VCO, kc	Description	Range
	10.5	Commutated (high-level)	
	14.5	Commutated (low-level)	
QB13	40.0	Y-axis vibration	+16g to -16g
QB14	70.0	Z-axis vibration	+16g to -16g

(e) Transmitter 230.4 (link 3)

Measurement	VCO, kc	Description	Range
KA03	1.3	Vertical (Y-axis)	+3g to -3g
KA02	1.7	Lateral (X-axis)	+3g to -3g
KA01	2.3	Longitudinal (Z-axis)	-3g to +19g
QD10	3.0	X-axis vibration	+4g to -4g
	10.5	Commutated (high-level)	
QC24	40.0	Z-axis vibration blast shield (TY)	+16g to -16g
QC26	52.6	Radial vibration	+32g to -32g
QC23	22.0	Radial vibration	+100g to -100g
QD11	3.9	Y-axis vibration	+4g to -4g

TABLE 3-II.- LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS FOR GT-1

Measurement	Description	Instrumentation range
0001	Thrust chamber valve position subassembly 1	0 to 100 percent open
0002	Thrust chamber valve position subassembly 2	0 to 100 percent open
0003	Thrust chamber pressure, subassembly 1	0 to 1,000 psia
0004	Thrust chamber pressure, subassembly 2	0 to 1,000 psia
0005	Gas-generator chamber pressure, subassembly 1	0 to 750 psia
0006	Gas-generator chamber pressure, subassembly 2	0 to 1,000 psia
0007	Turbine speed, subassembly 1	0 to 40,000 rpm
0008	Turbine speed, subassembly 2	0 to 40,000 rpm
0009	Turbine inlet temperature, subassembly 1	0 to 2,500° F
0010	Fuel pump discharge pressure, subassembly 1	0 to 1,500 psia
0011	Fuel pump discharge pressure, subassembly 2	0 to 1,500 psia
0012	Turbine inlet temperature, subassembly 2	0 to 2,500° F
0013	Fuel pump inlet temperature, subassembly 1	0 to 200° F
0014	Fuel pump inlet pressure, subassembly 1	0 to 100 psia
0015	Oxidizer pump discharge pressure, subassembly 1	0 to 1,500 psia
0016	Oxidizer pump discharge pressure, subassembly 2	0 to 1,500 psia
0017	Oxidizer pump inlet pressure, subassembly 2	0 to 200 psia

TABLE 3-II.- LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS
FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0021	Fuel pressurant orifice inlet pressure, subassembly 2	0 to 500 psia
0022	Fuel pressurant orifice inlet temperature, subassembly 2	0 to 500° F
0023	Oxidizer pump inlet temperature, subassembly 2	0 to 200° F
0026	Oxidizer pressurant orifice inlet subassembly 2	0 to 1,000 psia
0027	Oxidizer pressurant orifice inlet temperature, subassembly 2	0 to 500° F
0028	Bootstrap fuel venturi inlet pressure, subassembly 1	0 to 1,500 psia
0029	Bootstrap fuel venturi inlet pressure, subassembly 2	0 to 1,500 psia
0030	Bootstrap oxidizer venturi inlet pressure, subassembly 1	0 to 1,500 psia
0031	Bootstrap oxidizer venturi inlet pressure, subassembly 2	0 to 1,500 psia
0032	Power-on TCVPSVORS (87FS2) stage I	Bilevel
0033	Pressure, oxidizer standpipe, subassembly 1	0 to 200 psia
0034	Pressure, oxidizer standpipe, subassembly 2	0 to 200 psia
0035	Piston motion, fuel surge chamber, subassembly 1	0 to 7 in.
0036	Piston motion, fuel surge chamber, subassembly 2	0 to 7 in.
0037	Pressure, fuel surge chamber, subassembly 1	0 to 100 psia
0038	Pressure, fuel surge chamber, subassembly 2	0 to 100 psia

TABLE 3-II. - LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS

FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0050	Fuel sensor shutdown, stage I	Bilevel
0052	Fuel sensor outage, stage I	Bilevel
0053	Fuel sensor outage, stage I	Bilevel
0054	Fuel sensor high, stage I	Bilevel
0055	Fuel sensor high, stage I	Bilevel
0056	Oxidizer sensor high, stage I	Bilevel
0057	Oxidizer sensor high, stage I	Bilevel
0058	Oxidizer sensor outage, stage I	Bilevel
0059	Oxidizer sensor outage, stage I	Bilevel
0060	Fuel sensor shutdown, stage I	Bilevel
0150	Travel actuator 1, pitch, stage I	± 1.25 in.
0151	Travel actuator 2, yaw-roll, stage I	± 1.25 in.
0152	Travel actuator 3, yaw-roll, stage I	± 1.25 in.
0153	Travel actuator 4, pitch, stage I	± 1.25 in.
0154	Pressure, hydraulic system (primary), stage I	0 to 4,500 psia
0155	Fluid-level, hydraulic reservoir (primary), stage I	0 to 100 percent
0156	Hydraulic fluid temperature (primary), stage I	0 to 300° F
0157	Pressure, hydraulic system (redundant), stage I	0 to 4,500 psia
0158	Fluid-level, hydraulic reservoir (redundant), stage I	0 to 100 percent
0159	Hydraulic fluid temperature (redundant), stage I	0 to 300° F

TABLE 3-II.- LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS
FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0169	Acceleration, axial, vehicle compartment 5	± 10g
0171	Acceleration, lateral, vehicle compartment 5	± 2g
0172	Acceleration, vertical, vehicle compartment 5	± 2g
0230	Rate gyro output, pitch, stage I (primary)	± 12.5 deg/sec
0231	Rate gyro output, yaw, stage I (primary)	± 12.5 deg/sec
0232	Rate gyro output, roll, stage I (primary)	± 12.5 deg/sec
0233	Rate gyro output, pitch, stage I (redundant)	± 12.5 deg/sec pitch
0234	Rate gyro output, yaw, stage I (redundant)	± 12.5 deg/sec yaw
0235	Rate gyro output, roll, stage I (redundant)	± 12.5 deg/sec roll
0355	APS/IPS (Comp. 4)	0 to 35 v d-c
0356	Subassembly 1, MDTCPS A and B (B.C.)	Step voltage
0357	Subassembly 2, MDTCPS A and B (B.C.)	Step voltage
0358	Subassembly 1, MDTCPS A and B	Bilevel
0359	Subassembly 2, MDTCPS A and B	Bilevel
0364	Fuel tank pressure (A), stage 1	0 to 50 psia
0365	Fuel tank pressure (B), stage 1	0 to 50 psia
0366	Oxidizer tank pressure (A), stage 1	0 to 50 psia
0367	Oxidizer tank pressure (B), stage 1	0 to 50 psia

TABLE 3-II.- LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0501	Thrust chamber valve position, subassembly 3	0 to 100 percent (open)
0502	Thrust chamber pressure, subassembly 3	0 to 1,000 psia
0503	Turbine inlet pressure, subassembly 3	0 to 1,000 psia
0504	Turbine speed, subassembly 3	0 to 40,000 rpm
0505	Turbine inlet temperature, subassembly 3	0 to 2,500° F
0506	Fuel-pump discharge pressure, subassembly 3	0 to 1,500 psia
0507	Fuel-pump inlet pressure, subassembly 3	0 to 100 psia
0508	Fuel-pump inlet temperature, subassembly 3	0 to 300° F
0509	Oxidizer pump discharge pressure, subassembly 3	0 to 1,500 psia
0510	Oxidizer pump inlet pressure, subassembly 3	0 to 100 psia
0512	Fuel pressurant orifice inlet pressure, subassembly 3	0 to 500 psia
0513	Fuel pressurant orifice inlet temperature, subassembly 3	0 to 500° F
0514	Oxidizer pump inlet temperature, subassembly 3	0 to 300° F
0517	Bootstrap fuel venturi inlet pressure, subassembly 3	0 to 1,500 psia
0518	Bootstrap oxidizer venturi inlet pressure, subassembly 3	0 to 1,500 psia
0519	Power on TCVPSVORS (91FS2), stage II	Bilevel
0540	Fuel sensor high, stage II	Bilevel
0541	Fuel sensor high, stage II	Bilevel
0542	Oxidizer sensor high, stage II	Bilevel

TABLE 3-II.- LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS
FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0543	Oxidizer sensor high, stage II	Bilevel
0544	Fuel sensor shutdown, stage II	Bilevel
0545	Oxidizer sensor shutdown, stage II	Bilevel
0546	Fuel sensor outage, stage II	Bilevel
0547	Fuel sensor outage, stage II	Bilevel
0548	Oxidizer sensor outage, stage II	Bilevel
0549	Oxidizer sensor outage, stage II	Bilevel
0550	Oxidizer sensor shutdown, stage II	Bilevel
0551	Fuel sensor shutdown, stage II	Bilevel
0650	Travel actuator 5, yaw, stage II	± 0.505 in.
0651	Travel actuator 6, pitch, stage II	± 0.505 in.
0652	Travel actuator 7, roll, stage II	± 1.43 in.
0653	Pressure, hydraulic system, stage II	0 to 4,500 psia
0654	Fluid-level hydraulic reservoir, stage II	0 to 100 percent
0655	Hydraulic fluid temperature, stage II	0 to 300° F
0670	Acceleration, axial, vehicle compartment 1	± 10g
0671	Acceleration, lateral, vehicle compartment 1	± 2g
0672	Acceleration, vertical, vehicle compartment 1	± 2g
0673	Skin temperature	0 to 600° F

TABLE 3-II. - LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS

FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0674	Skin temperature	0 to 600° F
0675	Skin temperature	0 to 600° F
0676	Skin temperature	0 to 600° F
0677	Skin temperature	0 to 600° F
0678	Skin temperature	0 to 600° F
0679	Skin temperature	0 to 600° F
0680	Skin temperature	0 to 600° F
0681	Skin temperature	0 to 600° F
0682	Skin temperature	0 to 600° F
0683	Calorimeter slug temperature	0 to 600° F
0684	Calorimeter slug temperature	0 to 600° F
0685	Calorimeter slug temperature	0 to 600° F
0686	Calorimeter slug temperature	0 to 600° F
0687	Calorimeter slug temperature	0 to 600° F
0699	Acceleration, axial, low range	± 0.5g
0720	Displacement gyro output, pitch, stage II	± 6 deg
0721	Displacement gyro output, yaw, stage II	± 6 deg
0722	Displacement gyro output, roll, stage II	± 6 deg
0723	Rate gyro output, pitch, stage II (primary)	± 12.5 deg/sec
0193	Rate gyro output, yaw, stage II (primary)	± 12.5 deg/sec
0725	Rate gyro output, roll, stage II (primary)	± 12.5 deg/sec
0726	25 v d-c power supply voltage	0-39.4 v d-c
0727	800 cps power supply voltage (primary or redundant)	20 to 30 v a-c

TABLE 3-II. - LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0728	TARS discrete (stage 1 gain change)	Bilevel
0729	Autopilot output pitch sub-assembly 3 (primary or redundant)	± 0.7 v d-c
0730	Autopilot output, yaw, sub-assembly 3 (primary or redundant)	± 0.7 v d-c
0731	Autopilot output, roll, sub-assembly 1 (primary or redundant)	± 1.2 v d-c
0732	Displacement gyro torquer monitor, pitch	± 0.80 v d-c
0733	Displacement gyro torquer monitor, yaw	± 0.80 v d-c
0734	Displacement gyro torquer monitor, roll	± 0.80 v d-c
0735	TARS discrete (arm stage 1 shutdown sensor)	Bilevel
0736	Rate gyro output, pitch, stage II (redundant)	± 12.5 deg/sec
0737	Rate gyro output, yaw, stage II (redundant)	± 12.5 deg/sec
0738	Rate gyro output, roll, stage II (redundant)	± 12.5 deg/sec
0739	TARS discrete (arm stage II shutdown sensors)	Bilevel
0740	TARS discrete (guidance initiate)	Bilevel
0741	IPS staging arm timer actuation	Bilevel
0746	Rate beacon-guidance + 30 volt supply	0 to 5 v d-c
0747	Pulse beacon-guidance + 15 volt supply	0 to 5 v d-c
0748	Decoder-guidance + 10 volt supply	0 to 5 v d-c
0749	Rate beacon-guidance received signal no. 2	0 to 5 v d-c

TABLE 3-II. - LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS
FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0750	Rate beacon-guidance received signal no. 1	50 to 85 dbm
0751	Rate beacon-guidance PH detect	Step voltage
0752	Rate beacon-guidance power output	Step voltage
0753	Pulse beacon-guidance MAG current	Step voltage
0754	Pulse beacon-guidance AGC	-10 to -65 dbm
0755	Pitch output-guidance	± 100 percent
0756	Yaw output, guidance	± 100 percent
0757	Decoder guidance discrete no. 1	Bilevel
0758	Decoder guidance discrete no. 2	Bilevel
0759	Decoder guidance discrete no. 3	Bilevel
0760	Decoder guidance discrete no. 4	Bilevel
0762	Autopilot output pitch subassembly 1 (primary)	± 1.2 v d-c
0763	Autopilot output, yaw, subassembly 1 (primary)	± 1.2 v d-c
0764	Autopilot output, pitch, subassembly 2 (primary)	± 1.2 v d-c
0765	Autopilot output, yaw, subassembly 2 (primary)	± 1.2 v d-c
0766	Adapter package output, pitch	± 6.0 v d-c
0767	Adapter package output, yaw	± 6.0 v d-c
0768	Adapter package output, roll	± 6.0 v d-c
0769	Autopilot output, yaw subassembly 1 (redundant)	± 1.2 v d-c
0770	Autopilot output, pitch, subassembly 1 (redundant)	± 1.2 v d-c

TABLE 3-II.- LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS
FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0771	Autopilot output, pitch, subassembly 2 (redundant)	± 1.2 v d-c
0772	Autopilot output, yaw, subassembly 2 (redundant)	± 1.2 v d-c
0773	IGS stage I gain change discrete	Bilevel
0780	AGC command receiver no. 1 RSS and burst	10 to 40 μv
0781	AGC command receiver no. 2 RSS and burst	10 to 20 μv
0782	Engine cut-off receiver no. 1	Bilevel
0783	Engine cut-off receiver no. 2	Bilevel
0784	AGC range channel MISTRAM transponder	-40 to -110 dbm
0785	AGC calibration channel, MISTRAM transponder	-40 to -110 dbm
0786	RF output range channel, MISTRAM transponder	Calibration voltage
0787	RF output calibration channel, MISTRAM transponder	Calibration voltage
0788	Phase detector calibration channel MISTRAM transponder	Calibration voltage
0789	Phase detector range channel MISTRAM transponder	Step voltage
0800	IPS bus voltage	15 to 35 v d-c
0801	APS bus voltage	0 to 37.5 v d-c
0802	a-c bus voltage phase A (400 cps)	105 to 125 v a-c
0803	a-c bus frequency phase A	380 to 420 cps
0804	IPS battery current	0 to 150 Amps
0805	APS battery current	0 to 150 Amps
0810	Instrument voltage, compartment 2	0 to 6 v d-c

TABLE 3-II. - LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0811	Temperature bridge power supply compartment 2	35.5 to 45 v d-c
0812	Signal conditioner package temperature	0 to 200° F
0813	PCM mercury cell voltage	1.35 v d-c
0814	PCM mercury cell voltage	1.35 v d-c
0815	PCM mercury cell voltage	1.35 v d-c
0816	Signal conditioner power supply, positive regulated	29.9 to 30.1 v d-c
0817	Signal conditioner power supply, negative regulated	29.9 to 30.1 v d-c
0842	Pitch SMRD (BH bypass)	Bilevel
0843	Pitch SMRD (BL bypass)	Bilevel
0844	Yaw SMRD (BH bypass)	Bilevel
0845	Yaw SMRD (BL bypass)	Bilevel
0846	Roll SMRD (BH bypass)	Bilevel
0847	Roll SMRD (BL bypass)	Bilevel
0848	Overrate warning	Bilevel
0853	Subassembly MDTCPS A and B	Bilevel
0856	Shutdown lockout timers 1 and 2	Bilevel
0858	Shutdown switches reset monitor	Bilevel
0859	APS/IPS compartment 2 (RSP)	15 to 30 v d-c
0861	Subassembly MDTCPS A and B (B.C.)	Step voltage
0862	IPS staging	Bilevel
0863	APS staging	Bilevel
0868	Fuel tank pressure (A) stage II	0 to 75 psia
0869	Fuel tank pressure (B) stage II	0 to 75 psia

TABLE 3-II.- LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS
FOR GT-1 - Continued

Measurement	Description	Instrumentation range
0870	Oxidizer tank pressure (A) stage II	0 to 75 psia
0871	Oxidizer tank pressure (B) stage II	0 to 75 psia
0872	Transfer to redundant control system (A)	Bilevel
0873	Transfer command to redundant control system (A)	Bilevel
0874	Transfer to redundant control system (B)	Bilevel
0875	Transfer command to redundant control system (B)	Bilevel
0876	APS to spacecraft	0 to 35 v d-c
0877	Pitch SMRD-B (B.C.)	Step voltage
0878	Yaw SMRD-B (B.C.)	Step voltage
0879	Roll SMRD-B (B.C.)	Step voltage
0880	Subassembly hydraulic switch-over command	Bilevel
0881	Subassembly hydraulic switch-over command	Bilevel
0882	Spacecraft switchover command (A)	Bilevel
0883	Spacecraft switchover command (B)	Bilevel
0884	APS/IPS compartment 2 (Eng. subassembly 3)	15 to 35 v d-c
1003	Thrust chamber pressure, subassembly 1	0 to 1,000 psia
1004	Thrust chamber pressure subassembly 2	0 to 1,000 psia
1017	Oxidizer pump inlet pressure (T-0 to 87FS2-5) subassembly 2	0 to 200 psia
1086	Pressure oxidizer tank dome centerline stage 1 (87FS2-5 to stage separation)	0 to 100 psia

TABLE 3-II. - LAUNCH-VEHICLE INSTRUMENTATION MEASUREMENTS

FOR GT-1 - Concluded

Measurement	Description	Instrumentation range
1169	Acceleration, axial, compartment 5 (T-0 to 87FS2-5)	± 10g
1189	Vibration tandem actuator (axial) stage I (T-0 to 87FS2-5)	± 100g
1502	Thrust chamber pressure, subassembly 3 (87FS2-5 to 87FS2+3)	0 to 1,000 psia
1651	Travel actuator 6, pitch, stage II (87FS2-5 to 87FS2+3)	± 0.503 in.
1670	Acceleration, axial, compartment 1 (T-0 to 87FS2-5)	± 10g
1692	Vibration MOD III rate beacon, axial, (T-0 to 87FS2-5)	± 30g
1861	Subassembly 3 (MDTCPS A and B) (B.C.) (87FS2-5 to 87FS2+3)	Step voltage
1862	IPS staging (87FS2-5 to 87FS2+3)	Bilevel

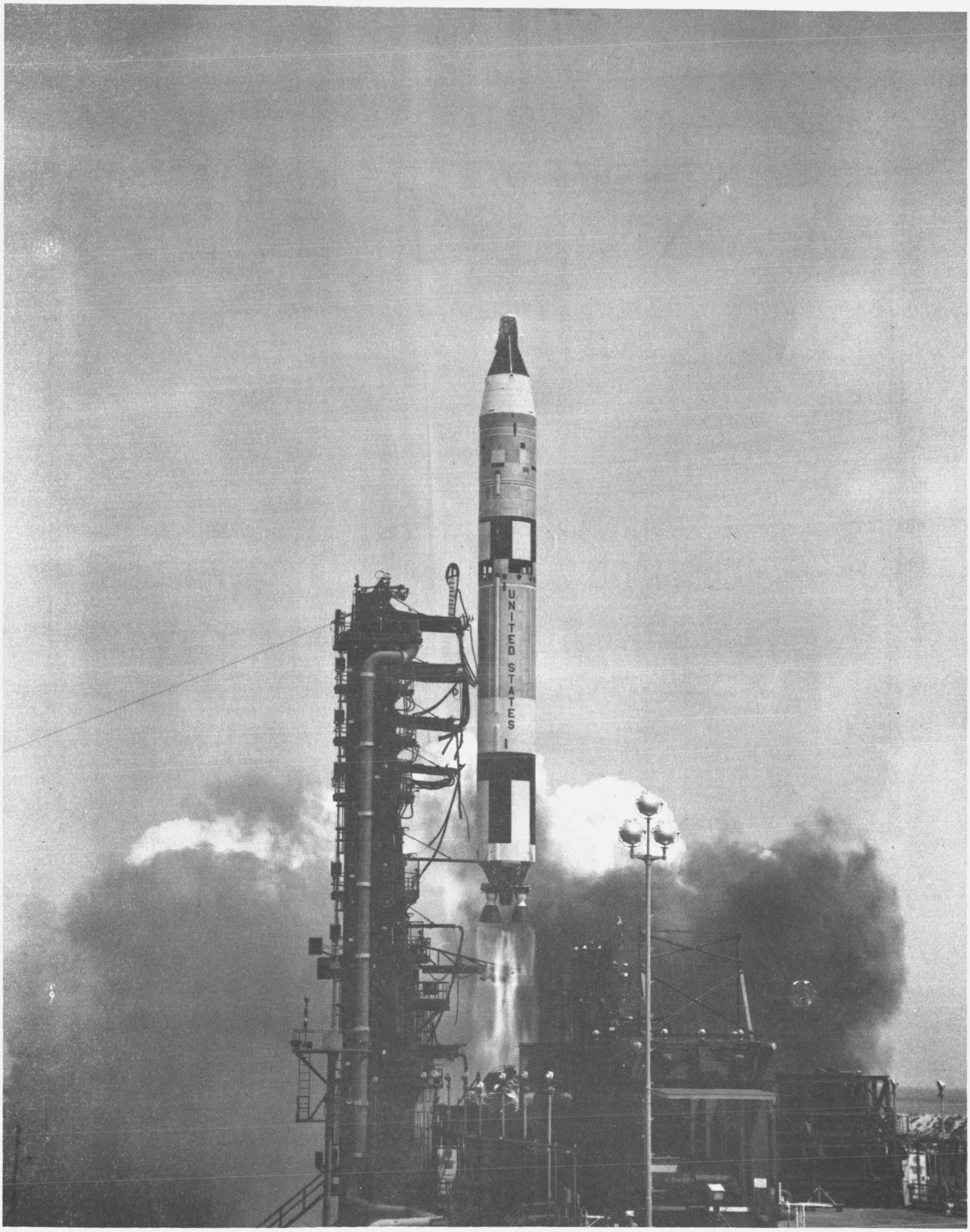


FIGURE 3-1 GT-1 LIFT-OFF CONFIGURATION

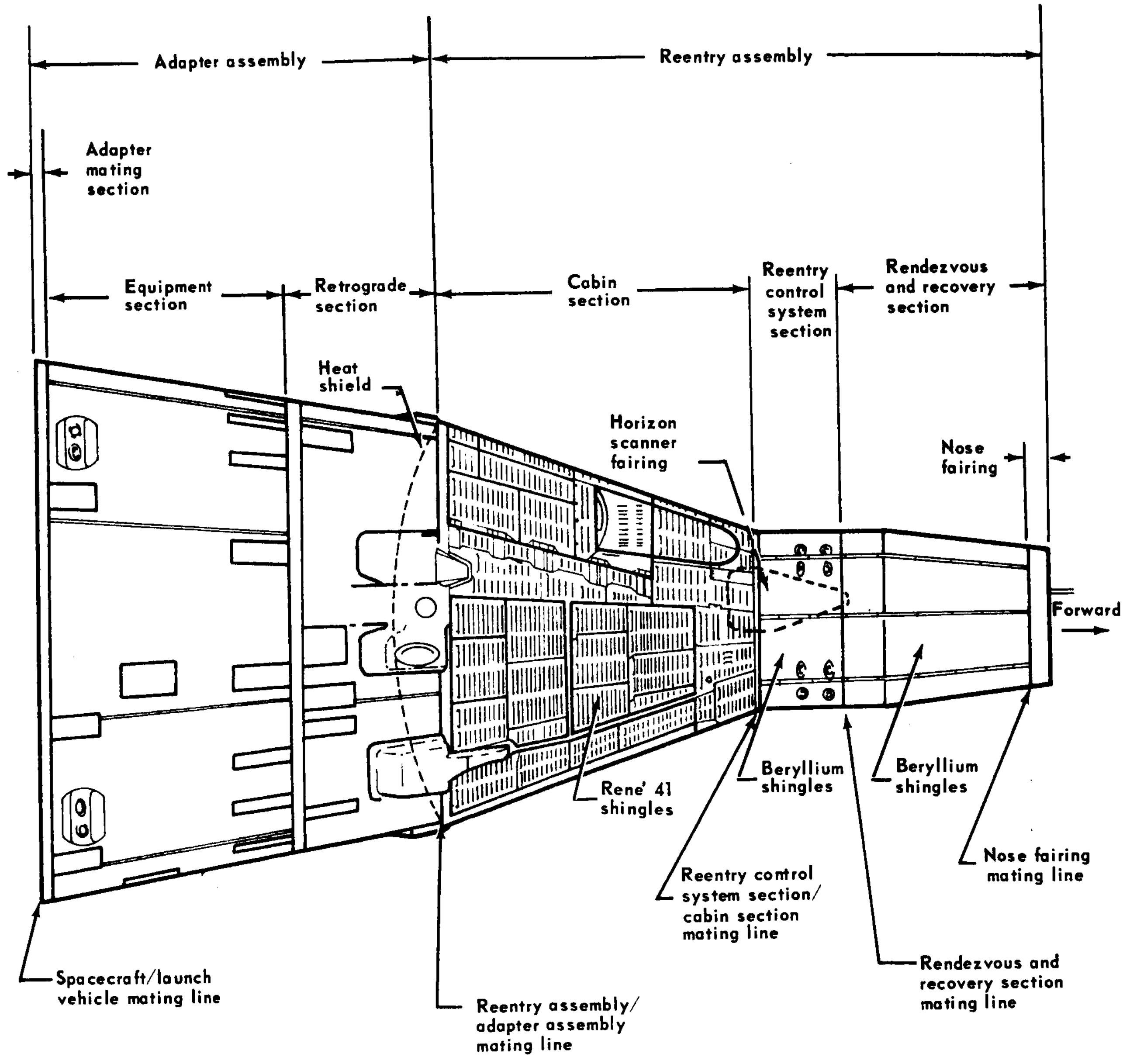


FIGURE 3-2 GEMINI SPACECRAFT

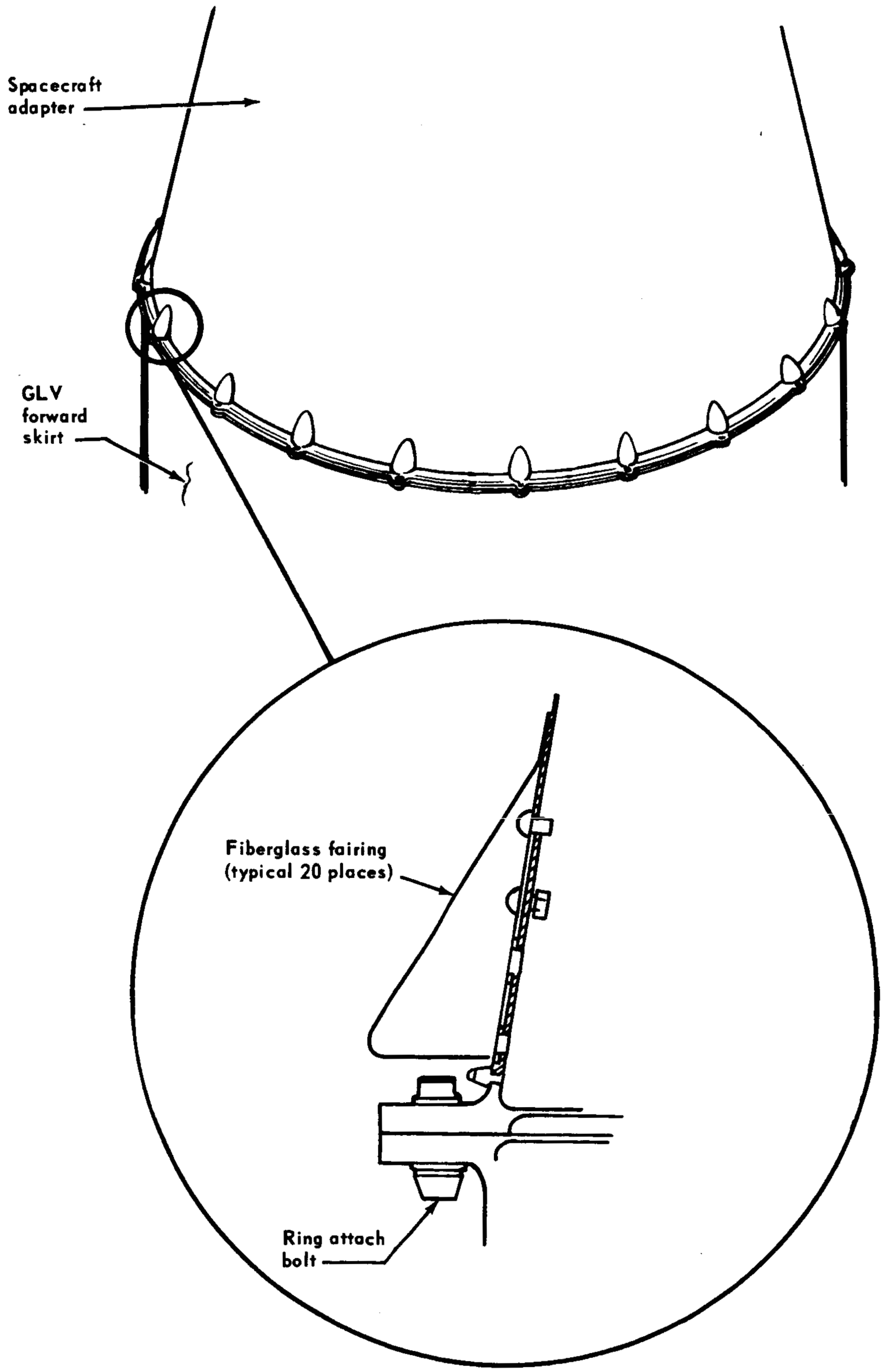


FIGURE 3-3 GLV-SPACECRAFT INTERFACE

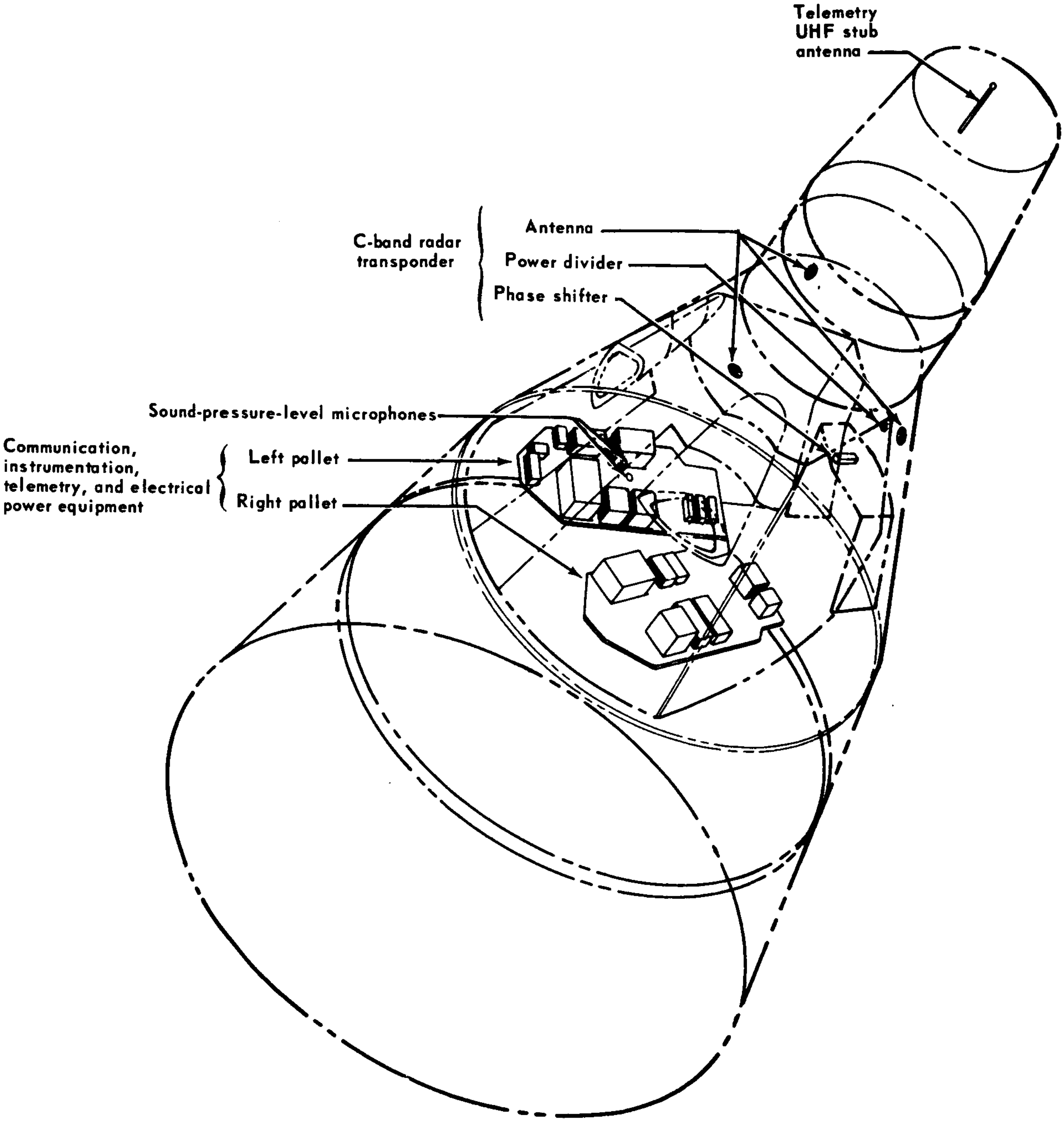
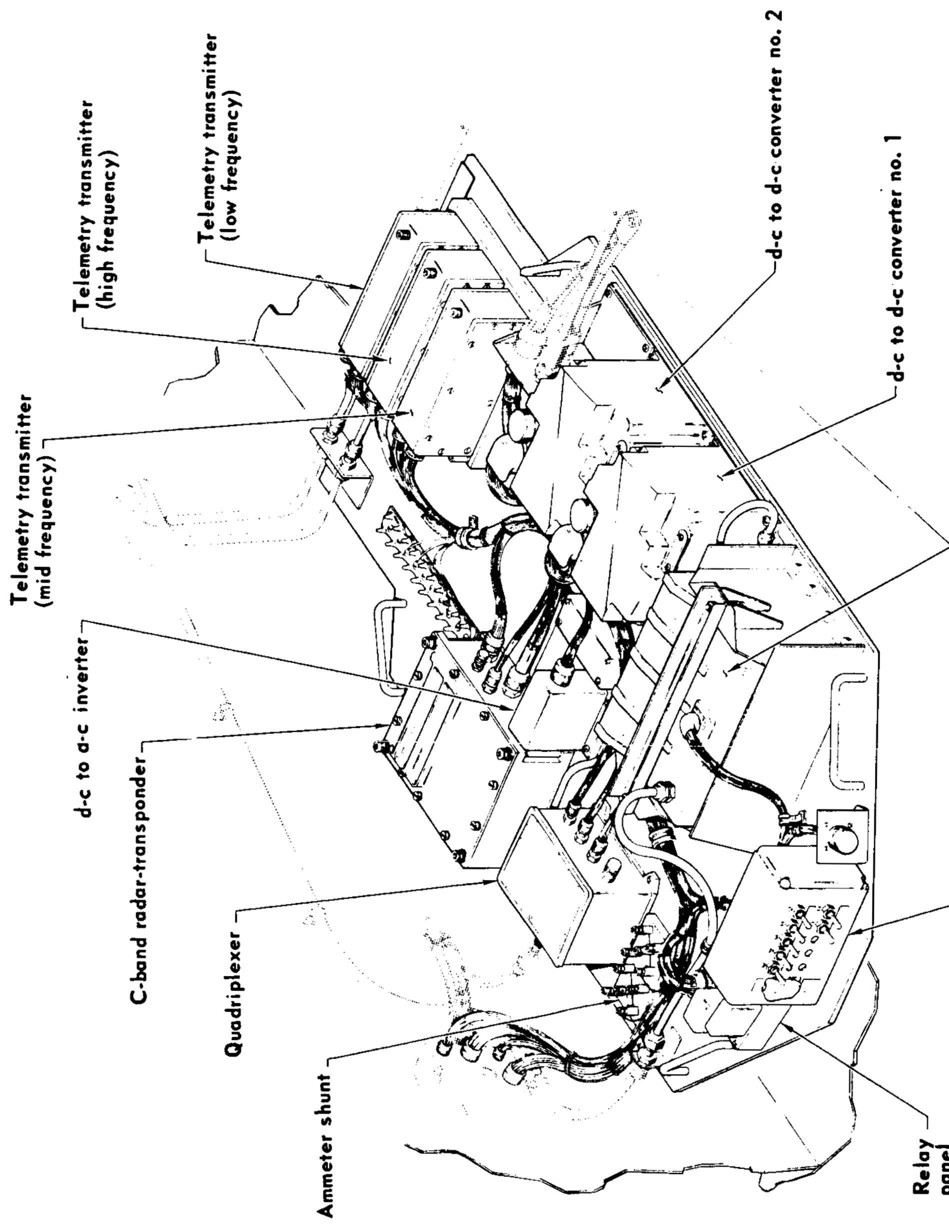


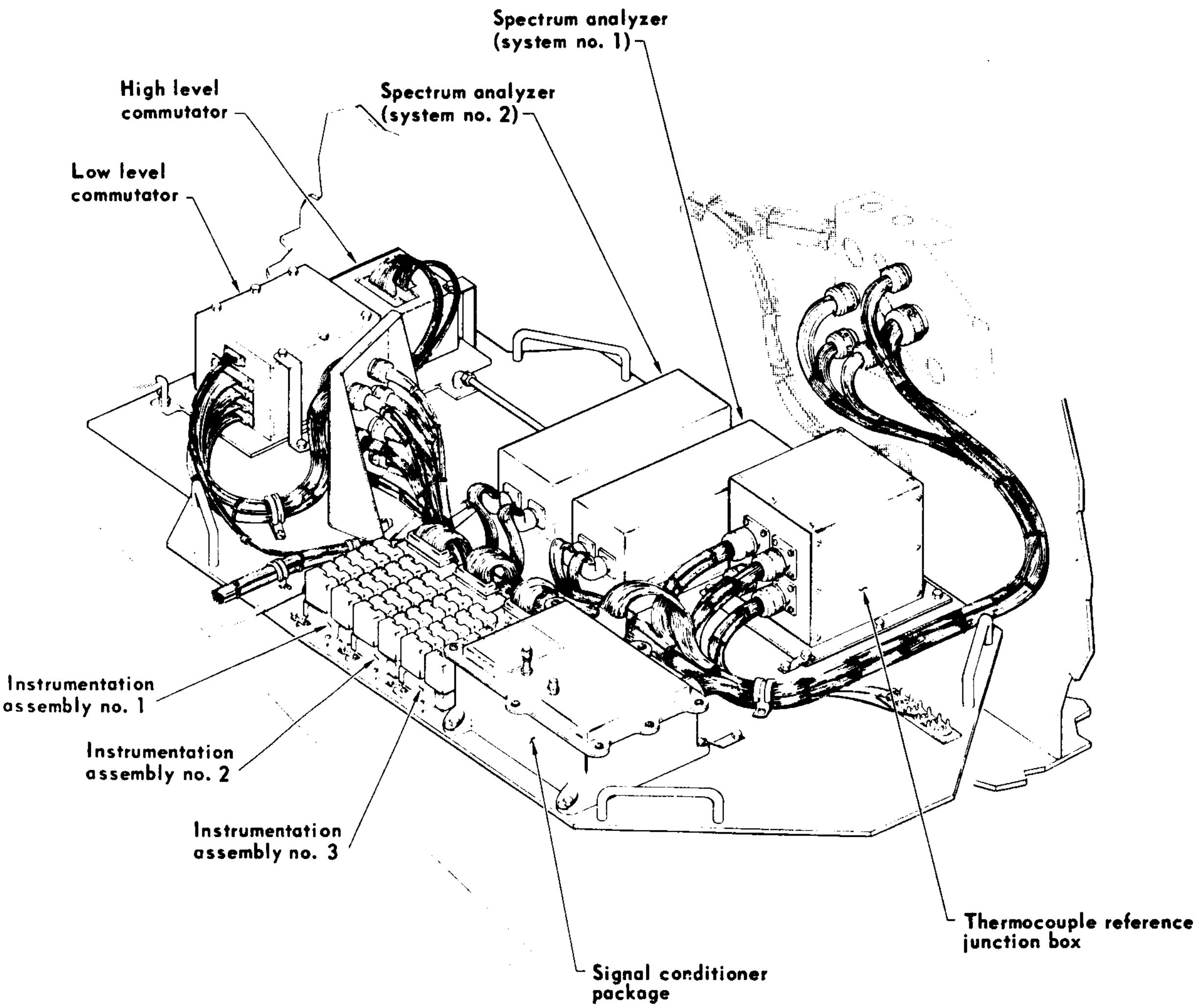
FIGURE 3-4 INSTRUMENTATION, COMMUNICATION, TELEMETRY, AND ELECTRICAL SYSTEM EQUIPMENT LOCATION ON SPACECRAFT



(a) Left pallet

FIGURE 3-5 SPACECRAFT PALLETS

NASA-S-64-3069



(b) Right pallet

FIGURE 3 - 5 CONCLUDED

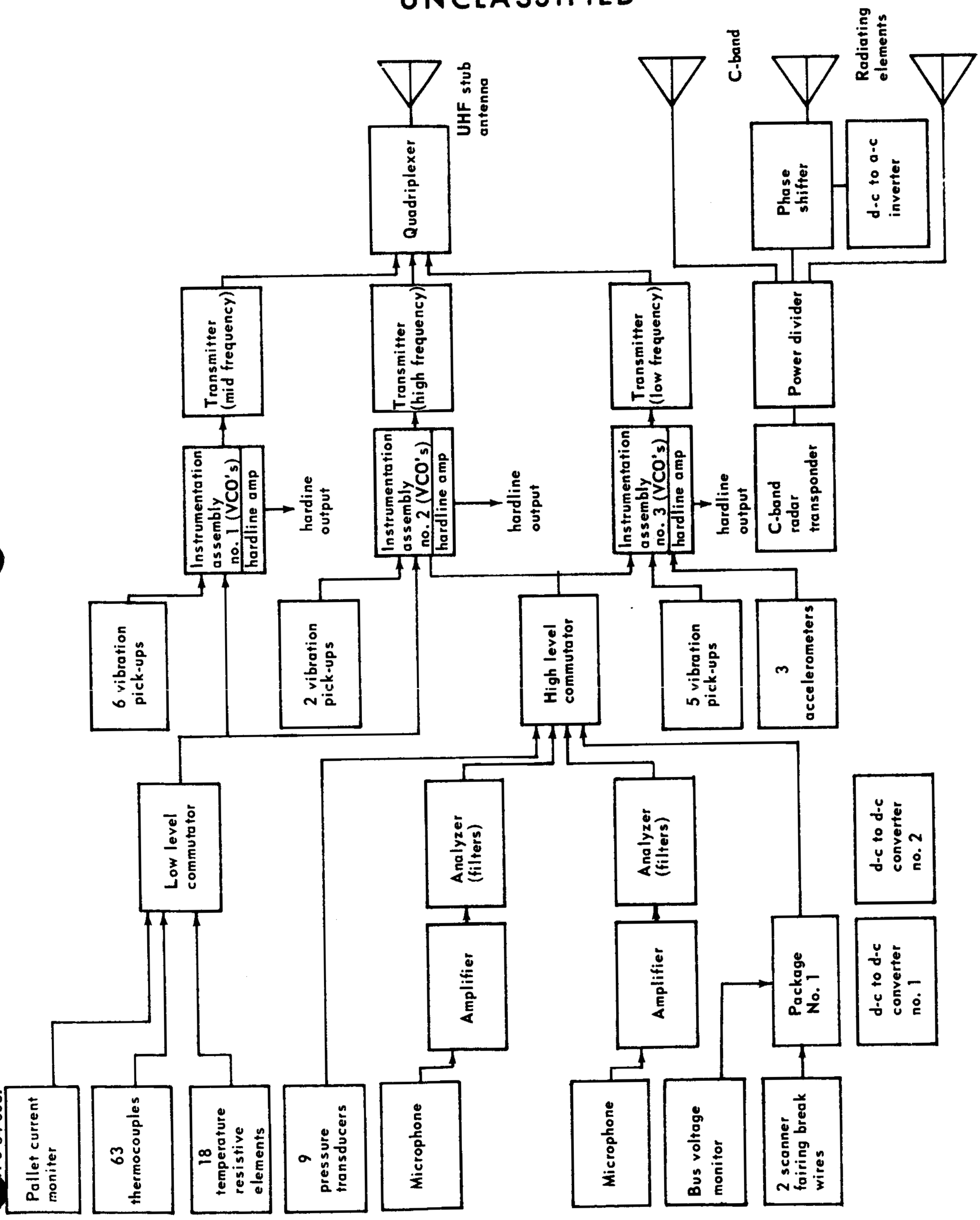


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FIGURE 3-6 INSTRUMENTATION AND COMMUNICATION SYSTEM BLOCK DIAGRAM

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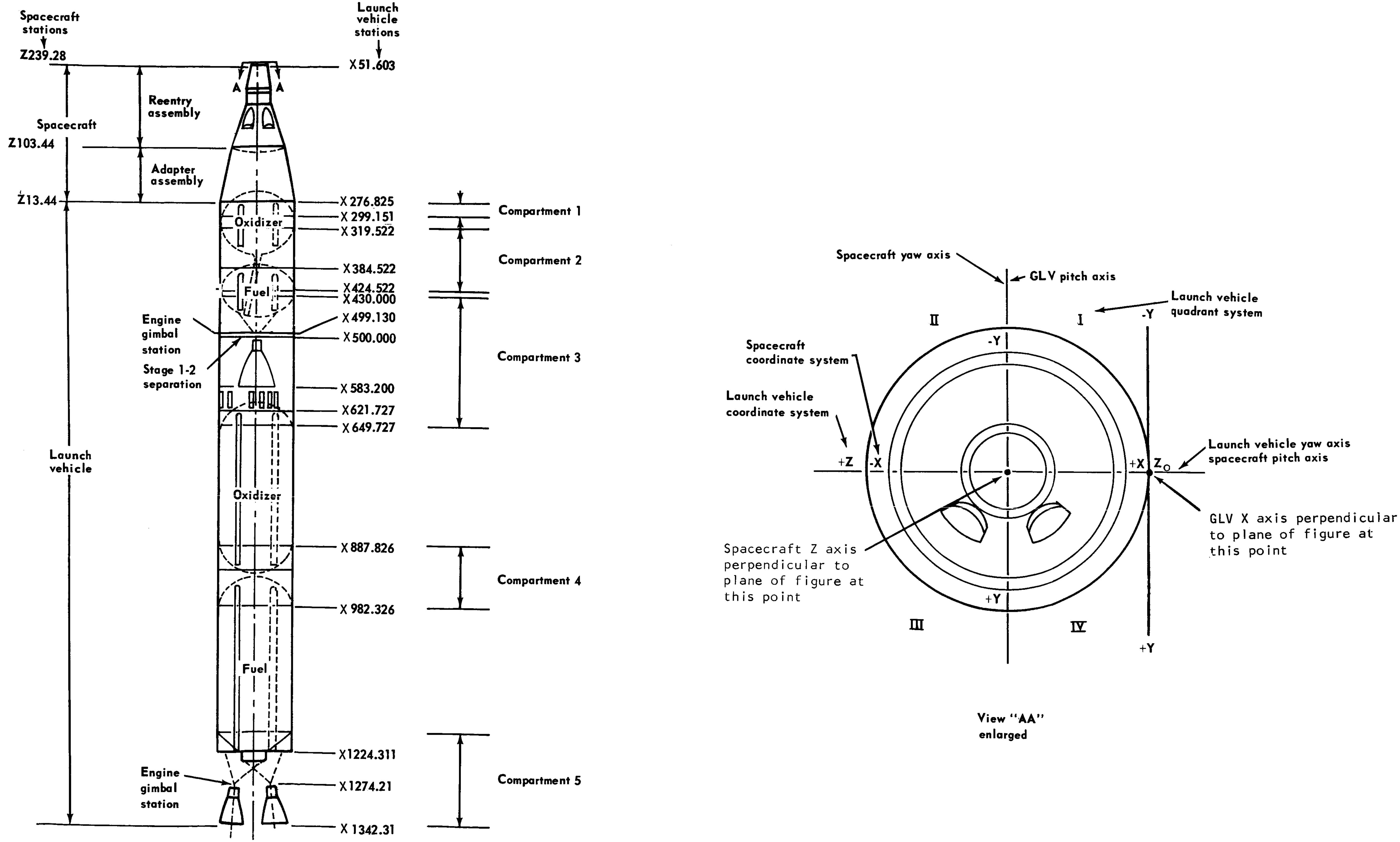

FIGURE 3-7 GEMINI - TITAN VEHICLE

UNCLASSIFIED

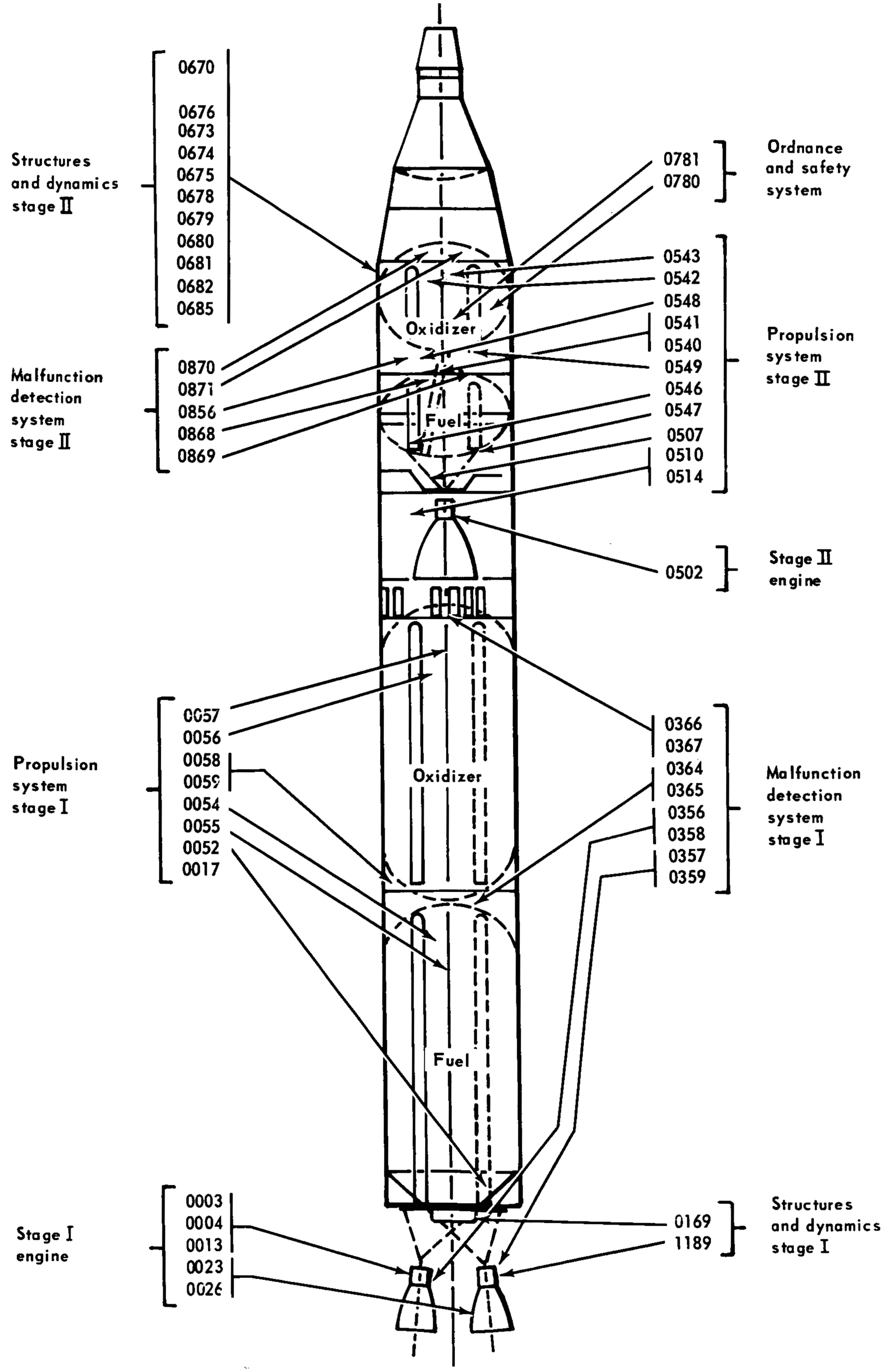

FIGURE 3-8 PERTINENT GLV SENSOR LOCATIONS FOR GT-1 MISSION REPORT

UNCLASSIFIED

4.0 MISSION DESCRIPTION

The GT-1 mission was designed to demonstrate the flight compatibility of the Gemini spacecraft and the Gemini launch vehicle (GLV) combination by a launch into a low earth orbit. It was planned that the launch and orbital environments be similar to those which will be experienced in the future Gemini manned orbital missions. Thus, the GT-1 mission was designed so that the second-stage GLV and the spacecraft would be inserted into an elliptical orbit with a perigee of 87 nautical miles and an apogee of 161 nautical miles referenced to a spherical earth. The launch azimuth of 72.0° was selected to provide an optimum orbital ground track over the Manned Space Flight Network radar and telemetry stations. These stations would provide tracking and telemetry data for orbit determination and systems performance verification. Neither separation of the spacecraft from the second stage of the GLV nor its recovery was planned. Although the trajectory was designed to have an orbital lifetime of several days, the mission was to be considered terminated after three orbital passes.

The actual GT-1 mission was essentially nominal, and all test objectives were satisfied. The launch and insertion into orbit were within tolerance, and a 20 ft/sec overspeed condition at insertion produced an increase of 11 nautical miles in apogee altitude. The radar tracking data indicate that the velocity gained from SECO to insertion was apparently 35 ft/sec higher than expected.

On the second orbital pass, skin tracking radar units detected small objects near the main vehicle. These small objects apparently reentered after a few orbital passes. It was concluded that these objects were small in size and weight and their loss did not appreciably affect the size and weight of the spacecraft and stage II GLV combination.

Active tracking and passive mission support were extended past the planned three orbital passes until it was assumed that the spacecraft and stage II GLV combination had reentered to verify that the loss of these small objects did not affect the ballistic number of the original vehicle which was placed into orbit.

4.1 ACTUAL MISSION

The launch of the GT-1 space vehicle was initiated from the GLV sequencer at T-3.682 seconds by sending an engine start signal to the vehicle. The vehicle was released 2 seconds after both engines reached 77 percent of full thrust.

Lift-off is defined as the time at which the pad disconnects separated after the vehicle had lifted 1.5 inches. The time of lift-off was recorded as 11:00:01.69 a.m. e.s.t. The vehicle rose vertically for 23 seconds. During the vertical rise phase, the vehicle was rolled from a launch stand azimuth of 85.2° to a flight azimuth of 71.9°. At LO + 23.0 seconds the first pitch down rate of 0.6718 deg/sec was initiated by the GLV flight control system. The second pitch-down rate of 0.4687 deg/sec began at LO + 88.1 seconds, and another pitch-down rate of 0.2734 deg/sec was imposed at LO + 118.8 seconds. This maneuver continued through stage separation and ended at 162.0 seconds when the radio guidance system (RGS) was enabled. The GLV flight control system then maintained a zero pitch rate until the first RGS command was received at LO + 168.8 seconds.

The staging sequence was initiated by the stage I thrust-chamber-pressure switch (TCPS) at LO + 153.988 seconds. This signal initiated the following events:

(a) Stage I engine shutdown

(b) Stage II engine ignition

(c) Power removed from the stage I rate gyros

(d) Initiation of a control system gain change

(e) Initiation of separation of stage I from stage II

Results from the telemetry data indicate that separation occurred 0.7 second after the TCPS signal. The second-stage thrust-chamber-pressure build-up was complete at 1.2 seconds after the TCPS signal. This time agrees with the simulation procedure used in the prediction of the nominal trajectory.

The stage II shutdown signal was sent by the ground guidance computer at LO + 339.194 seconds. The backup shutdown signal (ASCO) was also sent at this time. Telemetry data indicate that the stage II shutdown signal occurred at LO + 339.23 seconds. Physical shutdown of the GLV occurred with propellant valve closure starting at LO + 339.25 seconds and continued until LO + 339.67 seconds. At LO + 339.67 seconds the thrust chamber pressure in the second stage engine was approximately zero. At LO + 339.4 seconds, the first indication of thrust decrease was shown in the accelerometer data.

Acceleration data were reduced during the stage II tail-off period and indicated that there was a gain in velocity in excess of the predicted. It is also noted that the second-stage shutdown signal was 3.73 seconds later than the predicted time. Telemetry records indicate that the stage II chamber pressure was lower than nominal, which would account for some of the extended burning time. However, a final analysis cannot be made until a postflight trajectory can be reconstructed.

Insertion, by definition, occurred at second stage shutdown signal time + 20 seconds (LO + 359.230 seconds). The orbit was determined by the Manned Space Flight Network, and characteristics of the orbit are presented in section 4.3 of this report. This section shows that the apogee was 11 miles higher than predicted. This increase in apogee is attributed to the 20 ft/sec excess velocity at insertion. The mission was terminated on the third orbital pass; however, tracking was continued.

