

REPORT NO.GDC | BKF65-042

PROJECT FIRE INTEGRATED POST FLIGHT EVALUATION REPORT

FLIGHT II

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Project FIRE

N66-15225

(ACCESSION NUMBER)

180

(PAGES)

(THRU)

1

(CODE)

31

(CATEGORY)

(NASA CR OR TMX OR AD NUMBER)

GPO PRICE \$ _____

CFSTI PRICE(S) \$ _____

Hard copy (HC) 5.00Microfiche (MF) 1.00

NASA
LANGLEY RESEARCH CENTER

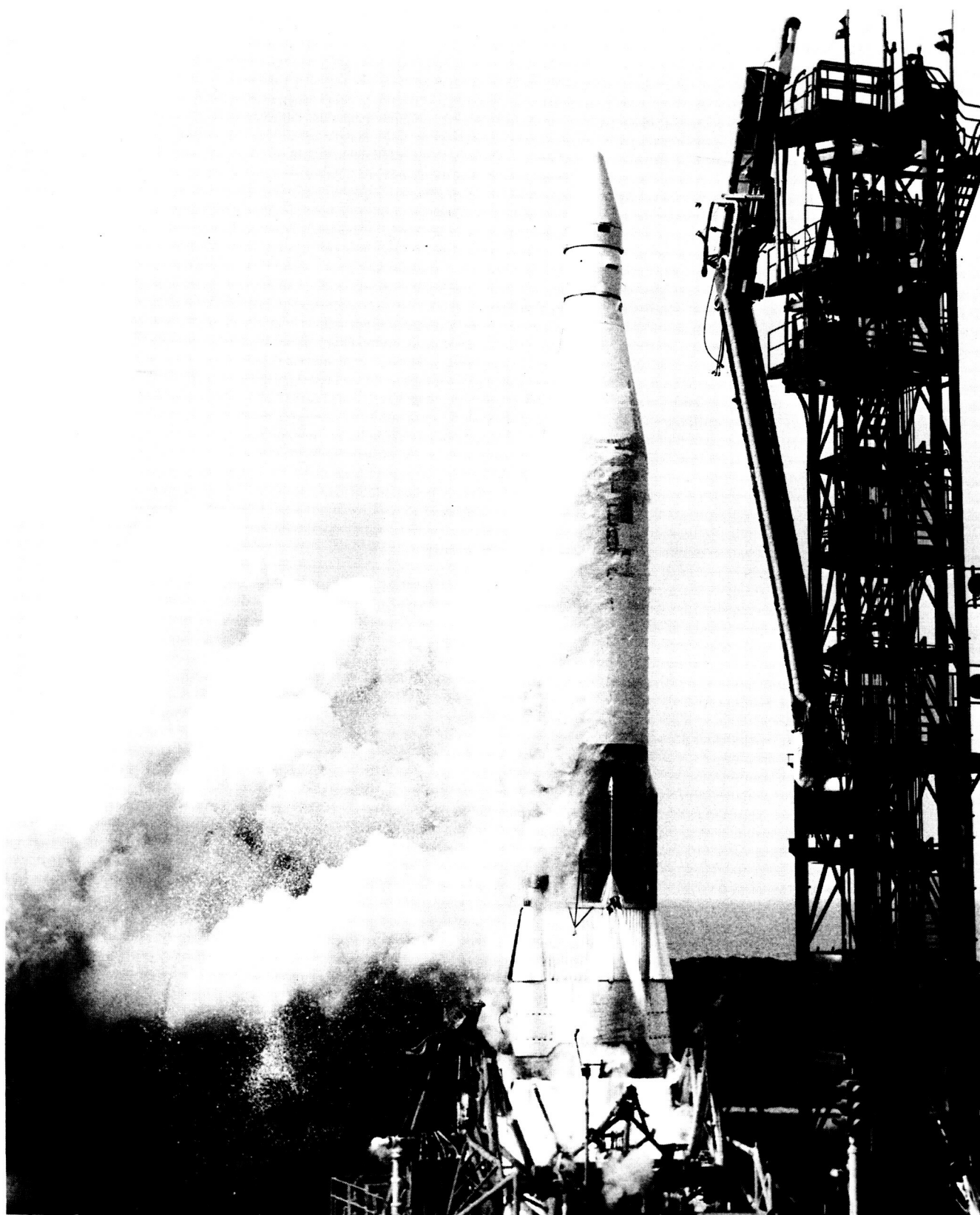
PROJECT FIRE
INTEGRATED POST FLIGHT
EVALUATION REPORT

FLIGHT NO. II

REPORT NO. GDC | BKF65-042
NASA CR66000

24 SEPTEMBER 1965

Compiled by
CONVAIR DIVISION OF GENERAL DYNAMICS
TEST EVALUATION GROUP



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INTRODUCTION

The second Project FIRE space vehicle was successfully launched from Complex 12 at Cape Kennedy, Florida at 16 hours 54 minutes 59.703 seconds, Eastern Standard Time on May 22, 1965 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 re-entry 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, 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 Convair, a Velocity Package produced by Ling-Temco-Vought/Astronautics (containing an Antares II A5 rocket motor), and a Re-entry Package produced by Republic Aviation Corporation. A photograph of the assembled space vehicle is presented in the frontispiece of this report.

The Atlas injected the FIRE spacecraft into a precise ballistic trajectory along the Eastern Test 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 Re-entry Package to 37,239 feet per second for re-entry into the earth's atmosphere 4,257 nautical miles down-range near Ascension Island. A more detailed account of flight events is given in Part 2. Sequence of events times listed on Page 2-3-3 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 Re-entry 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 Re-entry 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 re-entry radio signal "blackout" period. All data were obtained by remote methods since the Re-entry 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 re-entry area gathered the re-entry data.

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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 CONVAIR
INTEGRATED REPORT NO. GDC/BKF65-042**

SECTION 1SUMMARY

The Project FIRE Flight II trajectory provided the desired experimental conditions at the 400,000-foot re-entry test point. All flight events occurred as planned and within allowable time limits.

A large quantity of re-entry data of excellent quality was obtained from radar, telemetry, and optical sources. These data comprise radiation and temperature time history measurements for both re-entry package forebody and afterbody, afterbody pressure, and radio attenuation information. The measurements ensure the complete achievement of all the mission objectives, principally the determination of total and radiative heating rates to a blunt body re-entering the atmosphere at a velocity of 37,000 feet per second.

Performance of the Atlas launch vehicle was excellent. The Atlas successfully injected the Project FIRE II spacecraft into a specified ballistic trajectory of the termination of powered flight. Guidance computer, radar performance, and launch vehicle operating characteristics were well within the expected limits. Spacecraft separation was satisfactorily accomplished.

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

The downrange tracking facilities indicated close agreement with the expected Antares II A5 performance.

Re-entry package operation throughout the flight was excellent with no significant performance deviations. All of the recorded re-entry data were obtained from multiple playbacks after blackout. In addition, there was comprehensive data coverage for the periods before and after blackout.

PART 2

MISSION TRAJECTORY

General Dynamics/Convair
Integrated Report No. GDC/BKF 65-042

By Flight Reentry Programs Office
NASA/Langley Research Center

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

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INTRODUCTION

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.

As a result of the postflight evaluation of data obtained on flight 1, the trajectory for flight 2 was slightly altered. The alteration consisted of reducing, by 85 n.mi., the ground range from the launch point to the reentry test point and changing the flight azimuth to provide a closer passage of the ground track to Ascension Island. The overall result of the trajectory change was to enhance optical coverage during the experimental period.

In order to meet the experimental requirements for flight 2, a nominal trajectory was designed which provided a velocity of 37,255 fps and a reentry angle of -14.959° at an altitude of 400,000 feet. The reentry point was located 4,250 n.mi. downrange from the launch site on a heading of 122.96° from true north and a ground-track minimum passage distance of 53.4 n.mi. southwest of Ascension Island.

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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.

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SUMMARY

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 nominal and actual parameters at the reentry point. As noted in the table, the differences between the nominal 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. mi.
Nominal	1617.16	37,255	-14.959	4250.0
Actual *	1617.74	37,239	-14.738	4256.8
Difference	+0.59	-16	+0.221	+6.8
Tolerance			±1.0	-5, +15

*Obtained from a 2-hour quick look trajectory based on Ascension Island TPQ-18 radar data.

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DISCUSSION OF DATA

SECTION 3

DISCUSSION OF DATA

The Project Fire Flight 2 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.

The actual flight data were obtained by reducing the Ascension Island TPQ-18 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. 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 TPQ-18 radar data and onboard accelerometer measurements. The TPQ-18 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 2.1 percent. This difference is attributed to the accelerometer measurement capabilities and the data transmission system accuracies.

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DISCUSSION OF DATA

Reentry Phase

The reentry phase of the trajectory is shown in figures 2-3-9 and 2-3-10. Figure 2-3-9 presents the variation of altitude, velocity, and flight-path angle with time. The relationship between the reentry ground track and Ascension Island is presented in figure 2-3-10. Since the TPQ-18 radar did not track the reentry package during the experimental period, due to C-band blackout, the trajectory during this period was obtained by computer simulation from the last good radar point. The computer simulation merged smoothly with the radar data obtained after emergence from blackout and the two impact points agree very closely. The computer simulation therefore is considered to be an interpolation of the reentry trajectory during blackout and is the source of the actual data for this phase of flight.

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 nominal. As can be seen from figure 2-3-10, the minimum ground track passage from Ascension during the experimental period was 51.3 n.mi. or about 2.1 n.mi. closer than the expected ground track. The actual impact point was approximately 85 n.mi. southeast of Ascension or about 7.2 n.mi. downrange and 2 n.mi. cross-range 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 2 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.

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DISCUSSION OF DATA

Project Fire Sequence of Events
(Flight 2)

Event description (in-flight sequences)	Expected time, sec	Actual time, sec
Enable V/P ignition interlock (signal transmission)	126.5	127.0
V/P timer start (signal transmission)	294.83	294.386
V/P shroud jettison (signal transmission)	295.5	295.284
Uncage V/P gyros (signal transmission)	302.87	304.59
S/C separation (onboard programmer)*	308.73	308.73
S/C separation (signal transmission)	308.37	310.25
Start V/P pitch program	335.33	334.87
End V/P pitch program	435.5	435.03
Start R/P separation timers	1,538.63	1,538.0
Fire spin rockets	1,545.63	1,545.0
Ignite Antares II-A5 delay squib	1,545.63	1,545.0

*S/C separation was activated by the backup signal from the onboard programmer. It was known that the backup signal would activate this event if the vernier phase was longer than 17.5 seconds, which was the case on this flight.

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DISCUSSION OF DATA

Project Fire Sequence of Events - Continued
(Flight 2)

Event description (in-flight sequences)	Expected time, sec	Actual time, sec
V/P shell separation	1,548.63	1,548.0
Antares II-A5 ignition	1,551.63	1,551.34
Antares II-A5 burnout (main thrust termination)	1,583.6	1,583.0
R/P separation	1,611.63	1,610.43
Arrival at 400,000 ft altitude	1,617.16	1,617.74
Tumble motor ignition	1,617.63	1,616.43
Begin T/M blackout		1,624.7
Begin C-band radar blackout		1,629.0
Start reentry timer (10.4g deceleration)	1,638.56	1,639.11
First heat-shield ejection (pyro-fuse link signal)	1,641.56	1,642.12
Second heat-shield ejection (pyro-fuse link signal)	1,646.56	1,647.53
End T/M blackout		1,655.1
Disable record and erase head	1,661.21	1,661.48
Activate failover switch	1,696.21	1,696.11
R/P impact	1,931.9	1,934.3

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DISCUSSION OF DATA

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 similar to those which were made for flight 1. Measurements of pressure and temperature were made with instrumented Goddard payloads launched on Nike-Apache sounding rockets. The results of these soundings were not available at the time of publication of this report.

