NASA
LANGLEY RESEARCH CENTER

PROJECT FIRE

INTEGRATED POST FLIGHT EVALUATION REPORT

FLIGHT NO. 1

REPORT NO. GDA | BKF64-018

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Prepared by

SLV TEST EVALUATION AND GUIDANCE SOFTWARE GROUPS

GENERAL DYNAMICS | ASTRONAUTICS

A DIVISION OF GENERAL DYNAMICS CORPORATION

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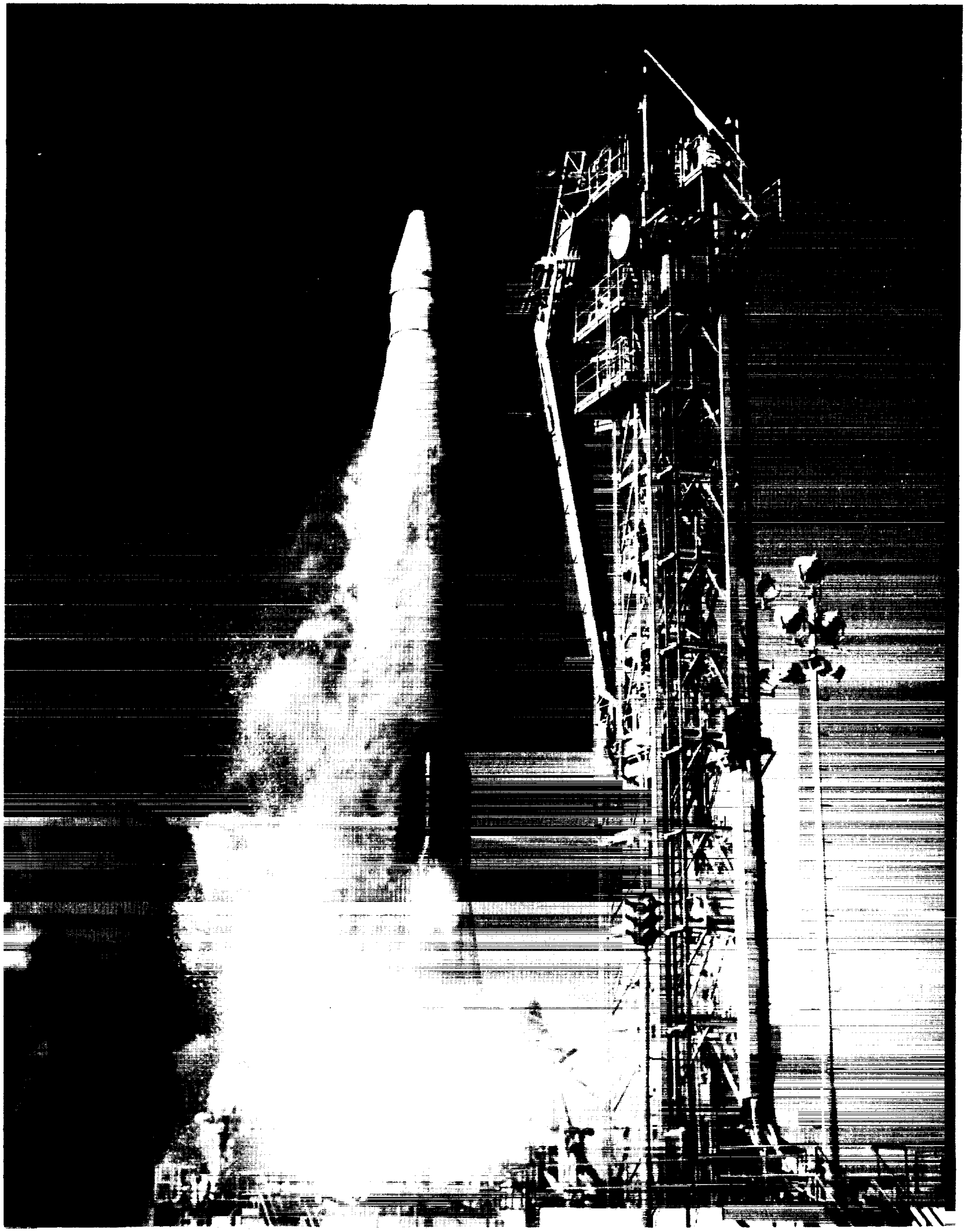

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INTRODUCTION

The first Project Fire space vehicle was successfully launched from Complex 12 at Cape Kennedy, Florida at 16 hours 42 minutes 25.536 seconds, Eastern Standard Time on April 14, 1964 by the National Aeronautics and Space Administration. The specific purpose of this flight was to obtain data on convective and radiative heating, radio signal attenuation, and material behavior during reentry into the earth's atmosphere at a velocity near 37,000 feet per second.

Project FIRE is a program of the National Aeronautics and Space Administration, Office of Advanced Research and Technology and is managed by the Langley Research Center. The spacecraft and tracking and data acquisition systems are also managed by Langley Research Center. The launch vehicle system is managed by the Lewis Research Center assisted by Goddard Launch Operations.

The Project FIRE space vehicle consisted of an Atlas Launch Vehicle produced by General Dynamics/Astronautics, a Velocity Package produced by Ling-Temco-Vought/Astronautics (containing an Antares II A5 rocket motor), and a Reentry Package produced by Republic Aviation Corporation. A photograph of the space vehicle is presented in the frontispiece of this report.

The Atlas injected the FIRE spacecraft into a precise ballistic trajectory along the Atlantic Missile Range. Upon Atlas separation the spacecraft was oriented to the proper Antares ignition attitude by the Velocity Package control system. At a predetermined time, following Atlas separation, the Antares rocket motor was ignited, accelerating the Reentry Package to 37,972 feet per second for reentry into the earth's atmosphere 4,335 nautical miles downrange near Ascension Island. A more detailed account of flight events is given in Part 2. Sequence of events times listed in Table 2-3-1 of Part 2 may vary slightly from those given in other parts of this report but should be considered the standard for the sake of future consistency.

A unique composite heat shield, consisting of two jettisonable phenolic asbestos layers sandwiched between three beryllium calorimeters, was used on the Reentry Package to provide three measurement periods during the heat pulse (see parts 3 and 4).

Two solid-state telemetry transmitters provided the primary sources of Reentry Package data. One transmitter provided real time data while the other relayed data on a delayed time basis which had been stored on tape during the reentry radio signal "blackout" period. All data were obtained by remote methods since the Reentry Package was not designed for recovery. Optical, radar, and telemetry tracking and receiving equipment located on Ascension Island and on ships and airplanes deployed in the reentry area gathered the reentry data.

The purpose of this integrated report is to present summary results concerning the flight of the space vehicle and the operation of its systems and subsystems. No research results are included.

PART 1

MISSION SUMMARY

GENERAL DYNAMICS/ASTRONAUTICS
INTEGRATED REPORT NO. GDA/BKF64-018

SECTION 1

SUMMARY

The Project FIRE Flight No. 1 trajectory provided the desired experimental conditions at the 400,000 feet reentry test point. All flight events occurred as planned and within allowable time limits.

A large quantity of reentry data was obtained from radar, optical and telemetry sources. Although a malfunction caused a decrease in the signal strength of the Reentry Package delayed-time transmitter, data for about seventy-five percent of the heat-pulse period were recovered. These data comprise radiation and total heating measurements for both Reentry Package forebody and afterbody, afterbody pressures, and radio attenuation information. An abrupt impulse, of presently unknown origin, was imparted to the Reentry Package near the middle of the data period. The resulting oscillations will make definitive analysis of the heating measurements difficult for the time period covering the second of the three data segments obtained during the heat pulse.

Performance of the Atlas launch vehicle was excellent. The launch vehicle successfully injected the Project FIRE spacecraft into a specified ballistic trajectory at the termination of powered flight. Spacecraft separation was satisfactorily accomplished.

The down range tracking facilities indicated close agreement with the reentry stage target conditions predicted at the termination of launch vehicle guidance. Guidance computer, radar performance and launch vehicle characteristics were well within the expected operating regions.

All Velocity Package flight objectives were accomplished in a completely satisfactory manner and no inflight problems were encountered.

In general, the Reentry Package performance was excellent. A description of those deviations in perforamce which did occur follows.

The roll rate gyro failed to function throughout the flight. This will complicate (although it will not preclude) the definition of reentry motion.

A spin rate of about 3 RPS after Reentry Package separation has been determined from signal strength records.

A slight yaw rate was induced in the Reentry Package at separation. At this time, there was no output from the pitch rate gyro, indicating a probable malfunction. Later in the reentry, the yaw rate increased to a larger amplitude and the pitch rate gyro began to respond with both gyros eventually oscillating stop-to-stop.

At Velocity Package spin up, the delay time data transmission deteriorated, resulting in cyclic loss of delayed time data. Sections of about two and one-half tape play backs of the reentry data were received at Ascension Island. An additional section of data was received by a ship in the impact area.

Late in the reentry phase the offset total radiometer bias shifted approximately 0.5 volt and the internal calibration lamp pulse disappeared. However, these data appear recoverable by adjusting the radiometer calibration curve to account for the bias shift.

PART 2

MISSION TRAJECTORY

General Dynamics/Astronautics
Integrated Report No. GDA/BKF 64-018

By Flight Reentry Programs Office
NASA/Langley Research Center

Approved by: David G. Stone
D. G. Stone
Manager, Flight Reentry Programs Office

SECTION 1

INTRODUCTION

Project Fire is a high-velocity flight experiment designed to investigate the environment of vehicles entering the earth's atmosphere at velocities slightly higher than lunar return velocities. The primary purpose of the experiment is to determine the total heat-transfer rates and the hot-gas radiance on a blunt-faced reentry body. The entry angle selected was a compromise between the steeper values needed to enhance the gas radiation level and the shallower flight paths which would insure survivability of the reentry package and the state-of-the-art instrumentation.

The reentry trajectory parameters chosen for the experiment were a velocity of 37,000 fps or higher, and a flight-path reentry angle of -15° at 400,000 feet altitude. The space vehicle was launched from Cape Kennedy along the Atlantic Missile Range to permit reentry into the Ascension Island area. The reentry was located to utilize the Ascension Island tracking, data acquisition, and optical instrumentation. Launch of the space vehicle was timed to insure that complete darkness would prevail in the Ascension Island area during the experimental period.

In order to meet the experimental requirements, a nominal trajectory was designed which provided a velocity of 37,335 fps and a reentry angle of -14.987° at an altitude of 400,000 feet. The reentry point was located 4,335 n.m. downrange from the launch site on a heading of 122.78° from true north and a ground-track minimum passage distance of 63.6 n.m. southwest of Ascension Island. Subsequent to the generation of the nominal trajectory, later performance and weight data were obtained for the reentry stage Antares II-A5 solid-propellant rocket motor and the reentry package. An expected trajectory was generated, based on this updated information. The expected trajectory provided a velocity of 37,968 fps, a reentry angle of -14.974°, and a ground range of 4,335 n.m. at the 400,000-foot reentry point. The actual flight data, therefore, are compared with the expected trajectory parameters in this report.

INTRODUCTION

To achieve the desired reentry trajectory, the launch vehicle guidance system was required to place the spacecraft on a coast ellipse such that it would pass through a predetermined point in space. The predetermined point is the point at which the velocity package motor ignites to accelerate the reentry package to the desired reentry velocity. The velocity package control system was required to provide the correct ignition attitude based on a reference provided by the launch vehicle. Ignition at the proper altitude was to be accomplished by a velocity package timer which was started by the launch vehicle guidance system.

The purpose of this part of the report is to summarize the extent to which the trajectory objectives were achieved.

SECTION 2

SUMMARY

The launch vehicle and velocity package produced a flight closely approximating that which was predicted. Complete radar tracking throughout the flight to reentry package separation enabled a highly accurate definition of the actual trajectory that was flown.

The following table provides a comparison between expected and actual parameters at the reentry point. As noted in the table, the differences between the expected and actual values indicate an extremely accurate trajectory.

Reentry Test Point
(400,000 feet altitude)

	Elapsed time, sec	Velocity, fps	Reentry angle, deg	Ground range, n. m.
Expected	1644.96	37,968	-14.974	4335.05
Actual	1647.42	37,971	-14.608	4344.55
Difference	+2.46	+3	+.366	+9.5
Tolerance			±1.0	-5, +15

SECTION 3

DISCUSSION OF DATA

The Project Fire Flight 1 trajectory results will be discussed in four phases: the launch and coast phase from lift-off to ignition of the Antares II-A5 rocket, the acceleration phase from Antares II-A5 ignition to separation of the reentry package, the reentry phase from the reentry test point to impact, and the mission sequence of events. In addition, atmospheric data in the reentry area are presented.

The actual flight data were obtained by reducing the Ascension Island FPS-16 radar measurements to trajectory parameters. Data from onboard accelerometers were reduced to trajectory parameters for comparison with the reduced radar parameters.

Launch and Coast Phase

Performance of the launch vehicle is shown in figures 2-3-6 and 2-3-7. Figure 2-3-6 presents altitude as a function of elapsed time and ground range from the launch site. Figure 2-3-7 presents velocity and flight-path angle as a function of elapsed time. A review of the above-mentioned figures graphically indicates that the launch vehicle provided a near nominal ascent and coast trajectory.

Acceleration Phase

Figure 2-3-8 presents the variation of velocity with time during Antares II-A5 burning. This figure compares the expected velocity variation with that obtained from reduced FPS-16 radar data and onboard accelerometer measurements. The FPS-16 radar data indicate that the actual trajectory during this phase of flight was nearly that which was expected. The accelerometer data indicate a difference from the expected velocity increment of about 2.8 percent. This difference is attributed to the accelerometer measurement capabilities and the data transmission system accuracies.

Reentry Phase

The reentry phase of the trajectory is shown in figures 2-3-9 and 2-3-10. Since the FPS-16 radar data became noisy at reentry package separation, the reentry trajectory was extended from the last good data point to impact by computer simulation. The impact point resulting from this simulation agrees very well with that given by the range; therefore, the actual data presented for this phase of flight were obtained from the computer simulation.

The actual trajectory data in terms of velocity, flight-path angle, and altitude with respect to time, as shown in figure 2-3-9, differ slightly from the expected. As can be seen from figure 2-3-10, the minimum ground track passage from Ascension during the experimental period was 60 n.m. or about 3.6 n.m. closer than the expected ground track. The actual impact point was approximately 169 n.m. southeast of Ascension, or about 15.6 n.m. downrange and 4.8 crossrange from the expected. These differences are attributed to attitude errors of the reentry stage at Antares II-A5 ignition.

Sequence of Events

The Project Fire Flight 1 sequence of events is presented in the following table, and a graphic illustration of the events is presented in figure 2-3-11. The table covers the major spacecraft events from launch through reentry package impact. All launch vehicle events from lift-off to spacecraft separation occurred within allowable limits of their expected times, however, these events are omitted from the table for security classification reasons. It should be noted that certain event times given in other parts of this report may differ slightly from the values listed in the table, since variations in event times will occur when different sources of information are used. Therefore the information contained in this table should be used as the standard and should supersede times given for similar events in other parts of this report.

Project Fire Sequence of Events
(Flight 1)

Event description (in-flight sequences)	Expected time, sec	Actual time, sec
Enable V/P ignition interlock (signal transmission)	129.9	128.0
V/P timer start (signal transmission	294.81	295.38
V/P shroud jettison (signal transmission)	295.5	298.0
Uncage V/P gyros (signal transmission)	302.8	306.0
S/C separation (signal transmission)	308.3	311.5
Start V/P pitch program	319.31	319.88
End V/P pitch program	420.84	421.39
Start R/P separation timers	1,567.2	1,567.57
Fire spin rockets	1,573.95	1,574.31
Ignite Antares II-A5 delay squib	1,573.95	1,574.31
V/P shell separation	1,576.95	1,577.32
Antares II-A5 ignition	1,580.2	1,580.31
Antares II-A5 burnout (main thrust termination)	1,613.18	1,613.13

Project Fire Sequence of Events - Continued
(Flight 1)

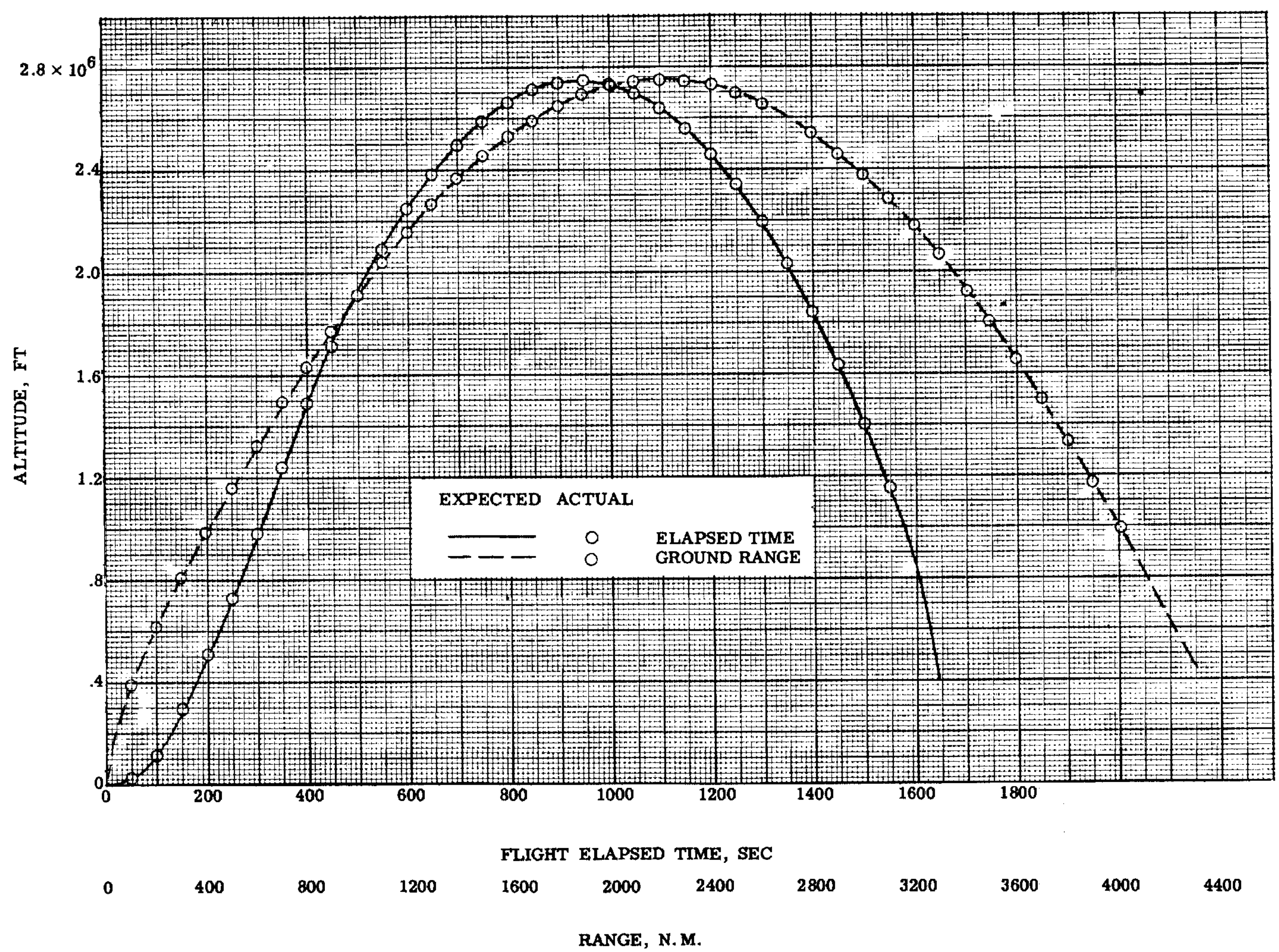
Event description (in-flight sequences)	Expected time, sec	Actual time, sec
R/P separation	1,640.2	1,640.46
Arrival at 400,000 ft altitude	1,644.95	1,647.4
Tumble motor ignition	1,646.2	1,644.4*
Begin T/M blackout	1,655.4	1,653.9
Begin C-band radar blackout		1,660.2
Start reentry timer (10g deceleration)	1,666.0	1,666.6
First heat-shield ejection (signal)	1,669.0	1,669.6
Second heat-shield ejection (signal)	1,676.0	1,676.6
End T/M blackout	1,682.0	1,686.8
Disable record and erase head	1,688.65	1,689.0
Activate failover switch		1,851.5
R/P impact	1,970.5	1,965.7

*From motion-picture film that was not time-correlated with range timing.

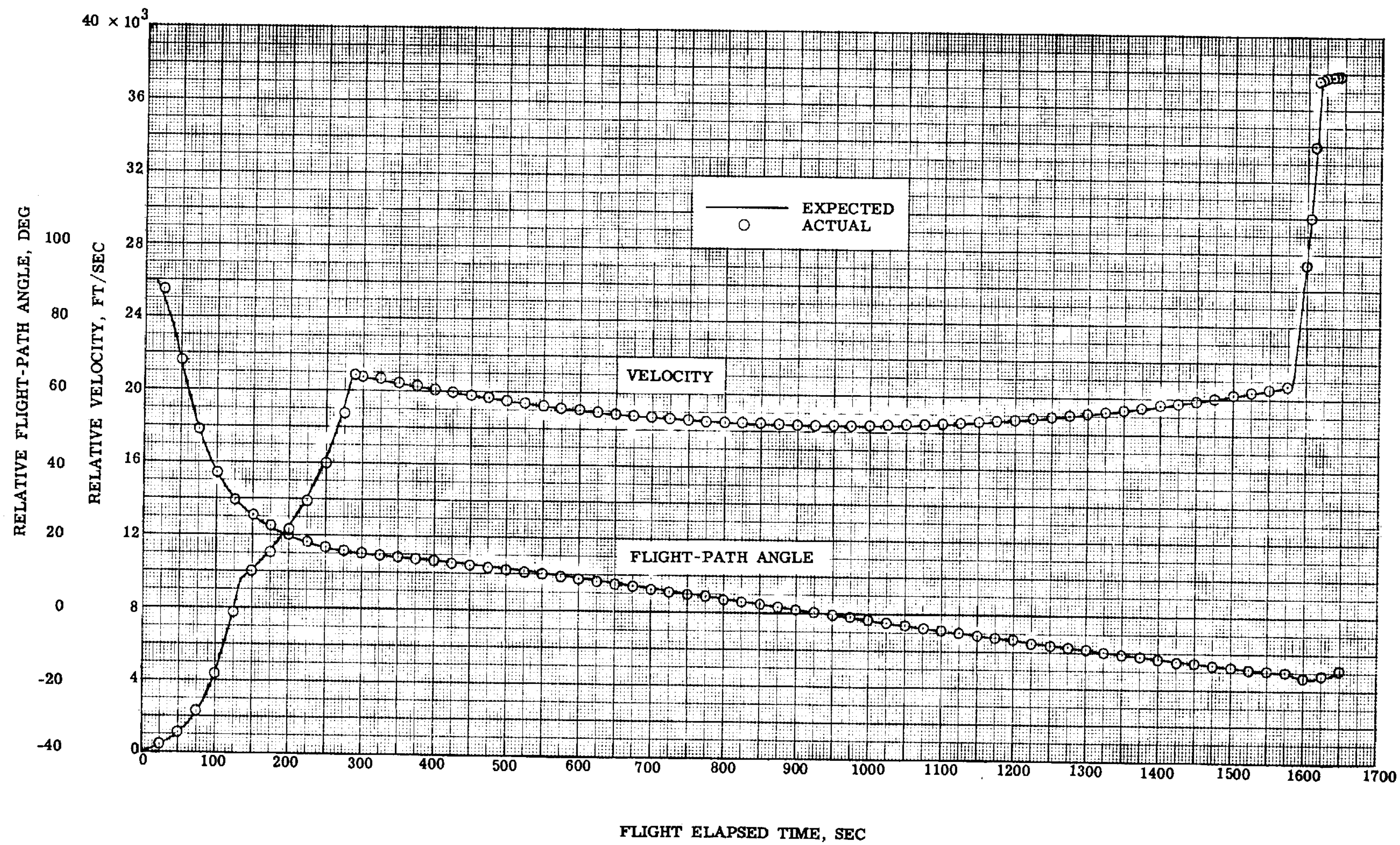
Atmospheric Data

In order to define the atmospheric environment through which the experiment was conducted, arrangements were made to conduct atmospheric soundings in the Ascension Island area immediately after conclusion of the experiment. Measurements of pressure and temperature were made with instrumented Goddard payloads launched on Nike-Apache sounding rockets. The results of these soundings are presented in figures 2-3-12 through 2-3-14. The sounding data are compared with U.S. Standard Atmosphere of 1962 which was used in the generation of the expected trajectory and the extended flight trajectory discussed in the reentry phase.

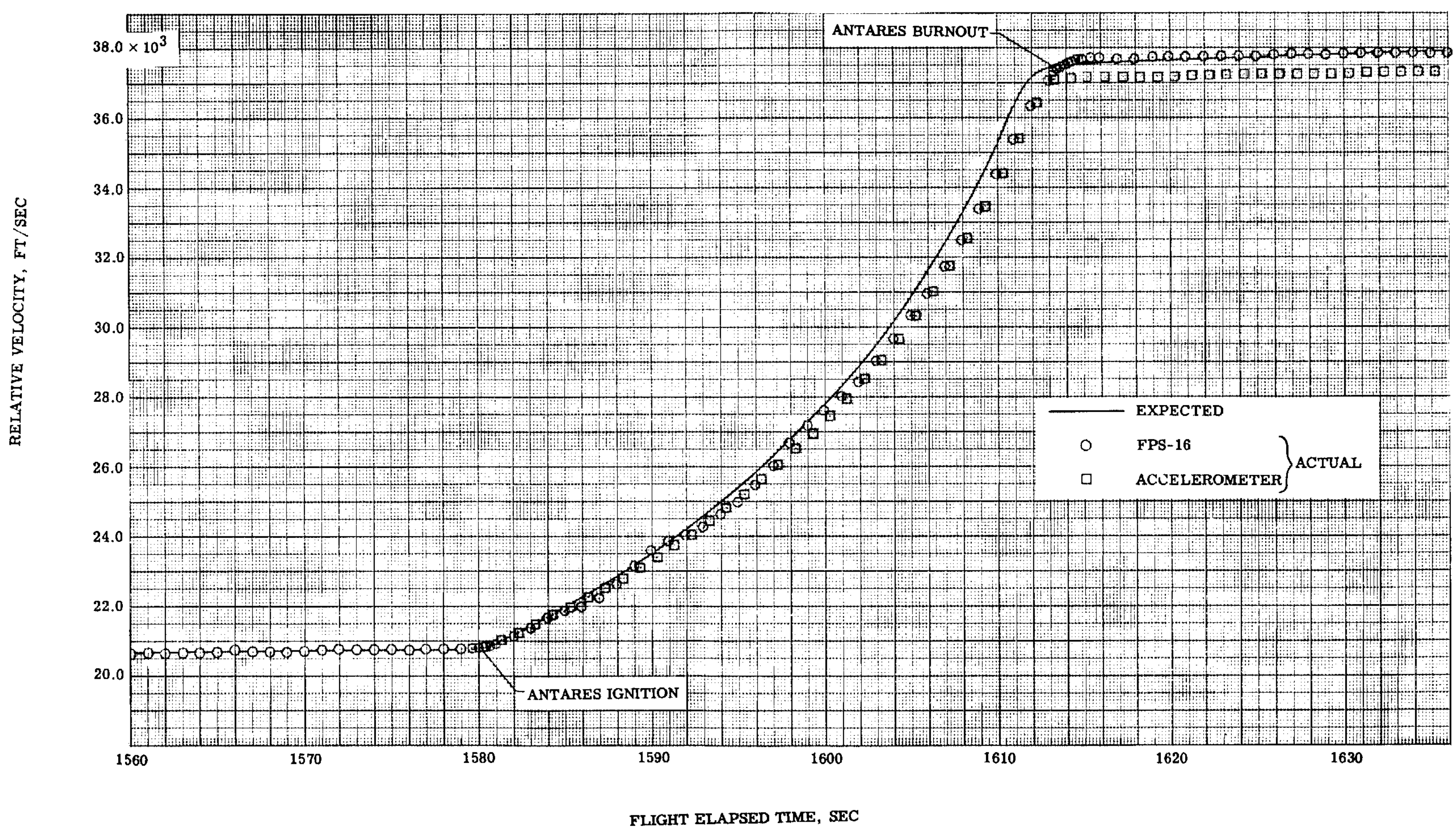
VARIATION OF ALTITUDE WITH GROUND RANGE AND FLIGHT ELAPSED TIME



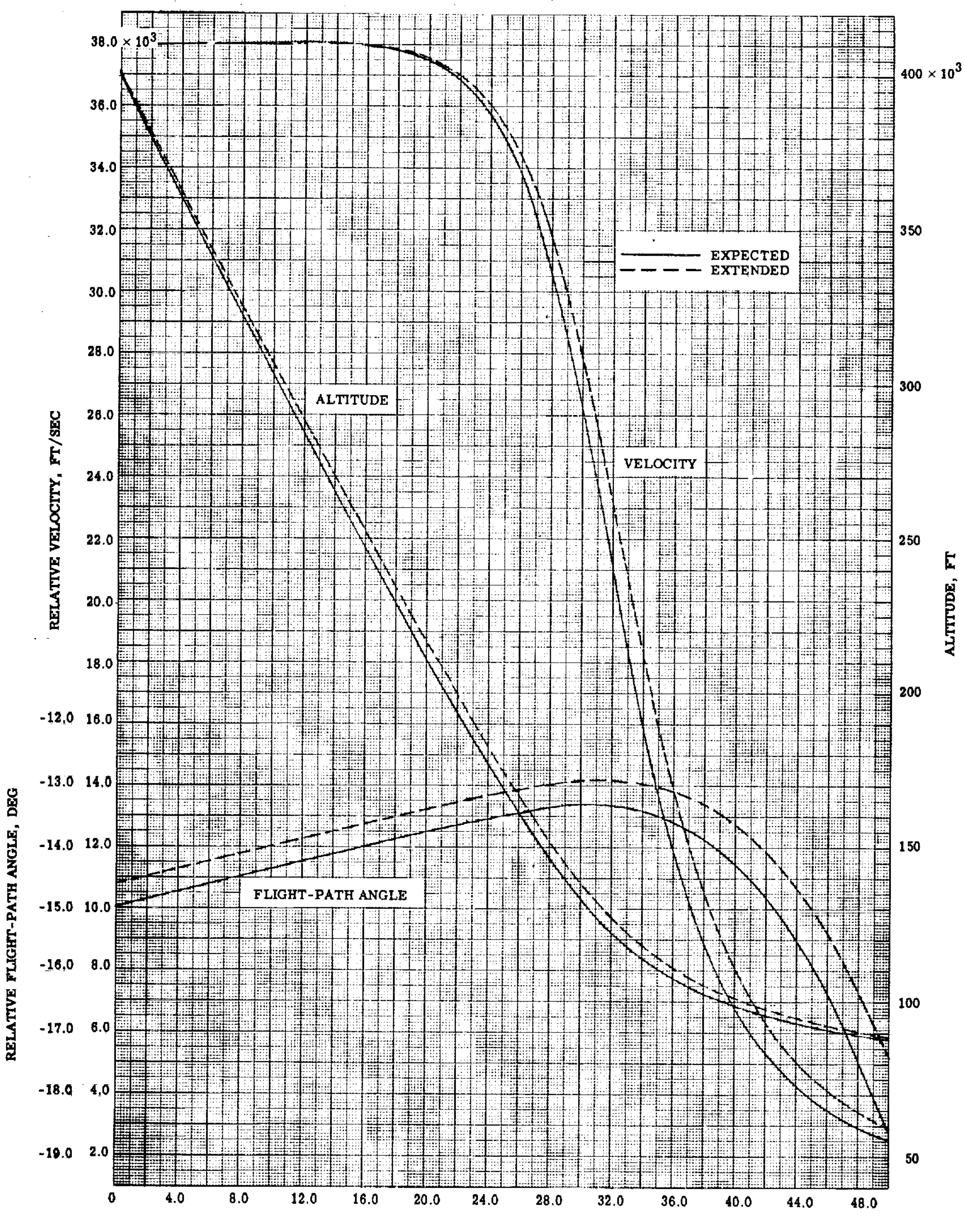
VARIATION OF VELOCITY AND FLIGHT-PATH ANGLE WITH ELAPSED TIME



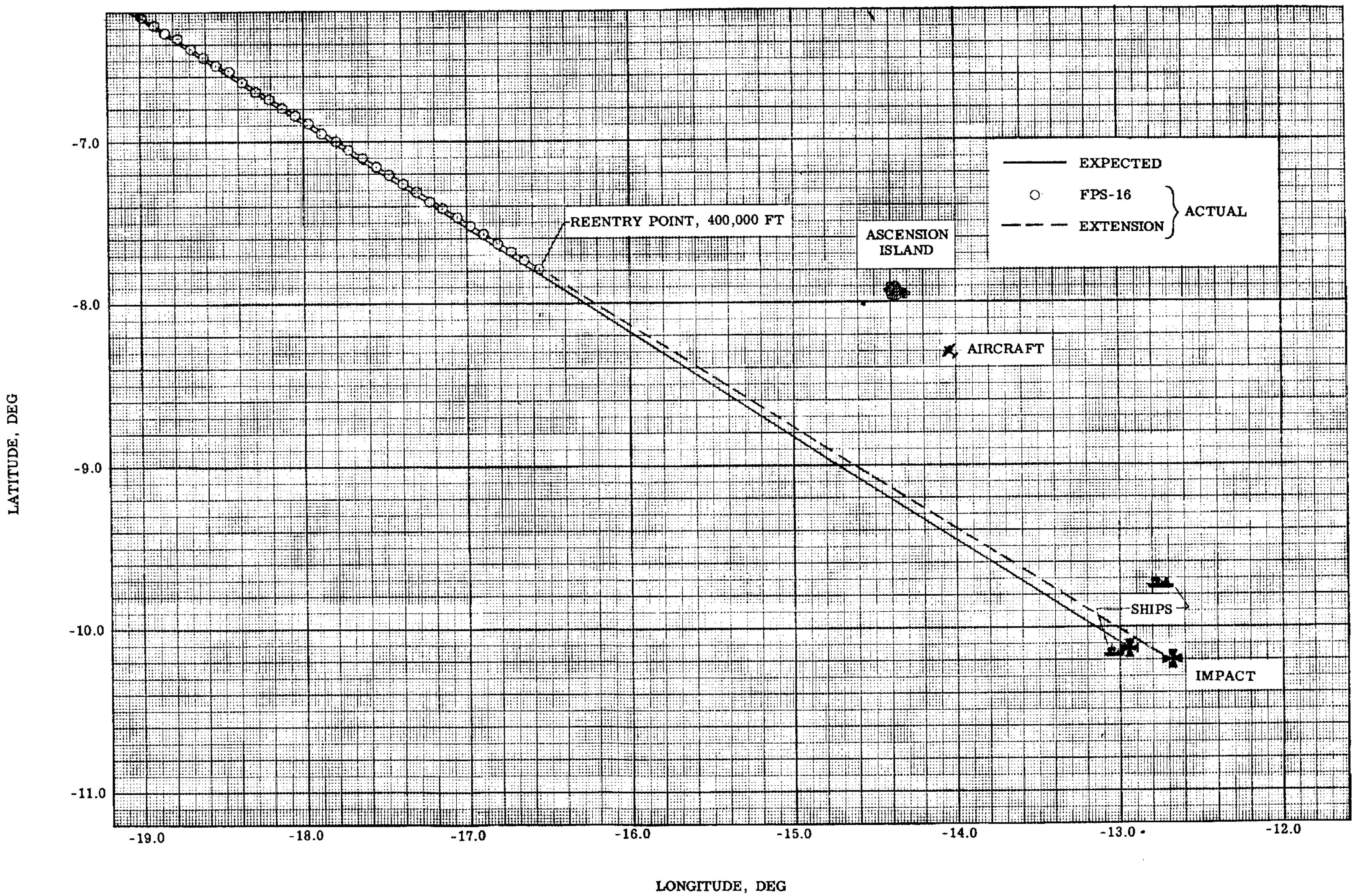
VARIATION OF VELOCITY WITH TIME DURING ANTARES II-A5 BURN



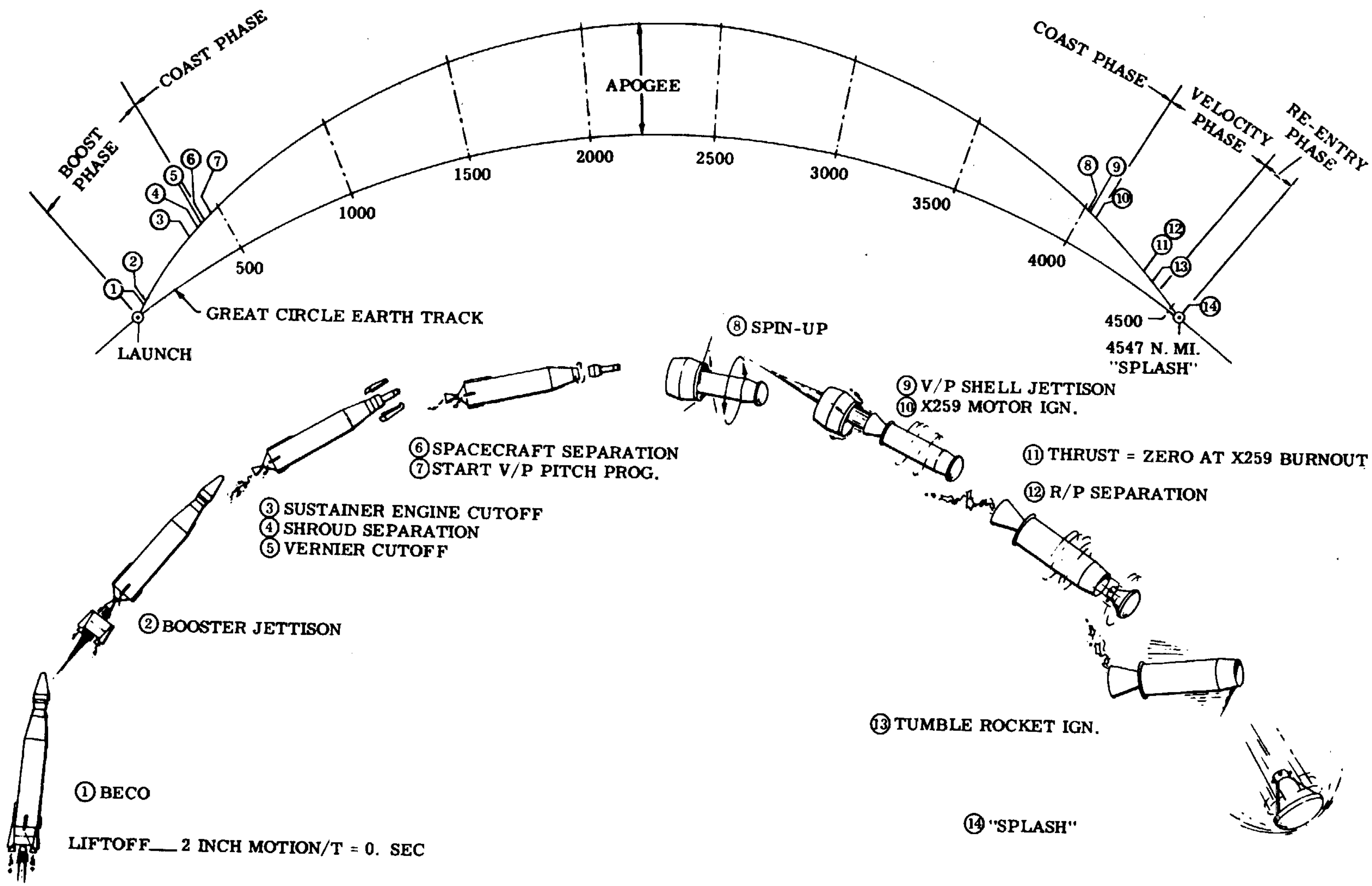
VARIATION OF ALTITUDE, VELOCITY, AND FLIGHT-PATH ANGLE WITH TIME FROM 400,000 FT