Radar data on the second pass indicated that the vehicle was tumbling, which was expected. The tumbling rate was calculated to be approximately 2 rpm. During the second pass, two small objects were noted by a radar observation. The existence of these objects was confirmed by other radar stations. The objects, which were observed to be lower and faster than the main vehicle, are believed to have reentered between the fifth and seventh orbital passes. Objects with these orbital characteristics would have small weight-to-size ratios with ballistic numbers in the range of 2.0 to 2.5. The lifetime of the main vehicle experienced no measurable change and, hence, the loss of these objects did not appreciably affect the size or weight of the spacecraft and stage II GLV combination. The radar data from the first orbital pass indicated a lifetime of 60 orbital passes, and the last radar station which tracked the vehicle indicated that reentry occurred on the 64th orbital pass in the South Atlantic area.

4.2 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 and are within the expected tolerances. The largest difference, 3.73 seconds at the second stage shutdown signal, was within the expected deviation of ± 7.0 seconds.

The time of the first-stage engine shutdown signal (BECO) is defined as the time at which the staging control relays, which have been activated by the TCPS, actuate the motor-driven staging switches when the thrust in either stage I engine decays below 77 percent. The staging switches in turn shut down the engines. The time of the second-stage engine shutdown signal (SECO) is defined as the time at which the MOD III guidance system in the GLV causes an electrical current to activate the thrust chamber valve position solenoid valve override switch (TCVPSVORS). The events defined above, because of common usage, will be referred to in this report and throughout the Gemini Program as BECO and SECO, respectively.

4.3 FLIGHT TRAJECTORIES

The trajectories referred to as "planned" are preflight-calculated nominal trajectories supplied by Aerospace Corporation, and the trajectories referred to as "actual" are based on the Manned Space Flight Network tracking data. In both the planned and actual trajectories, the Patrick Air Force Base model atmosphere below 25 nautical miles, and the 1959 ARDC model atmosphere above 25 nautical miles were used. The earth model used was the Fischer Ellipsoid. A three orbital-pass ground track of the GT-1 mission is presented in figure 4-1. The altitude-longitude profile for the launch and three orbital passes is presented in figure 4-2. These two figures show that the actual profile was close to the nominal.

4.3.1 Launch

The launch trajectory data shown in figure 4-3 are based on the real-time output of the range safety impact predictor computer (IP-7094), which used the missile trajectory measurement (MISTRAM) system, FPS-16 and TPQ-18 radars, and the guided missile computer facility (GMCF) no. 1 which used the MOD III radar. The data from these tracking facilities were used during the time periods listed in the following table:

Facility	Time after lift-off, sec.
Cape Kennedy, IP-7094 and FPS-16	0 to 48
Patrick Air Force Base, TPQ-18	48 to 65
GMCF	65 to 399

The actual launch trajectory is compared with the planned launch trajectory in figure 4-3. It can be seen from the figure that the actual launch trajectory up to SECO was slightly slow in velocity and low in altitude and flight-path angle. At BECO, the actual velocity, altitude, and flight-path angle were lower than the planned by 83 ft/sec, 2,655 feet, and 0.05°, respectively. At SECO the actual velocity, altitude, and flight-path angle were again slightly lower than the planned by 15 ft/sec, 2,461 feet, and 0.04°, respectively. However, at SECO plus 20, the actual velocity was 20 ft/sec higher than the planned and the altitude and flight-path angle were low by 2,424 feet and 0.14°, respectively. Thus, the radar tracking data indicate that the velocity gained from SECO to insertion was apparently 35 ft/sec higher than expected. Since the radar data at SECO are of a preliminary nature, a more detailed study is necessary to resolve the velocity gained during GLV tail-off and to correlate it with the measured acceleration data.

4.3.2 Orbital

The orbital portion of the trajectory is shown in figure 4-4. The planned orbital trajectory was obtained by beginning with the nominal insertion condition supplied by Aerospace and integrating forward for three orbital passes. The actual orbital portion of the trajectory was derived by starting with the payload position and velocity vector obtained at the beginning of the second pass over Bermuda, as determined by the Goddard Space Flight Center (GSFC) computer by using the Manned Space Flight Network tracking data. The Bermuda vector was integrated backward along the flight trajectory to orbital insertion (defined as SECO plus 20 seconds) and forward to the end of the third orbital pass. These integrated values were in good agreement with the position and velocity vectors determined by the GSFC computer for passes near Woomera, Australia, during the first and second passes and Eglin Air Force Base at the end of the third pass; thus the validity of the integrated orbital portion of the flight trajectory was established.

A comparison of the actual and planned trajectory parameters is given in table 4-II. The flight parameters at SECO plus 20 seconds, as determined from the orbital data, show that the actual velocity was higher by 20 ft/sec than was planned. This higher velocity at orbital insertion resulted in a higher apogee altitude and a longer orbital period. From orbital radar position and velocity vectors obtained during the first three orbital passes, an estimated lifetime of the GT-1 payload was calculated to be 60 orbital passes.

Since, as shown in table 4-II, the actual velocity at SECO was less than planned by 15 ft/sec and the actual velocity at SECO plus 20 seconds was greater than planned by 20 ft/sec, a velocity of 35 ft/sec greater than expected was obtained during the second stage tail-off.

[redacted]

TABLE 4-I.- SEQUENCE OF POWERED-FLIGHT EVENTS

Event	Planned time, sec	Actual time, sec	Difference, sec
Stage I engine ignition signal	-3.20	-3.68	-0.48
Lift-off	0.00	0.00	0.00
Roll program start	9.92	10.00	+0.08
Roll program end	20.48	20.50	+0.02
Pitch rate 1	23.04	23.00	-0.04
Shutdown lockout timers expires	40.00	39.60	-0.40
Pitch rate 2	88.32	88.10	-0.22
Control gain change no. 1	104.96	104.80	-0.16
Pitch rate 3	119.04	118.80	-0.24
Arm stage I engine shutdown circuitry	144.64	144.00	-0.64
Staging arm timer operates	145.00	145.20	+0.20
Staging I engine shutdown signal and stage II engine ignition signal (BECO)	153.12	153.99	+0.87
Separation begins	153.81	154.70	+0.89
Stage II malfunction detection thrust chamber switch makes	154.01	154.80	+0.79
Pitch rate 3 ends	162.56	162.00	-0.56
Initiate radio guidance enable	162.56	162.00	-0.56
1st radio guidance command received	169.00	168.80	-0.20
Stage II engine shutdown circuitry enable	315.50	316.30	+0.80
Stage II engine shutdown signal (SECO)	335.50	339.23	+3.73
Stage II malfunction detection thrust chamber switch unmakes	336.06	339.60	+3.54
Stage II engine shutdown signal + 20 seconds	355.50	359.23	+3.73

[redacted]

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

Condition	Planned	Actual	Difference
SECO			
Time from lift-off, sec	335.50	339.23	3.73
Geodetic latitude, deg North . .	30.5328	30.5433	0.0105
Longitude, deg West	72.1632	72.0570	-0.1062
Altitude, feet	531,021	528,560	-2,461
Altitude, nautical miles	87.4	87.0	-.4
Range, nautical miles	455.4	476.8	21.4
Space-fixed velocity, ft/sec . .	25665.3	25650	-15.0
Space-fixed flight-path angle, deg	0.01915	-0.017	-0.036
Space-fixed heading angle, deg East of North	77.6860	77.585	-0.101
SECO plus 20 seconds			
Time from lift-off, sec	355.50	359.23	3.73
Geodetic latitude, deg North . .	30.8178	30.8352	0.0174
Longitude, deg West	70.6834	70.5555	-0.1279
Altitude, feet	531,433	529,009	-2,424 [a](4,800)
Altitude, nautical miles.	87.5	87.1	-0.4 [a](±0.8)
Range, nautical miles	533.8	541.5	7.7
Space-fixed velocity, ft/sec . .	25,765.3	25,785.4	20.1 [a](±32)
Space-fixed flight-path angle, deg	0.0177	-0.1251	-0.1428 [a](±0.20)
Space-fixed heading angle, deg East of North	78.4784	78.3759	-0.1025

[a]Tolerances at orbital insertion

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

Condition	Planned	Actual	Difference
Orbital parameters			
Perigee altitude, statute miles .	100.6	99.6	-1.0
Perigee altitude, nautical miles.	87.5	86.6	-0.9
Apogee altitude, statute miles. .	186.4	198.9	12.5
Apogee altitude, nautical miles .	162.0	173.0	11.0
Period, min:sec	89:05	89:16	00:11
Inclination angle, deg	32.54	32.59	0.05
Maximum conditions			
Altitude, statute miles	186.4	198.9	12.5
Altitude, nautical miles	162.0	173.0	11.0
Space-fixed velocity, ft/sec . .	25,765.3	25,789.3	24.0
Earth-fixed velocity, ft/sec . .	24,448.9	24,473.1	24.2
Exit acceleration, g	7.3	7.4	0.1
Exit dynamic pressure, lb/sq ft .	751	779	28

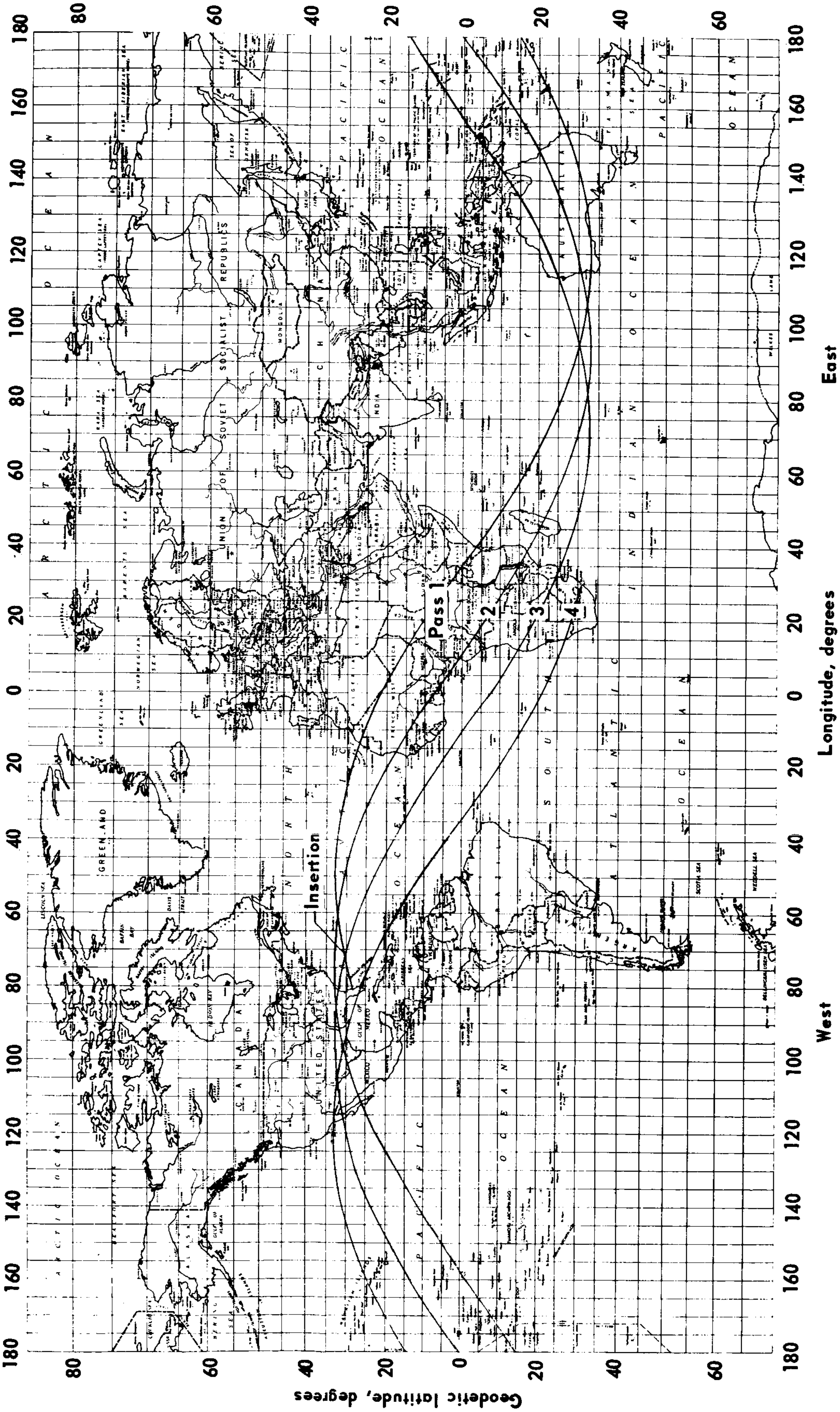

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FIGURE 4-1 GROUND TRACK FOR THE GT-1 ORBITAL MISSION

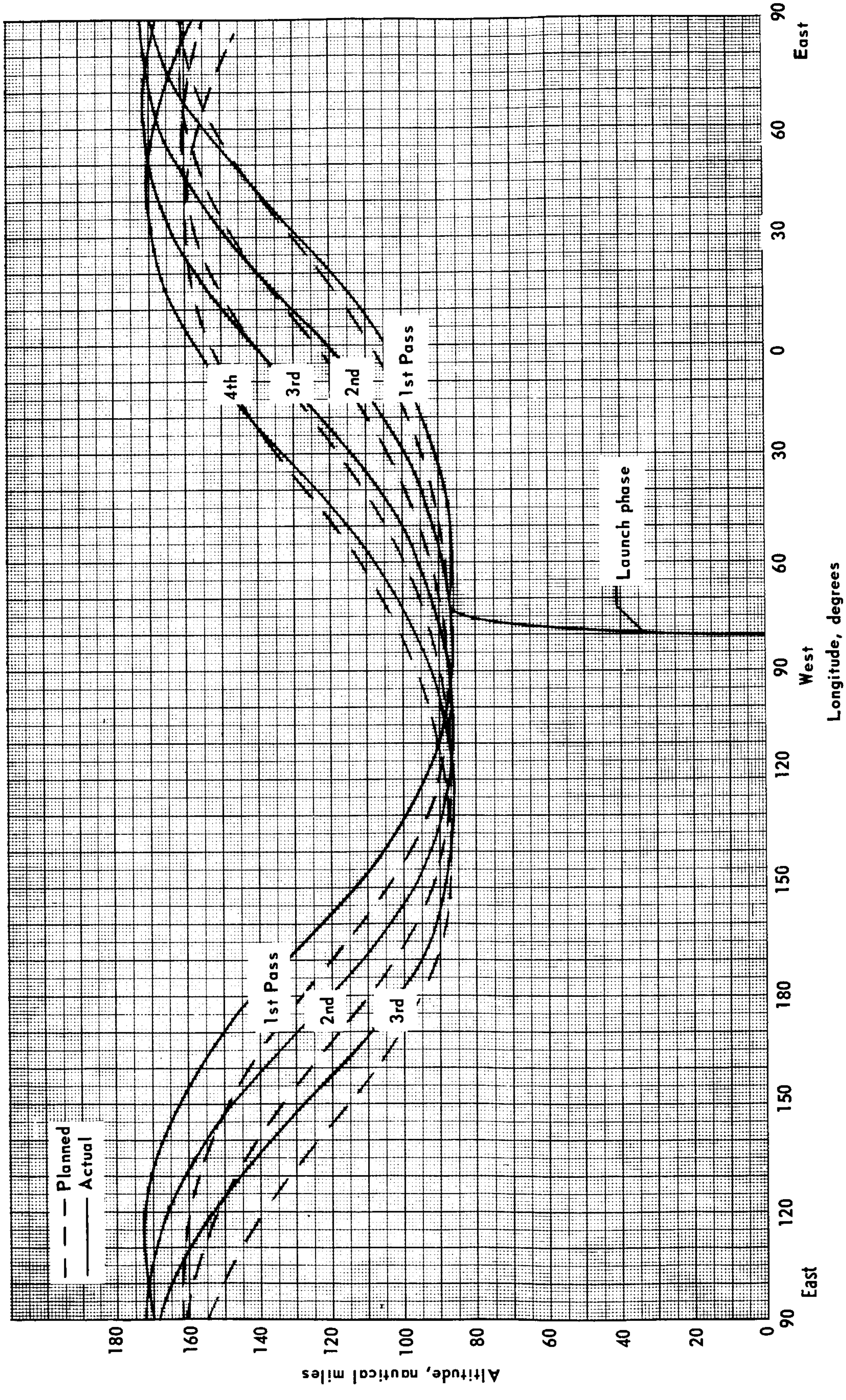

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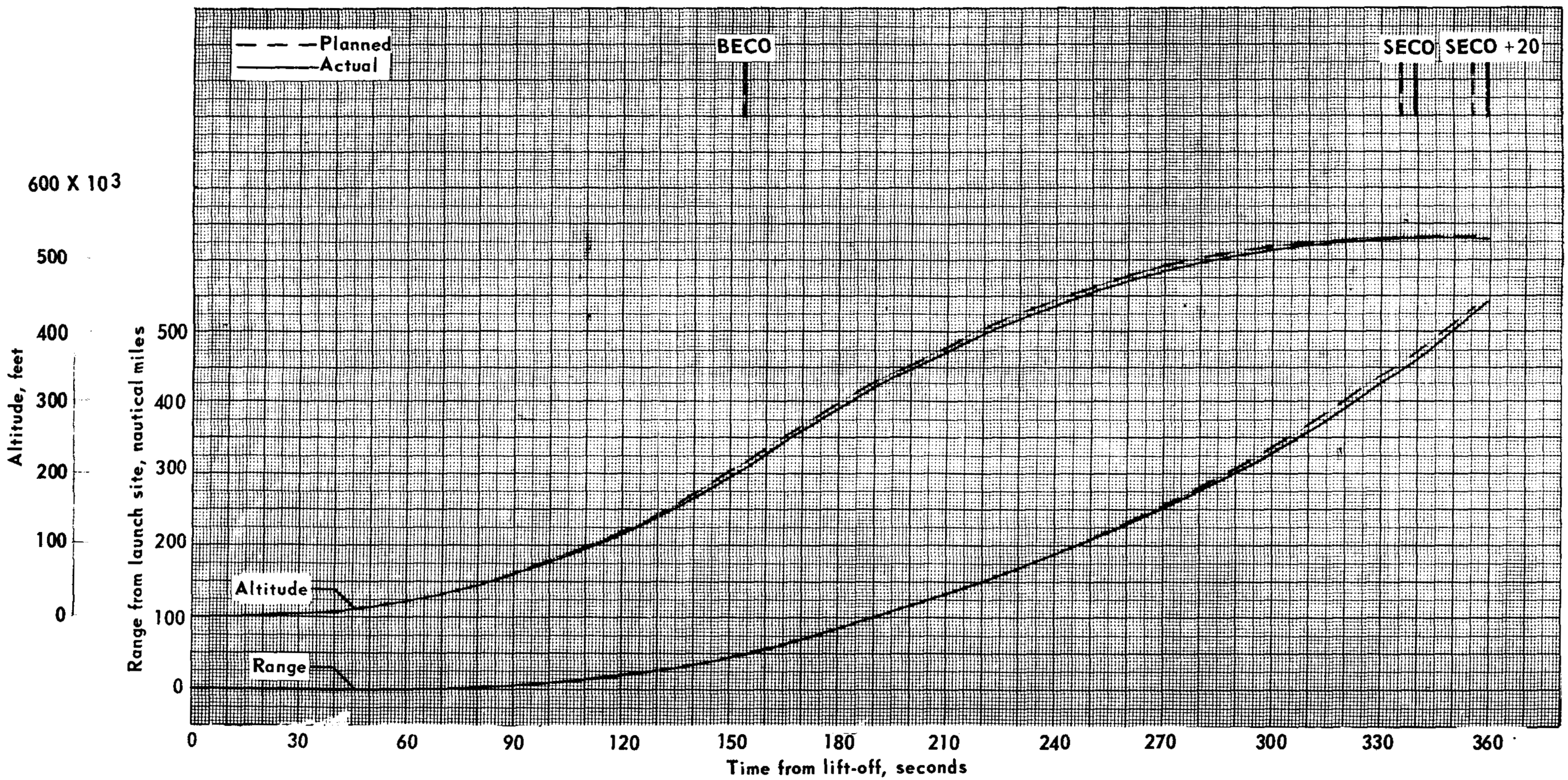
FIGURE 4-2 ALTITUDE - LONGITUDE PROFILE

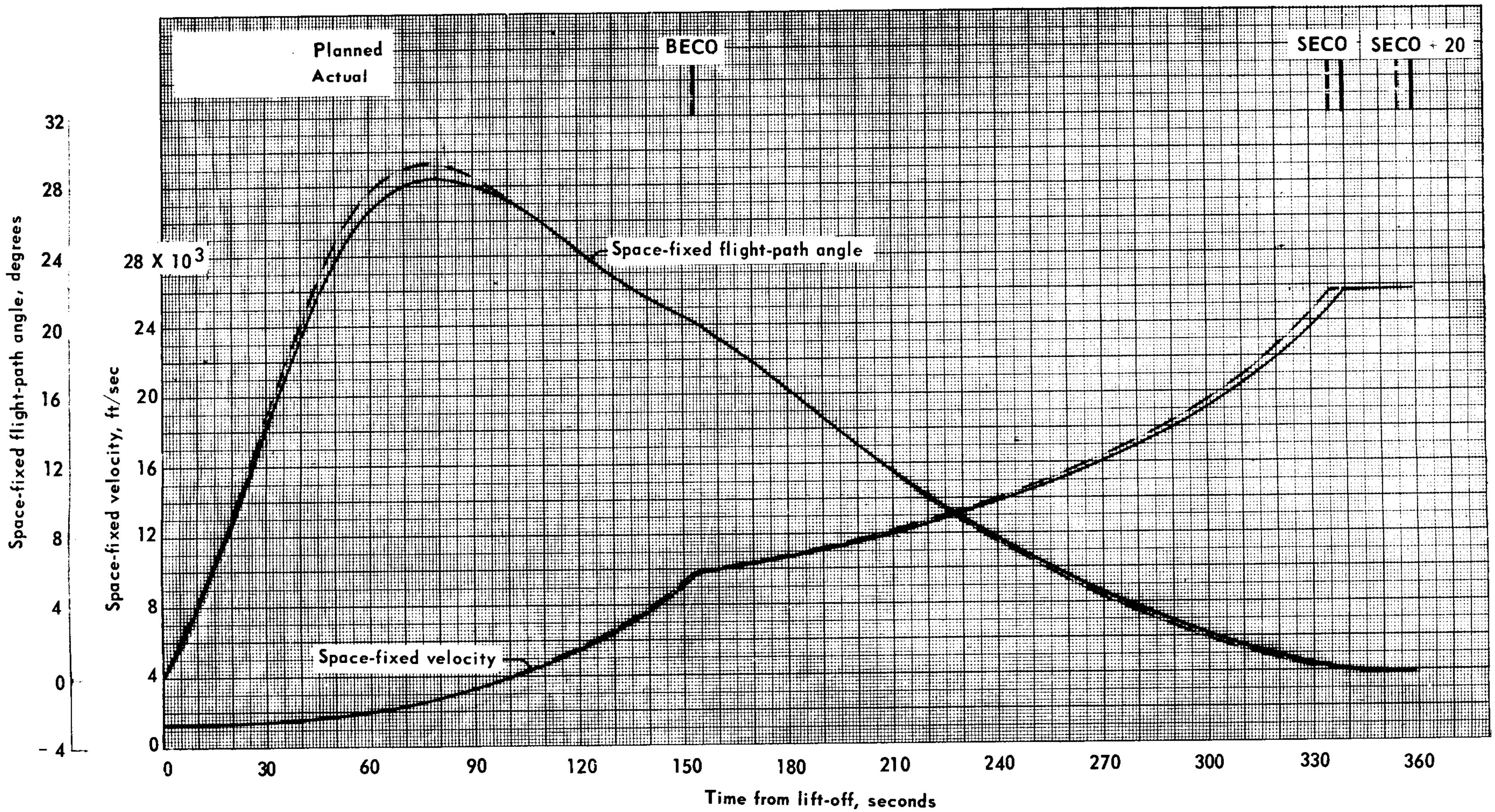
NASA-S-64-3074



(a) Altitude and range.

FIGURE 4-3 TIME HISTORIES OF TRAJECTORY PARAMETERS FOR GT-1 MISSION LAUNCH PHASE.

NASA-S-64-3075

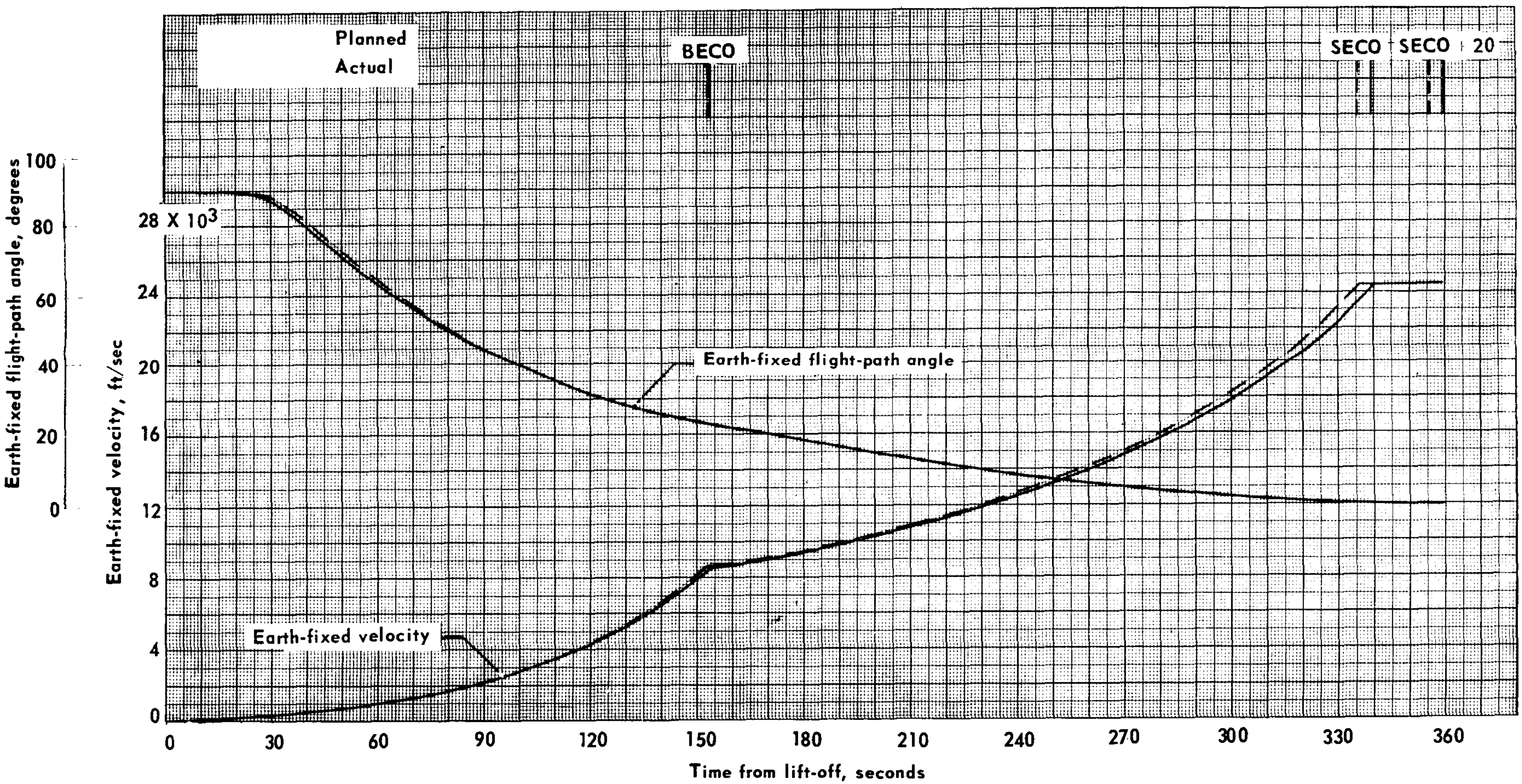


(b) Space-fixed velocity and flight-path angle.

FIGURE 4-3 CONTINUED

[redacted]

NASA-S-64-3076

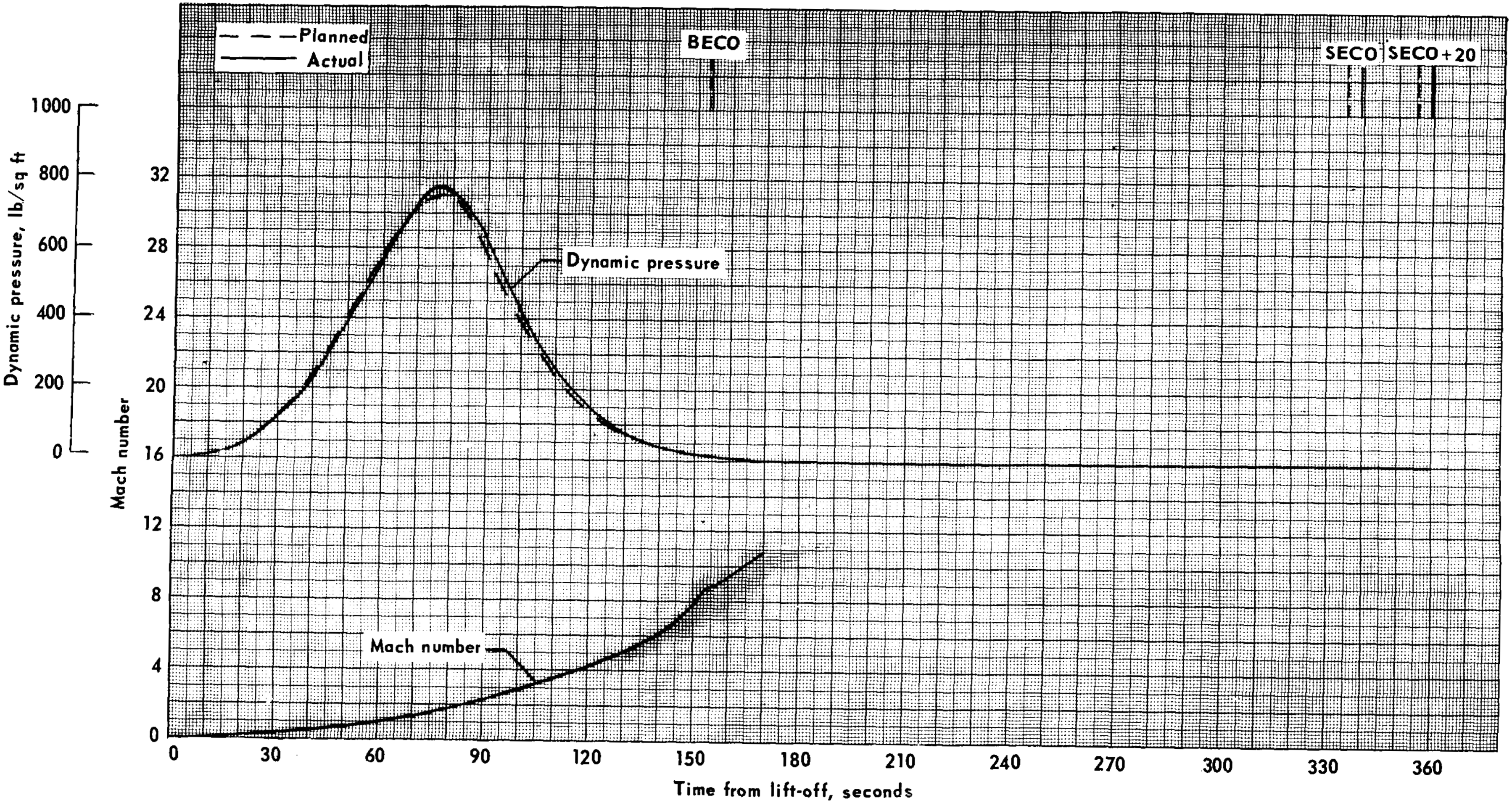


(c) Earth-fixed velocity and flight-path angle.

FIGURE 4-3 CONTINUED

[redacted]

NASA-S-64-3078

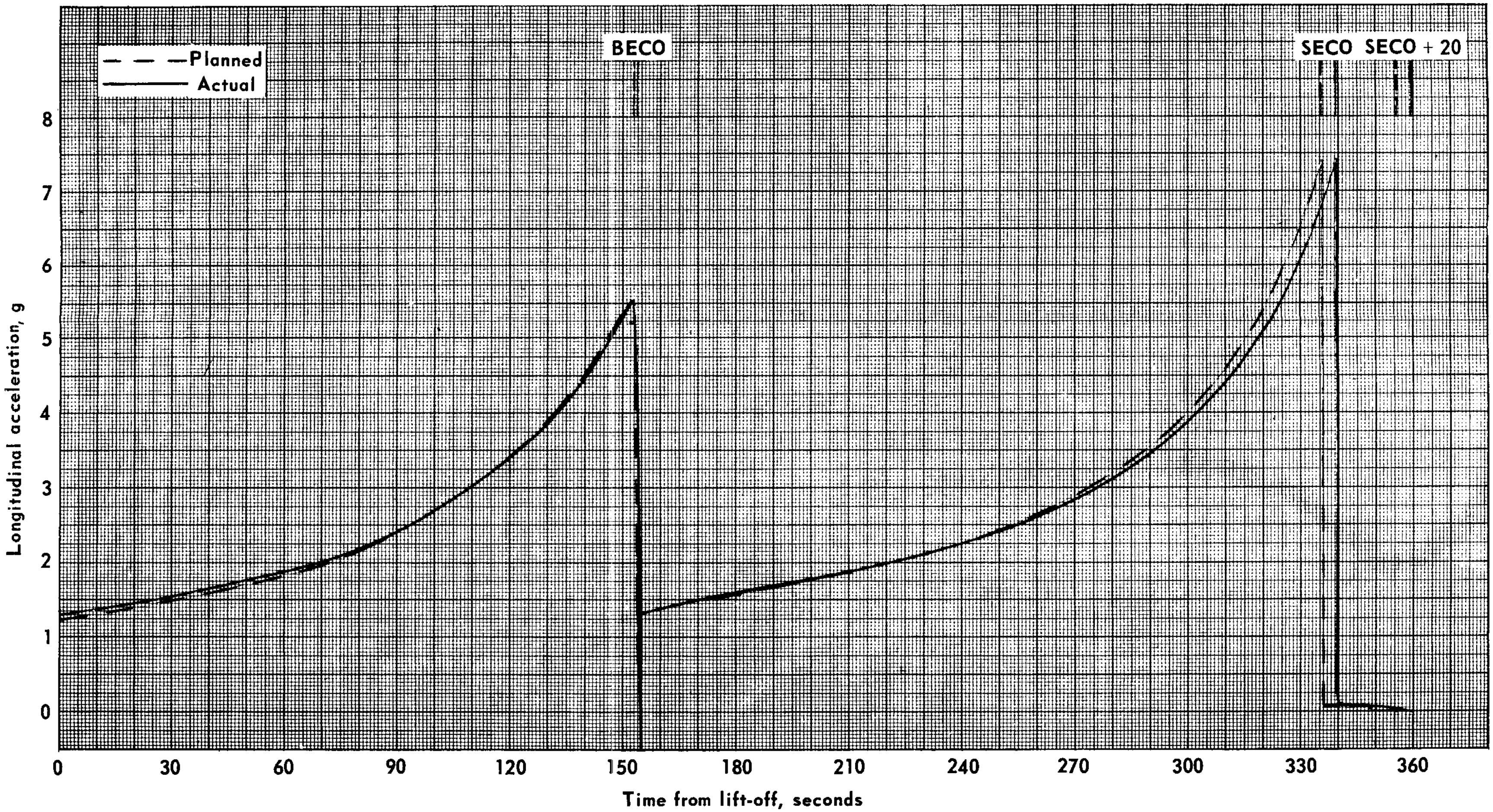


(d) Dynamic pressure and mach number

FIGURE 4-3 CONTINUED.

[redacted]

NASA-S-64-3077

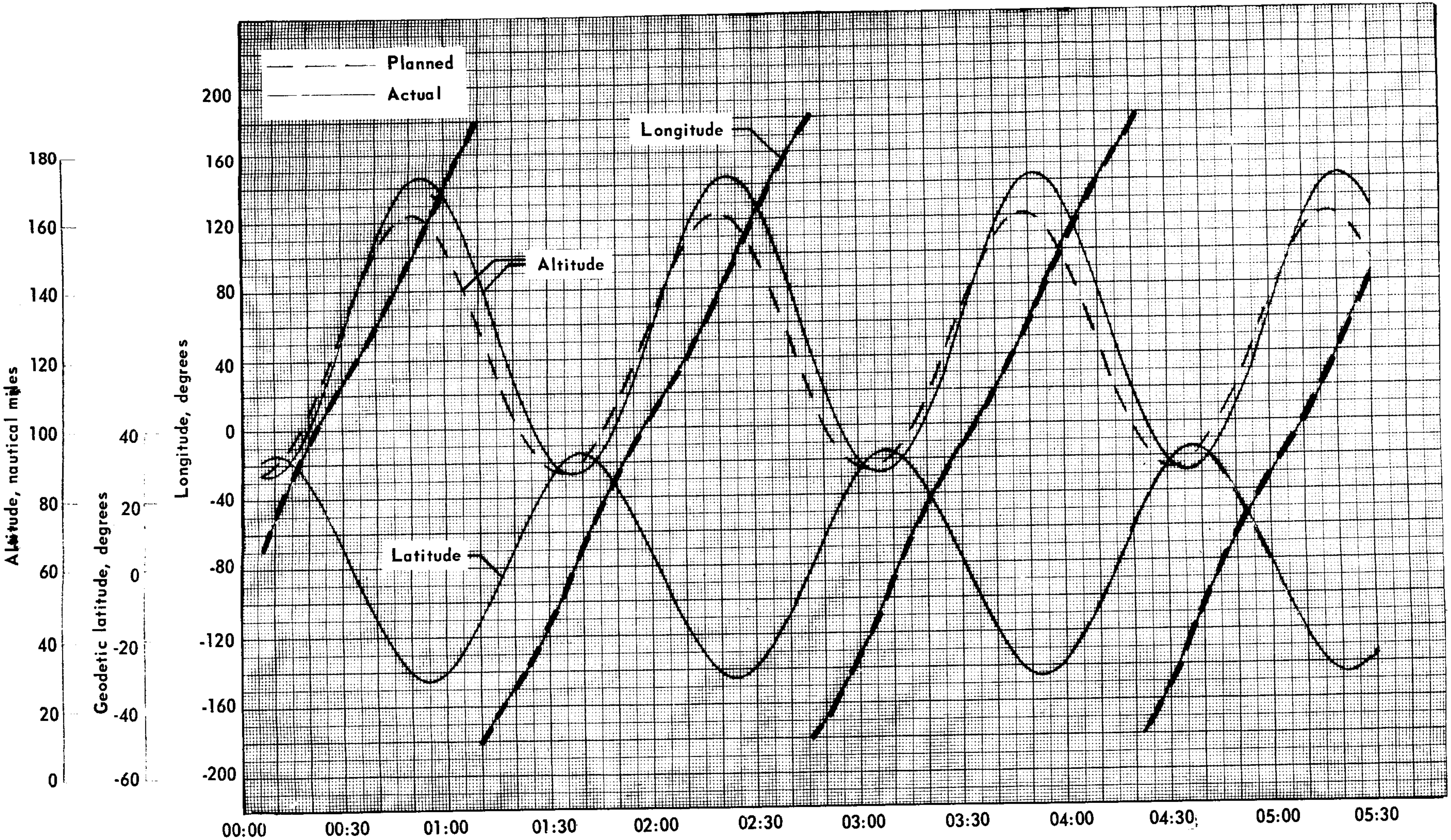


(e) Longitudinal acceleration

FIGURE 4-3 CONCLUDED.

[redacted]

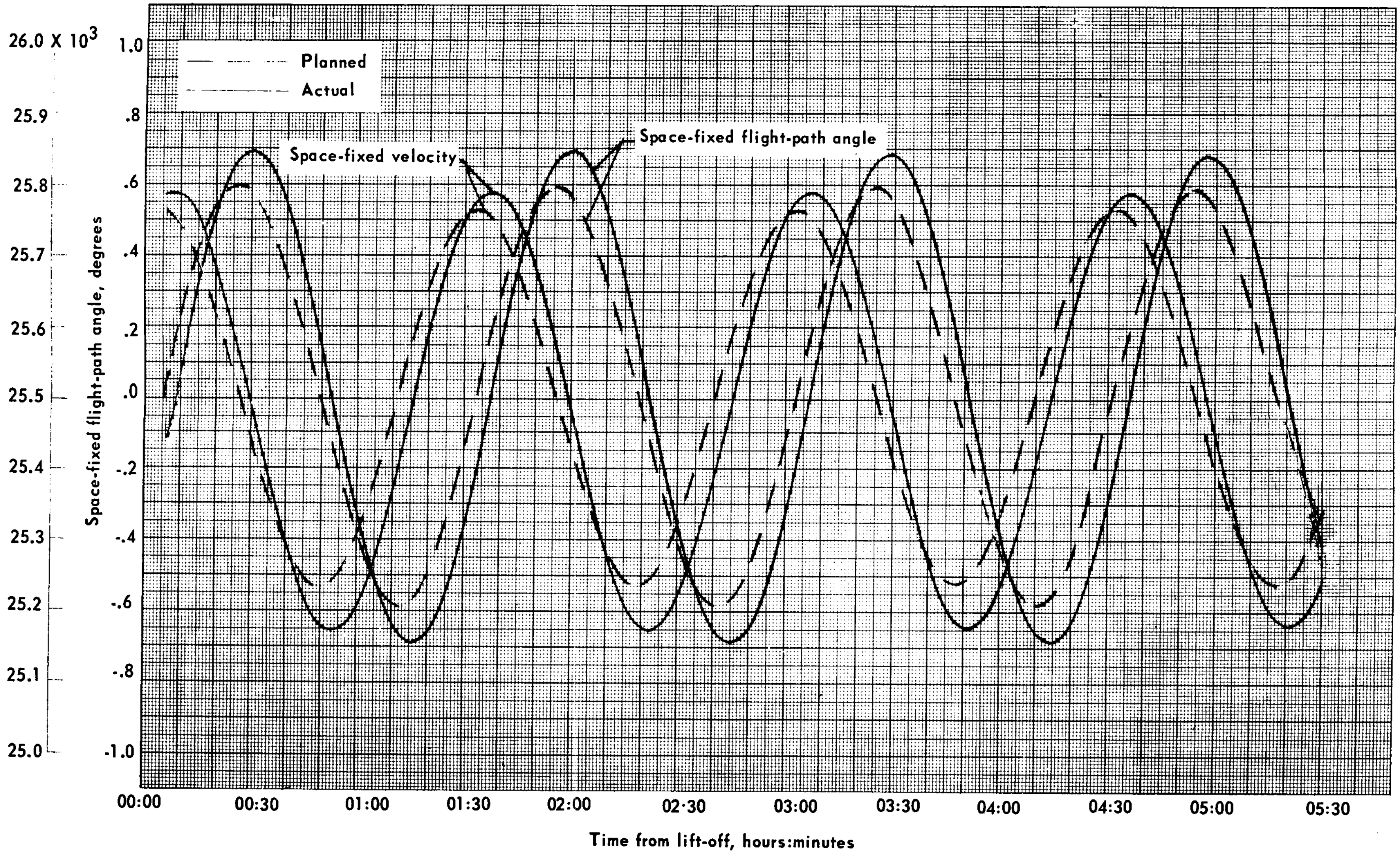
NASA-S-64-3081



(a) Latitude, longitude, and altitude.

FIGURE 4-4 TIME HISTORIES OF TRAJECTORY PARAMETERS FOR GT-1 MISSION, ORBITAL PHASE

NASA-S-64-3073



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

FIGURE 4-4 - CONCLUDED

5.0 VEHICLE PERFORMANCE

The performance of the Gemini space vehicle on the GT-1 mission was satisfactory. The launch vehicle performed within its specification limits in all respects and inserted the spacecraft into orbit within the prescribed limitations. A first-order mission objective was satisfied in that the structural integrity of the launch-vehicle and spacecraft combination was demonstrated.

While the loads environment encountered was not severe, the reaction of the airframe to the applied loads served to indicate the validity of the design. The performance of the engine subsystem and the related propellant, hydraulic, and pressurization systems was satisfactory with no serious anomalies.

Some misalinement of the stage II thrust vector with the vehicle center of gravity was indicated by the performance of the flight control and guidance system. An investigation is required.

On a number of telemetry traces, there was evidence that a transient had occurred approximately 11 seconds after stage II engine cut-off. It is thought to have originated in the engine system. While no appreciable significance is attached to the transient at this time, it must be investigated in an attempt to achieve an understanding of its cause and potential consequences.

The flight control and guidance systems satisfactorily accomplished their task, as evidenced by the orbit achieved. There was no switchover of the flight control system. The insertion conditions attained by the guidance system represented approximately a 2σ dispersed case, and some corrective action is required in this area, since the MOD III radar data available to the system were of excellent quality. The GT-1 mission demonstrated the first active guidance of a launch vehicle by the MOD III radio guidance system (RGS) at an elevation angle as low as 7.0°, and the performance of the radar and airborne antenna systems was very good.

The malfunction detection system (MDS) performed properly and would have provided correct information to the spacecraft MDS panel.

The longitudinal oscillation phenomenon, which is characteristic of the Titan II missile, was effectively suppressed on this flight by the oxidizer standpipe and the mechanical accumulator in the fuel supply lines of the first stage. The levels achieved at the critical frequency of 11 cps were well within the limits previously established. Some evidence of buildup of g-forces in the spacecraft late in the first-stage flight was noticed, but the condition was of very short duration and at a frequency of 17 cps. This condition is not considered to be significant to the manned environment.

The production spacecraft structure which was launched by this flight served to demonstrate the adequacy of the structural design and the launch-phase thermal analysis. While the flight environment was not severe, the data obtained from the structural instrumentation agreed well with the design parameters. The structural temperatures achieved verified the preflight predictions.

The cabin-pressure relief valve operated properly, and cabin pressure decayed at a very slow rate throughout the monitored three orbital passes, indicating a cabin seal well within the specification.

The reports of radar tracking of multiple objects in orbit cannot be confirmed by any of the telemetered data obtained from this flight. To the contrary GLV instrumentation indicated normal performance until loss of signal at insertion plus 80 seconds, and all of the spacecraft instrumentation gave indications of being in operating condition through the end of the third orbital pass. Any serious structural failure would have been reflected in some instrumentation loss. A further study of this situation is required in an attempt to determine the source of the objects detected.

5.1 SPACECRAFT

5.1.1 Structure

5.1.1.1 General.- Among the first-order test objectives of the GT-1 mission are the following which are related to the spacecraft structure:

(a) To determine exit heating conditions on the spacecraft and launch vehicle

(b) To demonstrate the structural integrity and compatibility of the spacecraft and Gemini launch vehicle combination from launch through orbital insertion

(c) To demonstrate the structural integrity of the Gemini spacecraft from launch through orbital insertion.

In the following paragraphs, results are presented which show that these objectives have been satisfactorily met, and their significance as related to the structural design criteria for the Gemini spacecraft is evaluated. It should be pointed out that this mission created only a single set of conditions which had a small statistical chance of producing the design limit loading. Actually, the mission was near nominal in all respects; thus average, rather than extreme, loading was imposed on the structure.

A few of the more interesting and salient results which are discussed in this section are presented briefly here. Low-frequency vibration of the environmental control system (ECS) pump package showed a response which was an order of magnitude lower than the worst case predictions. The adapter shell modes were also lower than the worst case predictions by a factor of 10. With respect to low-frequency longitudinal accelerations, there was a period in which the amplitude in the spacecraft exceeded the acceptable level of ± 0.25g. It consisted of two waves, one having a duration of 2 seconds and one of 1 second just before BECO. The average level during the 2-second period was

0.3g, with two single cycles reaching ±0.4g. Because of its short duration and higher frequency (17 cps) as compared with peak levels of 11 cps previously experienced on Titan II vehicles, the 2-second period does not appear serious, but it will be evaluated by the appropriate Manned Spacecraft Center personnel. No supplemental report is planned. Launch temperatures over the entire spacecraft averaged somewhat lower than the predicted nominal case. The cabin-pressure regulator operated properly, and cabin pressures were maintained at an acceptable level through three orbital passes. The wealth of telemetry data recorded during the orbital passes provides adequate proof of the structural integrity of the spacecraft throughout the mission.

5.1.1.2 Quasi-steady state loading.- The so-called quasi-steady state loading is that due to drag, inertia, aerodynamic lift, and low-frequency transonic buffet bending. The total loading is that due to the above quasi-steady state conditions and that due to high-frequency dynamic loading, that is, loading due to structural vibratory response to fluctuating pressures, which is discussed in subsection 5.1.1.3. The dynamic loading is a maximum near a Mach number of 1.0, whereas the quasi-steady state loading is a maximum near maximum dynamic pressure q_{max} which occurred on this flight at 17 seconds after the spacecraft reached a Mach number of 1.0.

Having stated previously that the quasi-steady state loads reach a maximum near maximum dynamic pressure, it must be pointed out that the various elements making up the total quasi-steady load vary in a different manner. Axial loads due to drag and aerodynamic lift are a strong function of dynamic pressure and occur close to q_{max}, but buffet bending is more predominant near a Mach number of 1.0 and inertia loading steadily increases through the high load region as a result of propellant depletion and increasing thrust due to the increasing altitude. Still another variable is the radial differential pressure loading which is not only a function of α and q, but also a strong function of compartment "internal-to-ambient" pressure differential due to venting lag. This radial load is the most critical adapter-ring quasi-steady state load and occurs in the structural design trajectory about 10 seconds after q_{max}.

5.1.1.2.1 Trajectory data for maximum loading region: Dynamic pressures between 400 lb/sq ft and the maximum value of 797 lb/sq ft were encountered between 50.5 seconds and 100.8 seconds after lift-off at Mach numbers of between 0.67 and 2.96. Figure 5-1 shows the actual and predicted time histories of these parameters. The flight dynamic pressure was calculated by using radar velocity data and atmospheric density as measured on the day of the flight, but it was uncorrected for winds. Wind magnitude and direction at the time of maximum dynamic pressure was 98 ft/sec and 302° from North. Mach number was calculated from the calculated dynamic pressure and ambient pressure as measured with a sounding rocket prior to flight. Definitive angle-of-attack data for the flight were not available at the time of writing of this report. For purposes of structural evaluation, an estimate of angle-of-attack was obtained. For the Mach number 1 period at LO + 65 seconds, the maximum $q\alpha$ period at LO + 69 seconds, and the maximum q period at LO + 78 seconds, the angles of attack in pitch were +0.15°, +0.42°, and +0.37°, respectively. The corresponding angles of attack in yaw were +1.96°, +2.86°, and +1.67°. The resultant total angles of attack were +1.97° at LO + 65 seconds, +2.89° at LO + 69 seconds, and +1.72° at LO + 78 seconds.

5.1.1.2.2 Longitudinal loading for actual trajectory conditions: The quasi-steady state loads can be summed as an axial load P and a moment M. They can further be combined by the relationship, $P_E = -P \pm \frac{2M}{R}$, where P_E is the equivalent axial load on one side of the vehicle, P is the axial load due to drag and inertia, M is the moment due to angle of attack, and R is the radius of the section. The negative sign is used for the compression side and the positive sign for tension.

In January of 1964 a production adapter and launch vehicle upper skirt and oxidizer dome assembly was subjected to 125 percent of design limit load and moment. These conditions resulted in an axial load of 26,000 pounds and a moment of 2.7 million in-lb. Additional moment was then added until failure was produced. The specimen failed in tension at the interface connection at an equivalent load of 136,000 pounds. The compression load was 184,000 pounds at the time of failure. During the GT-1 flight at 65 seconds after lift-off with an estimated angle of attack of +1.97°, the equivalent compression load was 35,200 pounds. At 78 seconds after lift-off and with an angle of attack of 1.72°, the compression load was 43,540 pounds. At 69 seconds during the time of maximum $q\alpha$, the angle of attack was 2.89° and the equivalent compression load was 44,470 pounds. No tension was present for these conditions. Hence, it is seen that the longitudinal loading for the GT-1 flight was less than 25 percent of the structural capability.

5.1.1.2.3 Radial loading for actual trajectory conditions: For low angles of attack, the critical stress occurs as compression in the inboard flanges of the rings, resulting from a combined loading of differential pressure, primary bending and axial load, and equipment inertial loads. The critical stress was lower than nominal on the GT-1 mission because it was relieved by the higher than predicted internal pressure in the adapter.

By using the angle-of-attack information discussed in subsection 5.1.1.2.1, the stresses may be estimated as follows: For an angle of attack of 1.97°, an adapter "internal-to-ambient" pressure of 1.4 psi, and a dynamic pressure of 660 psi, the critical stress in the Z28 ring was 1,580 psi; and for an angle of attack of 1.72°, an adapter differential pressure of 2.1 psi, and a dynamic pressure of 7.95 psi, the critical stress in the Z28 ring was 1,320 psi. For the maximum αq condition the critical stress was 1,440 psi. Allowable compressive stress in the inboard flange of the Z28 ring, as demonstrated by the test, was 38,300 psi.

5.1.1.3 Dynamic Loading.- The vibrations of the spacecraft 1 structure were measured by 13 accelerometers arranged to provide information regarding adapter shell modes, pump package local modes, longitudinal vibration effects from the launch vehicle, body bending modes, and spacecraft equipment dynamic environment (table 5-I). In general, oscillograph records of each accelerometer time history indicated an appreciable response from ignition to about

5 seconds after lift-off, when ground reflection effects predominate, then a quite period up to about LO + 35 seconds, a buildup to peak activity at about LO + 65 seconds, another quiet period after LO + 100 seconds, and finally, some low-level, low-frequency responses just prior to BECO and through SECO. These data were reduced with narrow-band analyzers at both Manned Spacecraft Center in Houston, Texas, and the contractor's facility in St. Louis, Missouri, to provide the g rms, power spectral density, and probability information necessary to evaluate the structure's dynamic performance.

The following subsections present an evaluation of the measured response for the actual flight conditions as compared with the predicted response.

5.1.1.3.1 Adapter shell modes: A ground-vibration survey of the spacecraft was conducted at the contractor's facility in July of 1963. It was found that the equipment section of the adapter had 12 significant resonant ring modes within the frequency band of 126 cps to 213 cps. The retrograde section had 11 modes ranging between 184 cps and 550 cps. An analysis was made of the ring mode response by using these measured mode shapes, measured damping, and fluctuating pressures and co-power and quad-power cross correlations between various areas of the adapter as measured in the transonic wind tunnel of Ames Research Center. The ring subjected to the most critical stress was determined to be the Z28 ring, with an estimated response of 22.8g rms. Accelerometer QC22 was installed on this ring at stringer 41 to check this radial response, and accelerometer QC23 was installed at ring Z82 and stringer 31 to verify the radial response of the adapter retrograde section. The maximum response of ring Z28 in flight, measured by accelerometer QC22 in the 100 to 225 cps band, was 2.5g rms measured at 66 seconds after lift-off. Accelerometer QC23 showed that ring Z82 had a maximum response of 14.5g rms at lift-off and 16.7g rms at 65 seconds after lift-off.

Figure 5-2 and figure 5-3 are power spectral density plots in (g rms^2/cps) for accelerometers QC22 and QC23, respectively, during the peak time period of 63.50 to 67.45 seconds after lift-off. Examination of these figures explains the low responses measured. For the accelerometer QC22, it can be seen that very little power exists in the 100 to 225 cps band which contains all the significant resonant modes from ring Z28. Ring Z82, with resonant modes for 184 to 550 cps, shows the higher response which can be expected from this stiffer structure since it can react to the high-frequency energy in this band. Because of its greater strength, however, stresses in ring Z82 are reasonably low.

5.1.1.3.2 Pump package local modes: The GT-1 spacecraft adapter had a 122-pound module attached directly to the shell structure of the equipment section of the adapter. Although the weight was lead ballast in GT-1, it represented the weight and center of gravity of the radiator pump installation for later spacecraft. Ground tests prior to the flight had shown that four low-frequency local modes, 44.6 cps, 50.8 cps, 58.5 cps, and 66.2 cps, were associated with this installation. By using the measured mode shapes and the fluctuating pressure and cross correlation data from the wind-tunnel test at Ames Research Center, the maximum radial response at the intersection of Z41 ring and stringer 68 was estimated by MSC to be 7.1g rms. Accelerometer QC26 was installed at this structural intersection, which is one of six mounting points of the pump package, to measure the radial response of the mass.

The QC26 accelerometer had a range of ±32g. High-frequency components in the order of 500 cps drove the accelerometer beyond these limits to about ±37g. Since not all peaks of the 500-cps random response reached these limits, however, the low-frequency response was still identifiable (see subsection 5.1.2.2.2 for details). The maximum responses in the 48 to 60 cps band measured during the GT-1 flight were 0.5g rms at lift-off and 0.7g rms at 67 seconds from lift-off. This low-frequency response was obtained by power spectral densities from 48 to 60 cps. No energy was discernable at the 44.6 cps and 66.2 cps resonances.

It is evident that the response of the pump package was much less than the predicted 7.1g rms. This low response, as in the case of ring modes previously discussed, was primarily the result of an extremely conservative analysis. However, with anticipated scatter from flight to flight, it is likely that the measured value could be higher by a factor of 2 or 3 on later flights.

5.1.1.3.3 Longitudinal vibration from the launch vehicle: Early Titan II flights indicated a strong, low-frequency longitudinal vibration that would be detrimental to manned flight. However, through much effort and the modifications incorporated into the launch vehicle, this effect has been substantially reduced in the more recent flights. The QA11 low-frequency accelerometer was installed in the GT-1 reentry assembly to measure this effect, and the QC17 accelerometer was installed near the interface of the spacecraft and launch vehicle to measure high-frequency launch-vehicle inputs. The QC17 accelerometer measured a peak at lift-off with an overall value of 2.9g rms, and again at 66 seconds after lift-off with an overall value of 2.6g rms. The frequencies were in excess of 100 cps and were not significant to the structure.