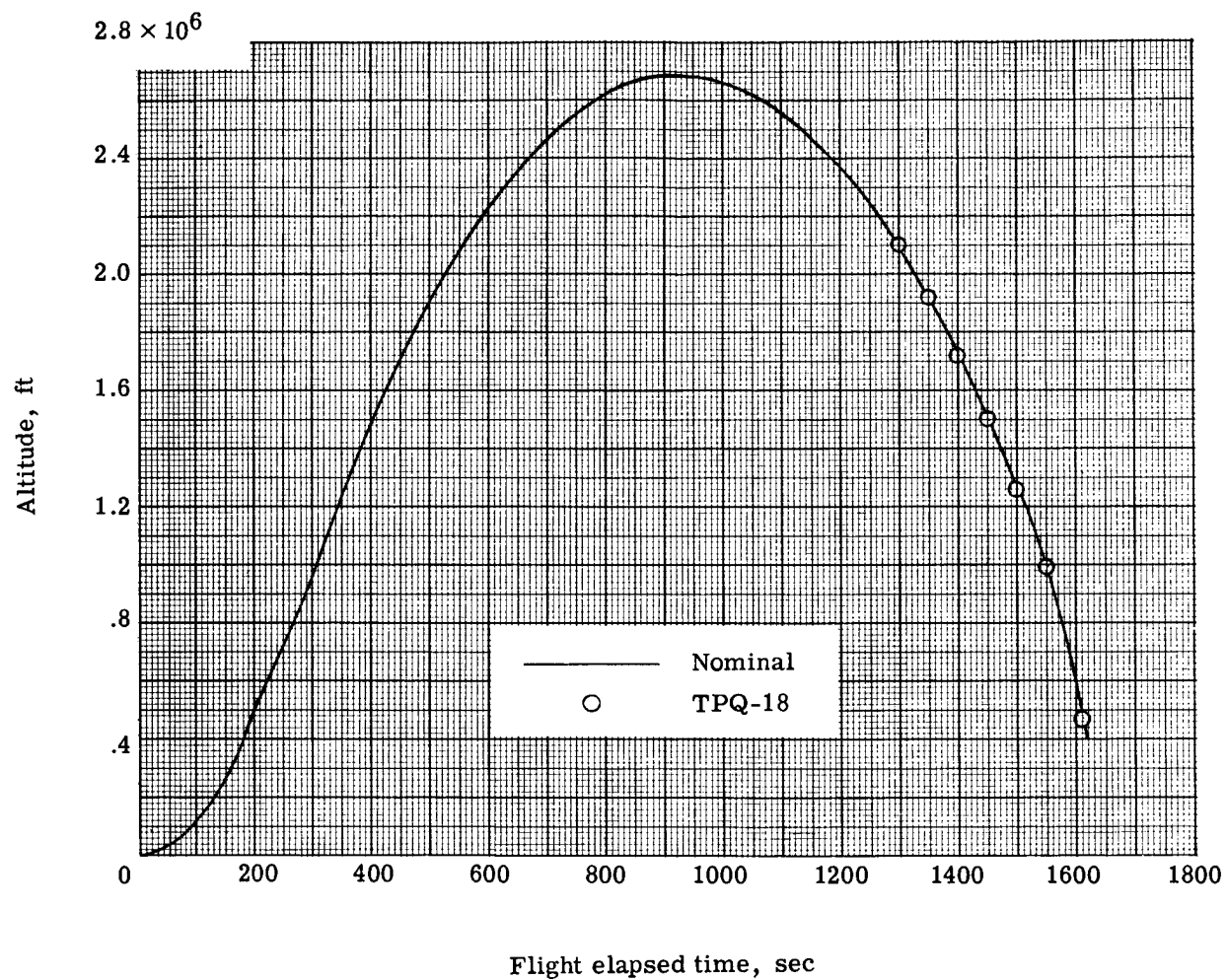
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FIGURE 2-3-6

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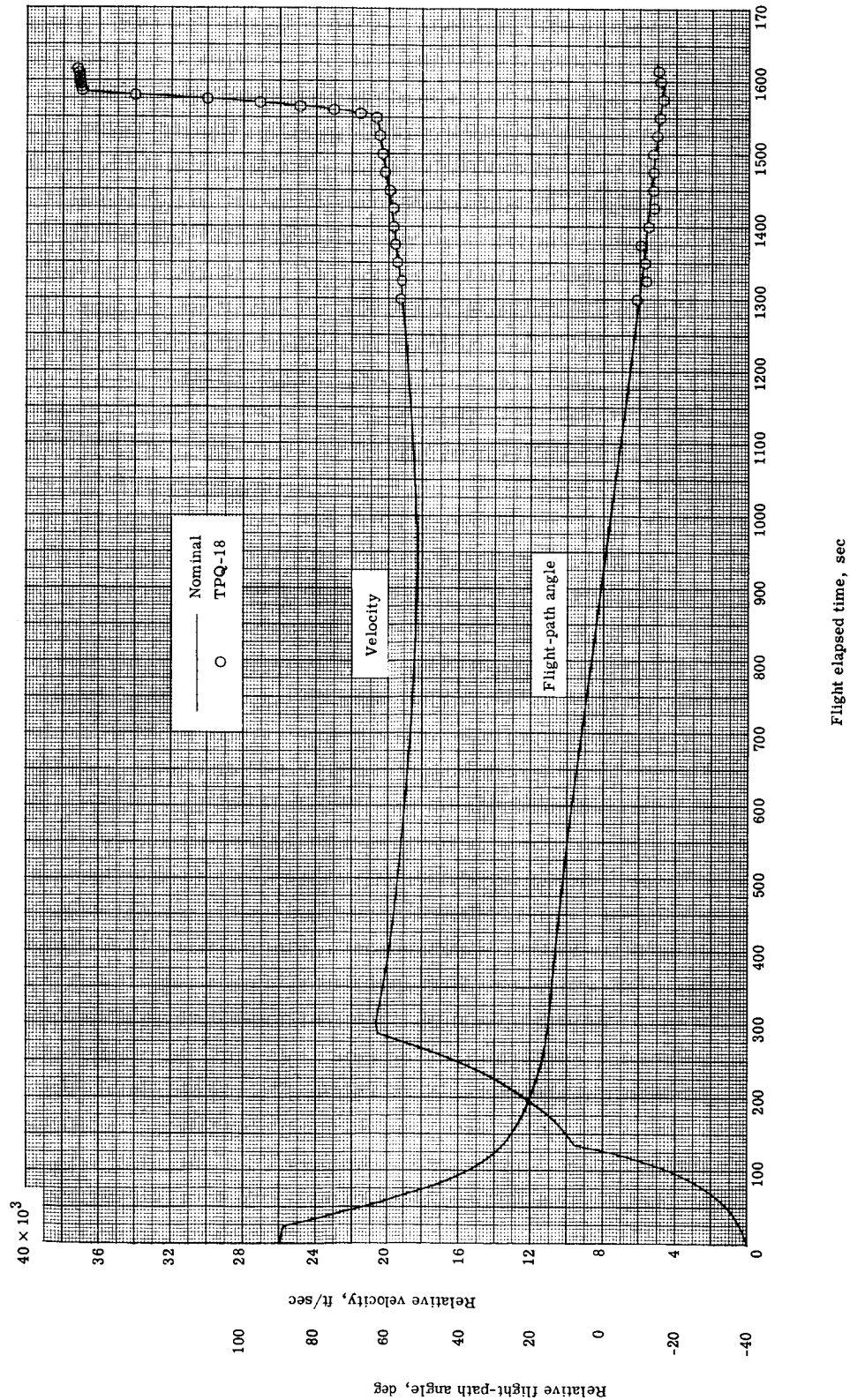
DISCUSSION OF DATA

VARIATION OF ALTITUDE WITH FLIGHT ELAPSED TIME



MISSION TRAJECTORY
 FIGURE 2-3-7
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 DISCUSSION OF DATA

VARIATION OF VELOCITY AND FLIGHT-PATH ANGLE WITH ELAPSED TIME



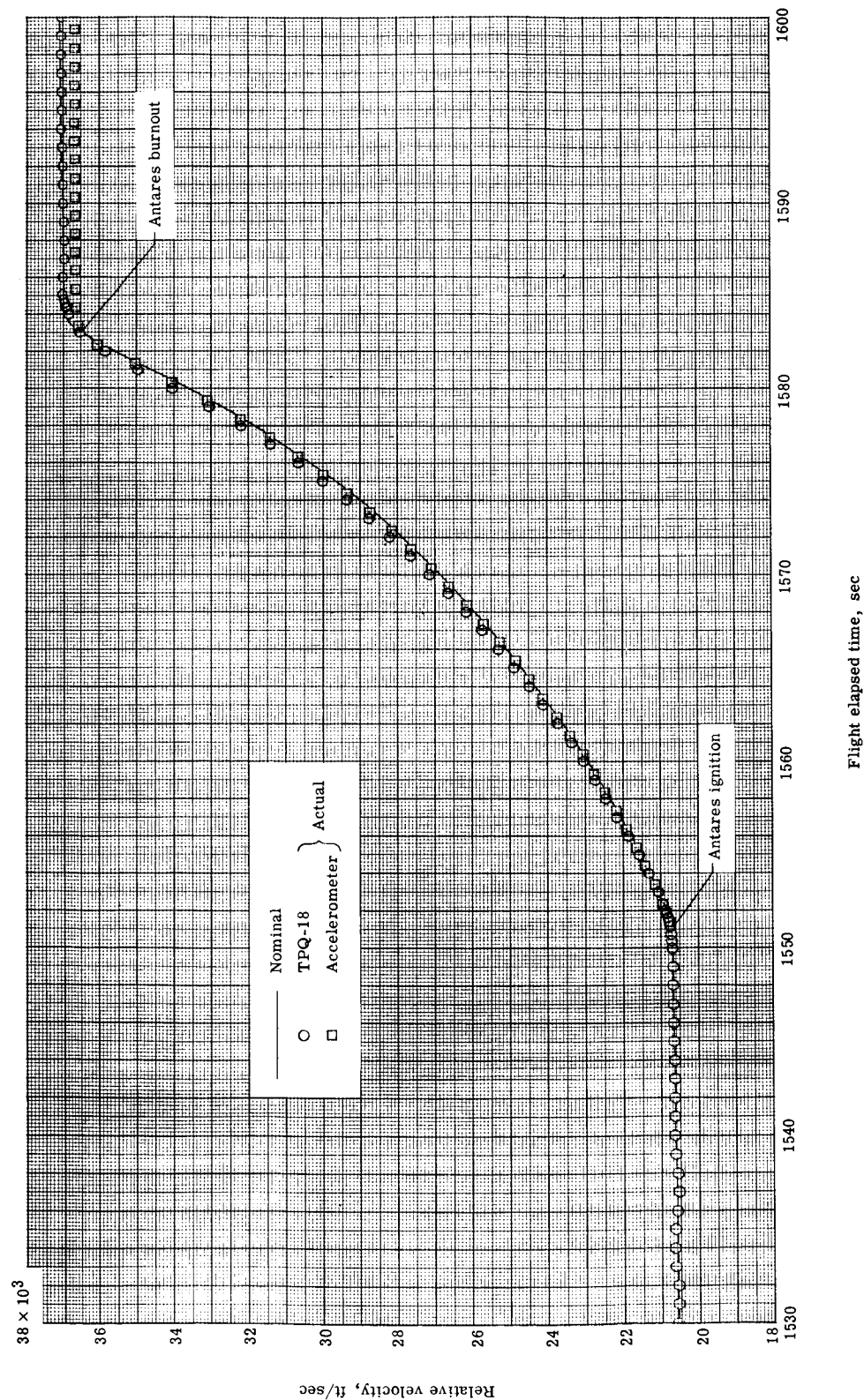
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FIGURE 2-3-8