REENTRY GROUND TRACK

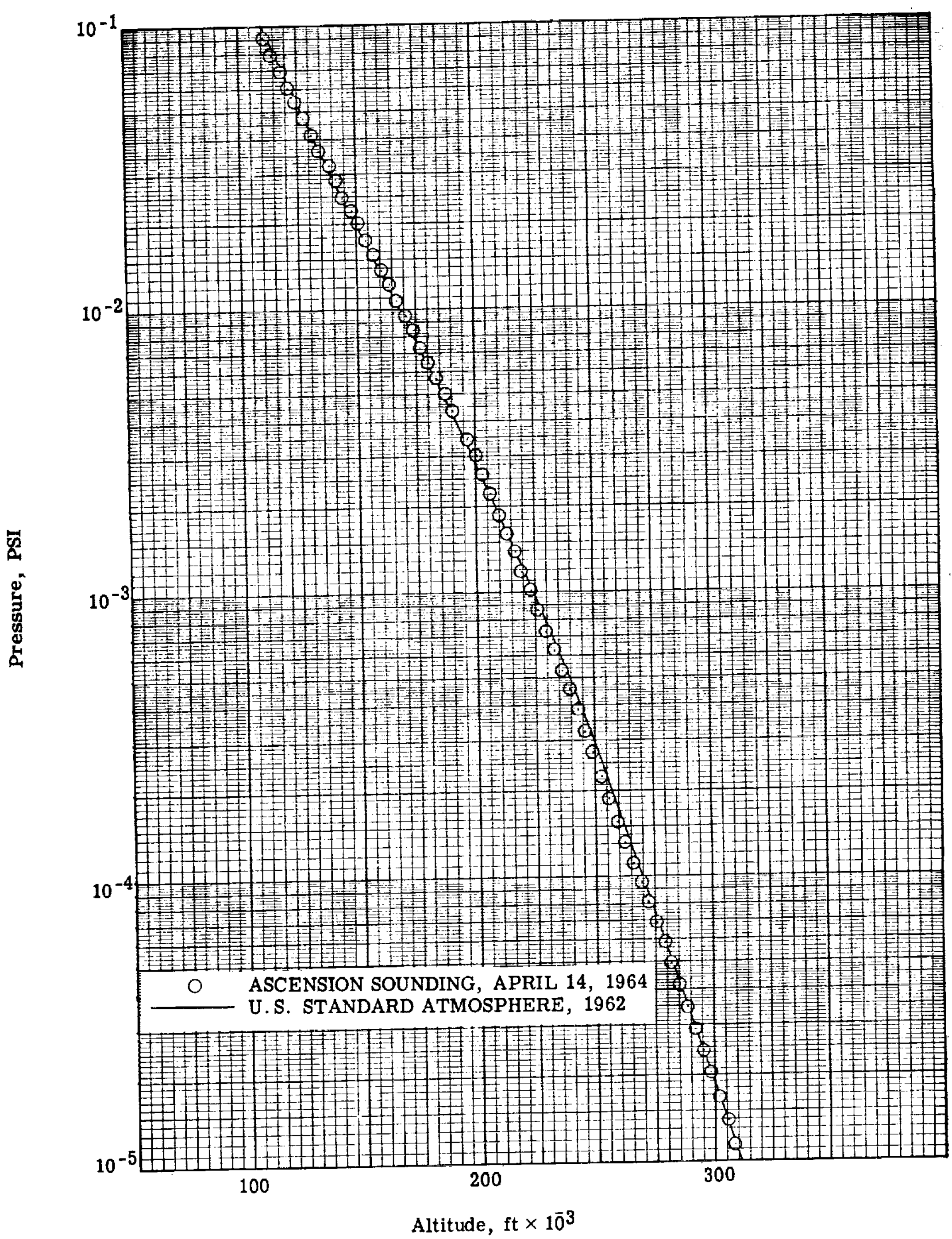


FLIGHT SEQUENCE OF EVENTS

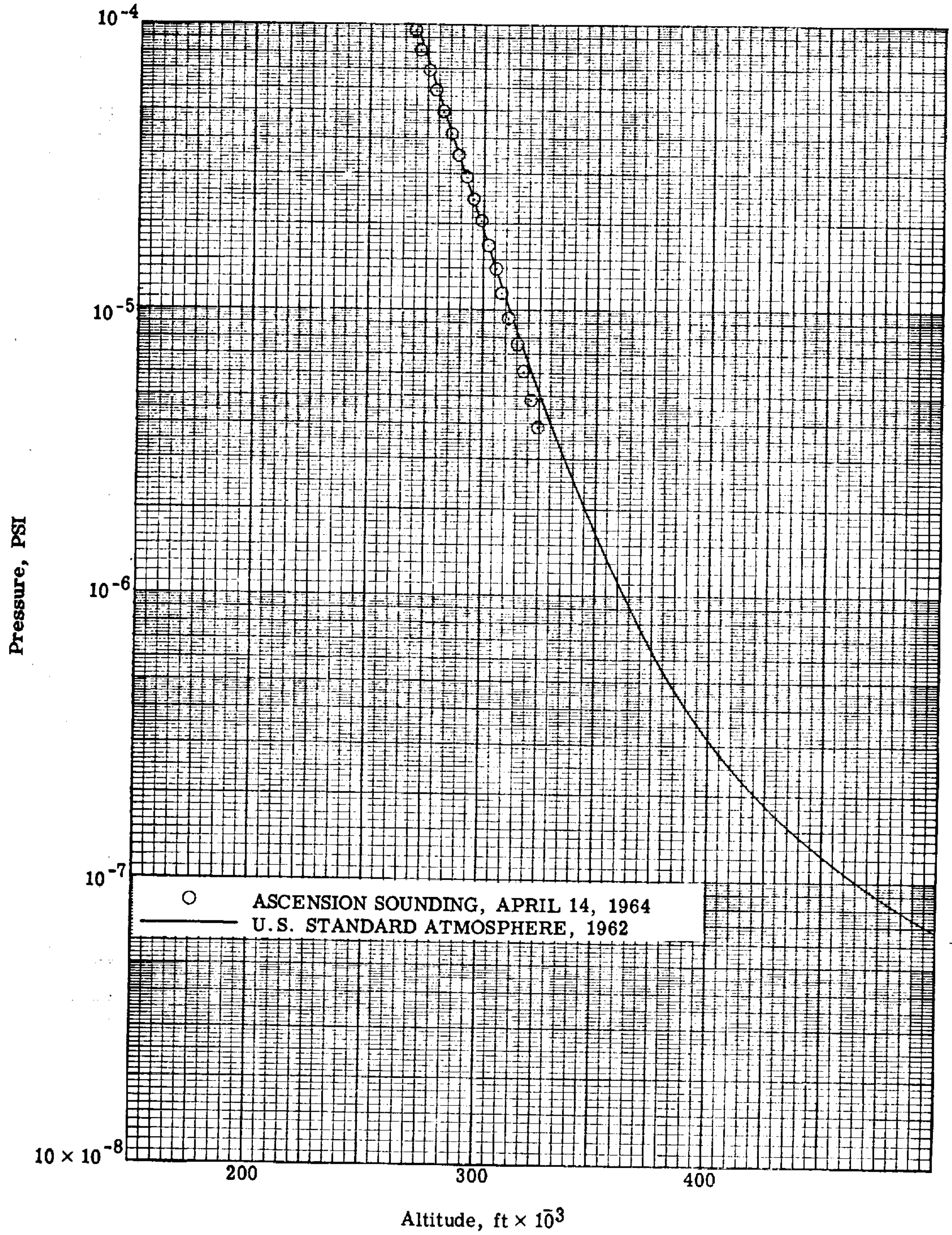


MISSION TRAJECTORY
FIGURE 2-3-12A
INTEGRATED REPORT NO. GDA/BKF 64-018
DISCUSSION OF DATA

ASCENSION SOUNDING. VARIATION OF PRESSURE WITH ALTITUDE

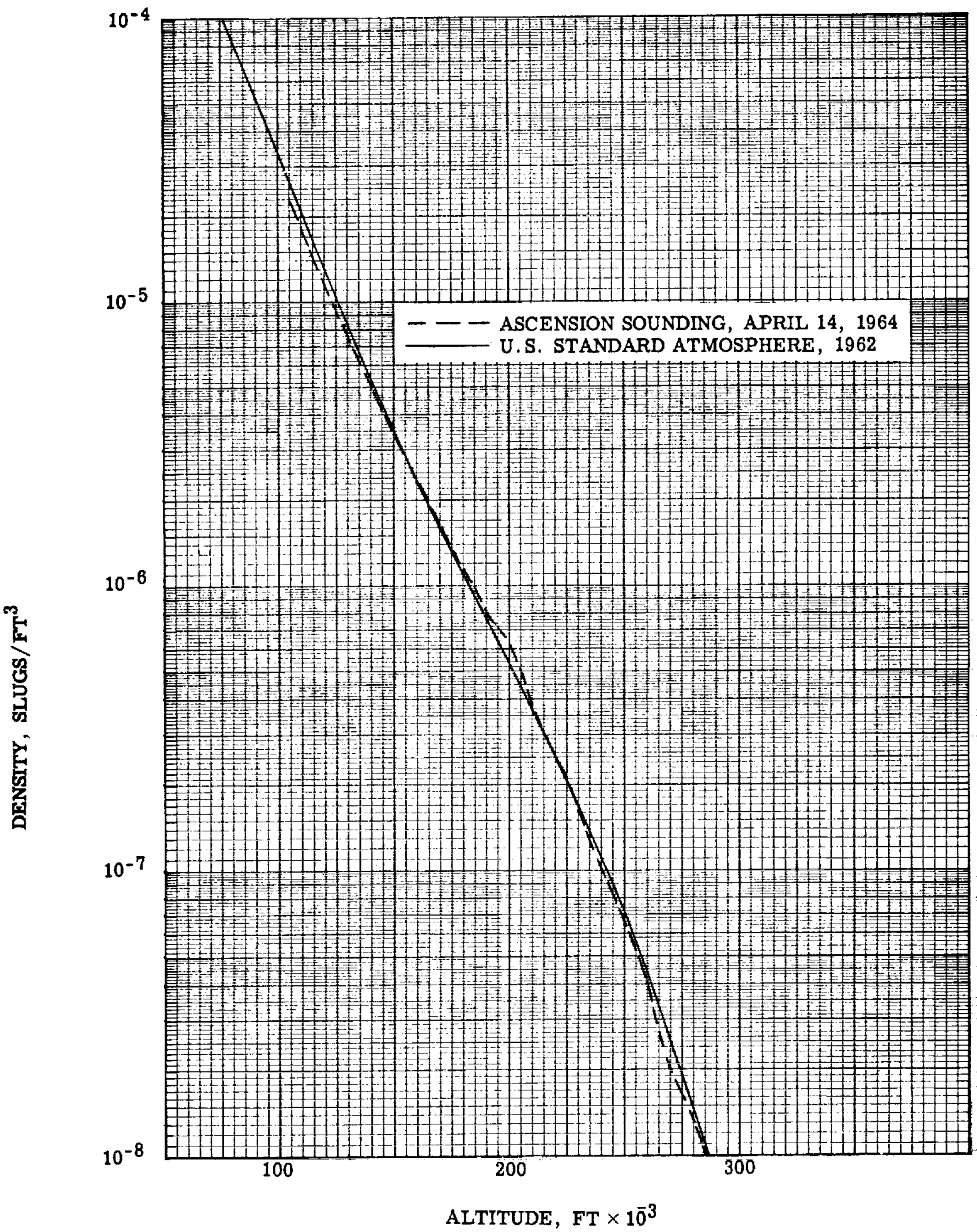


ASCENSION SOUNDING. VARIATION OF PRESSURE WITH ALTITUDE

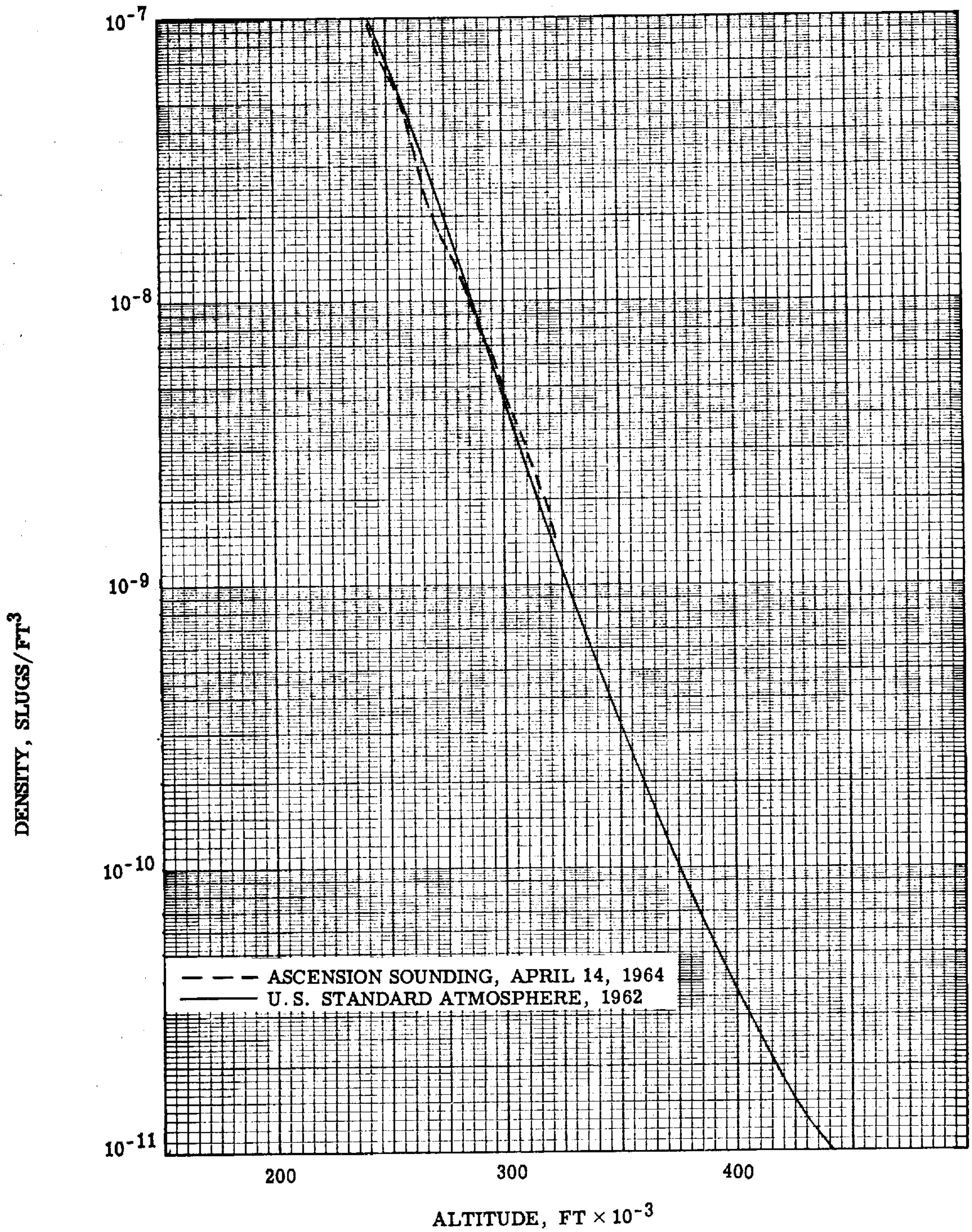


MISSION TRAJECTORY
FIGURE 2-3-13A
INTEGRATED REPORT NO. GDA/BKF 64-018
DISCUSSION OF DATA

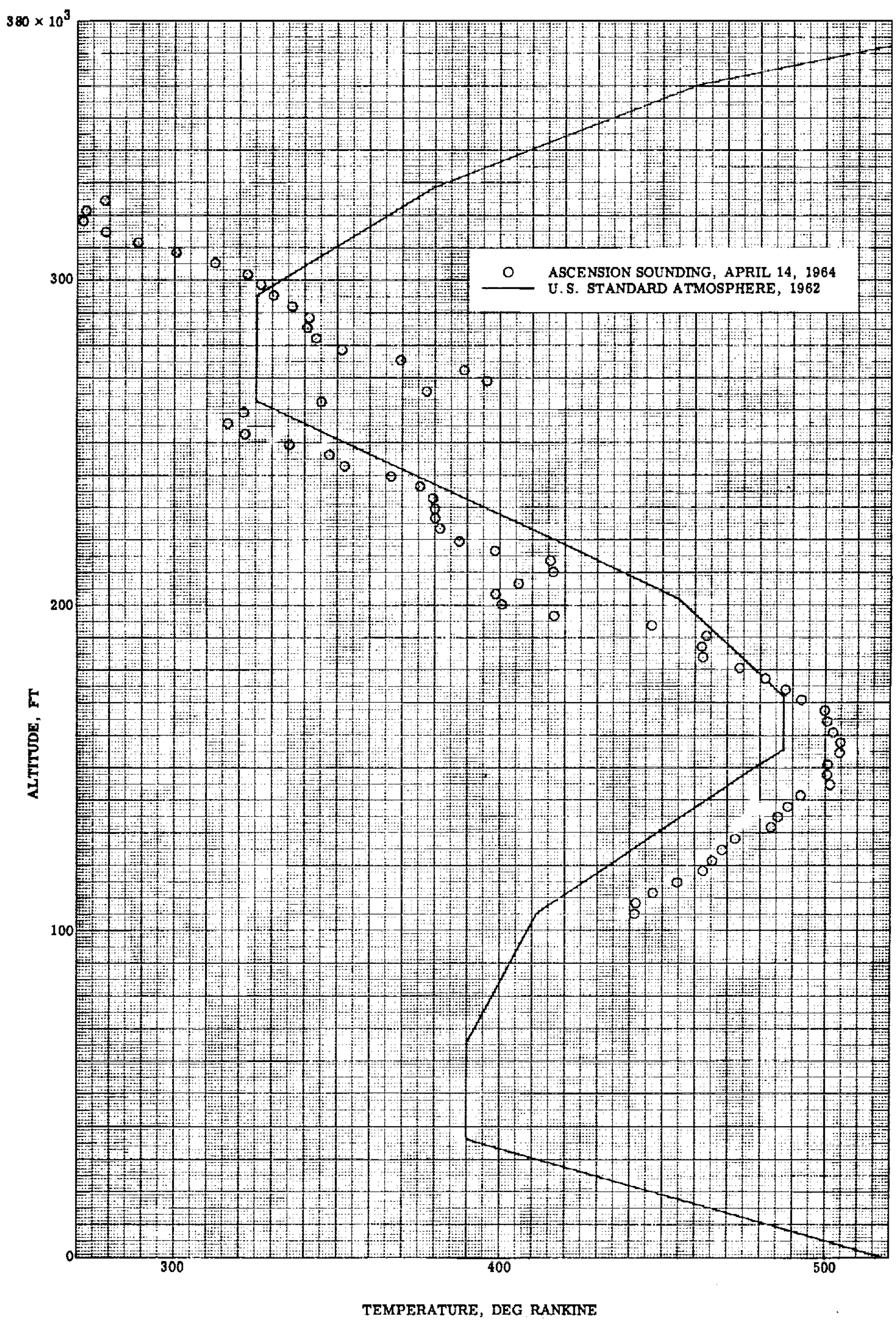
ASCENSION SOUNDING. VARIATION OF DENSITY WITH ALTITUDE



ASCENSION SOUNDING. VARIATION OF DENSITY WITH ALTITUDE



ASCENSION SOUNDING. VARIATION OF TEMPERATURE WITH ALTITUDE



PART 3

MISSION DATA EVALUATION

General Dynamics/Astronautics
Integrated Report No. GDA/BKF 64-018

By Flight Reentry Programs Office
NASA/Langley Research Center

Approved by: David G. Stone
D. G. Stone
Manager, Flight Reentry Programs Office

SECTION 1

INTRODUCTION

The primary objectives of the Project Fire mission were to measure the total and radiative heating rates on the forebody and afterbody of a blunt shape in the environment resulting from entry into the earth's atmosphere at a velocity of 37,000 feet per second. In addition, data were to be obtained on radio signal attenuation and afterbody pressures during reentry.

Because a single calorimeter cannot survive the heat of the entire reentry without surface melting, the forebody of the reentry package was constructed of six layers. The first, third, and fifth layers were made of beryllium and were instrumented with thermocouples to provide temperature time histories from which the total heating rates will be determined. The second, fourth, and sixth layers were ablative heat protection layers, the first two of which were jettisoned at appropriate times during the heat pulse to expose a fresh calorimeter to a clean environment. In this way, three data periods were planned during the reentry which would serve to define the heat pulse. Total radiometers, one located in the stagnation region and another located near the corner of the front face, measured the total radiant heating through quartz windows mounted in each of the forebody layers. In addition, a spectral radiometer measured the spectral distribution of the radiation at the stagnation point over a wavelength range of 2000 to 6000 Å. Because the life of the quartz windows was even shorter than that of the calorimeter in which it was mounted, valid hot-air radiance measurements were obtained only for three periods during the heat pulse. The expected data periods for the total heating and radiative heating measurements are shown in figure 3-1-3.

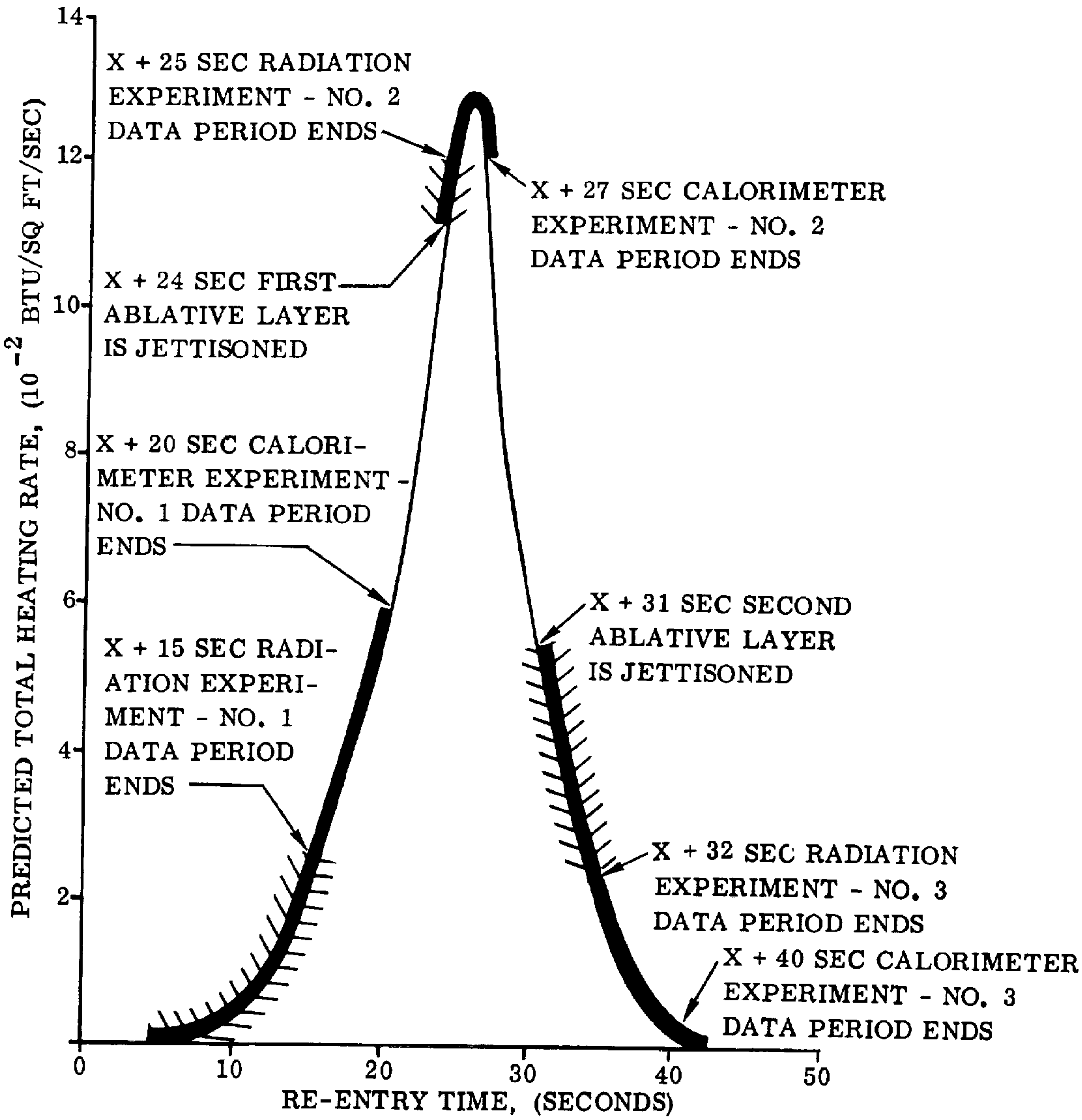
In addition to the measurements on the front face, temperatures of the afterbody surface were also measured. Angular rate gyros and accelerometers on all three axes were provided to determine the trajectory and body motions.

Total radiation to the afterbody was measured, as well as external pressures at two locations on the afterbody. To provide an indication of radio attenuation, the antenna voltage standing wave ratio was measured.

The data from the primary sensors were multiplexed into an FM/FM telemetry system. The data were broadcast continuously in real time. Rebroadcast of data after emergence from blackout was provided for by use of a time delay tape recorder. Considerable support was provided by instrumentation on the ground, and on ships and aircraft deployed in the reentry area.

The purpose of this part of the report is to summarize the plans for data acquisition, and the adequacy of the data coverage for accomplishing the mission objectives.

DATA PERIODS



NOTE: ALL TIMES ARE ESTIMATED
X = 0 TIMES START OF RE-ENTRY

TOTAL HEATING MEASUREMENT

RADIATIVE HEATING MEASUREMENTS

SECTION 2

SUMMARY

The first Project Fire experiment was launched April 14, 1964, at 1642 hours 25.536 seconds e.s.t., from complex 12 at the John F. Kennedy Space Center.

Except for the blackout period in the reentry area, the vehicle was tracked by radar for its complete 4,500-nautical-mile trajectory. A reentry velocity of 37,971 feet per second at an altitude of 400,000 feet and a reentry angle of -14.6° was achieved.

Although the NASA tracking telespectrograph at Ascension Island obtained no spectrographic data, a considerable amount of optical coverage was obtained from a number of stations, both on the ground and from aircraft. These data include trajectory information, events-type information, and spectrographic information.

Real-time telemetry records were excellent before and after blackout. Data covering about two-thirds of the reentry blackout period were transmitted by the delayed-time transmitter on emergence from blackout. These data include information on the radiation, total heating, and afterbody pressures. The reentry package received an abrupt impulse or disturbance in about the middle of the data period at a time near the end of the first calorimeter experiment. Oscillations of the reentry package following this disturbance were of a magnitude sufficient to influence the radiometer measurements. This influence was most prominent directly following the disturbance but by the time of the third beryllium calorimeter experiment the radiometer records show no variations due to the oscillations. These oscillations will make the interpretation and analysis of the reentry heating measurements difficult, particularly for the radiometer data during the time period near which the disturbance was noted.

SECTION 3

SUPPORT IN REENTRY AREA

The primary research data are gathered by the reentry package onboard instrumentation described in Part 4. In order to insure receipt of the data, track the reentry package, establish the occurrence of events, provide supporting information relative to spectra and wake characteristics, a vast amount of support equipment was necessary. Figure 3-3-4 indicates the facilities supporting the Fire reentry in the vicinity of Ascension Island. This figure is a satellite's eye view of the earth's surface showing the reentry trajectory in relation to Ascension Island and the deployment of the ships and aircraft. Two ships - an ARIS, the Gen. H. H. Arnold, and a CI-MA-VI, the Yankee - were on station to monitor the reentry. A third vessel, the DAMP ship, had to leave station because of fuel shortage and was not available for the flight. Six aircraft were deployed in the reentry area. The following table shows the support provided by each of the stations:

Ascension Island

Radar

FPS-16
TPQ-18
TTR

Optics

Telespectrograph
Ballistic, grating, streak, and chopped-streak cameras
IR tracker
IFLOT

Telemetry

TLM-18

Atmospheric soundings

Rawinsonde
Arcas
Nike-Apache

Ships

Radar

ARIS

Telemetry

ARIS
Yankee

Aircraft

Radar

BSD (2)

Optics

GSFC
BSD (2)
ARGMA

Telemetry

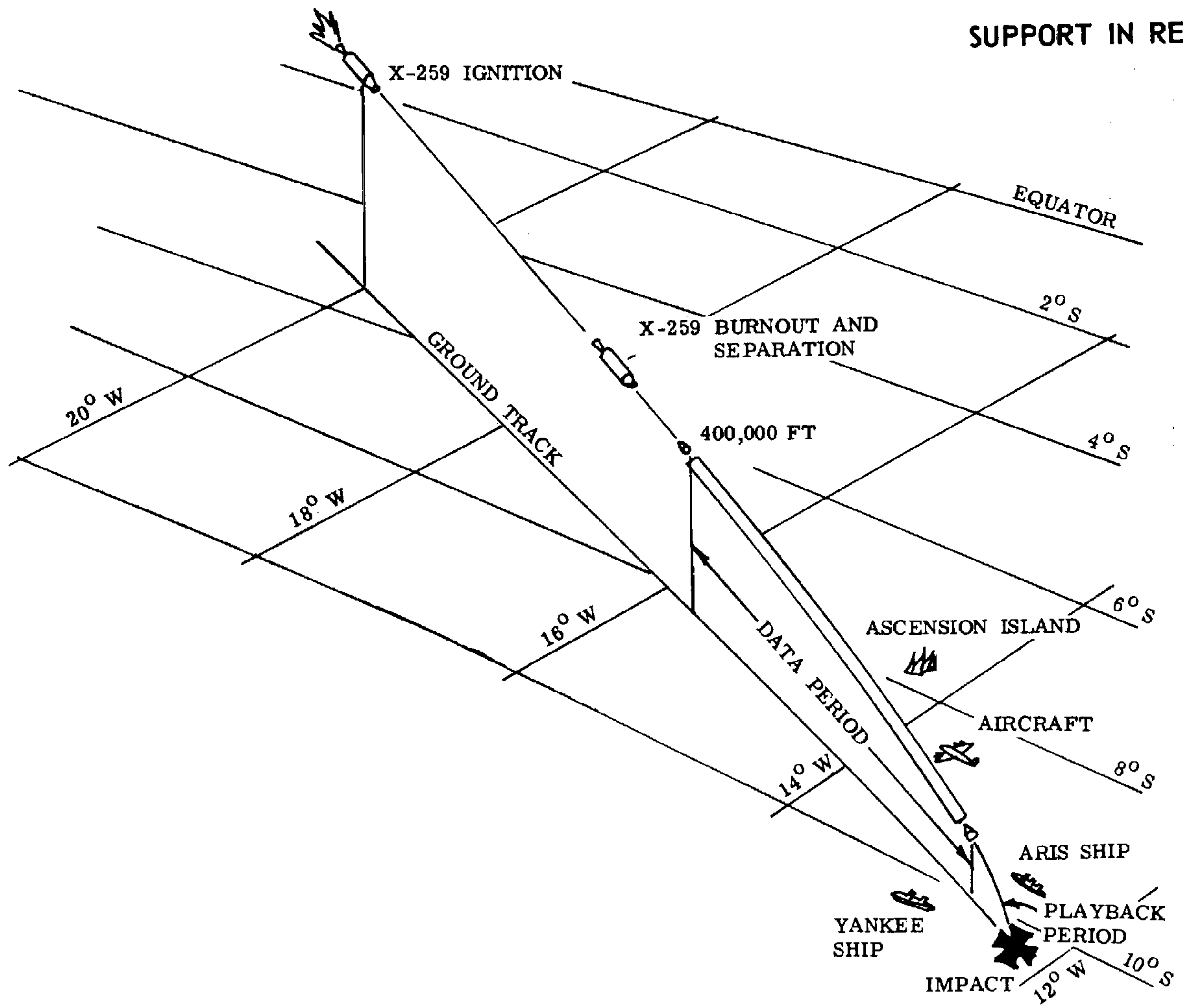
GSFC
AFETR (2)

The Ascension Island FPS-16, the TPQ-18, and Nike-Zeus target tracking radar (TTR) were utilized for obtaining position and velocity data. The TLM-18 was used to receive the onboard telemetry transmission. The Ascension optical instrumentation was provided to obtain events, position, and spectral data. In addition, Arcas sounding rockets and balloons launched by the range and Nike-Apache rockets carrying Goddard pitot-static devices provided accurate measurement of the atmospheric conditions from ground level to an altitude of 400,000 feet.

The two ships, ARIS and Yankee, were deployed to supply telemetry and radar support.

The six aircraft monitored the reentry to provide optical, radar, and telemetry backup information.

SUPPORT IN REENTRY AREA



SECTION 4

DATA ACQUIRED

Telemetry

The two reentry package frequencies (delay time, 237.8 mc; real time, 258.5 mc) were acquired by the Ascension Island TLM-18 at about T + 1260 and received until the start of blackout at T + 1654. These frequencies were reacquired by the TLM-18 on emergence from blackout at T + 1687. The real-time frequency was received until T + 1843, whereas the delay-time frequency was received until T + 1800.

The reentry package real-time signal received by TLM-18 at Ascension was of excellent quality. The reentry package delay-time frequency transmission system, however, suffered a malfunction at about the time of spacecraft spinup (T + 1574) which reduced the signal strength causing periodic dropouts and consequent loss of data during the 2-1/2 cycles of playback received by the TLM-18 at Ascension. At T + 1851.5 the delay-time transmitter output failed apparently as a result of the transmission system malfunction. This caused the failover switch to transfer the output of the delay recorder to the input of the real-time transmitter. (See Part 4 for additional discussion of the failover switch.) This failover enabled the CI-MA-VI (Yankee) ship to obtain partial data from the fourth and fifth playbacks, since this station was receiving the real-time frequency at that time. Ascension Island TLM-18 was the only station to receive good signals from both of the reentry package frequencies.

The Yankee ship received good signals from the reentry package real-time transmission from T + 1836 to T + 1873 seconds.

The ARIS received 6 minutes of delay-time transmission and 7-1/2 minutes of real-time transmission. These data were of very poor quality and no attempt was made to reduce them.

The Goddard and range aircraft received the reentry package real-time transmission, but these signals were also of poor quality and were not used.

Radar

The FPS-16 on Ascension Island gave valid track from T + 1178 to T + 1797, except for the blackout and postblackout reacquisition periods (T + 1660 to T + 1717). These tracking data indicated that the reentry package had a velocity of 37,971 feet per second at 400,000 feet altitude. The flight-path angle at that point was -14.6°. The package impacted at 10.227° south latitude and 12.666° west longitude which is within 18 miles of the planned impact point. It is estimated that the splash time was T + 1965.7. The velocity was about 1 percent greater than expected and the reentry angle was 0.4° shallower than planned.

The TPQ-18 on Ascension Island gave valid track from T + 1178 to beginning of blackout at T + 1654 but did not reacquire after emergence from blackout.

The Nike-Zeus target tracking radar (TTR) tracked the reentry package for 67 seconds starting at T + 1600 seconds. Its velocity data are in good agreement with those obtained from the FPS-16.

No tracking information was obtained from the C-, X-, or L-band radars on the ARIS.

Optics

The NASA telespectrograph (an optical instrument, developed for use with Project Fire, in which the light-gathering power of a 36-inch telescope is coupled with a slitless spectrograph to obtain a spectrographic record of the reentry) did not track quite well enough to obtain the desired spectrograph of the reentry package. Records from the photomultiplier tube and the events camera indicated that the instrument performed satisfactorily; however, tracking was not precise enough to maintain the reentry package image within the field of view (96 seconds of arc) for the time required to expose the infrared film used. The accumulated light recorded by the zero-order photomultiplier tube shows energy bursts within a 15-second period. This was some 50 to 100 times less energy than that required to expose the film.

The following data were obtained from still cameras at Ascension Island:

(a) Single frame streak exposure through grating - two K-24 cameras at station 12.2 and two at station 12.3.

(b) Single frame streak exposure - three 4 by 5 speed graphic cameras at station 12.2 and three at station 12.3.

(c) Single frame, chopped at 10/sec - three 4 by 5 speed graphic cameras at station 12.2 and three at station 12.3.

(d) Ballistic camera, synchronous mode at 10 pps - one camera at station 12.3.

(e) Single frame streak exposure through grating - one 18 by 18 grating camera at station 12.2 and one at station 12.3.

The Intermediate Focal Length Optical Tracker (IFLOT) provided 104 frames of data from its 35mm camera (color) and 166 frames from its 70mm camera (black and white). These data cover the time period from T + 1666 to T + 1716 seconds.

No data were obtained from the IR tracker.

The Goddard aircraft provided the following optical data:

(a) 35mm cine spectral camera, black and white.

(b) 35mm cine events, color and black and white.

(c) 16mm cine events, color.

(d) Single frame chopped at 10/sec - one RC-5 camera and one RC-7 camera.

(e) Single frame streak exposure through grating - one KG-24 with 600 lines/mm grating, one KG-24 with 400 lines/mm grating, one KG-24 with 150 lines/mm grating.

(f) Single frame streak - one K-37 camera.

All of these cameras were in operation from T + 1653 to T + 1680 seconds.

The TTR boresight camera provided 266 frames of usable 16mm black and white film.

In addition to the Goddard aircraft, two BSD aircraft (DC-6, DC-4 meteor) and one ARGMA aircraft (121K) supporting the Nike-Zeus TTR obtained optical and radar signature data.

Atmospheric Soundings

In order to determine the properties of the atmosphere through which the reentry took place, Nike-Apache sounding rockets carrying Goddard pitot-static tube payloads were launched 4 hours and 12 hours after impact of the reentry package. The launches have provided accurate information on the variation of the density, pressure, and temperature with altitude up to an altitude of about 400,000 feet.

SECTION 5

TELEMETRY COVERAGE

As pointed out previously, and as detailed more fully in Part 4, the delayed-time transmission system suffered a reduction in signal strength which resulted in dropouts and consequent partial loss of data obtained during the 32.9-second-long blackout period. A number of playbacks of the delayed-time data were received after emergence from blackout, and since the dropouts for each playback do not coincide, it is possible to dovetail the four playback cycles received and thus recover the data for a large portion of the blackout period.

Figure 3-5-4 illustrates the amount of data recovered in relation to the significant events. The reentry package separated from the spent Antares II motor at 1640.5 seconds, but the experiment period is considered to begin at an altitude of 400,000 feet which is reached at 1647 seconds. The experiment is completed at 1689 seconds, at which time the "erase-record" function of the onboard continuous loop tape recorder is disabled. Thus, the total duration of the experiment is 42 seconds of which 32.9 seconds occur during blackout.

The principal reentry events and the times of their occurrence are listed as follows:

Event	Time from lift-off, sec		Time from 400,000 feet, sec	
	Planned	Actual	Planned	Actual
R/P separation	1640.2	1640.46	---	---
400,000 feet	1644.95	1647.4	0	0
Timer start	1666.0	1666.6	21.05	19.2
Eject 1st phenolic layer*	1669.0	1669.6	24.05	22.2
Eject 2nd phenolic layer*	1676.0	1676.6	31.05	29.2
Disable "erase-record"	1688.65	1689.0	43.17	41.6

* Times given are for ejection signal; nominal time required for ejection of phenolic layers is 0.5 second.

It can be seen that the phenolic asbestos layers were jettisoned and thus provided the three measuring periods during the experiment. However, the timer was apparently started about 1.8 seconds earlier in the experiment than was planned (with respect to the 400,000-foot reentry point), with the result that the phenolic asbestos layers were ejected earlier than planned. Although the second and third experiment periods were therefore initiated earlier than planned, they still occurred during the heating period of interest.

Excellent reception was obtained for the real-time transmission, both before and after blackout, so that good data were obtained from this link for the first 7 seconds and the last 2.2 seconds of the total data period.

The data obtained from each of the four delayed-link playbacks superimposed on the real-time scale are shown on the figure 3-5-4. The blank spaces represent the periodic loss of data. The four cycles are added together on the lower line to give the resultant telemetry coverage. This line indicates that about two-thirds of the blackout period of 32.9 seconds is covered by usable telemetry records and that about three-quarters of the total data period of 42 seconds is covered by usable telemetry records.

Figure 3-5-5 shows the research data available from the telemetry in relation to the performance of the onboard measurement systems. Also shown for reference are curves giving the theoretical stagnation-point heat fluxes ($\dot{q}$) to which the calorimeters and the radiometer were exposed. These curves are estimated on the basis of the measured characteristics of the atmosphere at Ascension Island and the actual reentry trajectory.

The radiometers functioned continuously as indicated by the top bar, and the quartz windows in each beryllium calorimeter remained optically clear for approximately the brief periods indicated by the narrow bar, as expected.

The second bar indicates the exposure times and useful life of the three successive beryllium calorimeters as obtained from the flight measurements.

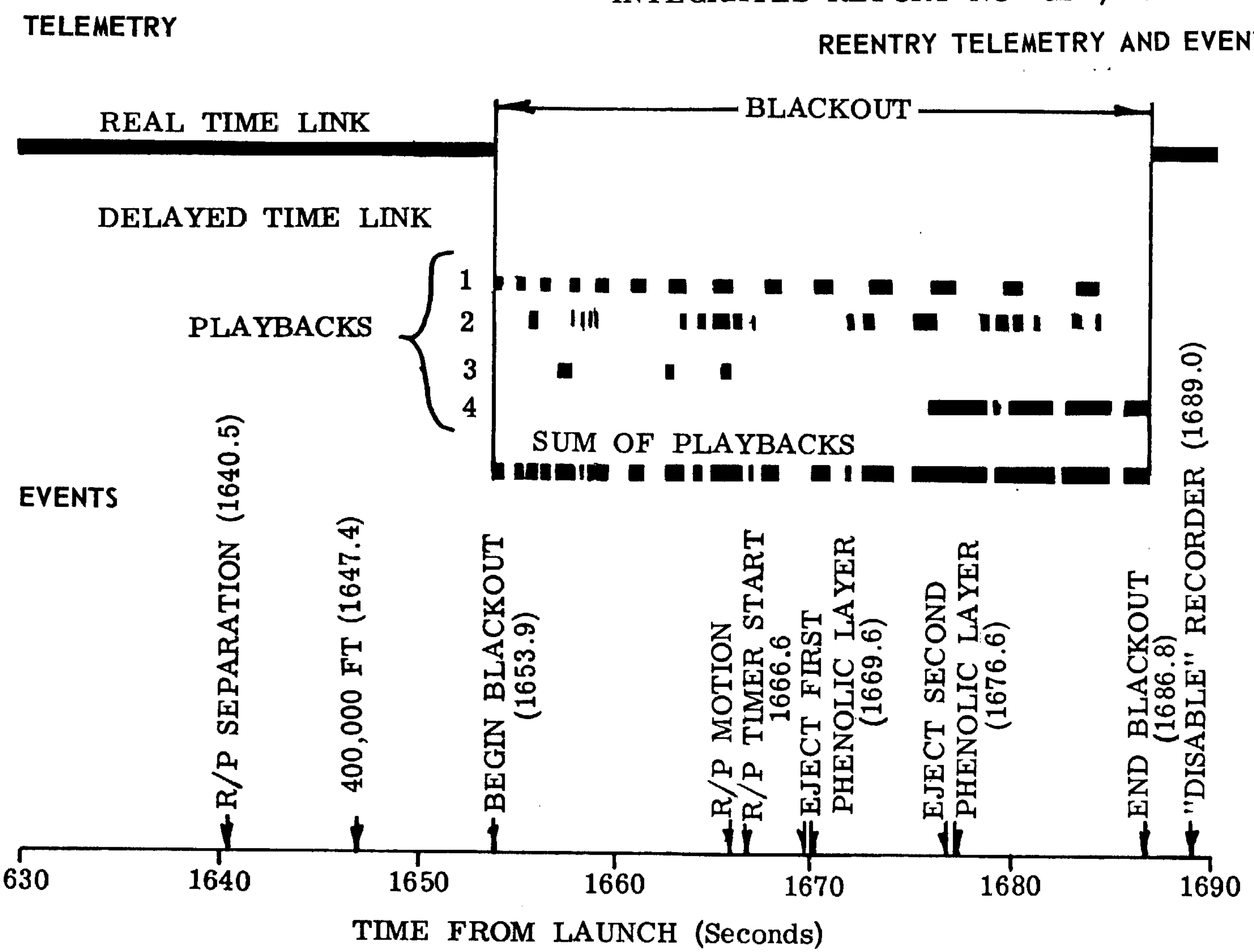
Afterbody temperature sensors functioned continuously.

Afterbody pressure sensors functioned as planned to follow the aerodynamic pressure rise up to the limits of the sensors (which were off scale during maximum pressure) and followed the pressure drop toward the end of the measurement period. The lengths of the measurement periods are imposed by the range of the instrument.