Discrete longitudinal accelerations of small amplitude occurred between 75 seconds after lift-off and BECO on data channel QA11. The pulsations occurred in waves with increasing amplitude and frequency from 0.08g peak half amplitude and 9.0 cps at 78 seconds after lift-off to 0.4g peak half amplitude and 17 cps just before BECO (at 151 seconds).

A further evaluation of the measured low-frequency longitudinal accelerations and a discussion of their significance is presented in subsection 5.2.1.1.

5.1.1.3.4 Body bending modes: The fluctuating aerodynamic pressure encountered during launch excites the low-frequency bending modes of a missile. This phenomenon may be defined as the "buffet bending" effect. The effect is manifested as a buffet bending moment which is additive to the quasi-steady state bending moment resulting from normal angle-of-attack excursions during launch. Accelerometers QD10 and QD11 were installed in the rendezvous and recovery section of the GT-1 spacecraft to measure the lateral and pitching response, respectively, of these modes. Another set of accelerometers, QA09 and QA10, were installed to measure lateral and pitching response, respectively, in the conical section of the spacecraft. During the flight, the maximum recorded values from these 1 to 30 cps accelerometers occurred between 60 and

70 seconds after lift-off. The values were 0.10g rms for QA09, 0.18 g rms for QA10, 0.20g rms for QD10, and 0.37g rms for QD11.

Between BECO and SECO, lateral and axial accelerations occurred in the reentry assembly in short waves which increased in amplitude and frequency as the flight progressed. The waves first occurred in the lateral directions at 1.6g peak amplitude and 23 cps. The waves gradually shifted to the axial direction and increased to about 0.4g peak amplitude and 28.5 cps at 250 seconds, and then decreased to about 0.3g peak amplitude and 34 cps at 281 seconds after lift-off.

An evaluation of these body-bending modes for the complete vehicle (launch vehicle and spacecraft) and the structural interface moments and loads is presented in subsection 5.3 of this report.

5.1.1.3.5 Spacecraft equipment dynamic environment: The spacecraft equipment is being qualified to a random vibration spectrum which was derived from an envelope of several measurements taken on the Mercury Program flights. The spectrum has an overall 20 to 2,000 cps acceleration level of 12.6g rms. Accelerometers QB12, QB13, and QB14 were mounted on hard structure in the left-hand equipment bay of the spacecraft to measure the tangential, radial, and longitudinal vibration, respectively. On the GT-1 flight, the maximum overall responses of these accelerometers occurred at 67 seconds after lift-off and were 2.2g rms for QB12, 3.8g rms for QB13, 2.2g rms for QB14, and 2.4g rms for QC24.

Figure 5-4 presents a plot of the spectral density analysis of the radial vibration on the conical section and of the longitudinal vibrations on the blast shield at 66 seconds after lift-off with the equipment-qualification launch spectrum superimposed. The qualification spectrum has peak densities at frequencies somewhat lower than those which are indicated to predominate by the GT-1 data. In addition, the spectral density analysis indicates that accelerometer QB13 experienced a spectral density above the qualification requirement in a narrow band centered at 565 cps. It is not a region of concern since the narrow band does not add a significant amount of energy in this region when compared with the qualification spectrum. Longitudinal vibrations above 1.2 kc as measured by accelerometer QB14, although lower than those measured by accelerometer QB13 in the low-frequency regions, are presented in figure 5-4.

Preflight predictions of the Gemini vibrations were based on data measured during the flights of the Mercury Program. The Mercury data indicated that a statistical scatter could be expected in the overall levels from flight to flight, as reported in reference 4. The data measured on spacecraft 1 are below the mean value of 6.4g rms presented in reference 4 for the flight dynamic pressure of 797 lb/sq ft.

5.1.1.4 <u>Thermal environment</u>.- The thermal environment of the GT-1 launch trajectory has been evaluated, and the present indication is that the heating parameters were close to the nominal mission trajectory, supplied by Aerospace (case 31). It should be pointed out that both the nominal mission trajectory and the actual launch trajectory were considerably less severe from the heating standpoint than the design heating trajectory (ref. 5).

The Patrick Air Force Base atmosphere was used for design temperature calculations. For a discussion of the atmospheric properties and winds measured on the flight day, see subsection 12.2.

The thermal environment was measured by 81 temperature sensors located on the GT-1 reentry assembly and adapter assembly in areas which would best show longitudinal and circumferential variations as well as the effect of outer skin protuberances. This instrumentation measured the outer skin and internal temperatures at designated locations from lift-off through the third orbital pass. In the evaluation of temperatures in the following discussion, only a representative number of temperature locations are discussed. The only temperature sensor which failed to function was on the adapter, and it was not essential to the thermal analysis. No temperature sensors were located on the spacecraft heat shield, which is used for reentry heat protection, since the GT-1 mission was primarily an exit structural test and the spacecraft was not separated from the second stage of the Gemini launch vehicle (GLV).

5.1.1.4.1 Distribution of peak measured temperatures: Figure 5-5 shows the distribution of measured peak temperatures on the upper half of the GT-1 spacecraft, and figure 5-6 shows the distribution of measured peak temperatures on the lower half of the spacecraft. Temperature trends discussed in this subsection are those for areas of the spacecraft which are least affected by outer skin protuberances. The effects of outer skin protuberances on temperatures for each reentry assembly section are discussed in subsections 5.1.1.4.2 to 5.1.1.4.5.

Definite temperature variations in the longitudinal direction were observed. In figure 5-5, outer skin temperatures on the cabin section are shown to increase longitudinally toward the adapter sections on the upper half of the spacecraft. In figure 5-6, outer skin temperatures on the cabin section on the lower half of the spacecraft are also shown to increase longitudinally in the direction of the adapter sections. On the adapter retrograde section, temperatures on the upper half of the spacecraft (fig. 5-5) increased in the longitudinal direction as the retrograde separation fairing was approached. On the adapter equipment section, temperatures decreased as the adapter-GLV separation ring was approached. On the lower half of the spacecraft adapter, the same temperature variations also occurred, as shown in figure 5-6.

In areas of uniform outer skin thickness, very little circumferential temperature variation was noted. On the conical section, the small variation can be seen in figure 5-5 as measured by PD47 and PD52 on the upper half of the spacecraft, and in figure 5-6 as measured by PD09 and PD06 on the lower half of the spacecraft. Temperatures on the adapter sections indicate little circumferential variation. On the reentry control system (RCS) and the rendezvous and recovery (R and R) sections, any circumferential temperature variation is directly related to the varying thickness of the circumferential beryllium shingles.

Temperature measurements were recorded by the Manned Space Flight Network and the Atlantic Missile Range tracking stations during the first, second, and third orbital passes; however, only data recorded at Cape Kennedy Telemetry building III were available for evaluation during the preparation period of this report. Three intervals of approximately 6 minutes each, as recorded at Cape Kennedy, were available. A supplemental report will be issued at a later date and will present an evaluation of orbital temperature measurements.

Temperature sensors on the reentry assembly indicated a slight temperature rise during the three successive orbital passes. The vehicle was exposed to sunlight for a greater period of time on each successive orbital pass prior to passing over Cape Kennedy. Also, temperatures recorded from the left side of the spacecraft were generally higher. The highest temperatures measured on the cabin, RCS, and R and R sections were 235° F, 128° F, and 220° F, respectively; these temperatures were recorded during the third orbital pass.

Minimum orbital temperatures for the spacecraft are not available since the lower limit of the instrumentation range for the thermocouples was the reference junction temperature and the lower limit for the resistive element was 0° F, and these limits were reached. Each of the measurements from outer skin of the adapter indicated that minimum measurable temperatures had been reached.

Heating rates have been calculated from the measured temperature-time histories by a computer using a one-dimensional thermal model. The accuracy of these deduced heating rates depends primarily on the complexity of the structure because of the one-dimensional nature of the computer program. For this reason, the adapter sections of the spacecraft do not lend themselves well to analysis because of the complex nature of the structure. Heating rates calculated from the measured temperature time histories on the RCS section were low, being in the range of 0.55 to 0.80 Btu/ft^2-sec, while heating rates on the R and R section varied from 0.50 to 1.25 Btu/ft^2-sec. The maximum heating rates experienced by the spacecraft occurred on the cabin section and were in the range of 0.40 to 1.50 Btu/ft^2-sec with a maximum heating rate of 1.88 Btu/ft^2-sec for temperature sensor PD46, located aft of the window depression.

5.1.1.4.2 Cabin section of reentry assembly: Figure 5-7 shows measured temperature time histories at three locations on the cabin section. These three temperature time histories illustrate the highest, most typical, and lowest temperature areas of the cabin section. Also shown is a design temperature prediction corresponding to PD06 at a location in the area of Z = 131.0. This prediction is discussed in subsection 5.1.1.4.6. The maximum temperature of 530° F, shown for PD06, is indicative of temperatures over much of the cabin section. A maximum temperature of [illegible] F is shown at PD46, which was located just aft of the window depression. This maximum temperature showed the effects of the flow-field change caused by the window depression. High heating in this area was also indicated by wind-tunnel data. Sensor PD50, which measured the lowest cabin-section temperature, 205° F, was located near the junction of the cabin and RCS sections. Low heating in this area was also indicated by wind-tunnel data. Temperature sensor PD48 was located in a high-temperature region, just aft of the horizon scanner fairing. Window temperatures, as represented by a maximum of 230° F at PD30, were lower than expected.

Since the cabin-section structure and insulation are designed for the more severe reentry heating condition, it was not unexpected that all measured internal temperatures, including inner skin temperatures and inner window-pane temperatures, indicated no temperature rise during the launch period.

5.1.1.4.3 RCS section of the reentry assembly: Figure 5-8 shows RCS section measured temperature time histories at two locations, PC07 and PC15. These two temperature time histories represent the most typical temperature (PC07) on the centerline on the lower half of the spacecraft (BY) and the temperature (PC15) in an area which shows the effect of the horizon-scanner fairing. Also shown is a design temperature prediction corresponding to PC07 at a location in the area of Z = 189. The maximum temperature of 110° F, shown for PC07, was indicative of temperatures over much of the bottom half of the RCS section. The maximum temperature of 135° F, shown for PC15, was measured in the vicinity of the horizon-scanner fairing, and this maximum temperature shows the effects of the flow-field change caused by the horizon-scanner fairing. The horizon-scanner fairing temperature, measured by PD19, reached 250° F, which was much lower than the design temperature.

The critical internal temperature area of the separation pyrotechnics at Z = 191.97 indicated no temperature rise during the launch period.

5.1.1.4.4 R and R section of reentry assembly: Figure 5-9 shows the measured temperature time histories at two locations on the R and R section. These two temperature time histories represent the most typical BY temperature (PB05) and the lowest measured temperature (PB14) on the R and R section. Also shown is a design temperature prediction corresponding to PB05 at a location in the area of Z = 217. The maximum temperature of 155° F, shown for PB05, is indicative of temperatures over much of the bottom half of the R and R section. The lowest peak in the R and R section was 145° F measured at a location aft of the nose fairing (PB14). Low heating in this area was also indicated by wind-tunnel data.

5.1.1.4.5 Adapter sections: Figure 5-10 shows measured temperature time histories for the adapter sections at three locations. These three outer skin locations are along the BY centerline and show typical temperature time histories. The highest measured temperature in this region (PG06) was 320° F.

Internal ring temperatures at Z= 70 indicated little temperature rise by showing an average maximum temperature of 150° F. The pyrotechnics at Z = 103 indicated no temperature rise. Maximum temperatures on the adapter-GLV internal structural ring at Z = 13 averaged 160° F. The adapter sections, which are primarily designed to withstand launch heating and structural effects, indicated temperatures lower than predicted and thereby assured the design margins.

5.1.1.4.6 Design temperature predictions: Figure 5-6 to figure 5-10 shows predicted temperatures for the cabin section in the vicinity of PD06 at Z = 189, the RCS section in the vicinity of PC07 at Z-189, the R and R section in the vicinity of PB05 at Z = 217, and the adapter sections in the area of PF11 at Z = 83. These predictions are based on conservative design assumptions rather than being an actual best estimate. For the analysis of these design launch temperatures, representative areas of the spacecraft were chosen rather than analyzing each measured temperature location individually. This approach is logical because the Gemini spacecraft is designed for the more severe reentry conditions, and launch temperatures are not critical. These temperature predictions were made by using the nominal mission trajectory supplied by Aerospace (case 31) and the Patrick Air Force Base atmosphere. Measured temperatures on the cabin section in the vicinity of PD06 agree fairly well with predictions. Measured temperatures in other areas of the spacecraft are all lower than design predictions. The data do not indicate any problems which would necessitate redesign or affect any other mission in an adverse manner.

5.1.1.5 Acoustic environment.- Spacecraft 1 contained two audio microphones located on the overhead beam within the pressurized cabin. Output from the microphones was fed to onboard octave band analyzers to supply commutated octave band and overall data. The two systems were complementary in order to provide coverage of the range from 105 to 155 db for the overall level and from 90 to 150 db on the octave bands. The peak level was 127.5 db at LO + 68 seconds and had the octave band distribution indicated in figure 5-11. For comparison, figure 5-11 presents the levels at LO + 5 sec when ground reflection effects were maximum.

These measured values were within the range of expected values and compared with a qualification spectrum of 135 db for cabin equipment.

5.1.1.6 Pressure environment.- Pressure transducers were located at nine places in the spacecraft for this mission. Absolute pressures were measured by all of the transducers except one. This transducer measured differential pressure between the cabin and the forward compartment. The transducer locations are shown in figure 5-12.

Measurements made by the transducers were recorded from 15 seconds prior to lift-off until 00:07:15 g.e.t. Measurements were also recorded during the first, second, and third orbital passes when the spacecraft passed over stations where a telemetry signal could be received. Except for the pressure measurement between the cabin and the forward compartment, the period of primary concern for pressure measurement is between lift-off and LO + 130 seconds. During this period of a launch phase, internal pressures in closed compartments do not decrease at a consistent rate, and a differential pressure on the structure can occur. The pressure differential between the cabin and the forward compartment increased from zero at lift-off to normal operating pressure during the first 51 seconds of the launch phase. Typical pressure measurements during the launch phase are shown in figure 5-13. An additional measurement made after the third orbital pass indicated that the cabin pressure was 4.8 psi. Based on a time of 04:49:00 g.e.t. and a cabin volume of 75 cu ft, the leak rate after the third orbital pass was computed to be 700 cc/min. The design leak rate for spacecraft 1 was 1,000 cc/min.

The measured incremental pressure of the adapter above ambient pressure is shown in figure 5-14. Also shown are the upper and lower boundaries of the allowable pressure and the predicted pressure based on a design venting configuration with 15.023 sq in. of leakage area. The actual leakage area, taken from measured data of a pressure test of the adapter after it was mated to the launch vehicle, was calculated to be approximately 9 sq in. Since the measured leakage area was less than the predicted area, the measured pressure was greater than the predicted pressure as shown in figure 5-14. From a structural, load bearing standpoint, it is desirable that the incremental pressure on the adapter be near the upper boundary for ring stress.

The computed differential pressure between the conical section equipment bay and the landing gear bay was negligible. The differential pressure between these two bays was assumed to be zero for design purposes.

5.1.1.7 Assessment of overall structural performance.- The GT-1 flight demonstrated, for a given set of conditions, the compatibility of the spacecraft—launch-vehicle integrated structure. It was not a test of design strength in that the flight conditions were far short of the conditions used as design criteria in the design of the structure. High wind shears and gusts, switchover transients, and high angles of attack in the abort situation which dictated the design strength were not encountered on this flight. The GT-1 flight demonstrated the important fact that the dynamic response of the structure was extremely low for the given set of conditions to which the vehicle was exposed. It is important to point out, also, that since structural dynamic response predominates at low angles of attack, the set of conditions to which the GT-1 vehicle was exposed was very near that required to give maximum response. However, the statistical nature of vibration does not permit one flight to establish firmly the behavior of those to follow; therefore, some later flights may not be as free from vibration as the GT-1 flight. Later flights may very well produce twice, or even three times, the response of the GT-1 flight. Some Mercury flights varied by a factor of two, but with the first Gemini flight showing a measured response of less than one-tenth the structural capability, satisfactory structural performance has been demonstrated as far as the vibration is concerned.

This flight also demonstrated that the structural temperatures in the adapter assembly are lower than predicted, and that all unpressurized compartments vented satisfactorily, without approaching either the upper or lower design pressure limits.

From the results presented in the previous paragraphs, it is estimated that the GT-1 vehicle experienced about 20 to 25 percent of its stress capability, which is an indication of a considerable allowance for structural loading under the more severe conditions that may be encountered in future flights.

5.1.2 Major Systems

5.1.2.1 C-band radar transponder performance.- The C-band transponder system furnished accurate range information throughout four orbital passes for the ground radar tracking network. Amplitude modulation, produced by the phase shifter, was reported from several stations. Although signal varied because of spacecraft tumbling, all stations were able to maintain automatic tracking. The duration of the transponder signal reception varied from about 2 to 6 minutes per pass. Several stations reported an increase in signal strength after the spacecraft passed the point of closest approach. Signal strengths up to 45 decibels above the noise level were observed. Transponder code spacing remained nominal throughout the mission, and frequency drift remained within tolerance. During the entire recorded flight of three orbital passes, the C-band transponder and its associated radiating subassembly performed in accordance with the design intent. The capability of this portion of the communications system to enable a satisfactory determination of spacecraft position in space when used with the planned ground tracking equipment was adequately demonstrated.

5.1.2.2 Telemetry and Instrumentation System.-

5.1.2.2.1 Telemetry: Excellent telemetry transmission was obtained during this mission. During the launch phase, the signal strengths varied as would be expected as a result of launch vehicle flame attenuation until LO + 154.0 seconds when flame attenuation ceased, and a good signal was received. At LO + 154.8 the separated launch-vehicle first stage evidently fell between the ground receiving antennas and the spacecraft causing a telemetry dropout of 0.5 second. Normal signal strengths were again obtained at LO + 155.3 seconds, and began gradually decreasing as the spacecraft moved farther away until loss of signal (LOS) occurred at LO + 438.0 seconds.

A comparison of TM received RF signal strengths obtained from Telemetry II or MCC, and Grand Bahama Island and Bermuda ground stations for the 230.4 mc to 246.3 mc links during the flame build-up and telemetry dropout time periods is given in figure 5-15. It can be seen that although flame build-up and telemetry dropout occurred at the Cape Kennedy tracking stations, the different look angles used by the Grand Bahama Island station provided acceptable telemetry reception. Table 6-I presents the telemetry reception times at various tracking stations.

5.1.2.2.2 Instrumentation: Results of test data gathered during the GT-1 mission indicate satisfactory operation was realized in all measurement areas. Complete loss of data was experienced on only 1 of the 131 parameters and a partial loss of data occurred on a vibration parameter as discussed below.

Temperatures - A general survey of temperature data indicates that the readings obtained were considerably lower than expected. A wealth of information was made available since only one of the temperature sensors failed to operate satisfactorily during the mission. One of the adapter retrograde section thermocouples was discovered to be inoperative prior to the final simulated mission, but no attempt was made to repair the sensor because it was inaccessible when launch vehicle and spacecraft were in the mated condition. It was not considered a mandatory parameter for flight since there were two other thermocouples in close proximity. (Refer to subsection 5.1.1.4 for detailed temperature results).

Pressures - Pressure readings experienced during the flight were as expected. The absolute pressure transducers (0 to 15 psia), beginning at lift-off, indicated near full-scale readings (sea-level pressure) and fell off to zero at a gradual rate as the spacecraft gained altitude. The cabin differential pressure transducer (0 to 6 psid) which was referenced to the forward compartment, indicated zero at the beginning of the flight. The indication increased to full scale at altitude and then dropped back to the expected nominal reading for the remainder of the telemetered flight. (Refer to subsection 5.1.1.6 for more detailed pressure readings).

Vibration - Data results indicate that all vibration systems were operating satisfactorily. Full-scale values for the sensors appear to have been well chosen for most of the high and mid-frequency systems since most of the available range of the systems was being utilized. However, one of the pickups mounted in the area of the environmental control system (ECS) pump package exceeded the full-scale indication during the period of maximum dynamic pressure and resulted in some loss of data. A close review of the data shows that the output from the crystal accelerometer is approximately 20 to 27 percent (extrapolated figure) higher than the full-scale (±32g) reading established for the charge amplifier. The high-input charge capacitance of the accelerometer drove the transistorized charge amplifier into saturation which resulted in the clipping of all data peaks at approximately 15 percent (±37g) above the full-scale setting. All data above the 15 percent level cannot be determined accurately and will result in attenuation of the power spectral density plots by a factor approximately equal to the amount of data clipped by saturation (the difference between the point of clipping, which is 15 percent of full-scale setting, and the 20 to 27 percent figure of data which is extrapolated to be over the full-scale setting of ±32g). This situation would tend to indicate that all data are attenuated by a factor of approximately 10 percent.

Low-frequency vibration systems appeared to be well within their operational range and functioned properly in all cases despite the relatively low readings obtained in comparison with the full-scale readings that had been set up for anticipated higher levels. (Refer to section 5.1.1.3 for more detailed vibration analysis.)

Accelerations - The low-frequency (0 to 1 cps) static accelerometers mounted in the cabin area functioned satisfactorily. The Z-axis accelerometer reading showed a gradual rise from the 1.0g reading prior to lift-off to a peak at BECO, dropped off sharply, again rose gradually to a second peak at SECO, and then dropped to zero for the remainder of the telemetered flight. The signals from the X- and Y- axis accelerometers were of a relatively low amplitude, but they were adequate for determining proper operation and accurate data. (Refer to section 5.1.1.3 for more detailed acceleration analysis.)

Acoustic noise - Data results indicate that the acoustic systems functioned satisfactorily and obtained valid data throughout the telemetered flight. The data results indicate that the acoustic noise level never reached a level high enough to excite system 2 (set for 129 to 155 db). The summation of the eight octaves between 37.5 to 9,600 cps on system 1 is higher than the reading obtained on the overall channel of system 1. This was expected since high cross-talk, due to the imperfect filters (see figure 5-16), allowed energy associated with a given octave to be present in adjacent octaves and resulted in high readings. (Refer to section 5.1.1.3 for more detailed information).

5.1.2.3 Electrical.- The electrical system performed as expected during the mission. The lift-off amperage and voltage were 6.3 and 24.0 respectively. A slight increase in current to 6.5 amps was normal and was due to temperature effects. The current and voltage remained essentially the same during the entire mission (that is, three orbital passes). From the results obtained during preflight test of a similar battery, the spacecraft battery was expected to have a lifetime of 48.5 amp-hours. Launch-complex testing of the system resulted in a consumption of 0.61 amp-hour during the precount, and 1.50 amp-hour during the count. The life of the battery would then be expected to be 7.1 hours, assuming that the current remained at 6.5 amps.

5.1.2.4 Umbilicals.- At the time of engine ignition, T-3.5 seconds, the reentry section umbilical cable was electrically ejected and a drop-weight actuation signal was simultaneously given. The cable and connector normally fall away from the spacecraft and downward until the drop weight snatches them away. During the launch of the GT-1 vehicle, the cable twisted as it fell freely downward and away for 0.5 to 0.75 second until it was below the two coolant umbilical lanyards. At this time, the drop-weight actually moved, snatched the reentry section umbilical away, causing it to snag the coolant umbilical lanyard. This action caused a 2.5-second premature release of the coolant umbilical; however, there was no detrimental effect on the mission. On future spacecraft, this premature release cannot occur since all three umbilicals will be released simultaneously with initiation of exploding the launch vehicle hold-down nuts.

5.2 GEMINI LAUNCH VEHICLE PERFORMANCE

The Gemini launch vehicle 1 used for the GT-1 mission functioned normally throughout the powered-flight phase of the mission.

The function of the low-level propellant sensors to initiate cut-off was disabled for this flight. Staging was normal, and the second-stage engine shutdown discrete signal was sent by the radar guidance system at LO + 339.194 seconds and caused SECO to occur at LO + 339.23 seconds. Tail-off was accomplished in approximately 14 seconds. Insertion velocity was 25,785 ft/sec at a flight-path angle of -0.1251° and an altitude of 87.1 nautical miles.

An evaluation of the performance of the individual systems is reported in the following sections.

5.2.1 Structure

The launch-vehicle structure was satisfactory throughout powered flight. Longitudinal oscillation, skin temperature, and flight loads are of particular interest and are therefore discussed in detail in the following paragraphs.

5.2.1.1 A longitudinal oscillation instability has been observed on Titan II flights. This instability is characterized by a sustained 11-cps oscillation in the first longitudinal structural mode beginning at approximately LO + 110 seconds and ending at LO + 130 seconds. The phenomenon is associated with a structural-propulsion system interaction in which structural axial oscillations are propagated as pressure and flow perturbations in the propellant feed system. As a result of these perturbations, an oscillatory thrust which tends to maintain the oscillation is fed back into the structure. Critical combinations of structural and propulsion system parameters, which vary with flight time, result in various degrees of instability. It has been shown analytically and verified by flight-test results that the oscillation can be suppressed by the insertion of a piston-type fuel accumulator and an oxidizer standpipe in the propellant feedlines.

The GLV-1 was equipped with these devices. A review of flight data indicates that a sustained longitudinal oscillation of critical amplitude did not occur. However, intermittent response was noted with a maximum value, in compartment 1, of ±0.2g at LO + 151 to 153 seconds at a frequency of 17 cps. In compartment 5 the corresponding maximum response was ±0.85g. A time history of g-levels during first-stage flight in compartment 1 is shown for comparison in figure 5-17 with g-levels experienced on two previous research and development flights of Titan II missiles equipped with suppressors.

Longitudinal oscillations were also measured on the spacecraft bulkhead, where crew seats will be attached. Figure 5-18 presents data measured in the spacecraft with launch-vehicle data for comparison. The values shown are in fair agreement. except at LO + 153 seconds when the spacecraft vibration peaked at ±0.4g and launch-vehicle vibration at ±0.2g. Figure 5-19 shows the oscillograph traces at LO + 153 seconds for the spacecraft accelerometer QA11 as compared with those in compartments 1 and 5 of the launch vehicle. The spacecraft maximum magnitude was measured on only one cycle of oscillation having a duration of 0.06 second, occurring within a series of approximately four cycles with magnitudes of ±0.3g. The increase in magnitude for the launch vehicle to the spacecraft may be from two sources: one, amplification through the adapter structure, and two, a change in the longitudinal node location. Although the adapter structure has been shown by test to have no amplification at frequencies down to 35 cps, the possibility exists for amplification at 17 cps. In studying the first longitudinal mode shape of the entire vehicle, it is found that the measurements from the three accelerometers agreed fairly well with predicted response amplitude relationships. The response of GLV compartment 5 was large in amplitude and opposite in phase to that measured by the forward accelerometers, and the spacecraft accelerometer measured about twice the amplitude measured in GLV compartment 1. In figure 5-20 this two-to-one relationship is partially explained in that the node of the first bending mode, at LO + 148 seconds, is about as far aft of the GLV compartment 1 accelerometer as the spacecraft accelerometer is forward of it. Early in the flight, the node was much further aft, which explains the fact that the GLV compartment 1 and spacecraft accelerometers read very close to the same amplitude at LO + 78 seconds. This trend, however, does not appear to account for the total difference. In view of the short duration and higher frequency of the response measured in the spacecraft, it is not considered to represent a serious detriment to the pilot environment.

5.2.1.2 Skin temperature.- The outer surface of the launch vehicle forward skirt was protected from excessive aerodynamic heating with an ablative coating of sprayed-on silicone rubber, and temperatures were recorded by thermocouples on the inner surface at locations shown in figure 5-21. The maximum temperature attained was 157° F as compared to 325° F predicted, based on protuberance heating. It now appears that the addition of more insulation, for which a weight increase of 25 pounds has been budgeted, can be eliminated. In fact, consideration could be given to reducing the amount of insulation now on this area.

5.2.1.3 Flight loads.- Using the measured flight wind profile and predicted flight trajectory, a $q\alpha$ value of approximately 32 percent of the design was obtained.

Response of the structural vibratory modes of low frequency was less than predicted through the transonic region. There was a slight indication of first bending mode response and only a small amplitude indication of second bending mode, giving a peak modal moment of 200,000 in-lb on the launch vehicle. The spacecraft-launch-vehicle structural interface loads and moments are discussed in subsection 5.3.

The amplitude of random noise measured by the accelerometer in compartment 1 was approximately the same as that produced by the engine acoustic intensity at ignition, indicating that the response of the forward skirt to transonic buffeting is no greater than that due to engine acoustics.

At various times in the trajectory, intermittent response of structural, slosh, and engine mode frequencies were noted on both the spacecraft and launch-vehicle lateral accelerometers. The amplitudes of these responses were small; and resulting loads, when combined with other incident loads, were well below design levels. A more detailed analysis of the GT-1 flight data will be performed by the GLV integrating contractor. Using flight measurements, spacecraft interface and launch-vehicle loads will be computed for the times of staging, maximum transonic buffeting, and the maximum normal load conditions.

5.2.2 Propulsion

5.2.2.1 Engines.- No significant events related to the engine subsystem occurred between the wet mock simulated flight and launch of the GT-1 vehicle.

During the flight, the performance of the YLR-87-AJ-7 and YLR-91-AJ-7 engines was satisfactory during both steady-state and start-shutdown transients. The stage I engine started at 10:59:58.196 a.m. e.s.t., and lift-off occurred 3.47 seconds later. No low-level longitudinal acceleration effects were discernable from the engine pressures. The MDS system on the engines performed as planned. The 87FS2 signal (BECO) was initiated as planned by actuation of the thrust chamber pressure switch (TCPS) due to oxidizer exhaustion, with a normal shutdown. Stage II start and burning time were as planned, and a normal shutdown was initiated by SECO guidance command. The thrust, specific impulse, and mixture ratios achieved as obtained from preliminary data reduction by the GLV integrating contractor are presented in table 5-II.

Telemetry data on flight performance obtained during the launch-vehicle flight were used directly as input data for an analytical model designed to aid in further detailed evaluation of vehicle performance. Each subassembly was analyzed separately in determining thrust-chamber and gas-generator flow rates and mixture ratios and autogenous pressurant flow rates. Results were obtained in terms of time-history curves of those parameters. Inspection of these curves indicated nominal operation of all engines assemblies throughout powered flight of the vehicle. Figures 5-22 and 5-23 are thrust-chamber pressure curves for stage I and stage II.

The following subassembly 2 parameters are presented:

(a) Gas generator oxidizer flow rate (fig. 5-24)

(b) Gas generator fuel flow rate (fig. 5-25)

(c) Thrust chamber oxidizer flow rate (fig. 5-26)

(d) Thrust chamber fuel flow rate (fig. 5-27)

(e) Oxidizer and fuel pressurant flow (fig. 5-28)

It is concluded that the performance of the engine subsystem during GLV-1 flight was satisfactory.

5.2.2.2 Tail-off.- The data obtained indicated that the shutdown transients for both stages were normal and as predicted. The shutdown transients for stage I engines as well as other staging events are illustrated on figure 5-29. Transients occurring during engine shutdown for stage II are shown on figure 5-30. For the purpose of accurately determining the total impulse generated after stage II shutdown, or thrust tail-off, the data in figure 5-31 are given. From these data it was determined that the total impulse occurring after SECO, which is defined as the time the signal to shutdown the stage II engine is received at the shutdown bus from the airborne guidance discrete signal decoder, compared closely with the predicted value. Aerospace predicted a value of 43,500 ± 6,300 lb-sec for the tail-off thrust occurring after SECO; whereas the value actually obtained by using the data shown in figure 5-31 amounted to 45,800 lb-sec. As a comparison, the GLV integrating contractor obtained a value of 45,020 lb-sec in these calculations.

5.2.2.3 Pressurization.- On F-1 day while performing launch preparations, the vent plug in the stage II fuel autogenous pressurization line was removed and pressure in the line was detected. Suspecting a leaking burst disc, decay leak checks were performed from both sides of the disc assembly, with negative results. Hence, the disc was retained for flight. The pressure had been locked in during a previous subsystem functional test.

The tank pressurization was acceptable during powered flight and remained between maximum and minimum limits. Stage I fuel-tank pressure began nominally, dropped to the minimum at LO + 65 seconds, and then rose to nominal. Stage I oxidizer and stage II fuel tank pressures remained slightly lower than nominal, with the latter showing some recovery toward the end of flight. During stage I powered flight, the stage II oxidizer suction pressure (measurement no. 510) gradually rose to the limit of the measurement and remained there until 87FS2 (BECO). This rise was a result of the combined effect of vehicle acceleration and ullage pressure acting on the liquid head. During stage II powered flight, pressure on this tank began at a nominal level, then decayed below nominal, and reached the minimum for the last 20 seconds of flight.

One anomaly was noted in the stage I autogenous system. The oxidizer pressurant orifice inlet pressure (Po_{Poi}), measurement no. 26, showed an unexpected rise in pressure beginning at approximately LO + 126 seconds. The pressure rapidly increased from 520 psia to 587 psia at LO + 141 seconds and decayed only slightly from that time until 87FS2. This anomaly was not reflected in stage I oxidizer tank pressure, nor in propellant pressures. A curve of this parameter showing the anomaly is presented in figure 5-32. Curves of the actual tank ullage pressure superimposed on the pressure limits for stage I and stage II are included in figure 5-33.

Performance of the airborne pressurization system during flight was entirely satisfactory. The anomaly in the autogenous system had no effect on system performance. It is recommended that this parameter be monitored closely for this phenomenon on future flights and that the performance of the Po_{Poi} transducer on previous Titan II flights should be investigated.

Due to the behavior of the stage II oxidizer suction pressure, the range of this measurement should be extended to account for the pressure rise, which is considered normal.

5.2.2.4 Propellants.- On F-1 day, due to a problem in the propellant transfer and pressurization system (PTPS), the oxidizer for stages I and II were loaded at temperatures of approximately 1.5° (stage I) and 3.3° (stage II) higher than desired temperature. This created a differential temperature between propellants, which would result in a mixture-ratio variation in flight. The resultant sacrifice in payload capability was discussed with the GLV integrating contractor, and efforts were made to retard the rise rate of the stage II oxidizer by providing full curtain cover plus a plastic film covering over the tank barrel section up until the time for erector lowering. By so doing, the differential temperature was brought well within acceptable limits by launch time.

The propellant temperature rise rates in the tanks during flight were as follows: stage I fuel, 7° F; stage I oxidizer, 16.5° F; stage II fuel, 9° F; and stage II oxidizer, 4° F. The outage on stage I was 926 pounds of fuel, and on stage II the outage was 286 pounds of oxidizer. The burning time margin left in stage II was 3.27 seconds, the fuel residual was 465 pounds, and the oxidizer risidual was 1093 pounds. These data are contained in table 5-III.

Although the shutdown capability of the propellant tank low-level sensors were deactivated on GLV-1, there was instrumentation which showed the performance of the low-level sensors, as well as all other tank-level sensors. An analysis of the sensors was made from the output of this instrumentation.

The level sensors on the stage I fuel tank were erratic in operation. The main problem was in the sensors indicating cover when they were uncovered. The level sensors on the stage II fuel tank were erratic during staging and intermittent from first uncover indication to SECO. One high-level sensor gave an intermittent indication for 158 seconds during stage II operation; the other high-level sensor indicated cover for the last 165 seconds of powered flight when it was actually uncovered.

The level sensors on the stage I oxidizer tank were normal except for erratic operation after the sensors were uncovered during staging.

The level sensors on the stage II oxidizer tank were erratic during staging but normal at all other times.

The stage I shutdown sensors were not armed until LO + 139.52 seconds, and the stage II shutdown sensors were not armed by the electronic timers in the TARS package until LO + 322.56 seconds. For instrumentation purposes, the data may be considered erratic but useful.

During the flight, the performance of the propellant feed system was satisfactory in all respects. Prior to flight, a problem occurred with propellant temperatures which requires corrective action. The initiating problem in the PTPS is being investigated. However, in the event of a reoccurrence of this problem or others affecting propellant temperature, an additional element of control is needed to adjust propellant temperatures after propellants are on board the vehicle. To this end, it is recommended that the erector curtain system be closely scrutinized for necessary design changes to make possible the complete closure of the curtains for temperature control when necessary. Also, the possibility of designing and building a set of external insulating blankets for the tanks should be considered.

5.2.3 Flight Control and Hydraulic System

5.2.3.1 Flight control. - The overall operation of the flight control system was satisfactory during stage I and II flight. All programed flight control discretes, including the roll and pitch programs, occurred within acceptable limits resulting in a near-nominal flight.

A yaw offset appeared at LO + 61 seconds and remained for 24 seconds, requiring a 1° engine gimbal correction. This occurrence was observed during the region of maximum dynamic pressure and was caused by aerodynamic conditions.

At separation, a shift in the output level on pitch, yaw, and roll displacement gyros and actuators was noted; however, it did not adversely affect the operation of the flight control system. A gradual change in these attitude error signals was observed during the progression of the stage II flight. Such a change could be due to a change in center-of-gravity location associated with the depletion of propellants. Preliminary investigations have indicated that the shift at separation was a result of a slight misalinement of the engine thrust vector. Section 5.2.4 details the effects of this condition on the launch-vehicle performance. A detailed investigation will be conducted by the launch-vehicle integrating contractor and a supplementary report will follow.

There was no degradation noted in rate gyros (primary and secondary), although the stage I rate gyros were slightly "noisier" than the stage II gyros. The rate gyro "noise" is a peculiarity of the rate gyro telemetry signal and as a result, a modification has been proposed by the launch-vehicle integrating contractor to reduce this effect on the second mission.

5.2.3.2 Hydraulic System. - An examination of the data obtained during the flight indicates that the stage I and II hydraulic systems performed normally and as expected, except for one item which is presently being investigated. This item is that the buildup of pressure in the secondary hydraulic system at start of stage I was undesirably high. Within one second after engine start, the hydraulic pressure increased to 4,500 psia, which is the limit of the pickup, and remained there for 0.2 second before decreasing to a normal pressure of 3,000 psia. This same pressure peak or spike was also noted during startup of the secondary hydraulic system in the preflight SCF. The system was tested to a pressure of 5,000 psia prior to launch of the GLV-1. This pressure buildup does not occur on the Titan II, which uses only a single hydraulic system; or on the GLV primary hydraulic system. The launch-vehicle integrating contractor is currently investigating the effects of this short-duration pressure buildup to determine what measures should be taken to correct the situation.

A chronological examination of pressure traces of the primary and secondary hydraulic systems on stage I indicates that approximately 2 minutes before launch, the secondary hydraulic system was pressurized to 3,150 psia using the airborne electrical motor-driven pump which requires ground power for operation. About 10 seconds later, through the use of the hydraulic selector valve, the stage I hydraulic system was switched from the secondary to the primary hydraulic system which built up to a pressure of 3,150 psia in a period of about 2 seconds. During primary pressure buildup, the secondary hydraulic pressure started returning to zero pressure which is normal. About 0.2 second after engine ignition, the secondary hydraulic system pressure started building up rapidly, as previously discussed. The secondary system pressure returned to about 3,000 psia by lift-off. A normal damped oscillation in primary hydraulic system pressure was observed during the transition from the electrically driven pump to the turbine driven pump. The pressure varied from a maximum of approximately 3,400 psia to a minimum of approximately 2,480 psia and lasted from 87FS1 (stage I engine start) to LO + 5 sec. This pressure of 2,480 psia is well above the switchover point of 1,500 ± 50 psia. The hydraulic system operated in the 2,640 to 3,000 psia range throughout the remainder of stage I operation, with no excessive demands on the system, indicating minimum gimbal action. It should be noted that at approximately LO + 70 sec, the hydraulic fluid temperature started to rise at an approximately steady rate until 87FS2 (BECO). At LO + 70 sec, the temperature was approximately 105° F, and at 87FS2 the temperature was about 184° F. The secondary system went from approximately 95° F to about 167° F during the same interval. Associated with this temperature rise, an increase in fluid level and a decrease in pressure was observed which was within the previously described limits.

Chronologically, the stage II hydraulic pressure started building up 0.3 second after stage II engine ignition and increased to a pressure of 3,870 psia within 1.6 seconds. Four seconds later, the pressure had returned to about 3,000 psia where it remained with no indication of excessive demand during the remainder of stage II operation. In the 15-second period after stage II engine cut-off, the pressure gradually declined to 1,700 psia, and then in 1 second dropped to 400 psia indicating actuator movement at that point. Movement readings on the yaw and pitch actuators indicated a movement of about 1°. Nineteen seconds after SECO, the hydraulic pressure returned to about 900 psia, where it stayed during the remaining 70 seconds in which data were recorded. This increase and sustaining of hydraulic pressure is possibly due to fuel tank venting through the autogenous line to the gas generator.

5.2.3.3 Post-SECO vehicle transient.- At approximately 11.2 seconds after SECO, several measurement parameters in both the spacecraft and the launch vehicle exhibited a transient which lasted approximately 0.85 second. Engine parameters, spacecraft and launch-vehicle accelerations, rate gyro outputs, and autopilot parameters were affected.

The vehicle rates were noted prior to and after the transient and there were no noticeable rate changes resulting from the phenomena. The exact cause of the disturbance is not known at this time; however, an investigation has been initiated to establish its source. All telemetry parameters showing the transient in the GLV and spacecraft are presented in figure 5-34 and 5-35, respectively.

This disturbance caused a 0.2° pitch-up and a 0.5° yaw-right engine movement and was also reflected on the low axial accelerometer compartment 2 (measurement no. 0699) as a 0.2g peak-to-peak transient. Preliminary investigation indicates that the cause of this disturbance was an abrupt short-duration increase in the low-level thrust emanating from the roll-control nozzle being fed from the gas generator. The gas generator at this time in flight is fed from the fuel-tank venting through the autogenous line.

The above mentioned theory is not expected to be conclusive at this time; however, this disturbance is considered worthy of a supplemental report which will be the result of the investigation now in process.

5.2.4 Guidance

The launch-vehicle radio-guidance system performed well and guided the second stage to approximately 2σ cutoff conditions which were within the acceptable 3σ limits. As shown in table 4-II, the trajectory was 20 ft/sec high in space-fixed velocity, 71.6 ft/sec high (negative) in crossrange velocity, 2,424 feet low in altitude, and 0.143° low in flight-path angle. Except for crossrange velocity which would have to be corrected by the spacecraft orbital attitude and maneuver system because of its direct implication on rendezvous, errors of this magnitude are of minor significance to the Gemini mission.

The axial vibration recorded during stage I operation indicated a series of transients with frequencies of approximately 10 to 30 cps, and a maximum half amplitude of 4g. The time interval between 3 and 5 seconds after lift-off showed a g-level of 1.2g rms with a peak level of 3.9g rms.

5.2.4.1 Open-loop guidance.- The open-loop guidance system for the first 162.0 seconds after lift-off consisted of programed events in the roll, pitch, and yaw channels provided by the primary flight control system. Table 5-IV presents a comparison of planned and actual event times and vehicle rates.

The Gemini launch vehicle (GLV) had a launch azimuth heading of 71.9° after completion of the roll program rather than the planned 72.0°. The programed events occurred within acceptable limits. As can be seen in section 4.3, a near nominal first-stage flight was flown. The errors at BECO were 83 ft/sec low in velocity, 2,655 feet low in altitude, and 0.05° low in flight-path angle.

5.2.4.2 Closed-loop guidance.- The guidance system acquired the track beacon of the launch-vehicle, tracked in the monopulse automatic mode, and was locked on continuously from lift-off to LO + 414.8 seconds. Track was maintained to an elevation angle of 0.78°; however, the gradual deterioration of the processed data quality indicates that the data after SECO + 10 seconds were of questionable value.

Rate lock was continuous, except for a momentary interruption at staging, from LO + 46.6 sec to LO + 397.2 (47.8 seconds after SECO). Lateral rate noise at SECO was 0.03 ft/sec. Steering commands were transmitted as planned at LO + 168 seconds. Following the initial full-scale pitch-down steering command (2 deg/sec) which is a normal characteristic of the guidance equations, the steering gradually returned to a relatively small and constant pitch command after 18.5 seconds. A pitch-down command of about 5.3 percent persisted until SECO, producing a continuous pitch rate of 0.106 deg/sec. The initial yaw commands were of small magnitude, with the command over the first 40 seconds of steering amounting to a negative correction of 0.181°. During the final 90 seconds of steering, the yaw correction amounted to about 0.371° in the opposite direction. SECO occurred at LO + 339.2 sec, which was 3.7 seconds later than planned, and at an elevation angle of 6.67° as compared with a planned 6.85°. At SECO plus 20 seconds, tumbling velocities were 0.25 deg/sec pitch down, 0.65 deg/sec yaw left, and 0.3 deg/sec roll clockwise (CW).

The computing system, in conjunction with the RGS track, rate, and airborne systems, completed all launch operations in a normal and satisfactory manner. The inertial guidance system updates were also properly computed and transmitted to the computer register at the correct times.

In figure 5-36 the velocity and flight path angle are shown in the region of SECO. The launch-vehicle radio-guidance system (RGS) data are shown in figure 5-37 to illustrate the data quality during the time of the go-no-go computation. In figure 5-37, the variation of flight-path angle with velocity is the type of display used by the Flight Dynamics Officer in the Mission Control Center for the orbital go-no-go decision. The launch-vehicle guidance data indicated a "go" condition. Since MISTRAM I (Valkeria) data in the region of SECO were not provided to GSFC from the Range Safety Computer (IP-7094) the redundant orbital determination was not made (see section 6.3).

The insertion errors at SECO plus 20 seconds were higher than anticipated, considering the good quality of the RGS radar data. An explanation of these errors can be attributed to the continually changing pitch and yaw displacement of the vehicle attitude, with respect to the gyro reference, that occurred after staging (see section 5.2.3) due to thrust center-of-gravity alinement. Normally, the RGS would correct for a constant attitude displacement because of its closed-loop nature. However, these displacements changed with time, and the smoothing characteristics of the guidance equations prevented recognition of these changes. Preliminary launch-vehicle simulations on a digital computer, using the GT-1 RGS equations and the measured pitch and yaw attitude displacements with no simulated radar noise, produced flight-path angle, altitude, and yaw velocity errors of the same magnitude and direction as the actual insertion conditions.

Due to the results of this investigation, improvements in engine alinement and modification of the guidance equations are deemed necessary to correct this situation.

5.2.5 Electrical Power System

The electrical power system performed satisfactorily during countdown and powered flight. The APS steady-state voltage ranged from 29.3 volts to 29.9 volts. The IPS steady-state voltage ranged from 28.3 volts to 29.2 volts. Specification values are 29 ± 2 volts. Staging currents appeared nominal, with TM dropout just after or during the latter portion of the current excitation to the staging nuts and stage II start cartridge. The APS bus experienced a temporary drop to a minimum or 27.3 v d-c. The corresponding IPS drop was a minimum of 26.9 volts. The span of the disturbance was approximately 1 second.

A drop in frequency of the static inverter from a steady-state value of 401.0 cps to a steady-state value of 400.3 cps occurred just after SECO. There was no change in inverter voltage at this time.

The power remaining in each battery at SECO was estimated to be 5.3 ampere-hours; that is, the time remaining for supplying nominal voltage after SECO was about 12 to 14 minutes.

5.2.6 Malfunction Detection System

The malfunction detection system (MDS) performed satisfactorily throughout the flight. During the first- and second-stage flight, the vehicle experienced no turning rates that approached the MDS overrate limits. The nine spin motor rotation detection (SMRD) measurements and the overrate warning measurement show that the rate switch package operated properly. No switchover commands were noted on the inputs to the malfunction detection package (MDP) power amplifiers, and there were no outputs to the flight control system.

The malfunction detection thrust chamber switch pressure (MDTCPS) for subassembly 1 was monitored by measurements (no. 358) for the underpressure warning, and (no. 356) for the A and B back contacts. The switches picked up at 595 psia at T-1.76 seconds when the engine started. The specification value for the switch pickup is a maximum of 595 psia. The first switch dropped out at 510 psia at LO + 154.6 seconds, as shown by measurement no. 356. The specification dropout value is 515 to 585 psia.

The MDTCPS for subassembly 2 monitor measurements were no. 359 for the underpressure warning and no. 357 for the A and B back contacts. The switches picked up at a pressure of 585 psia at T-1.68 seconds when the engine started. Upon thrust chamber pressure decay, the first switch dropped out at 495 psia at LO + 154.6 seconds, as shown by measurement no. 357. The second switch dropout point for each engine subassembly cannot be observed since the first switch to drop out opens the back contact monitor circuit. Measurement no. 356 and no. 357 are sampled 100 times per second; therefore, the switch dropout times shown by the measurements could be up to 10 ms later than actual. The switch dropout points may have actually been within specification limits.

The MDTCPS for subassembly 3 was monitored by measurements no. 853 for the underpressure warning and no. 861 for the A and B back contacts. The switches picked up properly when the engine started; however, the actuation pressure cannot be measured since the switching occurred during the TM dropout period at staging. The MDTCPS for subassembly 3 dropout point was at 535 psia at LO + 340.25 seconds. The nominal dropout value is 550 ± 50 psia.

The MDS tank sensors operated properly during flight. The difference between the A and B sensors on any one tank was well within the specified 0.150 v d-c output difference. The maximum difference in output was between the A and B sensors on the stage II oxidizer tank just prior to SECO. This value was an output difference of 0.06 v d-c. The actual tank-pressure measurements are evaluated in subsection 5.2.2.

The shutdown lockout timers operated properly as shown by measurement no. 856. They timed on at LO + 40.6 seconds. The specification is LO + 40 (±2) seconds.

The two 145-second staging arm timers operated properly, and thereby prevented the engine chamber pressure circuit from presenting a light at staging. The circuits which operate the spacecraft MDS "stage" telelight operated properly on GT-1. Both the IPS and APS circuits were pulled in by the 87FS2 signal (BECO) and dropped out by separation of the interstage lanyard. The presence of the signal was indicated by telemetry for about 0.9 second on each circuit. Numerous signal circuits of the flight-control switching system were telemetered during flight. The data from CKMTA Telemetry II are difficult to interpret because of the similarity between the bilevel signal and the telemetry dropout signals. Preflight checks of the switchover system demonstrated that it was operable during the countdown. No switchover occurred during this flight. Therefore, the brief signals indicated on the telemetry data are telemetry dropouts and do not represent actual events in the switchover circuits.

5.2.7 Instrumentation

5.2.7.1 Airborne.- The airborne instrumentation system performed satisfactorily during the flight test of GLV-1. The airborne system featured one PCM-FM link and one FM-FM link with an associated airborne tape recorder. The tape recorder did recover data during the 400-milliseconds RF blackout at staging. A total of 240 measurements were programed into GLV-1. Of this number, no measurement was lost in its entirety. The following comments, however, are noteworthy:

(a) The measurement (no. 0035) for the piston motion in fuel-surge chamber SA1 and the measurement (no. 0036) for piston motion in fuel surge chamber SA2 became erratic at lift-off. Data were received from these measurements, but they were of practically no value because of wheel slippage on both piston shafts.

(b) The measurement (no. 0685) for the calorimeter slug temperature was normal up to LO + 280 seconds, at which time it dropped out completely; however, this time was well beyond the period of critical skin temperatures.

(c) The measurement (no. 0510) for the oxidizer pump inlet pressure, SA3, reached the maximum scale at LO + 140.3 seconds. This condition has been observed on Titan II research and development flights and the use of a pressure transducer with upper limits of 150 psia instead of 100 psia has been considered.

(d) The measurement (no. 1189) for the vibration tandem actuator, axial, was lost at LO + 119 seconds. Data received prior to this time are valid.

(e) The measurement (no. 0026) for the oxidizer pressurant orifice inlet pressure, SA2, provided an unexplained increase in level at about LO + 125 seconds. This anomoly is discussed in greater detail in subsection 5.2.2.3.

5.2.7.2 <u>Ground Instrumentation</u>.-

5.2.7.2.1 Complex 19: The performance of all land-line instrumentation, including special monitoring of the engine start cartridge temperature, was satisfactory during the prelaunch phase of the mission with the following minor exceptions:

(a) The record-input level on tape recorder 2 decreased at approximately T-200 minutes as a result of a faulty rectifier tube in the connecting chassis power supply. The tube was replaced during a period when recording was not required.

(b) One channel of a Sanborn recorder was unbalanced before the recorder started during the countdown. The channel was adjusted and recalibrated before the recorder start sequence of the countdown.

During the countdown, land-line data acquisition was 100 percent. After launch, the blockhouse telemetry ground station received valid data from the airborne system through LO + 425 seconds, with only two significant dropouts prior to staging. The first dropout of 3 seconds was just prior to 87FS2 (BECO) and the second dropout of 0.3 second was at 91FS1 (stage II engine ignition). Each of these dropouts are normal and are caused by flame attenuation. During stage II powered flight, a third significant dropout occurred as a result of fixed antenna orientation relative to the flight path.

The weight and thrust measuring system performed satisfactorily during the four propellant loadings for the SCF test and two propellant loadings for the prelaunch test. The Z4 load cell began losing output in December 1963 and was changed after the SCF. It was noted that throughout all loading tests, a difference of approximately 0.5 percent existed between the flowmeter preset and the blockhouse weight readings.

5.2.7.2.2 Telemetry Building II at the Cape Kennedy Missile Test Annex (CKMTA): Eight minor dropouts were noted in the PCM flight data, and one dropout was noted in the FM-FM flight data recorded at Telemetry Building II. Representative times, durations, and probable causes of these dropouts are shown on table 5-V. All dropouts can be attributed to reduced signal strengths at the ground receivers due to the vehicle attitude and flame attenuation.

Satisfactory telemetry data were received from lift-off through LO + 438 seconds, which is better than expected. Ten minor dropouts in the PCM flight data and one dropout in the FM-FM flight data were recorded at the Grand Bahama Island telemetry station. The times, durations, and causes of these dropouts are shown in table 5-VI. The use of a quad-helix antenna at the Grand Bahama Island accounts for the slightly reduced performance at that station as compared with performance at the CKMTA Telemetry Building II which used a TLM-18 antenna.

5.2.8 Range Safety System

The range safety system for GT-1 functioned as planned. No operational or equipment discrepancies occurred.

5.2.8.1 Performance of airborne equipment.- The performance of the airborne equipment was satisfactory. Telemetry information from the airborne equipment substantiated proper operation.

5.2.8.1.1 Command receivers: Both command receivers operated satisfactorily throughout powered flight, and telemetry was obtained to loss of signal (LOS) at approximately 438 sec after lift-off. The receiver automatic gain control (AGC) traces agreed completely and indicated occurrence of the following events:

(a) At LO + 100 seconds, transfer from the low-power transmitter at Cape Kennedy to the high-power transmitter at Grand Bahama Island.

(b) At LO + 200 seconds, transfer from the high-power transmitter at Grand Bahama Island to the high-power transmitter at San Salvador.

(c) At LO + 290 seconds, antenna transfer at San Salvador from Sterling I to Sterling II. Measurements 780 and 781 indicated marginal dropout for 0.35 second due to this transfer.

The engine cut-off bilevel telemetry channels from the receivers showed no anomalies.

Receipt of the ASCO signal by the receiver in flight was not telemetered.

After SECO, two null points on the telemetry records of receiver AGC indicated tumbling of the vehicle. The nulls occur about 22 seconds apart.

5.2.8.1.2 Flight termination system: The flight termination system functions verified during the countdown showed no anomalies. None of the functions of the flight termination system were used during the GT-1 mission.

5.2.8.2 Ground complex performance.- The operation of all range safety ground equipment was satisfactory throughout powered flight and operated as planned, with minor exceptions. Range operations were nominal, and all countdown events were met without delay to the count.

The Range Safety Officer's (RSO) displays for real-time flight evaluation were as follows:

(a) Redundant plot-board displays of real-time radar, impact prediction, and vertical flight-path profile

(b) Real-time telemetry of GLV attitude

(c) Closed-circuit television

A postflight evaluation of the plot-board data indicates that the actual flight path was well within the 3σ expected flight path for all parameters. Since MISTRAM data system accuracies have not yet been determined, both the MISTRAM I site (Valkaria) and MISTRAM II (Eleuthera) were operated on a limited commitment basis. The range had planned to track with MISTRAM I until LO + 360 seconds and then switch to MISTRAM II. Prior to the test, an Operations Requirement (OR) change was submitted requesting switchover to be made at LO + 292 seconds. MISTRAM I acquired acquisition track subsystem (ATSS) track at LO + 22.6 seconds and tracked until LO + 386.8 seconds with two dropouts of 0.4 second at BECO and 0.8 second at switchover. It was selected by automatic data and selection program (ADASP) for impact prediction (IP) during the following periods from lift-off:

58.0 to 58.3 seconds
86.5 to 90.2 seconds
91.2 to 91.8 seconds
95.8 to 101.5 seconds
105.0 to 156.0 seconds
174.8 to 245.0 seconds
245.4 to 319.8 seconds.

At LO + 292 seconds (the requested switchover time), MISTRAM II did not have track and MISTRAM I had seven-level data at this time. At 319.7 seconds, MISTRAM II had ATSS track, and switchover was made. MISTRAM II had 6 seconds of precision measuring subsystem (PMSS) data and was selected by ADASP at 427.3 seconds.

The performance of the MISTRAM transponder was nominal throughout the flight, and telemetry data on its operation extended to LOS at about LO + 438 seconds. Operational mode changes, station handoff, and loss of lock were clearly discernable on the telemetry signals. In addition, multipath effects are evident just prior to lift-off. A possible cause is the flame-bucket cooling water. These effects did not interfere with transponder operation.

5.2.9 Pyrotechnics

Since the new destruct initiator with the short-off position was not available in time for the GT-1 flight, a modified Titan II initiator having a short across the detonators was used.