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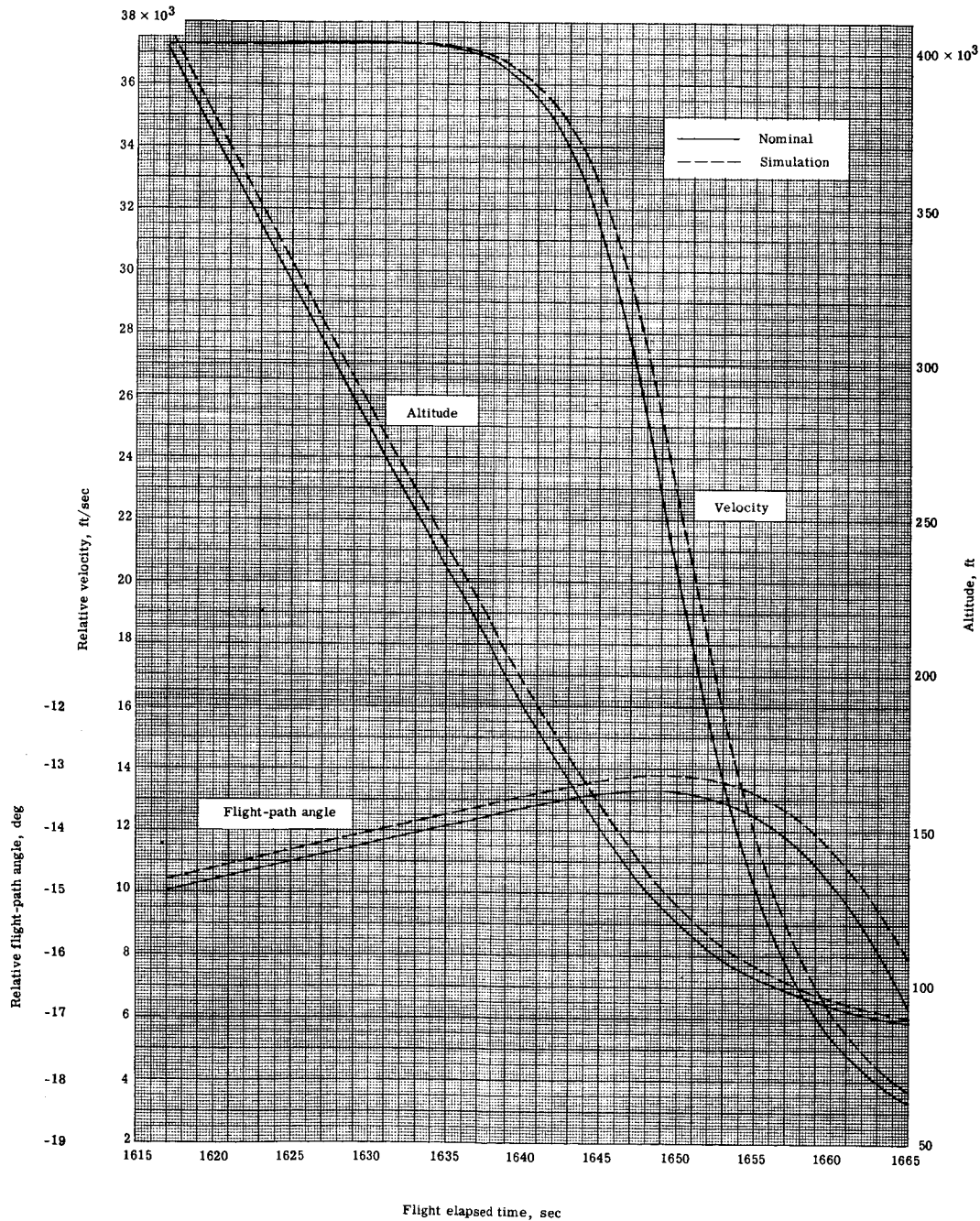
DISCUSSION OF DATA

VARIATION OF VELOCITY WITH TIME DURING ANTARES II-A5 BURN



MISSION TRAJECTORY
FIGURE 2-3-9
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DISCUSSION OF DATA

VARIATION OF ALTITUDE, VELOCITY, AND FLIGHT-PATH ANGLE



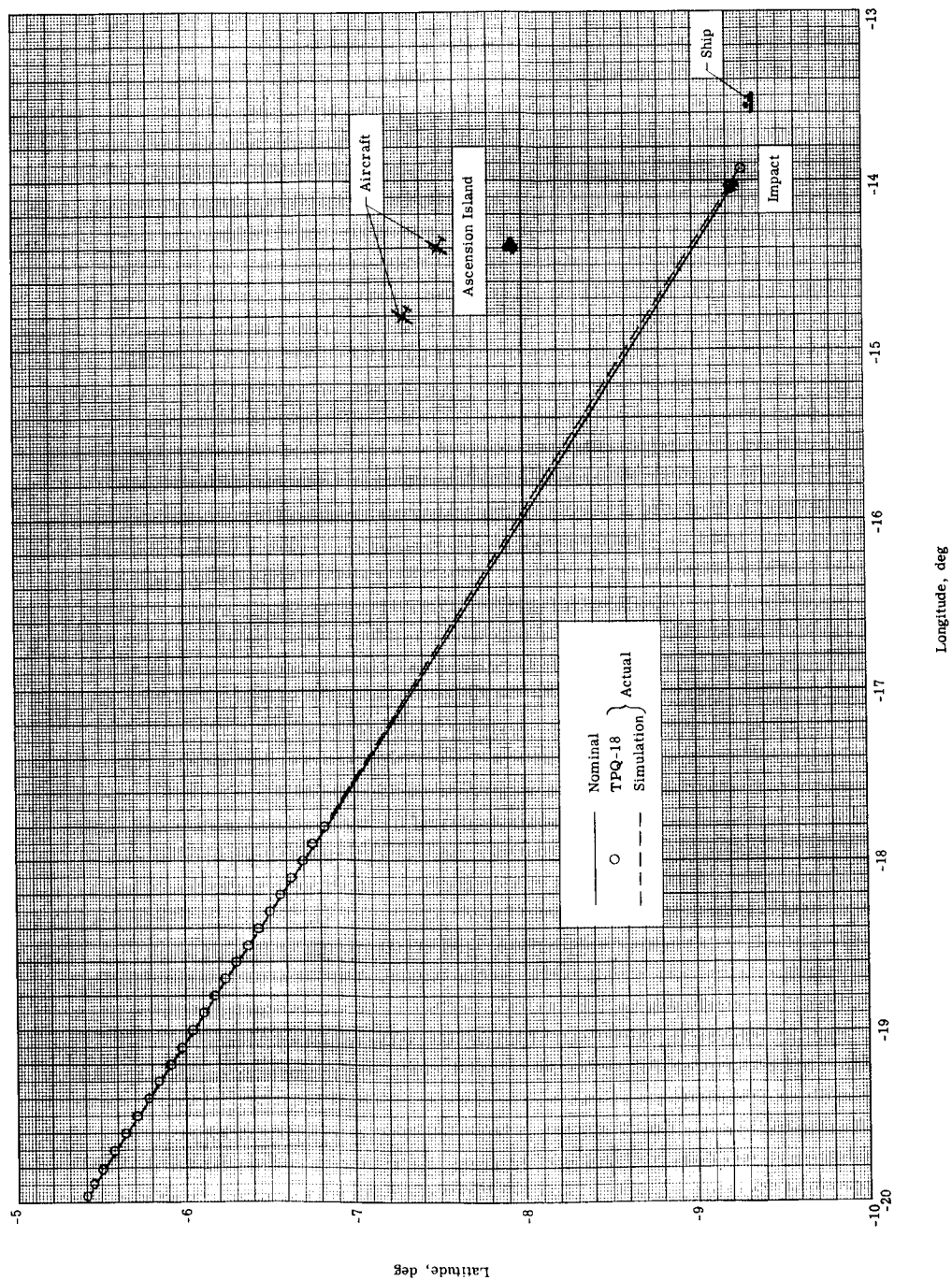
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FIGURE 2-3-10

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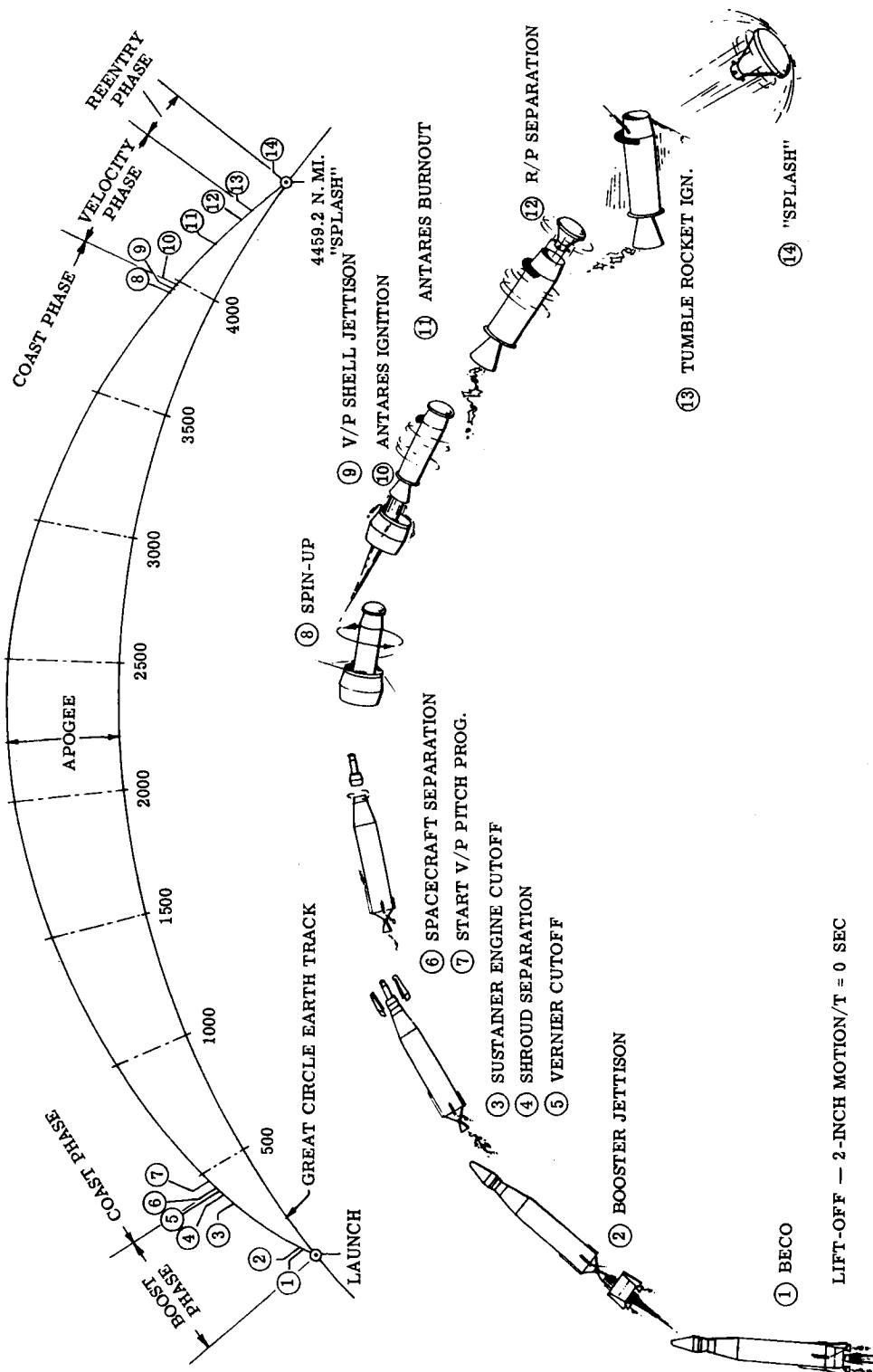
DISCUSSION OF DATA