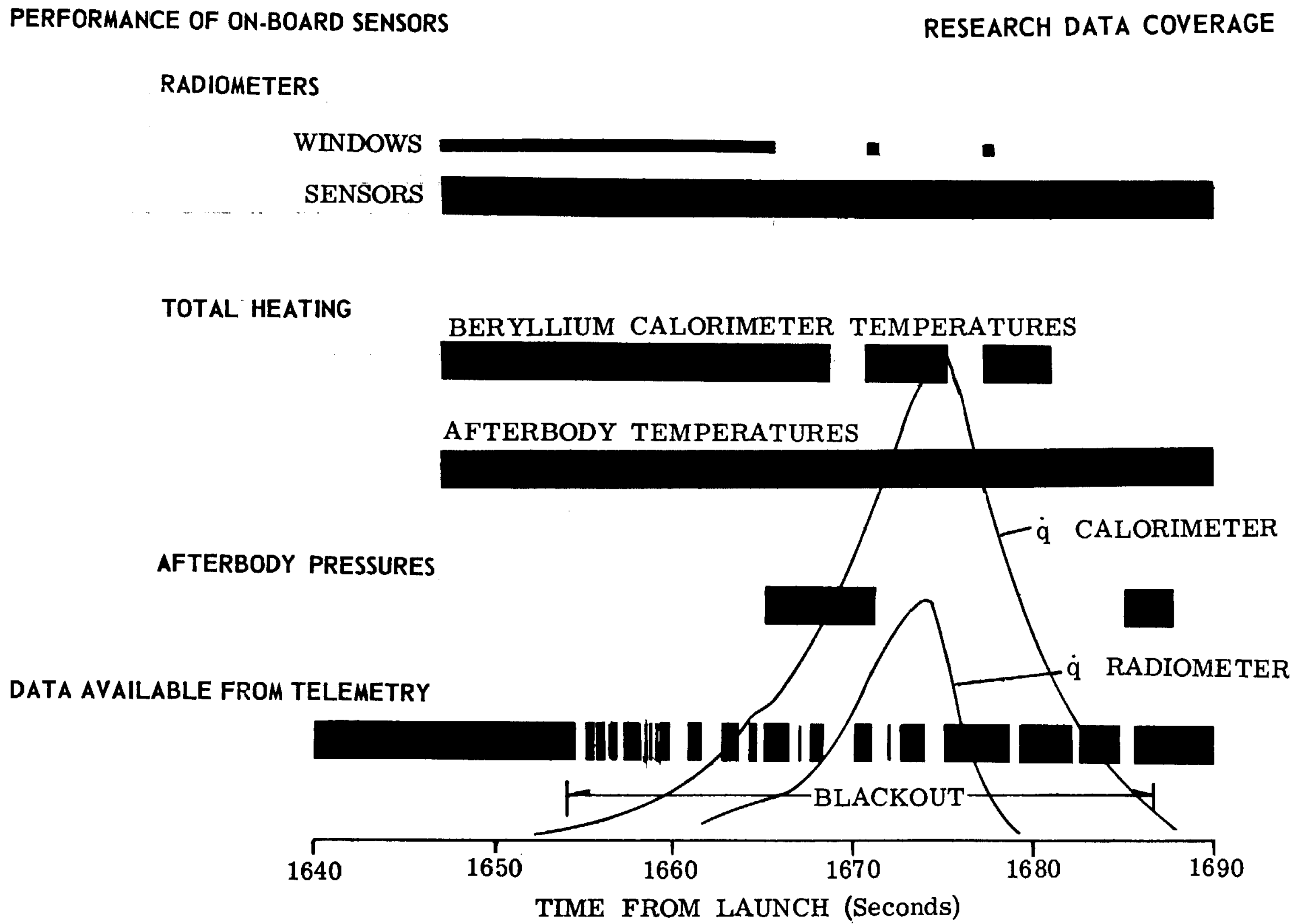
As indicated previously, the phenolic asbestos layers were jettisoned about 1.8 seconds earlier in the heat pulse than planned. The consequence of the earlier jettisoning may be seen from figure 3-5-5. The period for acceptable radiometer data after the first phenolic layer is ejected lies to the left of the peak of the radiation curve, and therefore the radiation levels seen by the instrument are considerably less than the peak value. Thus, it will not be possible to establish with any certainty the maximum radiation heating rate but the levels and shape of the radiation heating rate curve can be established.

Preliminary examination of the Fire data has indicated that the reentry package received a sharp impulse or disturbance in about the middle of the data period at T + 1666 seconds, as shown in figure 3-5-4 by the tick labeled "R/P motions." Examination of records indicates that the oscillations experienced by the reentry package immediately following this disturbance were of a magnitude sufficient to influence the radiometer data. This influence was most prominent directly following the disturbance but by the time of the third beryllium calorimeter experiment the radiometer records show no variations due to the oscillations. The analyses of the Project Fire reentry heating measurements, particularly the radiometer data for the time period directly following the disturbance, are greatly complicated by the presence of these body motions as the interpretation of the measurements is dependent upon a definition of the body motions. The assessment of the reentry package body motions, however, is complicated by the fact that the oscillations exceeded the range of the yaw-rate gyro, the roll-rate gyro was inoperative, and the pitch-rate gyro operated erratically, as described more fully in Part 4. More refined theoretical methods are currently being employed in an attempt to compute the complete motion time history and evaluation of data is continuing in an attempt to determine the cause of the disturbance.

REENTRY TELEMETRY AND EVENTS



RESEARCH DATA COVERAGE



PART 4

REENTRY PACKAGE PERFORMANCE

GENERAL DYNAMICS/ASTRONAUTICS
INTEGRATED REPORT NO. GDA/BKF64-018
REPUBLIC REPORT NO. 499-32-II

APPROVED BY: E. Uttendorfer for I. Singer

I. SINGER, PROGRAM MANAGER
PROJECT FIRE

SECTION 1

DESCRIPTION

The Project FIRE Reentry Package Subsystem consists of two airborne packages: 1) an adapter, and 2) the Reentry Package.

Reentry Package Adapter

The adapter forms the transition from the Velocity Package Subsystem to the Reentry Package and houses the reentry package separation system, the Antares II adapter tumbling system, and the umbilical connector.

The separation system consists of a coil spring and an explosive nut for deploying the Reentry Package. It also utilizes a tumbling rocket mounted on the Reentry Package adapter to increase the separation distance between the Reentry Package and the spent reentry stage. The separation system power is provided by a redundant pair of remotely activated batteries, controlled by contact closures in a pair of redundant timers, each of which is started by a contact closure from the Velocity Package prior to reentry stage separation.

Figure 4-1-5 shows the Reentry Package mounted on the adapter, which in turn is attached to the Velocity Package. The tumbling rocket is shown on the adapter wall.

Reentry Package

The Reentry Package (R/P) may be considered to consist of a number of subsystems which are briefly described in the following.

Structural

The R/P is made up of a forebody and an afterbody, shown schematically (with the adapter) in Figure 4-1-6. The forebody and the afterbody are joined at a pressure-cooker, lid-type sealed joint. The forebody is an aluminum structure covered on the outside with a composite heat shield and reinforced with an instrument mounting grid which is welded to the inside. The composite heat shield consists of alternate layers of beryllium and phenolic asbestos. The afterbody consists of an aluminum fiberglas structural combination covered with a laminate of Min-K and phenolic asbestos which is coated with a Sylgard formulation.

Primary Power

Primary power is supplied to the Reentry Package subsystems by five types of batteries: 1) the auxiliary battery located in the aft portion of the velocity package, which supplies the inflight instrumentation power until approximately V/P spin-up; 2) the instrumentation battery located in the R/P, which supplies instrumentation power after power transfer from the auxiliary battery; 3) the C-band beacon battery located in the R/P, which supplies C-band power; 4) a pair of heat shield ejection batteries located in the R/P, which supply power to the pyrofuze link; and 5) the previously noted separation and tumbling system batteries.

Data Sensing

Data sensing is accomplished by a variety of sensors located within the R/P. Their purpose is to measure temperatures resulting from the heat flux incident upon the exterior of the R/P; measure radiant energy resulting from the heated shock layer; sense R/P motion during flight; provide a time reference to correlate all R/P events; measure external pressures to assist in flow field analysis; and, by means of an internal pressure sensor and internal thermistors, to make available diagnostic tools in the event they are required. The locations of many of these sensors are shown schematically in Figure 4-1-7. The temperature is measured by calorimeters (not shown in Figure 4-1-7) which consist of three types: 1) 36 beryllium calorimeters, 12 of which are imbedded in each of the three beryllium heat shields along three radii, 120° apart, at four radial locations. Each calorimeter contains four thermocouples imbedded at various depths. 2) 20 phenolic asbestos calorimeters, 12 of which are imbedded in the outermost phenolic shield in a manner similar to that noted for the beryllium shields. The remaining 8 are similarly located in the second phenolic asbestos heat shield, with the exception that one radius is eliminated. Each phenolic asbestos calorimeter contains three thermocouples imbedded at various depths. 3) 12 gold slug-type calorimeters located along three longitudinal rows, 120° apart, in the afterbody. Each gold calorimeter has two thermocouples (one of which is redundant) located at the rear face of the gold slug.

The radiant energy is sensed by four radiometers, two of which are contained in a single unit called the spectral/total radiometer which measures the radiant energy in the stagnation region of the gas gap. The spectral radiometer continuously scans over a wavelength range of 2000 to 6000 Å, whereas the total radiometer senses the integrated radiant energy in the wavelength range of approximately 2000 Å to 4-6 microns as limited by the radiometer windows. The other two radiometers are of the total type, one of which is located in the outboard portion of the forebody and the other is located in the afterbody (see Figure 4-1-7).

The vehicle motion is sensed by an attitude sensor which consists of three rate gyros used to sense rates about each of the three R/P orthogonal axes, and five linear accelerometers. Three of the accelerometers are mounted along the R/P longitudinal axis to sense reentry decelerations and boost accelerations. The other two accelerometers are mounted in each of the two orthogonal axes.

The onboard time reference is obtained by means of a time code generator whose output is a continuous serial binary time code.

External pressure is sensed by each of two pressure transducers located in the afterbody. Each transducer has an associated power converter (see Figure 4-1-7).

The remaining diagnostic sensors are located at various positions within the R/P.

Data Acquisition

The data acquisition equipment,which prepares the sensed data for transmission to the ground loop, is comprised of the signal conditioner; 18 x 5, 30 x 2.5, and 30 x 5 PAM commutators; a PDM multicoder which contains three 90 x 10 PDM commutators; an FM multiplexer; and a delay recorder. Schematic location of these is shown in Figure 4-1-8.

The signal conditioner provides regulation, identification and calibration, monitoring, and pedestal generation. The 18 x 5 PAM commutator contains the accelerometer, roll rate, and internal pressure data; the 30 x 2.5 PAM commutator contains the diagnostic data (monitor point and internal temperature), plus the external pressure and radio attenuation data; the 30 x 5 PAM commutator contains afterbody temperature data. All of the forebody temperature data are contained in the 90 x 10 PDM commutators; in addition, some afterbody information is contained in the third PDM commutator.

The FM multiplexer combines the data signals into a complex waveform for modulating the VHF FM transmitters. IRIG channels 6 through 14, C, E, and a non-IRIG standard 100 kc are used. Yaw and pitch rate data are on channels 6 and 7; total radiometer data are on channels 8, 9, and 10; the time code is on channel 11; the 30 x 2.5 PAM data are on channel 12; the 18 x 5 PAM data are on channel 13; the 30 x 5 PAM data are on channel 14; spectral radiometer data are on channel C; and the 90 x 10 PDM data are on channel E. The 100 kc channel is used for tape speed compensation. The delay recorder stores one track of FM multiplexed for a nominal 45-second delay.

Data Transmission

Data are transmitted via two VHF transmitters - one real time and one delay time - which feed the antennas. The real time assigned frequency is 258.5 megacycles and the delay time frequency is 237.8 megacycles. Delay data are identical to real time except for the nominal 45-second delay. In the event of a delay time transmitter failure after the blackout period, a failover switch is included. It is enabled by the acceleration switch and timer (included for simplicity in the heat shield separation system), and transfers the output of the delay recorder to the input of the real time transmitter, if required. The output power of the delay time transmitter is monitored by an rf power sensor. The incident and reflected power from the real time transmitter is monitored by a directional coupler.

Heat Shield Ejection

The ejectable phenolic asbestos heat shields are each secured by a pyrofuze link which has a redundant set of initiators. An acceleration switch initiates a timer which provides power to a calorimeter switch. A breakwire switch in the system inhibits the firing in the event the beryllium has not melted. The calorimeter switch initiates pyrofuze firing and switching of PDM commutators. Power for the pyrofuze initiators is provided by the previously mentioned heat shield ejection batteries. The locations are shown in Figure 4-1-9.

C-Band Beacon

An onboard C-band beacon is provided to assist in trajectory tracking of the R/P. The beacon is powered by the previously mentioned beacon battery and has a four-port circulator which prevents interference between beacon interrogation and output signals. The beacon feeds an antenna mounted on the R/P adapter prior to R/P separation and an antenna in the R/P apex after separation. Locations of the equipment are shown in Figure 4-1-10.

Cooling

An onboard cooling package provides cooling for the ground and inflight operations of the R/P. Prior to lift-off, Freon 114 is used as the coolant and is supplied through the umbilical. After lift-off, water supplied from the reservoir in the cooling package is used. In both cases, the cooled air is passed through a manifold (see Figure 4-1-11).

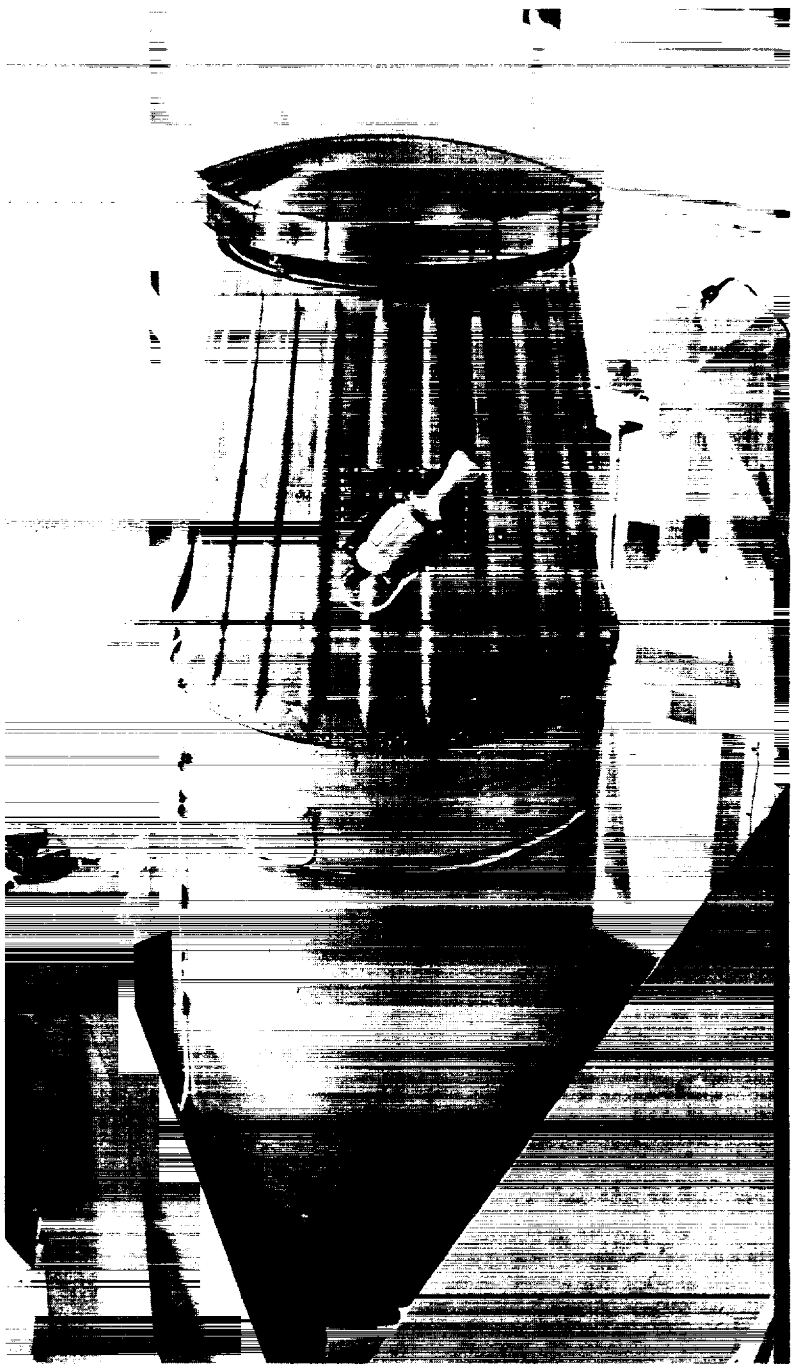
Figure 4-1-12 is a schematic block diagram showing the interrelation of the data sensing, data acquisition, data transmission, and heat shield ejection systems.

Figure 4-1-13 shows (schematically) a build-up of the R/P.

Figure 4-1-14 shows the R/P in the open condition, mounted in the handling rig, and with many of the previously mentioned components visible.

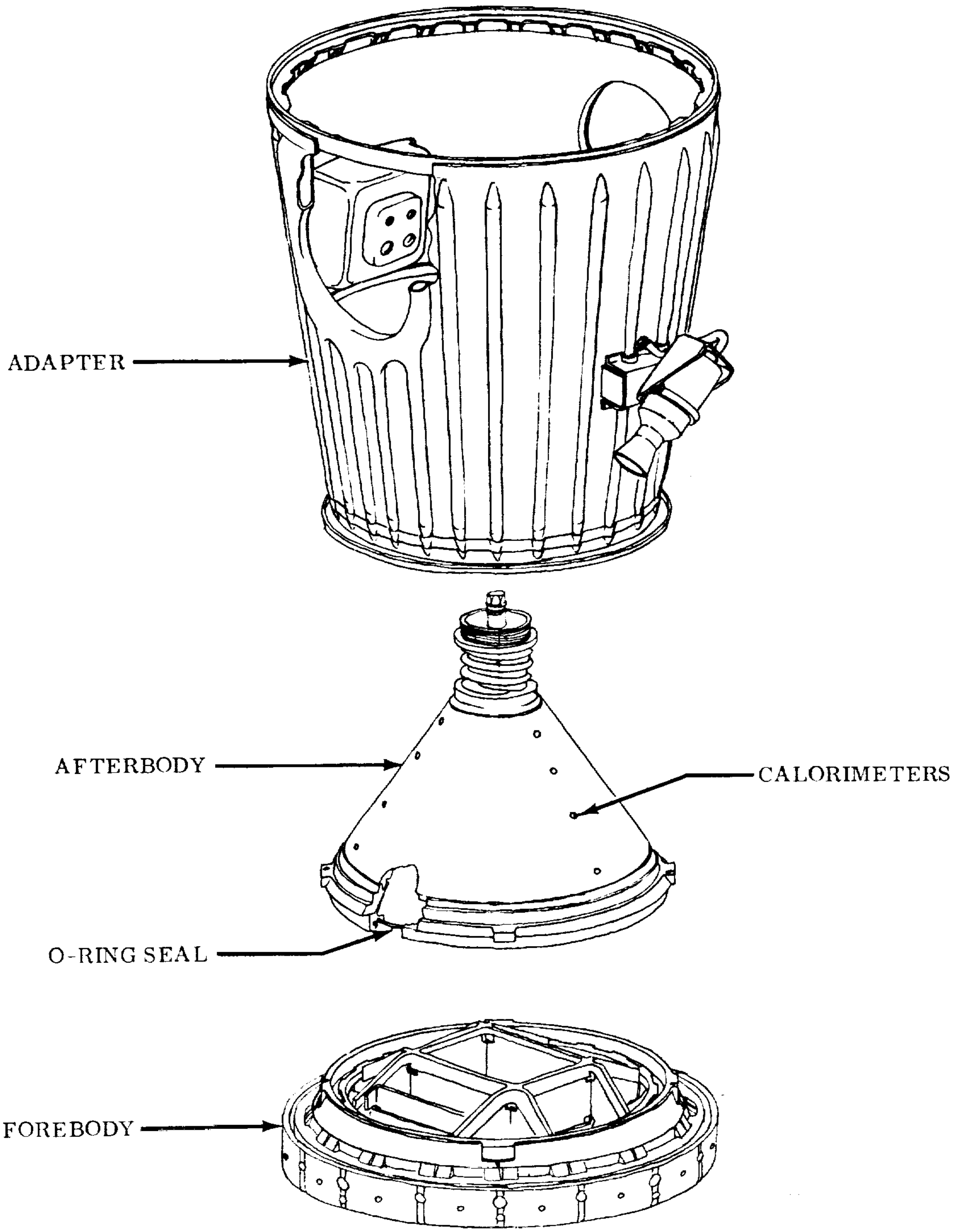
Figure 4-1-15 shows the R/P in the reentry flight configuration.

REENTRY PACKAGE MOUNTED ON THE ADAPTER

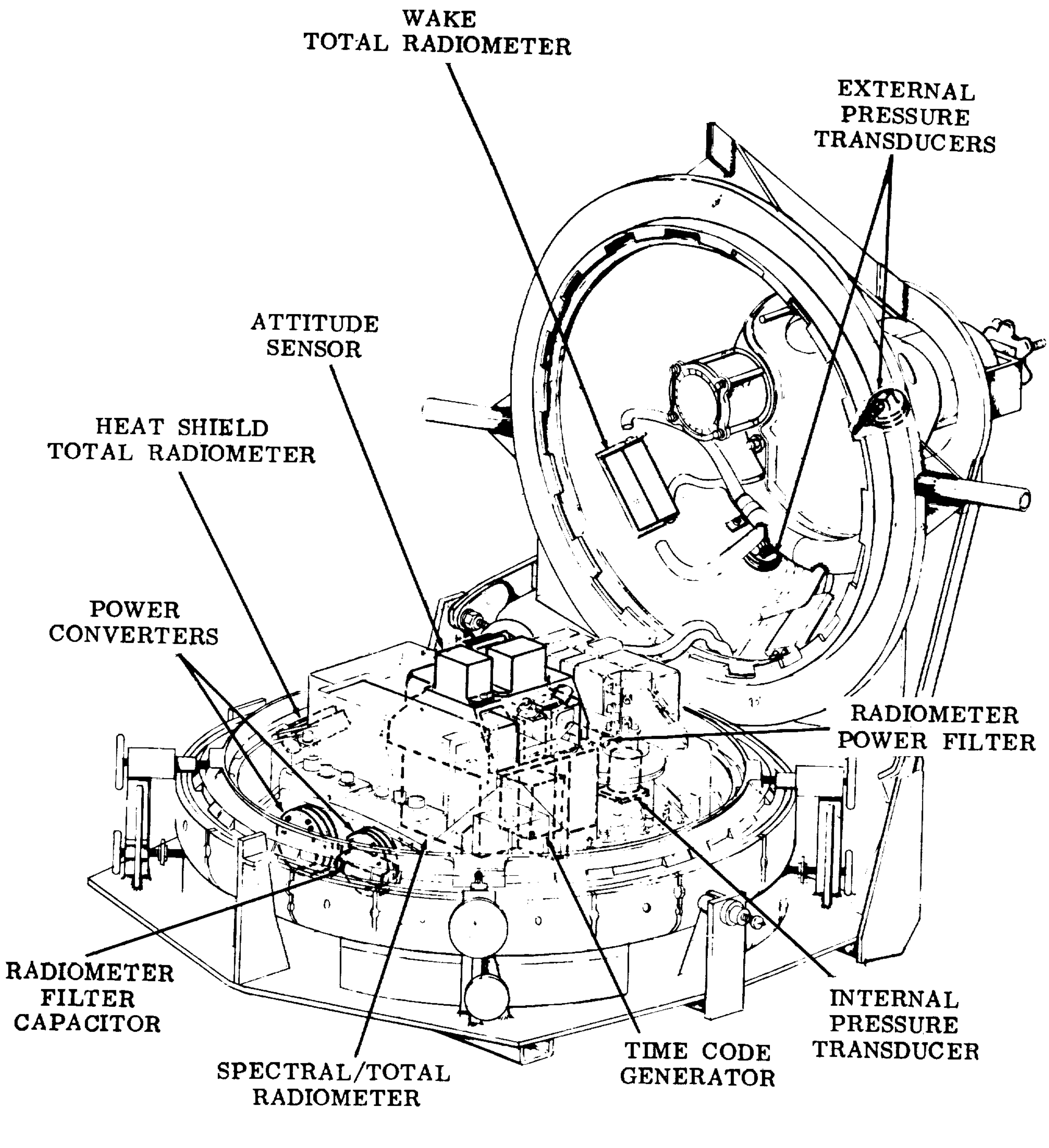


REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-6
INTEGRATED REPORT NO. GDA/BKF64-018
RAC REPORT NO. 499-32-II
DESCRIPTION

SCHEMATIC OF REENTRY PACKAGE AND ADAPTER

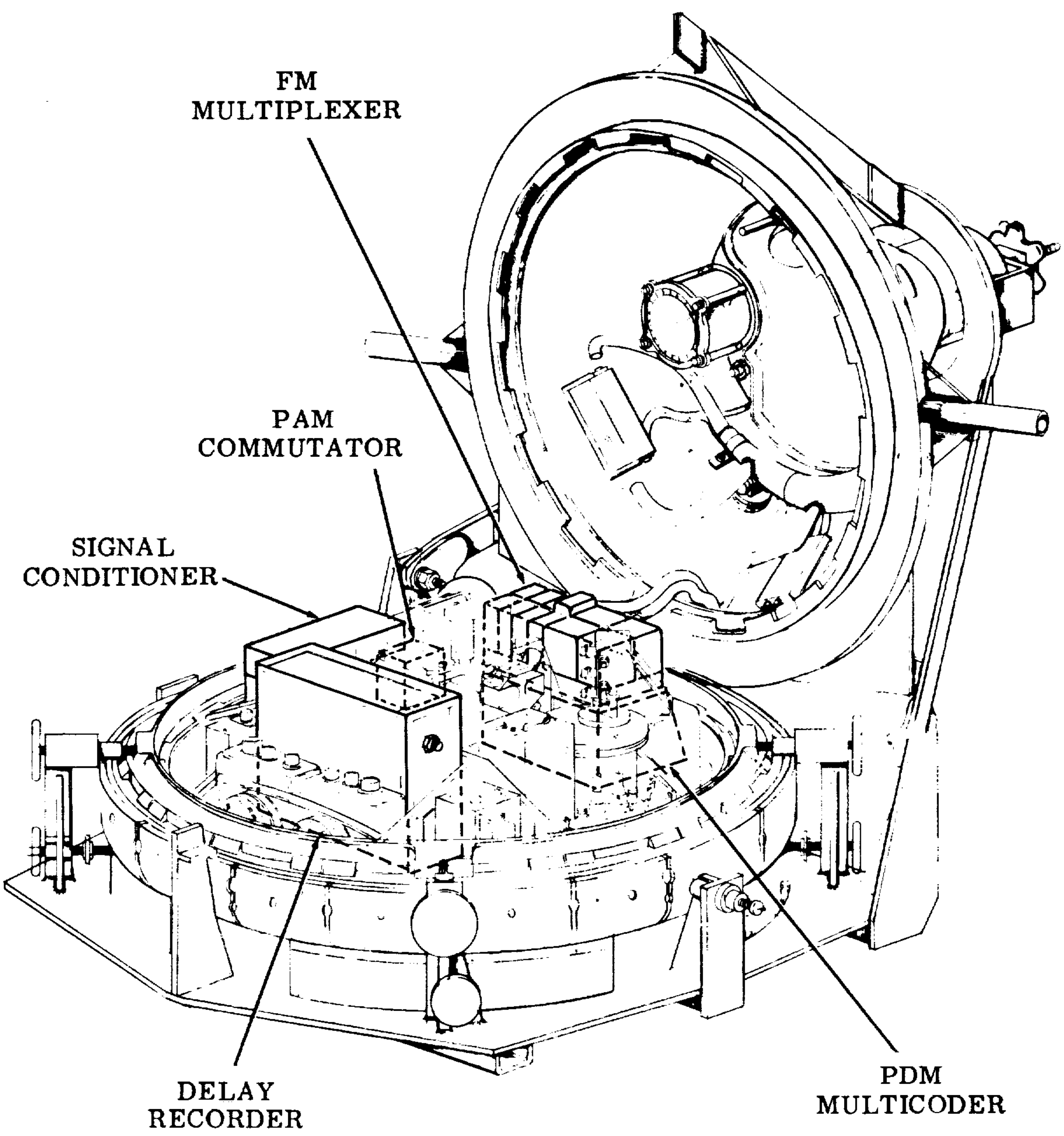


SCHEMATIC OF DATA SENSING SYSTEM

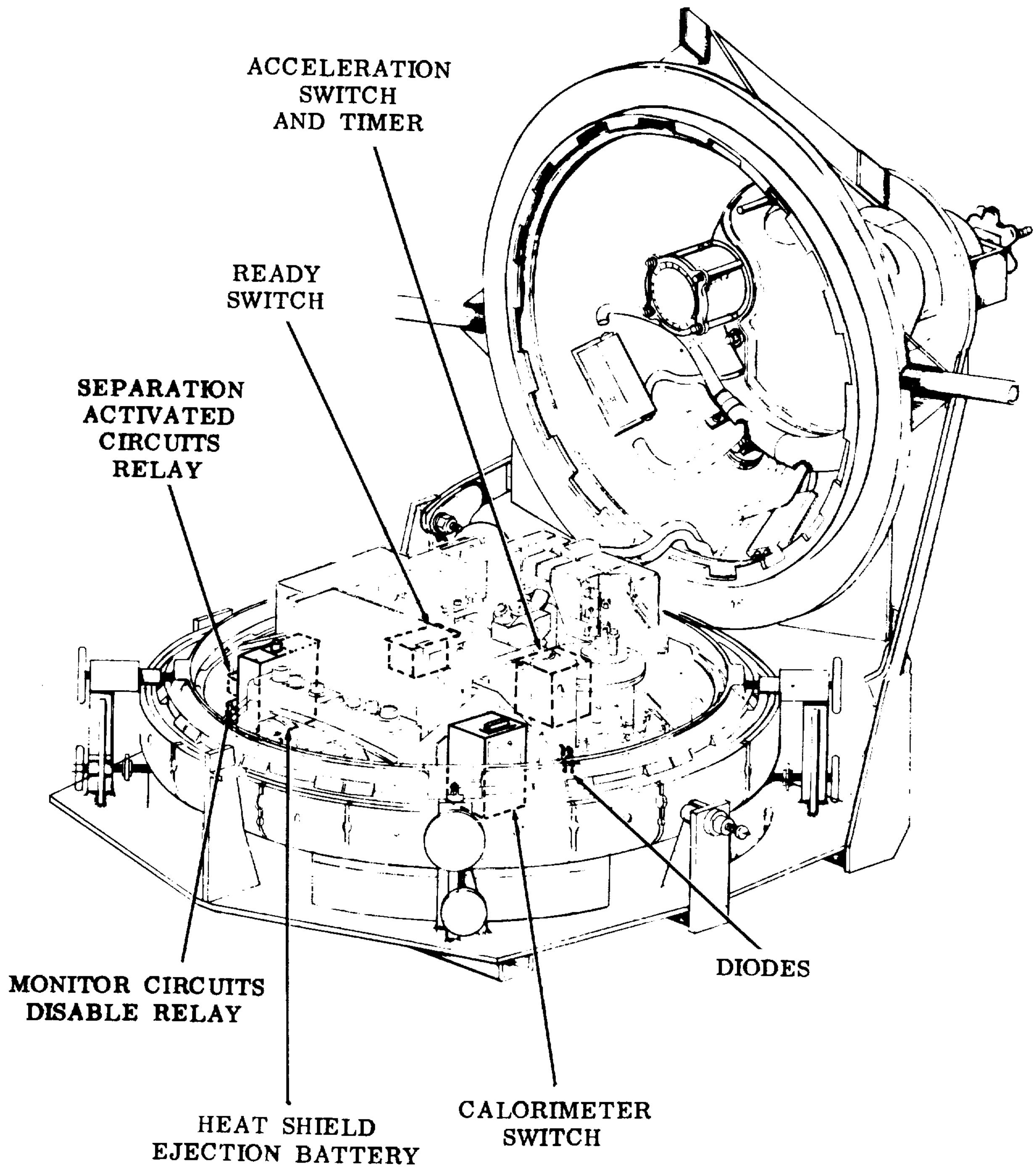


REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-8
INTEGRATED REPORT NO. GDA/BKF64-018
RAC REPORT NO. 499-32-II
DESCRIPTION

SCHEMATIC OF DATA ACQUISITION SYSTEM

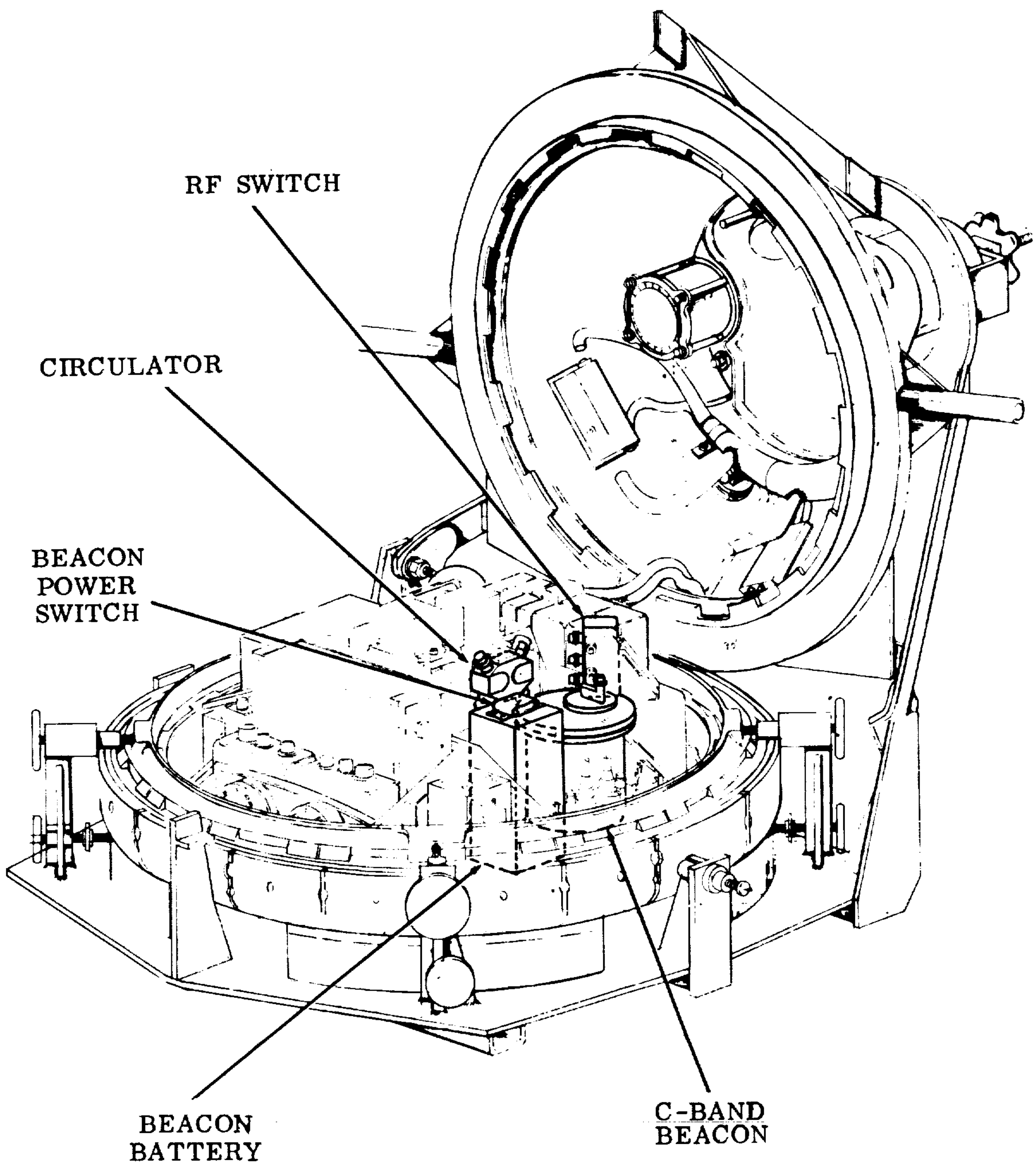


SCHEMATIC OF HEAT SHIELD EJECTION SYSTEM

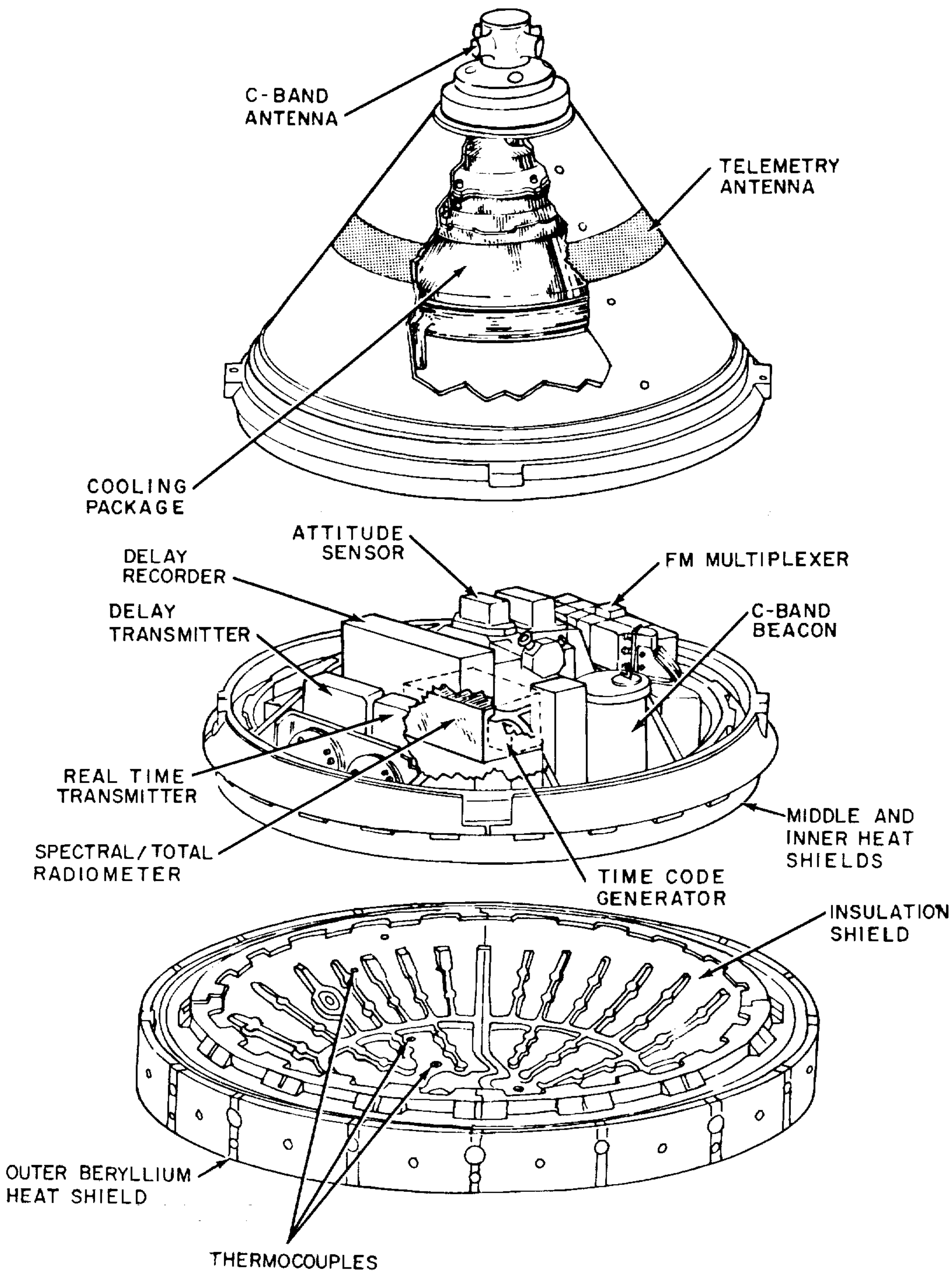


REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-10
INTEGRATED REPORT NO. GDA/BKF64-018
RAC REPORT NO. 499-32-II
DESCRIPTION

SCHEMATIC OF C-BAND BEACON SYSTEM

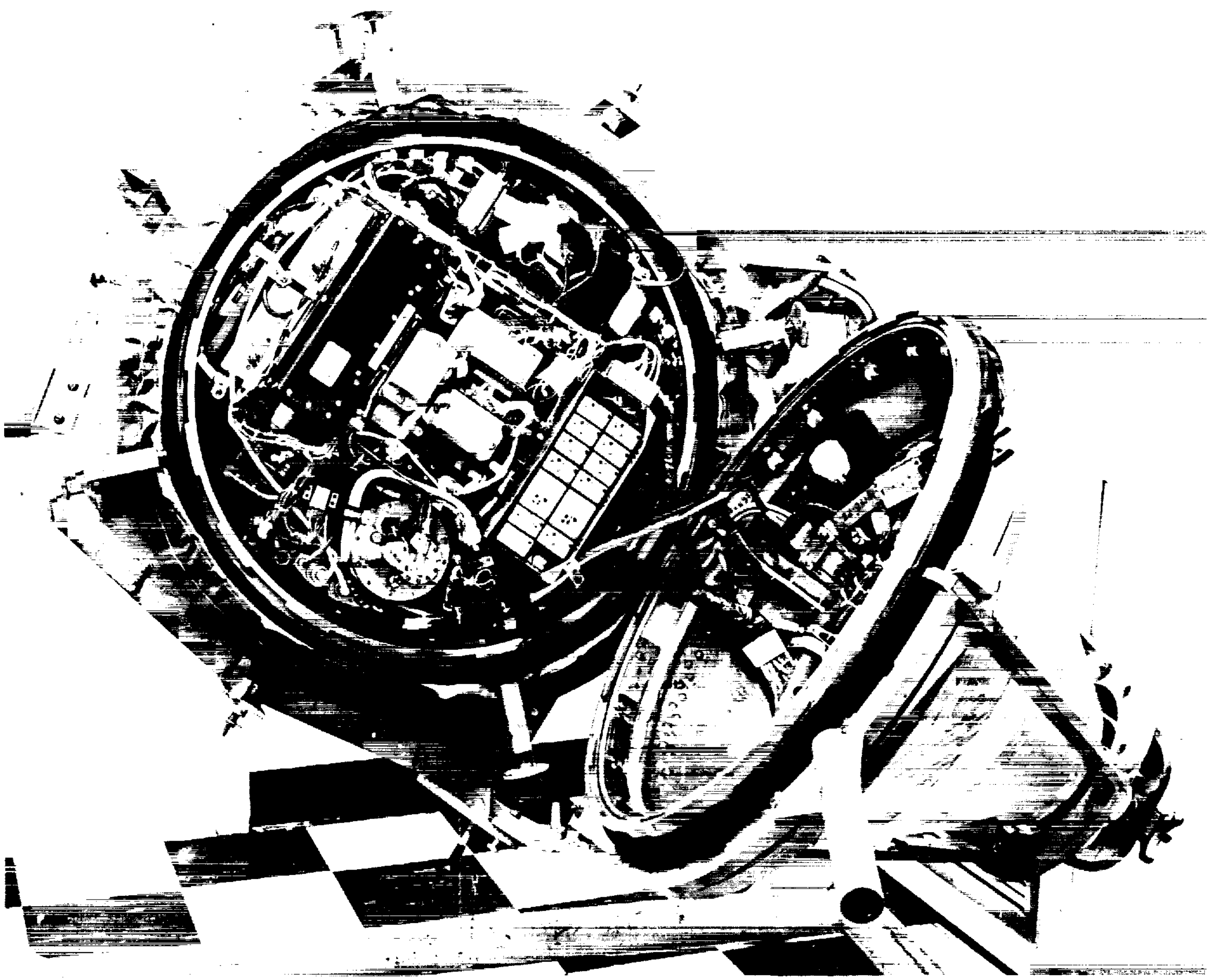


BUILD-UP OF THE REENTRY PACKAGE



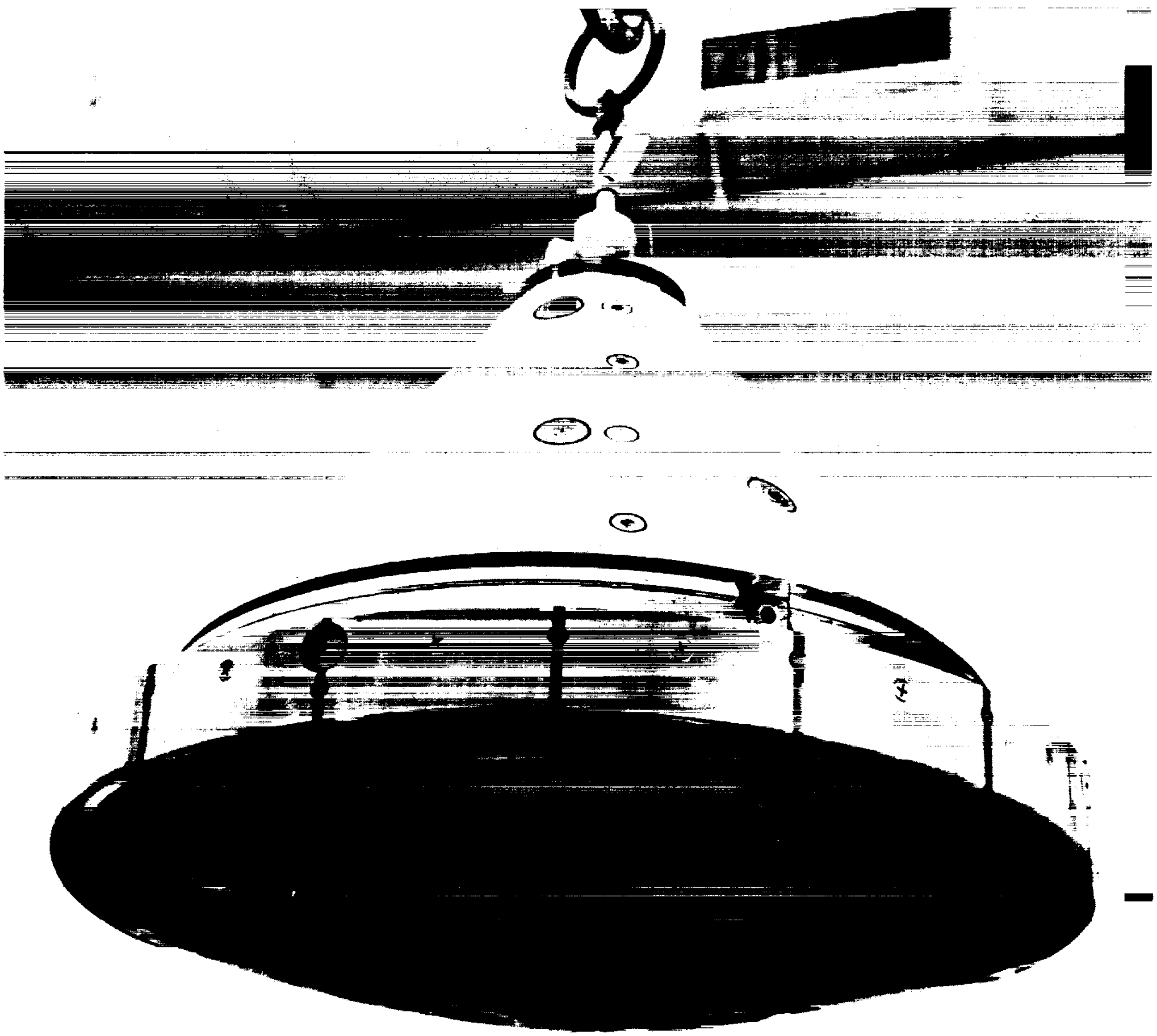
REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-14
INTEGRATED REPORT NO. GDA/BKF64-018
RAC REPORT NO. 499-32-II
DESCRIPTION

REENTRY PACKAGE OPENED AND MOUNTED IN THE HANDLING RIG



REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-15
INTEGRATED REPORT NO. GDA/BKF64-018
RAC REPORT NO. 499-32-II
DESCRIPTION

REENTRY PACKAGE IN THE REENTRY FLIGHT CONDITION



SECTION 2

ACCOMPLISHMENT OF FLIGHT OBJECTIVES

The purpose of the Reentry Package was to obtain data for the following five (5) primary flight objectives:

1) Definition of Total Heating
2) Definition of the Gas Cap Radiance
3) Determination of the R-F Signal Attenuation
4) Acquisition of Information on Materials Behavior
5) Definition of Reentry Motion

Since complete analysis of the flight data is beyond the scope of Contract NAS 1-1945, a quantitative review of the attainment of the flight objectives is precluded; however, a qualitative review is possible.