Because of an unexplained explosion of the stage I destruct system at separation during two Titan II flights, it was necessary to take measures to protect the GT-1 vehicle against possible causes. The entire explosive portion of the destruct system in stage I was wrapped with silicone and aluminum fiber-glass tape to protect it against acoustical vibration and heat.

The pyrotechnics on the Gemini launch vehicle for engine starting and lift-off functioned normally as indicated by the events recorder.

Solid-grain temperatures of the engine start cartridges used for the GT-1 mission had a minimum acceptable temperature for efficient operation. As set forth by the manufacturer, the start cartridge (serial number 0002490) on subassembly 1 and start cartridge (serial number 0000436) on subassembly 2 had minimum allowable temperatures of 54° F and 52° F, respectively. The start cartridge on subassembly 3 (serial number 859264) had the more critical minimum allowable temperature of 65° F.

To insure temperature control, an electric heater and a polyethylene bag were used on stage I start cartridges. Two thermocouples were mounted on the skin of each cartridge, and two were mounted in the engine compartment to check both cartridge and ambient temperatures. A heater blanket with a variable power supply was used to maintain temperature control of the start cartridge on stage II. Four thermocouples, one ambient and three to provide case temperature, were installed.

At T-108 minutes, the plastic bag and thermocouples on stage I cartridges were removed. At this time, the temperature on the cartridges were 81° F and 83° F with an ambient temperature of 72° F. When the heater blanket and thermocouples on the stage II cartridges were removed at T-95 minutes, temperatures on the forward and aft cartridge skin were 116° F and 100° F, with an ambient temperature of 70° F. The center thermocouple reading was lost at T-7 hours. Using a temperature decay curve, the temperatures of the start cartridges at lift-off were estimated to be 75° F for stage I and 90° F for stage II.

The specification time interval for start cartridge ignition permits a 200-millisecond differential. On GT-1, a 50-millisecond difference in stage I start cartridge initiation was noted in gas-generator chamber-pressure buildup of subassemblies 1 and 2. This difference was also verified in the thrust-chamber-pressure build-up of both subassemblies, as noted in section 5.2.2.

5.3 GEMINI LAUNCH VEHICLE-SPACECRAFT INTERFACE PERFORMANCE

The electrical and structural integrity of the interface between the Gemini launch vehicle (GLV) and the spacecraft was satisfactory throughout powered flight.

5.3.1 Electrical

The electrical interfacing circuits between the spacecraft and GLV were terminated on the spacecraft side with dummy loads except for the inertial guidance system (IGS) circuits which were tied back on the spacecraft side. The telemetry parameters of the interfacing functions were monitored by the GLV instrumentation system and showed normal operation.

5.3.2 Structural

All available evidence confirms that the interface structural connection between the spacecraft and the launch vehicle did not fail during the entire mission.

During the launch phase of the flight, the loads experienced by the interface were a small percentage of its structural capability. The most conclusive item in this evidence was the continued performance during the first three orbital passes of a spacecraft accelerometer (QC17) located on the structure at the interface.

During powered flight, the excitation of the combined vehicle's low-frequency bending modes caused fluctuating aerodynamic pressure which was measured by accelerometers in both the launch vehicle and the spacecraft. Figure 5-38 presents the comparison of the predicted body-mode frequencies with the frequencies derived from the flight instrumentation. The modal frequencies agree quite closely during the first-stage flight, which includes the period of maximum response.

By using the power spectral densities from the lateral sensors in the spacecraft for a 10-second time loop, and assuming discrete frequency response of the first three body modes, the maximum bending moment at the spacecraft-GLV interface was calculated to be 355,000 in-lb. This moment is equivalent to a compression load of 11,830 pounds. Static tests referred to in subsection 5.1.1.2 have demonstrated a capability in compression of 186,000 pounds for the spacecraft adapter and GLV-adapter interface. An independent evaluation of the bending moment at the interface was performed by using the output of the stage II rate gyros in combination with the computed moment transfer functions for rate gyro output. This computation resulated in a maximum value of approximately 100,000 in-lb. Both of these interface bending moment values measured at transonic speeds may be compared with the results of three wind-tunnel tests which were performed at the Ames Research Center. These tests established a maximum buffet bending moment of 280,000 in-lb at the spacecraft-GLV interface. While the variation between the values quoted in this report is large when the values are compared with each other, due to the varying degrees of conservatism in the analyses, these differences are insignificant when they are compared with the demonstrated capability of the joint. The larger value represents only 6.5 percent of the bending strength substantiated in the static test program.

TABLE 5-I.- GT-1 ACCELEROMETERS

Purpose	Accelerometer	Sensing level, g	Direction	Frequency response, cps	Spacecraft Station Location (a)
Adapter shell modes	QC22	± 100g	Radial	20 to 600	Z 28
	QC23	± 100g	Radial	20 to 600	Z 82
Pump package local modes	QC26	± 32g	Radial	20 to 600	Z 41
Launch-vehicle vibration effects	QC17	± 16g	Z	20 to 600	Z 16
	QA11	± 4g	Z	1 to 30	Z 100
Body bending	QD10	± 4g	X	1 to 30	Z 223
	QD11	± 4g	Y	1 to 30	Z 223
	QA09	± 2g	X	1 to 30	Z 100
	QA10	± 2g	Y	1 to 30	Z 100
Equipment environment	QC24	± 16g	Z	20 to 600	Z 70
	QB12	± 16g	Tangential	20 to 600	Z 113
	QB13	± 16g	Radial	20 to 2,000	Z 115
	QB14	± 16g	Z	20 to 2,000	Z 118

[a]For sensor locations, see reference 2.

[redacted]

TABLE 5-II.- COMPARISON OF ACTUAL AND PREDICTED ENGINE PERFORMANCE

Parameter	Predicted	Actual
Thrust (stage I), lb:		
5 sec from ignition	420,000	418,530
55 sec from ignition	450,000	443,270
140 sec from ignition	476,500	472,770
Specific impulse (stage I), sec:		
5 sec from ignition	258.8	257.3
55 sec from ignition		269.4
155 sec from ignition		281.1
Thrust (stage II), lb:		
57 sec from ignition	100,100	100,200
Specific impulse (stage II), sec:		
5 sec from ignition	310.2	309.9
57 sec from ignition		310.0
Burn time (stage I), sec:		
87FS1 (stage I engine ignition) to 87FS2 (BECO)	[a]157.7 [b](156.3)	157.46
Burn time (stage II), sec:		
91FS1 (stage II engine ignition) to 91FS2 (SECO)	[a]183.2 [b](182.4)	185.42
Burn time available at stage II	[a]2.4	3.27
Engine burnout, sec	[b](3.3)	

[a]Obtained from GLV integrating contractor.

[b]Obtained from Aerospace.

TABLE 5-III.- PROPELLANT WEIGHT DATA

	Predicted, lb		Actual, lb
Total propellant loaded			
Stage I fuel	90,254		90,264
Stage I oxidizer	171,215		171,225
Stage II fuel	21,940		21,940
Stage II oxidizer	38,936		38,936
Outage			
	Minimum	Maximum	
Stage I	0	1,321 (fuel)	926 (fuel)
Stage II	0	582 (fuel)	286 (oxidizer)
Residual propellant			
Stage I oxidizer	186		Not computed
Stage I fuel	447-1,768		1,271
Stage II oxidizer	87		1,093
Stage II fuel	209		465
Burning time margin (fuel)			
Stage II	---		3.27 seconds

[redacted]

TABLE 5-V.- INSTRUMENTATION DROPOUT RECORDED AT CAPE KENNEDY MISSILE TEST ANNEX TELEMETRY II

Time, sec after lift-off	Duration, sec	Cause
PCM Subsystem (244.3 mc)		
.558 to .708	.15	Lift-off
33.75 to 33.9	.15	Vehicle attitude
147.918 to 147.988	.07	Stage I flame attenuation prior to 87FS2 (BECO)
148.410 to 148.539	.129	Stage I flame attenuation prior to 87FS2
154.636 to LO + 155.198	.562	91FS1 (stage II engine ignition) flame attenuation
176.488 to 176.688	.20	Pitch down after Stage II guidance initiate
179.412 to 179.787	.375	Pitch down after Stage II guidance initiate
191.764 to 191.895	.131	Pitch down after Stage II guidance initiate
FM/FM subsystem (237.0 mc)		
154.728 to 155.078	.35	91FS1 flame attenuation

[redacted]

[redacted]

TABLE 5-IV.- PLANNED AND ACTUAL EVENT TIMES AND VEHICLE RATES

Event	Planned time after lift-off, sec	Actual time after lift-off, sec	Difference, sec	Planned rate, deg/sec	Actual rate, deg/deg	Difference, deg/sec
Pitch program start	9.9	10.0	0.1	1.25	1.17	-.07
Roll program end	20.48	20.5	0	1.25	1.17	-.07
Pitch program 1 start	23.04	23.0	0	-.6970	-.67	-.03
Pitch program 1 end	88.32	88.1	0.2	-.6970	-.67	-.03
Pitch program 2 start	88.32	88.1	0.2	-.47298	-.47	0
Pitch program 2 end	119.04	118.8	0.2	-.47298	-.47	0
Pitch program 3 start	119.04	118.8	0.2	-.26677	-.27	0
Pitch program 3 end	162.56	162.0	0.6	-.26677	-.27	0

[redacted]

TABLE 5-VI.- INSTRUMENTATION DROPOUT RECORDED AT GRAND BAHAMA ISLAND

Time, sec after lift-off	Duration, sec	Cause
PCM Subsystem (244.3 mc)		
127.408 to 127.506	.098	Unknown
154.673 to 155.150	.472	91FS1 (Stage II engine ignition) flame attenuation
307.263 to 307.328	.065	Unknown
309.408 to 309.488	.08	Unknown
349.970 to 350.423	.453	Unknown
368.558 to 368.923	.365	Unknown
374.270 to 374.320	.050	Unknown
398.378 to 398.478	.10	Unknown
399.384 to 399.478	.094	Unknown
417.458 to 417.532	.074	Unknown
FM/FM subsystem (237.0 mc)		
154.668 to 155.085	.417	91FS1 flame attenuation

[redacted]

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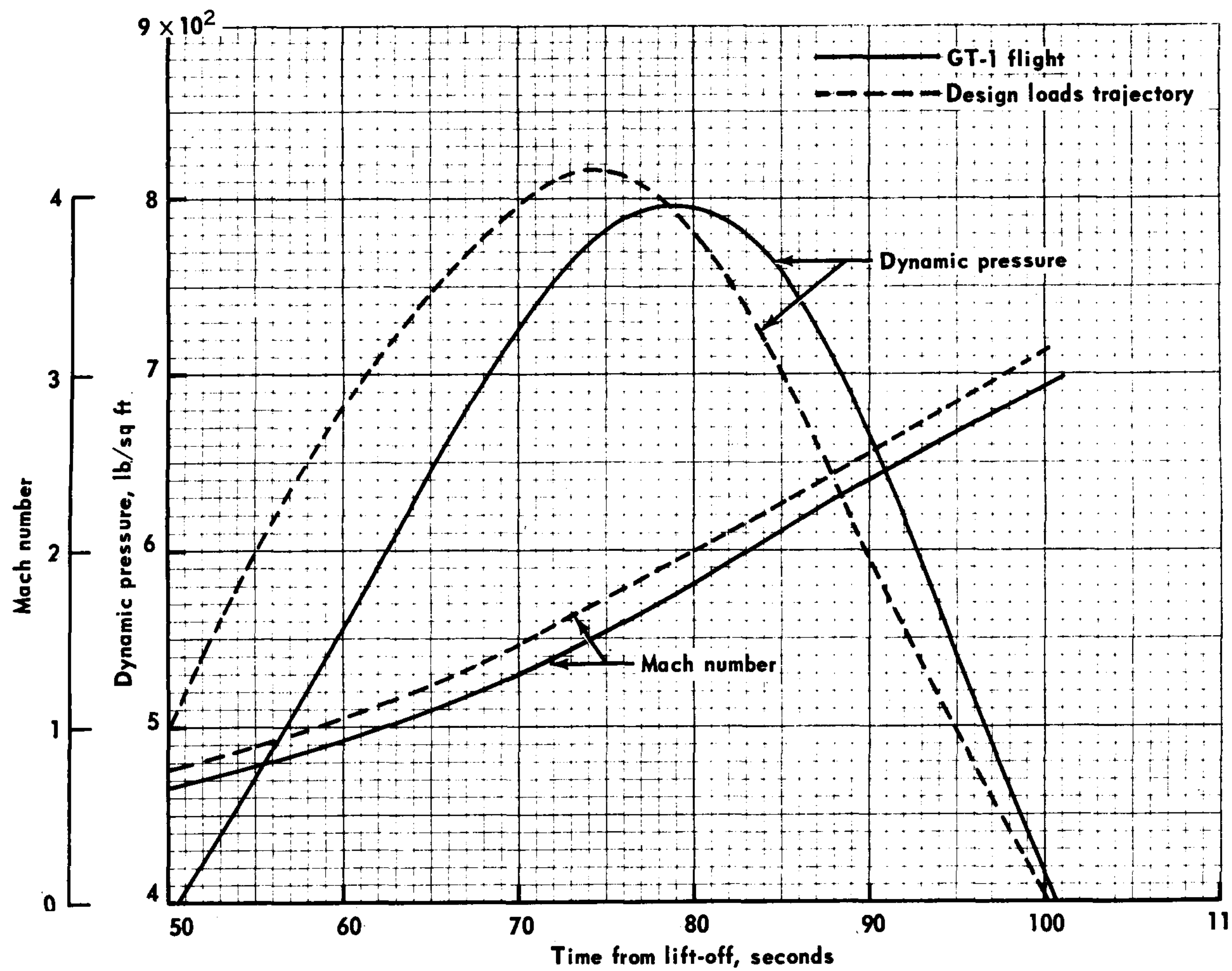

FIGURE 5-1 DYNAMIC PRESSURE AND MACH NUMBER TIME HISTORIES OF GT-1 FLIGHT AND DESIGN LOADS TRAJECTORY

[redacted]

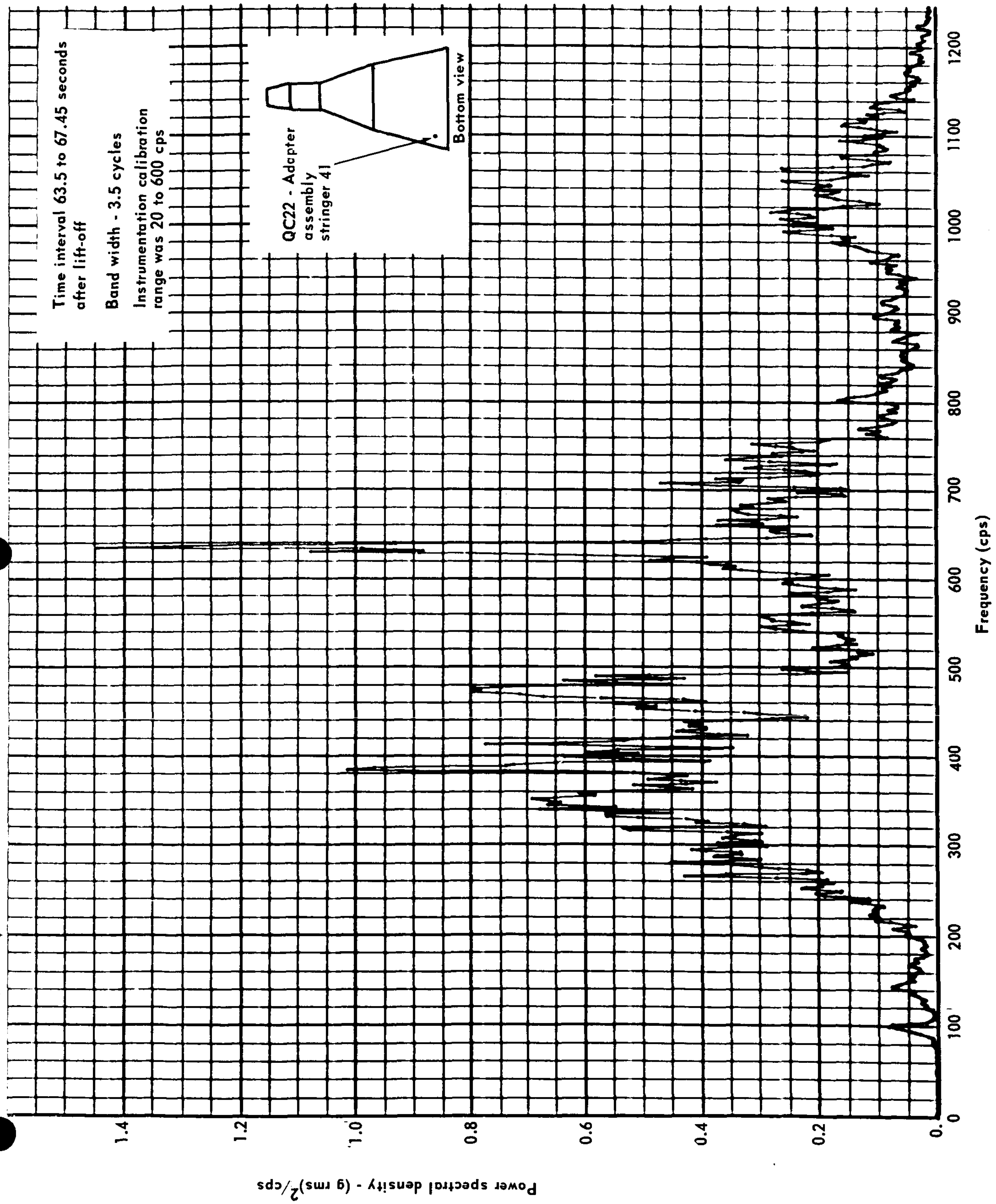

FIGURE 5-2 POWER SPECTRAL DENSITY, STATION Z28 RING, SENSOR QC22

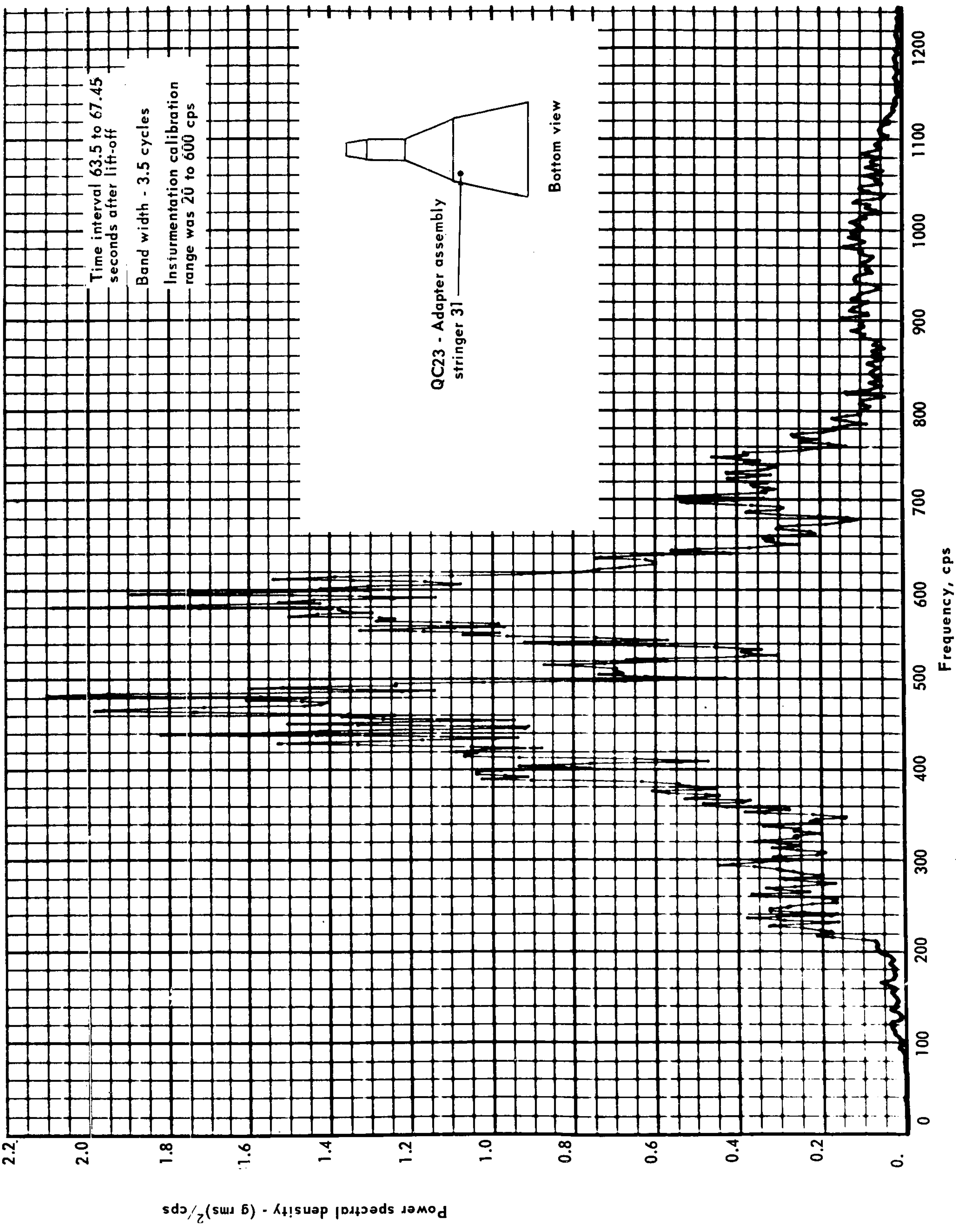

FIGURE 5-3 POWER SPECTRAL DENSITY, STATION Z82 RING, SENSOR QC23

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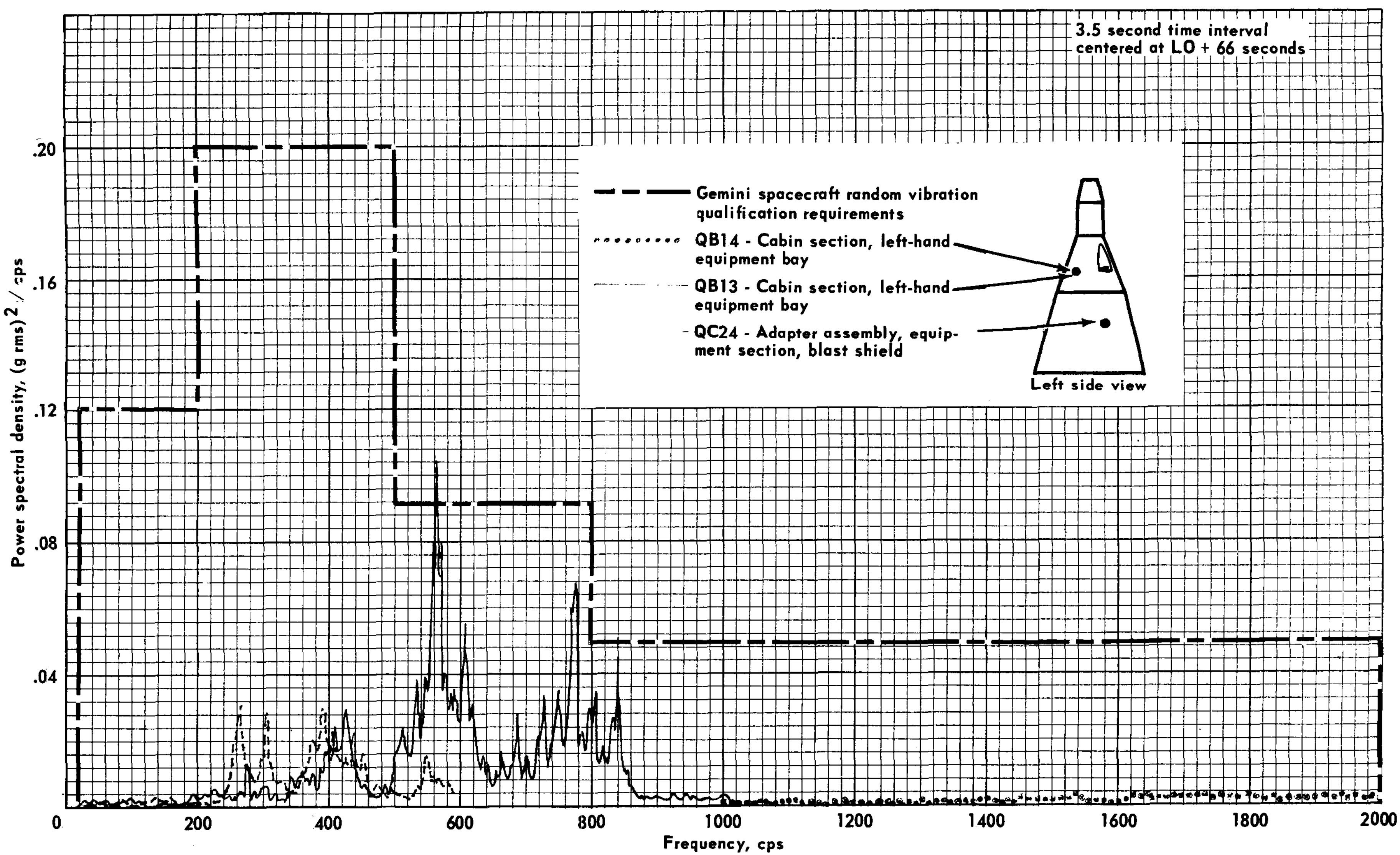

FIGURE 5-4 EQUIPMENT VIBRATION REFERENCE ACCELEROMETERS

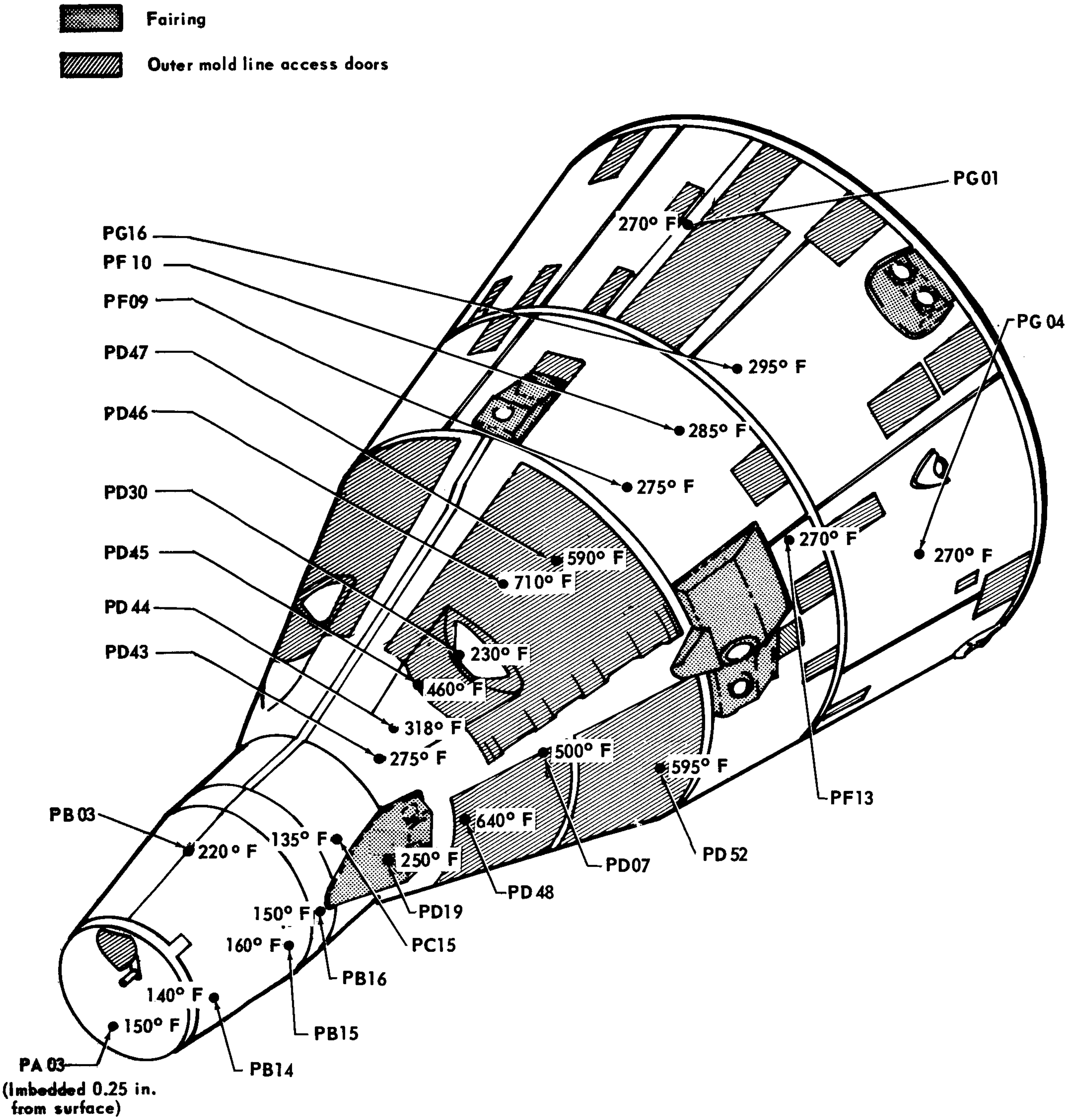

FIGURE 5-5 DISTRIBUTION OF PEAK MEASURED TEMPERATURES ON UPPER HALF OF SPACECRAFT

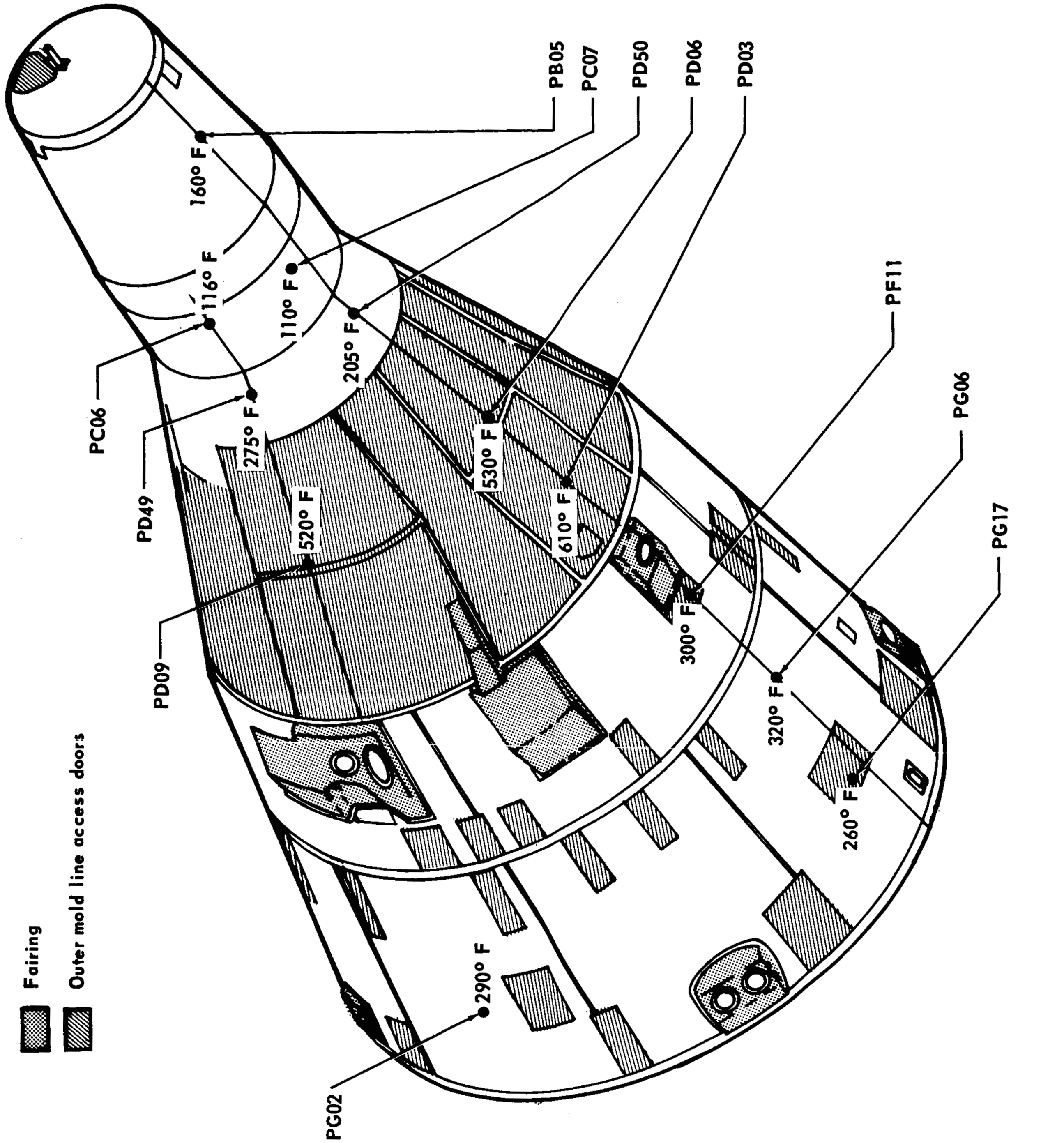

FIGURE 5-6 DISTRIBUTION OF PEAK
MEASURED TEMPERATURES ON LOWER HALF OF SPACECRAFT

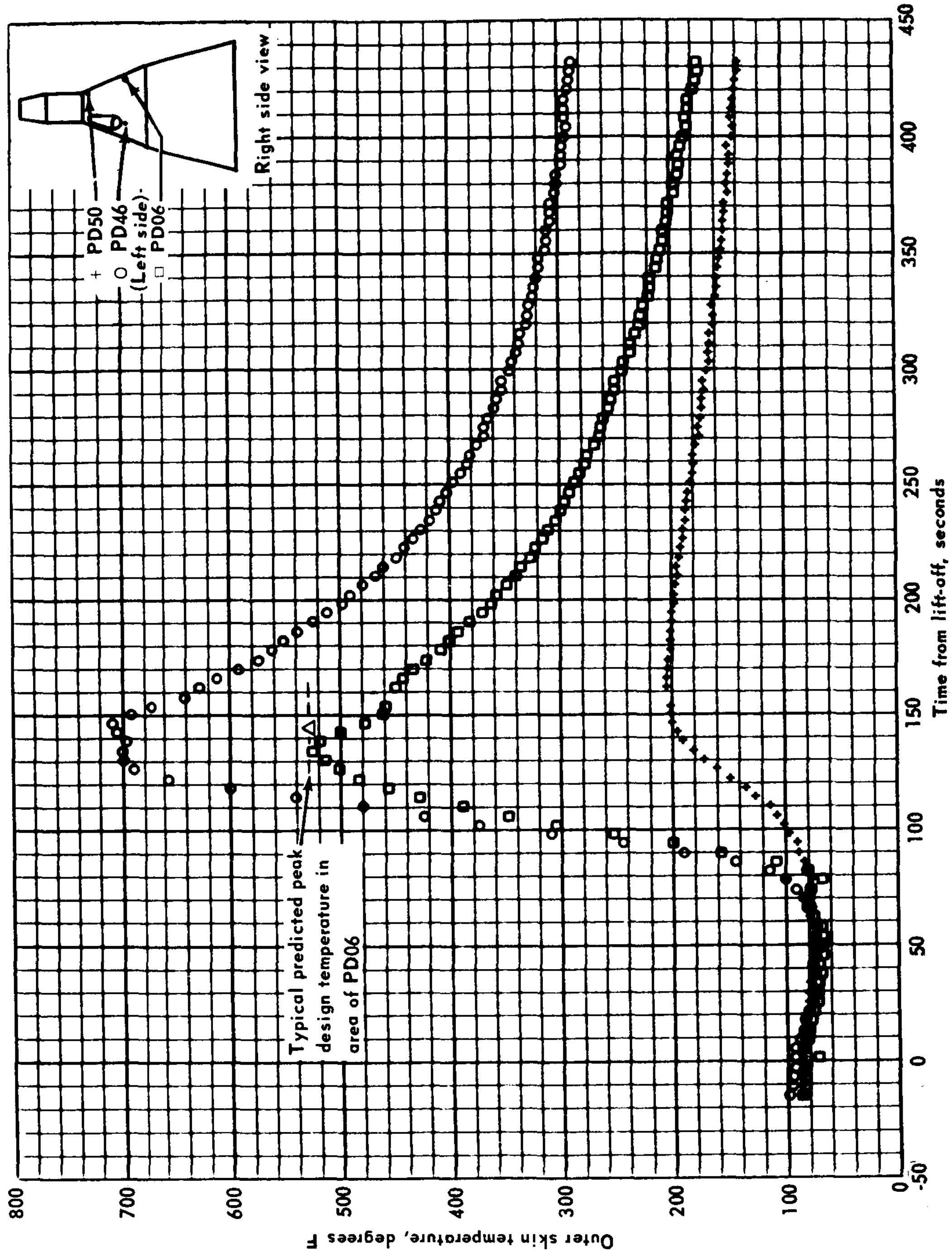

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FIGURE 5-7 GT-1 CABIN SECTION TEMPERATURES

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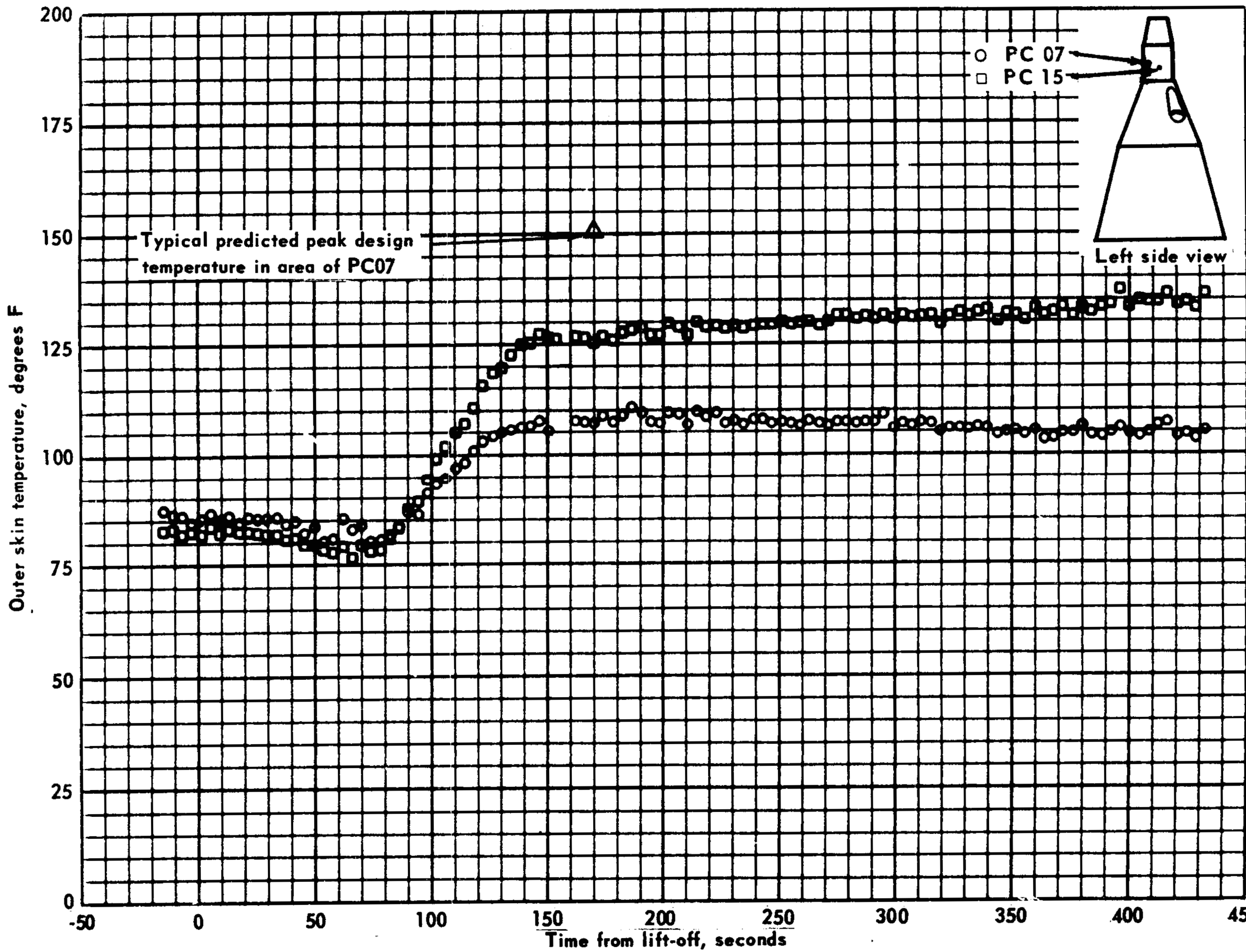

FIGURE 5-8 GT-1 RCS SECTION TEMPERATURES

NASA-S-64-3141

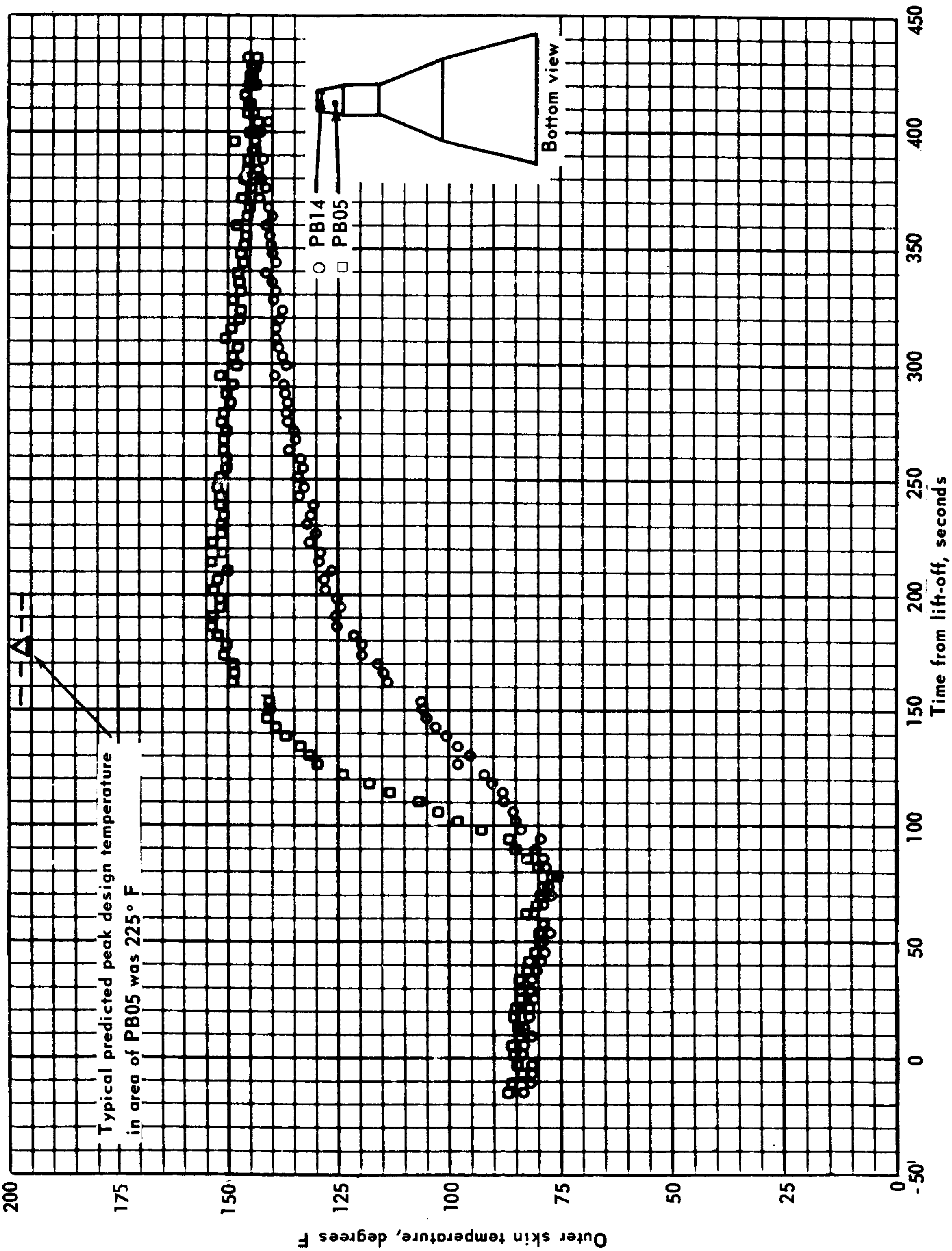

FIGURE 5-9 GT-1 R AND R SECTION TEMPERATURES

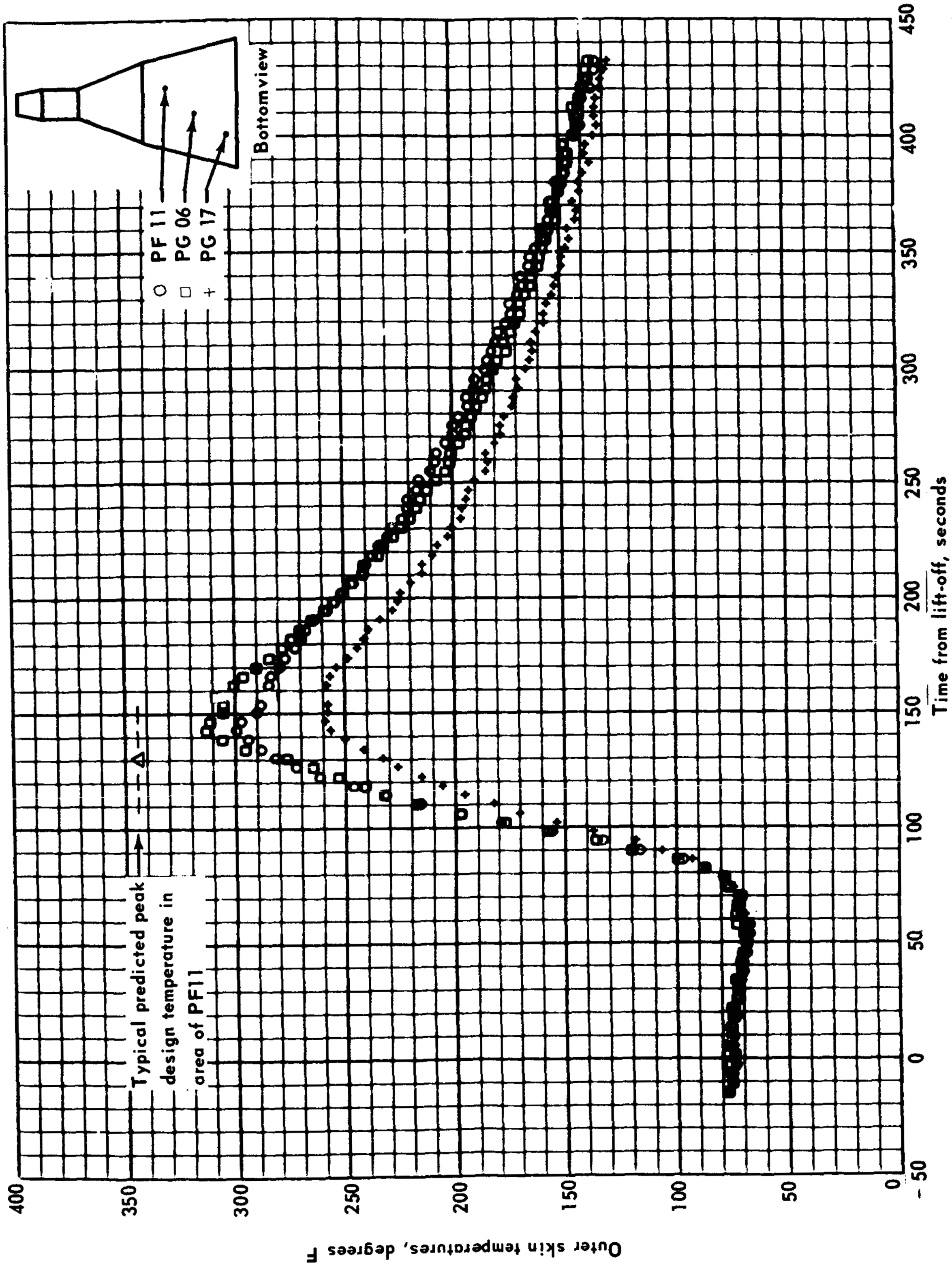

FIGURE 5-10 GT-1 ADAPTER MODULE TEMPERATURES

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NASA-S-64-3095

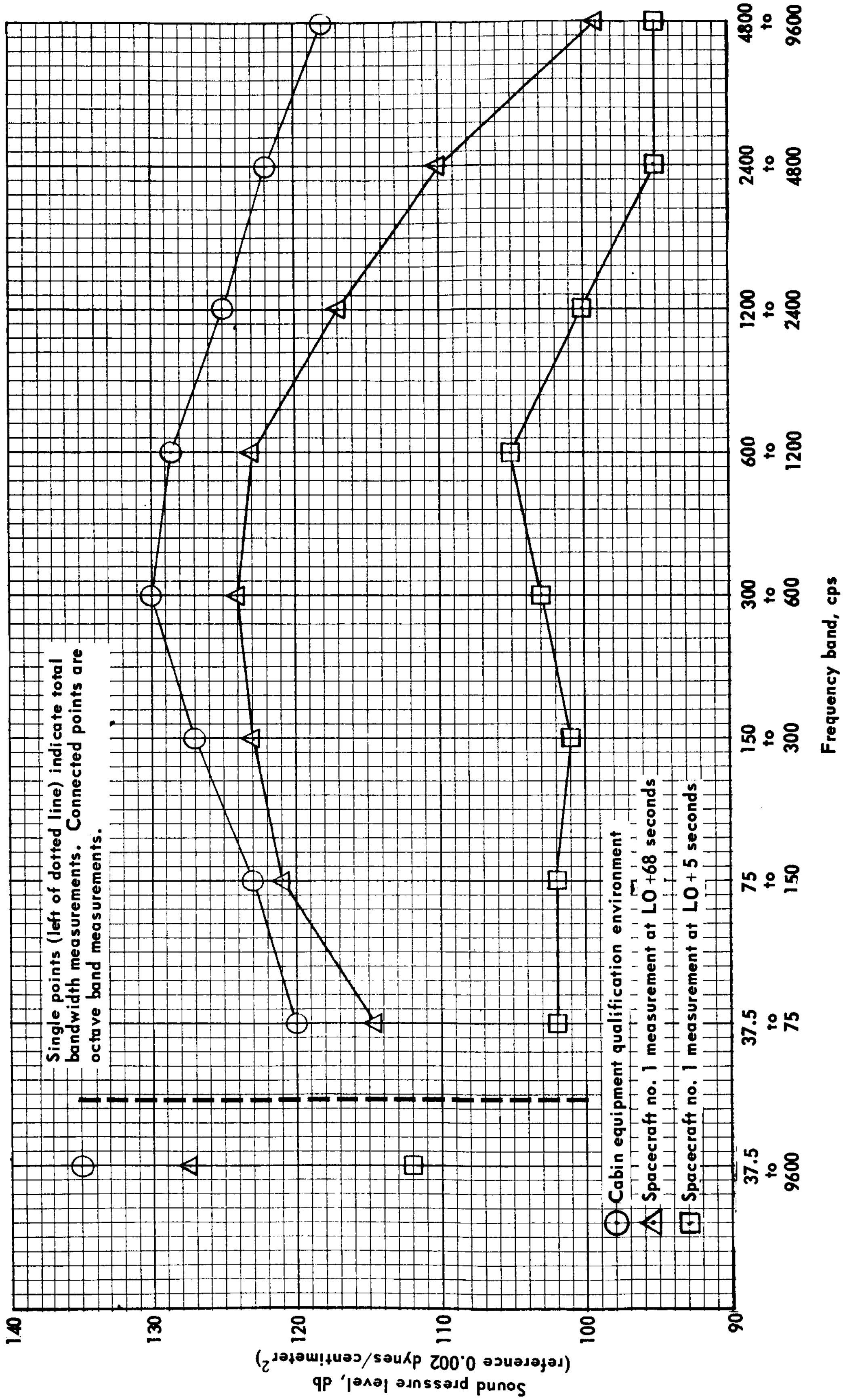

FIGURE 5-11 ACOUSTIC ENVIRONMENT - CABIN AREA

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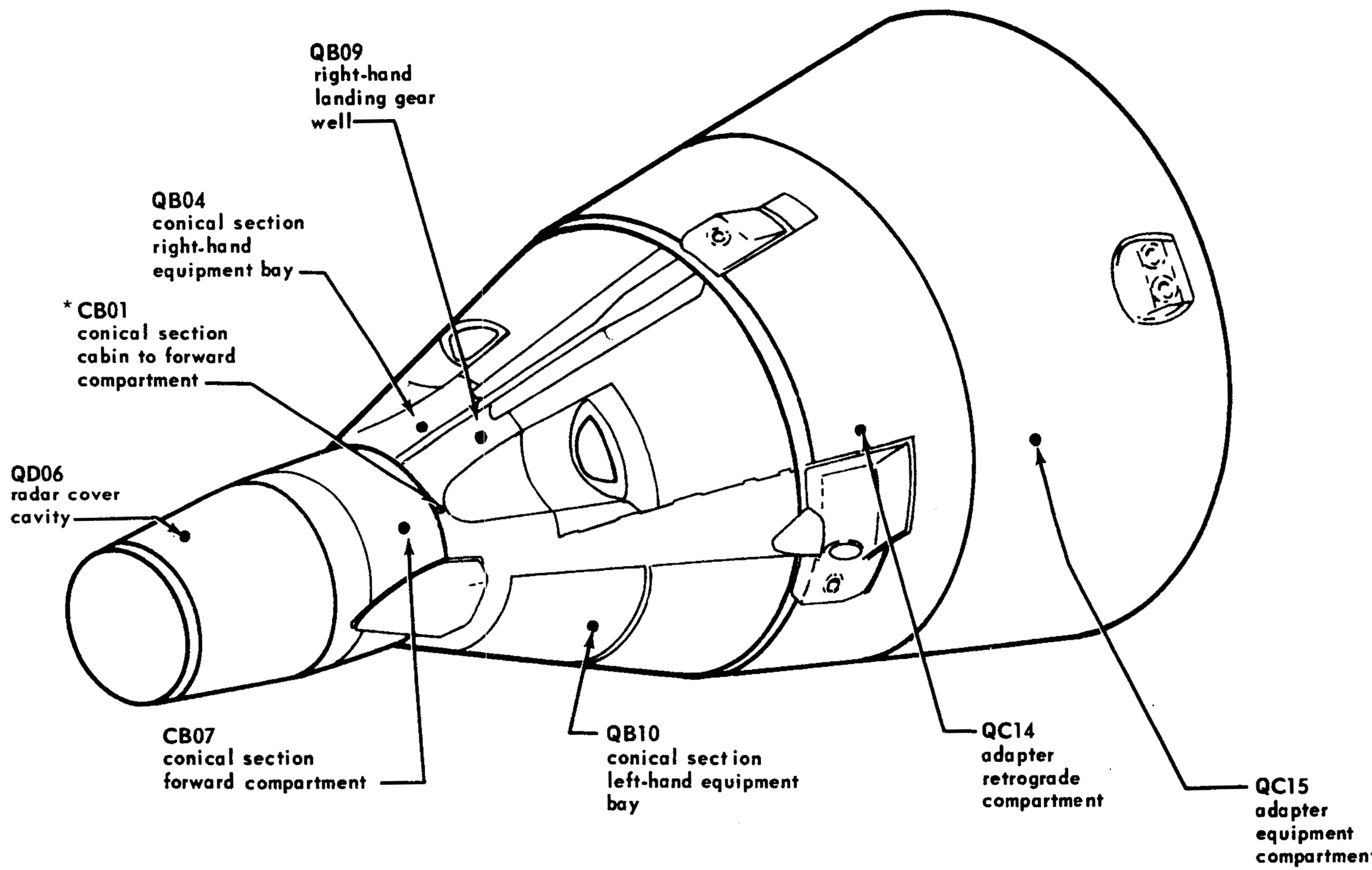


* Measures pressure differential between cabin and forward compartment; all other sensors measure absolute pressure

FIGURE 5–12 SPACECRAFT PRESSURE TRANSDUCER LOCATIONS

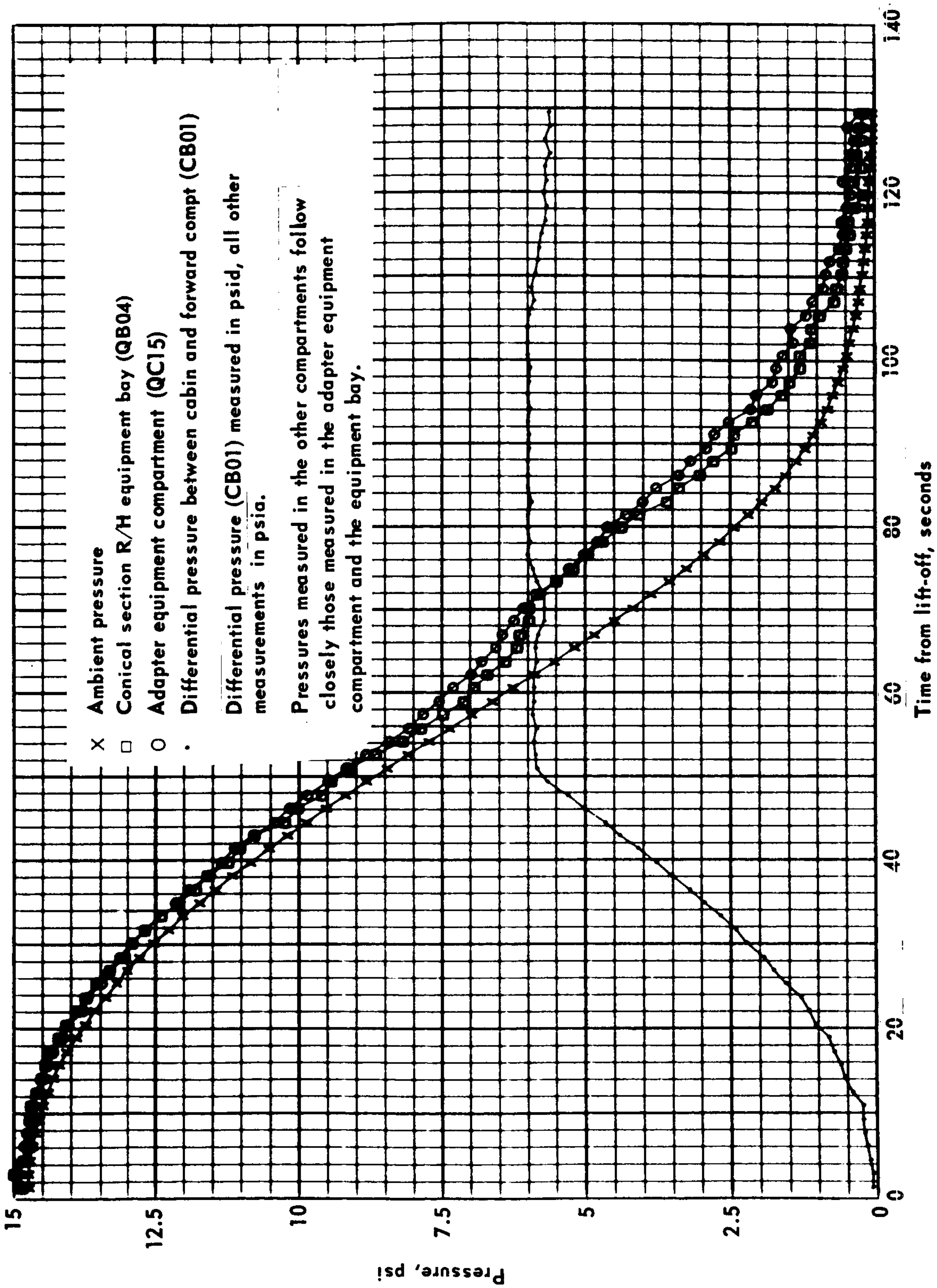
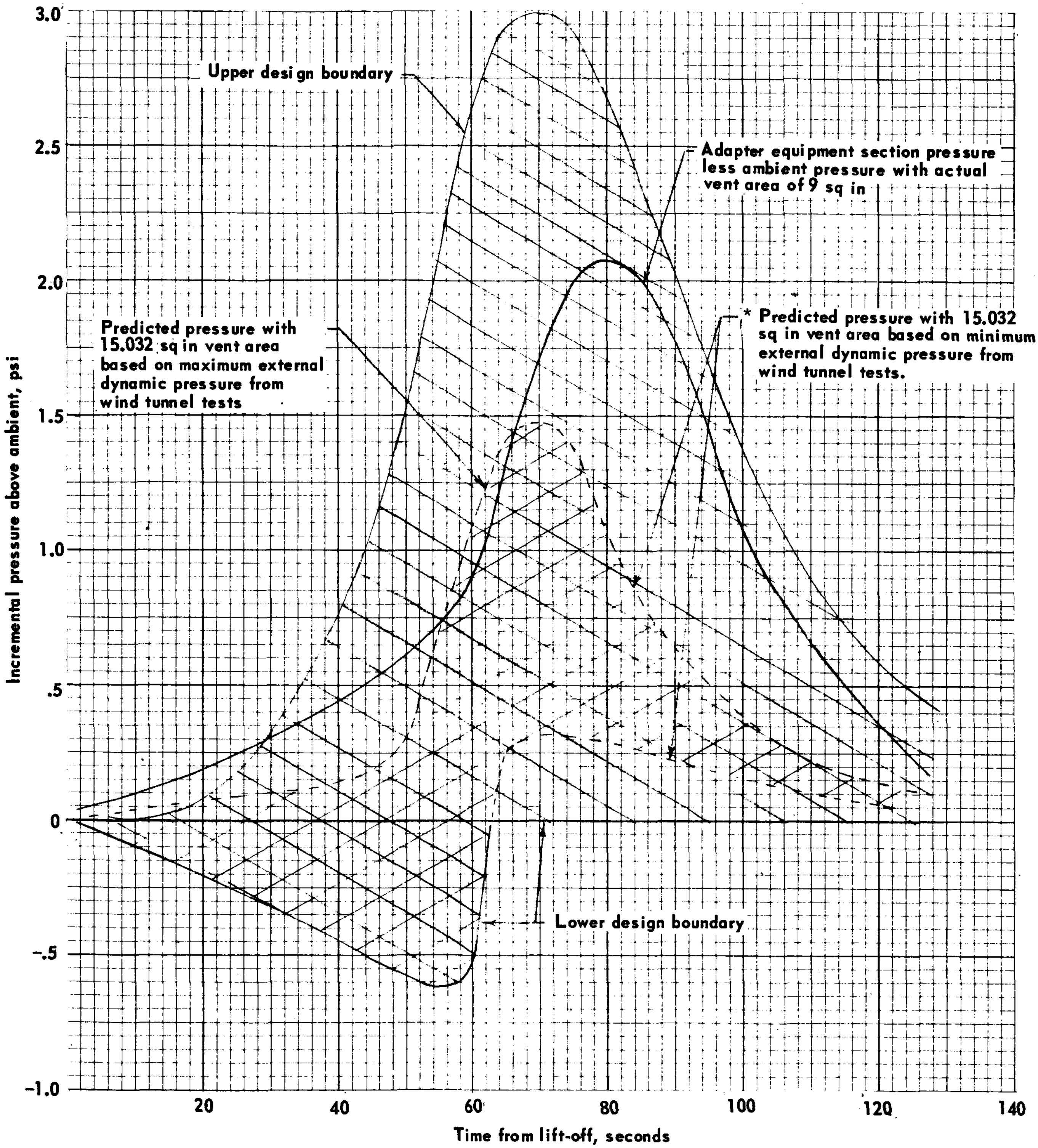


FIGURE 5 - 13 TYPICAL PRESSURE MEASUREMENTS DURING LAUNCH PHASE



Notes:
1. Ref. 6
*2. Based on design venting configuration of 15.023 sq. in.