REENTRY GROUND TRACK



MISSION TRAJECTORY
FIGURE 2-3-11
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DISCUSSION OF DATA

FLIGHT SEQUENCE OF EVENTS



PART 3

MISSION DATA EVALUATION

General Dynamics/Convair
Integrated Report No. GDC/BKF 65-042

By Flight Reentry Programs Office
NASA/Langley Research Center

Approved by: _____

David G. Stone

David G. Stone
Manager, Flight Reentry Programs Office

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3-3-5	Ascension Island Instrumentation
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MISSION DATA EVALUATION

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INTRODUCTION

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 pressure 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 a ship and four 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.

MISSION DATA EVALUATION
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INTRODUCTION

SECTION 1

INTRODUCTION

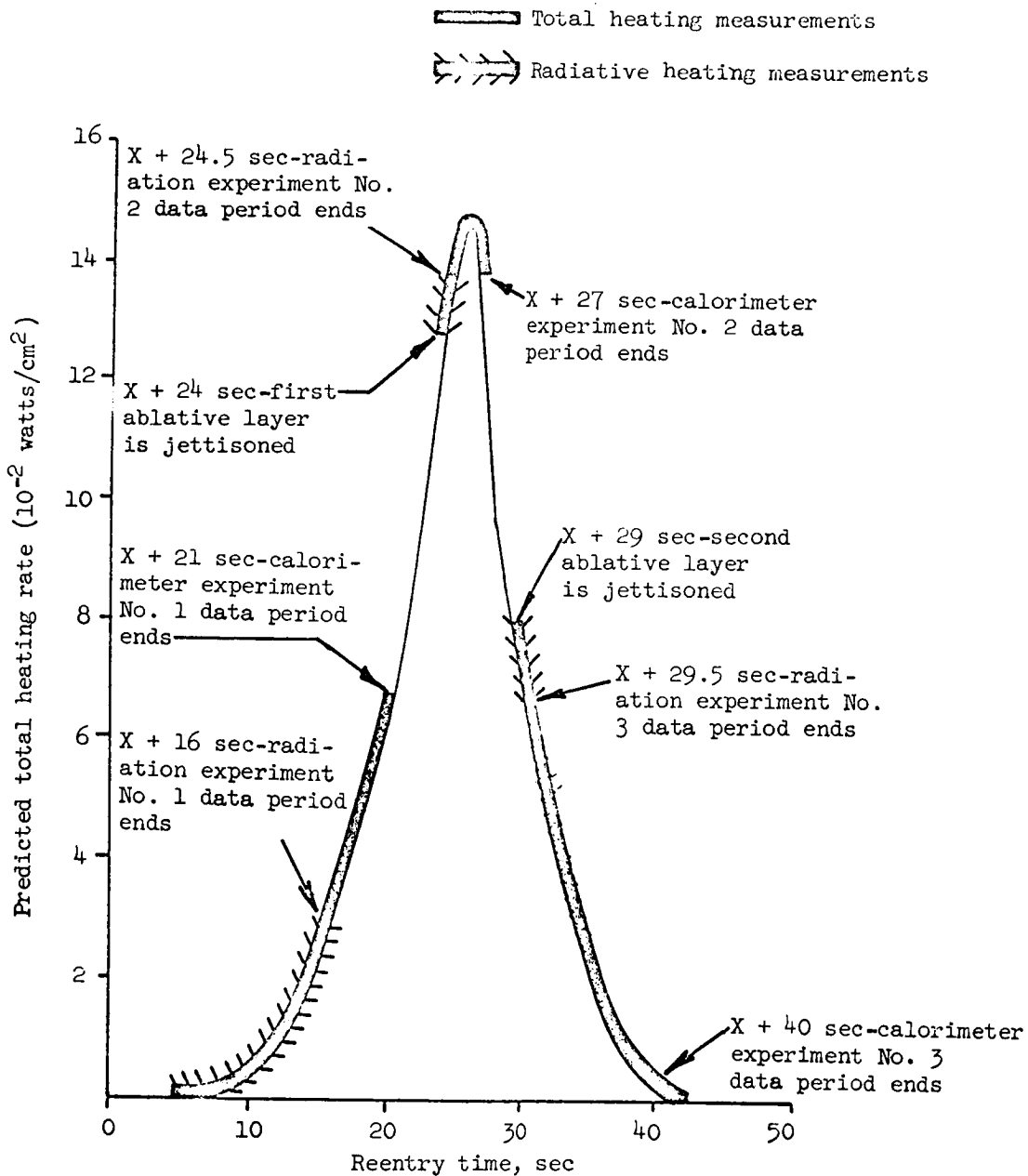
The primary objectives of the Project Fire mission were the determination of the total and radiative heating rates on the forebody and afterbody of a blunt shape in the environment resulting from entry into the earth's atmosphere at a velocity of 37,000 feet per second. In addition, data were to be obtained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

The project comprised two flights. The first one was successfully flown on April 14, 1964, and a considerable amount of reentry data was obtained (see reference 1). The purpose of the second flight was to improve the definition of the total and radiative heat flux curves obtained from Flight 1 by providing additional points of greater accuracy.

The reentry package was designed to obtain total heating by means of calorimeter measurements. 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 could 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 distribution of the radiation at the stagnation point over a wavelength range of about 2000 to 6000 Å. Because the life of the exposed quartz window is even shorter than that of the calorimeter in which it is mounted, valid hot-air radiance measurements can be 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.

MISSION DATA EVALUATION
 FIGURE NO. 3-1-3
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 INTRODUCTION

DATA PERIODS



Note: All times are estimated
 X = 0 time at start of reentry (400,000 ft)

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SUMMARY

SECTION 2

SUMMARY

The second Project Fire space vehicle was launched May 22, 1965, at 1654 hours 59.703 seconds e. s. t., from complex 12 at Cape Kennedy along the Air Force Eastern Test Range.

Except for the blackout period in the reentry area and a 28-second period during midcourse, the vehicle was tracked by radar for its complete 4,500-nautical-mile trajectory. A reentry velocity of 37,239 feet per second at an altitude of 400,000 feet and a reentry angle of -14.7° was achieved.

Although the NASA tracking telespectrograph at Ascension Island obtained almost no spectrographic data, extensive optical coverage was obtained from a number of stations, both on the ground and from aircraft. A completely clear sky during the reentry greatly facilitated this coverage. The data include trajectory information, events-type information, and spectrographic information. Exceptionally good visual coverage of the reentry was obtained by the range optical equipment located on Ascension Island.

The telemetry records from each of the reentry package links were excellent both before and after blackout. Four playbacks of the data recorded during blackout were received on the delay-time link and three on the real-time link both at Ascension Island and aboard the Range ship. The playbacks appear clean and essentially without drop-outs and should be capable of being reduced to engineering units by machine without any difficulty.

The timing of the data periods was such that measurements of the peak radiative heating were acquired during the second data period as planned. Although some body motions were induced during the experiment period, these are not believed to be large enough to have an appreciable effect on the data. Optical records indicate successful separation of the reentry package from the spent Antares motor case and successful jettisoning of the phenolic asbestos heat shields.

The data obtained will permit the full accomplishment of all the mission objectives.

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SUPPORT IN REENTRY AREA

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 plot of the reentry ground track showing its relation to Ascension Island and the deployment of the ship and aircraft. One ship, the Twin Falls (code name Uniform) was on station to monitor the reentry. Four aircraft were deployed in the reentry area. Figure 3-3-5 shows the Ascension Island instrumentation. The following table shows the support provided by each of the stations:

Ascension Island

Radar

FPS-16
TPQ-18
TTR
MOD II

Optics

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

Telemetry

TLM-18

MISSION DATA EVALUATION
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SUPPORT IN REENTRY AREA

Atmospheric soundings

Rawinsonde
Arcas
Nike-Apache

Ship

Radar

Twin Falls

Telemetry

Twin Falls

Aircraft

Optics

NASA-GSFC 232
NASA-GSFC 238

Telemetry

NASA-GSFC 232
NASA-GSFC 238
AFETR-Silver 1
AFETR-Silver 2

The Ascension Island radars (FPS-16, TPQ-18, and 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 measurements of the atmospheric conditions from ground level to an altitude of 325,000 feet.

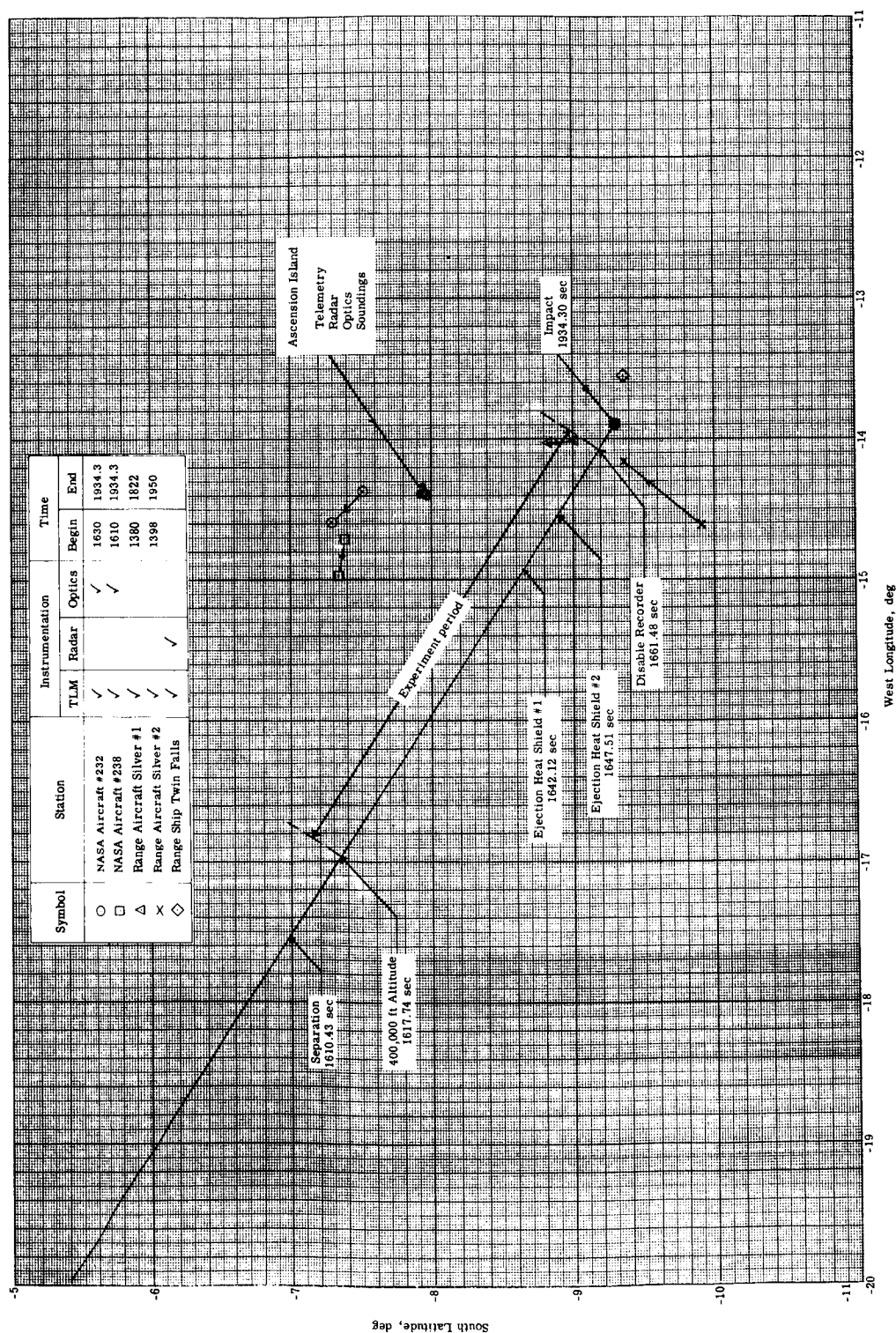
MISSION DATA EVALUATION
PAGE 3-3-3
INTEGRATED REPORT NO. GDC/BKF 65-042
SUPPORT IN REENTRY AREA