Although 100% data recovery during the telemetry blackout was not attained (deviations in performance are discussed later in this report), a substantial portion was obtained. Combination of the receiving station data and the various playbacks results in data recovery of about 65% (21.2 seconds out of a blackout period of 32 seconds).

Attainment of the five flight objectives is briefly summarized, as follows:

Flight Objective 1 - Temperature data were obtained from approximately 66% of the thermocouples.

Flight Objective 2 - All radiometers functioned throughout the flight.

Flight Objective 3 - The directional coupler functioned throughout the flight. In addition, information was gained from the sharp entry into and exit from telemetry blackout, as well as from the loss and recovery of C-band beacon data.

Flight Objective 4 - Time-temperature responses of the working thermocouples were obtained.

Flight Objective 5 - Data were obtained from all five accelerometers and from the yaw rate gyro. Partial data were obtained from the pitch rate gyro. This discrepancy, coupled with loss of the roll rate gyro, will complicate the definition of reentry motion; however, analysis of the Reentry Package telemetry strength indicates a roll rate of approximately 3 rps, compensating in part for the loss of the roll rate gyro.

SECTION 3

REENTRY PACKAGE FLIGHT SEQUENCE

The following is a tabulation of the planned and the actual R/P flight sequences. Since the actual times are so close to the planned times, comment is not required.

	Event	Planned	Actual
1.	Two-Inch Motion	T-0 sec	T-0 sec
2.	Start R/P Separation Timer	T+1567	T+1567.56
3.	Total Radiometer Zero Calibration (flag stop)	1569	------*
4.	V/P Spin-Up	1574	1574.31
5.	Completion of Switchover to Internal Battery Power (V/P separation)	1577	1577.32
6.	Antares II Ignition	1580	1580.31
7.	Antares II Burnout	-----	1613
8.	R/P Separation from Adapter	1640	1640.47
9.	Start of T/M Blackout	-----	1654
10.	Start of C-Band Blackout	-----	1660.2
11.	10g Reentry Deceleration Switch Closure	1667	1666.6
12.	Ejection Signal: First Phenolic Heat Shield	1670	1669.6
13.	Ejection Signal: Second Phenolic Heat Shield	1677	1676.6
14.	End of C-Band Blackout	-----	**
15.	End of T/M Blackout	-----	1686
16.	Disable of Recorder Erase/Record	1689	1689
17.	Failover	-----	1851.5

(Tabulation continued on Page No. 4-3-2)

REENTRY PACKAGE FLIGHT SEQUENCE

	Event	Planned	Actual
18.	Number of Delay Loop Cycles Received from TLM-18 at Ascension	2-1/2 to 3	2-1/2
19.	Delay Time Loop Length	45 sec	43.95 sec

* The flag stop was indiscernible.
**Data not available.

SECTION 4

PERFORMANCE DEVIATIONS

In general, the R/P performance was excellent. A description of those deviations in performance which did occur follows.

The quality of the delay transmission prior to V/P spin-up was excellent, with noise content well below 2%, indicating a minimum of recorder wow and flutter and more than adequate transmitter power and antenna efficiency. At V/P spin-up, both left-hand and right-hand polarization TLM-18 reception indicated a drop in signal strength to zero from the earlier levels of 80 μV and 50 μV, respectively. At Antares II ignition, a very short burst of signal was acquired on the left-hand track only. The left-hand polarization was again acquired 15.7 seconds later, followed by reacquisition of the right-hand polarization 5.4 seconds later. The signal levels were 100 μV and 85 μV, respectively.

At V/P burnout, delay transmission was again lost. Delay transmission was not reacquired until about 5 seconds after exit from the real time telemetry blackout, after which cyclic bursts occurred on both tracks. There was a transition of these bursts into a sinusoidal type fading above and below threshold until about 1798 seconds, following which no additional delay transmissions were received. There is no way to definitely establish the cause of this failure. However, an investigation has indicated that the failure is attributable to a broken co-ax connector or a broken trimicon connection. An attempt is being made to increase the reliability of this system through a mechanical "beef-up" of the antenna system.

At 1851.5 seconds, the failover function occurred, switching the delay recorder data to the real time loop. This resulted in an additional recovery of data. Combination of the receiving station data and the various playbacks results in data recovery of about 65% (21.2 seconds out of a blackout period of 32 seconds).

The roll rate gyro included in the attitude sensor failed to function throughout the flight. This will complicate the accomplishment of Flight Objective 5 (definition of reentry motion), although it will not preclude its attainment. Signal strength data indicate a roll rate of about 3 rps. Regarding this same flight objective, a slight yaw rate was induced at reentry package separation (approximately plus or minus 25 degrees per second). There was no motion indicated on the pitch rate gyro. Later in the reentry the yaw rate built up to larger amplitude, followed by a pitch rate build-up with both gyros eventually oscillating stop-to-stop. An investigation of the characteristics of the pitch gyro indicates that there was a partial failure, possibly caused by a temporary power interruption to the unit. This resulted initially in a no pitch indication. Monitoring diagnostics are being added to aid in detecting failures. In addition, stricter quality control is being imposed on the vendor.

During the reentry phase, the spectral, offset total, and stagnation total radiometers experienced bias shifts. In addition, the internal calibration lamp pulse of the offset total radiometer disappeared. However, these data appear to be recoverable by adjusting the radiometer calibration curve to account for the bias shift. The bias shift is attributed to rfi and will be eliminated through the use of better grounding and shielding, and more extensive use of rf feedthrough filters.

The output waveform of the 18 x 5 PAM commutator was not flat-top, somewhat complicating the automatic data processing; however, these data are all recoverable. Also, this problem existed prior to flight, and the shape of the waveform did not change during flight. The problem is currently under investigation and an attempt will be made to correct the situation.

SECTION 5

PERFORMANCE EVALUATION

With the exception of those deviations noted in the preceding section, the performance of the R/P and its included subsystems was excellent throughout the flight.

In order to assist in accomplishing the primary flight objectives, supporting instrumentation was included in the Reentry Package. This instrumentation consisted of ten monitor points, measurement of eight internal temperatures, measurement of two reference junction temperatures, measurement of internal pressure, and identification of multicoder sections.

All monitor points indicated normal vehicle functioning, with the exception of Monitor Point 5, which monitored the outboard radiometer power supply, and Monitor Point 8, which monitored the tape recorder current. Monitor Point 5 indicated a change in the radiometer power supply; however, this was a false indication, tied in with a failure of the intensity calibration lamp within the radiometer. Apparently the tape recorder current did fluctuate (Monitor Point 8), but this fluctuation did not affect the tape recorder function.

The internal temperature sensors indicated essentially no increase in any of the black box temperatures throughout the entire duration of the flight. The reference junction temperatures also indicated essentially no change. Monitoring of the internal pressure showed that the package remained within specification throughout the entire flight. Monitored functions indicate that the programmed signals for heat shield ejections occurred at their nominal times. Subsequent heat data analysis has verified that the heat shields did eject as planned.

PART 5

ANTARES II A5 PERFORMANCE

GENERAL DYNAMICS/ASTRONAUTICS

INTEGRATED REPORT NO. GDA/BKF64-018

APPROVED BY: ______________________

L. E. MUNSON
ASSISTANT PROGRAM DIRECTOR
FIRE PROGRAM OFFICE

SECTION 1

INTRODUCTION

The Project FIRE Spacecraft was placed into a ballistic trajectory by the Atlas launch vehicle. An Antares II A5 solid propellant rocket motor was used to provide the necessary impulse to increase the velocity of the reentry stage from 20,799 feet per second to the desired reentry velocity of 37,000 feet per second or greater.

The purpose of this part of the report is to present an evaluation of the Antares II A5 motor performance for Project FIRE Flight No. 1.

SECTION 2

SUMMARY

The performance of the Antares II A5 solid rocket motor for the first Project FIRE flight was completely satisfactory. The available flight data indicate that the actual performance closely approximated that which was expected, and that the velocity increment imparted to the Reentry Package provided a reentry velocity which satisfied mission requirements.

SECTION 3

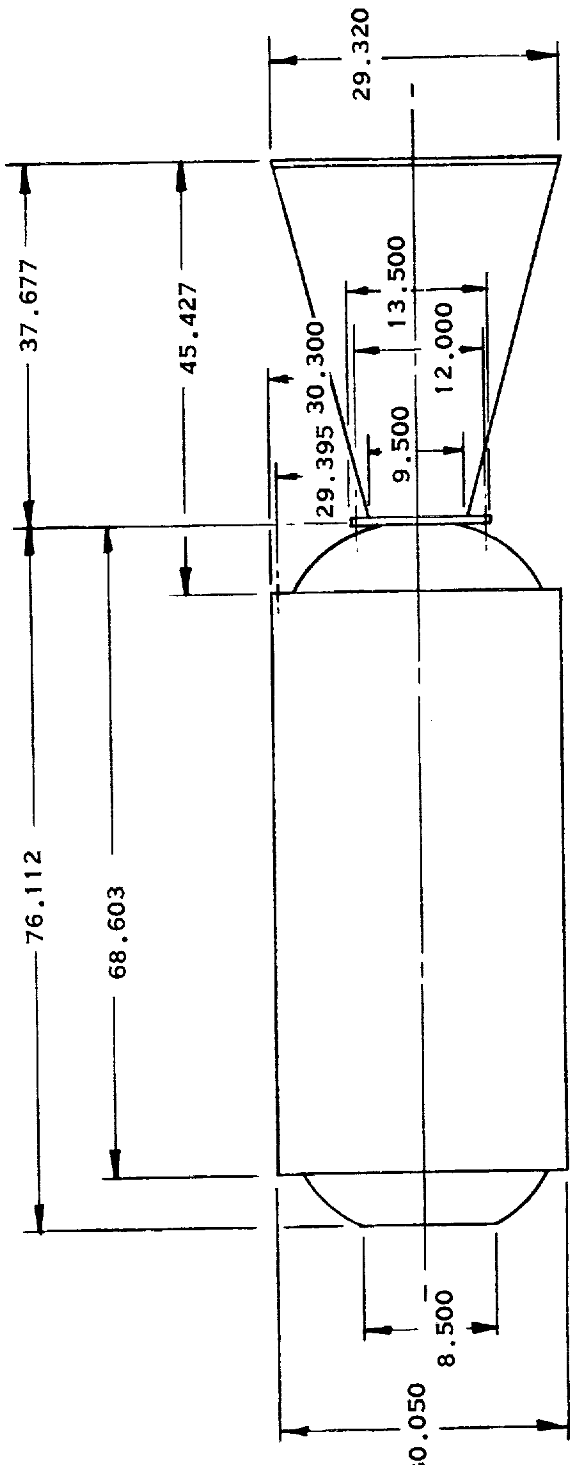
MOTOR DESCRIPTION

The Antares II A5 rocket motor is composed of a composite-modified, double-base solid propellant, bonded to a filament-wound glass fiber and epoxy resin case. Figure 5-3-2 is a sketch of the Antares motor giving its dimensions. The pre-ignition weight (including the FIRE payload) was 3071.03 pounds, and the burnout weight was 464.03 pounds. The following weights were used in the derivation of the Antares II A5 performance:

R/P	183.60 lbs
R/P Adapter	53.40 lbs
Motor Adapter and Balance Weights	27.03 lbs
Antares loaded weight	2807.00 lbs
Pre-Ignition weight	3071.03 lbs
Antares consumed propellant	-2607.00 lbs
Burnout weight	464.03 lbs

ANTARES PERFORMANCE
FIGURE NO. 5-3-2
INTEGRATED REPORT NO. GDA/BKF64-018
MOTOR DESCRIPTION

ANTARES MOTOR DIMENSIONS



SECTION 4

SOURCE OF DATA

FLIGHT DATA

Two primary sources of flight data were utilized to evaluate the Antares II A5 performance. The sources consisted of on-board acceleration measurements and radar tracking information. The on-board acceleration measurements were made by accelerometers mounted in the Reentry Package. A zero to +45g accelerometer was mounted along the longitudinal axis (thrust axis), and three ± 6g accelerometers were mounted along the longitudinal, transverse, and normal axes. Data from these accelerometers were obtained by telemetry through a commutated channel in the Reentry Package telemetry system which provided 5 data points per second. Radar tracking information was obtained from the Ascension Island FPS-16 radar.

PREFLIGHT PREDICTIONS

The predicted performance of the Antares II A5 motor was used to generate a preflight nominal trajectory. Subsequent to the publication of that trajectory, up-dated Antares II A5 weight and performance data were obtained, and the information was utilized to establish an expected trajectory. The expected trajectory results will be used for all comparisons with actual flight data.

SECTION 5

METHOD OF PERFORMANCE EVALUATION

INITIAL CONDITIONS

The Antares ignition point used in this evaluation is defined by the following parameters:

Altitude	982,377	Feet
Flight path angle	-15.32	Degrees
Velocity (earth relative)	20,799	Feet/Second
Pitch angle of attack	-5.932	Degrees

The values of the parameters listed above were based upon data calculated from launch vehicle cutoff conditions, prior to receipt of the FPS-16 radar data. Although these values do not agree exactly with the radar data, the errors resulting from this disagreement can be assumed to be small and will have a negligible effect on the performance evaluation.

COMPARISON OF ACCELEROMETER AND RADAR DATA

The accelerometer data were reduced to obtain a variation of velocity with time in order to facilitate a comparison with the radar data which were already in this form. Since the body angular rates were found to be small during the Antares thrusting, the $\pm$ 6g accelerometer data were not used in this evaluation. Reduction of the accelerometer data was accomplished by the following procedure:

A basic assumption was made that the propellant weight flow rates and thrust-time histories which occurred during flight were similar in character to the preflight predictions of these parameters. A multiplying factor was applied to the preflight thrust-time history, such that the resulting acceleration matched the acceleration value from the initial impulse obtained during flight. Using the multiplying factor obtained by this method, a complete acceleration versus time curve was generated by utilizing a computer program.

The resulting acceleration-time history was compared to the acceleration-time history obtained by telemetry during flight. The differences in the two acceleration-time histories (excluding the initial points, which were purposely made to coincide) could then be assumed to be caused by differences between the preflight and actual weight-flow time histories. Accordingly, a multiplying factor was applied to the preflight weight-flow tables and a second acceleration-time history was generated. This process was repeated until the computed acceleration-time history matched the acceleration-time history obtained in flight, and the total weight loss matched the predicted consummed weight.

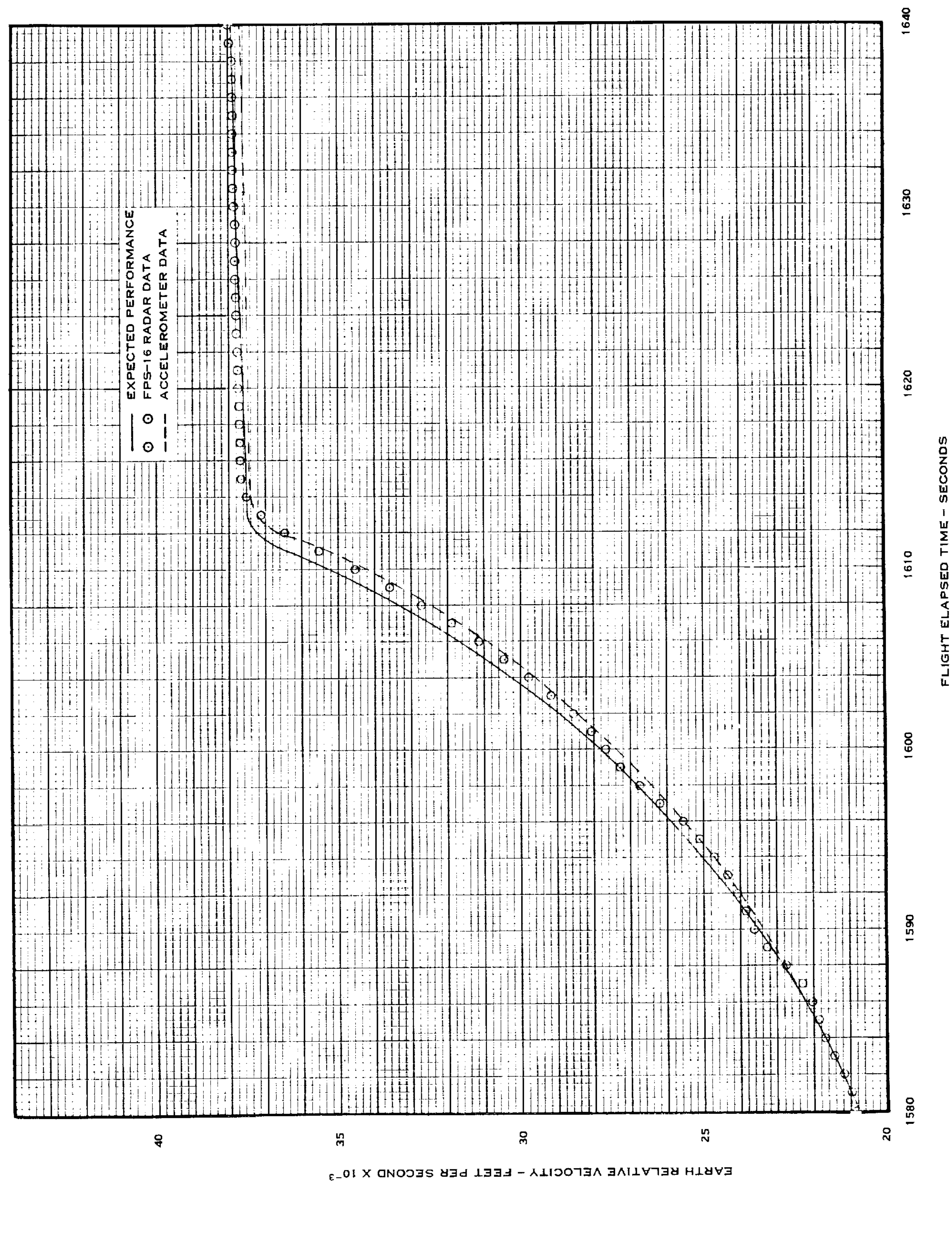
Figure 5-5-3 compares the resulting velocity-time history, obtained from the procedure stated above, to that obtained from the reduced FPS-16 data and the values obtained from the expected trajectory.

As can be seen in Figure 5-5-3 the accelerometer-derived velocities are considerably lower than the velocities obtained from the radar data. Also, the radar data compares closely with the expected velocity-time history. On the basis of this comparison, it was concluded that the velocity-time history obtained from the radar data would most accurately reflect the actual conditions during the flight.

METHOD OF DERIVATION OF ANTARES PERFORMANCE PARAMETERS

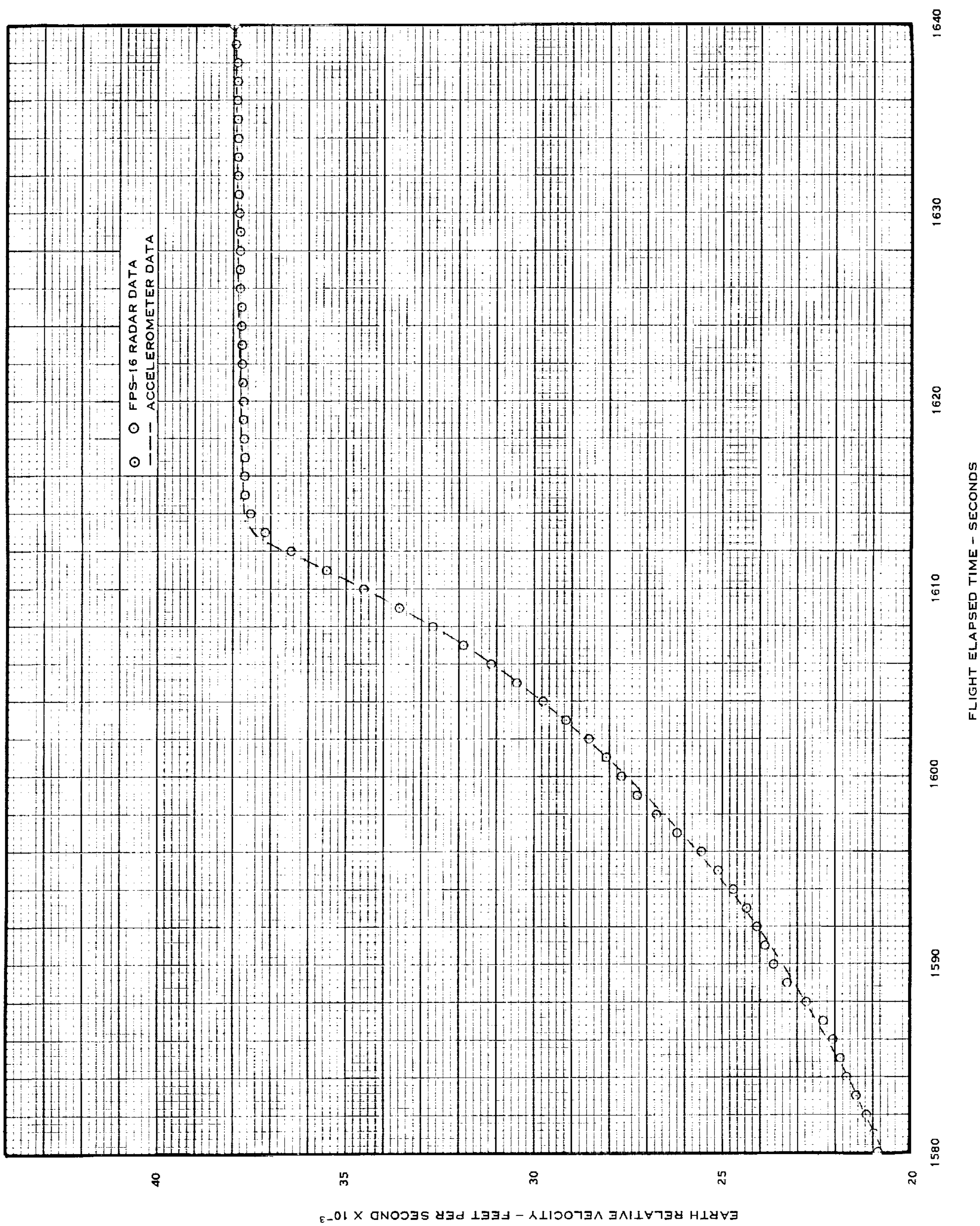
The motor performance, as reflected by the radar data, was obtained by generating a trajectory which produced a velocity-time history matching the one given by the radar data. A comparison of the velocity-time history obtained from this trajectory and that obtained from the radar data is shown in Figure 5-5-4. The match was achieved by using the weight flow rate obtained in the accelerometer match, and applying a multiplication factor to the thrust-time history. As can be noted in Figure 5-5-5, the acceleration values are greater than those obtained from on-board measurements, but the general shape of the profile is preserved. The thrust-time history, specific impulse, and total impulse of the Antares II A5 motor were derived from the final acceleration, weight-flow and weight-time histories resulting from the above evaluation.

ANTARES EARTH-RELATIVE VELOCITY DURING BURN

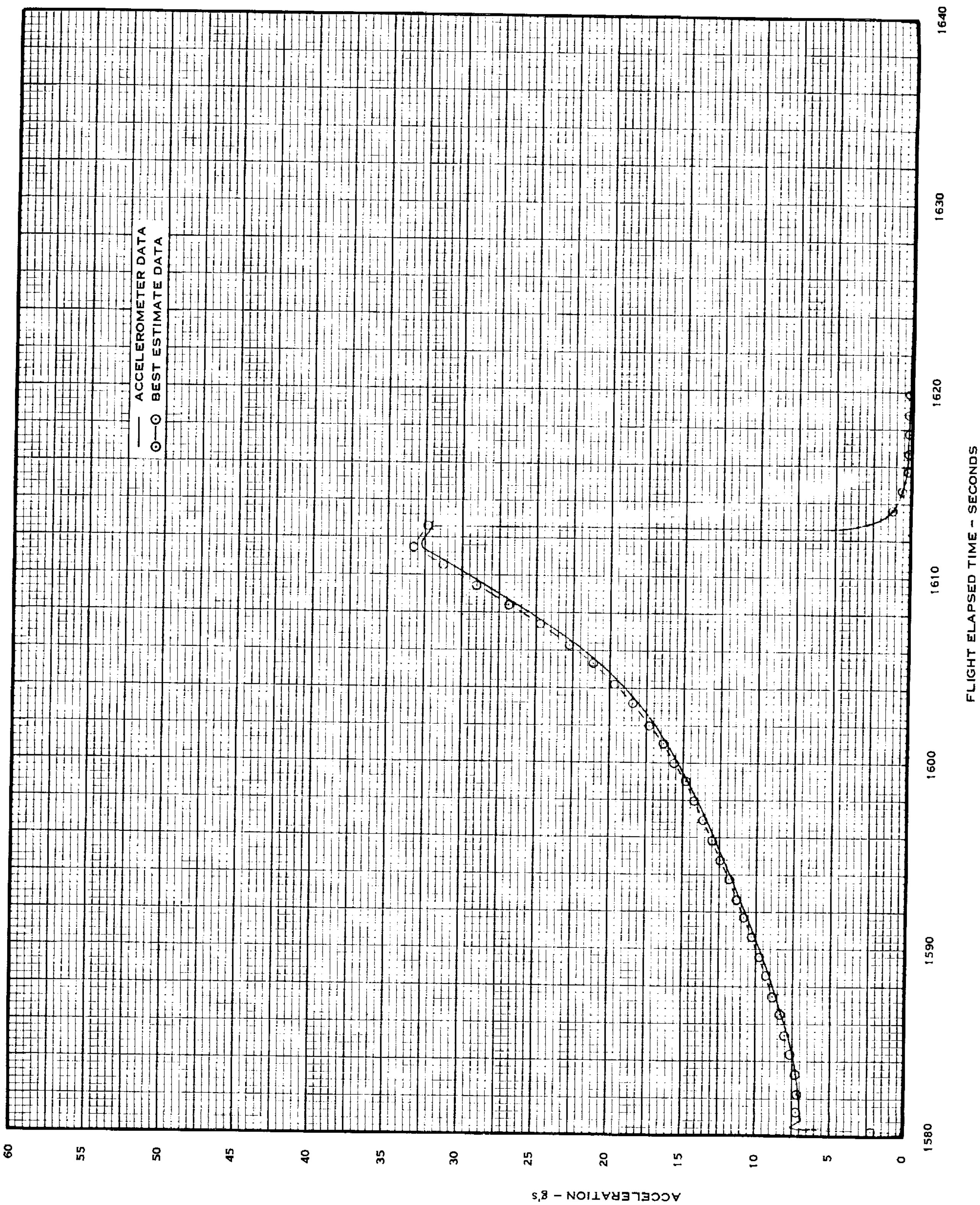


ANTARES PERFORMANCE
FIGURE NO. 5-5-4
INTEGRATED REPORT NO. GDA/BKF64-018
METHOD OF PERFORMANCE EVALUATION

COMPARISON OF BEST ESTIMATE AND RADAR VELOCITIES FOR ANTARES BURN



ANTARES AXIAL ACCELERATION DURING BURN



SECTION 6

ANTARES II A5 PERFORMANCE

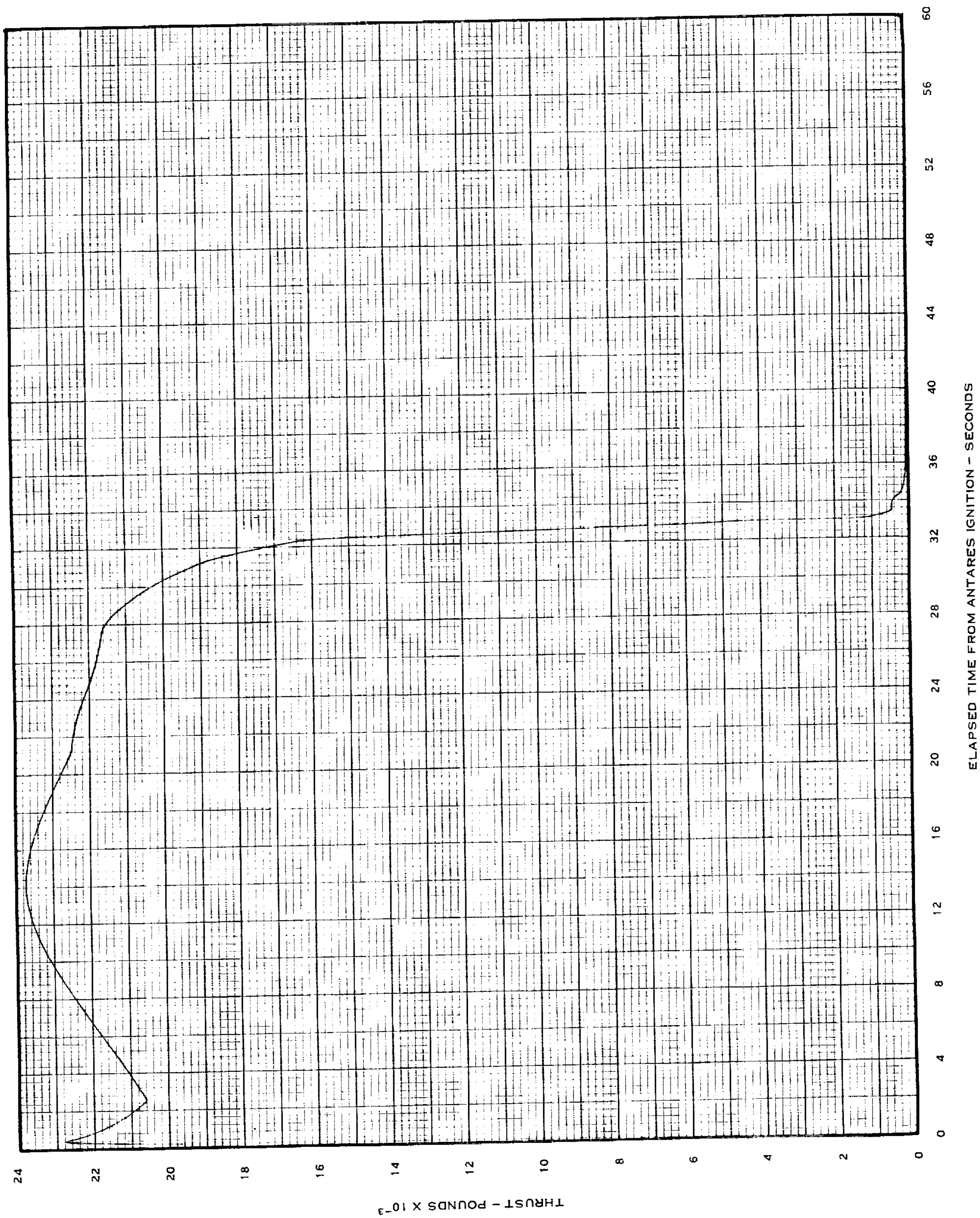
The results of the Antares II A5 performance evaluation are shown in Figures 5-6-2 and 5-6-3, and in Tables 5-6-1 and 5-6-2. Figures 5-6-2 and 5-6-3 present the variation of thrust and weight flow rate versus elapsed time from motor ignition respectively.

Table 5-6-1 presents time histories of thrust, flow rate and specific impulse in tabular form. The time increments have been chosen such that performance of the motor is adequately represented. It should be noted that the thrust tail-off shown in Figure 5-6-2 and Table 5-6-1 does not necessarily represent the actual tail-off during flight. This discrepancy occurs because the on-board accelerometer was not sufficiently accurate to define the tail-off in this region.

Table 5-6-2 presents consumed weight versus time and cumulative impulse versus time from motor ignition. Total impulse of the motor was 719,931.6 pound-seconds as compared to an expected value of 726,922.6 pound-seconds. The consumed weight average specific impulse was determined to be 276.17 pounds of thrust per pound of mass per second.

ANTARES PERFORMANCE
FIGURE NO. 5-6-2
INTEGRATED REPORT NO. GDA/BKF64-018
ANTARES II A5 PERFORMANCE

BEST ESTIMATE OF ANTARES THRUST



BEST ESTIMATE OF ANTARES WEIGHT FLOW RATE

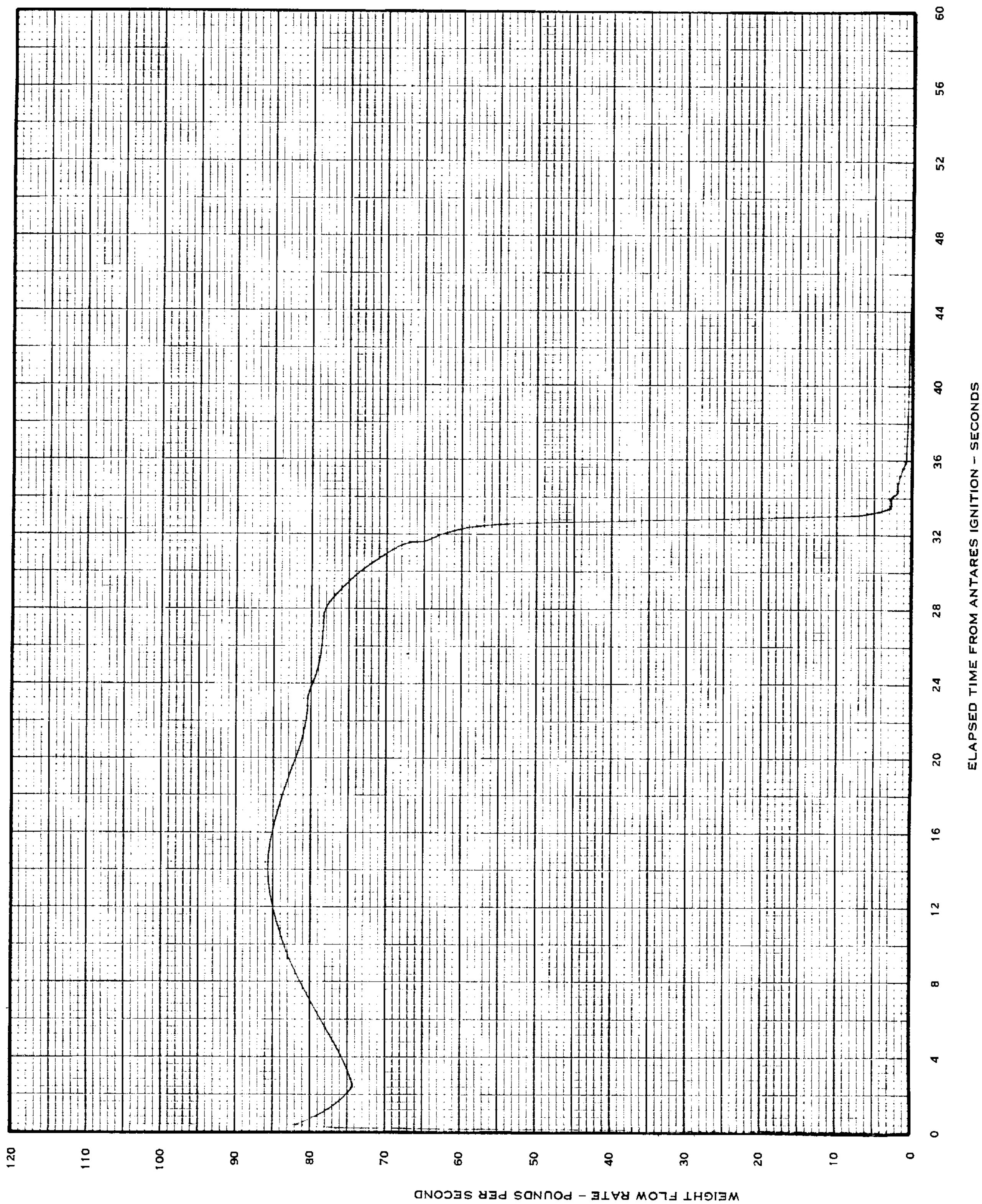


TABLE 5-6-1. BEST ESTIMATE OF ANTARES PERFORMANCE

Antares Burn Time (sec)	Thrust (lbs)	Weight Flow Rate (lbs/sec)	Instantaneous Specific Impulse (sec)
Ignition	0	0	-
0.2	21,226	76.54	277.4
0.4	22,771	82.10	277.2
1.0	21,786	78,55	277.3
2.5	20,599	74.28	277.4
4.6	21,226	76.54	277.4
7.4	22,288	80.36	277.5
11.0	23,349	84.19	277.3
14.4	23,754	85.66	277.3
17.0	23,426	84.47	277.4
21.2	22,481	81.06	277.5
23.3	22,268	80.29	277.5
25.7	21,832	78.72	277.4
26.7	21,769	78.50	277.3
27.7	21,630	78.28	276.6
29.7	20,506	74.21	276.4
30.8	19,482	70.64	276.0
31.2	19,065	69.14	275.5
31.3	18,814	68.23	276.0
31.6	18,254	66.21	275.8
21.8	17,541	63.62	275.8
32.1	17,008	61.69	275.8
32.2	16,614	60.27	276.3
32.4	16,167	58.65	276.1
32.5	15,081	55.00	274.3
32.9	4,855	27.0	179.8
33.1	1,205	7.0	172.0
33.3	792	4.7	168.5
33.5	456	2.8	162.8
34.0	455	2.8	162.4
34.4	263	1.9	138.3
35.2	119	1.5	79.3
35.9	57	0.8	71.3
39.8	33	0.5	71.8
45.8	0	0	-
60.0	0	0	-

Average I_{sp} = 276.2 sec.

TABLE 5-6-2. BEST ESTIMATE OF ANTARES WEIGHT HISTORY AND CUMULATIVE IMPULSE

Antares Burn Time (sec)	Consumed Weight (lbs)	Cumulative Impulse (lb-sec)
Ignition	0.0	0
2.09	157.80	43,086
4.18	314.66	86,585
6.27	476.37	131,403
8.36	643.84	177,827
10.44	816.14	225,591
12.53	992.30	274,435
14.62	1170.41	323,822
16.71	1348.04	373,089
18.80	1523.08	421,648
20.88	1609.28	445,562
22.97	1863.29	516,025
25.06	2029.92	562,251
27.15	2194.19	607,809
29.24	2355.31	652,371
31.32	2506.00	694,032
33.41	2598.39	719,036
35.50	2602.66	719,629
37.59	2604.26	719,745
40.72	2605.87	719,861
44.89	2606.82	719,930
50.11	2606.85	719,932
55.33	2606.85	719,932
60.00	2606.85	719,932

SECTION 7

CONCLUDING REMARKS

The evaluation of the Antares II A5 performance for the first Project FIRE flight indicates close agreement between the predicted and experienced values of total impulse for the motor. The accuracy of the predicted impulse is sufficient to allow the generation of preflight performance to a high degree of accuracy.