FIGURE 5-14 ADAPTER INCREMENTAL PRESSURE ABOVE AMBIENT

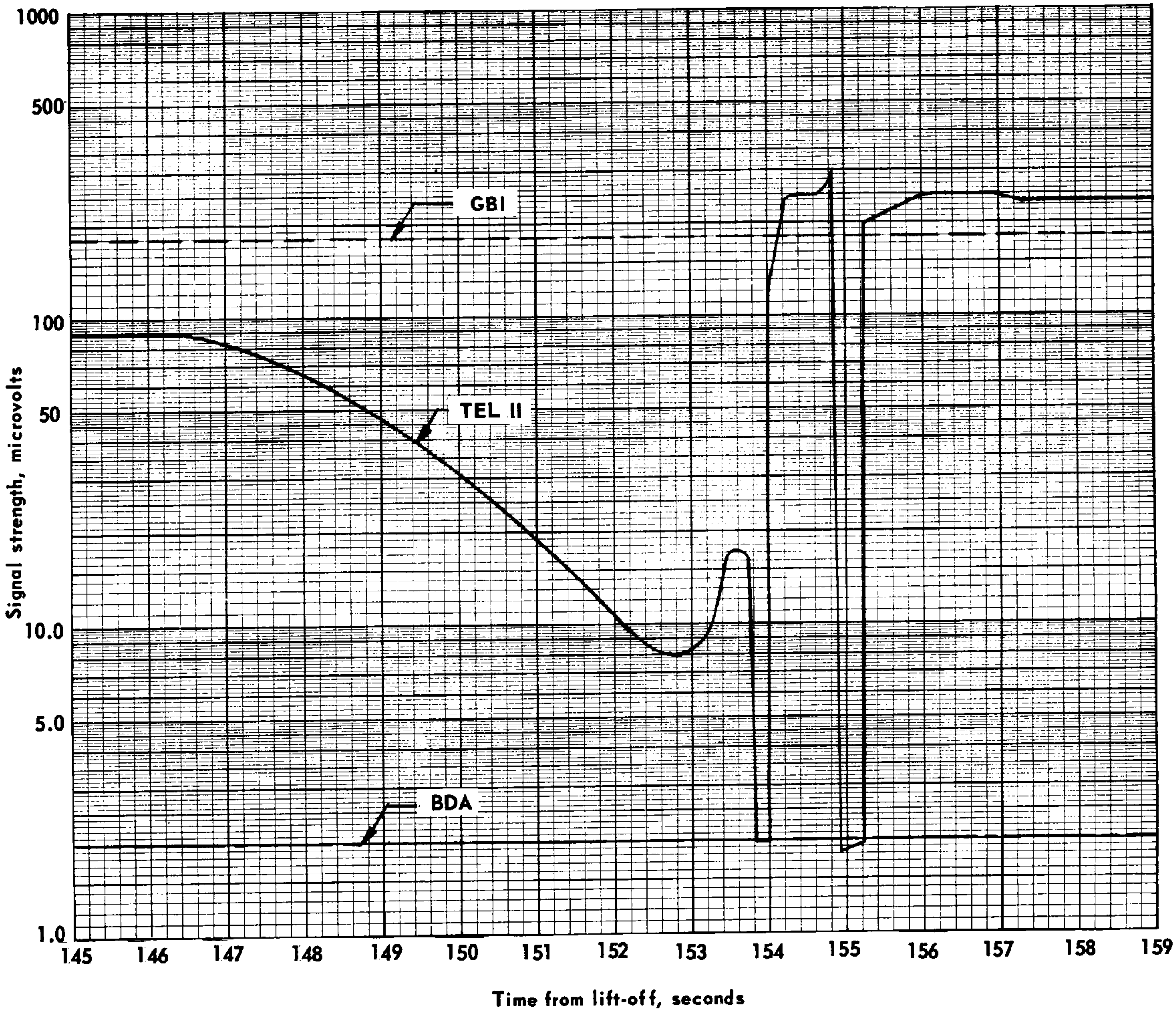


FIGURE 5-15

COMPARISON OF TELEMETRY SIGNAL STRENGTH ATTENUATION AT BECO OF 243 MC TELEMETRY LINK: CAPE KENNEDY (TEL II), BERMUDA (BDA), GRAND BAHAMA ISLAND (GBI)

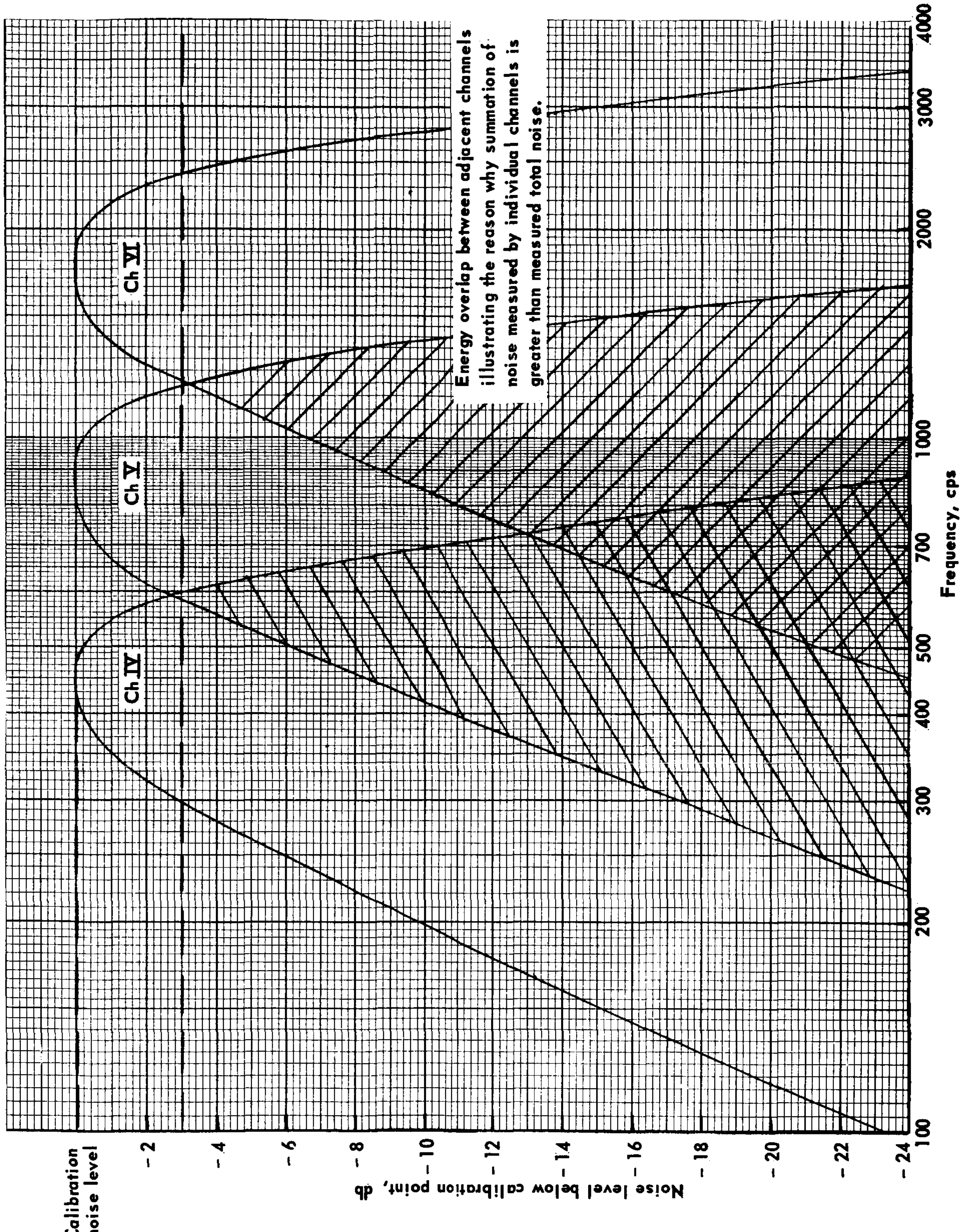


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FIGURE 5-16 TYPICAL ROLLOFF CHARACTERISTICS OF ACOUSTIC NOISE SPECTRUM ANALYZER

NASA-S-64-3096

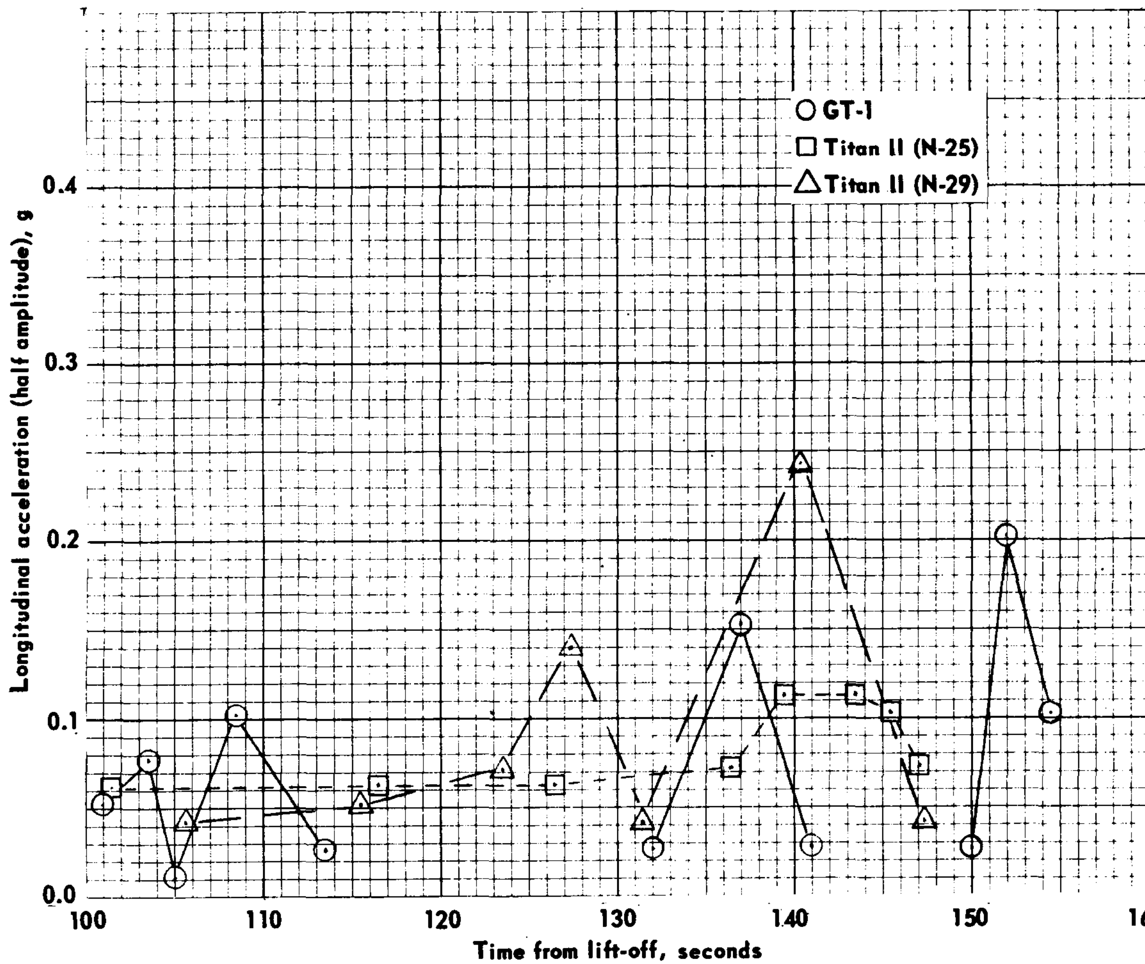

FIGURE 5-17 LONGITUDINAL OSCILLATION ON GT-1 AS COMPARED TO PREVIOUS RESEARCH AND DEVELOPMENT FLIGHTS OF TITAN II MISSILES WITH MODIFICATIONS TO SUPPRESS LOW-LEVEL LONGITUDINAL ACCELERATIONS.

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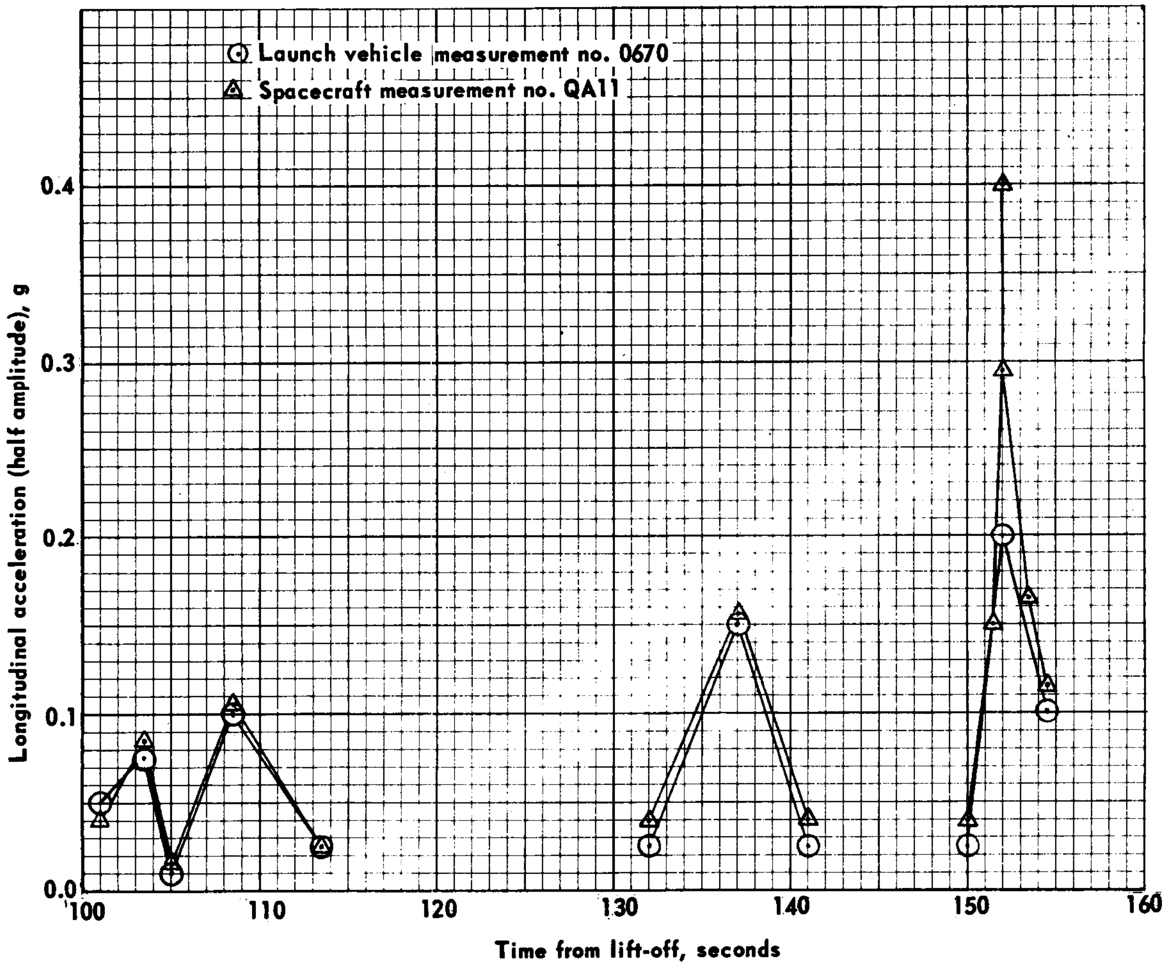

FIGURE 5-18 LONGITUDINAL OSCILLATION

[redacted]

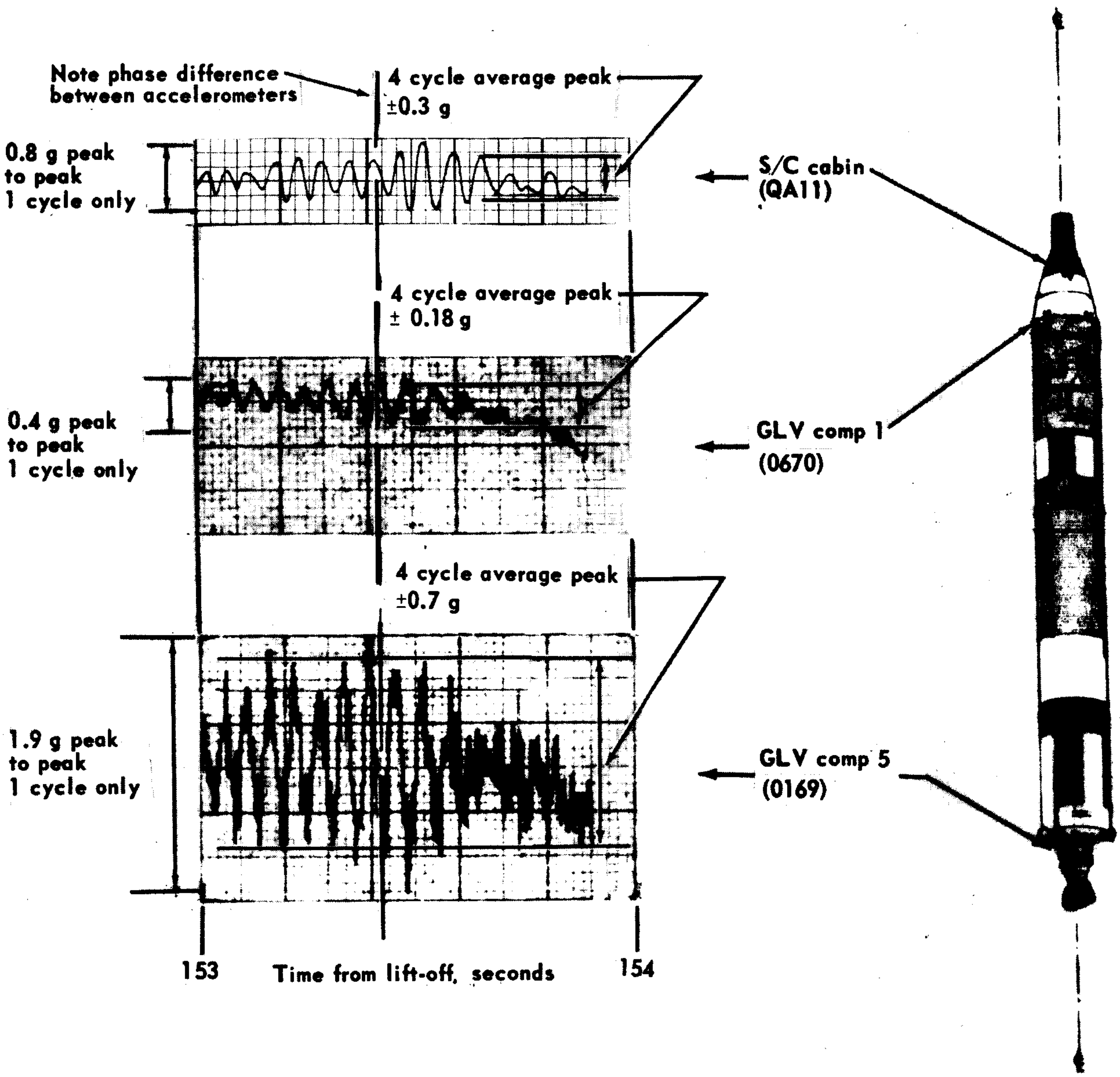

FIGURE 5-19 LOW FREQUENCY LONGITUDINAL OSCILLATION MEASUREMENTS

[redacted]

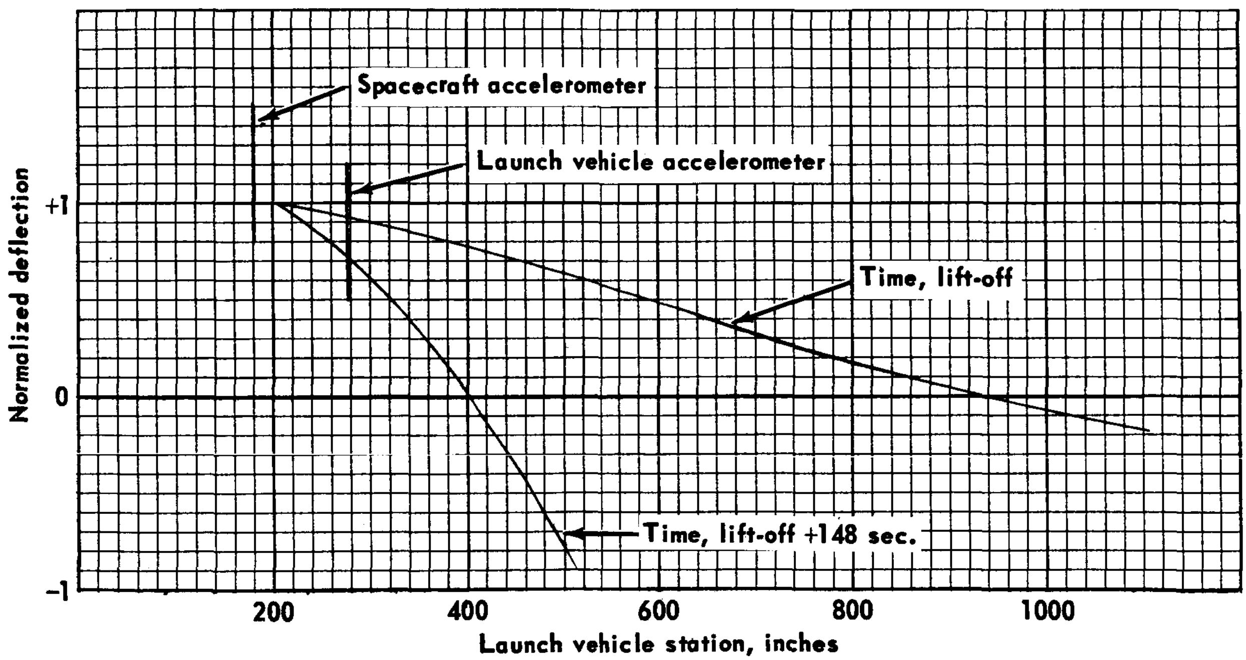

FIGURE 5-20 LONGITUDINAL MODE SHAPE

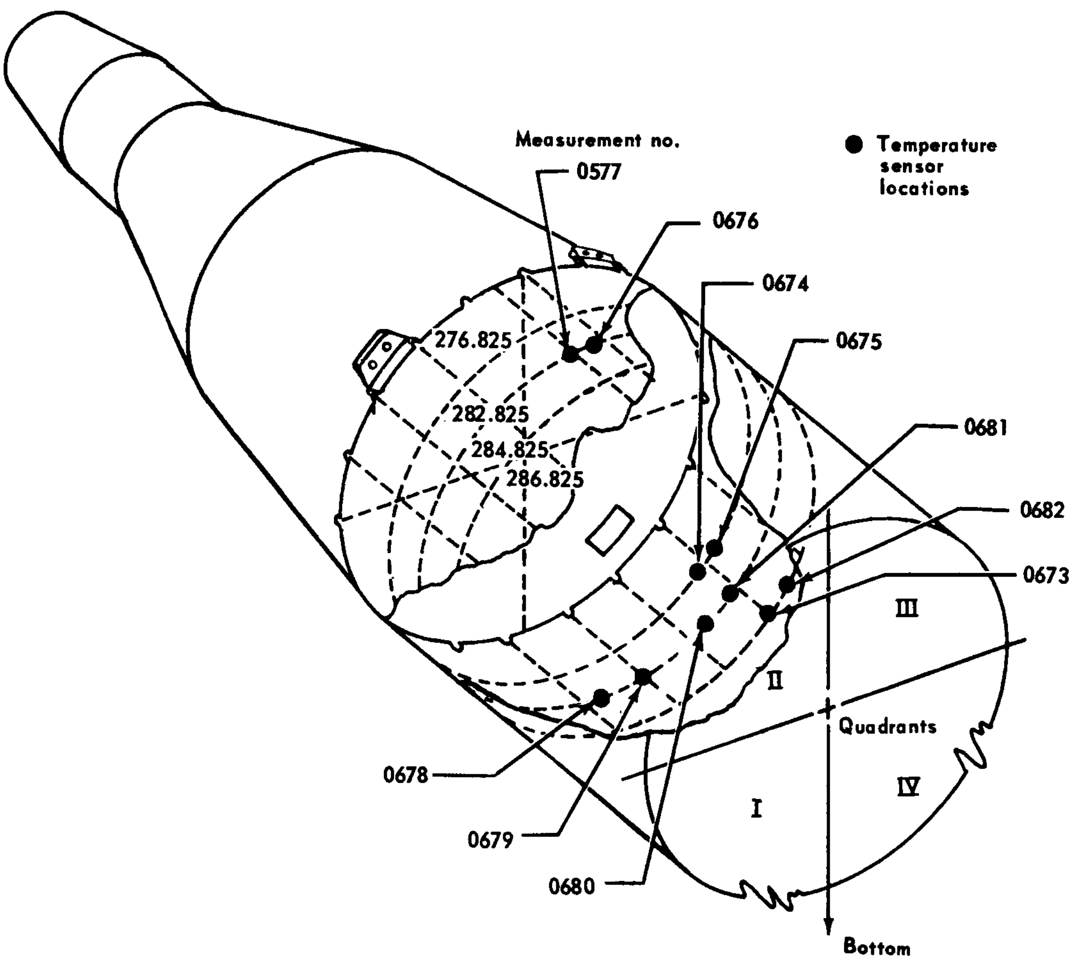

FIGURE 5-21 INTERFACE THERMOCOUPLE LOCATIONS

NASA-S-64-3128

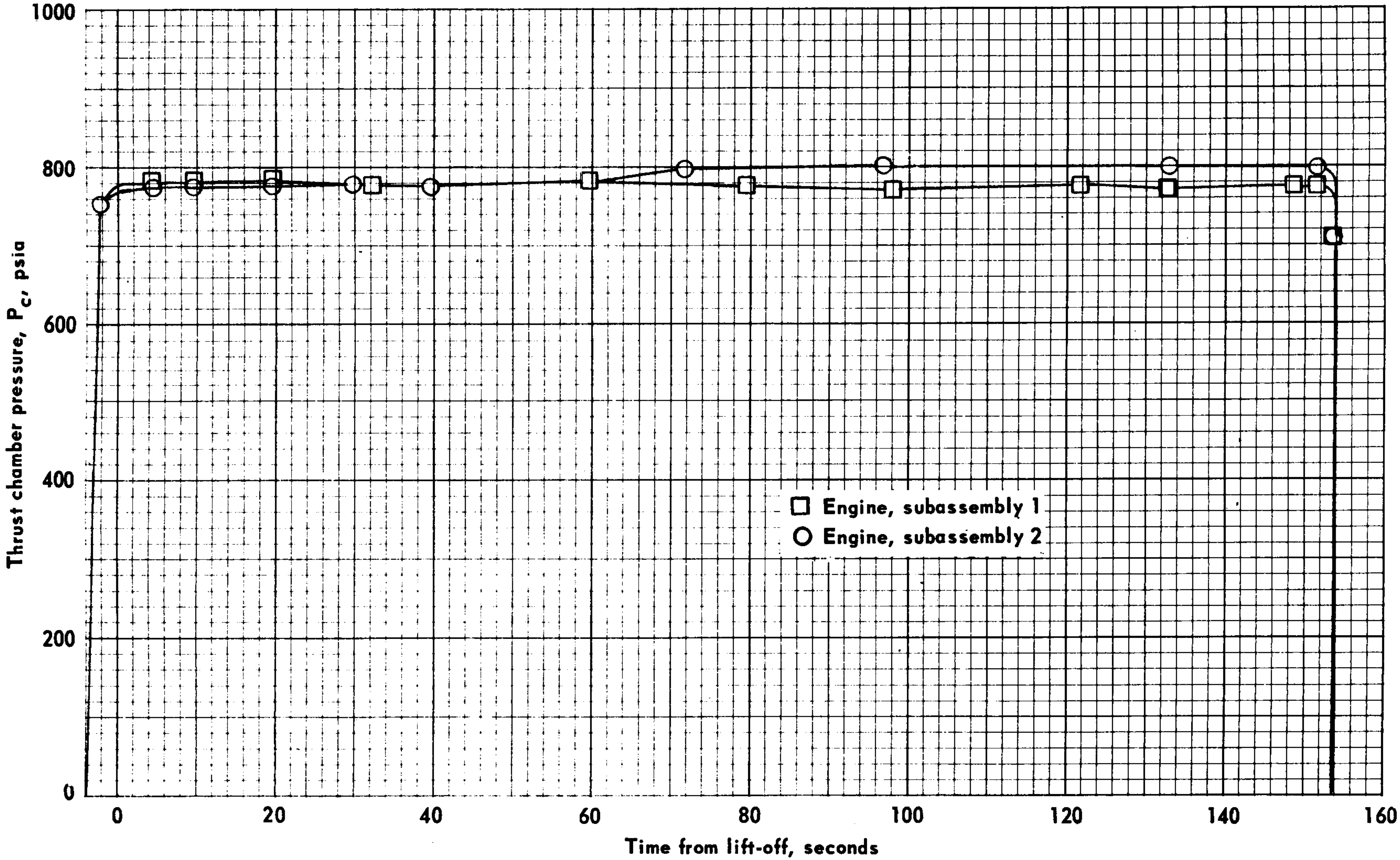

FIGURE 5-22 STAGE I THRUST CHAMBER PRESSURE

[redacted]

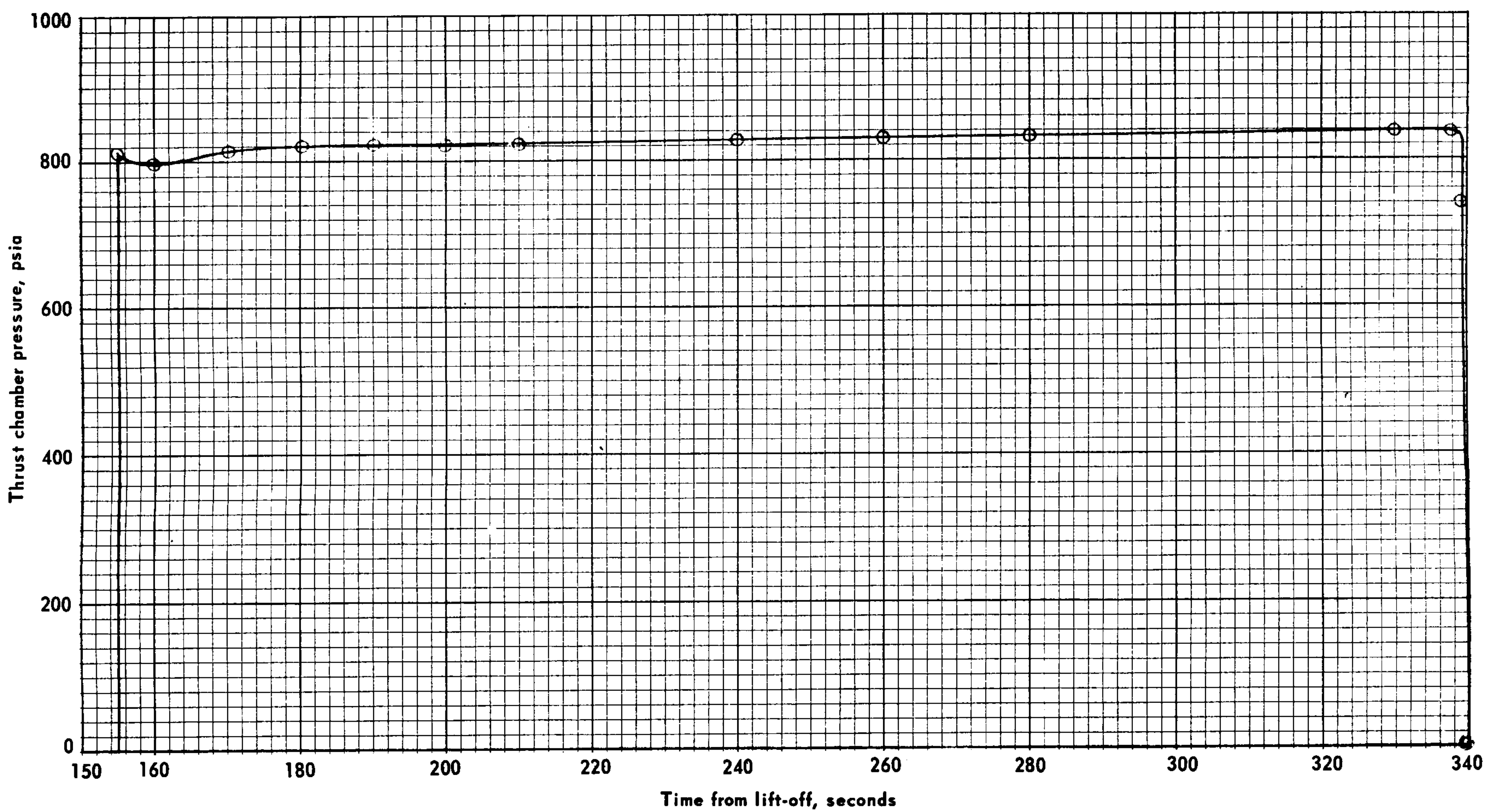

FIGURE 5-23 STAGE II THRUST CHAMBER PRESSURE

[redacted]

[redacted]

NASA-S-64-3291

Note that the time scales referenced to both engine ignition and lift-off have a dual scale factor (.25 seconds / division and .5 seconds / division). Discontinuities at points of scale factor change were caused by plotting equipment reaction to the scale factor change and are not true data discontinuities.

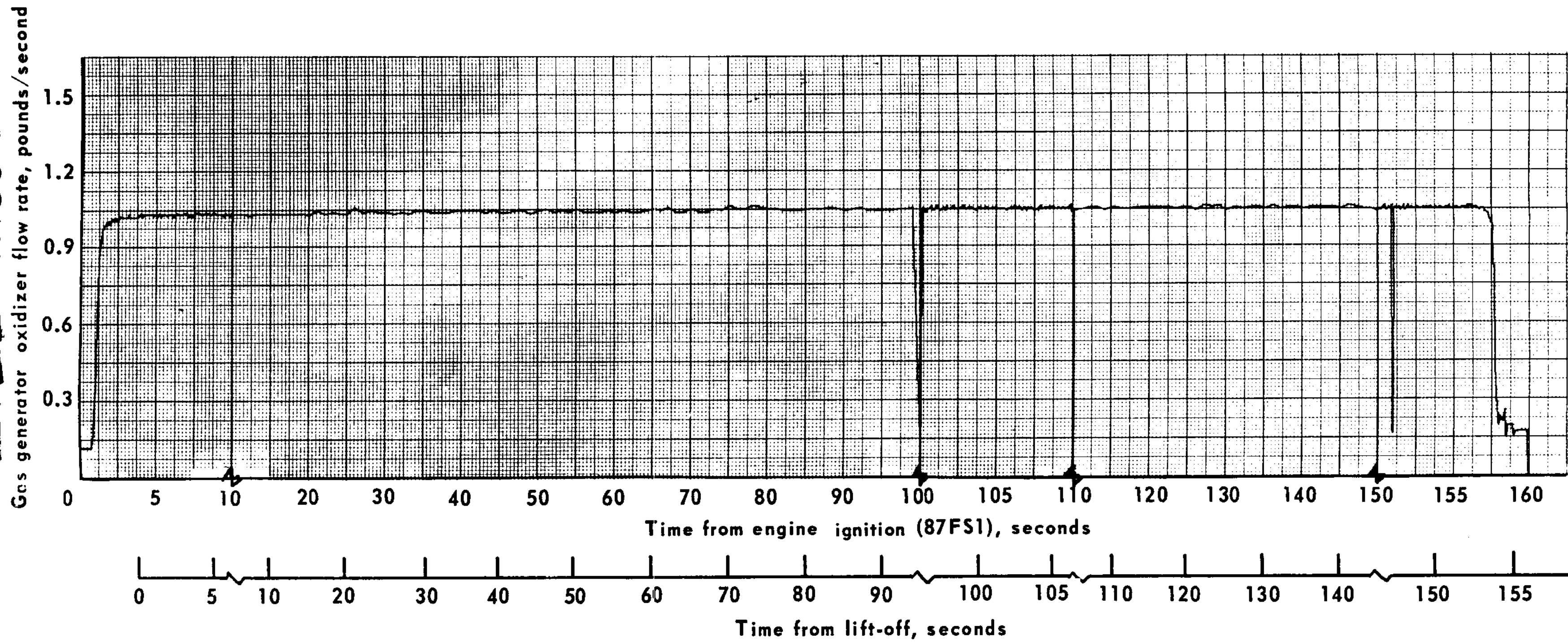

FIGURE 5-24 STAGE I, S/A-2 GAS GENERATOR OXIDIZER FLOW RATE

[redacted]

[redacted]

NASA-S-64-3290

Note that the time scales referenced to both engine ignition and lift-off have dual scale factor (.25 seconds / division and .5 seconds / division). Discontinuities at points of scale factor change were cased by plotting equipment reaction to the scale factor change and are not true data discontinuities.

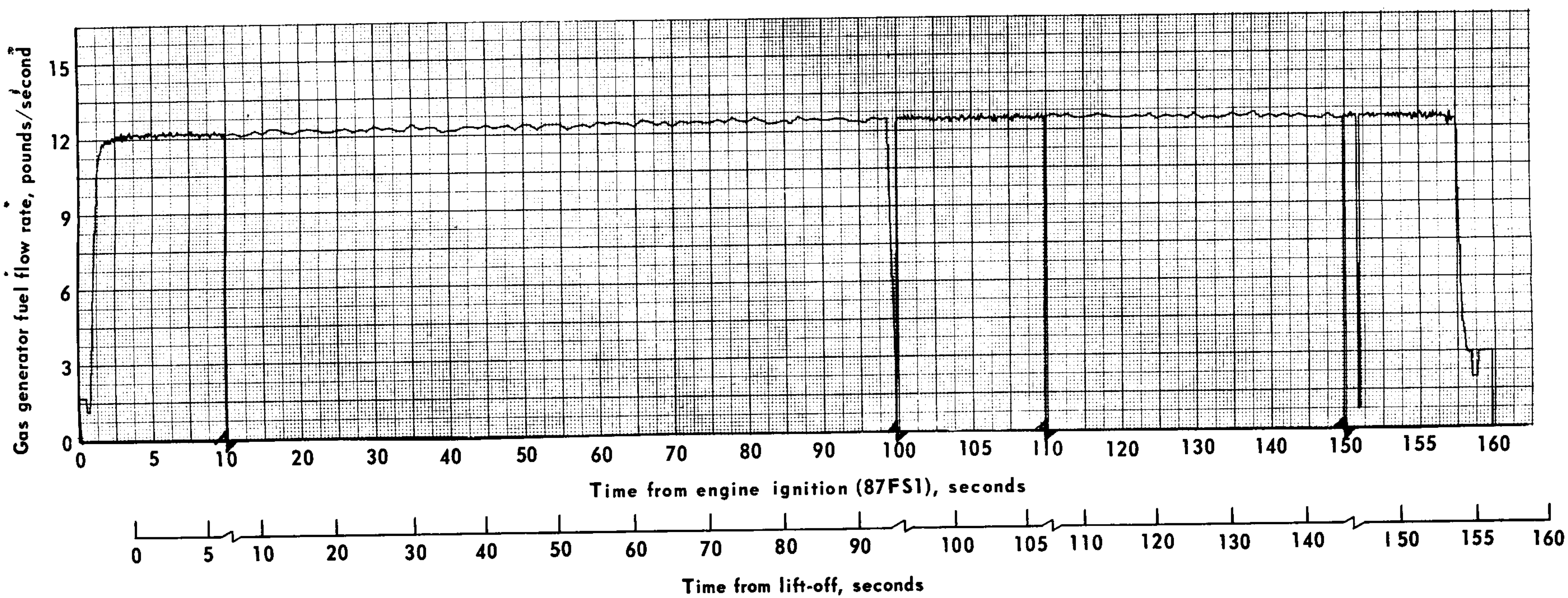

FIGURE 5-25 STAGE I, S/A-2 GAS GENERATOR FUEL FLOW RATE

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Note that the time scales referenced to both engine ignition and lift-off have a dual scale factor (.25 seconds / division and .5 seconds / division). Discontinuities at points of scale factor change were caused by plotting equipment reaction to the scale factor change and are not true data discontinuities.

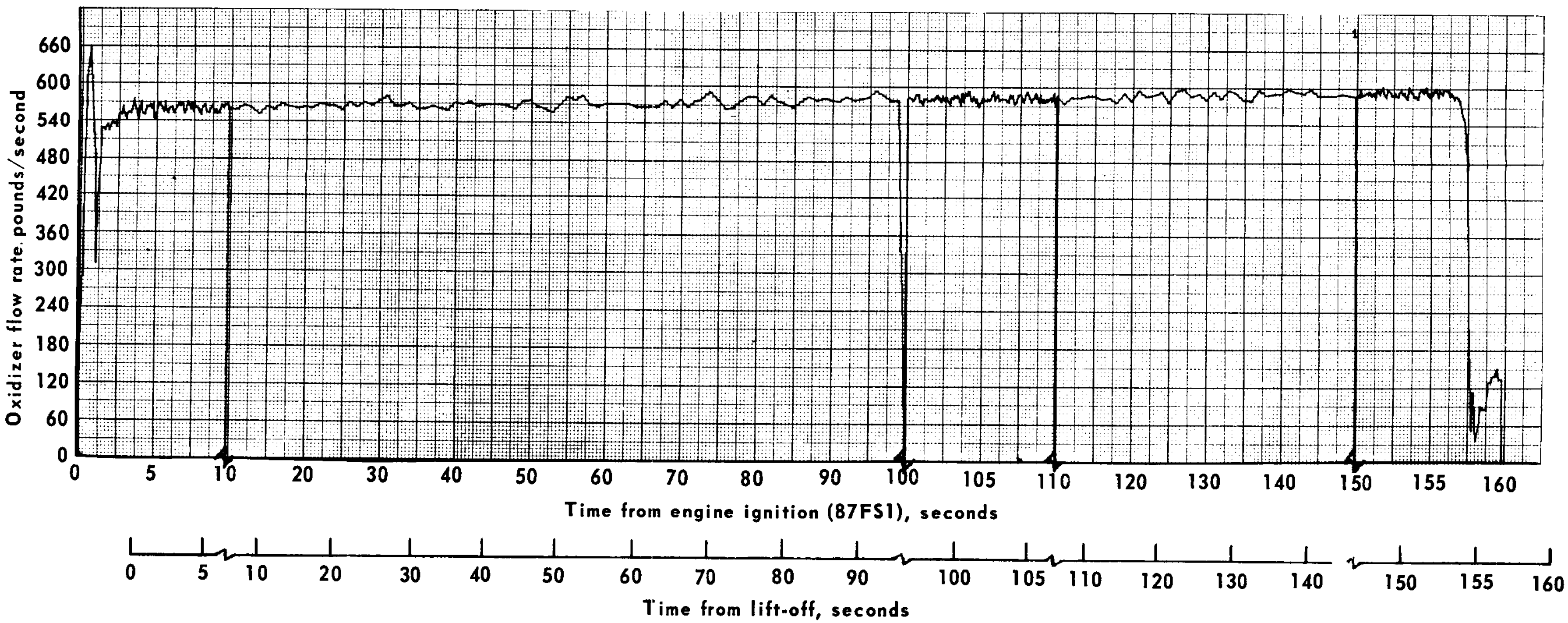


FIGURE 5-26 STAGE I, S/A-2 THRUST CHAMBER OXIDIZER FLOW RATE

NASA-S-64-3293

[redacted]

Note that the time scales referenced to both engine ignition and lift-off have a dual scale factor (.25 seconds / division and .5 seconds / division). Discontinuities at points of scale factor change were caused by plotting equipment reaction to the scale factor change and are not true data discontinuities.

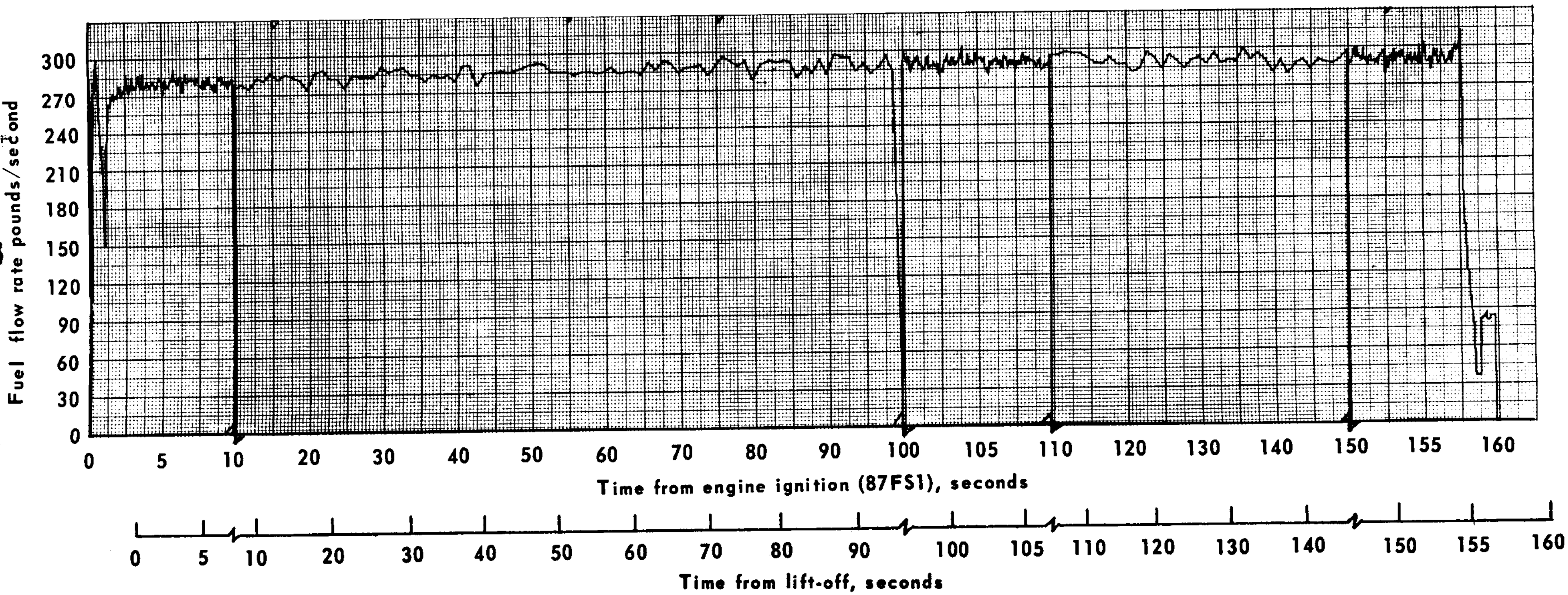


FIGURE 5-27 STAGE I, S/A-2 THRUST CHAMBER FUEL FLOW RATE

[redacted]

NASA-S-64-3292

Note that the time scales referenced to both engine ignition and lift-off have a dual scale factor (.25 seconds / division and .5 seconds / division). Discontinuities at points of scale factor change were caused by plotting equipment reaction to the scale factor change and are not true data discontinuites.