The ship, Twin Falls, was deployed to supply telemetry and radar support.

The four aircraft monitored the reentry to provide optical and telemetry backup information.

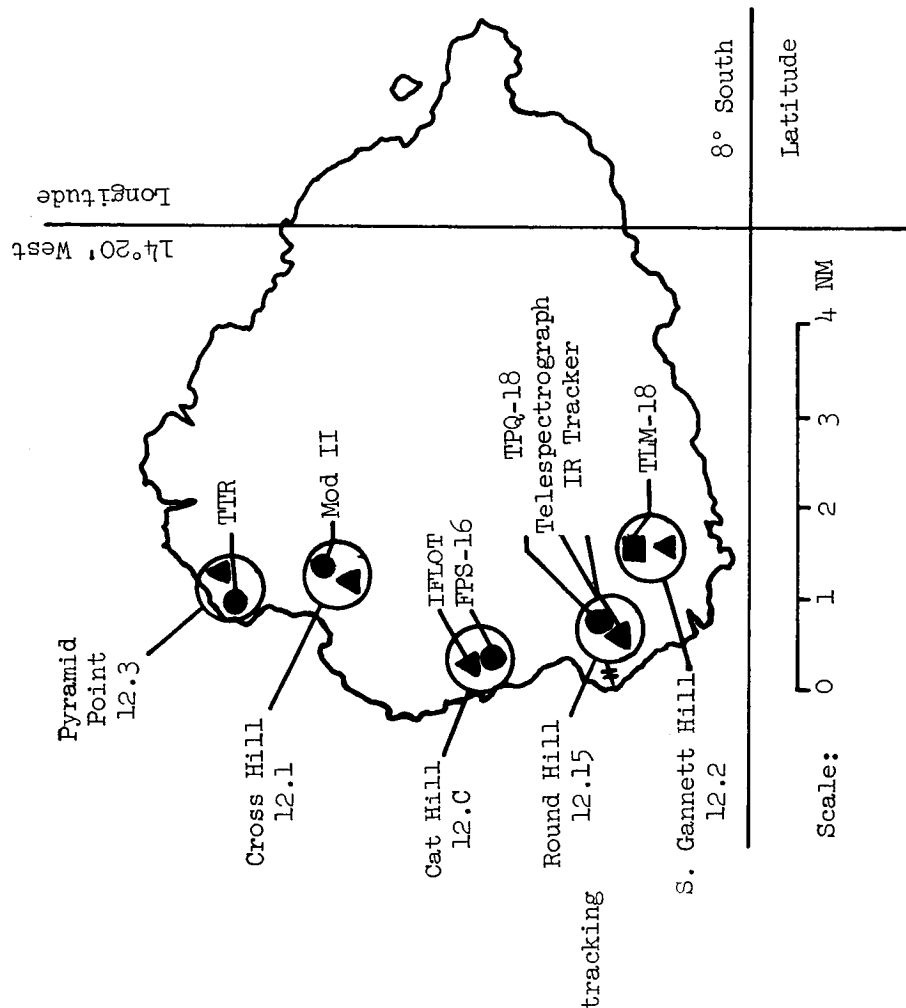
MISSION DATA EVALUATION
 FIGURE NO. 3-3-4
 INTEGRATED REPORT NO. GDC/BKF 65-042
 SUPPORT IN REENTRY AREA

REENTRY AREA SUPPORT



MISSION DATA EVALUATION
 FIGURE NO. 3-3-5
 INTEGRATED REPORT NO. GDC/BKF 65-042
 SUPPORT IN REENTRY AREA

ASCENSION ISLAND INSTRUMENTATION



CAMERA EQUIPMENT
Pyramid Point, South Gannet Hill, Cross Hill

- 3 - Ballistic cameras
- 3 - K-24, single frame streak exposure through grating cameras
- 3 - 4" X 5" Speed Graphic single frame streak cameras
- 3 - 4" X 5" Speed Graphic single frame chop exposure at 10/second

Telespectrograph Site

- 4 - 920 mm fixed spectral cameras
- 3 - K-37 spectral cameras
- 3 - K-37 streak cameras
- 2 - K-24 programmed cameras
- 1 - 70mm Flight Research grating camera
- 1 - 35mm Flight Research grating camera
- 1 - 16mm Miliken boresight camera
- 1 - 16mm Miliken events camera

SITE {
 CODES {
 RADAR
 TELEMETRY
 OPTICAL
 SOUNDING ROCKETS

SECTION 4

DATA ACQUIRED

Telemetry

The velocity package link (244.3 mc) was received essentially continuously from lift-off until the prescribed cutoff time of T + 1548 seconds except for about 20 seconds between T + 1129 and T + 1149. The two reentry package links (delay time, 237.8 and real time, 258.5) were received continuously until loss of signal at Antigua at T + 1075 seconds. The two links were acquired by the Ascension Island TLM-18 at T + 1149 and received until the start of blackout at about T + 1623. They were reacquired on emergence from blackout at T + 1655 and were received until T + 1849.

The Twin Falls received usable signals from the V/P link from T + 1170 to T + 1358 and again from T + 1393 to T + 1548. The real-time R/P link was received by the ship from T + 1170 to T + 1844 except for the blackout period between T + 1625 and T + 1657 seconds. Good signals from the R/P delay link were received by the ship for the following periods: T + 1170 to T + 1245, T + 1393 to T + 1625, and T + 1655 to T + 1844.

None of the four aircraft in the reentry area received telemetry data of usable quality.

The significance of the time periods of telemetry reception can be gaged by the fact, as indicated by the sequence of events presented in Part 2, that the onboard tape recorder erase-record function is disabled at T + 1661.48 and the tape loop containing the prime reentry data replays the data from that time until impact at T + 1934 seconds on the delay link (237.8 mc). In addition, the failover switch was activated at T + 1696.11 seconds so that the same reentry data from the tape loop was also being transmitted on the real-time link (258.5 mc) from this time until splash. It was thus possible to obtain a maximum of six complete playbacks on the delay link and five on the real-time link. Actually, four were obtained on the delay link and three on the real-time link.

MISSION DATA EVALUATION
PAGE 3-4-2
INTEGRATED REPORT NO. GDC/BKF 65-042
DATA ACQUIRED

Radar

The space vehicle and/or the spacecraft was tracked by radar (C-band beacon) for the entire flight from lift-off to shortly before splash, except for a period between T + 1184 and T + 1212, the blackout period between T + 1629 and T + 1655, and the period from T + 1835 to impact (T + 1934).

In the reentry area the FPS-16 at Ascension Island gave valid beacon track for the periods from T + 1212 to T + 1395, T + 1425 to T + 1610, and from T + 1655 to T + 1835. During blackout a short period of skin track from T + 1635 to T + 1651 was also obtained.

The TPQ-18 on Ascension Island gave valid track for the periods T + 1247 to T + 1629, T + 1724 to T + 1835. Blackout began at T + 1629. Reacquisition on emergence from blackout did not occur until T + 1724 seconds.

The target tracking radar (TTR) on Ascension Island skin-tracked first the spacecraft and then the reentry package for the periods T + 1497 to T + 1500, T + 1585 to T + 1630, T + 1670 to T + 1790.

The radar onboard the Range ship tracked for the following periods: T + 1299 to T + 1450, T + 1480 to T + 1580, T + 1739 to T + 1836.

Quick-look reduction of the Ascension Island FPS-16 radar data indicates that the reentry package had a velocity of 37,239 feet per second and a reentry angle of -14.738° at an altitude of 400,000 feet. This altitude, which is considered to be the start of the reentry experiment, was reached at T + 1617.74 seconds. The velocity was less than 0.5 percent lower than expected and the reentry angle was about 0.2° shallower than planned.

The package impacted at 9.289° south latitude and 13.938° west longitude. This position is within 9 miles of the expected impact point.

Optics

Viewing conditions for the optical equipment in the reentry area were essentially perfect, with complete darkness and cloudless sky from horizon to horizon during the reentry period. Figure 3-3-5 shows the location of the optical instrumentation on Ascension Island.

The NASA telespectrograph obtained approximately 3.5 seconds of data in the period from $T + 1631$ to $T + 1642$. cursory examination of this record indicates that it does not contain separate spectrographic data of the reentry package as was desired. Examination to the boresight film shows that the operator tracked for 10.75 seconds from $T + 1631$ to $T + 1642$ seconds. The failure of the telespectrograph to acquire more than the 3.5 seconds of data is associated primarily with the difficulty of tracking with an instrument of such narrow field of view (96 seconds of arc). The telespectrograph and its auxiliary equipment operated normally during reentry.

Four auxiliary cameras mounted on the telespectrograph consisted of:

- (a) 70 mm Flight Research camera with 150 L/mm grating
- (b) 35 mm Flight Research camera with 300 L/mm grating
- (c) 16 mm Miliken boresight camera, black and white
- (d) 16 mm Miliken events camera, color

These cameras obtained good data for 10.75 seconds to 13.4 seconds during reentry. The 70-mm spectral camera recorded a total of 25 seconds of data.

The still cameras at the telespectrograph site included:

- (a) Four 920 mm fixed spectral cameras with 300 L/mm grating
- (b) Three K-37 spectral cameras with gratings
 - Two 300 L/mm grating
 - One 600 L/mm grating

MISSION DATA EVALUATION
PAGE 3-4-4
INTEGRATED REPORT NO. GDC/BKF 65-042
DATA ACQUIRED

- (c) Three K-37 streak cameras
- (d) Two K-24 programed cameras

All of these cameras provided excellent data.