PART 6

VELOCITY PACKAGE PERFORMANCE

GENERAL DYNAMICS/ASTRONAUTICS
INTEGRATED REPORT NO. GDA/BKF64-018
LTV/A REPORT NO. 3-30000/4R-75

APPROVED BY: W.C. McMillin
W. C. McMILLIN
PROJECT MANAGER - PROJECT FIRE

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SECTION 1

INTRODUCTION

The first Project FIRE vehicle was successfully launched by the National Aeronautics and Space Administration from Cape Kennedy, Florida, at 1642 EST on 14 April 1964. Project FIRE Flight No. 1 was the first of a series of two launches planned by NASA/Langley Research Center for the purpose of obtaining data on total and radiative heating, radio signal attenuation, and material behavior during atmospheric reentry to provide basic knowledge of design criteria for reentry vehicles operating at lunar return velocities.

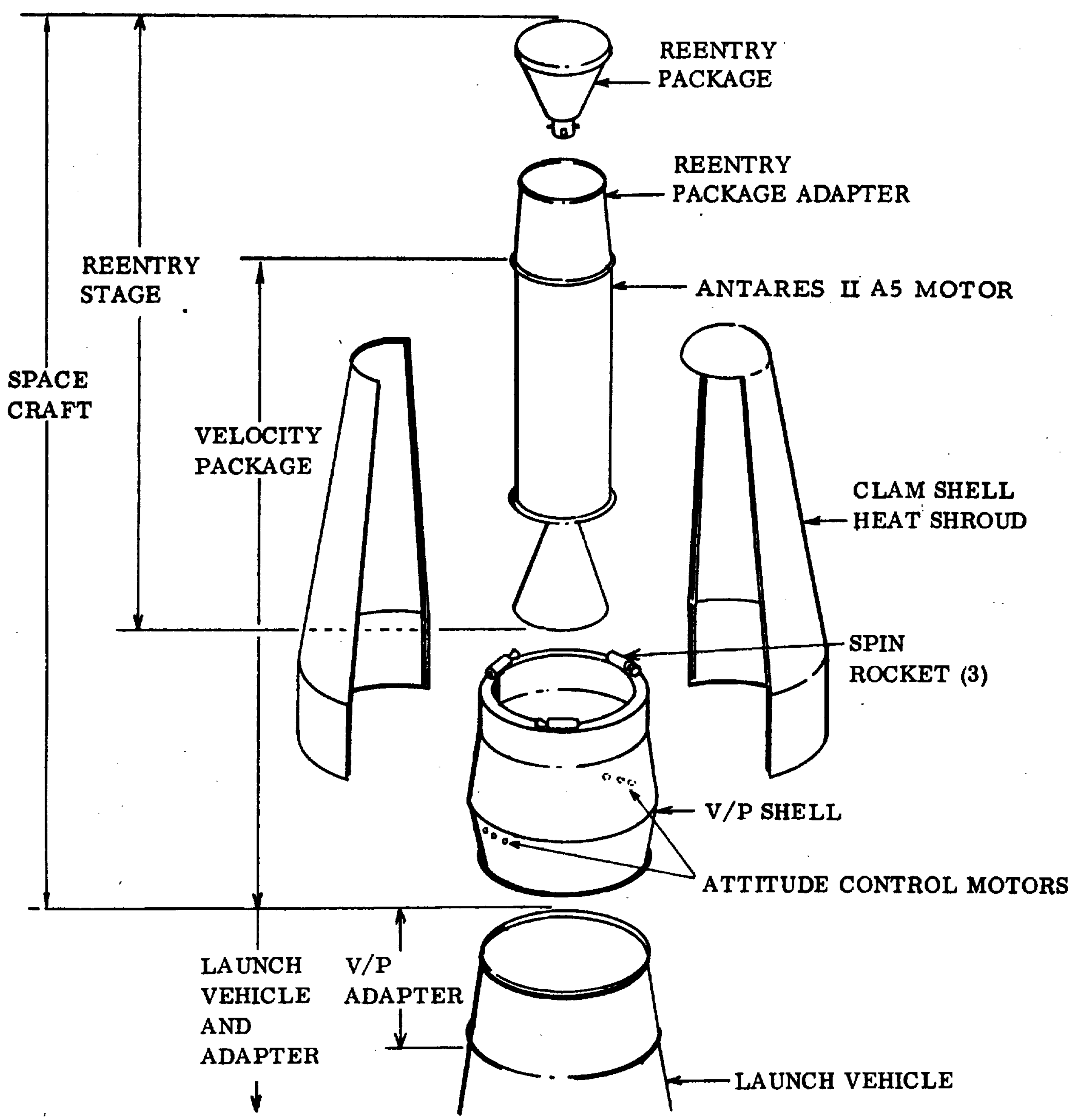
The Atlas D Launch Vehicle (L/V) placed the Spacecraft (consisting of a Velocity Package (V/P) manufactured by Ling-Temco-Vought/Astronautics (LTV/A) and a Reentry Package (R/P) manufactured by Republic Aviation Corporation (RAC)) into a ballistic trajectory along the Atlantic Missile Range; the Velocity Package then oriented the Spacecraft to the proper attitude and, at a pre-determined time, ignited the solid propellant rocket motor driving the Reentry Package back into the atmosphere at the desired velocity approximately 5,000 miles downrange near Ascension Island. All LTV/A flight objectives were satisfactorily accomplished.

The basic structure of the Velocity Package consists of two circular shells, one within the other. A metalite shelf located between the outer and inner shell sections provides support for the major part of the V/P equipment. A Velocity Package Adapter provides the structural and electrical interface between the Velocity Package and the Launch Vehicle and the Reentry Package Adapter provides the structural and electrical interface between the Velocity Package and the Reentry Package. Propulsion for the Velocity Package is provided by an ANTARES II A5 (ABL X-259) solid propellant rocket motor, manufactured by the Allegany Ballistics Laboratory. A heat shroud, manufactured for LTV/A by the Douglas Aircraft Corporation, protects the Spacecraft from aerodynamic heating during the boost ascent. Major components of the Spacecraft are shown in Figure 6-1-3 and a cutaway view is shown in Figure 6-1-4. The Velocity Package shell assembly and the Velocity Package with the heat shroud installed are shown in Figures 6-1-5 and 6-1-6, respectively. The Velocity Package also includes a guidance system for maintaining stability and control, a telemetry system for transmitting flight data, and an ignition/destruct system.

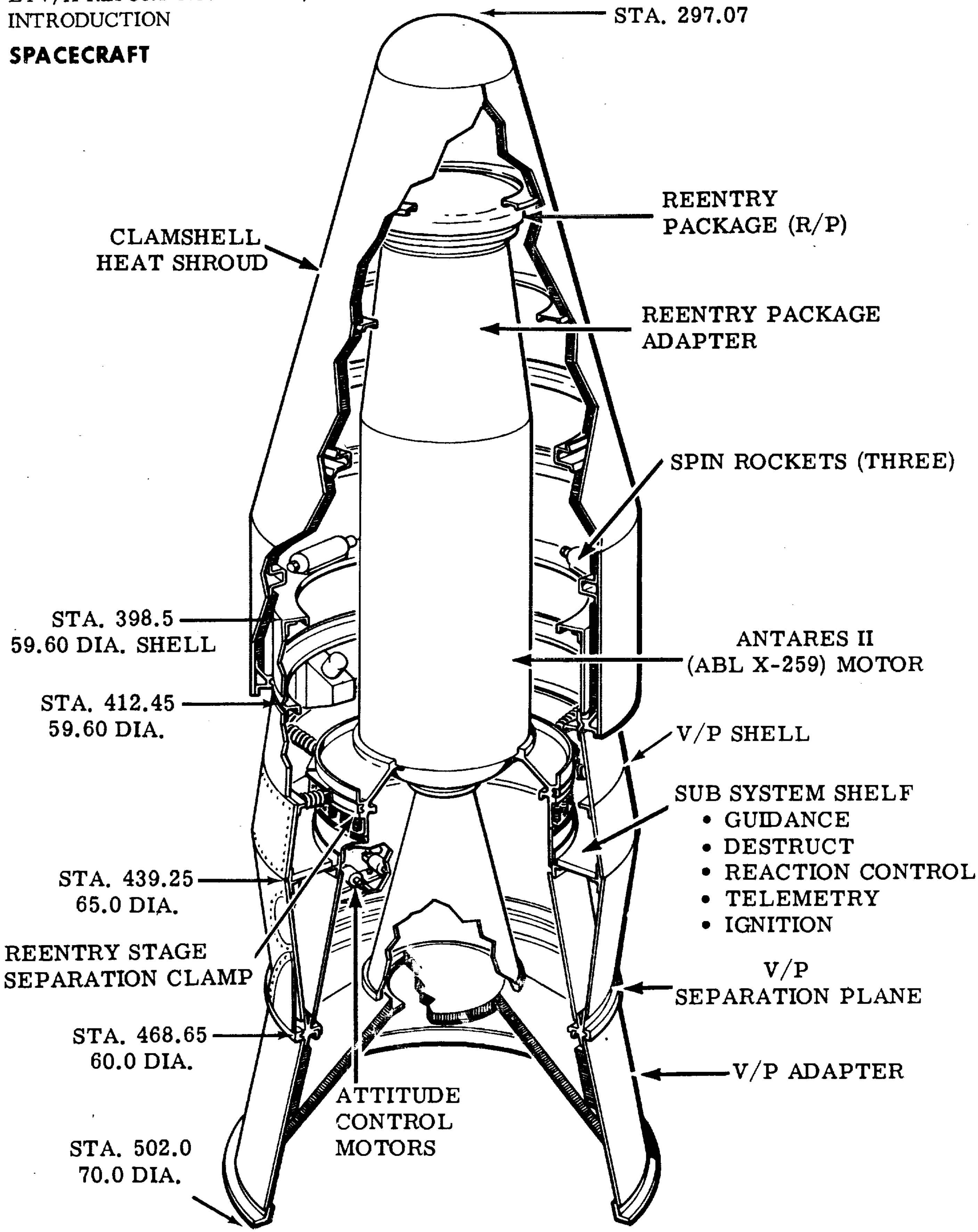
INTRODUCTION

The purpose of Part 6 of this integrated report is to present a summary of the reduced data and results achieved from Project FIRE Flight No. 1 as related to the Velocity Package only. The flight trajectory evaluation, the vibrometer analysis, and the ANTARES II A5 motor performance evaluation will be performed by others and reported elsewhere.

MAJOR COMPONENTS



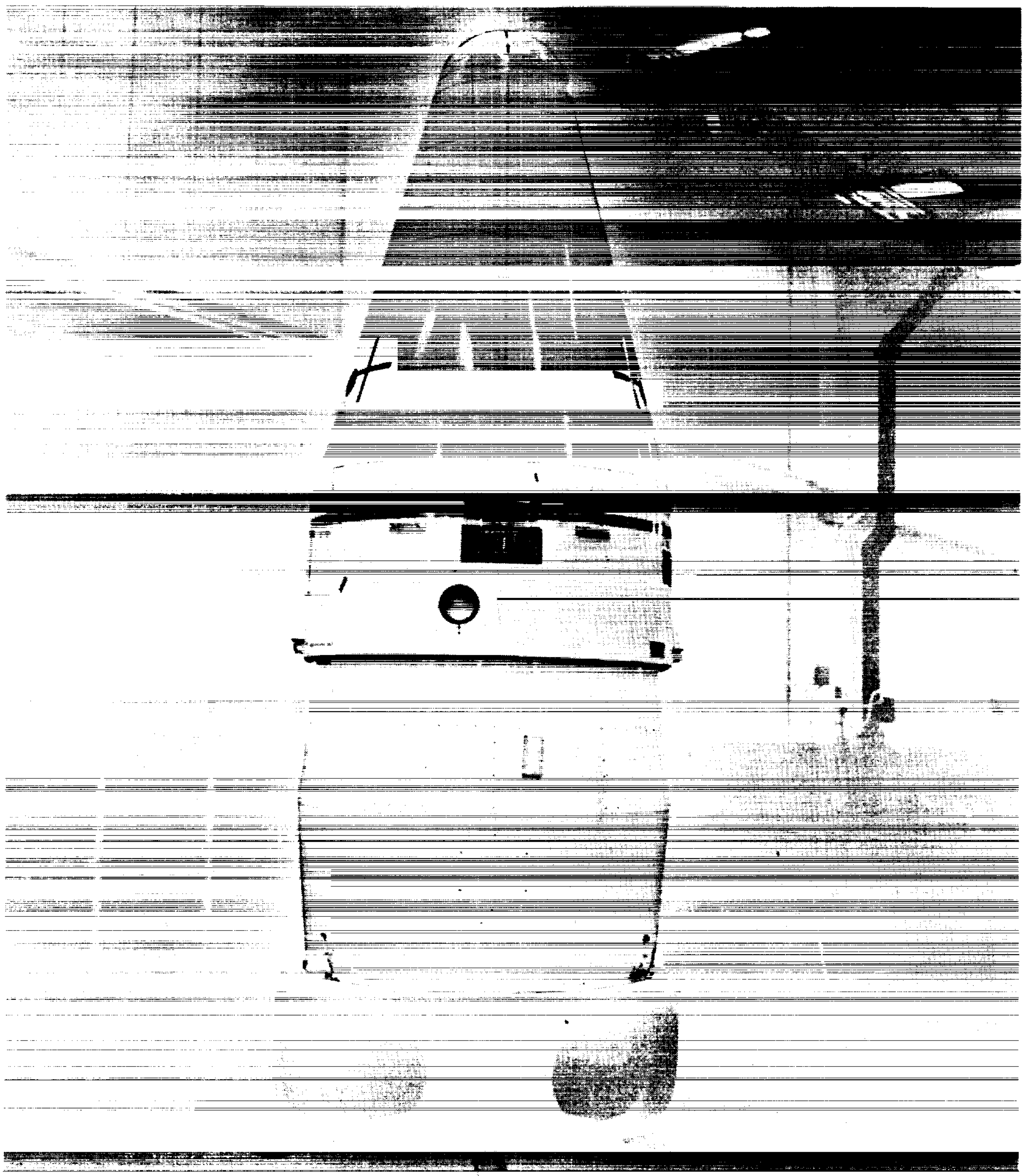
SPACECRAFT


STA. 297.07
REENTRY PACKAGE (R/P)
CLAMSHELL HEAT SHROUD
REENTRY PACKAGE ADAPTER
SPIN ROCKETS (THREE)
STA. 398.5
59.60 DIA. SHELL
ANTARES II (ABL X-259) MOTOR
STA. 412.45
59.60 DIA.
V/P SHELL
SUB SYSTEM SHELF
• GUIDANCE
• DESTRUCT
• REACTION CONTROL
• TELEMETRY
• IGNITION
STA. 439.25
65.0 DIA.
REENTRY STAGE SEPARATION CLAMP
V/P SEPARATION PLANE
STA. 468.65
60.0 DIA.
V/P ADAPTER
ATTITUDE CONTROL MOTORS
STA. 502.0
70.0 DIA.

VELOCITY PACKAGE SHELL ASSEMBLY

VELOCITY PACKAGE PERFORMANCE
FIGURE NO. 6-1-6
INTEGRATED REPORT NO. GDA/BKF64-018
LTV/A REPORT NO. 3-30000/4R-75
INTRODUCTION

VELOCITY PACKAGE



SECTION 2

SUMMARY

The Project FIRE Flight No. 1 vehicle (AMR Test 0225) was successfully launched from AMR Complex 12, Cape Kennedy, Florida, at 1642: 25.536 EST on 14 April 1964. The Velocity Package mission objective was to place the Reentry Package at a minimum velocity of 37,000 feet per second and a flight path angle of -15 degrees at an altitude of 400,000 feet. This objective was achieved satisfactorily as shown by the data in Part 2 of this integrated report.

Specific flight objectives assigned to the V/P Contractor, LTV/A, in support of the space vehicle system performance are itemized in the following table. All LTV/A flight objectives were satisfactorily accomplished.

FLIGHT OBJECTIVE	PRIORITY	REMARKS
Activate V/P ignition interlock	Primary	L/V telemetry records verified that the V/P interlock was activated at the proper time by the L/V discrete signal. The L/V backup signal occurred, but was not required.
V/P timer start	Primary	The V/P timer was started at the proper time by the L/V timer start discrete signal.
V/P gyros uncage	Primary	V/P gyros were uncaged at the proper time by the L/V discrete signal. This discrete signal was also a backup signal for the V/P timer start.

FLIGHT OBJECTIVE	PRIORITY	REMARKS
V/P guidance and control system functions	Primary	The V/P guidance and control system stabilized the spacecraft and oriented it to the correct attitude for reentry.
Control system unregulated pressure	Primary	The unregulated nitrogen pressure remained at a constant pressure (2985 psia) during boost. During the coast phase, the nitrogen pressure decreased approximately 100 psi.
Control system regulated pressure	Primary	A nitrogen regulated pressure of 335 psi was maintained during the flight.
Spin motors function	Primary	The spin motors provided the desired spin rate.
Heat shroud separation	Primary	A clean heat shroud separation occurred at the proper time. The L/V provided a backup signal, which was not required on this flight.
V/P separation from the L/V	Primary	A clean L/V-V/P separation occurred at receipt of signal from the launch vehicle.
Spin motors ignition	Primary	All three spin motor temperatures increased approximately 4°F three seconds after spin up. This verified that all three spin motors ignited.
Reentry Stage separation	Primary	The R/S separated from the V/P at the proper time. Within the limitations of the R/P instrumentation, no coning could be detected.
ANTARES rocket motor ignition	Primary	The R/P telemetry data verified that the ANTARES motor ignited at the proper time.

FLIGHT OBJECTIVE	PRIORITY	REMARKS
Thermal protection	Secondary	All temperature-instrumented V/P components operated well within their respective temperature limits.
Structural integrity of the V/P	Tertiary	The V/P did not have any specific instrumentation to verify structural integrity. The success of the flight indicates that the V/P structure provided the necessary rigidity for all V/P systems and that no structural failures occurred.

The V/P sequence of events is presented in the following table. Several events are referenced to nominal V/P timer start time. The V/P timer start time was assumed to have occurred 0.10 seconds prior to receipt of the timer start indication on V/P telemetry records since the time of the V/P timer start discrete signal was on an L/V commutated channel and could only be determined to ± 0.1 seconds. The shroud jettison, gyros uncage, and the L/V-V/P separation events occurred two seconds later than nominal time, being functions of the time that SECO occurred.

V/P SEQUENCE OF EVENTS

EVENT	SIGNAL SOURCE	NOMINAL TIME, SEC.	ACTUAL TIME, SEC.
Lift-off (2-inch motion) (2142:25.536 GMT)	L/V	T=0	T=0
V/P Timer Start Discrete	L/V	0*	+0.57*
V/P Timer Start Indication	V/P	+0.10*	+0.67*
V/P Shroud Jettison	L/V	+0.69*	+3.14*
Uncage V/P Gyros and V/P Timer Start Backup	L/V	+7.99*	+11.19
L/V-V/P Separation	L/V	T+308.30	T+311.53
Start Pitch Program	V/P	T+319.31	T+319.88
End Pitch Program	V/P	T+420.84	T+421.39
R/P Separation Timer Start Signal	V/P	T+1567.2	T+1567.57
Spin Motor Ignition	V/P	T+1573.95	T+1574.31
V/P-R/S Separation	V/P	T+1576.95	T+1577.32
ANTARES Ignition	V/P	T+1580.20	T+1580.31

* NOTE: Elapsed Time from V/P Timer Nominal Start Time

Special in-flight instrumentation was not installed to monitor the V/P batteries, however, the successful systems operation indicated that the battery performance was satisfactory. The 400-cycle inverter performance was satisfactory as evidenced by telemetry data and all guidance functions.

The V/P telemetry system in-flight performance was excellent. High quality data were recovered from all telemetry functions for the entire flight period that the V/P telemetry was programmed to operate. The telemetry system ceased transmitting as programmed at T+1577 seconds when the Reentry Stage (R/S) separated from the V/P.

Accuracy of the attitude reference and the programmer was not independently determinable, however, the reentry angle error of approximately 0.5 degrees indicated low drift and low initial misalignments of the reference, as well as accurate programming. The terminal error includes attitude errors at launch vehicle VECO, program errors, drift errors during coast, and separation errors.

The attitude reference, programmer, timer and inverter performed as expected; off-design operation of any one of these components would have resulted in significant time and/or angle errors. The reaction control system operation was satisfactory and within design limits. The motor valves operated normally upon command and the motor thrust was close to the predicted value. Nitrogen consumption was considerably lower than predicted.

Satisfactory spin motor performance resulted in an initial spin rate of 161 rpm compared to a predicted rate of 169$\pm$12 rpm. No special instrumentation was provided for the other pyrotechnic devices; however, satisfactory performance of their respective systems indicates that the devices functioned properly. The ignition system operated satisfactorily and the V/P received and responded to guidance primary signals rather than the backup signals. The destruct system performed satisfactorily during pre-launch checkout. The system was not required during the flight.

All temperature-instrumented V/P components operated well within their respective temperature limits.

The heat shroud separation was very clean. Minor disturbances were noted on all three vibrometer traces and on the V/P pitch and yaw traces at the time that the separation bolts fired. However, these disturbances were expected, and damped out within approximately 0.1 seconds. The V/P separated from the L/V cleanly, and with a very small tipoff effect. The maximum angular rates imparted to the V/P were approximately 0.6 deg/sec left in yaw, 1 deg/sec right in roll, and 0.5 deg/sec up in pitch, well within the predicted limits. The V/P separated from the Reentry Stage at the proper time. Within the limitations of the R/P instrumentation no coning could be detected.

Due to the high degree of success obtained on Flight No. 1 and the absence of any discrepancies or malfunctions evidenced on the flight records, it is recommended that no design or hardware changes be made to the Velocity Package.

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SECTION 3

TELEMETRY SYSTEM ANALYSIS

The V/P telemetry system in-flight performance was excellent. High quality data were recovered for all telemetry functions for the entire flight period that the V/P telemetry was programmed to operate. Ground Station 1 (TEL-2), at Kennedy Space Center (KSC), received useable data through T+765 seconds. Overlapping coverage was obtained between Station 1 and Station 9.1 (located at Antigua) and between Station 9.1 and Station 12 (located at Ascension Island). The telemetry system ceased transmitting as programmed at T+1577 seconds when the V/P separated from the Reentry Stage.

The telemetry ground stations at Cape Kennedy, Antigua, and Ascension Island obtained telemetry tape recordings which provided 100 percent V/P data coverage. Preliminary estimates of data coverage from AMR indicated that two AMR instrumentation aircraft and one AMR instrumentation ship ("Yankee") tracked the V/P telemetry (244.3 mc) after T+1577 seconds. This estimate was apparently in error since the oscillograph records made from the Ascension and Yankee tapes showed that the V/P transmitter ceased operation at T+1577 seconds as planned. The telemetry tape supplied to LTV/A from the Range ship ("Yankee") had only intermittent V/P telemetry signals from T+1500 to T+1577 seconds. Approximately 40 percent of the V/P data during this period was considered recoverable. During the period after T+1577 seconds, there was no evidence of a V/P telemetry signal. The tape supplied to LTV/A from a Range aircraft ("Silver 1") did not have any V/P useable telemetry data.

The Velocity Package telemetry parameters are listed on the following page.

VELOCITY PACKAGE

TELEMETRY INSTRUMENTATION SUMMARY

IRIG CHANNEL	SCO FREQUENCY KC	COMMUTATOR SEGMENT	MEASUREMENT CODE	MEASUREMENT	NOMINAL MEASUREMENT RANGE
3	0.73	N/A	GS-6	Yaw Displacement	± 10 deg.
4	0.96	N/A	GS-5	Pitch Displacement	± 10 deg.
5	1.30	N/A	GS-4	Roll Displacement	± 10 deg.
6	1.70	N/A	M-4	Event Matrix	On-Off
			(E-3)	Timer Start	
			(E-4)	Gyro Uncage	
			(E-5)	ANTARES Motor Ignition	
7	2.30	N/A	GS-8	Pitch Program Voltage	0-3.2 VDC
8	3.00	N/A	M-1	Upper Roll Matrix	On-Off
			(V-1)	Upper Left Valve	
			(V-2)	Upper Right Valve	
			(PS-1)	Upper Left Pressure	
			(PS-2)	Upper Right Pressure	
9	3.90	N/A	M-2	Lower Roll Matrix	On-Off
			(V-3)	Lower Right Valve	

VELOCITY PACKAGE TELEMETRY INSTRUMENTATION SUMMARY

IRIG CHANNEL	SCO FREQUENCY KC	COMMUTATOR SEGMENT	MEASUREMENT CODE	MEASUREMENT	NOMINAL MEASUREMENT RANGE
			(V-4)	Lower Left Valve	
			(PS-3)	Lower Right Pressure	
			(PS-4)	Lower Left Pressure	
10	5.40	N/A	M-3	Pitch Matrix	On-Off
			(V-5)	Pitch Up Valve	
			(V-6)	Pitch Down Valve	
			(PS-5)	Pitch Up Pressure	
			(PS-6)	Pitch Down Pressure	
11	7.35	N/A	GS-3	Yaw Rate	$\pm$ 10 deg/sec
12	10.50	N/A	GS-2	Pitch Rate	$\pm$ 10 deg/sec
13	14.50	N/A	GS-1	Roll Rate	$\pm$ 30 deg/sec
14	22.00	N/A	GS-7	400 cps Reference	-
15	30.00	5, 20	P-1	N_2 Tank Pressure	0-3500 psia
15	30.00	6, 21	P-2	N_2 Regulated Pressure	0-400 psia
15	30.00	8, 23	M-5	Event Matrix	On-Off

VELOCITY PACKAGE TELEMETRY INSTRUMENTATION SUMMARY

IRIG CHANNEL	SCO FREQUENCY KC	COMMUTATOR SEGMENT	MEASUREMENT CODE	MEASUREMENT	NOMINAL MEASUREMENT RANGE
			(E-1)	Heat Shroud Ejection	
			(E-2)	L/V-V/P Separation	
			(E-6)	V/P-R/S Separation	
15	30.00	9	T-1	ANTARES Motor Temperature	0-350°F
15	30.00	10	T-2	ANTARES Motor Temperature	0-350°F
15	30.00	11	T-3	ANTARES Motor Temperature	0-350°F
15	30.00	12	T-4	Spin Motor Temperature	0-350°F
15	30.00	13	T-5	Spin Motor Temperature	0-350°F
15	30.00	14	T-6	Spin Motor Temperature	0-350°F
15	30.00	15	T-7	Rate Gyro Temperature	0-350°F
15	30.00	16	T-8	PVE Temperature	0-350°F
15	30.00	17	T-9	T/M Transmitter Temperature	0-350°F
15	30.00	24	T-10	MIG Block Temperature	0-350°F
16	40.00	N/A	A-4	Vibration System B	± 25 "g"
17	52.80	N/A	A-5	Vibration System C	± 15 "g"
18	70.00	N/A	A-6	Vibration System A	± 30 "g"

SECTION 4

GUIDANCE SYSTEM ANALYSIS

General

During the active period of guidance system operation, overall performance was generally better in terms of system accuracy and fuel consumption than had been anticipated. The 400-cycle inverter performance was satisfactory as evidenced by telemetry data and all guidance timer functions were accomplished within 0.02% of their respective predicted times. Two factors contributed to reduction in fuel expenditure. First, the reaction control motors had shorter turn-on and turn-off times than predicted, and second, control about the roll-yaw axes occurred in an unanticipated manner. Allowances and expenditures are summarized in the following table:

MISSION PHASE	ALLOWANCE IMPULSE, POUND-SEC.	FLIGHT (CALCULATED) IMPULSE, POUND-SEC.
Capture	125.3	7.5
Pitch Program	7.0	7.1
Coast	133.2	55.1
Contingency	334.6	–

The actual capture maneuver was mild, with the maximum attitude error approximately 1/4 degree and the induced rate at capture not exceeding one degree per second. Since the capture allowance was based on "worse case" conditions, the comparison shown is not of direct significance. The pitch program allowances and the actuals are comparable. Coast requirements were significantly less than predicted.

The characteristics of the pitch control during a typical 75-second period are detailed in Figure 6-4-6. The limit cycle period during this interval is approximately 20 seconds, which conforms generally to the predicted period. However, the duty cycle is lower than expected. This result can occur in the presence of unsymmetrical limit cycles such as those indicated in Figure 6-4-6. The cause of the specific form of the asymmetry indicated in the pitch axis is not known. Significant differences in turn-off delays between pitch-up and pitch-down jets, with resulting differences in minimum pulse widths, would provide the general type of response indicated in the traces. An idealization of this effect is illustrated in Figure 6-4-7. The resulting time history corresponds qualitatively with the actual trace. The comparable behavior and impulse requirements per unit time for the symmetrical limit cycle with the same total average impulse per cycle are indicated in Figure 6-4-8.

The second anomaly which contributed to lower-than-predicted fuel consumption may be partially explained by interaction between the payload cooling system and the Velocity Package control system. The payload contains a "water boiler" cooling system which exhausts through a constant diameter tube exiting radially with respect to the vehicle longitudinal axis (Z-axis), 15° aft and at an angle of 23° with respect to pitch axis. Thus, the exhaust from the cooler will not produce a roll moment but will provide a small ($\sin 23^{\circ}$) component of the resulting moment about the pitch axis and a larger component (cox 23°) about the yaw axis, and a small translational velocity increment. The general characteristics of the pitch limit cycle indicate that the probably primary source of the reduced fuel consumption was asymetrical turn-off times for the jets, with the jet reaction of the payload cooling system contributing to some asymmetry in "up" and "down" propellant expenditure. A significant decrease in roll-yaw consumption may be attributed to the presence of the cooling system exhaust, since, except for one period of approximately 0.4 seconds at capture, the upper and lower left roll-yaw jets apparently did not actuate. Roll attitude control could be accomplished by differential thrusting periods between the upper right (UR) and lower right (LR) jets without actuating the upper left (UL) or lower left (LL) jets. An estimate of the impulse provided by the payload cooling system may be established by the impulse expenditures of the roll-yaw jets. The vehicle maintained the desired orientation; therefore, the angular impulse provided by the control jets must equal the angular impulse provided by the payload cooling system.

Roll-yaw jet, yaw moment arm, l_y	4.66 ft.
Payload cooling system, yaw moment arm, l_c	3.85 ft.
Roll-yaw impulse expenditure, I_y	40 lb-sec.
Mission Time (active control), t_m	1266 sec.

Assuming that additional moment sources were not present, the yaw angular impulse provided by the roll-yaw jets was approximately 190 ft lb-sec., which should equal the payload cooling system angular impulse. If it is further assumed that the output from the cooling system was constant, then for the 1266-second mission the yaw component of cooling system thrust (T_c) would be:

$$T_c = l_y I_y \ / \ t_m l_c \qquad \text{lbs}$$

$$= 0.038 \qquad \text{lbs}$$

Estimates of cooling system thrust have varied considerably. A coolant utilization of 0.003 lbs/sec and an exhaust velocity of 555 ft/sec establishes an upper bound on thrust of approximately 0.055 lbs which conforms generally to the requirements. However, the thrust estimate that was provided with this information was approximately 0.005 lbs for a "nozzle efficiency" of 10 percent. Other estimates have indicated thrust levels which varied between 0.002 and 0.2 lbs, depending on temperature, coolant utilization and the particular mathematical model used.

Attitude Reference

Accuracy of the attitude reference and the programmer is not independently determinable, however the reentry angle error of approximately 0.5 degrees indicated low drift and initial misalignments of the reference, as well as accurate programming. The terminal error includes initial condition attitude errors at launch vehicle VECO, program errors, drift errors during coast, and separation errors. It is therefore concluded that the attitude reference, programmer, timer, and inverter performed as expected, since off-design operation of any one of these components would have resulted in significant time and/or angle errors.

Reaction Controls Motor Valve Operation

Based on the flight data, the motor valves operated every time a command of sufficient duration was supplied. There are several instances where a valve was commanded to operate by the guidance and control system; however, before the chamber pressure could increase to close the pressure switch, electrical power was removed from the valve. In each case, the signal was applied to the valve for less than 10 milliseconds. Since the time required for the valve to open and the pressure switch to close is approximately 15 milliseconds, the motor operation is considered normal.

At two points on the trace, it appears that a pressure switch closed and then opened without an electrical command to the valve. Both indications occurred during periods of low telemetry signal strength. The first indication was present at the end of the Antigua record and overlap available from the Ascension record did not confirm the occurrence; it is, therefore, assumed to be noise. The second indication occurred shortly after Ascension assumed monitoring responsibility and the signal strength was still low. The characteristic of the trace and the condition of the telemetry signal were similar to the first occurrence. Because of the similarity and since the subsequent operation of the motor was normal, the second indication is also assumed to be noise.

Throughout the flight, the majority of the corrective commands about the yaw and roll axes were to the upper right and lower right motors. During the capture maneuver, the lower left roll motor fired for about 0.40 seconds total and then did not receive a firing command for the rest of the mission. There is no positive indication that the upper left roll motor was commanded to fire at any time during the mission.

Based on comparisons of valve command to chamber pressure switch closures, the response time of the motors was low. Flight data indicates the valve response times were less than 15 milliseconds which is well within the performance requirements of the system.

Reaction Controls Thrust Levels

The only method that is available for determining thrust levels is based on the pitch, roll and yaw displacement rates. Because of low instrumentation sensitivity, pitch displacement rate during the pitch maneuver was the only useable data. The thrust of the two pitch motors as calculated from this data is compared with the predicted values based on pre-flight checkout in the table on page 6-4-4. Chamber pressure and system dynamic pressure during flight were calculated from the indicated thrust levels as shown in the following table. The thrust levels and system regulated pressure during flight agree closely with the predicted. It should be noted that the accuracy of the calculated flight thrust levels is estimated to be about plus or minus five percent.

COMPARISON OF PREDICTED & ACTUAL SYSTEM OPERATING PARAMETERS

MOTOR	CONDITION	THRUST-POUNDS	CHAMBER PRESSURE-PSIA	SYSTEM DYNAMIC PRESSURE-PSIA
Pitch Up	Flight (Calc.)	5.36	234	321
	Predicted	5.07	221	303
Pitch Down	Flight (Calc.)	5.17	226	309
	Predicted	5.16	225	307

The only regulated system pressure that was monitored in flight was the total (no flow) pressure. This pressure, plotted versus flight time in Figure 6-4-9 averages about 335 psia. This compares favorably with the predicted pressure of 327 psia.

Reaction Controls Nitrogen Consumption

The amount of nitrogen that was used during the flight was determined by two different methods. The first method, based on the nitrogen tank pressure at the beginning and end of flight, assumed an isothermal expulsion process. A plot of this pressure versus flight time, based on telemetry data, is presented in Figure 6-4-9. In the second method the weight of nitrogen was based on the total motor firing times as determined from the telemetry data and a mean specific impulse measured in pre-flight tests. Based on telemetered unregulated nitrogen pressure data, 0.49 pounds of nitrogen were used during flight, compared to 1.03 pounds calculated from the motor firing times. Since 11.1 pounds of nitrogen were available for use at launch, only four to nine percent of the available nitrogen was used. Because the pressure decay in the tank is small over the entire mission (compared to the accuracy of the instrumentation) and heat input into the tank is unknown, the accuracy of the calculation based on the pressure trace is questionable.

Although the weights of nitrogen calculated by these two methods differ by a factor of two, it confirms that an extremely small amount of nitrogen was used. The difference between the calculated consumed weights of nitrogen is well within the accuracy of the instrumentation. The difference can be explained considering only the error contribution of the unregulated nitrogen pressure transducer ($\pm$1.2% of full scale or $\pm$42 psi). Instrumentation limitations are also considered responsible for the indication of unregulated system pressure rise at approximately 700 seconds flight time as shown in Figure 6-4-9.

Reaction Controls System Leakage

System pre-flight history and telemetered flight data indicate that the system experienced no significant leakage throughout the entire operation. The system was serviced two weeks prior to flight to 3100 psia. The system went through several countdowns in which the regulated system was pressurized and then was vented at countdown termination. At launch, the pressure of the tank was 2985 psia. As discussed above, the motor firings alone account for all of the telemetered pressure drop in the system. It is, therefore, evident that there was no significant leakage from the system.

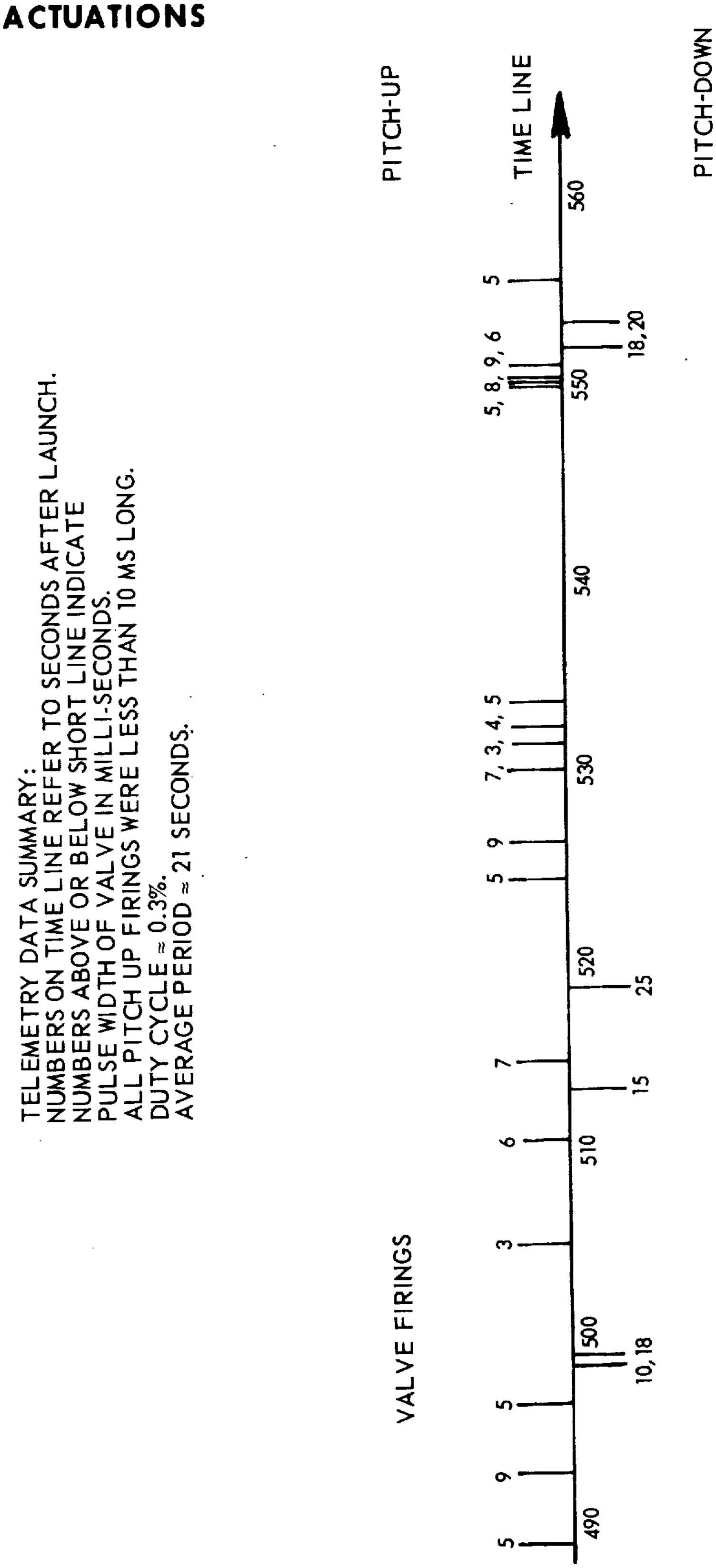
Conclusions - Reaction Controls

Based on review of the flight data, the following general comments can be made concerning the operation of the Reaction Control System:

(a) System operation was satisfactory and within design limits.
(b) The motor valves operated normally upon command.
(c) The thrust obtained from the motors was close to the predicted value.
(d) The nitrogen consumption was considerably lower than predicted.