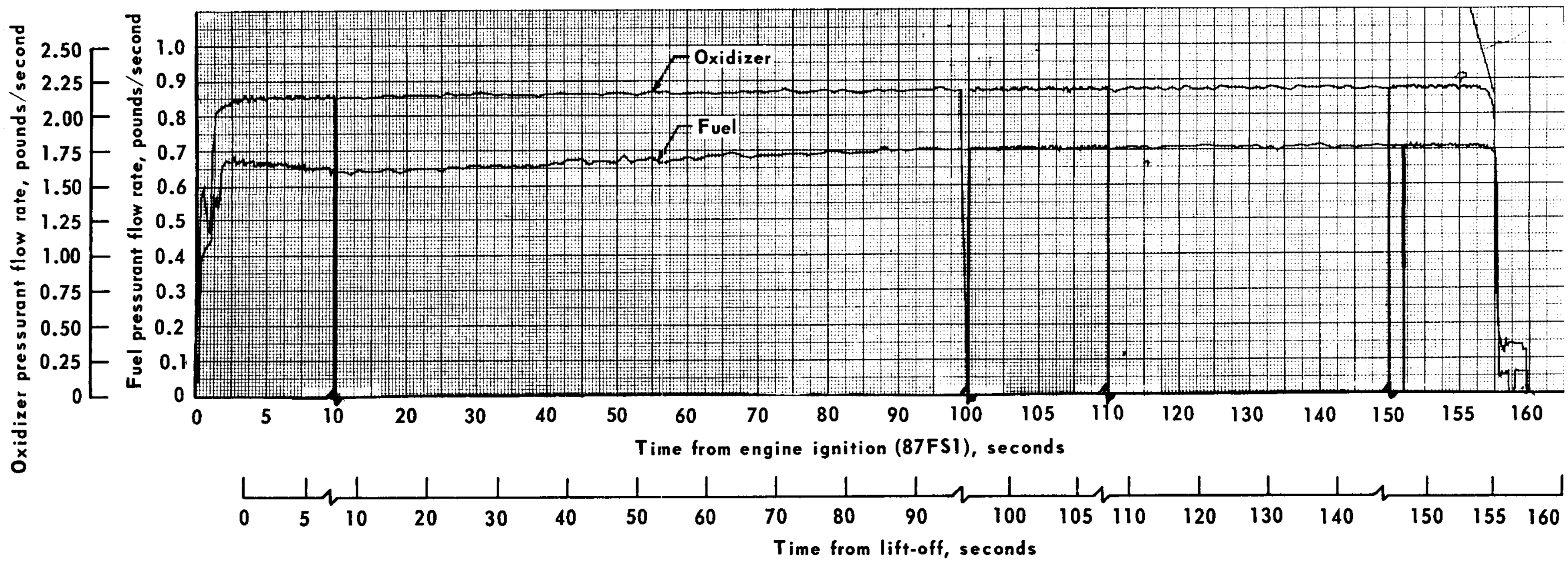


FIGURE 5-28 STAGE I PRESSURANT FLOW RATES

[redacted]

NASA-S-64-3130

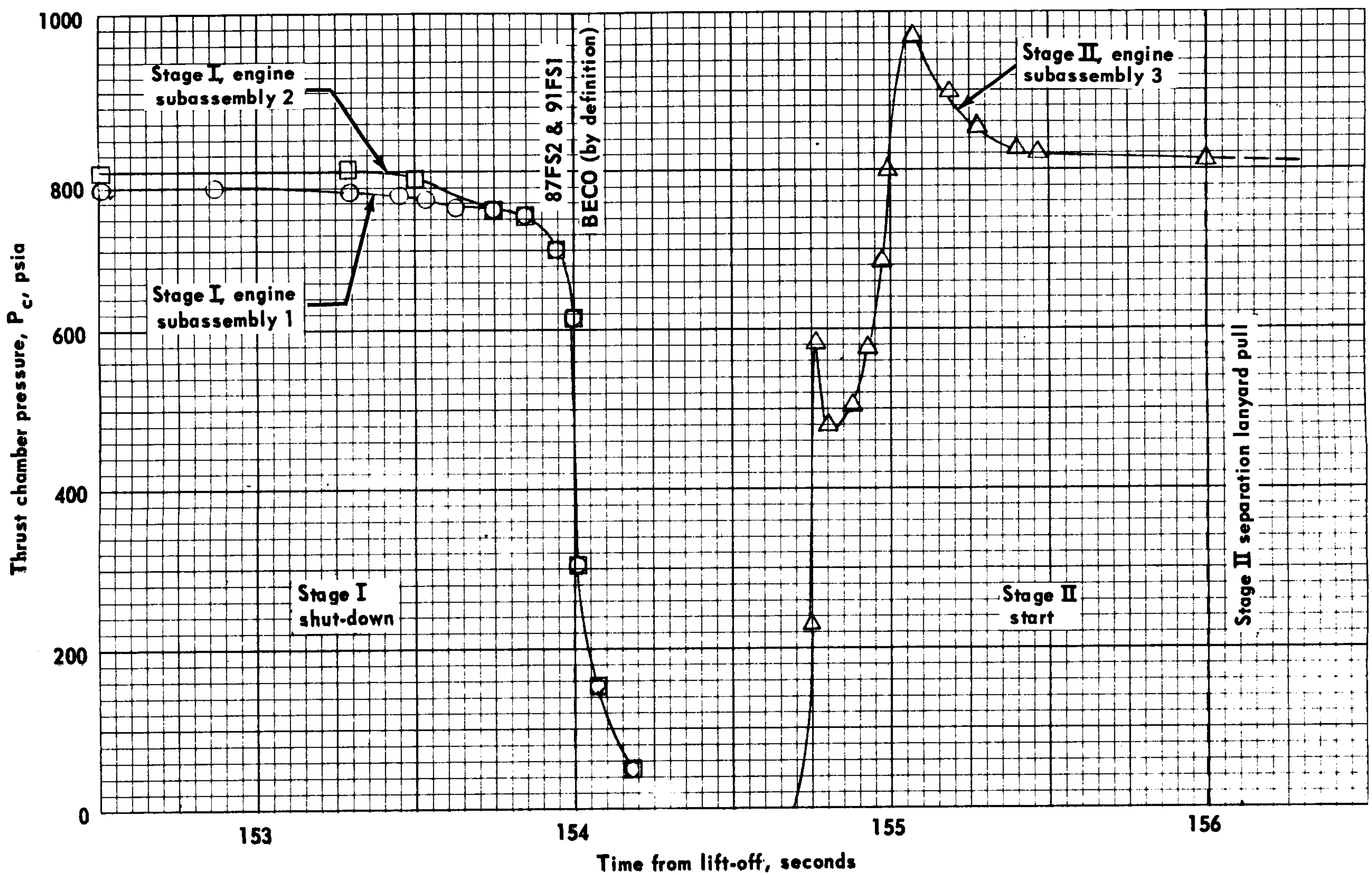


FIGURE 5-29 STAGE I AND STAGE II THRUST CHAMBER PRESSURE AT STAGING

NASA-S-64-3131

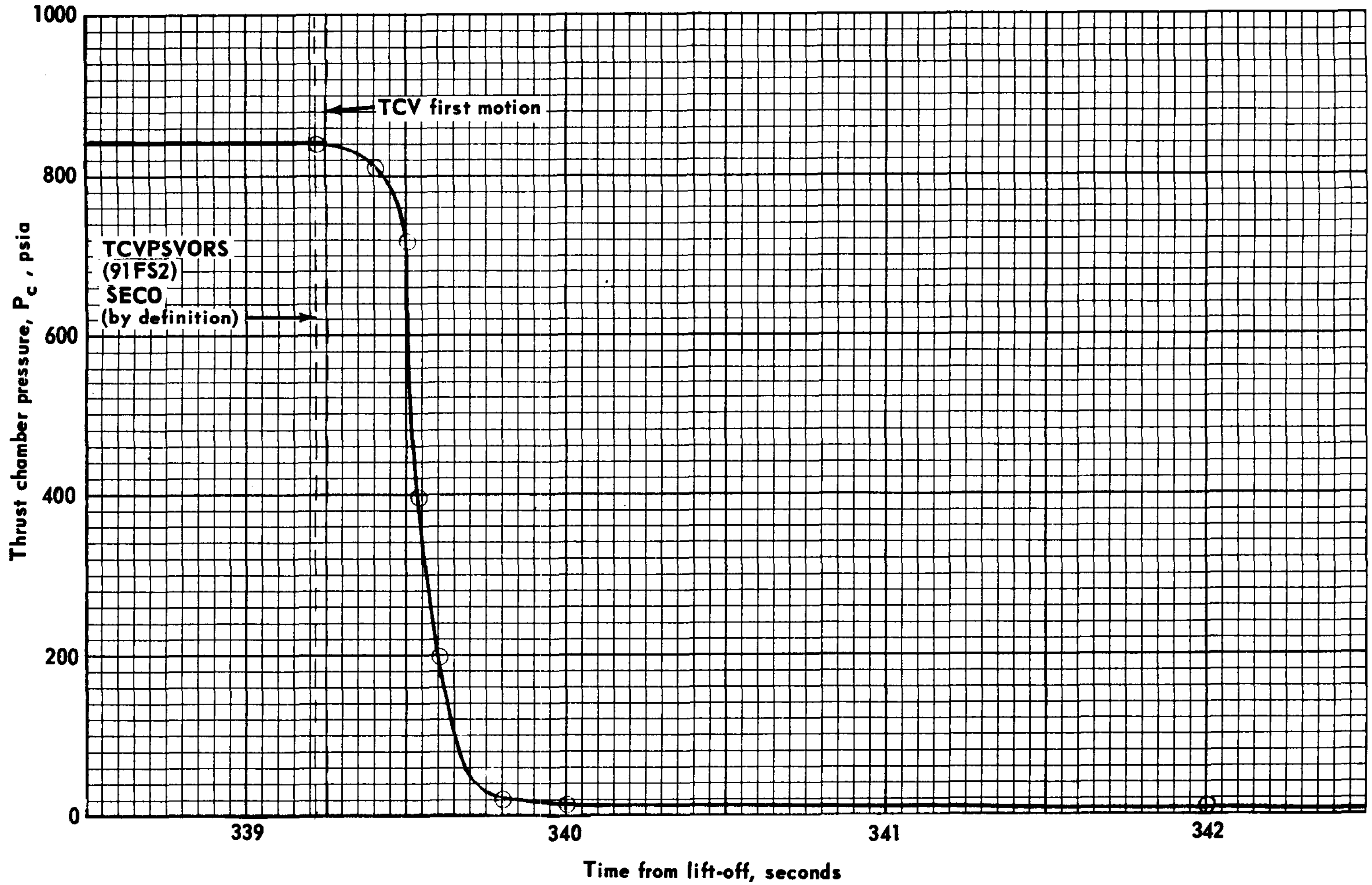
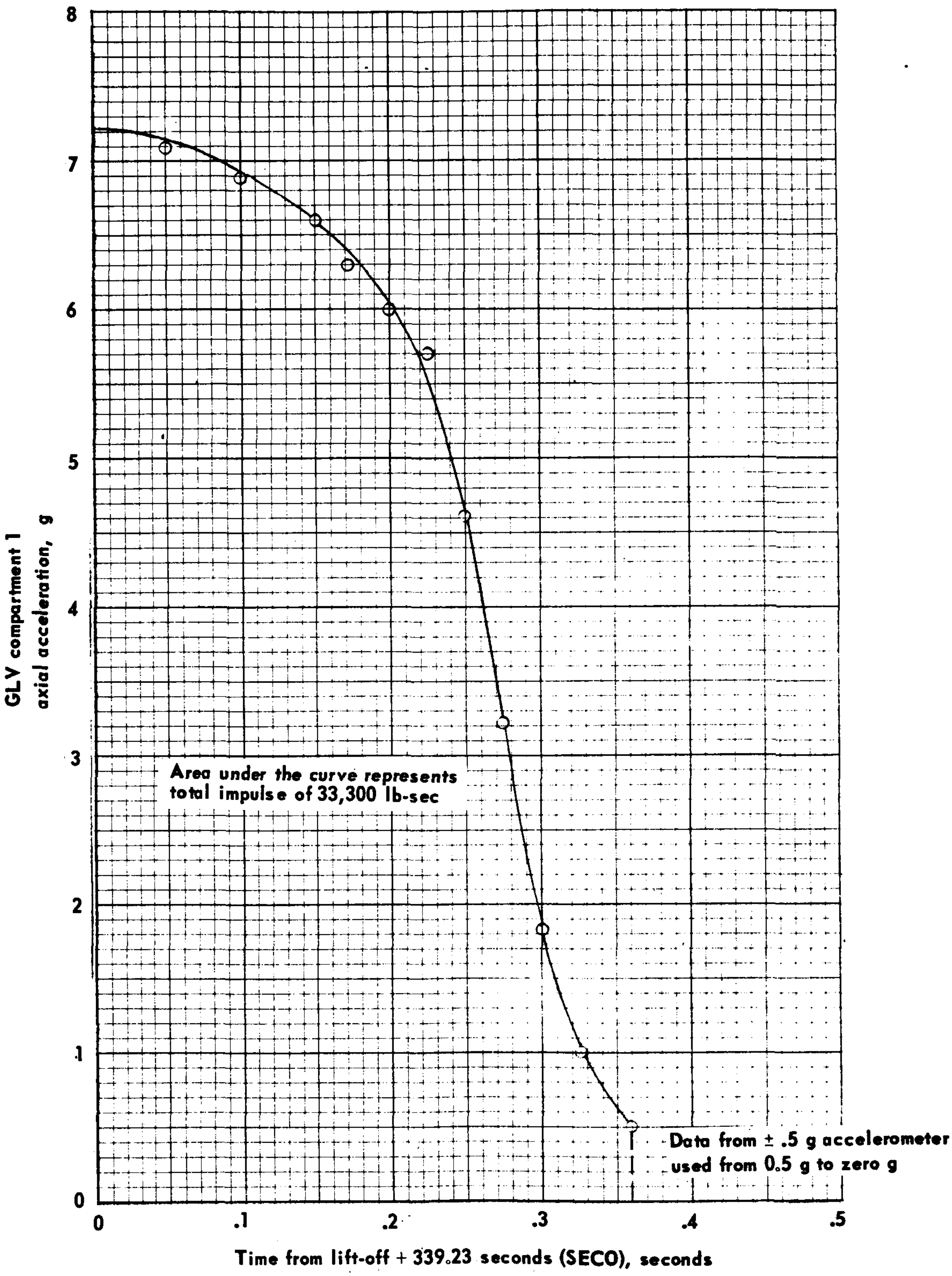


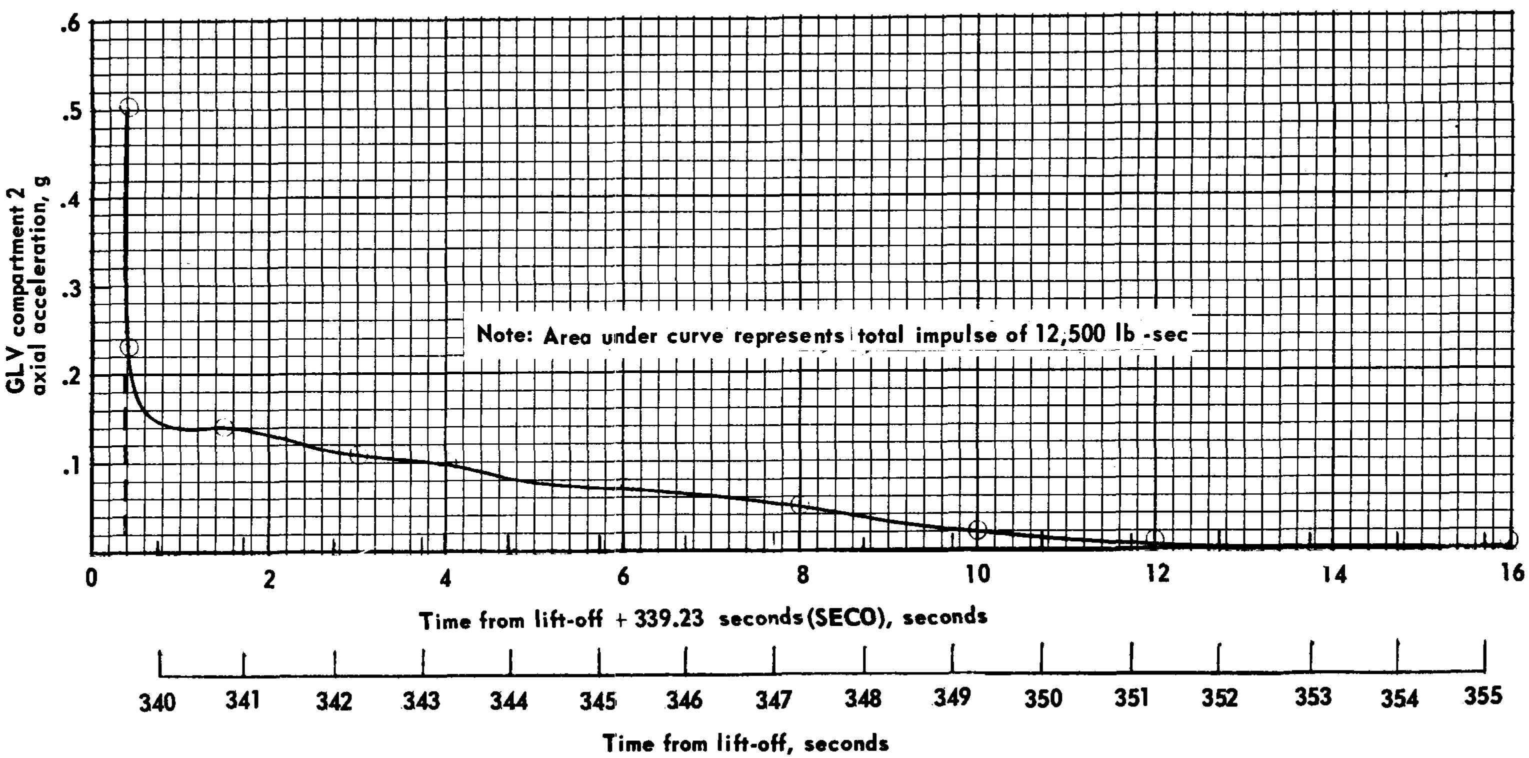
FIGURE 5-30 STAGE II ENGINE SHUT-DOWN TRANSIENT



(a) ± 10g accelerometer

FIGURE 5-31 STAGE II TAILOFF AXIAL ACCELERATION

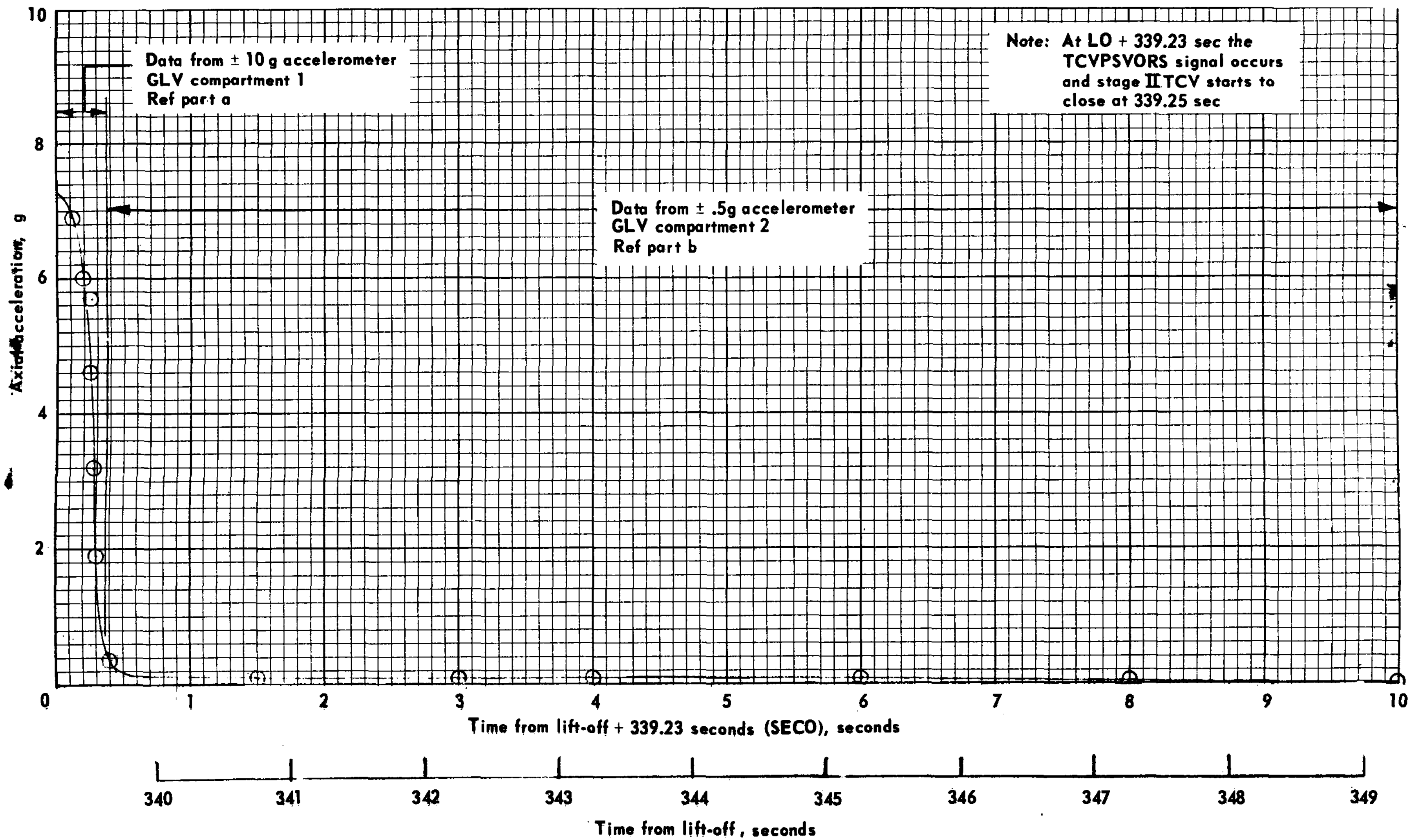
NASA-S-64-3133



(b) ± 0.5 g accelerometer

FIGURE 5-31 (CONTINUED)

[redacted]



(c) Total acceleration

FIGURE 5-31 CONCLUDED

[redacted]

[redacted]

NASA-S-64-3294

Note that the time scales referenced to both engine ignition and lift-off have a dual scale factor (.25 seconds / division and .5 seconds / division).

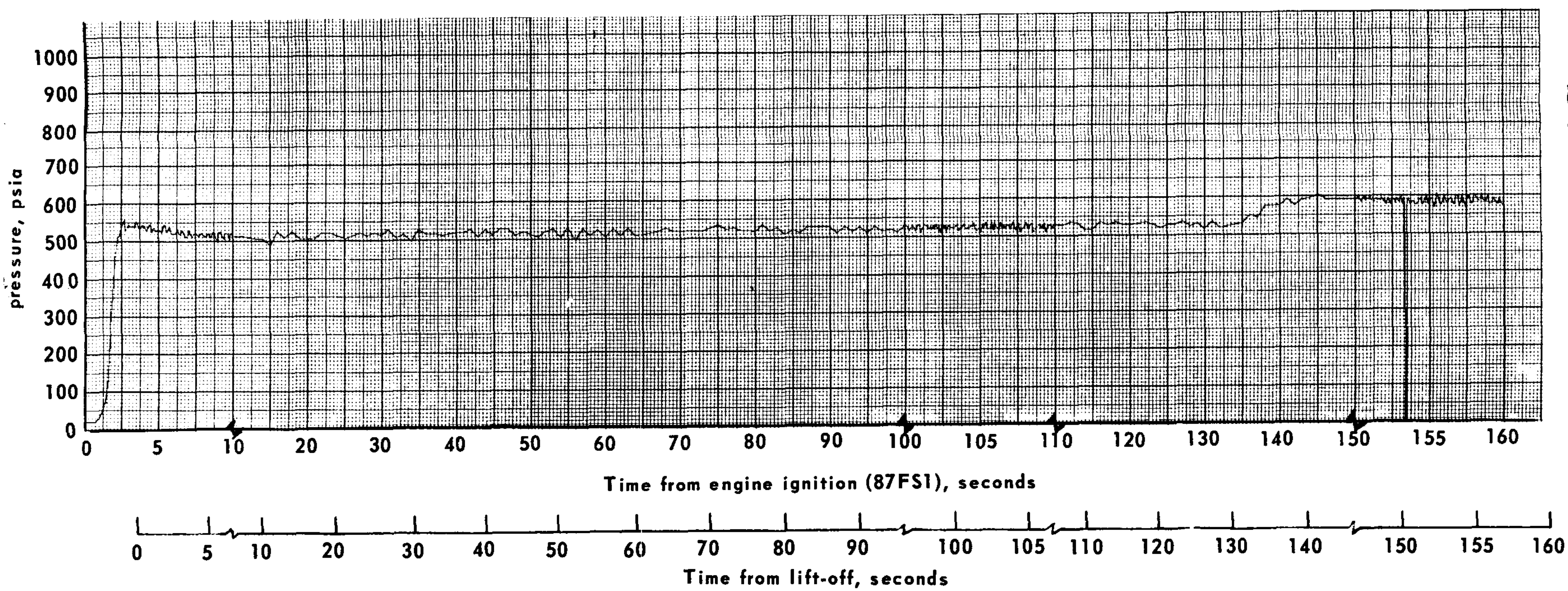
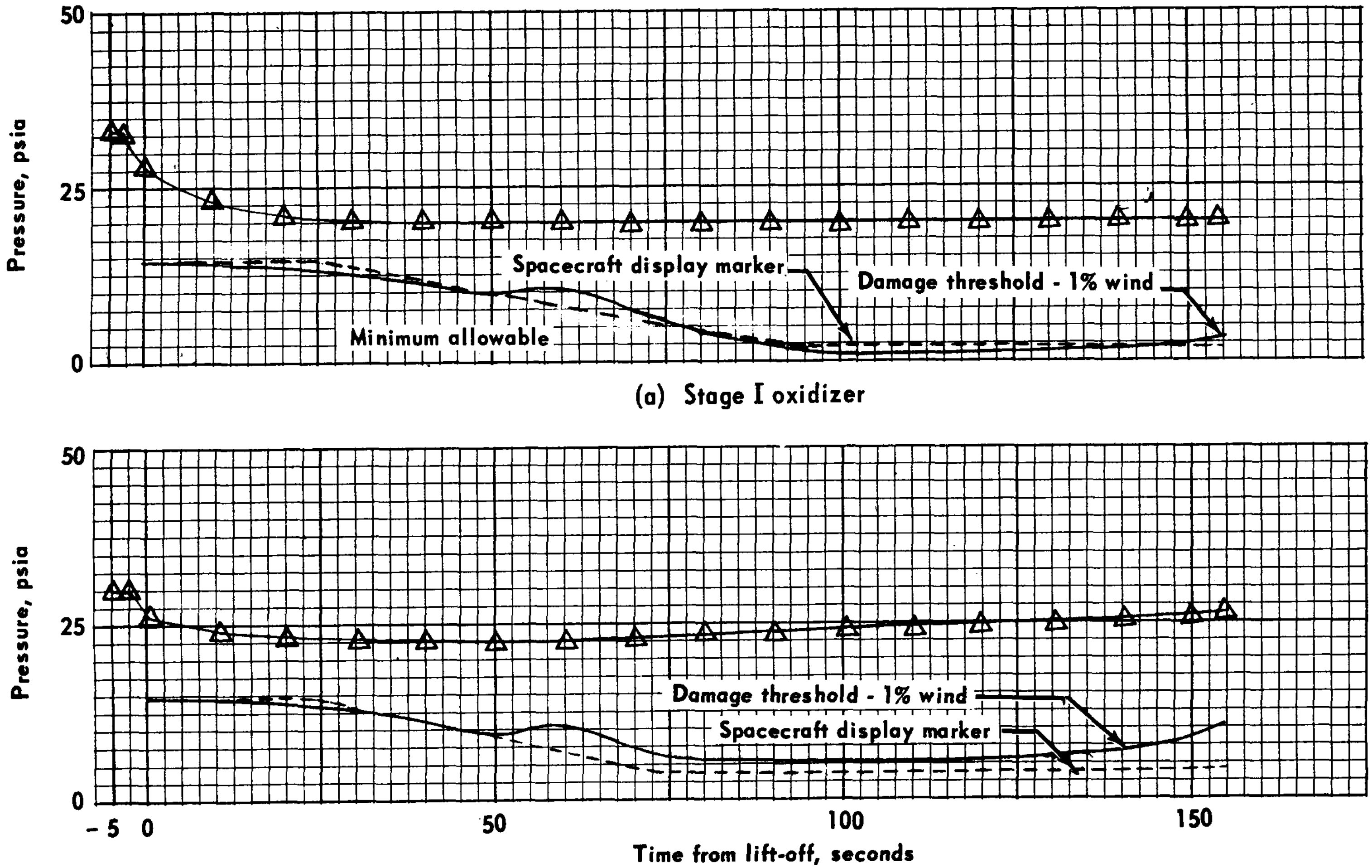


FIGURE 5-32 OXIDIZER PRESSURANT ORIFICE INLET PRESSURE

[redacted]

NASA-S-64-3286

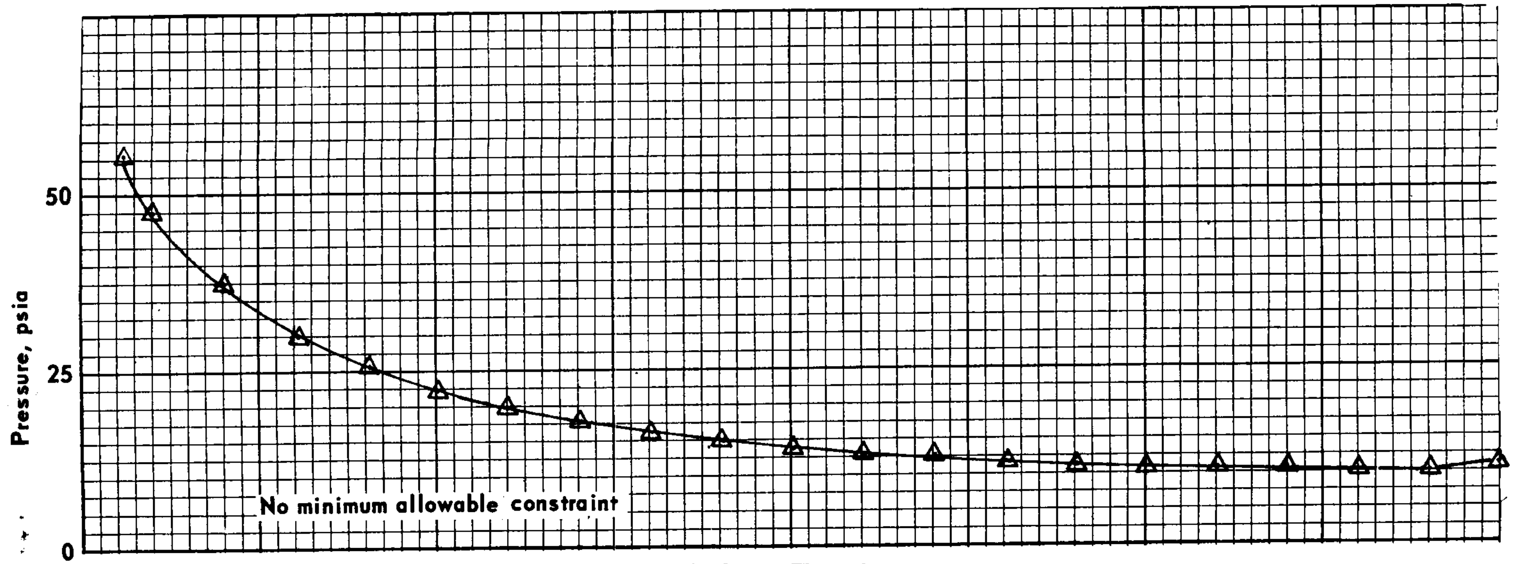


(a) Stage I oxidizer

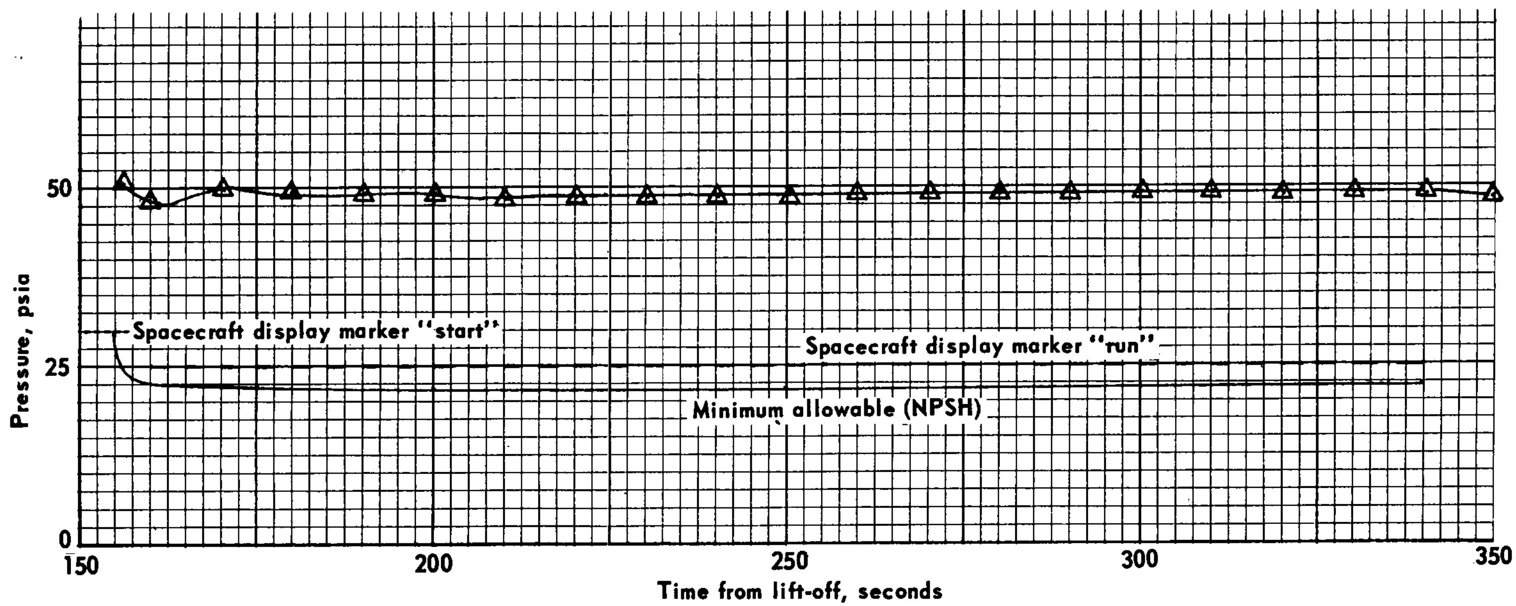
(b) Stage I fuel

FIGURE 5-33 GLV TANK PRESSURE

NASA-S-64-3192



(c) Stage II oxidizer



(d) Stage II fuel

FIGURE 5-33 CONCLUDED

[redacted]

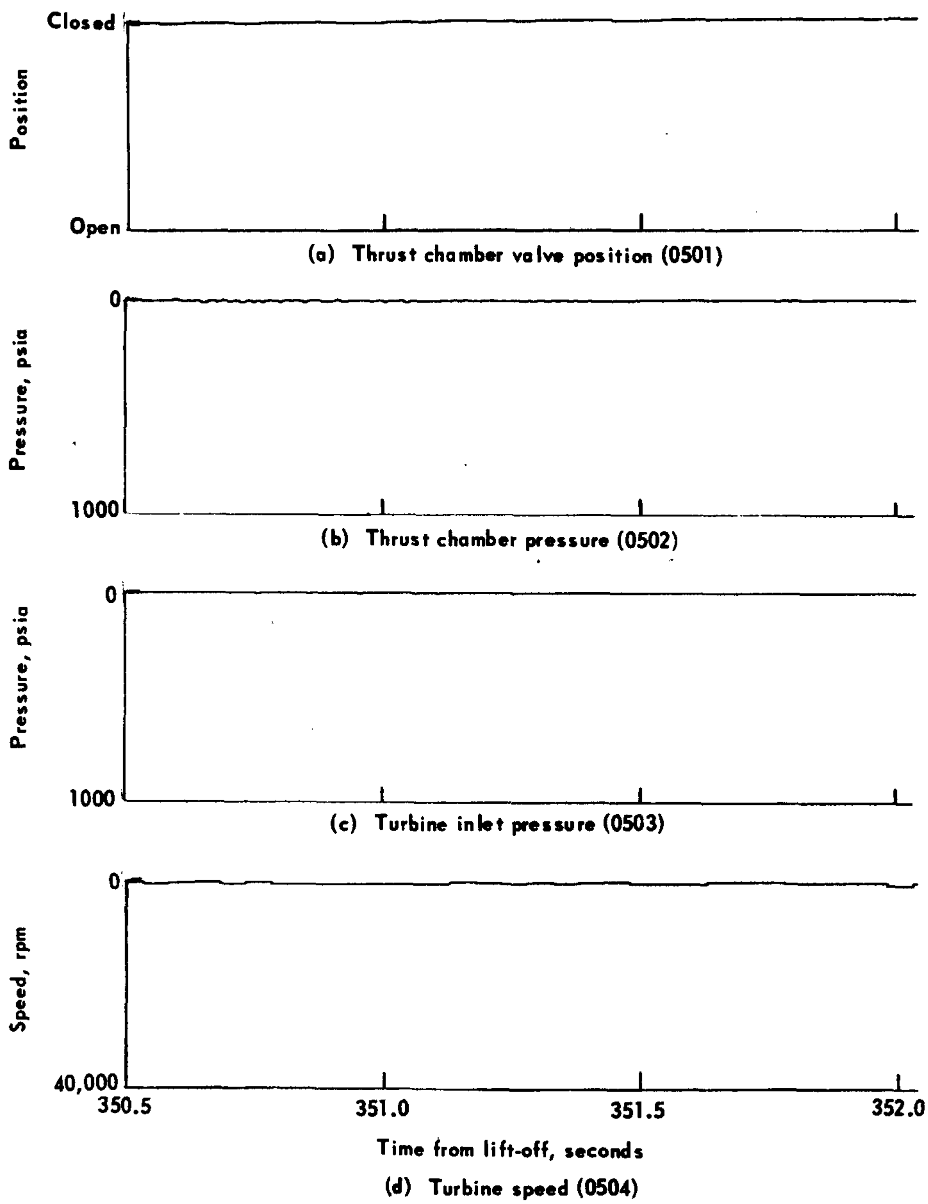
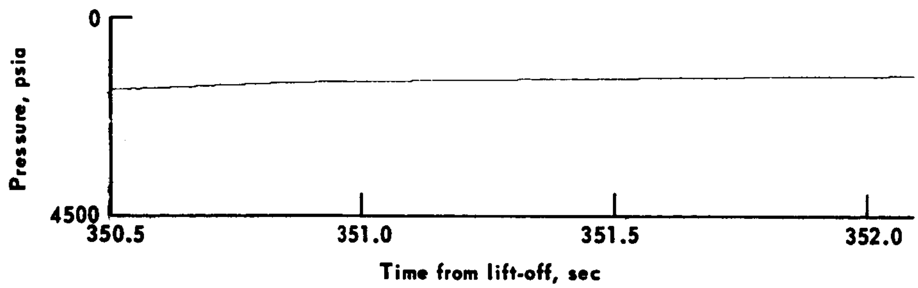


FIGURE 5-34 GLV STAGE II MEASUREMENTS AT LIFT-OFF +351 SECONDS

[redacted]

(e) Actuator no. 5, yaw (0650)

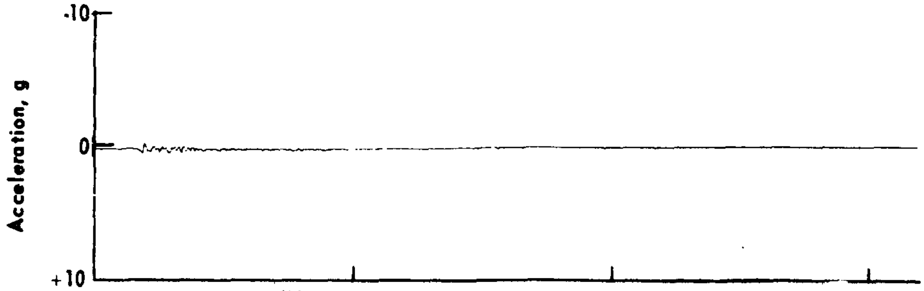
(f) Actuator no. 6 pitch (0651)



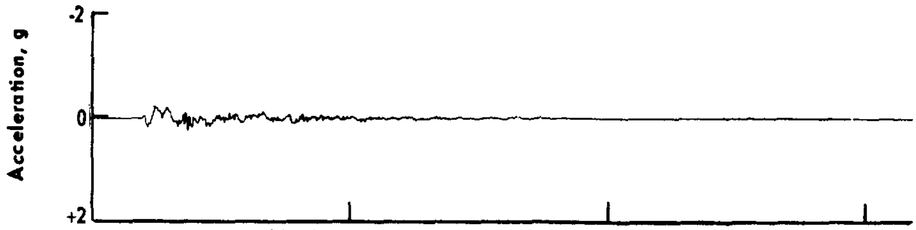
(g) Hydraulic pressure (0653)

FIGURE 5-34 CONTINUED

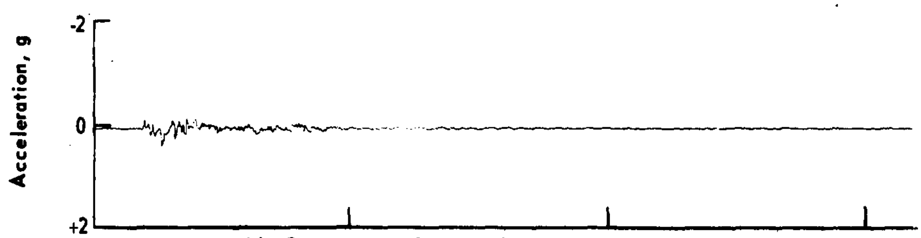
[redacted]



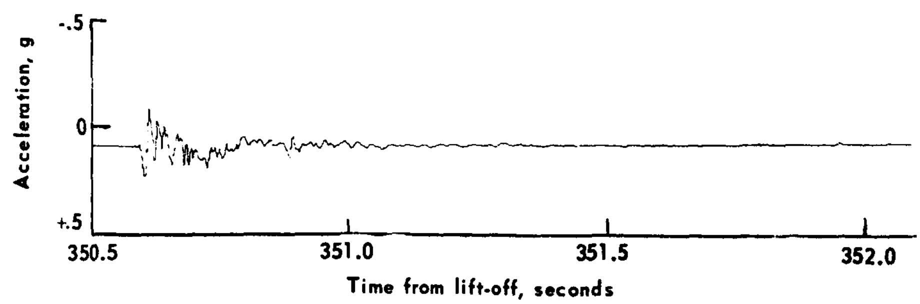
(h) Compartment 1 axial acceleration (0670)



(i) Compartment 1 lateral acceleration (0671)



(j) Compartment 1 vertical acceleration (0672)



(k) Compartment 2 axial acceleration (0699)

FIGURE 5-34 CONTINUED

NASA-S-64-3314 [redacted]

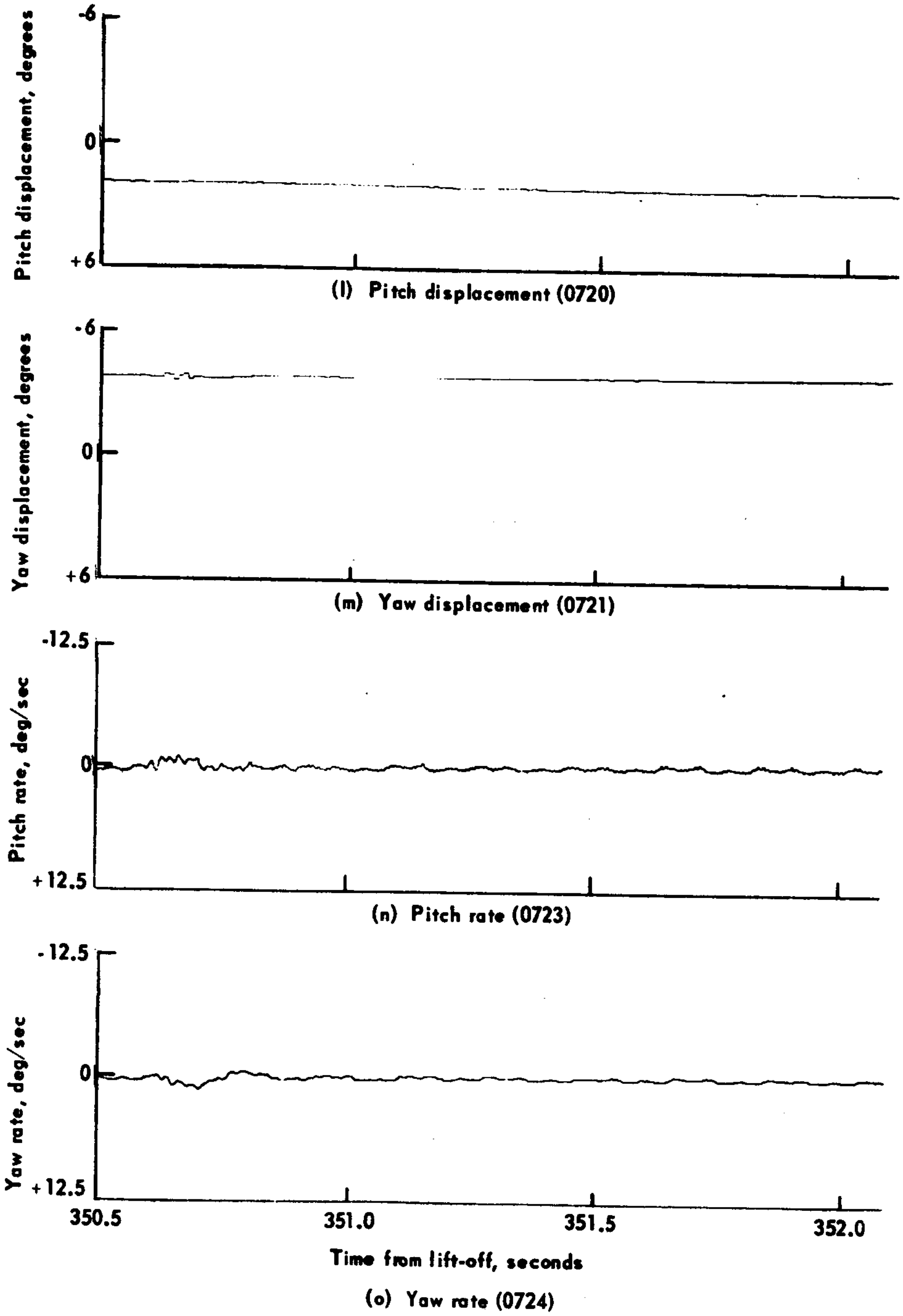
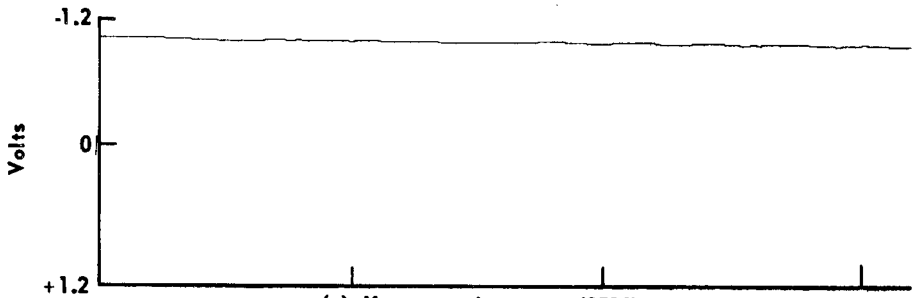


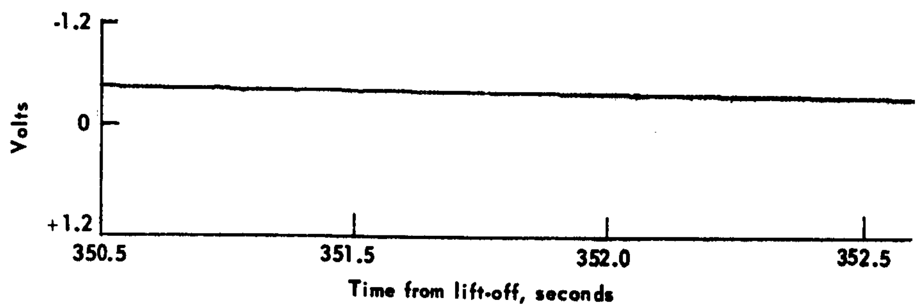
FIGURE 5-34 CONTINUED

[redacted]

(p) Roll rate (0725)



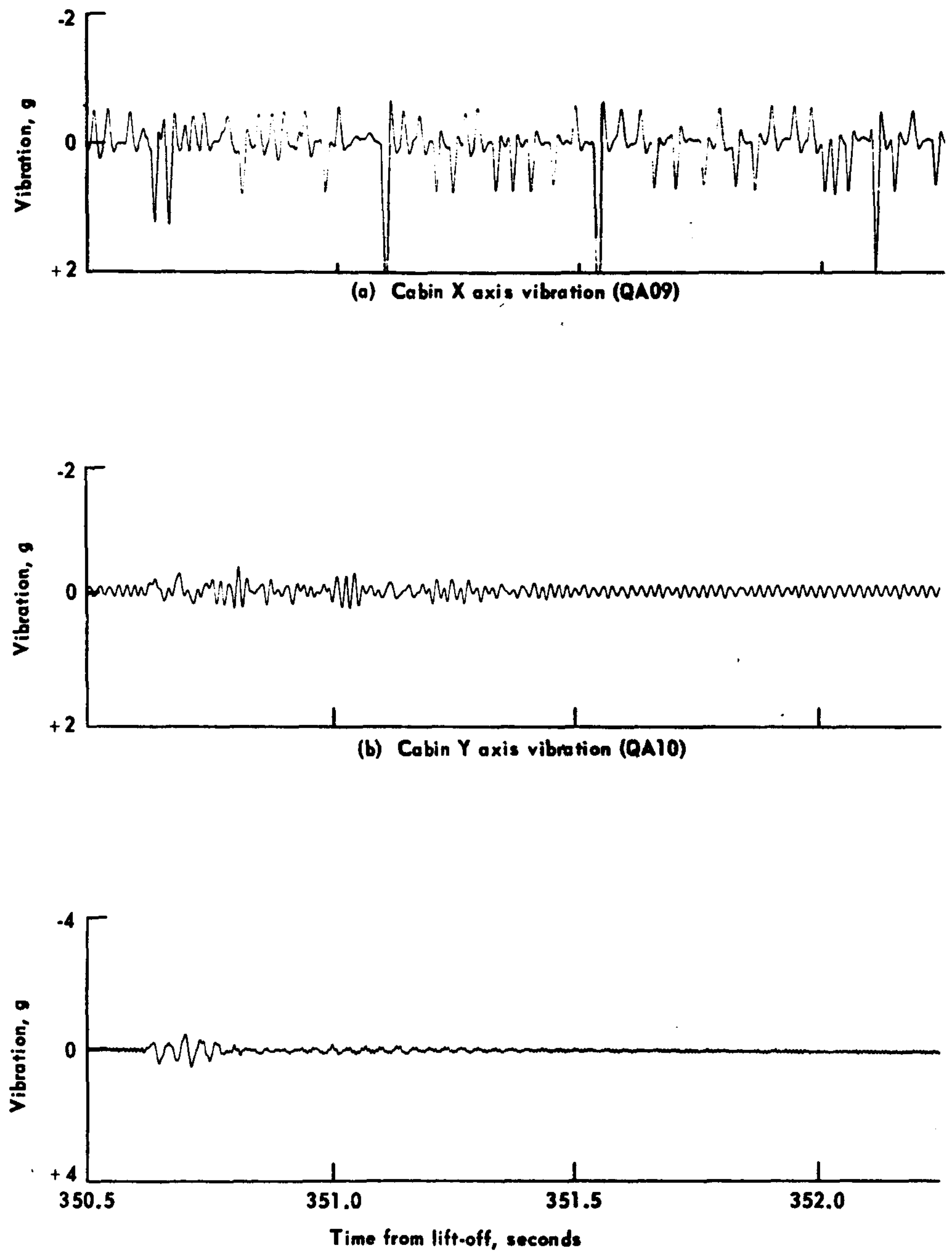
(q) Yaw autopilot output (0730)



(r) Roll autopilot output (0731)

FIGURE 5-34 CONCLUDED

[redacted]



(a) Cabin X axis vibration (QA09)

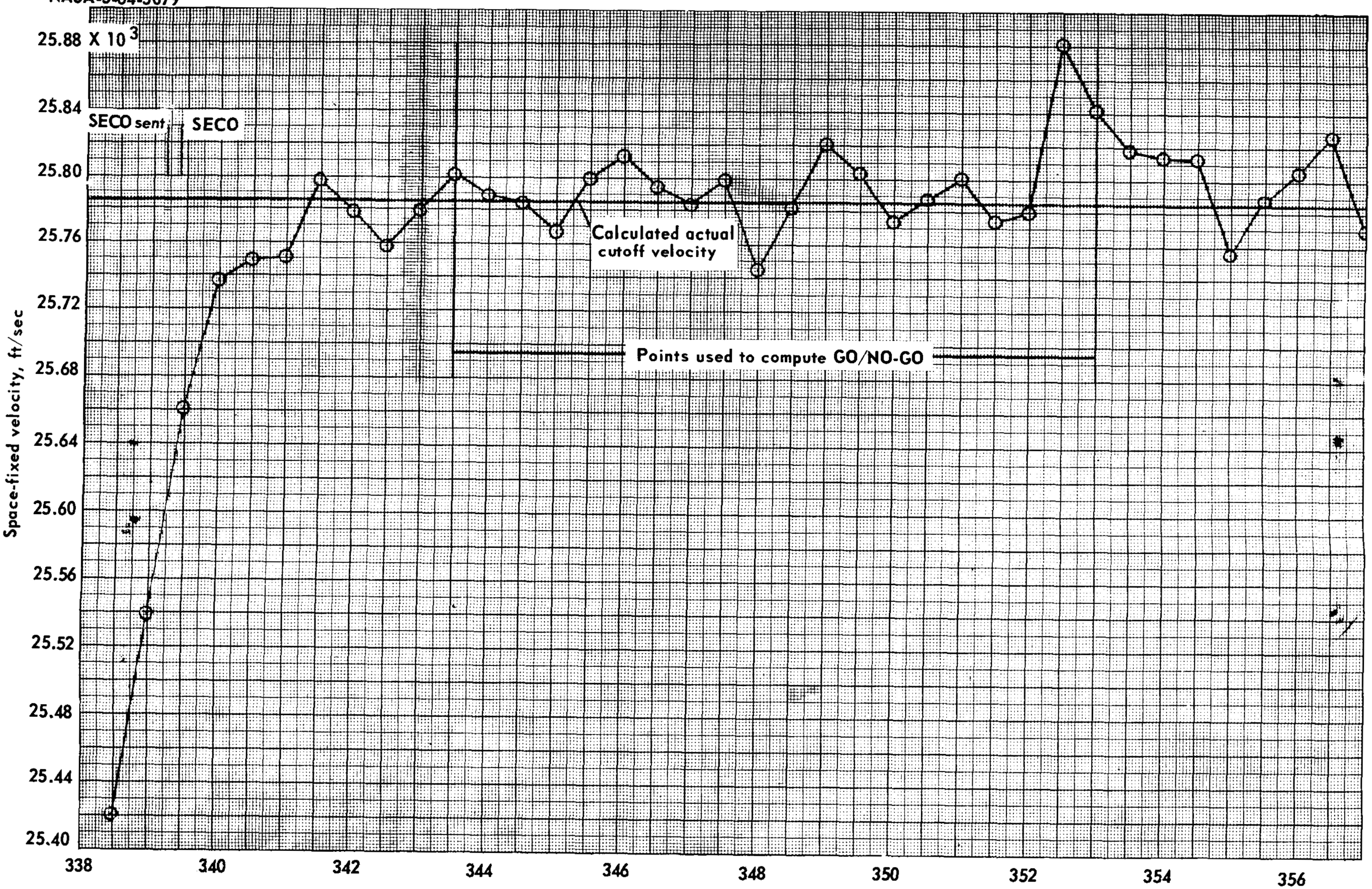
(b) Cabin Y axis vibration (QA10)

(c) Cabin Z axis vibration (QA11)

FIGURE 5-35 SPACECRAFT VIBRATION MEASUREMENTS AT LIFT-OFF +351 SECONDS

[redacted]

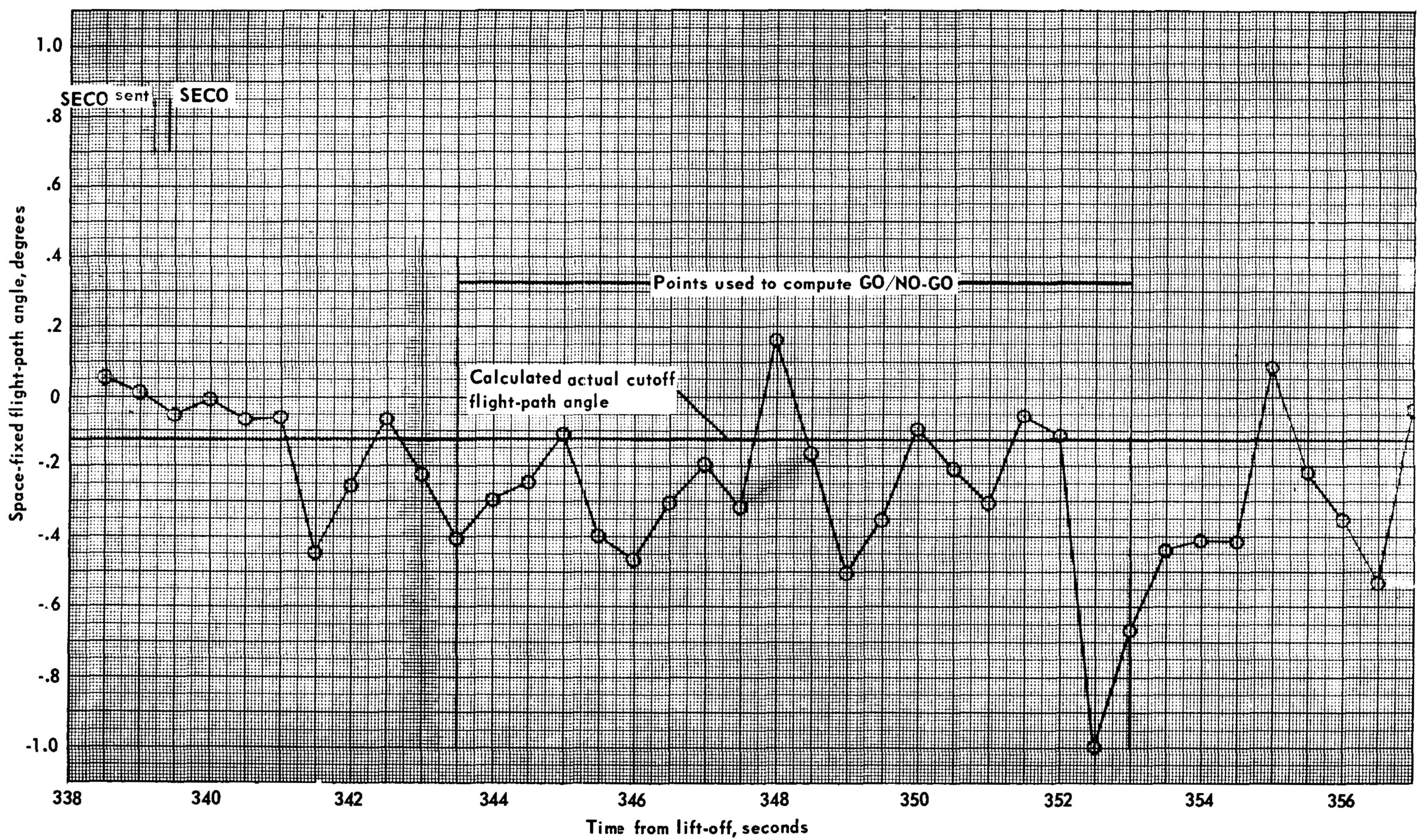
NASA-S-64-3079



(a) Space-fixed velocity.

FIGURE 5-36 SPACE-FIXED VELOCITY AND FLIGHT-PATH ANGLE IN THE REGION OF CUTOFF USING LAUNCH VEHICLE GUIDANCE DATA.

NASA-S-64-3080



(b) Space-fixed flight-path angle.