There were three other still-camera sites, Cross Hill (12.1), South Gannet Hill (12.2), and Pyramid Point (12.3). Each of these sites was instrumented with the following:

- (a) Three ballistic cameras (only two at Cross Hill)
- (b) Three K-24 single frame streak exposure through grating cameras
- (c) Three 4 by 5 speed graphic single frame streak cameras
- (d) Three 4 by 5 speed graphic single frame chop exposure at 10/second

All of these still cameras recorded reentry data. The shutters were open from T + 1550 to T + 1720 seconds.

The NASA-238 aircraft was equipped with the following:

- (a) One RC-5 ballistic camera
- (b) Six K-37 spectral cameras with gratings
 - One 150 L/mm
 - Three 300 L/mm
 - Two 600 L/mm
- (c) 70 mm Cine Spectrograph with 210 L/mm grating
- (d) 16 mm Miliken events camera

All this equipment functioned normally and obtained data. Item (c) recorded 18 seconds of data on the reentry package and about 11 seconds of booster reentry. All cameras were operated from T + 1630 to T + 1715 seconds.

MISSION DATA EVALUATION
PAGE 3-4-5
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DATA ACQUIRED

The NASA-232 aircraft carried the following optical instrumentation:

- (a) Eight K-24 spectral cameras with grating

Two 150 L/mm

Two 200 L/mm

Two 400 L/mm

Two 600 L/mm

- (b) Two K-37 streak cameras

- (c) One RC-7 ballistic camera

- (d) 70 mm Cine Spectrograph with 210 L/mm grating

- (e) 16 mm Miliken events camera

All cameras obtained valid data except four of the K-24 spectral cameras which did not have the reentry in the field of view. The Cine Spectrograph recorded 5 seconds of reentry package data and 16 seconds of booster data. All cameras were operated from T + 1630 to T + 1715 seconds.

The following films from the Intermediate Focal Length Optical Tracker (IFLOT) were obtained:

- (a) 16 mm black and white

- (b) 35 mm color from T + 1634 to T + 1648

- (c) 70 mm black and white from T + 1633 to T + 1655

- (d) 70 mm color from T + 1632 to T + 1650

The following films were obtained from the IR tracker:

- (a) 35 mm boresight black and white

MISSION DATA EVALUATION

PAGE 3-4-6

INTEGRATED REPORT NO. GDC/BKF 65-042

DATA ACQUIRED

- (b) 70 mm visible, black and white, from T + 1641 to T + 1647
- (c) 70 mm UV, black and white, from T + 1641 to T + 1645
- (d) 70 mm IR, black and white, from T + 1641 to T + 1652
- (e) 70 mm UV Cine Spectrograph, from T + 1629 to T + 1648
- (f) 70 mm IR Cine Spectrograph

Atmospheric Soundings

In order to determine the properties of the atmosphere through which the reentry took place, a Nike-Apache sounding rocket carrying a Goddard pitot-static tube payload was launched from Ascension Island about 4 hours after the reentry package impact. The payload was tracked and its telemetry data acquired by crews from the Universities of Michigan and New Mexico under contract to GSFC. These data in combination with those from rawinsonde and Arcas rocket soundings provide accurate information on the variation of density, pressure, and temperature with altitude up to an altitude of about 325,000 feet.

MISSION DATA EVALUATION
PAGE 3-5-1
INTEGRATED REPORT NO. GDC/BKF 65-042
DATA ASSESSMENT

SECTION 5

DATA ASSESSMENT

As pointed out previously, four replays of the prime experiment data measured during blackout were received on the delay link and three replays were received on the real-time link by the Ascension Island TLM-18. The Range instrumentation ship, Twin Falls, also received a similar amount of data. Excellent data for the periods before and after blackout were also obtained. An oscillograph record of one of the replays is shown in figure 3-5-3. This figure identifies the data obtained on each of the IRIG channels and notes a number of significant events during the experiment period. The record is clean and essentially without dropouts and should cause no problems in data reduction.

The experiment period is considered to start at an altitude of 400,000 feet and end when the tape recorder erase-record feature is disabled. The extent of the experiment period is 43.74 seconds, of which 30.4 seconds occur during blackout.

The yaw rate and pitch rate in the early portion of the record are essentially zero, indicating that separation of the reentry package from the burned-out Antares II motor had negligible tipoff effects. At about the time of melting of the first beryllium calorimeter there is indication of body motion as evidenced by small yaw and pitch rates. Larger rates are obtained during the third data period. However, a preliminary analysis indicates that the amplitudes of the motion are not great enough to have a major effect on the data. For example, the estimates indicate an angle-of-attack variation of only between 1° and 6° during the second data period and between 2.7° and 8.5° during the third data period.

The total radiometer measurements record the variation of the radiant heating during the experiment and (except for periods of window obscurement or deterioration) show a smooth rise and then decrease in the heating. It can be seen that the peak radiation occurs just after the beginning of the second calorimeter data period, indicating that the planned timing of the data periods was correct.

MISSION DATA EVALUATION
PAGE 3-5-2
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DATA ASSESSMENT

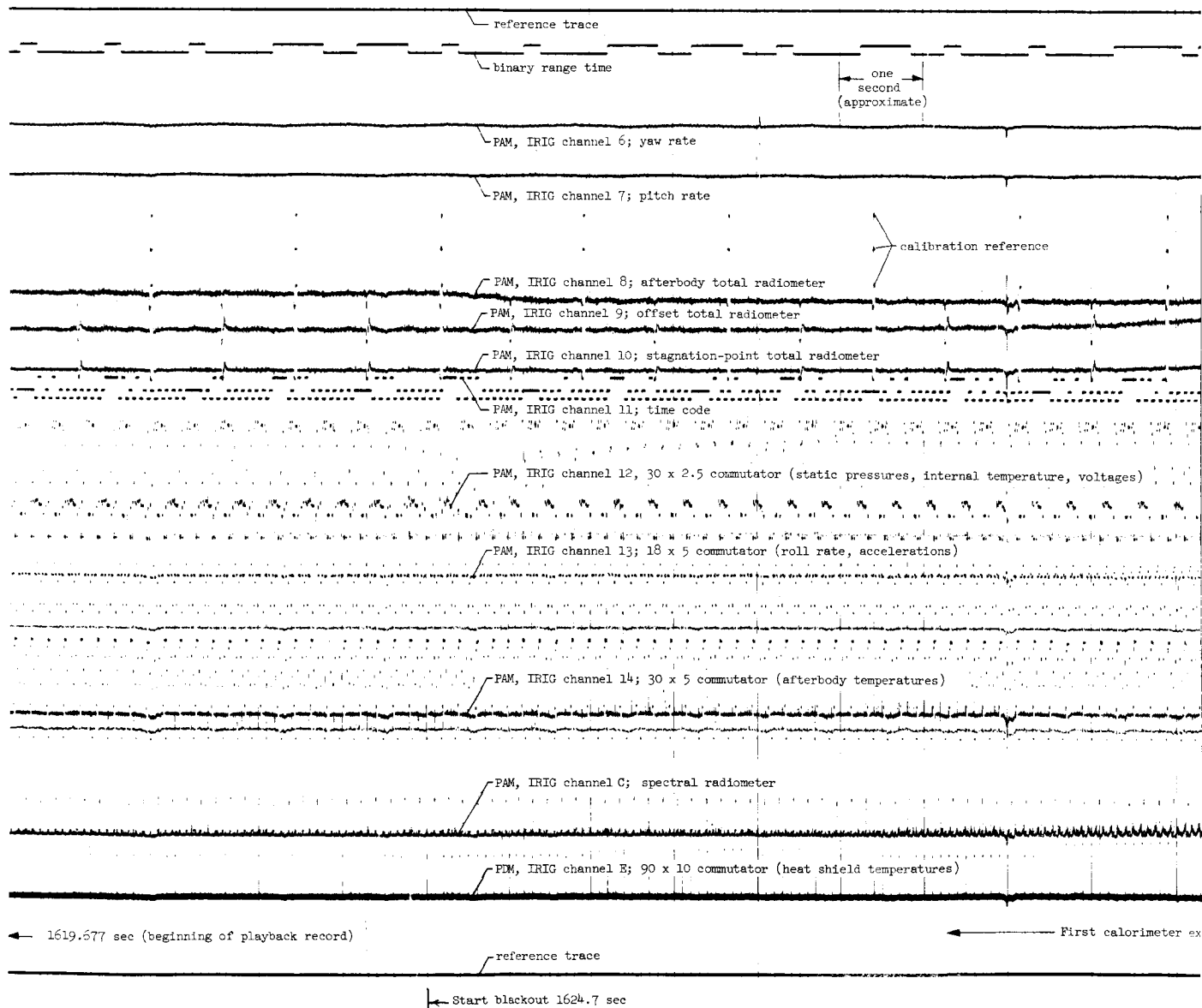
Preliminary readout of selected measurements from the PAM commutators showed smooth variation of the parameters with time. In no case were the ranges of the instrumentation exceeded. The latter statement is also true for all the instrumentation onboard the reentry package.

The spectral radiometer performed well and its record appears compatible with the total radiometer results. These data should provide useful information on the spectral distribution of the radiation.

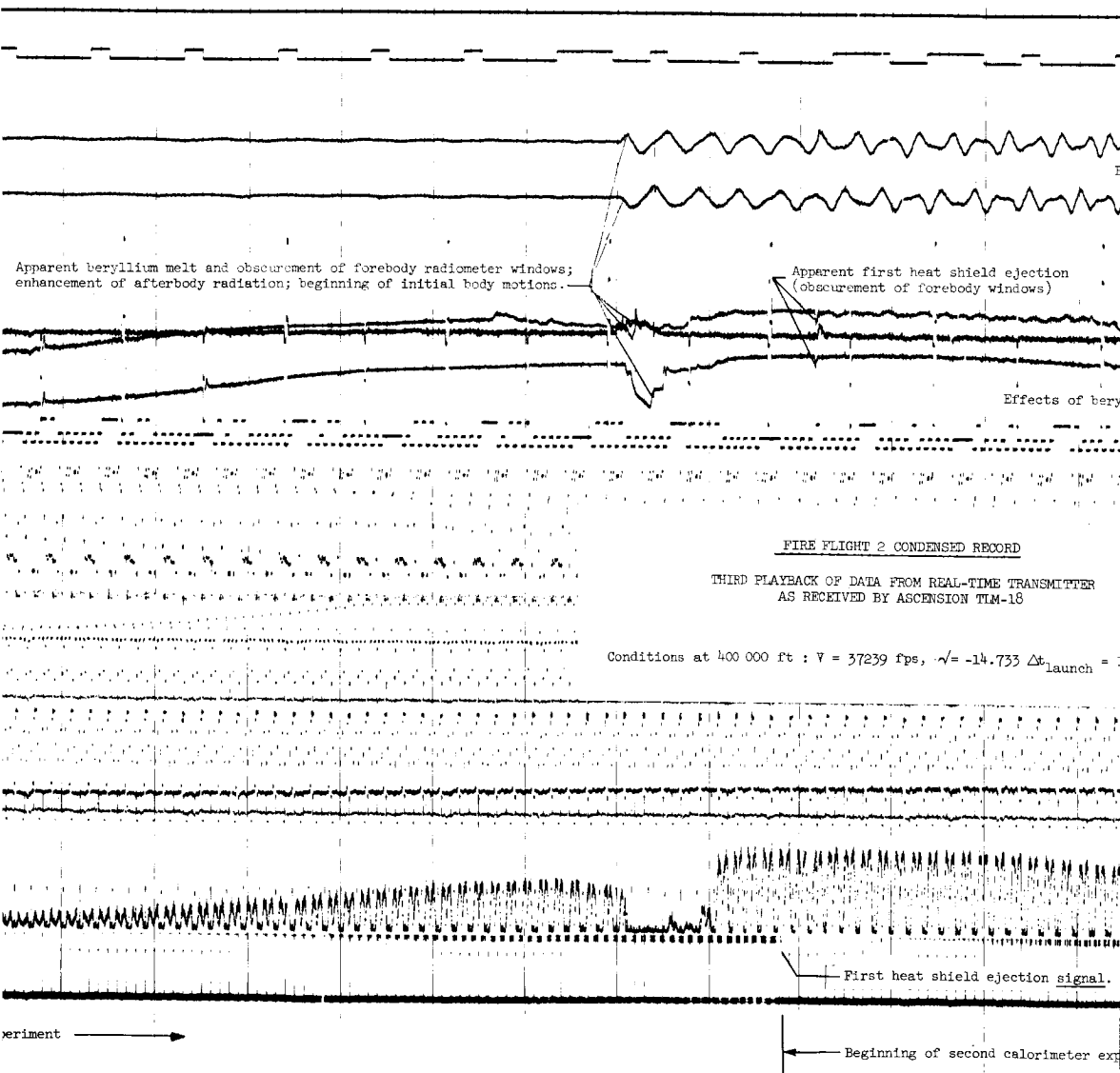
From the record of the heat-shield temperatures (PDM, 90 by 10 commutator) a continuous increase in calorimeter and heat-shield temperatures can be inferred from the progressive darkening of the pulses. Preliminary readout of the beryllium calorimeter temperatures indicates smooth variation of the data with only small scatter.