VELOCITY PACKAGE PERFORMANCE
FIGURE NO. 6-4-6
INTEGRATED REPORT NO. GDA/BKF64-018
LTV/A REPORT NO. 3-30000/4R-75
GUIDANCE

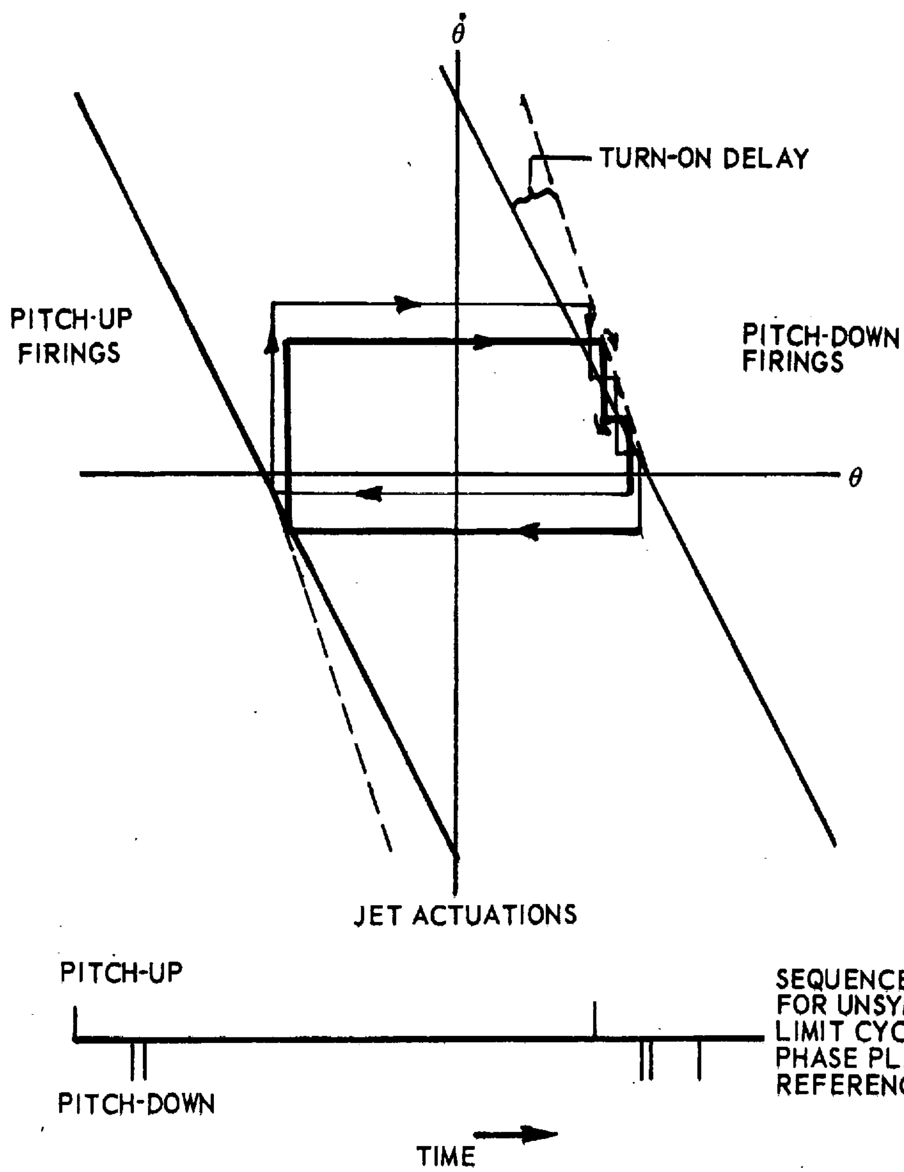
TYPICAL PITCH VALVE ACTUATIONS



IDEALIZED UNSYMMETRICAL LIMIT CYCLE

UNSYMMETRICAL LIMIT CYCLE DUE TO DIFFERENCES IN MINIMUM PULSE WIDTH

ASSUMPTIONS:
MINIMUM PITCH UP FIRING 2.5 TIMES MINIMUM PITCH DOWN FIRING
IMPULSIVE VELOCITY CHANGE



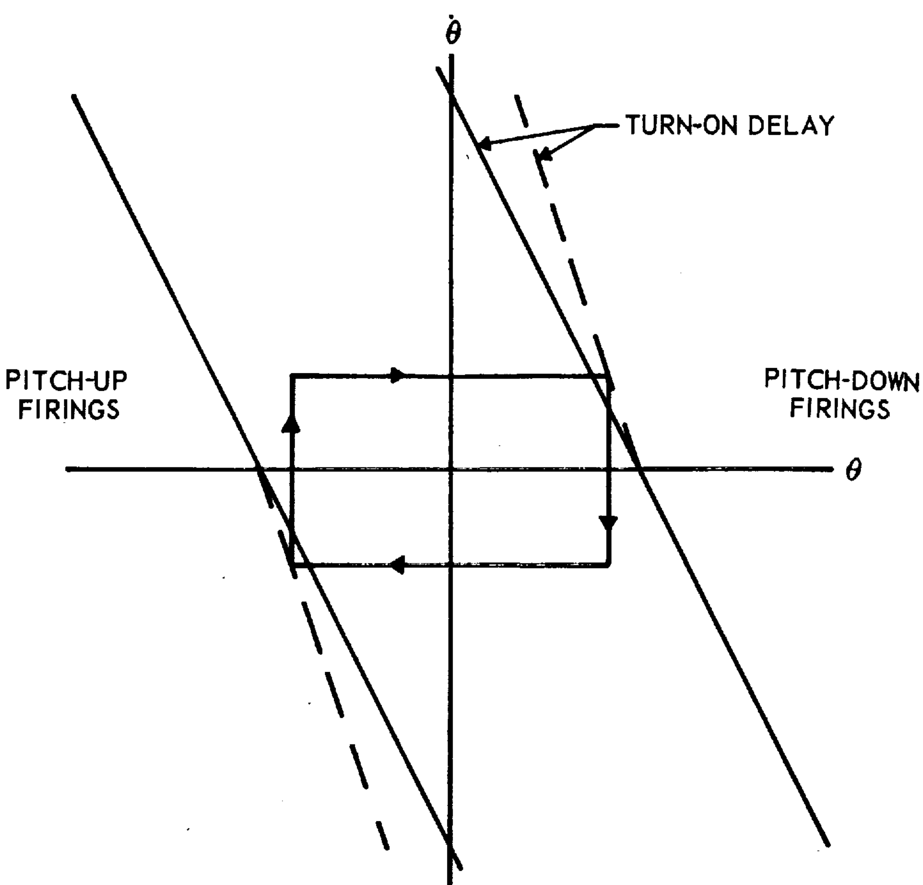
SEQUENCE OF MOTOR FIRING FOR UNSYMMETRICAL LIMIT CYCLE SHOWN IN PHASE PLANE
REFERENCE IMPULSE/UNIT TIME = 1

VELOCITY PACKAGE PERFORMANCE
FIGURE NO. 6-4-8
INTEGRATED REPORT NO. GDA/BKF64-018
LTV/A REPORT NO. 3-30000/4R-75
GUIDANCE

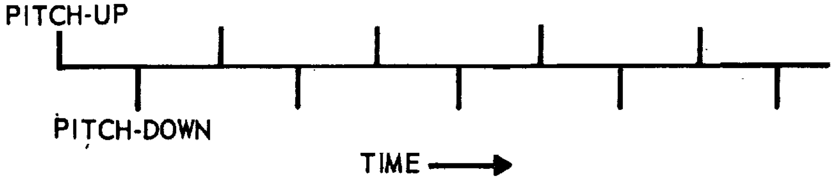
IDEALIZED SYMMETRICAL LIMIT CYCLE

SYMMETRICAL LIMIT CYCLE
WITH EQUIVALENT AVERAGE
IMPULSE PER CYCLE
(REF. FIGURE 6-4-6)

ASSUMPTION:
IMPULSIVE VELOCITY CHANGE



JET ACTUATIONS



SEQUENCE OF MOTOR FIRINGS
FOR SYMMETRICAL LIMIT CYCLE
SHOWN IN PHASE PLANE
REFERENCE IMPULSE/UNIT TIME = 1
UNSYMMETRICAL LIMIT CYCLE
(REF. FIGURE 6-4-7) EQUIVALENT
IMPULSE/UNIT TIME = 0.42

SECTION 5

PYROTECHNICS ANALYSIS

The ANTARES II A5 rocket motor ignited 6.00 seconds after receiving the ignition signal. This was within the design tolerance of 6.25 ± 1.0 seconds. A cursory review of the Reentry Package acceleration data indicates that the motor performed satisfactorily. The trajectory verified that sufficient thrust was provided so that the R/P exceeded the mission objective of 37,000 feet per second at reentry.

Satisfactory Velocity Package spin motor performance resulted in achieving an initial spin rate of 161 rpm. A slight increase in the spin motor temperature at spin-up confirmed that all three spin motors fired. Although no special instrumentation was provided for the other pyrotechnic devices, satisfactory performance of their respective systems indicates that the devices functioned properly.

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SECTION 6

IGNITION-DESTRUCT SYSTEMS ANALYSIS

Ignition System

The ignition system operated satisfactorily. The launch vehicle discrete (primary) times and the V/P event traces showed that the V/P received and responded to guidance discrete signals rather than the backup signals. This confirmed that all V/P ignition events were accomplished by ignition system No. 1 rather than by system No. 2, which operates from the launch vehicle backup signals.

Special in-flight instrumentation was not installed to monitor the V/P batteries, however, the successful systems operation indicated that battery performance was satisfactory. All battery voltages were normal at lift-off. The ignition-destruct batteries were activated at T-15 minutes and during load checks at T-11 minutes the voltages of ignition-destruct batteries No. 1 and No. 2 were 30.7 and 30.8 volts, respectively.

Destruct System

The destruct system performed satisfactorily during pre-launch checkouts. The system was not required during the flight.

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SECTION 7

STRUCTURAL SYSTEMS ANALYSIS

General

The V/P did not have any specific instrumentation to verify structural integrity. However, the success of Flight No. 1 indicates that the V/P structure provided the necessary rigidity for the various systems/components and that no structural failures occurred. An actual weight and balance summary is shown in the following table.

ACTUAL WEIGHT AND BALANCE SUMMARY

	Weight Pounds	Roll Z_{cg} In.	Pitch X_{cg} In.	Yaw Y_{cg} In.	Roll I_{zz} Slug-Ft^2	Pitch I_{xx} Slug-Ft^2	Yaw I_{yy} Slug-Ft^2
V/P Adapter (with clamp)	238.90	478.6	99.8	100.8	50.0	30.9	29.9
V/P Shell and Dynamic Balance Weights	789.03	435.6	99.8	99.9	124.7	111.2	109.6
ANTARES Ring Adapter and Dynamic Balance Weights	27.03	424.3	100.0	100.0	1.75	0.88	0.88
V/P Heat Shroud (S/N 00003)	294.93	367.6	100.7	100.0	42.5	88.9	83.1

Heat Shroud Separation

The heat shroud separation was very clean. Minor disturbances were noted on all three vibration accelerometer traces and on the Velocity Package pitch and yaw rate traces at the time that the separation bolts were fired. These disturbances were expected, however, and damped out within approximately 0.1 seconds. No other disturbance was detected during separation. The Launch Vehicle telemetry showed that the shroud separation was initiated at the proper time by the discrete signal. The L/V also provided the backup signal, which was not required on this flight.

V/P Separation From the V/P Adapter

The Velocity Package separated from the launch vehicle cleanly, and with a very small tip-off effect. The maximum angular rates imparted to the V/P were approximately 0.6 deg/sec left in yaw, 1 deg/sec right in roll, and 0.5 deg/sec up in pitch. The separation occurred properly at receipt of signal from the launch vehicle.

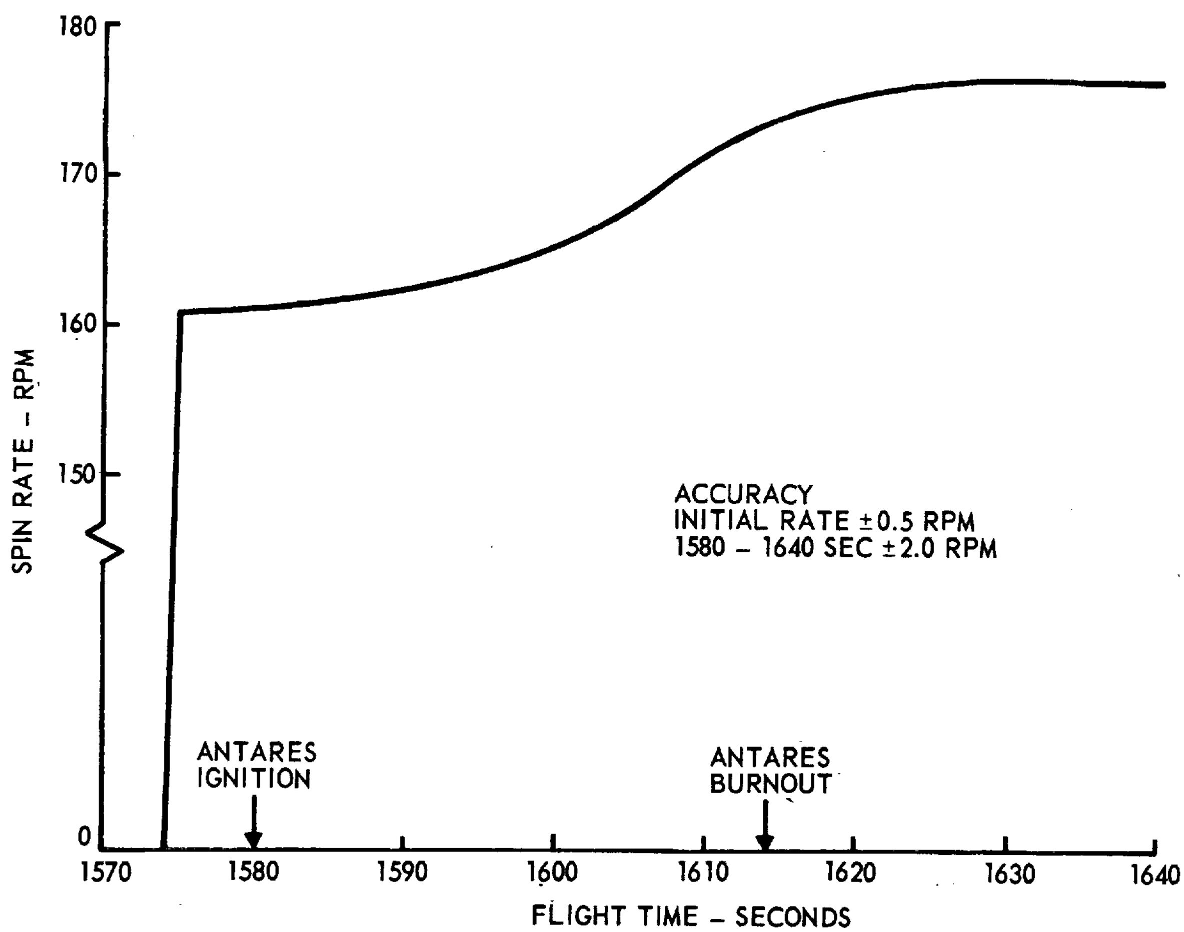
Spin-Up

A very slight disturbance was shown by the vibrometer traces at the time that the spin motors fired. The Velocity Package angular rate traces indicated a small disturbance when the spin motors fired, but did not show any coning during or after spin-up, thus indicating good thrust balance between the motors and good dynamic balance of the Velocity Package. The initial spin rate achieved was 161 rpm as determined from variations in the telemetry signal strength caused by antenna rotation. This spin rate is on the lower side of the predicted range of 169 $\pm$ 12 rpm. The impulse determined from test firings of this lot of spin motors at ambient temperatures was lower than the nominal value. This lower impulse would produce a spin rate very close to that recorded in flight. The spin rate increased during burning of the ANTARES motor to approximately 176 rpm at burn-out. This represents a 10 percent increase as compared to a maximum of 15 percent increase predicted before the flight. The measured spin rate as shown in Figure 6-7-3 indicates a slight increase in spin rate after ANTARES motor burn-out apparently due to accuracy limitations in data reduction or to outgassing effects, since no known torque is acting during this time.

Reentry Stage and V/P Separation

There was no indication of significant disturbance at the time of separation of the Reentry Stage from the Velocity Package. However, the longitudinal accelerometer trace was commutated and would not be expected to give an indication of short duration phenomena. The range of the rate gyros in the R/P was $\pm$ 170 deg/sec which would preclude the detection of very small angular rates induced at separation.

SPIN RATE


180
170
160
150
0
SPIN RATE – RPM
ACCURACY
INITIAL RATE ±0.5 RPM
1580 – 1640 SEC ±2.0 RPM
ANTARES
IGNITION
ANTARES
BURNOUT
1570
1580
1590
1600
1610
1620
1630
1640
FLIGHT TIME – SECONDS

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SECTION 8

THERMAL ENVIRONMENT ANALYSIS

General

The functional requirement of the FIRE Velocity Package thermal control system is to insure that all V/P systems and components operate within their design temperature limits throughout the pre-launch and coast phases of the mission. The thermal analyses of the Velocity Package showed that the following components were sufficiently marginal to warrant instrumenting for flight:

ITEM	TEMPERATURE, °F	
	Allowable	Calculated
ANTARES Rocket Motor	60 to 110	64 to 103
Spin Motors	-45 to 200	40 to 72.5*
Rate Gyro	185 max	186
PVE Unit (Poppet Valve Electronics)	348 max	281
Guidance Unit Assembly	252 max	242
Telemetry Transmitter	160** max	145

* Although calculated temperatures are well within limits, the temperature differences between motors are critical to prevent V/P "coning" during spin-up.

** Although the maximum allowable operating temperature is 160°F, it was desired that when the V/P was approximately mid-way between ground tracking stations, the transmitter should not exceed 120°F.

Figure 6-8-6, together with the following table, shows the location of the temperature sensor installations used to obtain telemetry data on the above components.

TEMPERATURE MEASUREMENTS MONITORED DURING FLIGHT

Code No.	Type of Measurement	Flight Sensor Location
T_1	ANTARES Case Range: 50°F to 250°F	Located within a sector of ± 10° from the V/P -Y-axis between stations 360 and 365
T_2	ANTARES Case Range: 50°F to 250°F	Located within a sector of ± 10° from the V/P +Y-axis between stations 360 and 365
T_3	ANTARES Base Range: 50°F to 250°F	Located at the ANTARES motor base
T_4	Spin Motor Case Range: 50°F to 250°F	Located on the forward (V/P reference) side of the spin motor
T_5	Spin Motor Case Range: 50°F to 250°F	Located on the forward (V/P reference) side of the spin motor
T_6	Spin Motor Case Range: 50°F to 250°F	Located on the forward (V/P reference) side of the spin motor
T_7	Equipment Operating Temperature Range: 50°F to 250°F	Located on the rate gyro base
T_8	Equipment Operating Temperature Range: 50°F to 250°F	Located on the base of the PVE unit
T_9	Equipment Operating Temperature Range: 50°F to 250°F	Located on the base of the telemetry transmitter
T_{10}	Equipment Operating Temperature Range: 50°F to 300°F	Located on the MIG block (NOTE: Vendor installed)

All temperature-instrumented V/P components operated well within their respective temperature limits. Performance of the thermal control system in no way impaired the operation and/or performance of any other V/P system or component. Based on these considerations and the close correlation between predicted and flight-recorded data, no modification or corrective action is required for the successful operation of the thermal control system on subsequent flights.

Prior to flight, the Prototype V/P, with installed operational systems, was tested in the LTV/A Space Environment Simulator (SES) as part of the V/P qualification program. The instrumentation for these tests included the above Flight Sensor Installation. From these tests, thermal data were obtained for the above components with the exception of the spin motors and the ANTARES rocket motor. The test consisted of three (3) thirty-minute cycles of 5 x 10^{-4} mm Hg simulated altitude. All components operated well within their respective temperature limits. Component temperature profiles showed that the transient thermal analyses were conservative, as expected. These SES temperature profiles form a part of the predicted performance of the V/P thermal control system.

The launch site ambient conditions at the time of launch were such that the initial V/P interior ambient temperatures were below the design maximum. Therefore, initial component temperatures were, excluding the PVE unit, 10 to 12°F below maximum values predicted from analyses but within 6°F of values predicted from SES testing. Difficulty in mounting a sensor unit directly under the PVE base at the point of highest heat flux used in analytical predictions resulted in the sensor being mounted on the edge of the base plate. Therefore, the initial PVE base temperature was considerably below the value predicted from analysis but within 13°F of that predicted from the SES test.

During boost, the heat shroud insulation blanket and the aluminized tape on the skin inner surface effectively prevented radiative heat transfer to interior components from the aerodynamically heated V/P exterior. This is shown in Figures 6-8-7 and 6-8-8 by the relatively flat shape of the left-hand portions of the component temperature profiles. Significant component heating from aerodynamic effects would have been evidenced by a steep temperature gradient during the time between launch and heat shroud ejection. Similarly, the absence of convective heating of components, either due to the hot V/P exterior surfaces or from the induction of boundary layer air, is evidenced by the flat slope of the temperature profiles during this flight period. Therefore, the thermal environment encountered by interior components not exposed directly to the space environment upon heat shroud ejection was almost entirely due to component internal heat generation. After heat shroud ejection, the environment of the ANTARES rocket motor and the spin motors is influenced by direct solar radiation. This is shown in Figure 6-8-7 by the gradually rising temperatures after approximately 400 seconds.

Spin Motors

The temperature of the three spin motors just prior to time of heat shroud ejection was approximately 75°F, with negligible temperature differences between motors. Upon shroud ejection, the motors received differing solar flux densities due to their varying circumferential positions. Each motor then experienced a slowly decreasing temperature until the time of V/P spin-up. The effectiveness of the white lacquer coating is shown by the average final spin motor temperature of approximately 69°F with a temperature spread of ±4°F between motors. The higher slope of the left-hand portion of the predicted upper limit curve results from the maximum initial spin motor temperature being above the solar radiation equilibrium temperature. Both predicted and flight data show that this equilibrium temperature is closely approached at the time of spin-up.

ANTARES Rocket Motor

The ANTARES motor case temperature profiles show the anticipated response characteristic both in slope and magnitude. All three sensor locations were at approximately 70°F at the time of heat shroud ejection. This is 23°F below the predicted upper limit due to the difference between the recorded initial temperature and the predicted initial upper limit. After shroud ejection, the two sensors represented by T_1 and T_2 received differing solar flux densities due to their varying circumferential positions, T_1 being more nearly normal to the sun. The temperature gradients of these two exposed sensors are very similar to the predicted upper limit curve slope. Thus, when corrected for initial temperatures, very close agreement is obtained between predicted and flight-recorded data with a final temperature of 82°F. The third sensor, T_3, was enclosed within the vehicle and remained cooler since it was heated only by adjacent components.

Rate Gyro Unit

The rate gyro unit is subjected primarily to internal heat generation. The temperature gradient for both the predicted and the flight-recorded data is fairly linear with the flight data showing a slightly lower slope than predicted in the analysis. This is attributed to conservation in the calculation of the gyro unit effective thermal mass. When corrected for the 10°F difference between predicted and flight-recorded initial temperatures, the final gyro unit temperature of 138°F is 30°F below the maximum predicted through analysis and approximately 26°F above that obtained during the SES test.

PVE Unit (Poppet Valve Electronics)

As previously discussed, mounting difficulties prevented location of the temperature sensor directly beneath the PVE base. Therefore, the predicted PVE temperatures were based on SES test results rather than analysis. Only a negligible difference in the temperature gradients exist between SES test and flight-recorded data. When corrected for a 15°F difference between predicted and recorded initial temperatures, the final flight-recorded PVE temperature of 140°F compares very well with the predicted value of 127°F.

Telemetry Transmitter

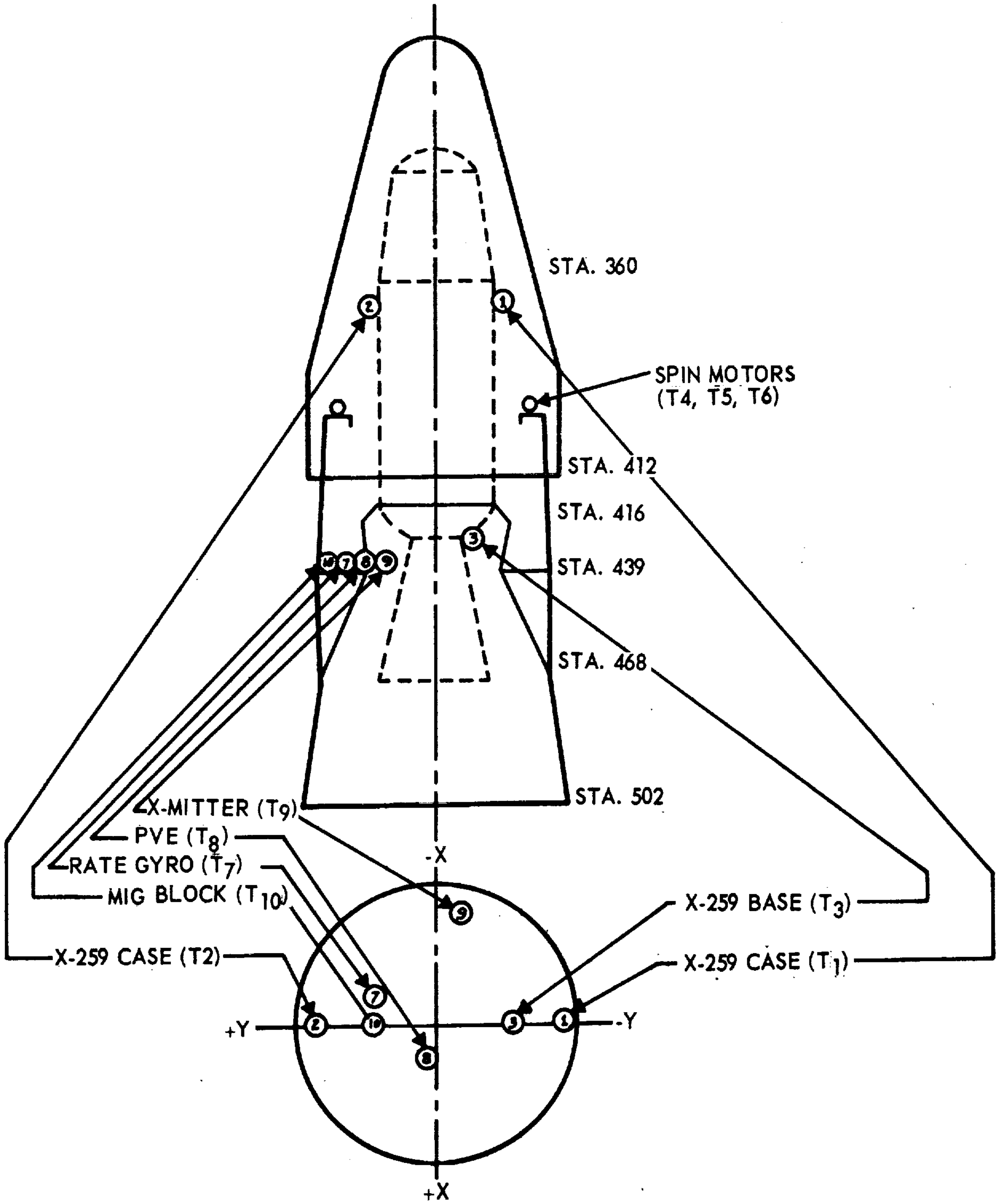
The telemetry transmitter in-flight temperature profile very nearly coincides with the predicted profile from the SES test. As anticipated, the temperature gradient resulting from the component internal heat generation is linear. The transmitter temperature at the time of the maximum output requirement midway between tracking stations was approximately 90°F which is well below the desired limit of 120°F. Final temperature of the transmitter was approximately 100°F compared with a maximum allowable temperature of 160°F.

Guidance Unit Assembly

The flight temperature profile shows that the MIG block temperature cycled about its control level at 180°F, which verified that the heaters maintained control of the MIG block temperature. Although the temperature of the guidance unit assembly itself was not recorded, the cycling of the MIG block temperature confirmed that the guidance unit assembly case temperature remained within the design operating limits.

VELOCITY PACKAGE PERFORMANCE
FIGURE NO. 6-8-6
INTEGRATED REPORT NO. GDA/BKF64-018
LTV/A REPORT NO. 3-30000/4R-75
THERMAL

TEMPERATURE SENSORS



PART 7

GUIDANCE SYSTEM PERFORMANCE

GENERAL DYNAMICS/ASTRONAUTICS
INTEGRATED REPORT NO. GDA/BKF64-018

APPROVED BY:

L. E. MUNSON
ASSISTANT PROGRAM DIRECTOR
FIRE PROGRAM OFFICE

SECTION 1

INTRODUCTION

Launch Vehicle (L/V) 263D was radio guided by the General Electric/Burroughs Mod III R&D ground guidance system located at Cape Kennedy. Guidance equations were generated specifically for the FIRE mission by General Dynamics/Astronautics. Because of the security classification of these equations and guidance system performance data, this part of the integrated report is limited to a word description of the results.

SECTION 2

DISCUSSION

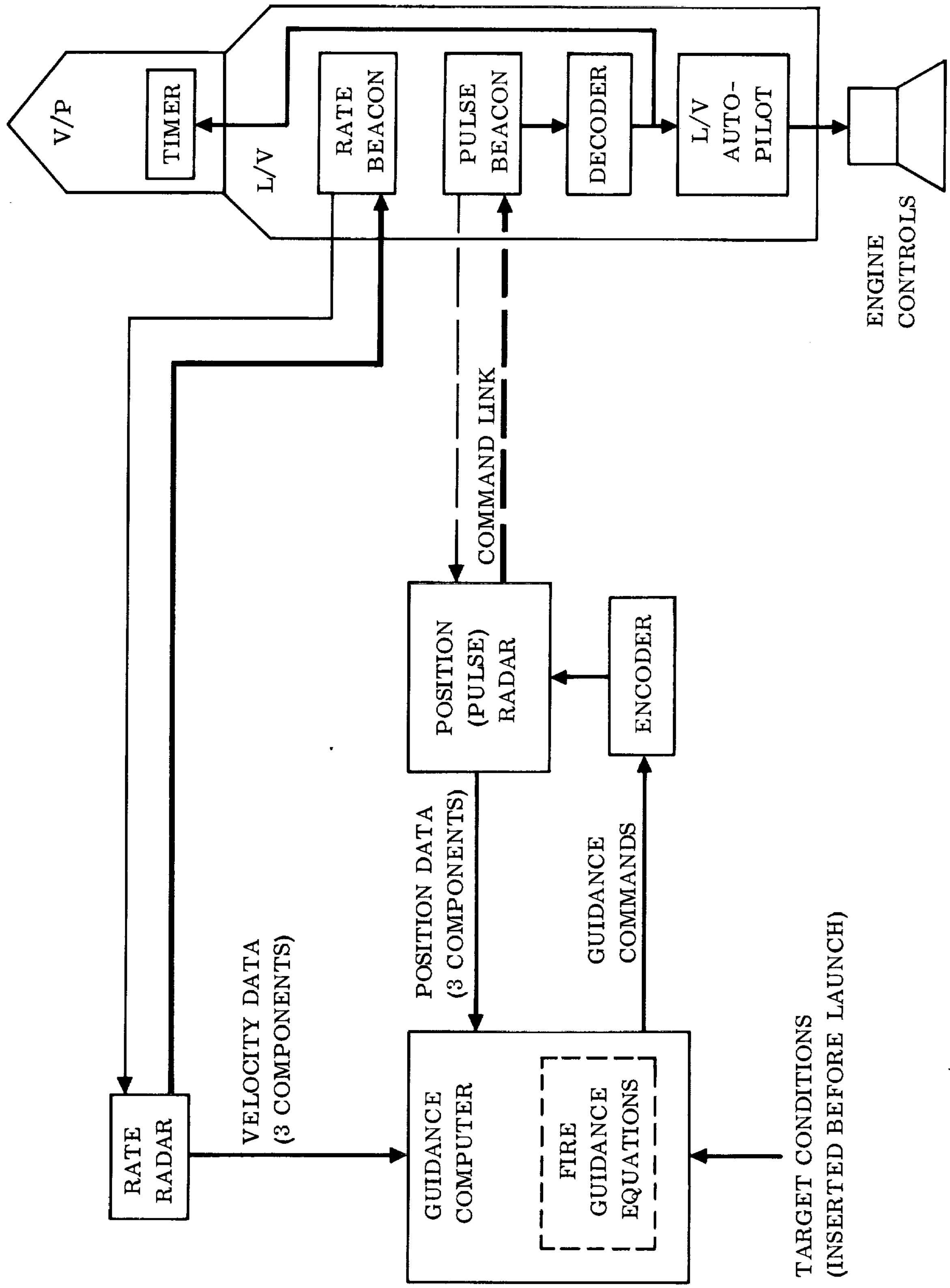
The basic techniques used in radio guidance involved controlling the attitude of the thrust vector, and hence the orientation of the velocity vector, through the use of steering commands which control the zero reference of the L/V autopilot in pitch and yaw. The magnitude of the velocity vector was controlled through the use of thrust termination. All guidance commands were transmitted from the ground over the command link provided by the Mod III radar system. Steering commands were transmitted in an analog fashion. Thrust termination and other guidance functions were in the form of discrete relay closures in the vehicle and were activated by discrete commands from the ground. Yaw steering on the FIRE mission controlled the lateral miss distance. Pitch steering was based on the semiminor axis of the desired coast ellipse which resulted in the proper flight-path angle at the target point. The velocity cutoff was determined by calculating the velocity required to intersect the target at the existing flight-path angle. During sustainer phase, the thrust attitude of the vehicle was continuously calculated. Sustainer thrust was terminated at the proper time to achieve the velocity required to satisfy the target conditions. Because of the relatively large propellent pad for this mission, a backup sustainer-cutoff capability was held in reserve during this flight. In the event of a guidance system failure, this command, generated by the range safety computer, would have been supplied to the L/V through the redundant range-safety command link. When sustainer cutoff occurred, the actual attitude was compared with the required pitch and yaw attitudes. A steering maneuver was made during the vernier phase to align the vehicle at the desired attitude. Also during vernier phase, a command based on the predicted time of flight to the target was sent to start the velocity package (V/P) timer. Another command, based on a fixed elapsed time from the sustainer cutoff discrete, was used to jettison the V/P nose fairing.

Figure 7-2-3 is a simplified block diagram of the overall guidance system. The guidance computer shown in this diagram contained equations which transformed measured radar quantities into the desired steering and discrete commands which caused the L/V to satisfy FIRE mission requirements. For this mission the requirements were 1) to place the spacecraft at a downrange target position and altitude with the proper velocity and flight path angle, 2) to start a timer in the V/P at the appropriate time to ignite the Antares II-A5 rocket at the target point, and 3) to provide the V/P with an accurate attitude reference at L/V and V/P separation. Active ground guidance was terminated with the transmission of the spacecraft separation discrete.

The guidance system also generated a command to stage the booster engines at the desired acceleration level and a command to enable the V/P pyrotechnic ignition-interlock circuits. The criterion for the latter command was that the discrete command be transmitted five seconds prior to booster cutoff.

The nominal vehicle trajectory was designed to achieve desired mission objectives with minimum assistance from the guidance system. Most of the guidance correctional capabilities were held in reserve in order to correct for possible vehicle perturbations.

SIMPLIFIED BLOCK DIAGRAM
OF RADIO GUIDANCE SYSTEM



SECTION 3

CONCLUSIONS

All L/V guidance objectives for FIRE Flight No. 1 were satisfied. The downrange tracking facilities indicated close agreement with the target conditions predicted at the termination of L/V guidance. Guidance computer and radar performance, and L/V operating characteristics were well within the expected limits. The backup auxiliary sustainer-cutoff command, generated by the range safety computer from AZUSA tracking data, was held in a standby condition for this flight. The performance of this command would have been satisfactory had it been required.

PART 8

LAUNCH VEHICLE PERFORMANCE

GENERAL DYNAMICS/ASTRONAUTICS
INTEGRATED REPORT NO. GDA/BKF64-018

APPROVED BY: ______________________
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SECTION 1

INTRODUCTION

The first Project FIRE launch vehicle, Atlas 263D, was successfully launched from the AMR, Complex 12, at 1642 EST on 14 April 1964. The Atlas space launch vehicle (produced by General Dynamics/Astronautics (GD/A)) placed the FIRE spacecraft (a Velocity Package (V/P) produced by Ling-Temco-Vought/Astronautics (LTV/A) and a Re-entry Package (R/P) produced by Republic Aviation Corporation (RAC)) into a precise ballistic trajectory calculated to place the R/P at a specified spatial location and time near Ascension Island. All GD/A test objectives were satisfactorily accomplished.

The purpose of Part 8 of this report is to present a summary of the results achieved from the Launch Vehicle (L/V) only as related to the Project FIRE Flight No. 1.

SECTION 2

LAUNCH VEHICLE CONFIGURATION

A brief description of the Atlas 263D FIRE launch vehicle systems is presented below:

An MA-5 rocket engine propulsion system consisted of a booster, sustainer, and vernier engine assembly. The booster engine utilized baffled injectors and the "wet start" procedure. A Kel-F liner was incorporated in the sustainer engine lox pump inlet adapter. The propulsion system engines were gimbal-mounted for control of vehicle attitude and direction in response to guidance system and autopilot commands. Only the booster engine used hypergolic ignition.

A jettison mechanism was carried to jettison the booster engine and associated fairings, pumps, lines, tanks, etc. The system consisted of 10 pneumatically-operated jettison fittings positioned around the tank section adapter ring and the necessary manifolds, lines, valves, wiring, and helium supply to actuate these valves. The flight programmer activated the system at the termination of booster engine flight phase.

The flight control system consisted of a gyro package, a filter-servoamplifier package, a programmer package, an excitation transformer (all mounted in the B1 equipment pod), a remote rate gyro package located at Station 675, and 10 hydraulic actuator assemblies connected from the thrust chamber to the vehicle structure. The gyro package contained the three displacement gyros and the associated electronic circuitry. The remote rate gyro package contained the roll, pitch, and yaw rate gyros. A gyro package and a remote rate gyro package were maintained as a matched set. The filter-servoamplifier package contained the filters, integrators, and 10 servoamplifiers. The hydraulic actuator assemblies included the hydraulic controllers and the position feedback transducers. The programmer package (a completely electronic unit) contained a digital clock, low and high power switches, roll and pitch program devices, and discrete logic circuitry. The excitation transformer provided the vernier bias voltage and excitation supply voltage to the feedback transducers. Staging backup was provided by an acceleration switch set for 7.80 g's. The gyro self-check system, consisting of a spin motor rotation detection (SMRD) system and self-test rate gyros, was incorporated in the gyro packages. The booster engine actuators were offset differentially 0.108 degree in yaw to cause a counterclockwise roll torque to neutralize clockwise roll torque caused by such factors as liftoff transients, booster turbine exhaust, and slight thrust vector imbalance.

The vehicleborne pneumatic system consisted of regulators, relief valves, six titanium helium bottles, and one fiberglass helium storage bottle. These bottles supply helium gas for booster stage propellant tanks pressurization, engine controls and staging pressure. During helium loading, the pressurized gas was chilled by liquid nitrogen to load the maximum weight of gas in the five booster helium bottles; and during flight, the gas was expanded by heating in a heat exchanger to provide maximum utilization of the gas. A pneumatically-operated, electrically controlled lox tank boiloff valve was installed, which had a nominal control range of 3.7 psig to 5.4 psig.

The vehicleborne hydraulic system included two independent subsystems to supply the operating pressure required to position the engine thrust chambers and for control of the sustainer engine head suppression, propellant utilization, and gas generator blade valves. The booster and the sustainer/vernier hydraulic systems each included a variable displacement pump, a reservoir, accumulators, actuators, and associated valves and plumbing. Vernier solo hydraulic power was supplied by two 25-cubic inch hydraulic accumulators. Check valves and pressure switches were incorporated in the booster and sustainer high pressure plumbing for added system reliability.

The electrical subsystem was composed of a 19-cell, 28-VDC main vehicle battery and a 115-VAC, 3-phase, 400-cps rotary inverter. A changeover switch provided for switching both AC and DC power from external ground power to internal battery and inverter.

A GD/A propellant utilization (PU) system, operating closed-loop, was used. This system is designed to regulate the oxidizer and fuel flows to the sustainer engine in order to maintain the proper balance of residuals in the propellant tanks. The PU system consists of two mercury manometers and a computer-comparator which includes a mass ratio error detector assembly and a PU valve controller assembly.

A type C coherent carrier transponder Azusa system consisted of one transponder canister, coaxial cable, and two antennas (tilted beam and modified Cape).

A range safety command system consisted of two receiver/decoders, each with self-contained power supplies and a single destructor unit.

The MOD IIIG solid-state vehicleborne guidance system, operating closed-loop, consisted of a rate beacon, pulse beacon, decoder, one flush antenna assembly and associated waveguide and cabling.

The Atlas airframe consisted of propellant tanks, a booster thrust section and two equipment pods. A special adapter section, provided by LTV, was attached to the forward end of the lox tank. Like the Atlas ICBM, the two retrorockets were mounted inside the No. 1 pod forward fairing.

A telemetry system consisting of one standard 17 channel PAM/FM/FM RF package, accessory package and associated antenna system was installed for monitoring areas of interest.

For a more detailed inspection of the FIRE launch vehicle systems, diagrams are included at the end of Section 4, Launch Vehicle Performance Summary.

SECTION 3

GD/A TEST OBJECTIVES

The following table presents the list of flight objectives which were scheduled for Atlas 263D and against which data was obtained and evaluated.

Description	Priority	Satisfied
Demonstrate the ability of Atlas to place the separable upper stage at a predetermined position and velocity in space as defined by the appropriate guidance equations. The MOD IIIG General Electric/Burroughs guidance subsystem will provide discretes and steering commands to achieve the trajectory defined by the guidance equations.	1	Yes
Determine Atlas systems flight performance utilizing telemetry data.	1	Yes
Demonstrate the structural integrity, during flight, of the Atlas portion of the assembled vehicle.	2	Yes
Obtain data on the Atlas trajectory and on the guidance equipment performance utilizing the MOD IIIG General Electric/Burroughs guidance system to generate the necessary flight control commands.	2	Yes
Demonstrate the ability of the Atlantic Missile Range support equipment to obtain external telemetry and tracking data throughout the vehicle powered flight.	2	Yes
Demonstrate that the Atlas flight programmer and MOD IIIG General Electric/Burroughs guidance system provided the correct commands for flight operations peculiar to this program.	2	Yes

Description	Priority	Satisfied
Demonstrate that the Atlas flight control system has the ability to stabilize and control the Atlas vehicle in proper response to guidance commands generated by the GE/Burroughs guidance system to achieve the desired trajectory.	2	Yes
Demonstrate that the Atlas flight control system has the ability to stabilize and control the Atlas vehicle during the flight programmer portion of the pitchover program.	2	Yes
Obtain data on the performance of the Azusa type C transponder and characteristics of associated airborne antenna.	3	Yes

SECTION 4

LAUNCH VEHICLE SYSTEMS PERFORMANCE SUMMARY

SPACECRAFT TRAJECTORY INSERTION

Guidance radar data indicated the FIRE spacecraft was injected into a specified ballistic trajectory (free-fall ellipse) at the termination of booster powered flight and separation was satisfactorily accomplished.

As interpreted at VECO, guidance radar data indicated that the insertion parameters placed the FIRE spacecraft into a proper ballistic trajectory so that ignition of the Antares IIA5 would occur very close to the planned nominal target point.

PROPULSION SYSTEM

The performance of the propulsion system was satisfactory. Normal operating characteristics were reflected in all system data. Engine cutoff commands were properly generated by guidance discrete signals.

System Redline parameters were within specified limits at engine start and are listed below:

TABLE 8-4-1. REDLINE PARAMETERS AT ENGINE START

Parameter	Units	Redline Limit	Engine Start Value
Booster lox regulator reference pressure	psig	576 to 596	586
Sustainer lox regulator reference pressure	psig	809 to 849	833
B2 Turbine inlet temperature	°F	> 0	89
S Turbine inlet temperature	°F	> 0	82
Lox temperature at breakaway valve	°F	< -283	-301

Inflight Booster Engine Performance

Performance of the booster engine was satisfactory. Telemetered system data displayed satisfactory trends and values throughout the booster operational mode. Booster engine data is tabulated below.

TABLE 8-4-2. BOOSTER ENGINE FLIGHT DATA

Measurement	Units	Liftoff	+10 seconds	BECO
B1 chamber pressure	psia	544	553	562
B1 pump speed	rpm	6113	6150	6187
B2 chamber pressure	psia	541	547	559
B2 pump speed	rpm	6122	6184	6184
BGG combustor pressure	psia	492	499	499
Lox regulator reference pressure	psia	603	603	593
B1 lox pump inlet pressure	psia	61	64	>100(1)
B1 fuel pump inlet pressure	psia	67	67	57
B2 lox pump inlet pressure	psia	60	66	>100(1)
B2 fuel pump inlet pressure	psia	69	68	57

NOTE: (1) Data above 100% IBW.