FIGURE 5-36 CONCLUDED

NASA-S-64-3082

[redacted]

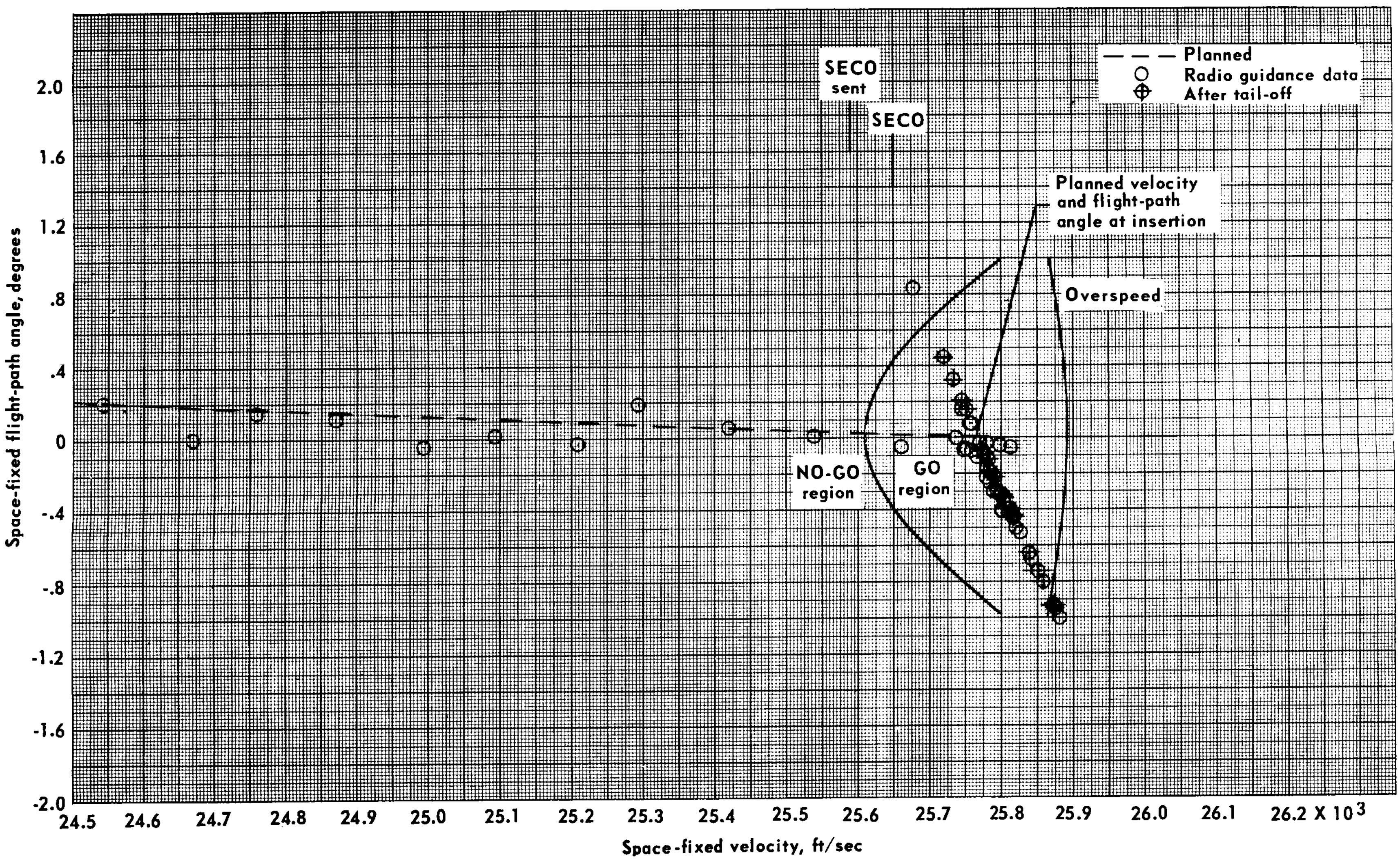


FIGURE 3-37 SPACE-FIXED FLIGHT-PATH ANGLE PLOTTED AGAINST SPACE-FIXED VELOCITY IN THE REGION OF CUTOFF

[redacted]

[redacted]

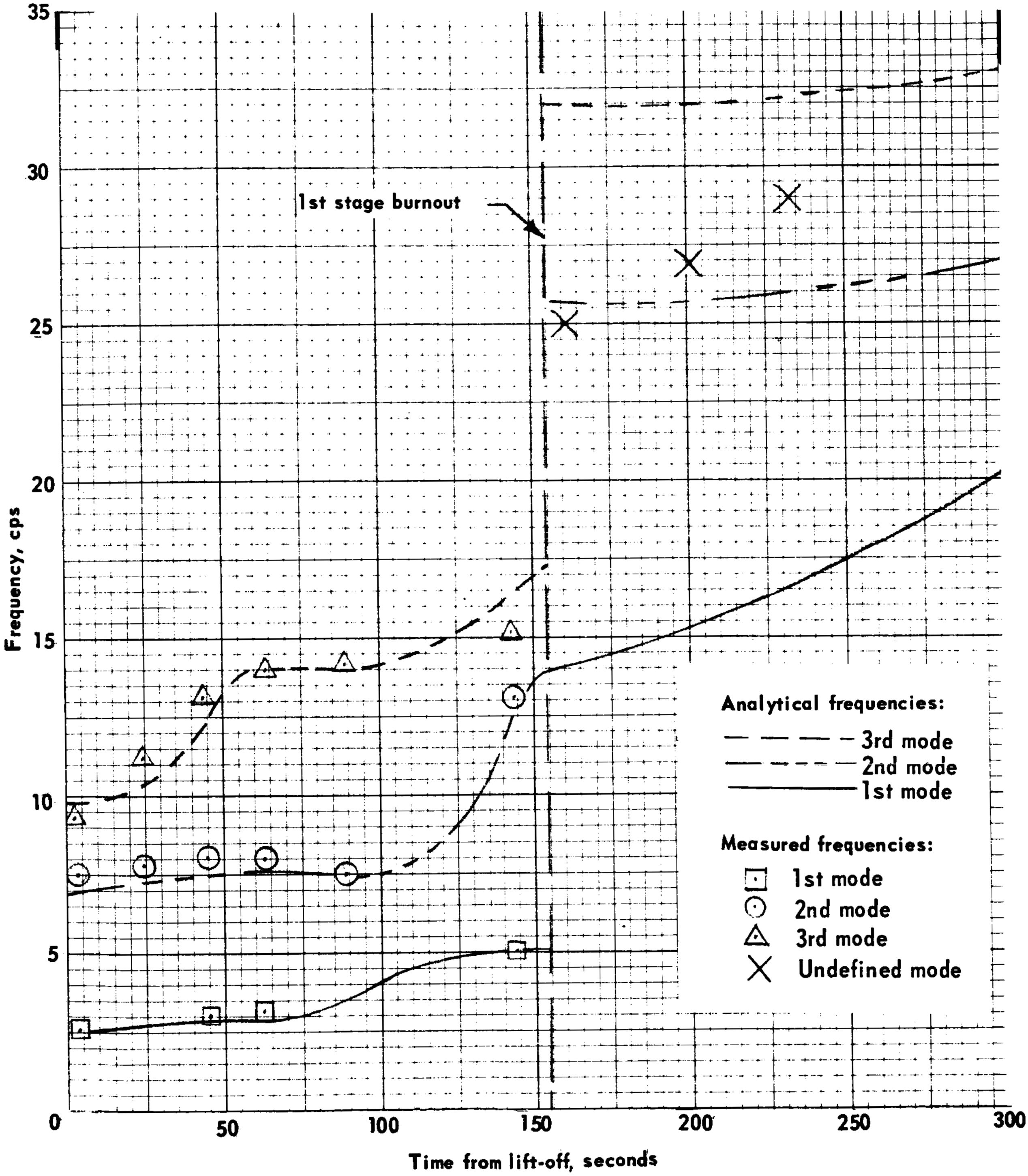


FIGURE 5-38

COMPARISON OF MEASURED LATERAL FREQUENCIES
WITH ANALYTICAL FREQUENCIES PRESENTED IN REFERENCE 7

[redacted]

6.0 MISSION SUPPORT PERFORMANCE

This portion of the GT-1 mission report consists principally of a narrative of events that occurred during the prelaunch, launch, and orbital phases as they were witnessed from the Mission Operation Control Room of the Mission Control Center (MCC). These events are centered about the spacecraft and launch vehicle countdown, the MCC countdown, the post-lift-off support from T-390 minutes through the third orbital pass of the spacecraft, and the Gemini tracking network countdown beginning at T-330 and terminating with reentry of the spacecraft during the 64th orbit. The network operations section includes an evaluated report of range equipment performance. From an overall viewpoint, the mission support was considered nominal. No serious problem areas developed to hinder the accomplishment of mission objectives. Efforts were made to exercise all mission command and control positions within the MCC Mission Operations Control Room in a manner similar to their functions during a manned orbital flight. This effort was beneficial in pointing out areas requiring procedural clarification and countdown integration.

6.1 PRELAUNCH OPERATIONS

6.1.1 Gemini Spacecraft

The space-vehicle launch operations were planned about a 390-minute split countdown, with a scheduled $23\frac{1}{2}$ hour hold at T-330 minutes to prepare the spacecraft for final countdown and to allow time to conduct launch-vehicle systems tests, to install the pyrotechnics mechanically, to hook up electrically all pyrotechnics except the stage I start cartridge and the destruct initiator, and to load the fuel and oxidizer. The second part of the split countdown started at 5:30 a.m. e.s.t., April 8, 1964, and progressed smoothly through lift-off at 11:00:01 a.m. e.s.t.

The space-vehicle launch countdown for GT-1 was planned around a normal Titan II countdown and a countdown which is anticipated to be the approach for subsequent GT missions. A precount was conducted to allow last-minute observations of systems operations before committing the space vehicle to the actual countdown. The $23\frac{1}{2}$-hour hold in countdowns for subsequent missions will be expanded to include spacecraft armed stray voltage checks, pyrotechnic hookup, and cryogenic soak. The remote power application in GT-1 was a practice feature for subsequent missions with pyrotechnics. The countdown from T-100 minutes until lift-off appears to be compatible for manned missions; however, from T-300 to T-140 minutes, the times will need expansion.

6.1.2 Gemini Launch Vehicle

The GLV countdown began on time at T-300 minutes and progressed to a successful lift-off with no holds.

The following items of interest occurred during the countdown:

(a) Prior to picking up the count, the temperature differential between fuel and oxidizer tanks of stage II was out of limits. The open north-side erector curtains were closed around the warmer oxidizer tank, and the tank barrel was wrapped with plastic film to reduce the rate of warmup. At the same time, curtains on two sides were opened at the fuel tank level to cause more rapid warmup. Monitoring was maintained and at T-75 minutes, the difference had decreased to less than 5° F.

(b) At T-80 minutes, the only elevator to the spacecraft level was reported to be inoperative. The launch-complex support crew located a gate that had jarred open and latched it to close the safety circuit. The elevator was operative by T-60 minutes, in time for all necessary operations.

(c) Late in the count, the flight control's Aerospace Ground Equipment console caused an erroneous readout of the roll valve drive amplifier. Reference to telemetry recordings and additional checks at the console indicated that the launch vehicle was responding properly, and the reading did return to normal before lift-off.

6.2 FLIGHT CONTROL

6.2.1 Gemini Spacecraft

6.2.1.1 Prelaunch phase.- The detailed GT-1 mission operations were initiated on March 27, 1964, at the mission briefing. All facilities and mission operations personnel were in a "ready" condition when the personnel were deployed to their stations on March 31. The wet mock simulated flight conducted at complex 19 was supported by the Mission Control Center (MCC) on F-6 day. Mission control launch simulations were conducted on F-5 and F-4 days. These simulations also were used as the first planned exercises in preparation for future missions. Fifteen launch-phase simulations were conducted in which the Guidance, Flight Dynamics, and Gemini Launch Vehicle (GLV) Systems Engineer consoles were exercised. These simulations were very useful in further defining the mission rules and standard operating procedures at those positions. The spacecraft simulated flight was completed on F-3 day, and spacecraft telemetry records were reviewed in MCC. A complete countdown network simulation was conducted on F-2 day. This test was used to verify the MCC countdown integrity and to prove network readiness. No major problem areas were noted.

The flight controller and network countdown was normal until the time of lift-off. The flight-control communications checks were completed by T-165 minutes. The MCC was in a "ready" status for tank pressurization at T-154 minutes.

The flight dynamics trajectory run at T-145 minutes began on schedule and all data and support facilities were acceptable during this test.

From T-138 to T-121 minutes, manual fuel cut-off and destruct command indications were received at the GLV Systems Engineer console in MCC. These commands were a part of the command setup checks performed by the Superintendent of Range Operations (SRO). The apparent lack of coordination during this test caused some concern. Subsequently, the problem was identified as a lack of understanding of normal range operating procedures by the launch-vehicle contractor and MCC personnel.

At T-38 minutes the Mistram II radar on Eleuthera Island in the Bahamas reported phase problems resulting from propogation conditions associated with heavy rainfall in that area. At T-20 minutes Mistram II data were declared acceptable. It was later reported that Mistram II was used as a data source late in powered flight, but the data were unacceptable. No suitable explanation of this problem was obtained during the mission period. The MCC received spacecraft telemetry at T-28 minutes, and MCC telemetry and displays were acceptable. The network voice was good during the check at T-15 minutes with the exception of a Carnarvon communications failure due to a line break south of the station. A procedure was established at the Adelaide, Australia, communications center to relay the lift-off time and acquisition data to Carnarvon via three amateur radio stations (Australian homesteaders). This procedure proved to be effective, although the data confirmation was greatly delayed.

The launch-vehicle and spacecraft internal power transfer was normal, and all MCC instrumentation was acceptable at lift-off.

6.2.1.2 Launch phase.- At first stage ignition, all launch-vehicle indications on the GLV Systems and Guidance consoles were at the expected values. For the first few seconds of flight, the primary hydraulic pressure indicated the normal fluctuations associated with engine gimballing, and then stablized. The indications of lateral velocity for the stage I flight were as predicted. The impact predictor (IP) was used as the selected data source until approximately LO + 55 seconds with smooth data indicating a nominal trajectory. MOD III radar data received via the Goddard Space Flight Center (GSFC) were used from LO + 55 seconds throught SECO. Data from all sources agree throughout the flight. The values presented in this section were observed and noted in the MCC for the purpose of real time monitoring. Values in other sections of this report which differ from the values found here are the result of an intensive postflight evaluation. These differences for the GT-1 mission, however, were of such a minor nature that they were not worthy of being discussed.

Spacecraft and GLV parameters displayed for flight control were monitored throughout powered flight. In the GLV, the primary and secondary hydraulic pressures dropped slowly from 3,100 psi to approximately 2,700 psi during stage I flight but remained within acceptable limits. The thrust-chamber pressure increased slightly. All electrical parameters in the GLV remained approximately constant except for normal variations in current in the auxillary power system (APS). These variations had been noted in previous tests, and the cause had been isolated to heater cycling. At lift-off the spacecraft main bus voltage indicated 24.8v d-c, and the main bus current was 6.3 amperes. The cabin pressure started to decrease at LO + 52 seconds, and the cabin pressure relief valve sealed off at 5.5 psid at LO + 126 seconds. The spacecraft telemetry readings remained acceptable throughout the launch phase. The break wires on the scanner fairing remained intact throughout the launch phase.

The shutdown enable signal was received at LO + 40 seconds, and the staging arm signal was received at LO + 145 seconds. First-stage flight appeared to be completely normal. At staging, a brief (0.5 second) telemetry dropout occurred, after which the BECO and stage separation lights were illuminated. At this time the second stage hydraulic pressure rose simultaneously with second-stage thrust-chamber pressure. These pressure increases indicated a normal staging sequence. Staging and guidance initiate lights illuminated on time and steering initiate command was noted at the normal time. The initial pitch steering was the same as predicted, and after the initial heavy pitch-down steering (characteristic of the guidance equations), the subsequent pitch steering was so light that it was not noticeable in the decoder output. Stage II steering was very good, and the trajectory was within the specification values during powered flight.

The longitudinal acceleration built up to 5.5g just prior to stage I fuel depletion. During second-stage flight, the GLV hydraulic pressure was very stable and thrust-chamber pressure rose slightly. All second-stage electrical measurements were normal. At SECO the thrust-chamber pressure dropped and the engine under-pressure warning and SECO lights illuminated. The hydraulic pressure remained at the engine running level for about 10 seconds, dropped to zero, then rose to one-third the former level. During second-stage flight, the fuel and oxidizer tank pressures were normal. Good agreement was observed between redundant sensors on both stages. At no time were switchover to secondary autopilot or spin motor rotation detection (SMRD) signals noted.

At SECO plus 11 seconds, an unexplained vibration was displayed in the 1.0g range on the low-frequency vibration measurements from the cabin and rendezvous and recovery (R and R) sections. It was particularly noticeable along the X- and Y-axes in the R and R section.

At SECO plus 20 seconds, the roll engine was observed to oscillate at about 2 cpm, and the observation was confirmed by an oscillation in roll error of approximately ±3° at the same period. This oscillation was thought to be a result of tank pressure blow down through the roll nozzle. In addition, from SECO to approximately SECO plus 10 seconds, the vehicle appeared to pitch at about 0.3 deg/sec. At SECO, there was a X-range velocity of 70 ft/sec to the left, resulting from a yaw-velocity error which occurred during the last 30 seconds of powered flight. The conditions at SECO were acceptable and a "go" decision was obtained directly from the guided-missile computer facility (GMCF) no. 1. From the GMCF data, the Goddard Space Flight Center (GSFC) determined that the flight-path angle was -0.3°, that the velocity was 159 ft/sec greater than that required for $1\frac{1}{2}$ orbital passes, and that the velocity was

40 ft/sec greater than planned. The Bermuda data confirmed the GMCF "go" decision. No IP solution was available; however, MCC obtained a SECO vector based on FPQ-6 data. The SECO conditions from the three sources were as follows:

Data sources	Points	Velocity, ft/sec	Flight-path angle, deg
GMCF	20	25,805.8	-0.3023
FPQ-6	25	25,733	+0.5006
Bermuda		25,794.8	-0.0946

At launch, Bermuda experienced a small problem with its acquisition aid due to the unexpected 300-kc telemeter RF carrier deviation. The deviation presented no telemetry recording operational problems as far as could be ascertained at launch time. However, it did result in a late acquisition of the radar transponder.

No excessive rises in outer skin temperature (maximum of 150°) were noted, nor were any high vibrations monitored. Longitudinal acceleration at SECO was 7.5g. At loss of signal (LOS) with MCC at the beginning of the first orbital pass, the spacecraft battery lifetime was estimated at approximately 7 hours.

6.2.1.3 Orbital phase.- The flight controllers' responsibilities during the orbital phase for this mission consisted of monitoring station acquisition of radar and telemetry signals. Since none of the data were decommutated and displayed at the remote stations, the remote stations did not analyze the data in real time. The MCC had three real-time recorders in the operations room which displayed cabin pressures; temperatures; battery voltages and currents; and miscellaneous temperatures, pressures, and accelerations.

Readouts of cabin pressure during the orbital phase were 5.4 psid at the end of the first orbital pass, 5.2 psid at the end of the second, and 4.7 psid at the end of the third.

An estimate of temperatures within the spacecraft was made based upon monitoring parameters PD57 and PD58 (cabin wall temperatures on the left and right side of the pressure vessel, respectively) and also PD29 (inner right-hand window temperature). The results were as follows:

	Cabin wall left, ° F	Cabin wall right, ° F	Inner pane, ° F
Before lift-off	83	74	99
MCC LOS - Beginning of first orbital pass	90	74	90
End first orbital pass	85	80	88
End second orbital pass	80	80	70
End third orbital pass	80	87	63

These values were read from sensors located on the inner cabin wall and window, and of course are much lower than those measured on the outer skin which are discussed in section 5.0.

All voice exchanges between the MCC and remote stations consisted of telemetry and transponder contact times, signal strength, estimated tumbling rates, and radar targets.

The orbit of the combination spacecraft and GLV stage II was determined to have a perigee of 86.5 nautical miles and an apogee of 173.1 nautical miles after the differential correction from the Bermuda station. At the end of three orbital passes the orbit was determined to have a perigee of 86.4 nautical miles and an apogee of 171.0 nautical miles. The orbital lifetime was predicted to be 60 orbital passes.

Flight control support was terminated at 4:00 p.m. e.s.t. at the end of the third orbital pass, with the completion of the mission debriefing over the Goddard conference loop.

6.3 NETWORK OPERATIONS

The network countdown was picked up at the Mission Control Center (MCC) at T-330 minutes. Initial computation and data flow instrumentation subsystems (CADFISS) tests with the Patrick Air Force Base radars and the Cape Kennedy radars had to be rerun, an occurrence which indicated a need for preplanning reruns on these two radars. This need became particularly evident when the Grand Bahama Island and San Salvador radars failed the slew tests. With only one circuit available to conduct the reruns, the rerun time allotted at T-110 minutes proved to be insufficient. Integrating the range countdown and the Goddard Space Flight Center (GSFC) computer at approximately T-225 minutes and attempting to find acceptable rerun times indicate a definite need to have separate rerun times for the T-330 minutes and T-260 minutes CADFISS rollcalls.

The Bermuda station reported two cases of frequency interference after T-180 minutes. The interference due to the Kindley Air Force Base Tower at Bermuda was resolved by a tower frequency change, and the other case was tolerated but not resolved.

At T-38 minutes the missile trajectory measurement (MISTRAM) II station at Eleuthera Island in the Bahamas had phase problems due to propagation. At T-20 minutes, MISTRAM II was declared to be in the "go" condition, and it remained in the "go" condition throughout the tracking interval. For more detailed information on MISTRAM II operation, refer to the Range Safety section of this report, subsection 5.2.8.1.1.

The initial camera-coverage reports were very poor, and the weather did not begin to clear until T-73 minutes. The reports after T-15 minutes indicated that camera coverage was adequate for launch. Three U.S. Navy A3D aircraft from the Jacksonville Naval Air Station, Jacksonville, Florida, provided lift-off coverage. Two aircraft from the Air Force Missile Test Center (one C-131 at an altitude of 6,000 ft and one at an altitude of 9,500 ft) and a C-54 aircraft from the Air Force Communications Squadron (AFCS) flying at an altitude of 10,000 feet obtained photographic coverage of the launch area. An Air Force Missile Test Center (AFMTC) B-57 aircraft in the launch area covered the ascending Gemini launch vehicle (GLV) from an altitude of 20,000 feet to stall altitude.

At lift-off, no major problems existed in the network. The communications and teletype circuits were excellent throughout most of the mission. One major communications problem occurred at the Carnarvon, Australia, station at 9:23 a.m. e.s.t. The problem was due to a line break approximately 65 miles south of Carnarvon. Communications to Carnarvon were unavailable until 2:02 p.m. e.s.t.; however, good communications were maintained throughout the mission with the Woomera, Australia, station.

Radar performance was good and transponder operation was continuous throughout the mission. At the end of the second orbital pass (2:06 p.m. e.s.t.), the White Sands tracking station reported that two objects were being tracked, with the second object 0.54 nautical mile higher, three times larger, and approximately 99 seconds behind the first object. The North American Air Defense (NORAD) space detection and tracking system (SPADATS) at Moorestown, New Jersey, confirmed that two objects were being tracked. The Laredo, Texas, tracking station for NORAD detected three objects: one large object and two small objects very close together.

On the third orbital pass, the Laredo, Texas, station obtained good tracking on the large object and hits on the smaller two (no estimate on size). Personnel at the Laredo station stated that the separation between the larger and smaller objects was approximately 105 seconds and that the separation between the two smaller objects was approximately 14 seconds. The White Sands Nike-Zeus station also verified the tracking of three objects.

6.3.1 Mission Period

6.3.1.1 Premission phase.- On April 2, 1964, the Manned Space Flight Network went on mission status. All modifications affecting the GT-1 mission had been completed and were operational prior to initiation of the countdown.

6.3.1.2 Countdown phase.- All mandatory and highly desirable network items were available at lift-off. At that time, camera coverage consisted of 14 metric, 39 engineering sequential, and 49 documentary cameras. All of these cameras operated successfully.

Tracking coverage obtained from the long-focal-length cameras was reported immediately after the launch as shown in the following table:

Cameras	Tracking time, sec	Comments
False Cape IGOR	LO + 0 to LO + 175	
Williams Point IGOR	LO + 15 to LO + 200	Intermittent clouds at an altitude of 40,000 feet
Cocoa Beach ROTI	LO + 10 to LO + 170	Momentary loss behind clouds at LO + 30 seconds
Patrick Air Force Base IGOR	LO + 4 to LO + 190	Light clouds up to an altitude of 40,000 feet
Melbourne Beach	LO + 70 to LO + 90 and LO + 100 to LO + 220	

6.3.2 Instrumentation Evaluation

6.3.2.1 Telemetry.- The coverage at all sites was from horizon to horizon with the exception of momentary dropouts caused by spacecraft tumbling. No data were lost except at the beginning and end of the pass where the signal level was at a minimum. Table 6-I gives the telemetry coverage for the mission.

6.3.2.2 Timing system.- The timing systems operated normally at all network stations, and no problems were reported.

6.3.2.3 Acquisition aid.- The performance of the automatically gimballed antenna vectoring equipment (AGAVE) acquisition aid was not satisfactory at most stations. A preliminary investigation indicates the cause to be excessive deviation of the signal transmitted from the spacecraft. The actual signal deviation on all three transmitters was greater than anticipated and was evidently beyond the tolerance of the acquisition aid system. This problem was recognized and more stations were able to acquire the spacecraft signals with minor difficulties.

The following stations reported either poor or no tracking with the AGAVE acquisition aid: Cooper's Island, Bermuda; Eglin Air Force Base, Florida (poor on two of three passes); Texas; Guaymas, Mexico; White Sands Missile Range, New Mexico; Canton Island; and Kano, Nigeria. The difficulty was primarily due to the wide-band characteristics of the transmitted signals.

Teltrack acquisition aids at Canary Islands and Guaymas, Mexico, although not used for the mission, were installed by station personnel in time to support the AGAVE systems. Their operation was superior to that of the AGAVE because of the wide-band capability. Tracking accuracy was not determined since the radars at these stations did not actively support the mission.

6.3.2.4 Radar.- Radar performance during the spacecraft transponder lifetime returns were noted from launch through the fourth orbital pass, after which no attempts were made to track the transponder. The transponder performance was considerably in excess of minimum requirements. (See table 6-II for C-band radar coverage.) As a consequence, radar coverage was adequate in the critical early stages of the mission.

Light to heavy amplitude modulation of the transponder return signals was reported by the network stations as a result of the wobbulator. The variation was probably due to spacecraft aspect-angle changes resulting from tumbling and flight geometry.

Most stations reported better tracking conditions after the point of closest approach (PCA) which also accounted for late signal acquisition at many of the stations.

Preliminary observations of skin tracking returns indicate that the high-power radar units scheduled for participation in orbital coverage have sufficient capabilities to fulfill this requirement.

6.3.2.5 Communications and frequency interference.-

6.3.2.5.1 Voice: The Canary Islands' voice communications were unusable from approximately 5:00 a.m. e.s.t. to 8:00 a.m. e.s.t. due to local commercial carrier equipment problems. For the rest of the mission, this link was usable but noisy at times. The Eglin Air Force Base voice communications failed from approximately 2:17 p.m. e.s.t. to 2:21 p.m. e.s.t. The cause of the failure is undetermined. Carnarvon lost voice communications from 9:23 a.m. e.s.t. until 2:02 p.m. e.s.t.

6.3.2.5.2 Teletype: The Carnarvon station suffered the only significant outage in the ground network. This outage lasted from 9:23 a.m. e.s.t. to 2:02 p.m. e.s.t. and was caused by lightning striking a carrier pole approximately 65 miles south of Carnarvon. Lift-off and acquisition messages were exchanged between Carnarvon and Adelaide through a local amateur radio network.

6.3.2.5.3 Frequency interference: There was no frequency interference to the radio-frequency (RF) links of the ground communications network. Two cases of frequency interference on the telemetry system were reported. Bermuda reported interference on the high-frequency telemetry at 8:10 a.m. e.s.t. and on mid- and low-frequency telemetry at 9:44 a.m. e.s.t.

Interference on mid- and low-frequency telemetry was traced to the Kindley Air Force Base Tower, which was operating on 236.6 mc. Arrangements were made through the Command Post at Kindley Air Force Base to change the Tower operations from 236.6 mc during the launch and subsequent passes of the spacecraft over Bermuda.

Interference on high-frequency telemetry was reported as originating from the direction of Bermuda television station ZBM and NAS Bermuda. ZBM TV was contacted, and the stations transmitters were momentarily secured; however, it was determined that neither ZBM nor NAS Bermuda were the sources of the interference. The interference is believed to originate from the Texas Tower at Bermuda, but the source has not been confirmed.

Clearance was not requested for these frequencies at Bermuda as specified in the premission plan.

California reported interference on low-frequency telemetry at 12:10 p.m. e.s.t., identified as voice transmission from the local Air Defense Command (ADC) station. The interference was cleared locally.

6.3.2.6 Command.- All command functions were performed satisfactorily with no failures occurring during the mission support period.

6.3.2.7 Computers.- During the GT-1 mission, the NASA Goddard Space Flight Center (GSFC) real-time computing system performed its task normally. For the complete mission, the "A" computer was selected as the primary output machine, with the "C" computer providing back-up support.

Only one problem was encountered during the launch phase of the mission. In attempting to switch from the impact predictor (IP) source to the guided-missile computer facility (GMCF) no. 1 source, the computer failed to perform the switch on command. When attempted later in the launch, the switch was successfully commanded. Although this problem was temporary, it was later identified that the time delays in the data used at GSFC produced difficulty in the computer logic. This situation will be rectified for future missions.

The insertion conditions provided by the GMCF indicated a slight overspeed in velocity; however, the Bermuda tracking data corrected the insertion conditions and provided a good orbit determination. When Woomera's tracking data were added to the solution, this orbit determination was confirmed as having an apogee of 173 nautical miles and a perigee of 86.5 nautical miles.

For the rest of the mission, tracking data were excellent and GSFC real-time computing support continued for six orbital passes. As of the fifth orbital pass, the GSFC computers estimated a lifetime of 56 additional orbits.

6.3.2.8 Photographic.- Photographic coverage, including the quantity of instrumentation committed and data obtained during the launch phase by the Atlantic Missile Range (AMR), is shown in table 6-III and discussed in the following paragraphs. Launch-phase photographic coverage was generally good in quality but intermittent in duration due to cloud obscuration and haze conditions in the launch area. Photographic data were obtained through the time of launch-vehicle staging and were available for a detailed photographic evaluation had it been necessary. Table 6-III presents the camera coverage in the launch area. The photographic coverage discussed in the following sections is based on film available for evaluation during the postlaunch reporting period.

6.3.2.8.1 Metric film: Metric film from 14 cameras was processed, and the results were tabulated by the AMR.

6.3.2.8.2 Engineering sequential film: The total engineering sequential coverage obtained from the launch phase is shown in table 6-IV. Locations of the engineering-sequential tracking cameras reviewed for this report are illustrated in figure 6-1. The duration of photographic coverage obtained from the engineering-sequential tracking cameras is illustrated in figure 6-2. The figure indicates the time interval for which the spacecraft, launch vehicle, and/or exhaust flame were visible to the tracking camera. Although photographic coverage was obtained through launch-vehicle staging, coverage near the region of staging is considered marginal due to haze increase and image reduction as a function of increasing slant range. Optimum camera coverage was obtained from lift-off through the region of maximum dynamic pressure.

Twenty-one engineering sequential films were selected for review, including 16mm and 35mm film from 10 fixed cameras and 11 tracking cameras.

Fixed camera coverage with respect to exposure and focus was generally good with the exception of one film underexposed and three films slightly out of focus. Film quality with respect to other camera and film defects was poor. Three films from fixed cameras indicated camera vibration prior to lift-off, one film indicated timing image bleeding into the picture frame, one camera recorded no legible timing, and four films showed film base scratches and emulsion gouges. Three of the fixed cameras indicated smooth umbilical disconnect of the spacecraft umbilicals, and confirmed the early disconnect of the coolant umbilical. Two fixed cameras showed explosive bolt action and launch-vehicle first motion. The five other fixed cameras showed closeup views of spacecraft and launch-vehicle displacement through lift-off.

The quality of the tracking camera coverage was generally good with respect to exposure, focus, and tracking except for two films. Film quality with respect to other camera and film defects was good with the following exceptions: one film with dense timing image, two films with timing image bleeding into picture frame, two films appearing grainy, and two films with emulsion gouges. Six of the tracking cameras showed launch-vehicle ignition and lift-off. Ten tracking cameras tracked through launch-vehicle staging. Tracking cameras located to the north and west of the launch complex were partially obscured by cumulus-cloud coverage, and tracking-camera coverage from the south of the launch complex was generally degraded due to cirrus-cloud obscuration and slant range.

6.3.2.8.3 Documentary film: Documentary film used for engineering evaluation of the launch sequence was provided by 35 mm aerial motion picture film taken by six specially designed aircraft in the launch area. Three aircraft did not acquire track and made no coverage. Two other aircraft acquired track at approximately 35,000 feet and tracked for a short duration; however, coverage was not adequate for evaluation. One aircraft did get very good coverage for a short duration after lift-off, but it lost track approximately 50 seconds after acquisition due to aircraft positioning. With the exception of this film, aerial photographic coverage was not adequate to support a detailed evaluation had it been necessary.

Some of the problems encountered which contributed to the poor performance of the specially designed aircraft requested for aerial photographic surveillance are as follows:

(a) Aircraft flight-path and position problems

(b) Poor acquisition as a result of an inability to see the launch vehicle at lift-off.

(c) Aerial camera manual tracking and buffeting problems

(d) Camera power failure

(e) Partial cloud obscuration

6.3.3 Documentation

The mission documentation and distribution of the documents was adequate in most cases. In the case of the Network Operations Directive 63-1, a serious problem could develop due to the distribution list of the supplement being the same as the distribution list of the basic document. The nature of the 63-1 supplements (network countdown, data acquisition, communications plan, etc.) is such that a widespread distribution is required, whereas there is only a limited short-lived interest in the basic document. All supplements should be separately bound and treated as independent documents made official by their title being included in Network Operations Directive 63-1.

TABLE 6-I.- TELEMETRY COVERAGE

Station	Duration of signal		Coverage, percent	Frequency	Orbital passes
	Acquisition, g.e.t. hr:min:sec	Loss, g.e.t. hr:min:sec			
Cape Kennedy	01:34:26 03:08:37 04:42:25	01:41:21 03:15:38 04:50:02	90 100 100	All All All	1 and 2 2 and 3 3 and 4
Grand Bahama Island	00:00:50 01:36:25 03:10:08 04:43:52	00:07:40 01:41:35 03:15:53 04:50:25	100 98 90 96	All All All All	1 2 3 4
Bermuda	00:04:24 00:04:40 00:04:40 01:37:49 03:11:48	00:11:00 00:11:00 00:11:00 01:44:53 03:18:53	90 90 90 100 100	230.4 246.3 259.7 All All	1 1 1 2 3
Canary Island	00:14:57 01:49:03	00:22:01 01:55:10	98 99	246.3 230.4	1 2
Kano, Nigeria	00:22:40 01:55:51	00:29:07 02:02:52	100 100	All All	1 2
Carnarvon, Australia	00:50:00 02:23:59 03:57:04 05:32:45	00:58:09 02:33:26 04:07:31 05:40:34	99 99 99 99	All All All All	1 2 3 4
Canton Island	1:10:09 1:10:09 1:10:09 2:44:48 2:44:45 2:44:42 2:44:48	1:17:45 1:17:45 1:17:45 2:51:38 2:51:38 2:45:55 2:51:38	95 90 95 95 95 20 95	230.4 246.3 259.7 230.4 246.3 259.7 259.7	1 1 1 2 2 2 2
Guaymas, Mexico	01:27:50 03:02:01 04:36:15	01:34:25 03:08:15 04:42:27	100 90 80	All All All	1 2 3
Texas	01:31:19 03:05:04 04:39:05	01:37:19 03:11:26 04:45:54	100 100 100	259.7 259.7 259.7	1 2 3
Eglin Air Force Base	01:33:11 03:07:13	01:40:00 03:14:09	95 95	All All	1 2
Hawaii	02:50:25 04:23:59	02:56:54 04:31:03	97 98	All All	2 3

TABLE 6-II.- C-BAND RADAR COVERAGE

Station	Duration of signal: Acquisition, g.e.t. hr:min:sec	Duration of signal: Loss, g.e.t. hr:min:sec	Mode
Cape Kennedy	00:00:00	00:06:15	Transponder
Patrick Air Force Base	00:00:00	00:06:06	Transponder
Grand Bahama Island	00:01:05	00:05:52	Transponder
San Salvador Island	00:02:06	00:06:44	Transponder
Bermuda	00:05:13	00:10:48	Transponder
Carnarvon	00:53:12	00:57:43	Transponder/ skin
Woomera	00:58:12	01:01:36	Transponder
California		01:32:14	Transponder
White Sands Missile Range	01:30:29	01:35:49	Transponder
Eglin Air Force Base	01:34:29	01:39:22	Transponder
Cape Kennedy	01:35:59	01:40:59	Transponder
Patrick Air Force Base	01:36:59	01:40:59	Transponder
Grand Bahama Island	01:38:36	01:40:51	Transponder
Bermuda	01:37:53	01:44:45	Transponder
Carnarvon	02:24:33	02:29:57	Transponder/ skin
Woomera	02:32:10		
California		03:05:06	Transponder
White Sands Missile Range	03:04:15	03:10:29	Transponder
Eglin Air Force Base	03:07:31	03:12:58	Transponder
Patrick Air Force Base	03:10:20	03:15:30	Transponder
Cape Kennedy	03:10:31	03:15:26	Transponder
Grand Bahama Island	03:12:13	03:15:39	Transponder
San Salvador Island			
Bermuda	03:11:59	03:18:42	Transponder
Carnarvon	03:58:22	04:05:11	Transponder/ skin
California			
White Sands Missile Range	04:37:11	04:40:17	Transponder
Eglin Air Force Base	04:41:10	04:48:00	Transponder
Patrick Air Force Base	04:43:16	04:49:41	Transponder/ skin
Cape Kennedy	04:42:56	04:49:36	Transponder
Grand Bahama Island	04:44:48	04:49:45	Transponder
San Salvador Island	04:46:35	04:51:35	Transponder
Antigua Island	04:50:14	04:53:32	Transponder
Carnarvon	05:33:43	05:40:29	Transponder/ skin
California			
White Sands Missile Range	06:11:43	06:16:24	Transponder

TABLE 6-III.- OPTICAL LAUNCH COVERAGE AT THE ATLANTIC MISSILE RANGE

Film type	Station	Number of items committed	Number of items obtained	Comments
Metric	1	14	14	
Engineering sequential	1	39	39	(1) Coverage degraded resulting from 10-percent cumulus and 100-percent cirrus cloud obscuration (2) Engineering sequential tracking camera coverage is shown in figure 6-1.
Documentary		49	49	

TABLE 6-IV.- AMR ENGINEERING SEQUENTIAL CAMERA DATA

Sequential film coverage (a)	Camera		
	Type	Location	Usage
Item 1.2-1	Fixed	Launch area	GLV-spacecraft and launch stand surveillance
Item 1.2-2	Fixed	Launch area	GLV-spacecraft and launch stand surveillance
Item 1.2-3	Fixed	Launch area	GLV-spacecraft and launch stand surveillance
Item 1.2-4	Fixed	Launch area	GLV-spacecraft and launch stand surveillance
Item 1.2-5	Fixed	Launch area	GLV-spacecraft and launch stand surveillance
Item 1.2-6	Fixed	Launch area	GLV-spacecraft and launch stand surveillance
Item 1.2-7	Fixed	Launch area	Possible leakage or spillage of fuel storage tank
Item 1.2-8	Fixed	Launch area	Possible leakage or spillage of oxidizer storage tank
Item 1.2-9	Fixed	Launch area	GLV-spacecraft movement and vibration at lift-off
Item 1.2-10	Fixed	Launch area	GLV-spacecraft movement and vibration at lift-off
Item 1.2-11	Fixed	Launch area	GLV-spacecraft movement and vibration at lift-off

[a] See reference 8.

TABLE 6-IV.- AMR ENGINEERING SEQUENTIAL CAMERA DATA - Continued

Sequential film coverage (a)	Camera		
	Type	Location	Usage
Item 1.2-12	Fixed	Launch area	Spacecraft observation at lift-off
Item 1.2-13	Fixed	Launch area	Spacecraft observation at lift-off
Item 1.2-14	Fixed	Launch area	Explosive bolt action and first motion of GLV
Item 1.2-15	Fixed	Launch area	Explosive bolt action and first motion of GLV
Item 1.2-16	Fixed	Launch stand (East)	Possible leakage or spillage of fuel
Item 1.2-17	Fixed	Launch stand (West)	Possible leakage or spillage of fuel
Item 1.2-18	Fixed	Launch stand (North)	Thrust chamber surveillance
Item 1.2-19	Fixed	Launch stand (South)	Thrust chamber surveillance
Item 1.2-20	Fixed	Umbilical tower, first level	3D1E and 3D2E launch-vehicle umbilical plug disconnect
Item 1.2-21	Fixed	Umbilical tower, second level	2DFVT launch-vehicle umbilical plug disconnect
Item 1.2-22	Fixed	Umbilical tower, fourth level	1DOVT umbilical plug disconnect

[a]See reference 8.

TABLE 6-IV.- AMR ENGINEERING SEQUENTIAL CAMERA DATA - Continued

Sequential film coverage (a)	Camera		
	Type	Location	Usage
Item 1.2-23	Fixed	Umbilical tower, fifth level	3B1E and associated umbilical plugs disconnect
Item 1.2-24	Fixed	Umbilical tower, sixth level	2B1E and 2B2E umbilical plugs disconnect
Item 1.2-25	Fixed	Umbilical tower, top level 1	Spacecraft top umbilical plugs disconnect
Item 1.2-26	Fixed	Umbilical tower, top level 2	Spacecraft lower umbilical plugs disconnect
Item 1.2-27	Tracking	Cape Kennedy	Spacecraft flight coverage
Item 1.2-28	Tracking	Cape Kennedy	Spacecraft flight coverage
Item 1.2-29	Tracking	Cape Kennedy	GLV-spacecraft flight coverage
Item 1.2-30	Tracking	Cape Kennedy	GLV-spacecraft flight coverage
Item 1.2-31	Tracking	Cape Kennedy	Engine section through GLV-spacecraft flight coverage
Item 1.2-32	Tracking	Cape Kennedy	Engine section through GLV-spacecraft flight coverage
Item 1.2-33	Tracking (IGOR)	False Cape	Engine section through GLV-spacecraft flight coverage
Item 1.2-34	Tracking (IGOR)	Williams Point	Engine section through GLV-spacecraft flight coverage

[a]See reference 8.

TABLE 6-IV.- AMR ENGINEERING SEQUENTIAL CAMERA DATA - Concluded

Sequential film coverage (a)	Camera		
	Type	Location	Usage
Item 1.2-35	Tracking (ROTI)	Cocoa Beach	Engine section through GLV-spacecraft flight coverage
Item 1.2-36	Tracking (IGOR)	Patrick Air Force Base	Engine section through GLV-spacecraft flight coverage
Item 1.2-37	Tracking (ROTI)	Melbourne Beach	Engine section through GLV-spacecraft flight coverage
Item 1.2-38	Tracking	Cape Kennedy	GLV-spacecraft flight coverage
Item 1.2-39	Fixed	Umbilical tower, sixth level	Umbilical cable cutter action

[a]See reference 8.

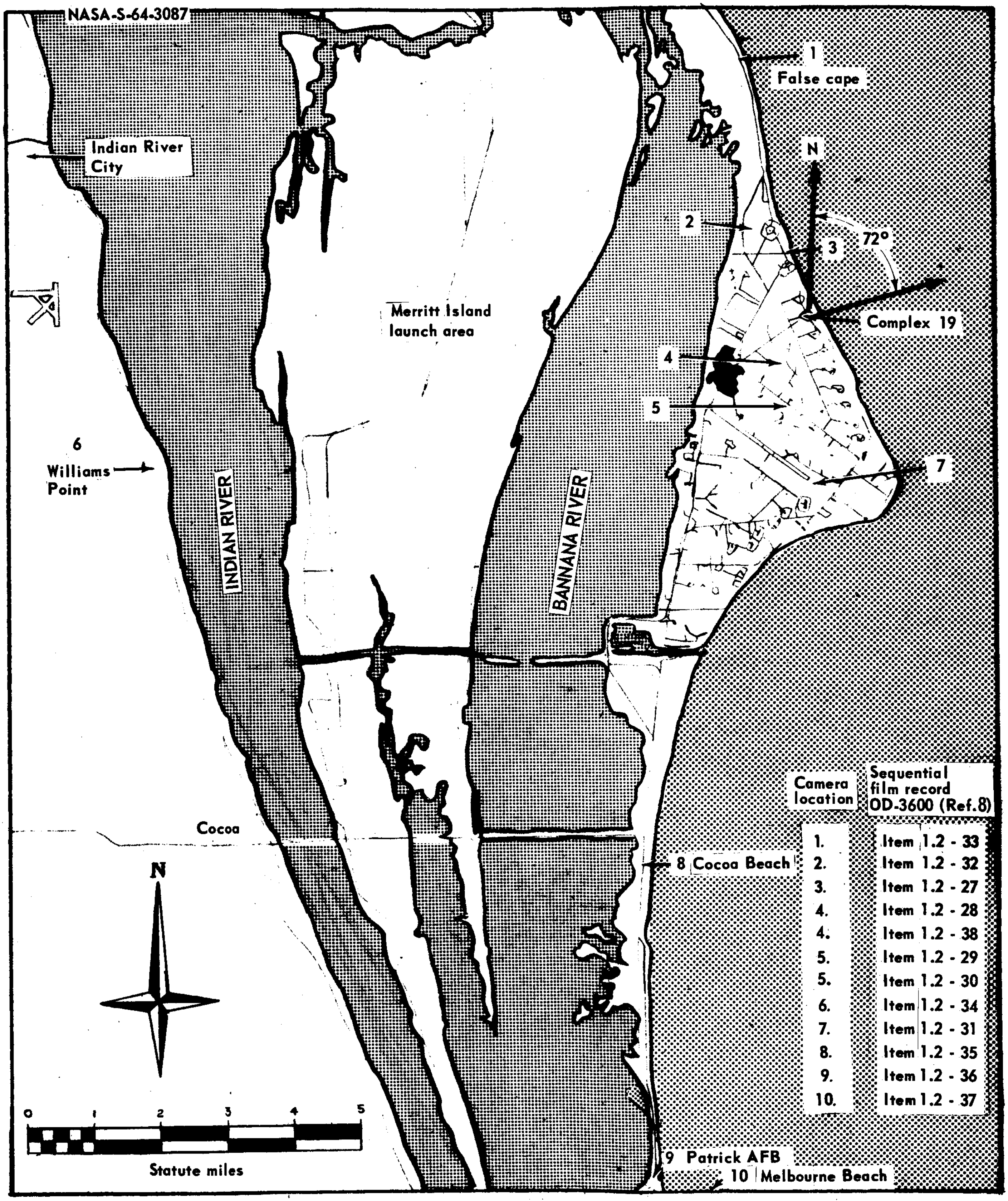


FIGURE 6-1 AMR ENGINEERING SEQUENTIAL TRACKING CAMERA LOCATIONS FOR GT-1

NASA-S-64-3086

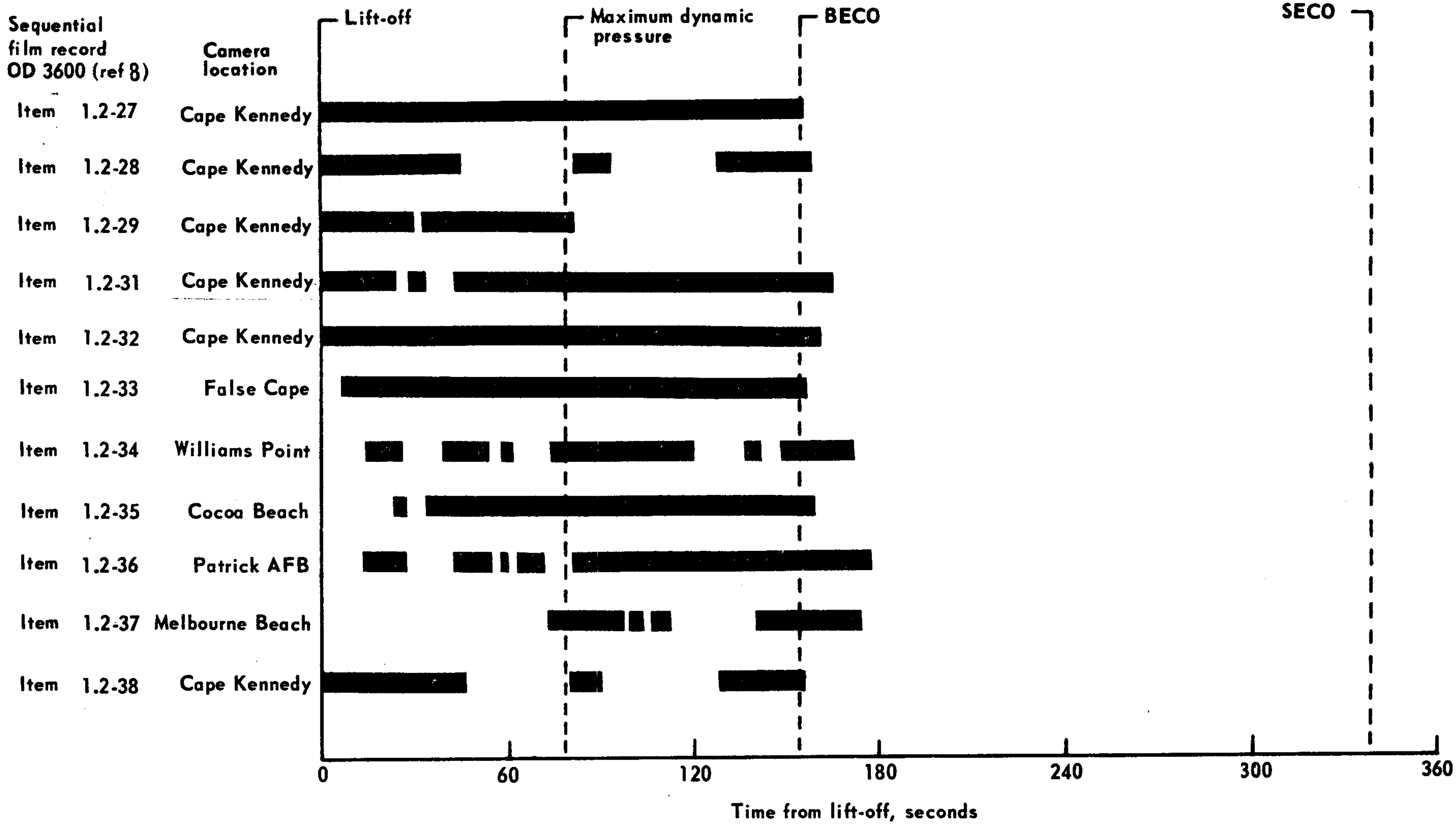


FIGURE 6-2 AMR ENGINEERING SEQUENTIAL TRACKING CAMERA COVERAGE

SECTION 7.0

THIS SECTION NOT APPLICABLE TO THIS REPORT.

SECTION 8.0

THIS SECTION NOT APPLICABLE TO THIS REPORT.

9.0 CONCLUSIONS

(a) The GT-1 mission demonstrated the flight compatibility of the Gemini spacecraft and the Gemini launch vehicle.

(b) The structure of the launch vehicle was adequate for the environment encountered, and its basic dynamic characteristics, as demonstrated in flight, agreed well with the predicted values for the maximum-load flight regime.

(c) The longitudinal oscillation which occurred during the flight was of low amplitude and well within the established limits. A slight build-up of the oscillation which was measured in the spacecraft just prior to staging is not considered significant.

(d) The malfunction detection system in the launch vehicle performed properly and would have supplied the correct information to the monitoring instruments on the instrument panel in the spacecraft.

(e) The launch trajectory, insertion into orbit, and the sequence of events were performed as planned, within the acceptable limits.

(f) The insertion into orbit was accomplished with approximately 2σ dispersions in velocity and flight path angle. The errors in yaw velocity and flight path angle are explainable by a shifting pitch and yaw attitude displacement which could have resulted from thrust misalinement of stage II. Further investigation of this problem is required. The error in velocity along the flight path has not been explained. Despite a guidance cut-off at 15 ft/sec below the required velocity, the final velocity achieved was 20 ft/sec in excess of the proper orbital velocity. Tail-off of the thrust from the stage II engine was close to the nominal value when measured from actual engine cut-off. An investigation of this situation is required.

(g) The GT-1 mission demonstrated the first active guidance of a launch vehicle by the Mod III radio guidance system (RGS) at an elevation angle below 7.0°. The noise in the radar measurement system up to SECO was of a low order of magnitude and did not contribute any significant guidance errors at insertion into orbit. The Mod III data were also used for real-time trajectory monitoring, and the data after SECO were used for the orbital go—no-go decision. The data were of proper quality to determine that an acceptable orbit was achieved. Except for atmospheric noise phenomena, which could occur on future missions and produce an increase in low-elevation-angle radar noise, the Mod III radar system should continue to demonstrate acceptable performance throughout the Gemini program.

(h) There was no switchover of the flight control system to the secondary system. The secondary system was operating properly.

(i) The propulsion system operated within its specification limits and gave nominal thrust and specific impulse. There was evidence of a transient in a number of telemetry traces at 11 seconds after stage II engine cut-off which could have been due to activity in the engine system. This problem also requires investigation.

(j) The temperatures measured on the launch vehicle in the region of the insulation on the forward adapter were lower than predicted. A study should be made to determine if the deletion of some insulation is possible.

(k) The MISTRAM II station at Eleuthera did not give satisfactory performance for range safety purposes or for use as a back up to the Mod III RGS for the go-no-go decision at insertion.

(l) The structure of the spacecraft maintained its integrity under the environment encountered on this flight. The severe local loading of the adapter rings which was predicted, due to the location of the environmental control system (ECS) pump package, did not materialize. The dynamic modal analysis of the structure was verified. The design thermal environment for the launch phase was verified. A complete analysis of the flight data should be conducted.

(m) During earth orbit there were definite indications that small objects became detached from the main vehicle, but this loss did not appreciably affect the size and weight of the main vehicle. There was no evidence in the flight data to indicate any failure of the spacecraft structure at any time during the first three orbital passes of the flight. Further investigation of this area is required.

(n) The cabin-pressure relief valve operated properly, and cabin pressures were maintained at an acceptable level throughout three orbital passes.

(o) Since minimum temperature data from only one station were available for evaluation of the orbital heating characteristics of the spacecraft during the report preparation period, a further evaluation must be conducted to correlate these data with data recorded at the remote stations.

10.0 RECOMMENDATIONS

(a) It is recommended that studies be performed and reports prepared on the following subjects:

Subject	Responsible organization	Completion date
Multiple objects in orbit	Gemini launch vehicle contractor with spacecraft contractor support through Manned Spacecraft Center	May 12, 1964
Post SECO transient	Gemini launch vehicle contractor	30 days after date of this report
Stage II thrust alinement	Gemini launch vehicle contractor	30 days after date of this report
Deletion of insulation from launch-vehicle adapter	Gemini launch vehicle contractor	30 days after date of this report
Detailed analysis of spacecraft structural loads	Spacecraft contractor	60 days after date of this report
Detailed analysis of spacecraft thermal loads	Spacecraft contractor	60 days after date of this report
Guidance system insertion accuracy	Aerospace	30 days after date of this report
Excessive pressure developed in secondary hydraulic system at engine ignition	Gemini launch vehicle contractor	30 days after date of this report

(b) Discussions should be instituted with the Range Safety Officer and Headquarters of the Atlantic Missile Range concerning the MISTRAM II station, and the action required to achieve adequate performance.

(c) The performance of the open-loop-monitored low-level sensors should be evaluated carefully to determine their feasability and a determination should be made about their further use on the Gemini launch vehicle.

(d) A close analysis should be made of temperature full-scale ranges for future flights so that more of the usable instrumentation range can be utilized.

(e) Due to the changing thrust misalinement in stage II and the resulting changes in vehicle attitude, it is recommended that this interaction with the RGS be fully investigated. This investigation should include the effects of reducing the smoothing of the radar data to determine if such changes in the vehicle attitude can be more rapidly and accurately determined without undue degradation in the guidance accuracy caused by radar noise.

(f) A discrepancy exists in the results of the evaluation of stage II engine thrust tail-off after SECO between the radar tracking data and the tail-off thrust energy calculated using telemetered accelerometer information. A detailed analysis should be made of this difference and the differences should be resolved.

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11.0 REFERENCES

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7. McDonnell Aircraft Corp.: Gemini Lateral and Longitudinal Vibration Investigations. Report no. A123 (Contract no. NAS 9-170), Sept. 17, 1963.

8. Air Force Missile Test Center: Project Gemini GLV/SC Launch. Operations Directive no. 3600, Nov. 15, 1963.

9. McDonnell Aircraft Corp.: Project Gemini Test Operations Production Spacecraft at Contractor Facility (Spacecraft No. 1). SEDR 308-1, May 20, 1963 (rev. Aug. 15, 1963).

12.0 APPENDIX A

12.1 VEHICLE HISTORIES

12.1.1 Gemini Spacecraft

Gemini spacecraft 1 checkout was accomplished in the following three phases:

(a) Spacecraft systems tests (SST) at the spacecraft contractor's facility in St. Louis.

(b) "Hangar" tests in the industrial area at Cape Kennedy.

(c) Tests at launch complex 19 at Cape Kennedy.

In this section, the chronology of phases (b) and (c) are discussed. The first phase of systems testing (SST) which was performed in St. Louis is described in detail in reference 9 and is not discussed in this section. However, the checkout chronology at the contractor's facility and a listing of significant events is presented in figure 12-1. Eighty-seven working days were required to complete systems testing. Manufacturing updating, trouble-shooting of the aerospace ground equipment (AGE), and correction of test procedures accounted for significant portions of the 87 days. In general, the spacecraft components performed well and to specification.

Gemini spacecraft 1 was delivered to Cape Kennedy on October 4, 1963. After a receiving inspection and a voltage standing wave ratio (VSWR) check, the instrumentation pallets were removed and delivered to the instrumentation laboratory for checkout. Instrument component checks and pallet instrument tests were performed in the laboratory while work on the spacecraft structure was being conducted in Hanger AF. The pallets were reinstalled in the spacecraft on November 26, 1963. Individual and integrated communications, instrumentation, and environmental control system (only cabin-pressure relief valve installed) tests were then performed. The schedule of the tests performed is shown in figure 12-2. Final industrial area testing of the spacecraft was completed with the performance of the confidence level test (SEDR H431-1) on February 12, 1964.

Following completion of the confidence level test, industrial area spacecraft work, which included hatch rigging checks, drogue parachute attachment point rework, sound-pressure-level instrumentation checks, beryllium-shingle fitting, etc., continued until March 3, 1964.

The spacecraft was transferred to complex 19 and placed on the spacecraft erector support assembly in the erector white room on March 3, 1964. Premating systems tests were successfully performed on March 4. Mechanical mating of the spacecraft with the Gemini launch vehicle (GLV) was performed on March 5.