Photographic records of the reentry confirm the indications of the telemetry that all systems functioned as planned. The reentry film shows that there was no interference with the reentry package by any part of the burned-out Antares motor and that the phenolic asbestos heat shields were successfully jettisoned at the proper time.

It is concluded that the data obtained will make possible complete achievement of all the mission objectives.

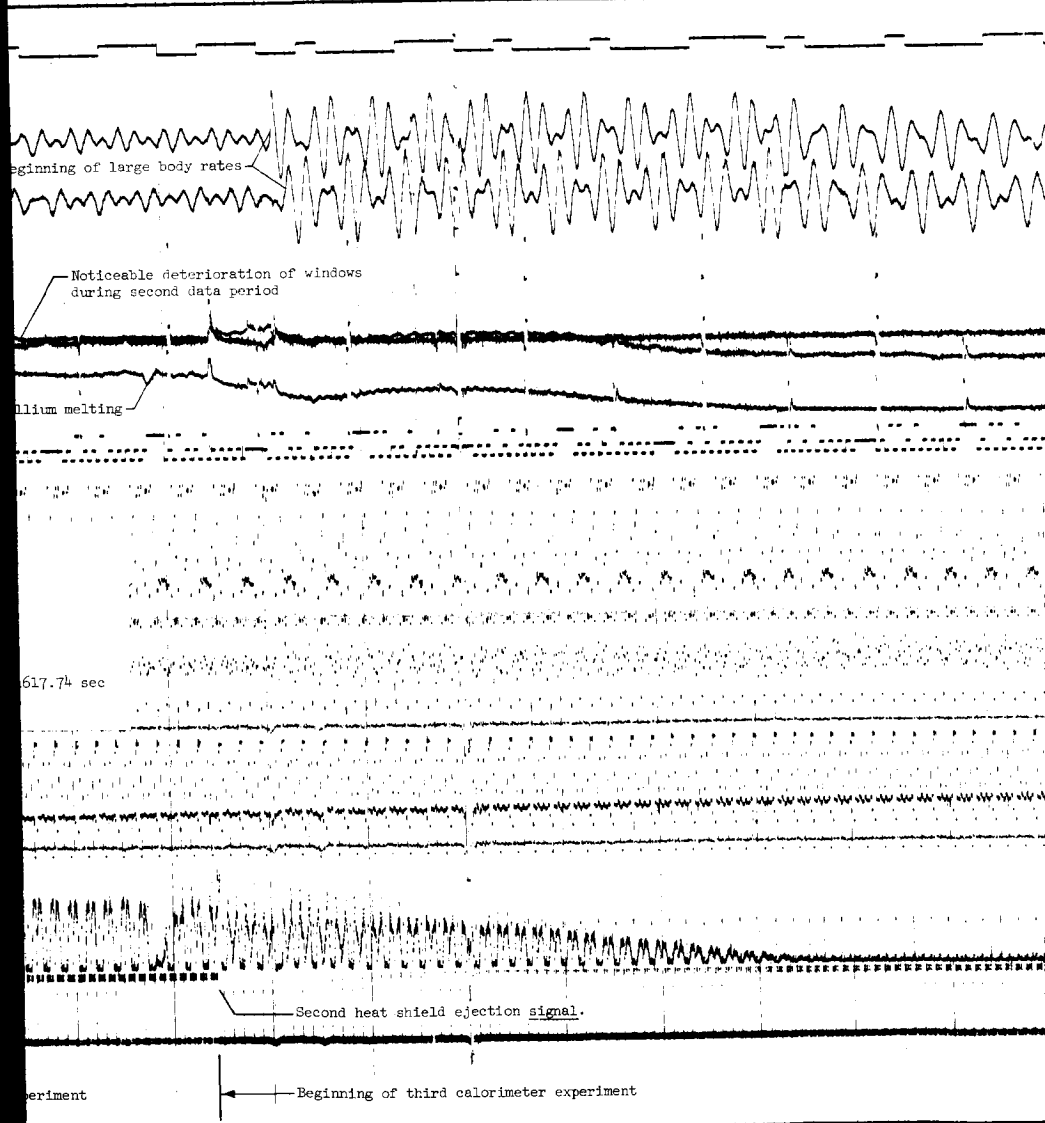


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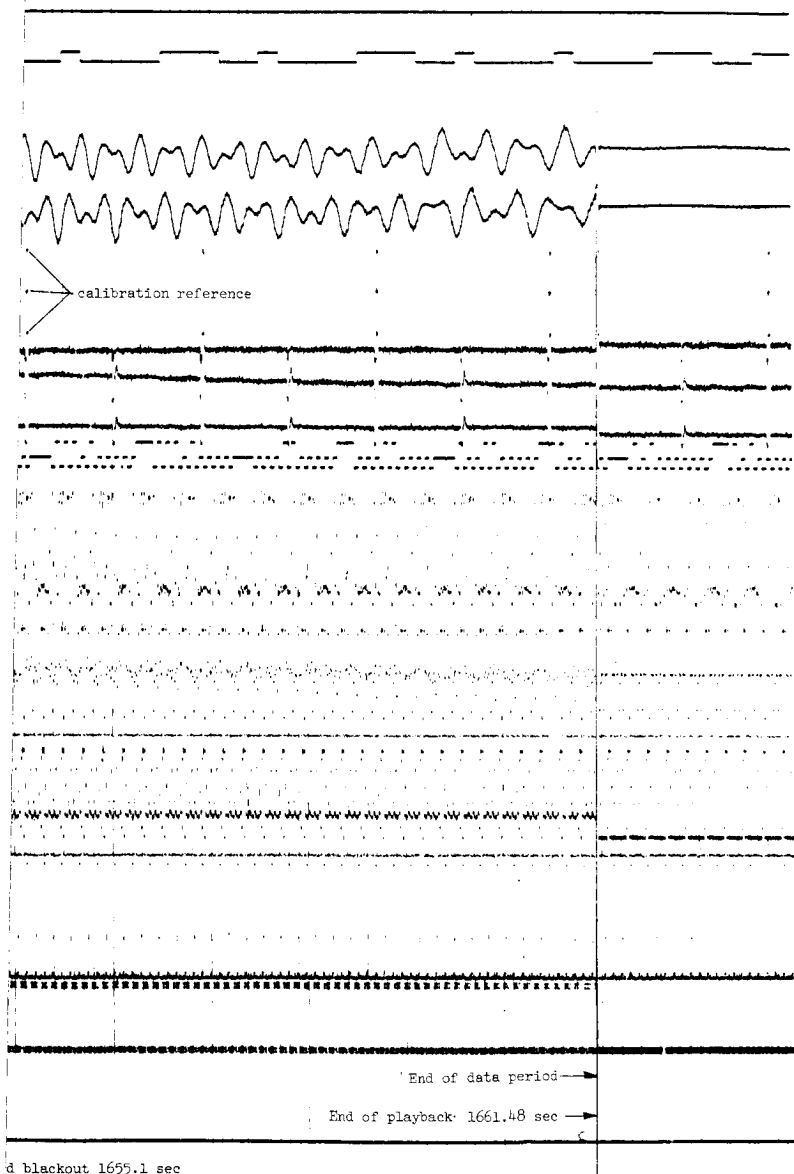


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MISSION DATA EVALUATION
FIGURE NO. 3-5-3
INTEGRATED REPORT NO. GDC/BKF 65-042
DATA ASSESSMENT

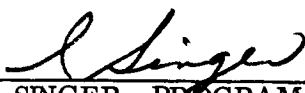
TELEMETRY PLAYBACK RECORD



PART 4

REENTRY PACKAGE PERFORMANCE

GENERAL DYNAMICS/CONVAIR
INTEGRATED REPORT NO. GDC/BKF65-042
REPUBLIC REPORT NO. 499-51-II

APPROVED BY: 
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. Following is a brief description of these packages.

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 two tumbling rockets 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.

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 4-1-5. The forebody and 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 fiberglass structural combination covered with a laminate of Min-K and phenolic asbestos which is coated with a Sylgard formulation.

REENTRY PACKAGE PERFORMANCE

PAGE NO. 4-1-2

INTEGRATED REPORT NO. GDC/BKF65-042

RAC REPORT NO. 499-51-II

DESCRIPTION

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 pressure 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 4-1-6. The temperature is measured by calorimeters (not shown in Figure 4-1-6) 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 cap. 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 4-1-6).

REENTRY PACKAGE PERFORMANCE
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RAC REPORT NO. 499-51-II
DESCRIPTION

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 a pressure transducer located in the afterbody. The transducer has an associated power converter (see Figure 4-1-6).

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 4-1-7.

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

REENTRY PACKAGE PERFORMANCE
PAGE NO. 4-1-4
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DESCRIPTION

time frequency is 237.8 megacycles. Delay data are identical to real time except for the nominal 45-second delay. To enable transmission of delay data over both real and delay time transmitters 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 the 35-second failover timer, and transfers the output of the delay recorder to the input of both transmitters. The incident and reflected power from the transmitters is monitored by two bi-directional couplers.

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 4-1-8.

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 4-1-9.

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 4-1-10). Following separation, the cooling package blower motor is turned off for the terminal portion of the mission, in order to retain an extra margin of instrumentation battery power.

Figure 4-1-11 is a schematic block diagram showing the interrelation of the data sensing, data acquisition, data transmission, and heat shield ejection systems.