Inflight Sustainer Engine Performance

Sustainer engine performance was also satisfactory. Engine thrust was calculated from chamber pressure data and an altitude thrust coefficient. This coefficient is dependent on the burning mixture ratio of the sustainer engine as indicated by the propellant utilization valve position. Sustainer engine data are tabulated below:

TABLE 8-4-3. SUSTAINER ENGINE FLIGHT DATA

Measurement	Units	Liftoff	BECO	SECO
Chamber pressure	psia	714	684	664
Pump speed	rpm	10,156	10,110	10,118
Fuel pump discharge pressure	psia	912	912	927
Gas generator discharge pressure	psia	649	640	640
Lox regulator reference pressure	psia	646	831	831
Lox pump inlet pressure	psia	68	113	79
Fuel pump inlet pressure	psia	73	69	41

Inflight Vernier Engine Performance

Vernier thrust chamber pressure reflected normal system operation. The engine lox and fuel tanks repressurized properly at BECO, and maintained nominal pressures to the end of recorded data. System data are as follows:

TABLE 8-4-4. VERNIER ENGINE FLIGHT DATA

Measurement	Units	Liftoff	BECO	SECO	VECO
V1 chamber pressure	psia	334	346	355	314
V2 chamber pressure	psia	342	346	352	312
Engine lox tank pressure	psia	608(1)	598(2)	694	603
Engine fuel tank pressure	psia	608(1)	603(2)	603	608

NOTE: (1) Prior to engine tanks vent.
(2) After engine tanks repressurization.

Total booster, sustainer, and vernier engine axial thrusts, as calculated from chamber pressure data, were in close agreement with the preflight simulation predicted thrusts as shown below.

TABLE 8-4-5. LAUNCH VEHICLE ACTUAL VS PREDICTED THRUSTS

	Liftoff	BECO	SECO	VECO
Booster Engine Thrust (pounds)				
Actual	307,900	366,300	---	---
Predicted	309,100	365,000	---	---
Sustainer Engine Thrust (pounds)				
Actual	56,600	79,200	78,400	---
Predicted	56,300	79,000	79,100	---
Vernier Engine Thrust (pounds)				
Actual	1,620	1,930	1,470	1,300
Predicted	1,720	1,980	1,470	1,380

FLIGHT CONTROL SYSTEM

Flight control system performance was satisfactory. The system generated the roll and pitch programs, accepted and executed guidance discrete and steering commands, generated the planned programmer switching functions and stabilized the vehicle throughout powered flight.

The pitchover maneuver was initiated at 15 seconds after liftoff and maintained through the end of the booster engine phase (refer to Table 8-4-6 below for Atlas 263D nominal pitch program). The actual pitchover angle at 133 seconds as evaluated by using MOD III tracking data, was -65.30 degrees. Comparison of this value with the nominal angle of -65.58 degrees indicates the vehicle's longitudinal axis was 0.28 degree high.

TABLE 8-4-6. VEHICLE 263D NOMINAL PITCH PROGRAM

Time (sec)	Programmer Output (volts)	Programmer Output Integral (volts-sec)	Rate (deg/sec)	Vehicle Angle (degrees)
		BOOSTER PHASE		
15	1.6	0.0	-0.602	0.0
30	2.0	24.0	-0.752	-9.02
45	2.1	54.0	-0.790	-20.30
55	2.0	75.0	-0.752	-28.20
65	1.8	95.0	-0.677	-35.72
75	1.6	113.0	-0.602	-42.49
85	1.3	129.0	-0.489	-48.50
100	0.9	148.5	-0.338	-55.84
120	0.6	166.5	-0.225	-62.60
133.192	0.0	174.415	0.0	-65.58

TABLE 8-4-6. VEHICLE 263D NOMINAL PITCH PROGRAM (Continued)

Time (sec)	Programmer Output (volts)	Programmer Output Integral (volts-sec)	Rate (deg/sec)	Vehicle Angle (degrees)
		SUSTAINER PHASE		
143.192	0.3	0.0	-0.1128	-65.58
288.500	0.0	43.59	0.0	-81.97

NOTE: The pitch program is based upon a nominal gyro torquing gain of 0.400 degree per volt-second, with an attenuation factor of 0.94 which gives an actual torquing gain of 0.376 degree per volt-second.

The booster pitch program ends 0.1 second after the BECO discrete or the "staging backup" acceleration switch signal, whichever occurs first.

The sustainer pitch program of -0.1128 degree per second was utilized from BECO discrete +10.0 seconds to SECO discrete.

Engine motion at mainstage ignition was small. The vehicle liftoff roll transient was clockwise 0.71 degree at a peak rate of 3.4 degrees per second. Atlas 263D employed the booster thrust chamber roll offset to reduce the roll magnitude at liftoff. Maximum aerodynamic loading occurred at approximately 66 seconds, requiring booster No. 1 and No. 2 thrust chamber deflections of +2.7 and +2.2 degrees, respectively, to maintain vehicle stability. The larger than usual deflections were due to the configuration of the upper stage and high winds aloft. Propellant slosh appeared in the yaw plane and coupled into the roll plane. Larger yaw transients occurred at BECO and jettison than is usual for Atlas space launch vehicles. Due to the mission constraints and a 0.5-second longer than nominal vernier solo, spacecraft separation resulted from the booster programmer backup at SECO +23 seconds rather than by the generated guidance discrete. However, this occurrence was anticipated by GD/A design and did not represent a problem.

Special Instrumentation

Additional instrumentation pertaining to this program's upper stage was added to the Atlas autopilot programmer switching functions and were telemetered to verify programmer switching operations during the flight. All programmer functions were generated satisfactorily.

GUIDANCE SYSTEM

Performance of the ground and vehicleborne Atlas guidance system was satisfactory. Steering commands were low in magnitude and deviations from the nominal trajectory were small. This Atlas SLV was not programmed to utilize booster-phase steering. Good track indications were received from 64.4 to 337.1 seconds and satisfactory rate flags were obtained from 52.0 to 336.0 seconds. The seven planned discrete commands were properly generated by the ground guidance station and properly received and decoded by the vehicleborne units.

Due to a 0.5-second longer than nominal vernier solo phase, the Separate Spacecraft discrete occurred at the same time as the programmer backup function (SECO +23 seconds). Accordingly, due to an additional decoder relay pickup time involved, the guidance discrete arrived after the function was initiated by the programmer. This sequence of possible events was anticipated prior to the flight and was a necessary consequence of the system constraints.

During the second launch attempt an erroneous enable V/P ignition interlock was generated by the computer at approximately T-15 seconds. An investigation resulted in a revision of the guidance station procedures during switching of the computer to Flight Ready mode to eliminate the transmission of the erroneous discrete.

PNEUMATIC SYSTEM

The pneumatic system adequately supported the flight. All pressurization and control functions were properly performed throughout flight. Two anomalies were observed in the pneumatics system prior to liftoff. The first anomaly consisted of oscillations in the vehicle lox tank ullage pressure between Phase III and engine start and is currently attributed to a F&G airborne ullage tank regulator which senses and corrects minor pressure variations before the pressure decrease is sufficient to initiate a standard leak/fill pressure correction. The second anomaly was an abnormal pressure spike (723 psig) in the ISS regulator discharge pressure at engine tanks vent 0.5 second prior to liftoff. The spike was observed for only one telemetry commutator segment; however, the pressure remained above the design criteria of 600 ± 32 psig for 4 seconds. The ISS regulator anomaly has been observed on Atlas Vehicles 105D and 215D. The exact reason for this pressure spike is not known; however it was not detrimental to the over-all performance of the regulator. The pressure spike occurred during the vehicle release and liftoff sequence. Since the vehicle was in motion this occurrence is considered to be an inflight anomaly.

Table 8-4-7 presents pressures that were monitored at significant times during flight.

TABLE 8-4-7. PROPELLANT TANK FLIGHT PRESSURE DATA (psig)

Parameter	-10 Seconds (1)	Liftoff (2)	BECO (2)	SECO	VECO
Vehicle Lox Tank Ullage Pressure - F1P	26.5	24.7	25.4	25.4	25.9
Vehicle Fuel Tank Ullage Pressure - F3P	59.4	58.4	58.5	41.0	41.0
Intermediate Bulkhead Differential Press. - F116P	15.21	13.98	12.44	14.74	15.21

NOTE:

(1) Static band parameters (internal pneumatic to engine start) are 24.3 to 26.7 psig for the vehicleborne lox regulator and 57.0 to 59.9 psig for the fuel regulator.

(2) Dynamic parameters (engine start to jettison) are 24.7 to 26.0 psig for the vehicleborne lox regulator and 57.0 to 59.9 psig for the fuel regulator.

HYDRAULIC SYSTEM

Hydraulic system performance was satisfactory. Telemetered data indicated that hydraulic pressure was properly maintained in the booster and sustainer/vernier subsystems throughout powered flight and supported all flight control, PU and engine control demands. Oil evacuation was initiated at -29.7 seconds in both subsystems. Telemetered data indicated fluctuations of the sustainer return pressure from T-51 seconds until initiation of oil evacuation. This can be attributed to pressure variations in the HSU reservoirs possibly due to minor fluctuations in the regulator output that pressurizes the reservoirs. The dual vernier solo accumulators maintained pressure for at least SECO +43.0 seconds at which time loss of the telemeter signal precluded recording of the complete pressure decay.

TABLE 8-4-8. HYDRAULIC SYSTEM PRESSURES (psia)

Measurement	Liftoff	BECO	SECO	VECO
Booster Hydraulic Pump Discharge Pressure	3190	3170	---	---
Booster Hydraulic System Return Pressure	78	72	---	---
Sustainer Hydraulic Pump Discharge Pressure	3010	3010	3010	---
Sustainer Hydraulic System Return Pressure	66	72	72	66

ELECTRICAL SYSTEM

Performance of the L/V electrical system was satisfactory. The main vehicle battery voltage and rotary inverter frequency and voltage were within specifications throughout Atlas powered flight. Main vehicle battery voltage Redline problems were encountered during the second launch attempt on 10 April 1964 and were resolved by ECN revisions of the Redline limits to values compatible with the high DC loads imposed on the battery. No further electrical Redline problems were encountered in the succeeding launch attempts.

TABLE 8-4-9. ELECTRICAL SYSTEM PARAMETERS AT SELECTED TIMES

Description	After P C/O	Liftoff	BECO	SECO	VECO	Tolerance
Vehicle DC Bus, (VDC)	27.2	27.2	27.9	27.9	27.9	26 to 30 VDC
Inverter Freq., (CPS)	400.2	400.2	400.2	399.9	399.9	394 to 406 CPS
Phase "A" Voltage, (VAC)	115.5	115.5	115.7	115.9	115.9	113.6 to 117 VAC

PROPELLANT UTILIZATION SYSTEM

Performance of the GD/A propellant utilization (PU) system was satisfactory. The PU valve responded properly to the Error Demodulator Output (EDO) signal throughout the flight. Response of the head suppression (HS) valve to the PU valve movements and to the effects of vehicle acceleration was also correct. Although an abrupt rise in the EDO signal occurred at 256.8 seconds (probably due to a fuel manometer dross ring) there was no adverse effect on the PU system since the PU valve is at or near the closed limit during this time of flight. The burnable residuals at SECO were 2,423 pounds of lox and 1,228 pounds of fuel which would have provided 12.74 seconds of additional sustainer engine operation with a fuel outage of 282 pounds. The predicted propellant residuals from the preflight trajectory simulation were calculated to be 2306 pounds of lox and 1135 pounds of fuel.

The calculated burnable residuals at SECO include the total lox remaining above the pump inlet, less the 70-pound nominal (for this program) lox depletion shutdown residual, and the total fuel remaining above the Station 1198 anti-vortex web. The residuals also include sustainer engine gimbal angle correction weights of 150 pounds of lox and 52 pounds of fuel.

AZUSA SYSTEM

Performance of the Azusa system was satisfactory. The angle cosines were switched to "fine" at 2 seconds and automatic track was established at 9 seconds. Azusa data was selected by the range IBM 7094 computer for impact predictions from 12 to 100 seconds and from 104 to 306 seconds.

RANGE SAFETY COMMAND SYSTEM

Performance of the range safety command system was satisfactory. Recorded signal strength at the receivers was adequate to provide command capability throughout the Atlas powered flight. The auxiliary sustainer cutoff signal (ASCO) generated by the Atlas ground guidance system after the normal SECO discrete command, was properly transmitted to the vehicle and decoded at 288.558 seconds. The manual fuel cutoff (MFCO) and destruct command signals were not required nor transmitted.

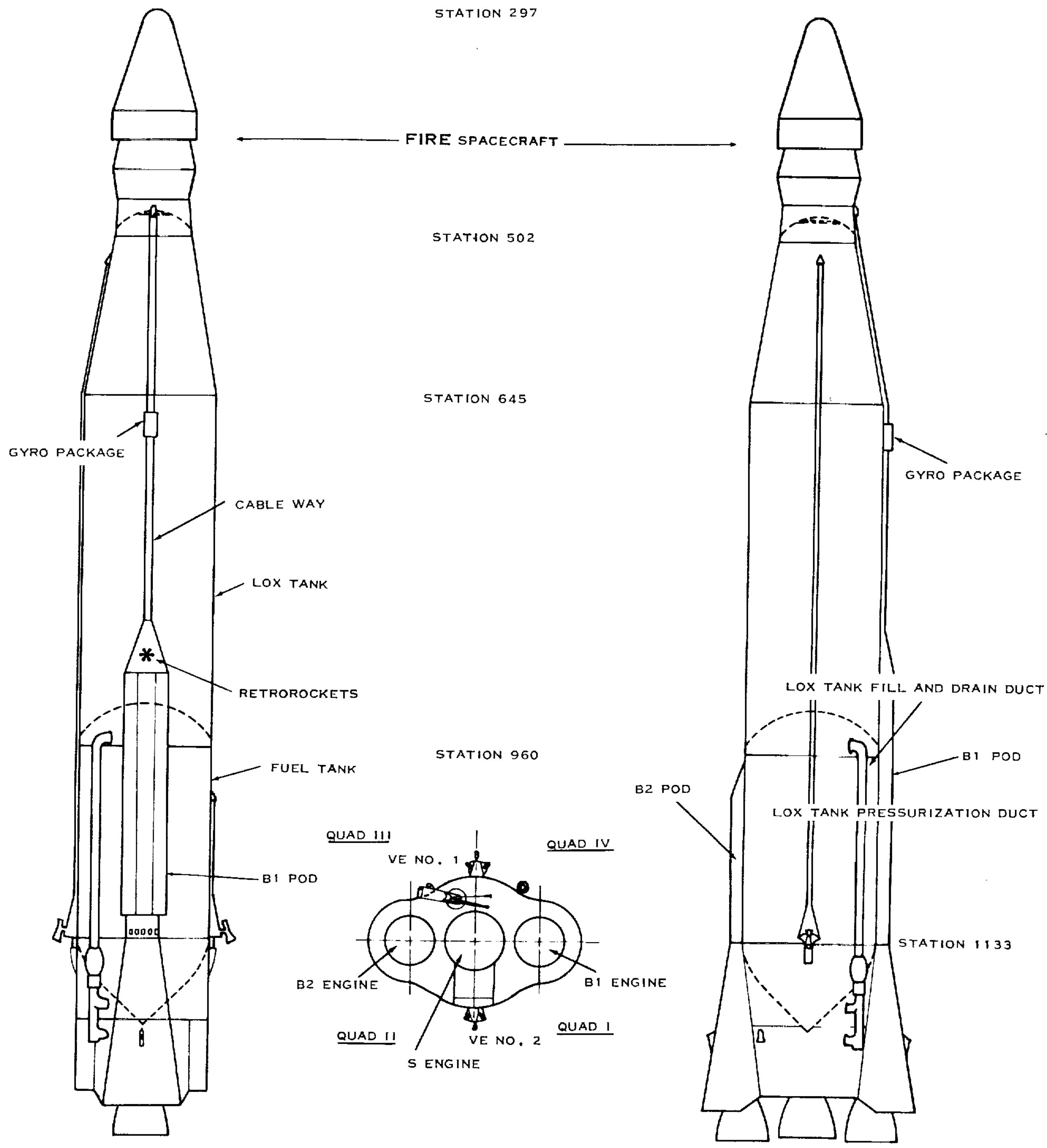
AIRFRAME SYSTEM

Vehicle structural integrity was maintained throughout powered flight and beyond spacecraft separation. The normal 5-cps longitudinal vehicle oscillations following liftoff attained a maximum amplitude of 0.68g (p-p) at 5 seconds and were damped by 23 seconds. The airframe was subjected to peak accelerations at BECO and SECO of 7.02g and 5.32g, respectively. Environmental conditions in the Atlas thrust section were satisfactory throughout flight with a maximum of 92°F recorded at staging by the Thrust Section Ambient temperature measurement in the Quad IV Area.

TELEMETRY SYSTEM

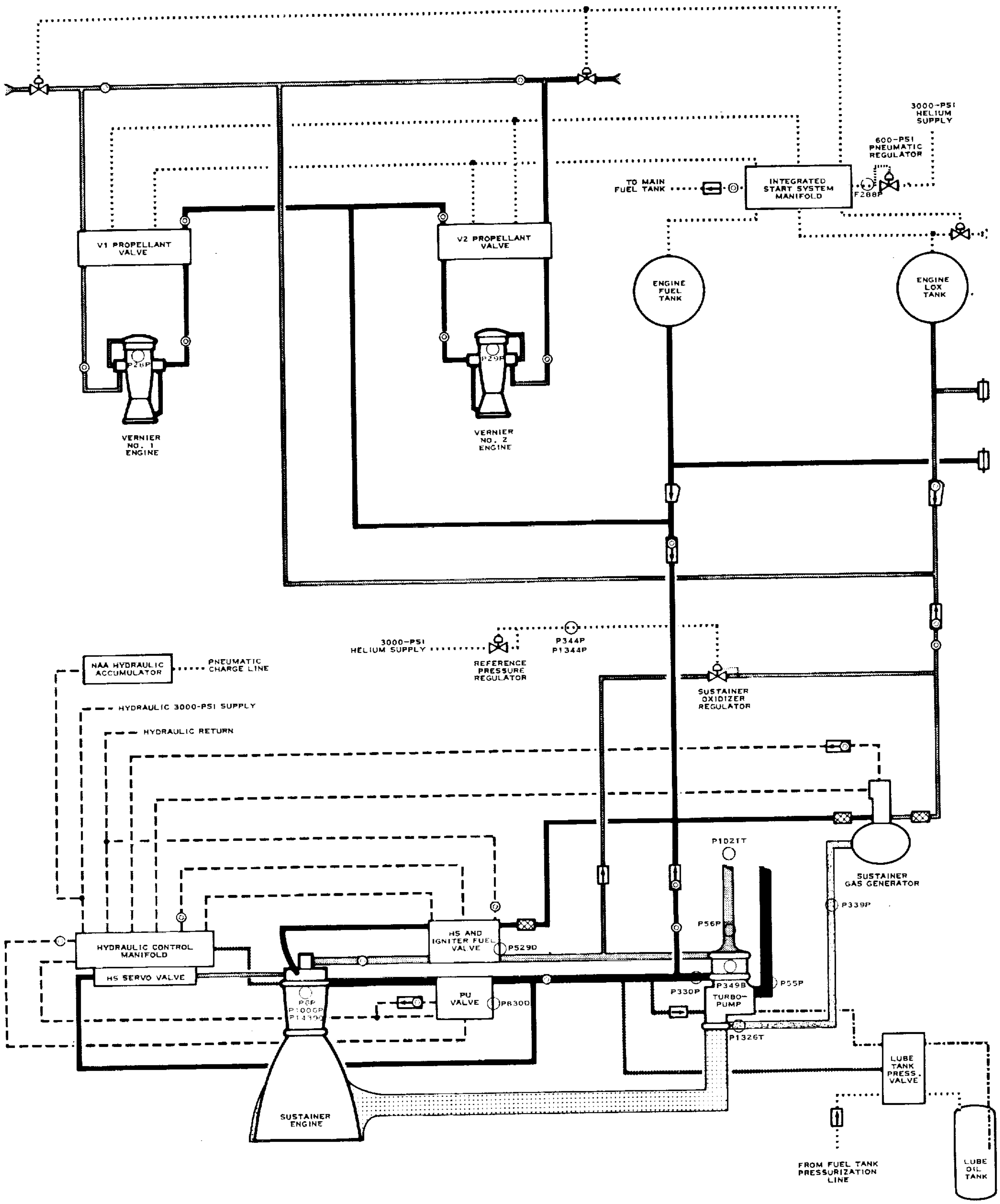
Operation of the telemetry system was satisfactory. One hundred three measurements were instrumented and all provided satisfactory data. Valid data signals were received beyond vehicle separation by the AMR telemeter station. However, at BECO a 20 db drop in telemeter signal strength was noted. Due to the AMR telemeter station high gain antenna system, no data was lost during Atlas powered flight. No other telemeter signal strengths, concerning this mission, were effected. The drop in telemeter RF signal strength may have been the result of degraded operation or total failure of the telemeter RF amplifier. The reliability of this unit has been excellent to date.