The spacecraft was demated from and remated to the GLV on March 9 for the replacement of a pressure transducer and a temperature pickup on the GLV oxidizer-tank dome. The spacecraft and GLV were again demated on April 3 to install an accelerometer near the simulated ECS pump package module in the adapter.

A more detailed history of the spacecraft is presented in the following subsections.

12.1.1.1 Instrumentation.- On arrival of the spacecraft at Cape Kennedy, the instrumentation pallets were removed and taken to the instrumentation laboratory for individual component checkout. The sound-pressure-level instrumentation was sent to the White Sands Missile Range for checkout and calibration. After individual component tests, pallet communication test (SEDR K341-1), pallet coolant test (SEDR K381-1), and pallet instrumentation test (SEDR K321-1) were performed. The instrumentation pallets were then weighed and replaced in the spacecraft.

An integrated system test (SEDR H430-1) was run after the pallets were installed in the spacecraft. During this test, an insufficient output was noted for the sound-pressure-level (SPL) system 2 (high-level system). The SPL system was removed from the spacecraft for bench checking, and the problem, which was found to be insufficient amplifier gain, was corrected. While the SPL system was removed from the spacecraft, the SPL filter system covers were modified. This modification was made to facilitate the removal of these covers for filter adjustment without removing the system from the cold plates.

The acoustic calibrator became inoperative during bench checking of the SPL system. The unit was repaired, but its output characteristics were modified and its calibration was invalidated. A spare microphone, calibrated at White Sands Missile Range (WSMR), was used in determining the new characteristics of the calibrator. In subsequent checks of the SPL system, an apparent drift in output of the SPL system was noted, and periodic checks of the system were made to monitor the indicated drift. A new acoustic calibrator which was less susceptible to background noise was obtained from the John F. Kennedy Space Center (KSC). The new acoustic calibrator was calibrated against a standard provided by KSC which was based on a standard from the National Bureau of Standards. The SPL was then calibrated with the new calibrator and subsequent checks of the SPL system indicated the drift to be minor since all check points fell within 0.5 db of this calibration.

Shortly after the spacecraft arrived at Cape Kennedy, eight accelerometers were removed and sent back to the vendor to insure that the locking sealer had been applied to the internal adjustment and mounting screws. The units were returned to Cape Kennedy in January of 1964. After a subsequent failure of a similar accelerometer during vibration testing at the contractor's facility at St. Louis, Missouri, the accelerometers were removed again and rechecked by contractor representatives at Cape Kennedy. After the accelerometers were checked and necessary sealing was added to internal screws, the accelerometers were recalibrated and reinstalled in the spacecraft. Three accelerometers had to be replaced in the process. One was replaced due to the extent of disassembly, another, due to a broken ground line between the sensor assembly and the filter assembly, and the third, due to out-of-specification sensitivity to accelerations perpendicular to the measuring axis of the instrument.

The following configuration changes were made in the spacecraft accelerometer instrumentation at Cape Kennedy. Accelerometers QC22 and QC23 were changed from ±16g to ±100g, and QA09 and QA10 were changed from ±4g to ±2g. The accelerometer assigned to QC25 was removed from the spacecraft and a ±32g accelerometer, designated QC26, was installed in the area near the simulated ECS pump package. Accelerometer QB14 (in left-hand equipment bay) was moved approximately 1.5 inches to agree with location of a reference accelerometer installed in spacecraft 2. The amplifiers for accelerometers QC17, QC22, and QC23 were relocated to one of the retrorocket beams in order to place them in a region in which they would not be subjected to an acceleration greater than that for which they were rated (20g).

An extra pair of break wires was added to the horizon-scanner cover, each electrically in parallel with one of the pair of break wires installed previously. The extra pair of wires was added for reliability after sled tests indicated a tendency for these wires to break prematurely.

Temperature measurements PC03 and PC07 were found to be reversed in location on the spacecraft, and due to cable lengths to the sensors, were left in these locations. Temperature sensor PF15 was noted to be inoperative 2 days prior to the simulated flight, but it was not repaired because of the delay in schedule which would have resulted.

12.1.1.2 Communications.- The C-band transponder was replaced twice at Cape Kennedy. The first transponder, which carried part no. 52-85707-7, was out of specification in the center frequency, power input, and radio-frequency output. This transponder was replaced with another, part no. 52-85707-9, which incorporated an improved capacitor network. Later information indicated that a resistor in a transponder of the same type had failed after heat soaking. Therefore, the spacecraft transponder was replaced with one incorporating resistors that had been checked for heat sensitivity.

Trouble was encountered in obtaining proper VSWR values and power division for the C-band transponder antenna system after the pallets were installed in the spacecraft. The power divider proved to have an internal fault (solder void) and was replaced with a unit that had been X-rayed and vibrated. Proper power distribution values were not obtained until the individual antenna cables were trimmed to the proper length. Information resulting from spacecraft 2 tests and inspection at the contractor's facility indicated that the coaxial cable connections in the transponder system may not have been properly assembled. All coaxial connections in the system in spacecraft 1, therefore, were disassembled and reworked to proper specification. (All telemetry coaxial connections were disassembled and reworked in the same manner).

The three TM transmitters on board the spacecraft were removed and replaced with production units.

The quadriplexer was removed from the spacecraft and replaced with a later version that incorporated a means for visual inspection of a suspect solder connection during manufacture of the unit.

During integrated GLV and spacecraft tests at the launch complex, spurious outputs from the GLV TM transmitters were noted. The spurious signals were examined, and it was determined that they would not interfere with the TM signal reception for this spacecraft because of their low signal strength and frequency.

12.1.1.3 Structural and mechanical.- Considerable effort was expended in obtaining a proper fitting of the spacecraft shingles. Final acceptable fitting of the beryllium shingles was achieved after the spacecraft was mated with the GLV. Four beryllium shingles were returned to the vendor for chemical milling to remove "crazing" (small shallow scratches or cracks) from the outer surface of the shingles. Information was received that any machining (such as drilling holes for a thermocouple installation) of beryllium shingles required chemical milling to relieve stress on the affected areas. This small-area chemical milling was done at Cape Kennedy. Three beryllium shingle retainers on the rendezvous and recovery (R and R) section of the spacecraft were redesigned to obtain proper contour fit at the transition from the conical to the cylindrical section.

One of the macron inserts supporting the ECS cryogenic oxygen bottle failed during the SST vibration of the spacecraft at the spacecraft contractor's facility. The failure appeared to have been caused by a relative rotation of the two halves of the insert with respect to each other. The insert was modified with a locking pin to prevent this rotation. During the subsequent vibration of a test specimen to verify the integrity of the modified installation, one of the welds of a sheet-metal attachment pad also supporting the oxygen bottle failed. All of the welded attachment pads were replaced with machined pads. During the next vibration of the test specimen, the macron insert mentioned earlier was discovered to have torn loose some of the honeycomb to which it was attached. The corrective action for this failure was to add a doubler to the lower face of the honeycomb at the insert. The test specimen was again vibrated. No further failures occurred. These changes were all incorporated into the spacecraft. The parts were fabricated in the spacecraft contractor's facility and flown to Cape Kennedy for installation.

The rigging of the spacecraft hatches was checked at Cape Kennedy. The rigging was found to be satisfactory; but in the process of checking, three parts were found to be broken (a latch pivot support, a positioner, and a handle hub housing) and were replaced.

The dummy shaped charge on the Z16 ring (spacecraft-GLV mating ring) was removed to add 16 rivets to the mating ring-adapter attachment rivet pattern to match the configuration tested at the contractor's facility at St. Louis, Missouri. Corrosion was found on the ring beneath the shaped charge.

The corroded area was cleaned, and the depths of the corrosion pits were measured and were found to be a maximum of 0.020 inch. A coating was applied to the ring before replacing the shaped charge to prevent further corrosive action between the shaped charge and the ring.

Other changes made to the spacecraft and work performed at Cape Kennedy are as follows:

(a) Minor modifications were made to both electrical umbilicals to provide break-away features.

(b) The drogue parachute cover attachment clips were redesigned to prevent breaking.

(c) The reentry assembly umbilical was rotated 180°to match the configuration on the gantry.

(d) An inert shaped charge was added at the separation line of the retrograde section and the equipment section.

(e) Four 1-foot-diameter counter bores were cut into the heat shield (through the ablative material only) to assure that the spacecraft would be destroyed during reentry.

(f) The horizon-scanner cover was reworked to provide better fit at the transition from the cylindrical to the conical section on the spacecraft.

(g) A portion of the metal strip was removed from the edge of the horizon-scanner cover to prevent interference with the C-band antenna, which was partially covered by the horizon-scanner cover.

(h) Seven of the coolant tubes were uncapped to inspect for corrosion; no corrosion was found.

(i) A fairing was installed (over water evaporator outlet) on the adapter to provide a specification configuration.

(j) The structural rings in the adapter were X-rayed in a search for possible cracks; no cracks were found.

(k) A special test was performed after the spacecraft was mated with the GLV to verify that the air leakage rate from the adapter was within allowable limits.

(l) The coolant system relief valve was permanently removed from the system (valve leaked) since an AGE valve served the purpose in ground tests, and the system was vented to ambient air in flight.

(m) The coolant-line quick disconnects in the spacecraft leaked during the confidence level tests (SEDR H430-1) and were replaced with bulkhead fittings.

(n) An extra layer of insulation was added to the top of the R and R section nose cover for exit heating protection.

12.1.2 Gemini Launch Vehicle

12.1.2.1 Gemini launch vehicle pre-Atlantic Missile Range (AMR).- Major weld fabrication on the Gemini launch vehicle (GLV) 1 (Contractor Serial No. B001, Air Force Serial No. 62-12556) was begun at the contractor facility in Denver, Colorado, in September 1962. All tanks were subjected to visual dye checks, radiographic inspection, and hydrostatic testing. The hydrostatic tests consisted of two cycles at 1.10 times the limit load.

The GLV-1 tanks were received by the contractor's Baltimore facility on October 10, 1962. The tanks were nitrogen purged and X-rayed to support stress analysis. Eddy current checks were run to verify heat affected zone isolation.

A Tank Roll-Out Meeting occurred on February 14 to 16, 1963, at which the contractor reported that cracks had been found in stage II tanks scheduled for GLV-2. Improved ultrasonic techniques were employed to reevaluate the GLV-1 stage II tanks, and a crack was found in the stage II oxidizer tank. This tank was rejected and returned to the contractor's Denver facility; the GLV-2 stage II oxidizer tank was used as the replacement. Delivery to the contractor's Baltimore facility was completed on March 1, 1963.

12.1.2.1.1 Operations at Baltimore, Maryland: Type "E" (dummy) engines were obtained on March 18, 1963, for use in developing tubing and wire runs. The "E" engines were removed, and flight engines were installed in stage II on May 7, 1963, and in stage I on May 17, 1963. Due to configuration differences between the flight engines and the "E" engines, five tubing runs had to be redeveloped. Installations were completed by May 21. Wiring continuity checks followed and were complete on May 25, 1963. During the installation of wiring clamps and the subsequent inspection, it was noted that clamps were damaged and in some cases sharp clamp edges were cutting their rubber coating with possible damage to wiring. All clamps were later removed from GLV-1 (in the vertical test facility) and replaced with approved parts.

Stage I was erected in the vertical test facility (VTF) on June 2, 1963, and stage II on June 9, 1963. Post-erection inspection was completed on June 12.

Subsystem functional verification tests (SSFVT) began on June 10, 1963. It should be noted that the contractor's in-plant operations are different from the operations at Cape Kennedy. During the in-plant operation, each system is separately controlled, and no automatic sequencing is employed for combined systems testing. Two of the systems are not fully tested in-plant: (a) propulsion, and (b) electrical (different power source configuration). These differences are not detrimental to satisfactory checkout for delivery to the launch site.

The SSFVT data were reviewed by the Air Force Space Systems Division (SSD) and Aerospace prior to beginning the electronic-electrical interference (EEI) tests on July 31, 1963. A total of 58 channels were monitored, 50 by oscillograph and 8 by oscilloscopes employing 35mm cameras for recording. On August 2, 1963, 20 channels were rerun to obtain additional data. The first dry combined systems acceptance test (CSAT) was performed on August 2 and was rerun on August 17.

EEI tests were performed prior to and concurrent with the CSAT testing. The additions of filters and grounds to the aerospace ground equipment (AGE) and airborne circuits resulted, and the final results of the test proved that the GLV was free of major EEI problems. EEI data were again reviewed by SSD and Aerospace. Another CSAT with EEI monitoring was performed on September 3 to clarify checkout procedures and recheck EEI results. During this period, umbilical drop tests were also performed.

The first official CSAT was run on September 6, 1963. The subsequent Vehicle Acceptance Team review disclosed the need for further testing or verification. In particular, documentation covering failure analysis and qualification of airborne hardware had not been completed. Finalization of outstanding items, closely coordinated by SSD and Aerospace, led to a preliminary CSAT on October 2, 1963, and the final official CSAT on October 4, 1963.

The Vehicle Acceptance Team critique was held on October 12, 1963, and the decision to ship the GLV-1 to AMR was announced. In the critique summation, awareness of the logistics problems concerning this first GLV was expressed. The large list of work to be done, principally due to delays from vendors of flight qualified units, was noted. It was also noted that the availability of the GLV at AMR, even with ground-test only units aboard, would expedite the program by permitting early vehicle launch-complex checkout for compatibility and final acceptance of complex 19.

At the conclusion of the Vehicle Acceptance Team critique, the following SSD policies were defined:

(1) Waivers permitted for GLV-1 will not be considered for GLV-2.

(2) Reasons for operator errors during CSAT must be investigated.

(3) Failure analyses must be monitered and expedited in a better fashion.

12.1.2.2 <u>Gemini launch vehicle at the Atlantic Missile Range</u>.-

12.1.2.2.1 Arrival and erection: This section contains a brief history of the GLV-1 from its arrival at AMR through the completion of launch preparations and launch. A detailed milestone schedule, showing the scheduled dates and actual accomplishment of the milestones, is presented in figure 12-3. This schedule, which is the working schedule used by the Gemini

Launch Vehicle Working Group, is referenced in the discussion to follow. This history of the GLV-1 at AMR describes the test sequence of events with reasons for schedule changes and denotes the hardware problems encountered during the tests and the resolutions of these problems.

The GLV-1 arrived at AMR on October 26, 1963. Following erection of stage I in the complete vehicle erector (CVE) and stage II in the second stage erector (SSE), cabling between the two stages was completed to establish the sequenced compatibility firing (SCF) configuration.

Prior to the application of power to the GLV flight systems, a limited EEI test was performed. Power was applied on November 13, 1963.

12.1.2.2.2 GLV testing at AMR: The combined systems test (CST) for SCF originally scheduled for December 13, 1963, was completed on December 31. The primary cause for the delay was incurred in completing the complex support systems for operational compatibility with the launch vehicle. A further delay resulted when the stage II turbo pump assembly (TPA) had to be returned to the contractor in Sacramento, California, for repair and recheck. The prerequisite testing was completed for a CST on December 31, 1963. Subsequently, later scheduled milestones changed.

Propellants were loaded on January 3, 1964, to accomplish the scheduled wet mock simulated flight test (WMSFT). Procedural errors encountered during the propulsion system tests and the flight control system tests made it necessary to discontinue the WMSFT at T-150 minutes. The count was continued to T-30 minutes to highlight any further problem areas and to provide training for the launch crew. The WMSFT was rescheduled for January 7, 1964, and was performed successfully on that date. The SCF scheduled on January 10, 1964, was discontinued at T-20 minutes due to an engineering uncertainty concerning the position of a prevalve actuator. The prevalves used for the SCF are for ground test only and incorporate an actuator mechanism to provide a back-up means to shut down the engines. The first-stage fuel prevalve closing mechanism was found to be out of position and would have prevented the valve from opening.

The prevalves were removed, adjusted, and reinstalled for a second SCF test on January 14. This test was canceled because the abnormally cold ambient conditions decreased the temperature of the start cartridge to a questionable flow rate level. A subsequent investigation of the flow rate as a function of ambient temperature resulted in raising the lower temperature-limit criteria from 35° F to 65° F. The SCF was rescheduled for January 21, 1964, and was successfully completed on that date (see fig. 12-4). The countdown, following a $2\frac{1}{2}$ hour delay, was performed with only an 8-minute hold to permit an instrumentation repatching to guarantee several blockhouse readings. The $2\frac{1}{2}$ hour delay before picking up the count was due to (a) a stage I fuel totalizer malfunction which necessitated tanking both stage I and II using the stage II totalizer sequentially, (b) a diesel-generator malfunction during precount operations, and (c) the requirement to inspect filters in the propellant lines prior to tanking. Because of the diesel-generator malfunction, the decision was made to use industrial power as the primary source to power the complex during launch operations.

Following the SCF, stage II erection in tandem was scheduled for January 27. However, during the post-SCF cleaning of the subassembly 3 turbo pump, the turbo rotor was found to be defective because of a separation at the base of one bucket. The TPA was shipped to Sacramento, California, for rotor replacement and dynamic balancing of the assembly. The turbine assembly was returned to AMR on January 29, but installation was delayed pending the receipt of a new seal running ring. The tandem erection was accomplished on January 31; the second-stage turbo pump reinstallation was completed on February 7.

During the period of January 23 through March 31, airborne electrical and mechanical modifications were in progress. The ground test only equipment on the GLV were replaced with flight units before beginning the individual subsystem functional verifications leading to launch. The EEI test equipment was installed to be used concurrently while the SSFVT was in progress. The low-frequency longitudinal accelerations surge tanks and related equipment were installed during this period.

The spacecraft structural simulator was erected on February 5, and the spacecraft umbilical drop weight tests were completed on February 14.

Subsystems functional verifications began on February 21.

The stage II TPA was returned to Sacramento, California, on March 2 for resolution of excessive rotor friction. The rotor was replaced and the TPA was returned on March 7 and reinstalled on March 8, 1964. Meanwhile, the spacecraft was mechanically mated with the GLV on March 5. Before the electrical mating with the spacecraft, a launch-vehicle reverification CST was performed on March 10.

On March 9 the spacecraft-GLV combination was demated to permit replacement of two launch vehicle transducers. One measurement, the gas pressure in the stage II oxidizer tank (PGOT2), was a mandatory landline transducer for launch and necessitated the work; the other measurement, one of the five skin temperature transducers, had failed previously and was replaced at this time.

The acceptance testing and calibration of the EEI test vans was completed on March 11, and EEI testing began on March 12. EEI test III was delayed when saturation-level noise appeared on two flight-control valve drive amplifiers during run 2 of EEI test II. This anomaly was resolved by adding low-pass filters in the inputs to the EEI recorders on all seven flight-control valve-drive amplifier measurements. Trouble shooting was accomplished by repeated runs of EEI test III and ended on March 25. Evaluation on March 26 of the EEI test III trouble-shooting runs and an abbreviated test with the fix (low-pass filters) resulted in an agreement that the intent of EEI testing had been accomplished. The EEI equipment was removed and a successful post-EEI systems reverification (CST) was performed on March 27.

The WMSL scheduled for April 1 was delayed to April 2 due to a malfunction in the terminal-room electrical switching system. The trans-o-matic switchover circuitry was restored, and the WMSL was completed on April 2.

On April 3, the propulsion system prevalves were changed to the flight articles. At this time, the spacecraft-GLV combination was demated again to relocate an accelerometer in the spacecraft. The simulated flight test/combined systems test (SFT/CST) was performed on Sunday, April 5.

12.1.2.2.3 Review meetings: The following management meetings were held to determine readiness of the GLV:

April 4 - Launch Vehicle Flight Readiness Review

April 6 - NASA Mission Review

April 7 - Status Review Team

The Status Review Team recommendation to commit the GLV-1 to launch was presented to the Flight Safety Review Board at 9:00 a.m. e.s.t. on April 7, 1964. The Flight Safety Review Board committed GLV-1 to launch with a scheduled lift-off time of 11:00 a.m. e.s.t. on April 8, 1964. The count-down was resumed on schedule and continued with no holds to a successful launch at 11:00:01.69 a.m. e.s.t.

12.2 WEATHER CONDITIONS

Weather conditions in the launch area were satisfactory for operations several days prior to and on the day of launch. On launch day, there was high thin cirrus haze condition which slightly affected the quality of the engineering film, but the condition was not serious enough to delay the mission. The effects of the cloud condition on the different types of camera coverage are given in subsection 6.3.1.2.

Weather observations in the launch area for lift-off taken at 11:00 a.m. e.s.t. were as follows:

Cloud cover -	
Cumulus at 2,000 feet altitude, percent coverage . .	10
Cirrus at 34,000 to 38,000 feet altitude, percent coverage	100
Wind direction, degrees	180
Wind velocity, knots	12
Visibility, miles.	10

Pressure, in. Hg 30.05

Temperature, °F 78.9

Dew Point, °F 67

Relative humidity, percent 64

Launch-area wind direction and velocity plotted against altitude is presented in figure 12-5.

Table 12-I indicates the atmospheric parameters measured in the launch area for various altitudes.

12.3 FLIGHT SAFETY REVIEWS

Flight Safety and Mission Review meetings were conducted to determine the flightworthiness of the spacecraft and launch vehicle for the GT-1 mission and to ascertain the readiness of all supporting elements.

12.3.1 Spacecraft

12.3.1.1 Pre-Flight Readiness Review.- After the main hangar systems tests of Gemini spacecraft 1 had been accomplished and prior to mating the spacecraft to the launch vehicle, a Pre-Flight Readiness Review was conducted on February 18 and 19, 1964, by the Gemini Program Office. Each system was reviewed, and the following is a summary of open items, deviations, qualification status, or program office direction as a result of the review.

12.3.1.1.1 "Environmental and coolant systems:" No open items existed in the "environmental and coolant systems." A deviation considered acceptable was the reseat pressure of the 5.35 psi differential (specification is a minimum of 5.4 psid) of the cabin-pressure relief valve (P/N 52-83700-85). Acceptance was based on the minor nature of the deviation and the primary function of the valve with no oxygen on board during the GT-1 mission. This valve was also not qualified for flight; however, the scheduled qualification-test completion date is late 1964 and the status of qualification will be carefully monitored for future spacecraft. Four coldplates, which were used for prelaunch cooling only, were accepted for flight on a waiver due to the unqualified status of these parts.

12.3.1.1.2 Electrical system: There were no open items or deviations in the electrical system. Three electrical system items were not qualified. Inverter P/N 52-83701-45 had completed only temperature and vibration testing, but the "X" qualification status was approved by the review team since this inverter was used in a special application for GT-1 only and its function was not critical on this mission. The 52-79721-5 circuit breaker qualification testing was not scheduled for completion until May 1964. The spacecraft contractor was directed to assure flight readiness of this item. The third item, relay P/N 52-79703-1, was to finish qualification testing on February 15, 1964. This item was required to be fully qualified before flight.

12.3.1.1.3 Communications: The only open item associated with the communication system of spacecraft 1 was the information desired by the Department of Defense (DOD) on the spurious signal characteristics of the C-band transponder. This information was transmitted from the spacecraft contractor to the Gemini Program Office and forwarded to the DOD. The 52-85103-1 antenna was fully qualified except for the salt-spray test scheduled for completion on February 21, 1964. The Gemini Program Office agreed that the salt-spray test should not present a problem on this mission. The TM transmitters and allied components (P/N 52-85713-63,-65,-67 and 52-88712-9) were scheduled for qualification testing including acoustic noise by April 1, 1964, as directed by the Gemini Program Office. (See subsection 12.3.1.1.4).

12.3.1.1.4 Instrumentation: In the instrumentation system, one open item existed on the calibration of the sound pressure level system 1. Several deviations, which were considered acceptable on spacecraft 1, but which the spacecraft contractor was directed to investigate on subsequent spacecraft, were excessive and out-of-specification noise on the outputs of the d-c to d-c converters when the power supply bus was spiked with $\pm$ 100v d-c, and excessive and out-of-specification noise on converter S/N 7 under minimum bus voltage and maximum loading conditions. The conditions which would produce these deviations would not occur on spacecraft 1. A Test Request was issued by the Gemini Program Office to qualify the filter bias unit (52-88712-9). The spacecraft contractor was also requested to pretest the spacecraft 1 transmitters for acoustic noise before the scheduled qualification tests were performed. The qualification testing of the d-c to d-c converter, P/N A05A006, had been completed, but the data had not been completely analyzed. This unit was required to be qualified before the flight. The need for qualification of the meter shunt, P/N 52-79717-3, under high-temperature conditions was waived for spacecraft 1.

12.3.1.1.5 Structures: The following open items existed on spacecraft structures:

(a) Two break wires, identical to those which had already been installed on the spacecraft, had inadvertently been broken during a sled test at the Naval Ordnance Test Station.

(b) The spacecraft contractor was asked to investigate the fit and alinement problems of the shingles.

(c) As a result of failures in the oxygen-bottle mounts during vibration tests at the contractor's facility, new mounts were being manufactured for installation in spacecraft 1.

(d) A Titan II/Gemini Spacecraft Interface meeting resulted in direction that a test be conducted to evaluate the sealing characteristics of the adapter area to assure controlled venting during launch. The test plans were presented and reviewed.

(e) The spacecraft contractor was requested to reevaluate the hatch flipper door pre-loads to determine the required and actual pre-loads on the spacecraft 1 flipper doors.

(f) The mandatory and priority temperature and pressure measurements for spacecraft 1 were to be reviewed by personnel from Manned Spacecraft Center and the spacecraft contractor at Cape Kennedy, and comments were to be submitted to the Gemini Program Office. Two conditions reviewed and considered acceptable on spacecraft 1 were the corrosion found between the flexible linear-shaped charge and the lower mating ring at station Z16.0, and external corrosion noted at the contractor's facility on the hatch actuators. The first condition was satisfactory after the area had been cleaned and further corrosion was prevented. The second condition was accepted because the hatch actuators did not function during this mission.

12.3.1.1.6 Spares: A survey of critical items at Cape Kennedy and of available equipment at the contractor's facility was requested to determine if a selected supply of spares could be made available at Cape Kennedy prior to launch.

12.3.1.2 Flight Readiness Review.- The Flight Readiness Review was held at 1:00 p.m. e.s.t. on April 3, 1964. All the Pre-Flight Readiness Review items had been corrected with the exception of the full qualification of the 52-79721-5 circuit breaker. The spacecraft contractor considered the circuit breaker to be flightworthy since it had successfully completed vibration, X-ray, salt-spray, and humidity testing. The review board concurred. Only two new open items were reviewed. In the instrumentation system, channel E of the high-frequency telemetry package had exhibited an apparent low deviation. Under direction of the board, subsequent investigation showed that the voltage controlled oscillator (VCO) of the channel had been incorrectly set because improper procedures had been used. It was corrected for flight. The other problem was a question on the structural integrity of the ECS pump package mount. The review board asked the spacecraft contractor and NASA to continue design analysis studies and sample structural tests. When these tests were completed prior to launch, an adequate margin of safety was confirmed. All other systems were approved ready for flight, pending the outcome of the final combined systems test. This test was completed satisfactorily on April 5, 1964.

12.3.2 Launch Vehicle

The Gemini launch vehicle 1 Flight Readiness Review was conducted at 1:00 p.m. e.s.t. on April 4, 1964. The Air Force Space Systems Division (AFSSD) and Aerospace presented the status of the launch-vehicle systems. Two unresolved items remained for further action.

A failure analysis on the spin motor rotation detection (SMRD) circuit of the stage I secondary rate gyro package was not complete, and a review of the hold-kill circuitry for possible out-of-sequence events due to relay closure times was requested by the Operations Director. Subsequent investigation by personnel of NASA and the launch vehicle contractor showed that a problem did not exist in the second item.

12.3.3 Mission

The GT-1 Mission Review Meeting was held at 9:00 a.m. e.s.t. on April 6, 1964. A brief presentation was made on the mission description and mission ground rules. The launch vehicle had two unresolved problems. The first was the autopilot rate gyro package SMRD circuit mentioned in subsection 12.3.2. The second was an investigation of the material in a Marman clamp T-bolt that was suspected of being weakened by stress corrosion. All other systems and support groups were found in readiness. A later inspection of the bolt on stage II ascertained that it was proper and not made of the suspect material.

12.3.4 Flight Safety Review

The F-1 day Flight Safety Review Board met on April 7, 1964. The Board was advised by AFSSD that all launch-vehicle problems, except receipt of the results of the failure analysis on the secondary autopilot rate gyro plug, had been resolved; however, these results had been confirmed by telephone, and the launch vehicle was approved for flight. On April 8, 1964, the GT-1 mission was satisfactorily accomplished.

12.4 SUPPLEMENTAL REPORTS

Supplemental reports will be issued as shown in table 12-II. The format for the supplemental reports will conform to the external distribution format of the NASA or contractor organization preparing the 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

In tables 12-III and 12-IV will be found a listing of the GT-1 mission data which have been made available to the Evaluation Team. Table 12-III includes the data obtained from the Atlantic Missile Range, the Goddard Space Flight Center, and the network stations; table 12-IV includes the data obtained from the John F. Kennedy Space Center, Manned Spacecraft Center, the spacecraft contractor, and the Aerospace Corporation. These date will be on file at the Manned Spacecraft Center, Computation and Analysis Division, Houston, Texas. The tables also indicate known additional mission data which will be available at a later date.

It should be noted that each supplemental report required in section 12.4 will also include a data availability schedule which will list all additional data generated in support of the supplement.

12.5.1 Atlantic Missile Range (AMR)

Delivery of data requested from the AMR was generally well ahead of schedule. Magnetic tape recordings, oscillograph charts, and signal strength recordings of telemetry received at Cape Kennedy were provided within 2 hours after lift-off. AMR also provided telemetry recordings for the first, second, and third orbital passes on LO + 1 day. Quick-look trajectory and special parameter data from the impact predictor recordings were received on LO + 1 day, but they did not include requested aerodynamic parameters. The latter were received on LO + 3 days. Final trajectory data were delayed several days due to a backlog of work. Data from down-range AMR sites were sent through normal channels as requested; this resulted in a delay in reduction of telemetry data for the time interval around BECO when stations at Cape Kennedy were unable to receive telemetry transmissions. Grand Bahama Island telemetry recordings were received on LO + 2 days. Some of the special requests for expanded oscillograph charts were delayed as a result of priority of support requirements for other vehicle launchings.

12.5.2 Goddard Space Flight Center (GSFC)

The GSFC network stations recorded telemetry data and tracked the spacecraft during the mission. The Bermuda tracking data for the second orbital pass (received via teletype) were used at Manned Spacecraft Center in Houston in calculating orbital trajectory data and insertion parameters. The Bermuda telemetry recording was lost in the mail and was not received until LO + 13 days. GSFC reduced the Mod III tracking data at Greenbelt, Maryland, and made the data available to the Evaluation Team on LO + 2 days.

12.5.3 Kennedy Space Center (KSC)

KSC made available quick-look plots of the spacecraft parameters in engineering units versus time within 2 hours after lift-off.

12.5.4 Manned Spacecraft Center (MSC)

All launch-phase spacecraft data reduced on the digital computer were completed ahead of schedule and were received between LO + 1 and LO + 5 days. MSC reduction of the spacecraft parameters included power spectral density plots for selected vibration measurements and calculation of heating rates for temperature sensors on the outer skin. Spacecraft data for the first, second, and third orbital passes were reduced and made available to the Evaluation Team on LO + 8 days.

MSC also computed and made available on LO + 12 days the power spectral density curves for six GLV vibration measurements.

12.5.5 Spacecraft Contractor

The spacecraft vibration analysis data, g rms and power spectral density plots, were available from the spacecraft contractor within LO + 5 days.

12.5.6 Aerospace Corporation

Aerospace made available to the Evaluation Team the GLV parameters in engineering units versus time on LO + 7 days. No power spectral density data nor machine analysis results were received in time for evaluation in this report.

TABLE 12-I.- LAUNCH-AREA ATMOSPHERIC PARAMETERS

Altitude, feet	Temperature, °F	Pressure, lb/sq ft	Density, slugs/cu ft
0	78.3	14.75085	0.002296
5,000	60.3	12.37430	0.001997
10,000	51.3	10.32980	0.001701
15,000	30.9	8.57385	0.001472
20,000	12.6	7.06150	0.001260
25,000	-2.9	5.77390	0.001066
30,000	-24.0	4.68350	0.000906
35,000	-45.9	3.75985	0.000766
40,000	-68.6	2.98120	0.000643
45,000	-91.3	2.33160	0.000534
50,000	-98.5	1.80380	0.000421
55,000	-101.9	1.39055	0.000328
60,000	-103.5	1.07010	0.000253
65,000	-87.7	.82795	0.000188
70,000	-78.3	.64670	0.000143
75,000	-74.9	.50750	0.000111
80,000	-68.3	.39875	0.000086
85,000	-59.8	.31465	0.000066
90,000	-48.5	.25085	0.000051
95,000	-35.3	.20010	0.000040
100,000	-28.3	.16095	0.000032

TABLE 12-II.- LIST OF SUPPLEMENTAL REPORTS

Number	Supplemental reports	Responsible organization	Completion date	Text reference section and/or remarks
1	Missile Flight Test Report GT-1	SSD and contractor	May 7, 1964	GE MOD III Performance and Interface
2	Burroughs Supplemental Data Report GT-1	SSD and contractor	May 7, 1964	Burroughs Computer and Interface
3	GLV Engineering Evaluation Report (GT-1)	SSD and Aerospace	May 7, 1964	Aerospace report containing detailed analysis, all systems, including summary of results, conclusions, and recommendations
4	Multiple objects in orbit	Gemini launch vehicle contractor with spacecraft contractor support through Manned Spacecraft Center	May 12, 1964	5.0
5	Post SECO transient	Gemini launch vehicle contractor	30 days after date of this report	5.0
6	Stage II thrust alinement	Gemini launch vehicle contractor	30 days after date of this report	5.0 5.2.4.2
7	Deletion of insulation from launch-vehicle adapter	Gemini launch vehicle contractor	30 days after date of this report	5.2.1.2
8	Detailed analysis of spacecraft structural loads	Spacecraft contractor	60 days after date of this report	5.1.1
9	Detailed analysis of spacecraft thermal loads	Spacecraft contractor	60 days after date of this report	5.1.1.4
10	Guidance system insertion accuracy	Aerospace	30 days after date of this report	5.0 5.2.4.2
11	Excessive pressure developed in secondary hydraulic system at engine ignition	Gemini launch vehicle contractor	30 days after date of this report	5.2.3.2
12	Manned Space Flight Network Analysis For GT-1 Mission	Goddard Space Flight Center		This report will contain a detailed analysis of the performance of the Manned Space Flight Network in support of the GT-1 mission

TABLE 12-III.- DATA AVAILABILITY SCHEDULE: ATLANTIC MISSILE RANGE, GODDARD SPACE FLIGHT CENTER, AND NETWORK STATIONS

Data Category	Data source																				
	Atlantic Missile Range							Goddard Space Flight Center and Network Stations													
	CNV	PAFB	GBI	SAL	GTI	ELU	VAL	GSFC	MCC	BDA	CYI	KNO	CRO	WOM	CTN	HAW	CAL	GYM	WHS	TEX	EGL
Telemetry																					
Magnetic tape recordings	▲		▲						▲	▲	X	X	X		X	▲	▲	X		▲	▲
Signal strength pen recordings	▲		▲		▲				▲	X	X	X	X		X	▲	▲	X		X	▲
Telemetry operations logs	▲		▲		X				▲	X	X	X	X		X	▲	▲	X		X	▲
Paper recordings	▲								▲												
Computer compatible tape (GLV PCM measurements)	▲		X																		
Radar tracking																					
Quick-look (printouts and magnetic tape)																					
Position, velocity, and acceleration	▲	▲						▲													
Special and aerodynamic parameters	▲	▲						▲													
Final data (printouts and magnetic tape)																					
Position, velocity, and acceleration	X																				
Special and aerodynamic parameters	X																				
Attitude data - pitch, yaw and roll	▲																				
Best estimate of trajectory (including optical)	X																				
Plotboard charts	▲		X	X					▲	X											
Magnetic tape recordings (raw radar data)	X	X	X	X																	
Radar operation function and event records	▲		X	X		▲	▲			X			X	X		▲			X		X
Optical and photographic																					
Cine fixed engineering sequential film	▲																				
Cine tracking engineering sequential film	▲																				
Cine theodelite (reduced)	▲																				
Airborne photographic film	▲																				
Meteorological																					
Surface weather observations	▲	▲	▲																		
Rawinsonde and wiresonde	▲	▲	▲																		
Other test data																					
Preliminary estimate of data coverage	▲	▲	▲	▲																	
MOD III system flight test reports	▲																				
MISTRAM/MOD III velocity comparison	▲																				
Command control records and logs	▲		▲	X					▲												
Acquisition aid records										X	X	X	X	X	X	▲	▲	X	X	X	▲
Voice recording of ground communications (1/4 in. tapes)									▲												

LEGEND: ▲ - Data available to the evaluation team.
X - Data to be available later.

TABLE 12-IV.- DATA AVAILABILITY SCHEDULE, KENNEDY SPACE CENTER, MANNED SPACECRAFT CENTER, SPACECRAFT CONTRACTOR, AND AEROSPACE

Data category	Reduction source				Comments
	KSC	MSC	SC	AERO	
Time histories, spacecraft	A	A			Launch and orbital telemetry measurements converted to engineering units and tabulated and plotted.
Time histories, Gemini launch vehicle				A	
Event listing, Gemini launch vehicle				A	Tabulation of bilevel GLV events and time of occurrance.
Thermodynamics, spacecraft		A			Heat flow rate calculations for each spacecraft outer skin temperature monitoring point based on temperature measurement data.
Vibration analysis, spacecraft, g rms		A	A		Overall plots of each vibration measurement in g rms versus elapsed time for the launch phase. Also, for each of the high frequency (20-600 cps and 20-2,000 cps) accelerometers, plots of g rms versus time by specific octave bands.
Power spectral densities		A	A		Power spectral density (PSD) plots for selected vibration measurements generated by an analog spectrum analyzer. In addition, MSC generated PSD plots by digital methods.
Vibration analysis, Gemini launch vehicle, power spectral densities		A			Power spectral density plots were computed on a digital computer for six vibration measurements.
Differential pressure, spacecraft		A			Plots of differential pressure (measured pressure minus free stream ambient pressure) versus elapsed time for the spacecraft pressure measurements.
Altimeter simulation, spacecraft		X			Plot of altitude versus time computed from telemetry measurements of static pressure and ICAO standard atmosphere data, and observed rawinsonde pressure and altitude data.
Audio noise by octave band, spacecraft		A			Plots of sound-pressure level in decibels versus octave bands for for selected times during the launch phase.

Legend: A - Data available to the evaluation team

X - Data to be available later

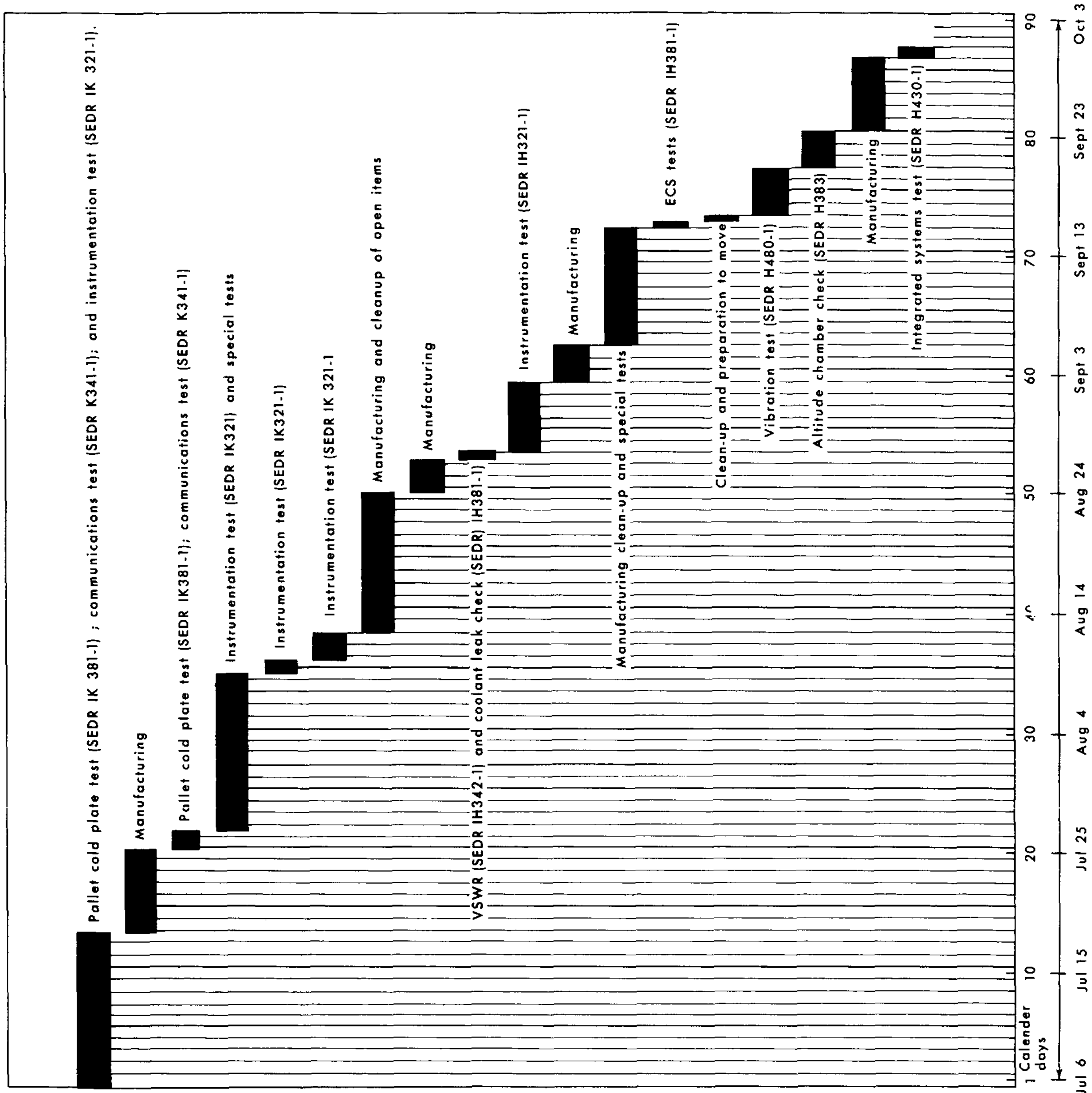

FIGURE 12-1 SPACECRAFT 1 SYSTEMS TEST-ACTUAL TEST SCHEDULE AT THE CONTRACTOR FACILITY

NASA-S-64-3088

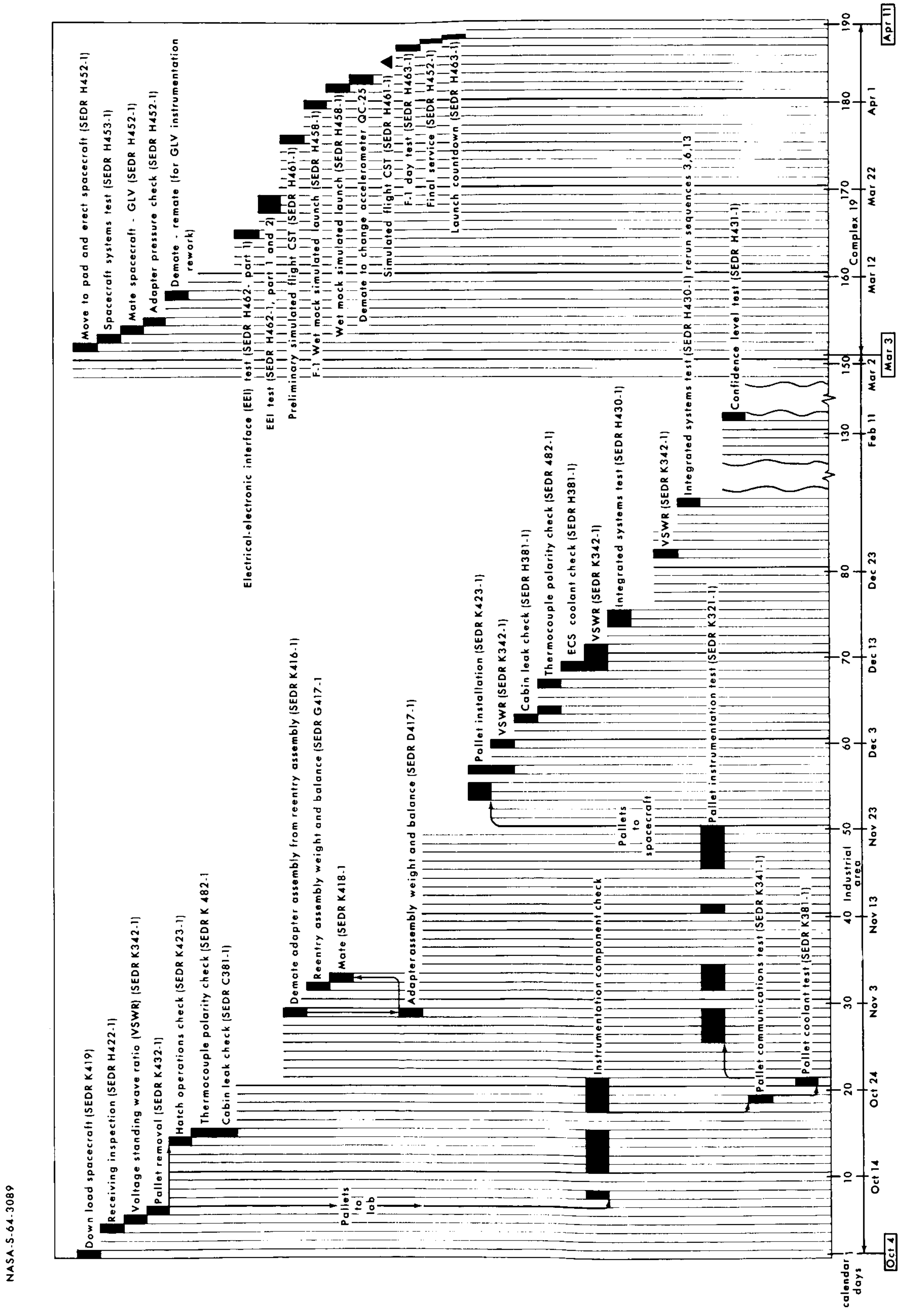

FIGURE 12-2 SPACECRAFT 1 SCHEDULE AT CAPE KENNEDY

NASA-S-64-3288

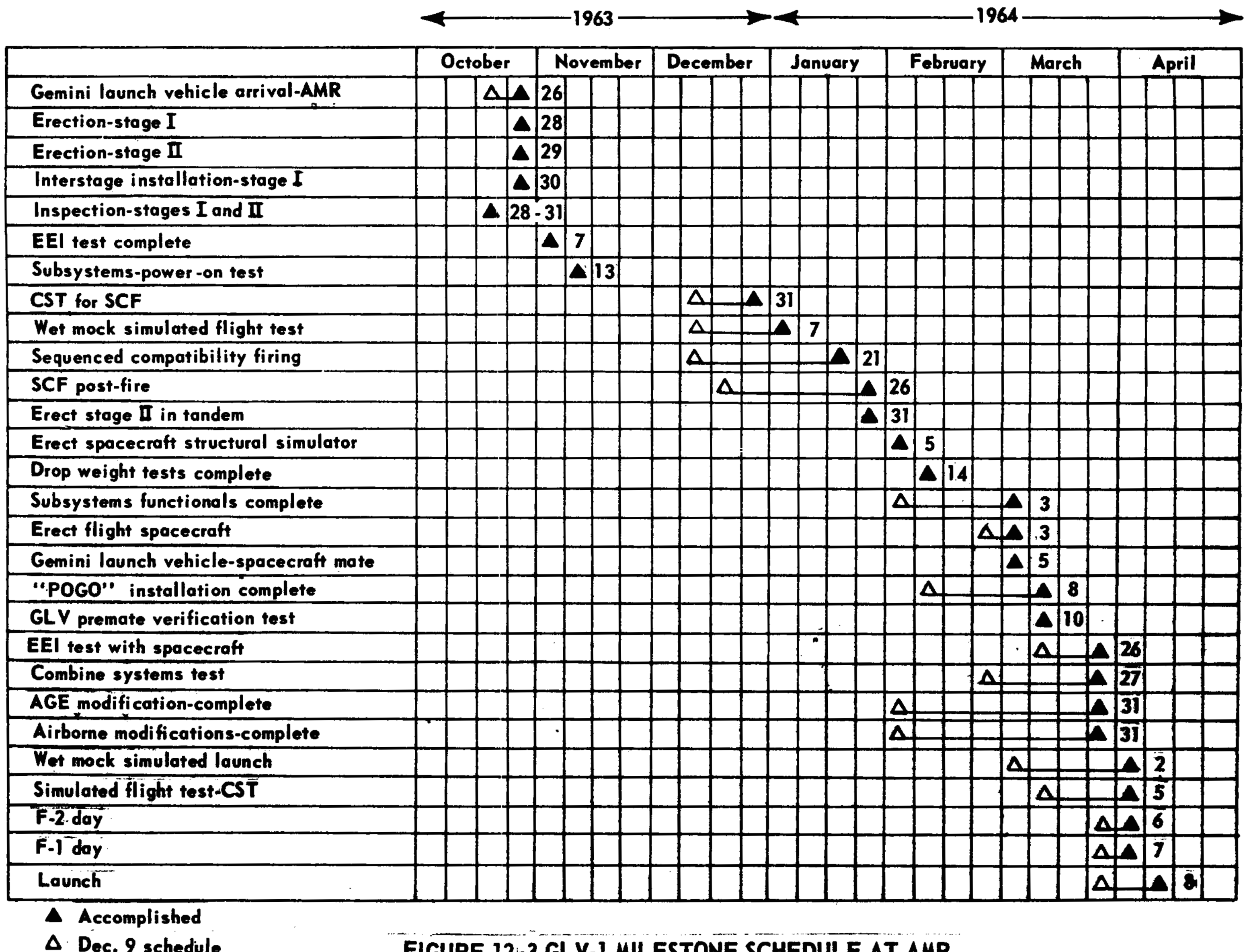

FIGURE 12-3 GLV-1 MILESTONE SCHEDULE AT AMR

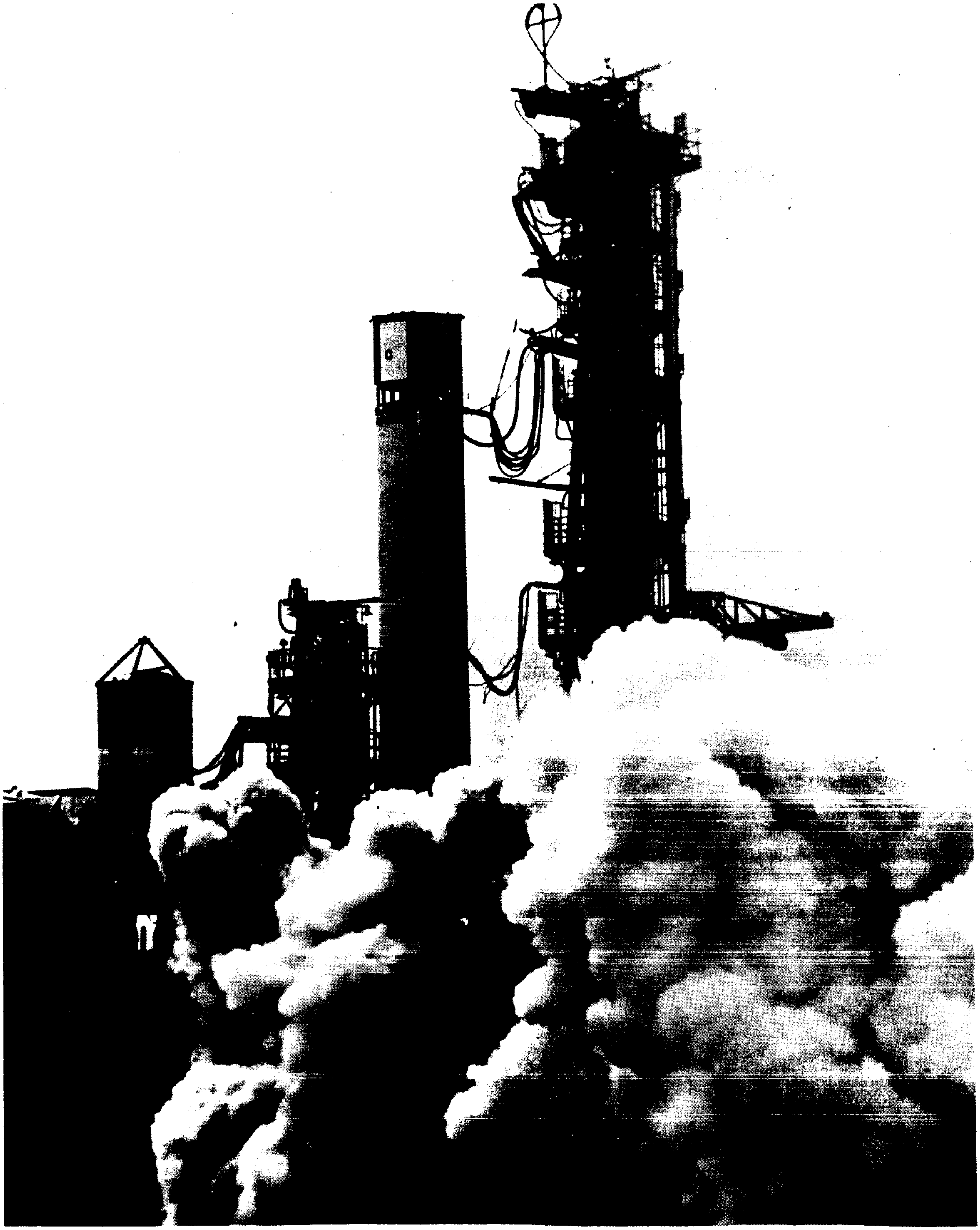

FIGURE 12-4 GLV-1 SEQUENCED COMPATIBILITY FIRING (SCF)

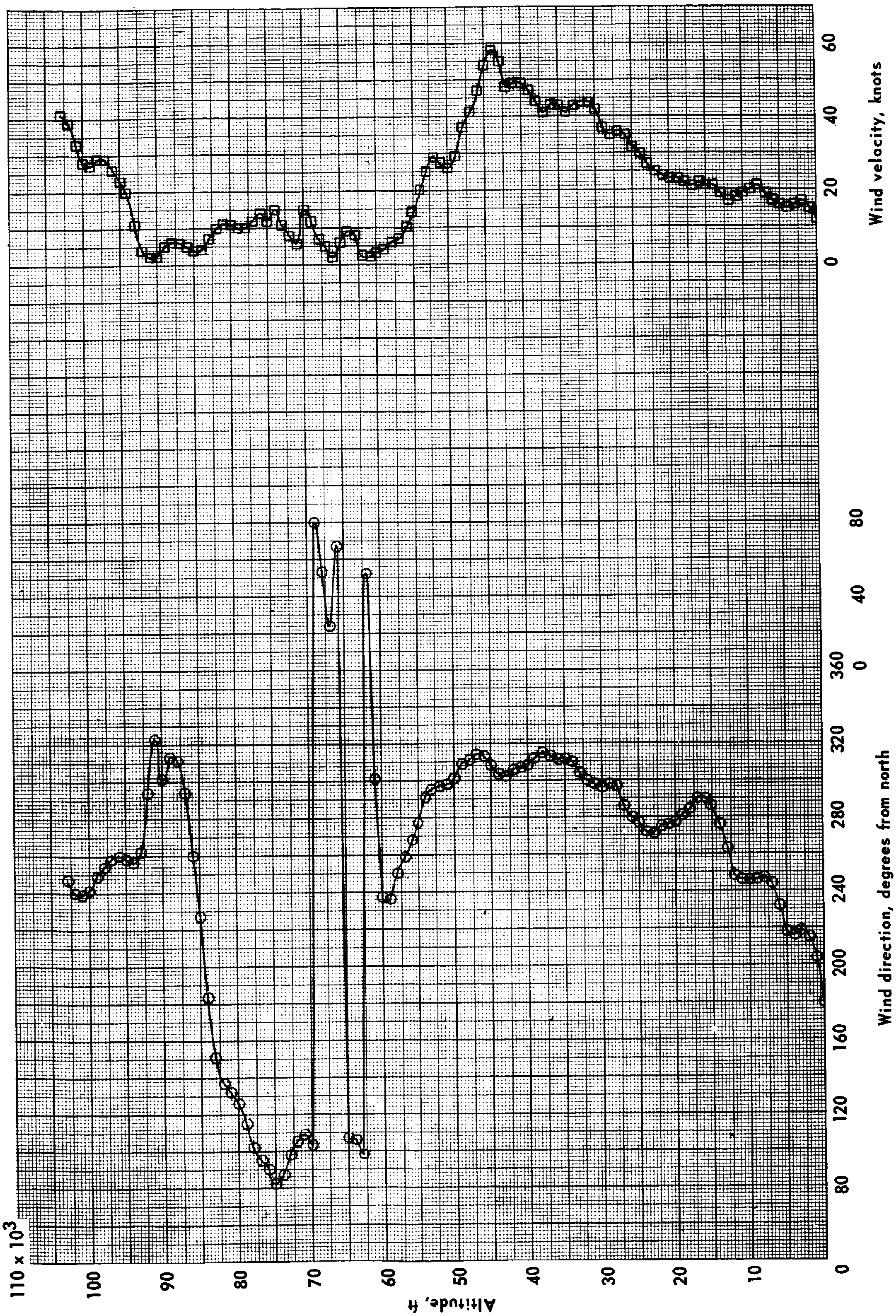

FIGURE 12-5 VARIATION OF WIND DIRECTION AND VELOCITY WITH ALTITUDE.

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