FIRE SPACE VEHICLE

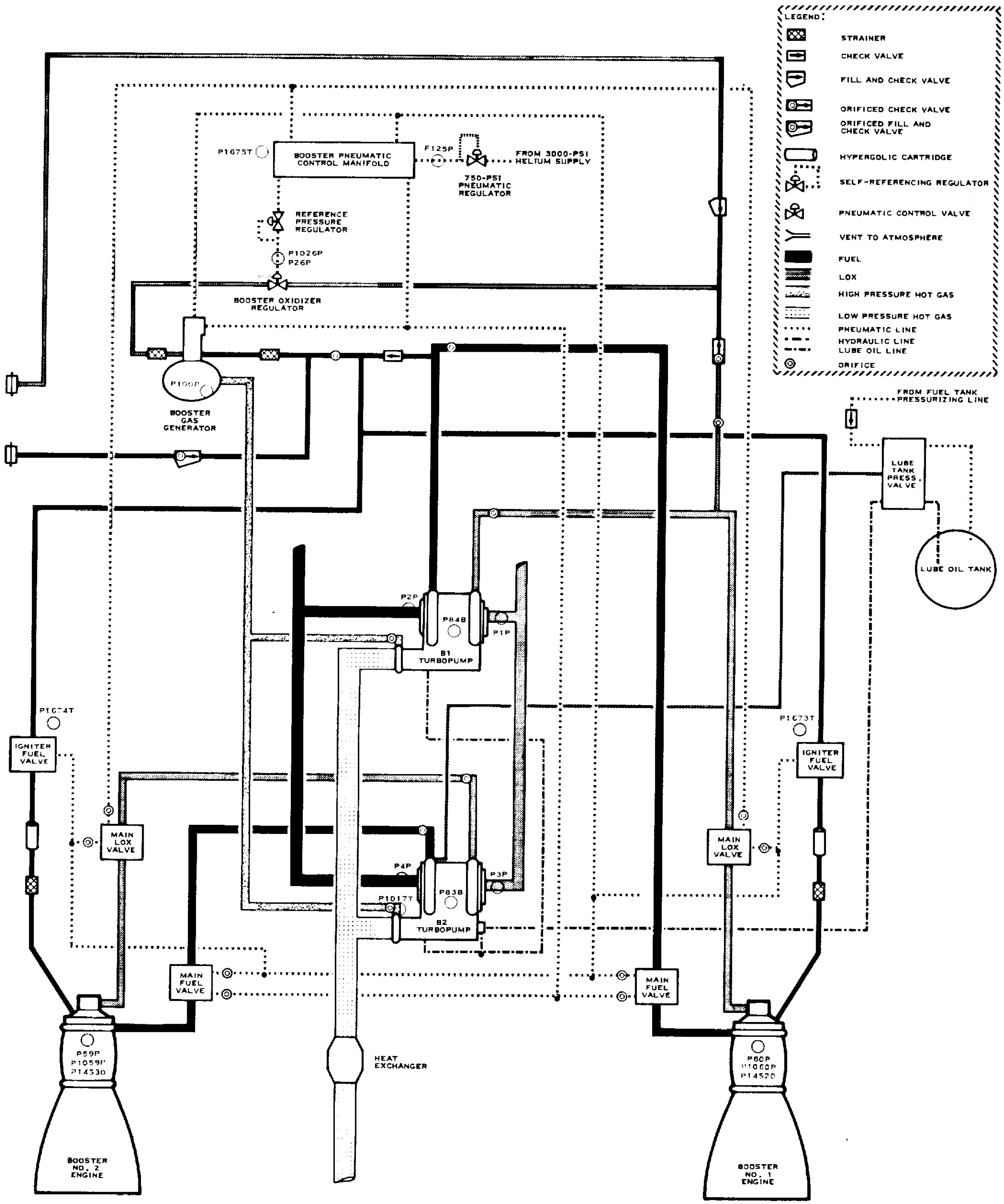


LAUNCH VEHICLE PERFORMANCE
FIGURE NO. 8-4-12
INTEGRATED REPORT NO. GDA/BKF64-018
SUMMARY

FIRE L/V SUSTAINER/VERNIER PROPULSION SYSTEM

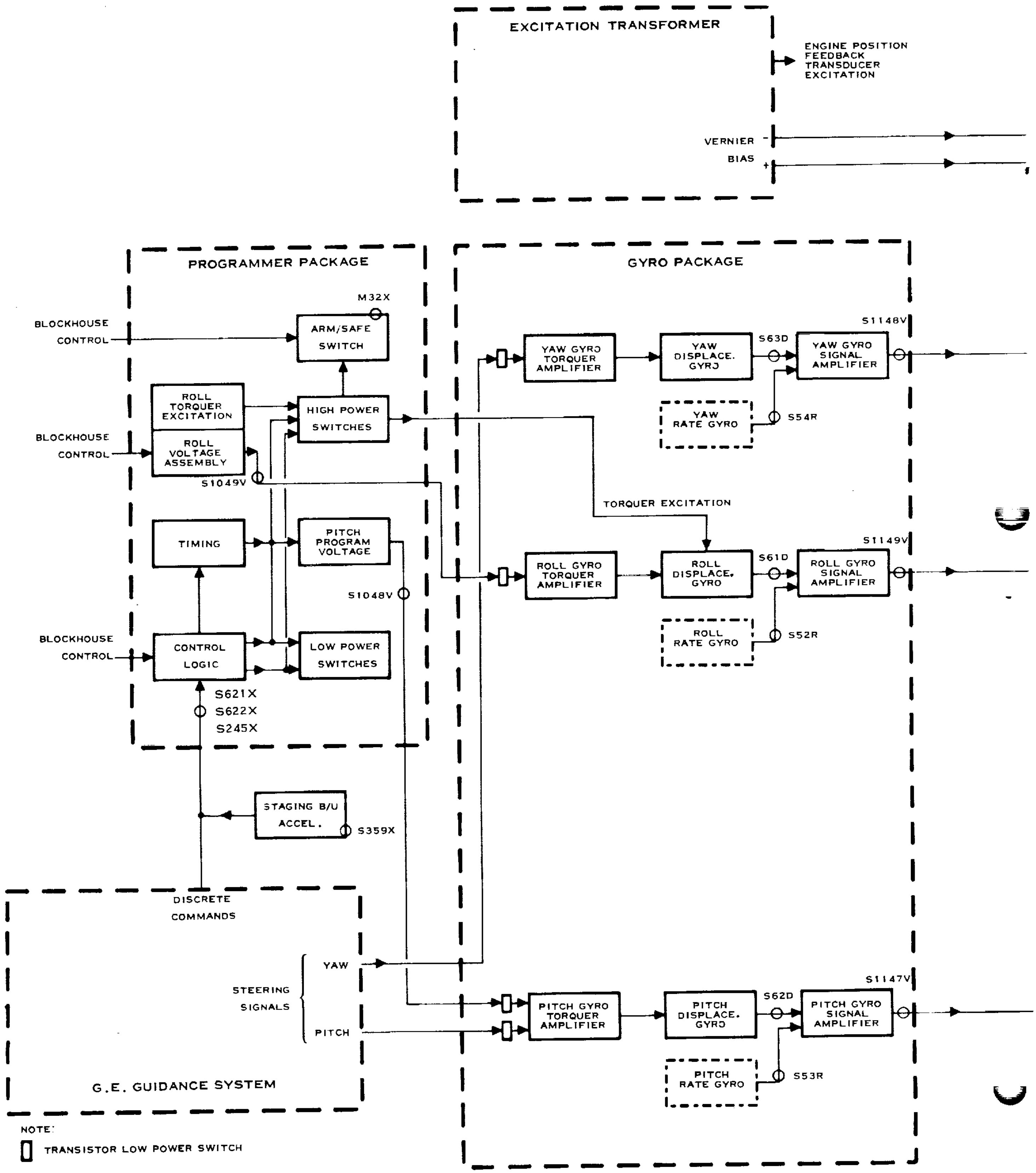


FIRE L/V BOOSTER PROPULSION SYSTEM

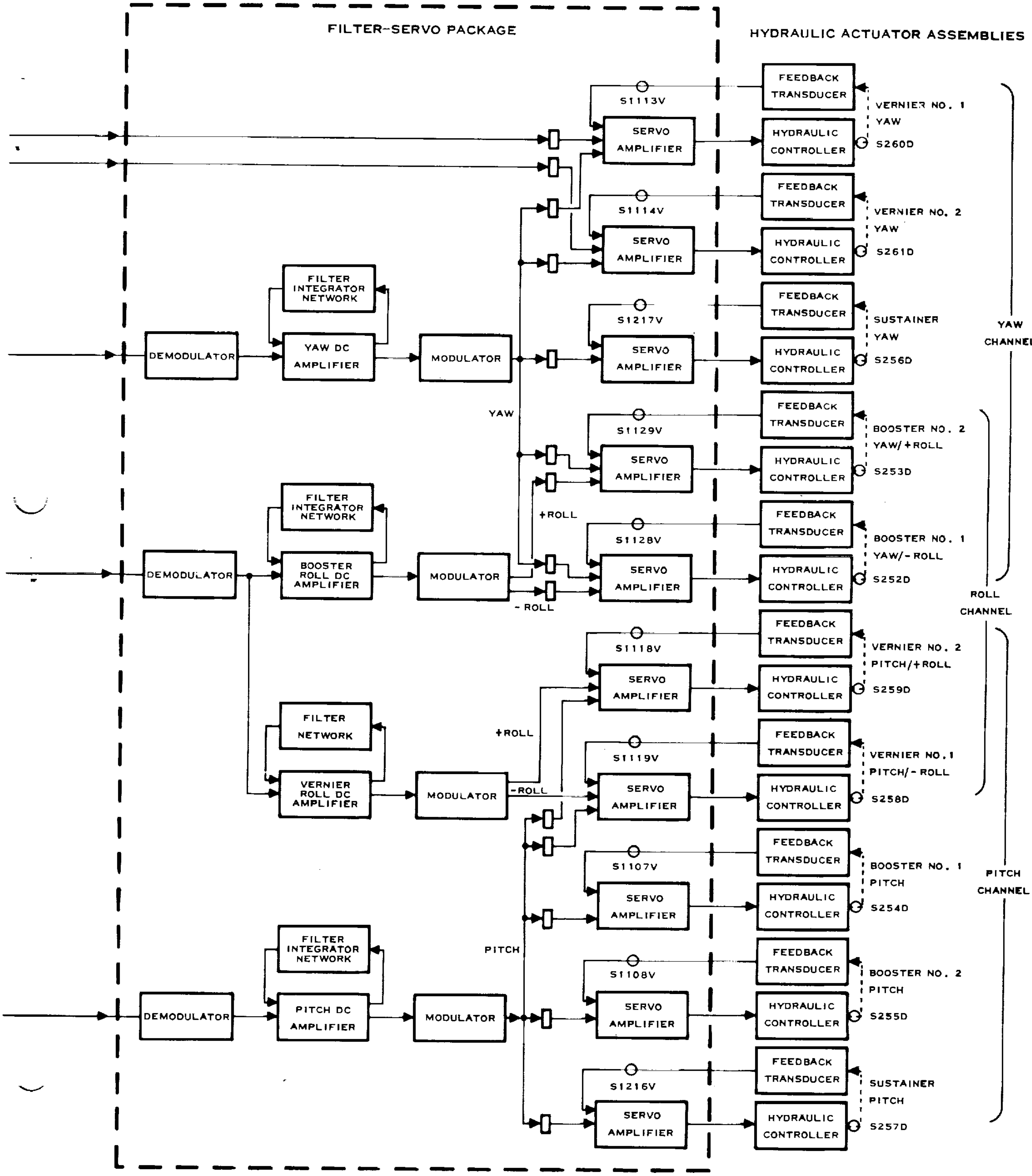


LAUNCH VEHICLE PERFORMANCE
FIGURE NO. 8-4-14
INTEGRATED REPORT NO. GDA/BKF64-018
SUMMARY

FIRE L/V FLIGHT CONTROL SYSTEM

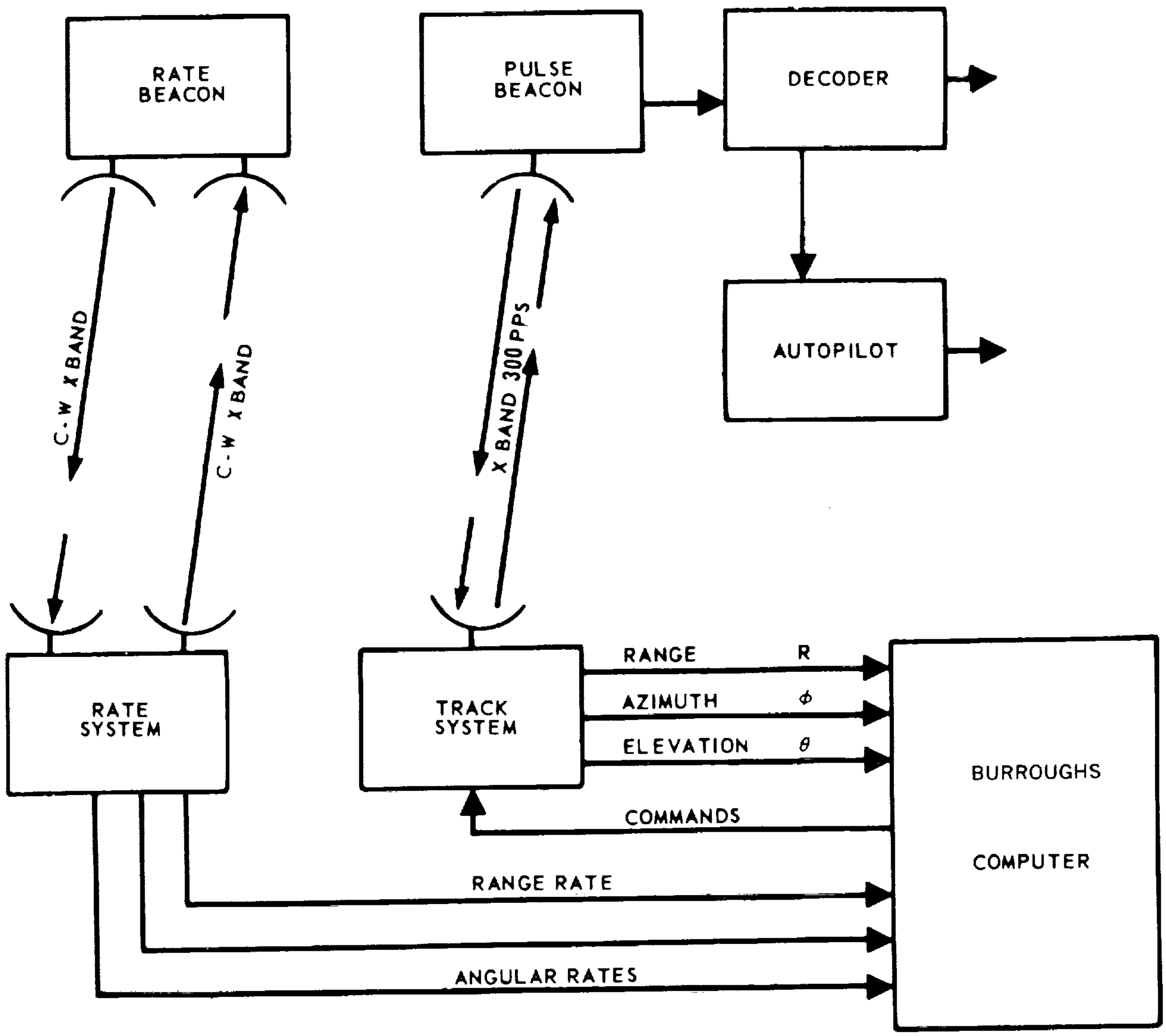


FIRE L/V FLIGHT CONTROL SYSTEM

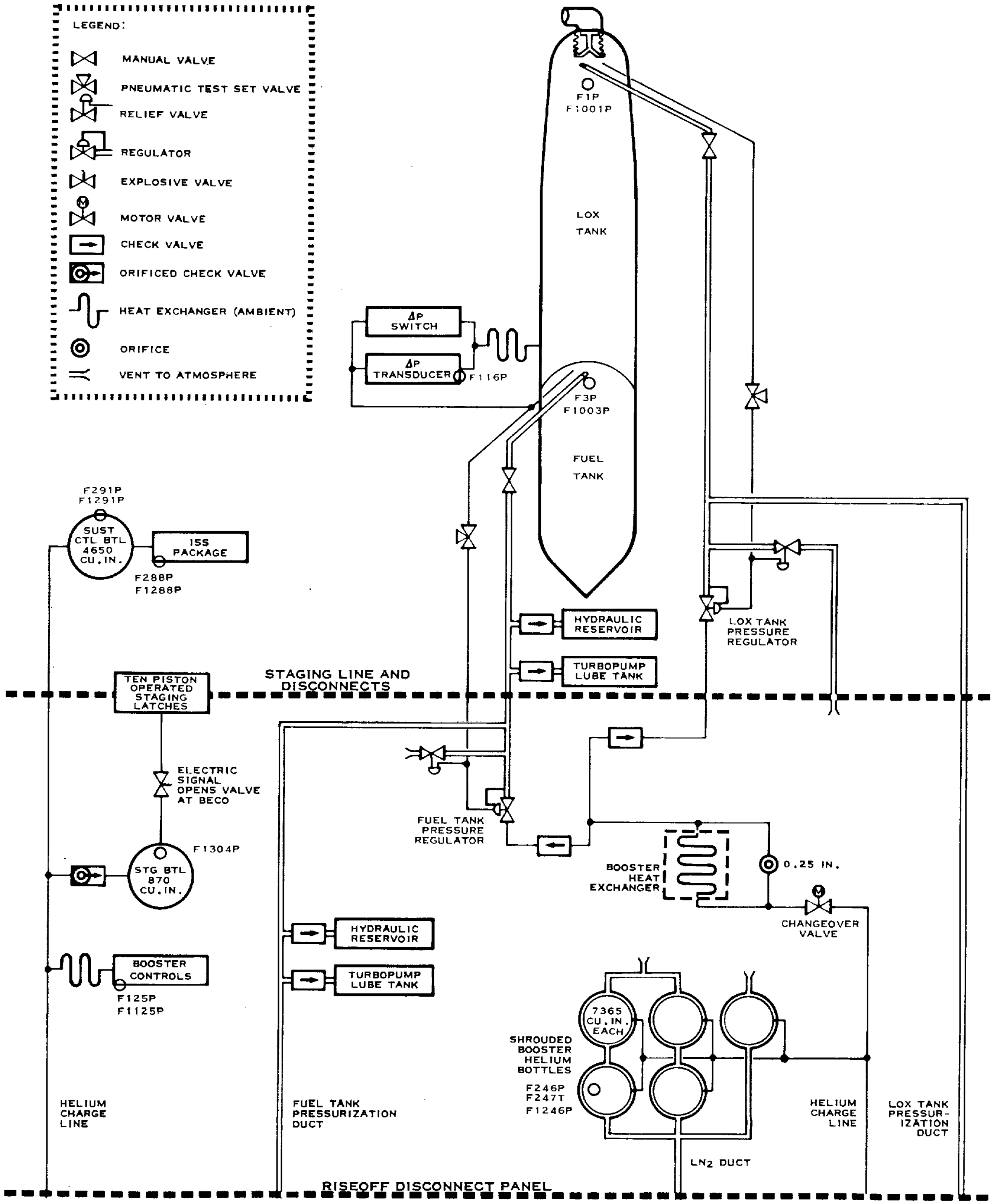


LAUNCH VEHICLE PERFORMANCE
FIGURE NO. 8-4-16
INTEGRATED REPORT NO. GDA/BKF64-018
SUMMARY

MOD III GUIDANCE SYSTEM

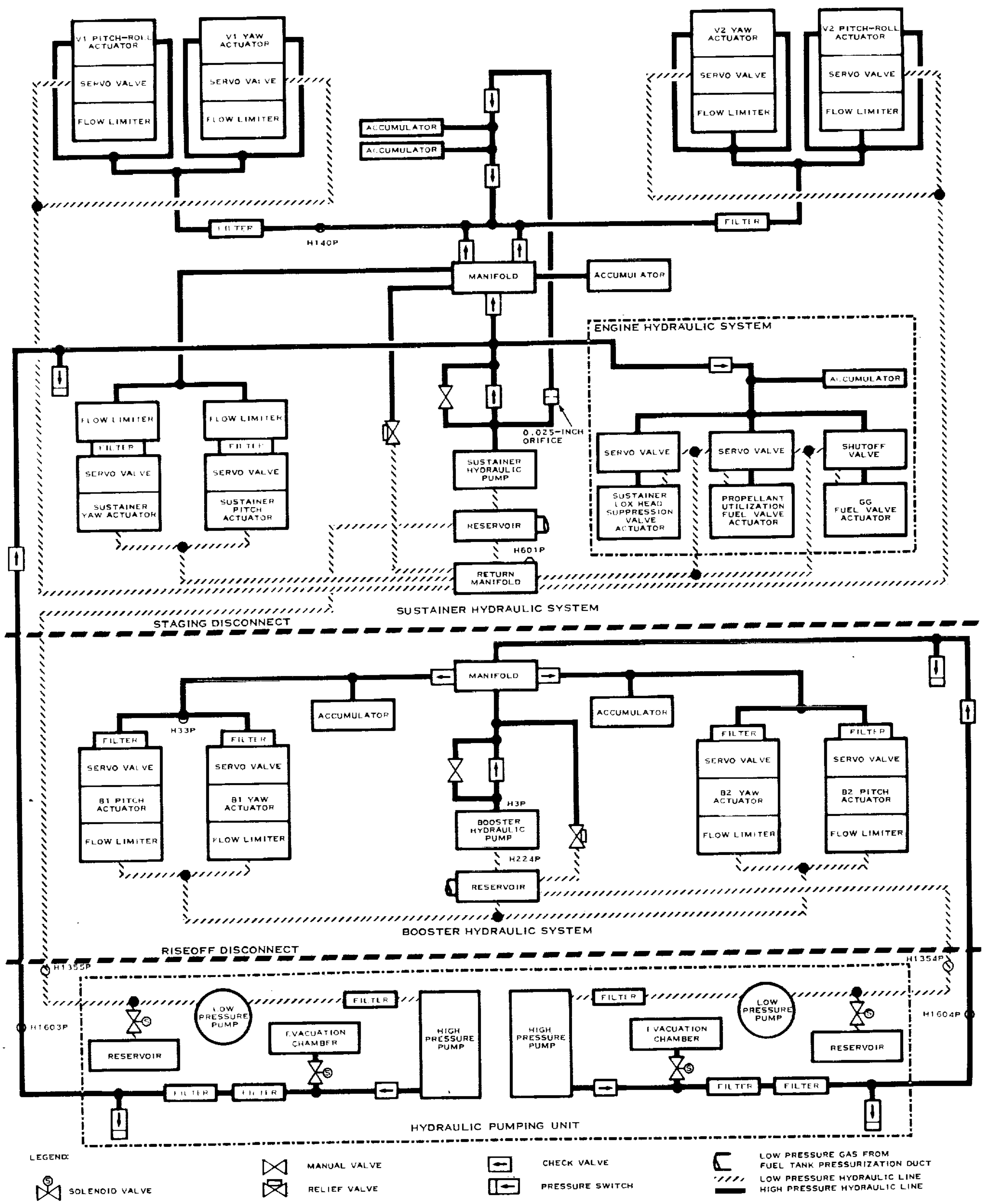


FIRE L/V PNEUMATIC SYSTEM

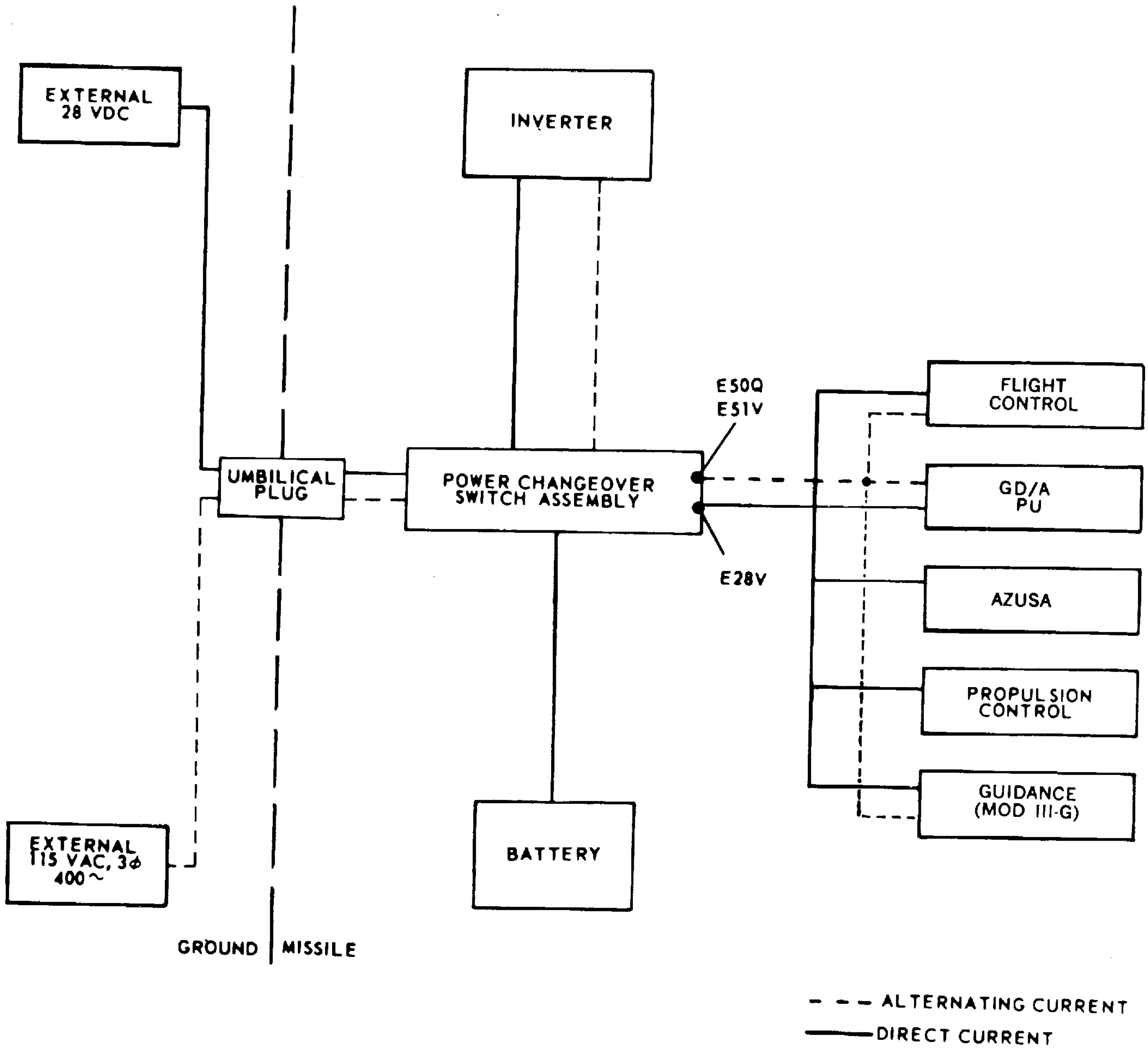


LAUNCH VEHICLE PERFORMANCE
FIGURE NO. 8-4-18
INTEGRATED REPORT NO. GDA/BKF64-018
SUMMARY

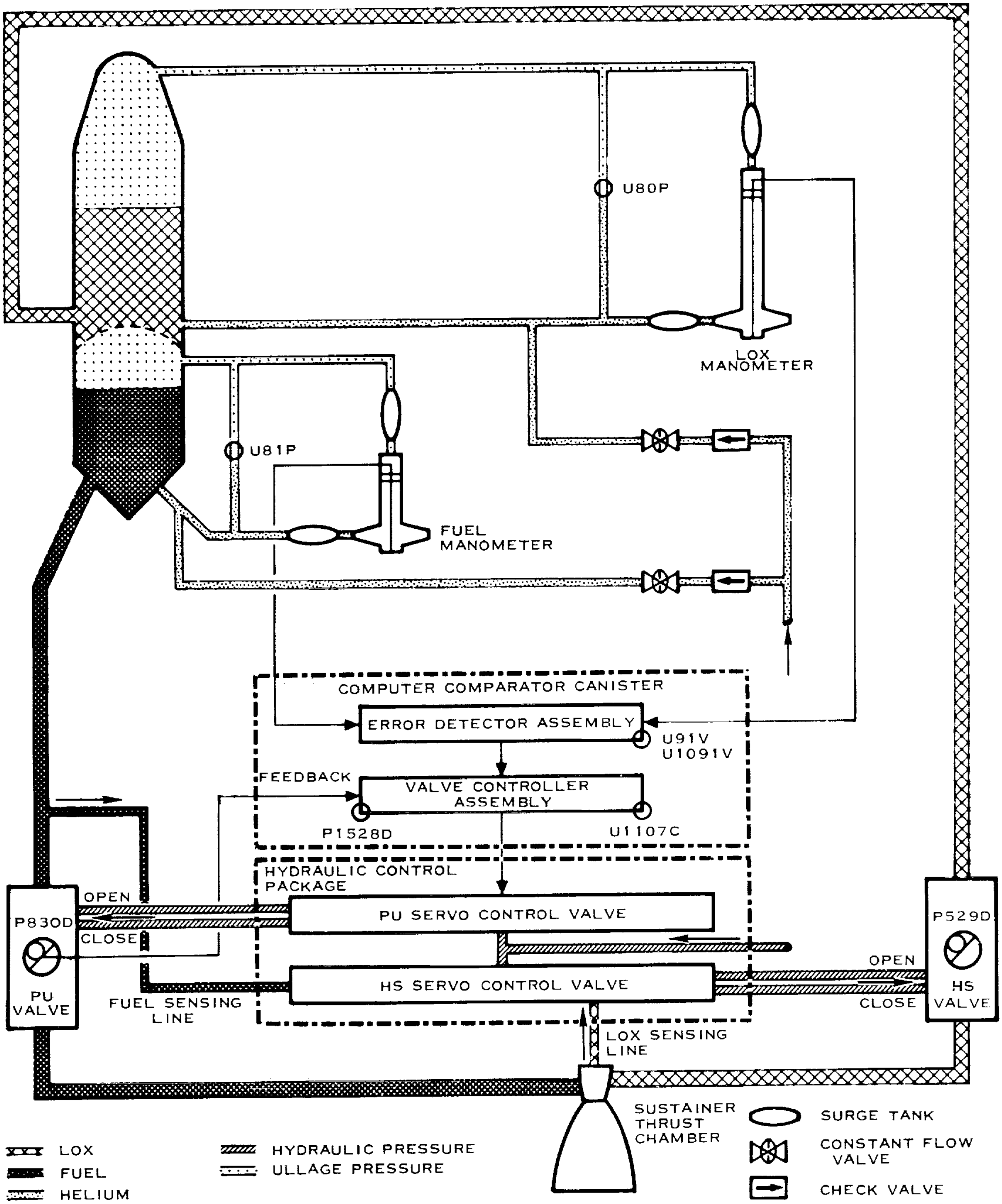
FIRE L/V HYDRAULIC SYSTEM



FIRE L/V ELECTRICAL SYSTEM

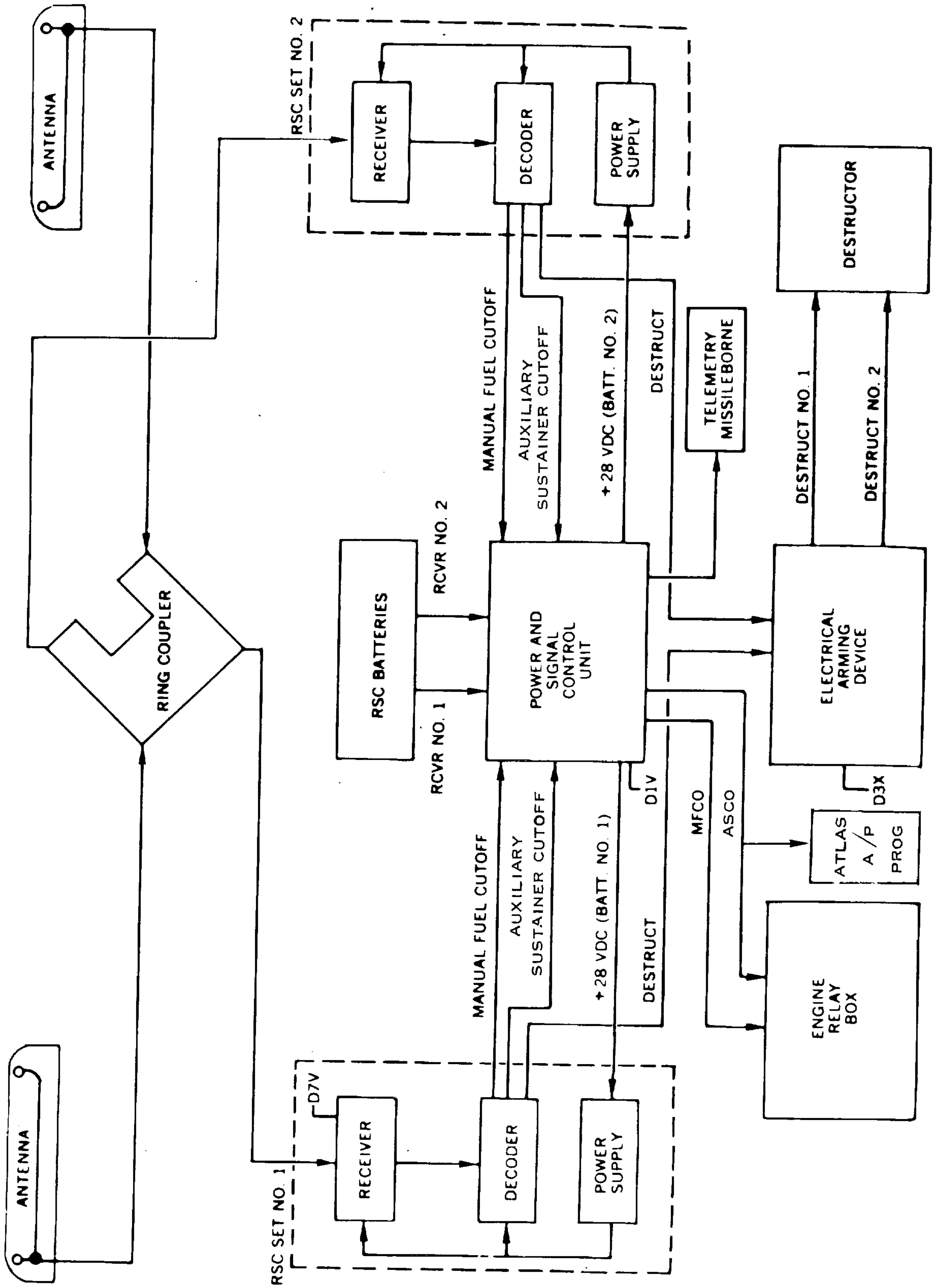


PROPELLANT UTILIZATION SYSTEM

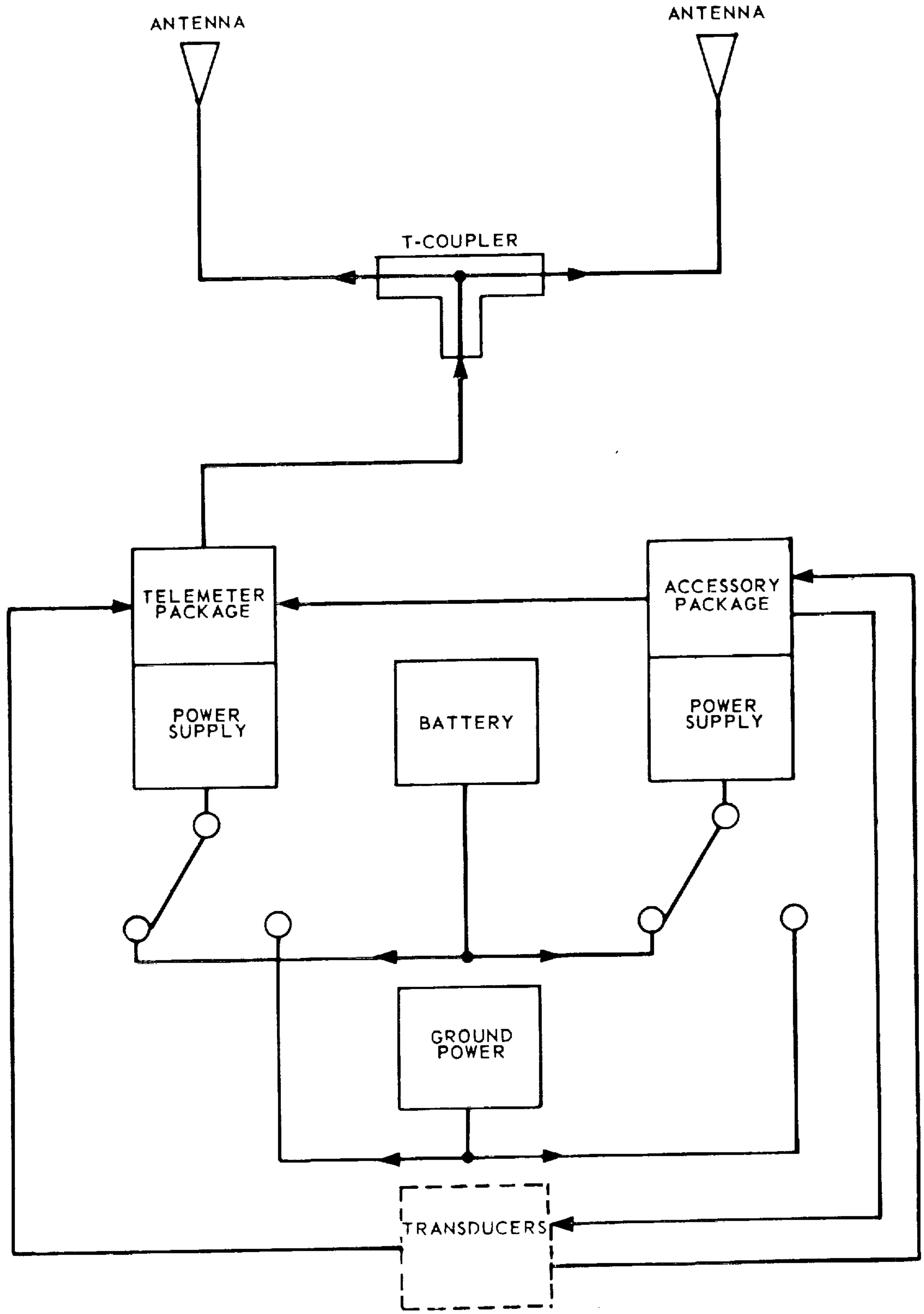


RANGE SAFETY COMMAND SYSTEM

RANGE SAFETY COMMAND SYSTEM



TELEMETRY SYSTEM



THIRD LAUNCH ATTEMPT

Range countdown P2-403-00-263 was initiated at 1120 hours EST on 13 April with a permissible launch window from 1530 to 2300 hours EST. The actual countdown duration was 505 minutes until abort, which occurred at T-40 minutes. The additional time was expended as follows:

1. At T-110 minutes, a 178-minute hold was observed due to adverse downrange weather. Conditions improved during the hold and the time count was resumed at 1628 hours EST.

2. At T-40 minutes a 10-minute hold was required to complete reinstallation of the sustainer radiation boot, removed to troubleshoot a faulty transducer. During the hold, problems developed in the re-entry package radiometer system. The countdown was aborted after a total hold of 127 minutes when it became apparent the downrange weather was rapidly degrading and the R/P radiometer problem could not be resolved prior to exceeding the launch window.

Major countdown events versus time are presented in Table 9-1-3.

FOURTH LAUNCH ATTEMPT

Range countdown P2-404-00-263 was initiated at 1122 hours EST on 14 April 1964 with an available launch window from 1622 to 2337 hours EST. The actual countdown duration was 320 minutes with vehicle liftoff occurring at 1642:25.520 hours EST as recorded by the blockhouse oscillograph recorder. The additional countdown time was expended as follows:

1. At T-165 minutes a 57-minute hold was required to replace a faulty GE airborne decoder.

2. At T-45 minutes a 2-minute hold was observed to verify downrange weather conditions.

3. At T-15 minutes a 5-minute hold was observed to re-verify downrange weather conditions.

4. At T-30 seconds an abort cutoff was received from the Launch Conductor when the track radar failed to regain automatic monopulse lock after switching from conical hold. The problem was isolated to a random occurrence when switching from conical to monopulse. The countdown was recycled to T-5 minutes after an 11-minute hold.

Major countdown events versus time are presented in Table 9-1-4.

TABLE 9-1-1. MAJOR COUNTDOWN TIME VS. EVENTS COUNTDOWN P2-401-00-263, 6 APRIL 1964

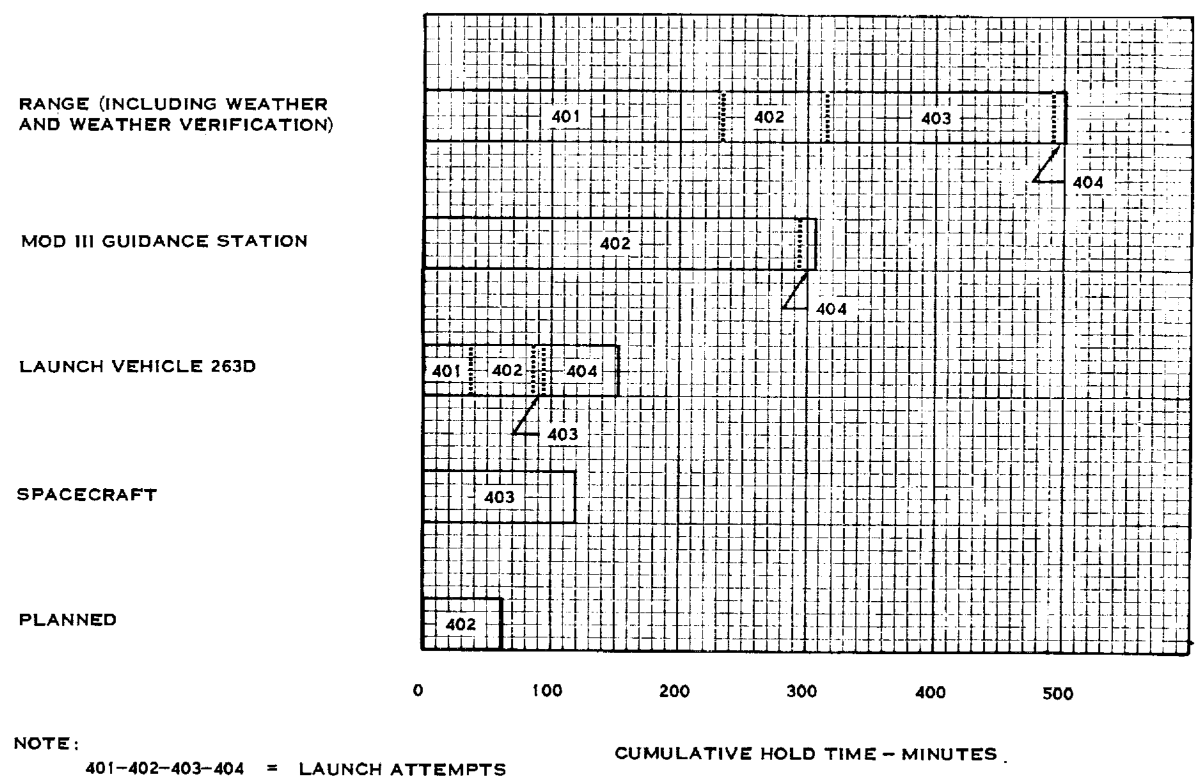
EST	T-TIME	EVENT
1033	T-240M	Start countdown.
1113	T-200M-Hold	Hold for problem with range safety command receiver. Receiver replaced and countdown recycled to T-230 minutes.
1149	T-230M	Countdown resumed.
1202	T-217M	Range safety command checks complete.
1249	T-170M	Autopilot checks complete.
1310	T-149M	Guidance command test #1 complete.
1321	T-138M	Telemetry batteries activated.
1329	T-130M	Battery checks complete.
1359	T-100M	Velocity and re-entry package squib checks complete.
1427	T-72M	Tower removal complete. Guidance command test No. 2 complete.
1454	T-45M-Hold	Hold for evaluation of downrange weather.
1850	T-45M-Abort	Countdown aborted due to continuing adverse downrange weather.

SECTION 1

SPACE VEHICLE COUNTDOWN

The 263D range countdown was scheduled for 240 minutes with an additional 60 minutes of planned hold at T-95 minutes. Countdown start was planned to occur 300 minutes prior to opening of the launch window (EST) to permit maximum utilization of the permissible launch duration. Time and duration of the launch window was dependent on altitude twilight conditions over Ascension Island.

Four countdowns were conducted in support of the scheduled launch. Thirteen holds were observed during the various launch attempts, with a total hold time of 1135 minutes. The cumulative hold time accrued by major area is presented below, followed by a detailed discussion of each range countdown.



FIRST LAUNCH ATTEMPT

Range countdown P2-401-00-263 was initiated at 1033 hours EST on 6 April 1964 with a window extending from 1533 to 2013 hours EST. The actual countdown duration was 497 minutes until abort, which occurred at T-45 minutes. The additional time was required as follows:

1. A 36-minute hold was observed at T-200 minutes to investigate a problem with the vehicleborne range safety receiver No. 2. The receiver was replaced and the countdown recycled to T-230 minutes.

2. The countdown was held at T-45 minutes due to deteriorating downrange weather. The launch attempt was aborted by the mission director after 236 minutes of hold. Abort occurred at 1850 hours EST. All systems were in a "Go" status at test termination.

A listing of major countdown events versus time is presented in Table 9-1-1.

SECOND LAUNCH ATTEMPT

Range countdown P2-402-00-263 was initiated at 1031 hours EST on 10 April 1964 with an established launch window from 1531 to 2333 hours EST. The actual countdown duration was 761 minutes until abort, which occurred at T-10 minutes. The additional time was required as follows:

1. A 79-minute hold was observed at T-95 minutes to verify downrange instrumentation status and weather conditions.

2. At T-40 seconds the countdown was held due to a Redline callout on main vehicle battery voltage. A series of battery load tests were performed and it was determined that replacement was not required. The countdown was recycled to T-9 minutes and resumed after a hold duration of 38 minutes.

3. At T-4 seconds an abort cutoff was received from the MOD III guidance blockhouse monitor due to receipt of guidance discrete 10 (velocity package ignition interlock) when no discrete was expected. Investigation revealed the condition was repeatable, but could be prevented by changing the track acquisition procedure. The revised procedure was tested and found satisfactory. During the investigation, lox was drained and the main vehicle and telemetry batteries replaced. The countdown was recycled to T-40 minutes and resumed after a hold duration of 294 minutes.

4. At T-10 minutes a hold was called to investigate a discrepancy in the commutated waveform from RF No. 1 Channel E. The countdown was aborted after 12 minutes due to adverse downrange weather and failure to readily solve the telemetry problem.

Major countdown events versus time are presented in Table 9-1-2.

PART 9

PREFLIGHT EVENTS

GENERAL DYNAMICS/ASTRONAUTICS
INTEGRATED REPORT NO. GDA/BKF64-018

APPROVED BY: ______________________
L. E. MUNSON
ASSISTANT PROGRAM DIRECTOR
FIRE PROGRAM OFFICE

TABLE 9-1-2. MAJOR COUNTDOWN TIME VS. EVENTS.COUNTDOWN P2-402-00-263, 10 APRIL 1964

EST	T-TIME	EVENT
1031	T-240M	Start range countdown.
1055	T-216M	Range safety command checks complete.
1141	T-170M	Autopilot checks complete.
1151	T-160M	Guidance command test No. 1 complete.
1256	T-95M-Hold	Battery activation complete. Start 60-minute scheduled hold. Tower removal in progress.
1356	T-95M	Countdown resumed.
1416	T-75M	Tower and test stand secure.
1446	T-45M-Hold	Hold to verify downrange weather and instrumentation status.
1605	T-45M	Countdown resumed.
1625	T-25M	Lox tanking in progress.
1632	T-18M	Range safety command tests complete.
1648	T-2M	Start flight pressurization.
1649	T-1M30S	Internal vehicle power.
1649	T-40S-Hold	Hold for Redline on main vehicle battery power. Recycle to T-9 minutes.
1727	T-9M	Resume countdown.
1734	T-2M	Start flight pressurization. Internal vehicle power.
1736	T-4S-Hold	Abort cutoff from MOD III guidance blockhouse monitor. Countdown recycled to T-40 minutes. Lox detanked, batteries replaced.
2100	T-40M-Hold	Conducting load tests on new vehicle batteries. Continuing analysis of guidance problem.
2145	T-40M-Hold	Guidance command tests complete. Tower removal complete.
2230	T-40M	Countdown resumed.
2245	T-25M	Start lox tanking.
2300	T-10M-Hold	Holding due to problems with Atlas telemetry.
2312	T-10M-Abort	Countdown aborted due to telemetry problems and continued degradation of downrange weather.

TABLE 9-1-3. MAJOR COUNTDOWN EVENTS VS. TIME.COUNTDOWN P2-403-00-263, 13 APRIL 1964

EST	T-TIME	EVENT
1120	T-240M	Start range countdown.
1142	T-218M	Range safety command test complete.
1230	T-170M	Autopilot checks complete.
1310	T-130M	Guidance command test No. 1 complete. Azusa check complete.
1330	T-110M-Hold	Hold for evaluation of downrange weather conditions.
1628	T-110M	Countdown resumed.
1733	T-45M	Guidance command test No. 2 complete. Tower removal complete.
1738	T-40M-Hold	Hold to complete reinstallation of sustainer engine boot. Hold extended due to spacecraft instrumentation problems.
1945	T-40-Abort	Countdown aborted due to excessive time required to complete spacecraft repairs and degrading downrange weather.

TABLE 9-1-4. MAJOR COUNTDOWN EVENTS VS. TIME, LAUNCH COUNTDOWN P2-404-00-263, 14 APRIL 1964

EST	T-TIME	EVENT
1122	T-240M	Start range countdown.
1144	T-218M	Range safety command test complete.
1159	T-203M	Retrorocket electrical installation complete.
1233	T-169M	Autopilot checks complete.
1237	T-165M-Hold	Hold for replacement of defective GE airborne decoder.
1334	T-165M	Countdown resumed.
1353	T-146M	Guidance command test No. 1 complete.
1359	T-140M	Main battery activated.
1424	T-115M	Battery load test complete.
1440	T-99M	V/P squib check complete.
1444	T-95M	Telemetry batteries activated.
1525	T-54M	Tower removal complete. Guidance command test No. 2 complete. Start helium load.
1534	T-45M-Hold	Hold to evaluate downrange weather.
1536	T-45M	Countdown resumed.
1541	T-40M	Start lox tanking.
1559	T-22M	Start RSC final checks.
1606	T-15M-Hold	Hold to evaluate downrange weather.
1611	T-15M	Countdown resumed.
1618	T-8M	RSC checks complete.
1623	T-3M30S	Telemetry internal.
1624	T-1M45S	Internal power.
1625	T-30S-Hold	Hold due to absence of Guidance Ready light from MOD III guidance ground station. Recycle to T-5 minutes.
1637	T-5M	Countdown resumed.
1638	T-3M30S	Telemetry internal.
1640	T-1M45S	Internal power.
1641	T-25S	Oil evacuation complete.
1642	T-3S	Ignition.
1642:25.536*		Two-inch motion.

NOTE: * As recorded by range telemetry.

SECTION 2

L/V PREFLIGHT ACTIVITIES

The purpose of this section is to summarize Atlas Launch Vehicle 263D preflight activities, including factory checkout, Hangar-J activities, and AMR Complex 12 checkout.

A satisfactory factory acceptance test was conducted on 13 August 1963. Atlas Launch Vehicle 263D was accepted by USAF and arrived at Cape Kennedy on 28 August 1963. Following receiving inspection Vehicle 263D was placed in the south bay of Hangar-J for prelaunch checkout. The first erection of Vehicle 263D was accomplished on 30 August 1963 at AMR Complex 12. Following Complex 12 preflight checkout Atlas 263D was removed and placed back in Hangar-J for storage. Vehicle 263D was again erected on 3 March 1964 at AMR Complex 12, where preflight checkout was again performed. A successful Simulated Launch was conducted on 3 April 1964. Following 3 unsuccessful launch attempts, Vehicle 263D was successfully launched on 14 April 1964.

SUMMARY OF FACTORY TESTING

Vehicle 263D was moved into vehicle checkout dock No. 12 on 20 June 1963. Systems level testing was performed from 21 June through 8 August 1963. One composite test was performed on 10 August with a partial retest satisfactorily performed on 13 August 1963. The following irregularities were observed on the test of 10 August 1963.

Test FC-CO-01-0020-001 - Guidance

Voltage proportional to pulse beacon AGC at -50 dbm interrogation level was 1.26 volts when 1.7 to 3.1 VDC was expected. Investigation revealed a component in the pulse beacon AGC telemetry conditioning circuitry was temperature sensitive. Since the guidance canisters are for ground tests only and the voltage proportional to AGC test is for telemetry comparison only, all composite test conditions were considered satisfied.

Range Safety Command

The range safety command tone channel delay times for the first and second manual and automatic fuel cutoff on Receivers No. 1 and No. 2 did not meet specifications. Investigation revealed that the oscillators in the modulation generator panel indicated slow response times. The panel was replaced and post composite testing indicated satisfactory pickup times.

AMR VEHICLE CHECKOUT

Vehicle 263D arrived at Cape Kennedy on 28 August 1963. After vehicle receiving inspection, the vehicle was placed in Hangar-J for checkout. Erection at AMR Complex 12 was performed on 30 August 1963, and 263D was prepared for simulated launch. On 4 October 1963 Vehicle 263D was removed from Complex 12 and stored in Hangar-J. Re-erection was accomplished on 3 March 1964 at AMR Complex 12 and Vehicle 263D was again made ready for launch. Table 9-2-1 lists significant Launch Vehicle 263D AMR Milestones.

Test P2-4CO-01-263 (Flight Acceptance Composite Test - FACT)

Booster FACT was run on 16 September 1963. The following discrepancies were noted:

1. Loss of yaw control. Autopilot U2 Package (Filter-Servoamplifier) was replaced.
2. GE rate beacon was replaced because the voltage proportional to RF power was below limits.
3. Umbilical P1002 did not eject, apparently due to excessive bending.
4. Pressurization Procedure omitted pressurization of propellant depletion switch; SECO occurred at Programmer SECO enable.

5. Autopilot 42-inch umbilical (P609) was not pulled until approximately 6.5 seconds. There was no engine roll response until then.

6. No Velocity Package separation backup from programmer. Programmer (U3) to be changed.

7. Booster No. 1 pump speed (P84B) was ten percent of information Bandwidth (IBW) when it should have been zero percent. Problem due to noise caused by improper shielding.

Test P2-4CO-02-263 (FACT)

Joint FACT was rerun on 27 September 1963. The following irregularities were observed:

1. Vernier No. 1 yaw (S260D) indicated loss of V1 engine position from 277.5 to 427.7 seconds.

2. One-sixteenth amp. fuse in Portable Destruct Test Set (PDTS) was found open after countdown.

3. E-A Pen 51 (Main Engines Complete) did not activate until 0.02 second after Pen 52 (Pre-release Cutoff Disarm). (Pen 51 should activate at, or just prior to, Pen 52.) Pen 56 (Main Engines Thrust Complete) activated in proper sequence 0.03 second prior to Pen 52.

Test P2-4CO-03-263 (FACT)

Booster FACT was rerun on 13 March 1964. The following discrepancies were noted:

1. The Azusa Transponder was changed due to NO-GO from the Range. Problem due to phase noise.

2. Booster No. 1 Fuel Pump Inlet (P2P) and Booster No. 2 Fuel Pump Inlet (P4P) indicated a helium leakage past the booster fuel pre-valve. The pre-valve was replaced.

3. RF No. 1, Channel 14 frequency decreased for short period (2-3 seconds) during test.

Test P2-4CO-04-263 (FACT)

Booster FACT was rerun on 24 March 1964. No discrepancies were noted.

Test P2-4CO-05-263 (FACT)

Joint FACT was rerun on 31 March 1964. The following irregularities were observed:

1. RF No. 1, Channel E sub carrier oscillator was drifting. The RF No. 1 canister was replaced.

2. Azusa NO-GO due to lack of frequency lock. The transponder was replaced.

3. RSC No. 1 fuse did not blow on first count (3-second signal). A second plus count (5-second signal) blew the Range Safety Command No. 1 fuse.

4. Ignition Interlock Backup (S293X) saw a signal on Arm side of Switch 18 at 5.7 seconds before Booster Engine Cutoff (BECO) during the Guidance Command Test (GCT). On a second Guidance Command Test the signal was seen at 28 seconds after BECO. Caused by Ling-Temco-Vought (LTV) signal from the Velocity Package (V/P) to the Arm side of the Arm Safe switch while in test mode (a normal occurrence which was not foreseen).

5. Rate Beacon RF output (G82E), raised 10 percent during Guidance Control (GCT) Test No. 1, and 5 percent during Guidance Control Test No. 2.

Test P2-4BN-01-263 (Dual Propellant Loading - DPL)

Dual propellant tanking test was performed on 18 September 1963. Fuel was tanked to a level 15 gallons above the 100 percent Propellant Loading Control Unit (PLCU) probe. Successful fuel tanking was accomplished with no leaks being detected.

Lox was tanked to 100 percent and sub-cooled topping was utilized for approximately seven minutes prior to flight pressurization. Lox tanking was secured with 174,150 pounds of lox aboard the vehicle. Successful lox tanking was accomplished.

Test P2-4BN-02-263 (DPL)

Dual propellant tanking test was run on 30 September 1963. Fuel was tanked to a level of 14 gallons above the 100 percent probe. Before lox tanking on 30 September 1963 the fuel overfill probe activated due to fuel expansion. Approximately 36 gallons of fuel was drained in order to place the fuel level below the 100 percent probe. Fuel tanking was satisfactorily secured and fuel remained on board until the Simulated Launch Test on 2 October 1963.

Lox was tanked to the 100 percent probe and sub-cooled topping was utilized for approximately 20 minutes prior to flight pressurization. One anomaly occurred during sub-cooled lox topping sequence. A Redline temperature on landline Measurement P1021T, Lox Temperature at Breakaway Valve was indicated at the start of sub-cooled lox topping. This Redline condition existed for approximately nine minutes while sub-cooled lox was being flowed through the 2-inch line. It is believed that this problem was caused by insufficient chilldown of the 2-inch line during fast flow lox tanking. The DPL was satisfactorily secured.

Test P2-4BN-03-263 (DPL)

Dual propellant tanking test was conducted on 18 and 19 March 1963. Fuel was tanked on 18 March 1963, utilizing the Propellant Loading Control Unit (PLCU) as the primary system and the totalizer and loadcells as the secondary system. Fuel tanking was accomplished satisfactorily and secured twelve gallons above the 100 percent PLCU probe. No leaks were detected in the vehicle fuel system.

Lox was tanked on 19 March 1963, utilizing the Propellant Utilization Control Unit (PLCU) as the primary system and the propellant utilization system as the secondary system. Lox tanking was satisfactorily secured with no problems being encountered.

Test P2-4MO-01-263 (Simulated Launch)

Simulated Launch was conducted on 2 October 1963. A Simulated Launch Test was performed on Vehicle 263D on 2 October 1963, at Complex 12, AMR. This vehicle was composed of Atlas Launch Vehicle 263D, a Prototype Velocity Package, and an AMR Model Re-entry Package. The test was conducted with the Re-entry Package (R/P), Velocity Package (V/P), and the launch vehicle in a flight-ready condition with the exception that no ordnance or batteries were aboard the launch vehicle. All test objectives were met satisfactorily. One significant problem, occurring in the Azusa system, necessitated replacement of the Azusa transponder.

The Azusa system was declared NO-GO because of excessive noise on the coherent carrier, making it difficult for the system to acquire and maintain lock. Six transponders were tested before it was possible to satisfy the range Azusa station requirements.

Test P2-4MO-02-263

A second Simulated Launch Test (conducted on 3 April 1964) was performed with all systems on the Launch Vehicle, Velocity Package, and Re-entry Package actively participating. All test objectives were met satisfactorily. No hardware problems were encountered.

TABLE 9-2-1. ATLAS 263D AMR MILESTONES

Event	Scheduled	Accomplished	Reason for Delay
Arrival	-	28 August 1963	
Erection	3 September 1963	30 August 1963	
Booster-FACT	16 September 1963	16 September 1963	
Dual Propellant Tanking	-	18 September 1963	
J-FACT	26 September 1963	27 September 1963	Launch control sequence panel electrical short
RFI	25 September 1963	30 September 1963	Weather
Dual Propellant Tanking	-	30 September 1963	
Simulated Launch	1 October 1963	2 October 1963	Weather
Removal from Tower	-	4 October 1963	
Re-Erection	3 March 1964	3 March 1964	
Booster-FACT	19 March 1964	12 March 1964	
V/P Mate	17 March 1964	18 March 1964	Complex evacuation
Fuel Tanking	20 March 1964	18 March 1964	
Lox Tanking	-	19 March 1964	
Booster-FACT	-	24 March 1964	Confidence rerun
R/P Mate	27 March 1964	27 March 1964	
RFI	30 March 1964	30 March 1964	
J-FACT	31 March 1964	31 March 1964	
Simulated Launch	-	3 April 1964	
Launch	6 April 1964	Terminated	Downrange weather
Launch	10 April 1964	Terminated	Downrange weather and Atlas telemeter
Launch	13 April 1964	Terminated	Spacecraft discrepancy and downrange weather
Launch	14 April 1964	14 April 1964	

PART 10

APPENDIX

GENERAL DYNAMICS/ASTRONAUTICS
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SECTION 1

GLOSSARY

Å	-	Angstrom unit
ABL	-	Allegheny Ballistics Laboratory
AGC	-	Automatic Gain Control
AMR	-	Atlantic Missile Range
ASCO	-	Auxiliary Sustainer Cutoff
BECO	-	Booster Engine Cutoff (Atlas)
B-FACT	-	Booster Flight Acceptance Composite Test
BGG	-	Booster Gas Generator
B1	-	Booster Engine No. 1 (Atlas)
B2	-	Booster Engine No. 2 (Atlas)
CPS	-	Cycles Per Second
db	-	decibels
dbm	-	decibels referenced to one milliwatt
DC	-	Direct Current
deg/sec	-	Degrees per second
DPL	-	Dual Propellant Loading
ECN	-	Engineering Change Notice
EST	-	Eastern Standard Time
$^{\circ}F$	-	Degrees Fahrenheit
FM	-	Frequency Modulation
g	-	unit acceleration of 32 ft/sec^2
GD/A	-	General Dynamics/Astronautics
GE	-	General Electric
HSU	-	Hydraulic Supply Unit
IBM	-	International Business Machine
IBW	-	Information Band Width
ICBM	-	Inter-Continental Ballistic Missile
IRIG	-	Inter-Range Instrumentation Group
ISS	-	Integrated Start System
I_{xx}	-	Moment of inertia about the X-X axis
I_{yy}	-	Moment of inertia about the Y-Y axis
I_{zz}	-	Moment of inertia about the Z-Z axis

J-FACT	-	Joint Flight Acceptance Composite Test
KC	-	Kilocycle (10^3)
KSC	-	Kennedy Space Center
Liftoff	-	Vehicle two-inch motion
Lox	-	Liquid oxygen
LTV/A	-	Ling-Temco-Vought/Astronautics
L/V	-	Launch Vehicle (Atlas)
M	-	Minutes
Max	-	Maximum
MFCO	-	Manual Fuel Cutoff
MOD	-	Model
N_2	-	Gaseous nitrogen
PAM	-	Pulse Amplitude Modulation
P C/O	-	Power Changeover
PDM	-	Pulse Duration Modulation
p-p	-	peak-to-peak
psi	-	pounds per square inch
psia	-	pounds per square inch absolute
psig	-	pounds per square inch gage
PU	-	Propellant Utilization
Pwr	-	Power
RAC	-	Republic Aviation Corporation
rf	-	Radio frequency
R-F	-	Radio frequency
RFI	-	Radio Frequency Interference
R/P	-	Reentry Package
rpm	-	revolutions per minute
R/S	-	Reentry Stage
RSC	-	Range Safety Command
S	-	Sustainer engine (Atlas) or second
SECO	-	Sustainer Engine Cutoff (Atlas)
SLV	-	Space Launch Vehicle (Atlas)
TLM	-	Telemeter
T/M	-	Telemeter
USAF	-	United States Air Force

VAC	-	Volts Alternating Current
VCO	-	Voltage Controlled Oscillator
VDC	-	Volts Direct Current
VECO	-	Vernier Engine Cutoff (Atlas)
VHF	-	Very High Frequency
V/P	-	Velocity Package
V1	-	Vernier Engine No. 1 (Atlas)
V2	-	Vernier Engine No. 2 (Atlas)
X_{cg}	-	Center of gravity distance from X-axis
XMITTER	-	Transmitter
Y_{cg}	-	Center of gravity distance from Y-axis
Z_{cg}	-	Center of gravity distance from Z-axis
μV	-	Microvolt (10^{-6})