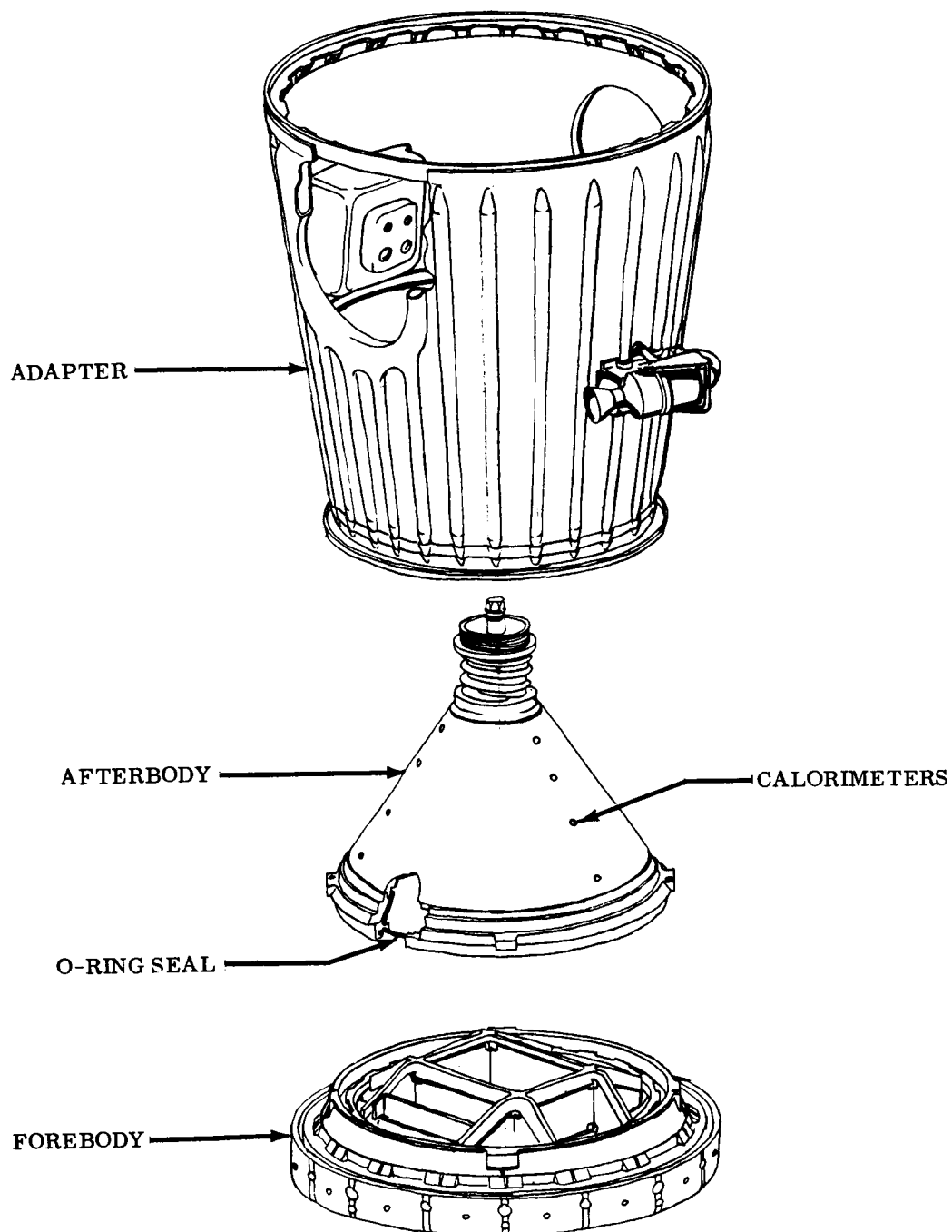
Figure 4-1-12 shows (schematically) a build-up of the R/P.

Figure 4-1-13 shows the R/P in the open condition, and with many of the previously mentioned components visible.

Figure 4-1-14 shows the R/P in the reentry flight configuration.

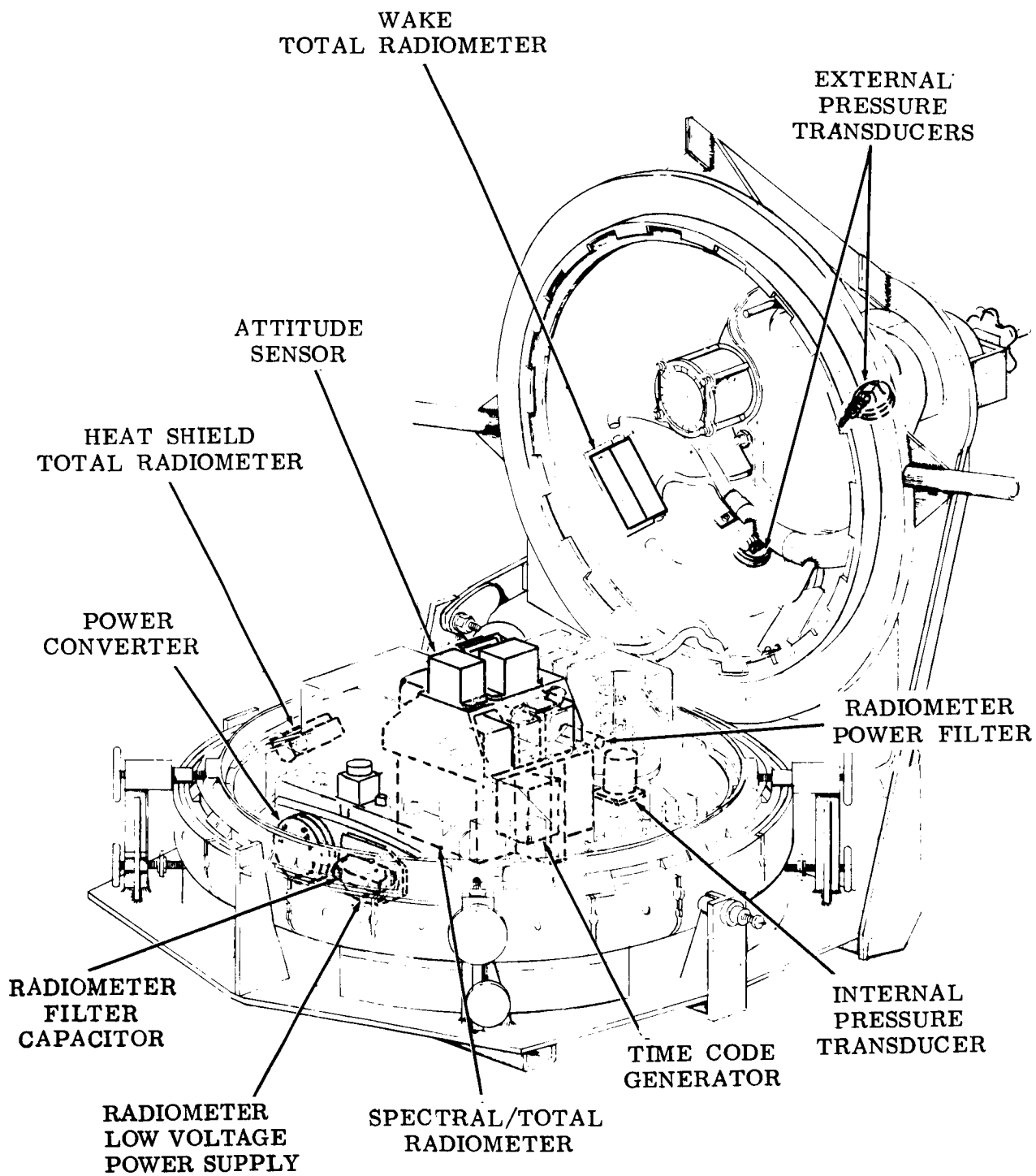
REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-5
INTEGRATED REPORT NO. GDA/BKF65-042
RAC REPORT NO. 499-51-II
DESCRIPTION

SCHEMATIC OF REENTRY PACKAGE AND ADAPTER



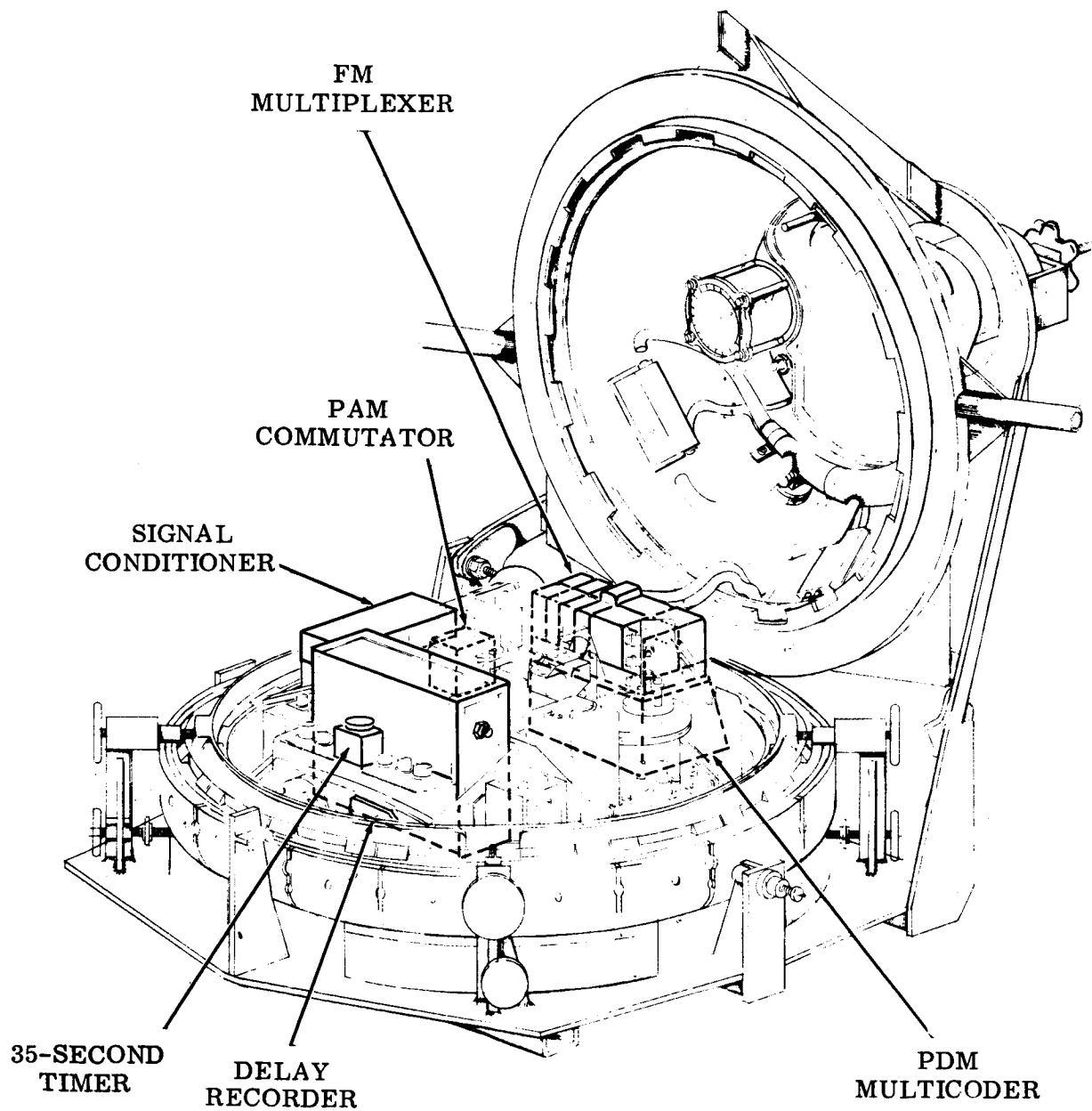
REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-6
INTEGRATED REPORT NO. GDA/BKF65-042
RAC REPORT NO. 499-51-II
DESCRIPTION

SCHEMATIC OF DATA SENSING SYSTEM



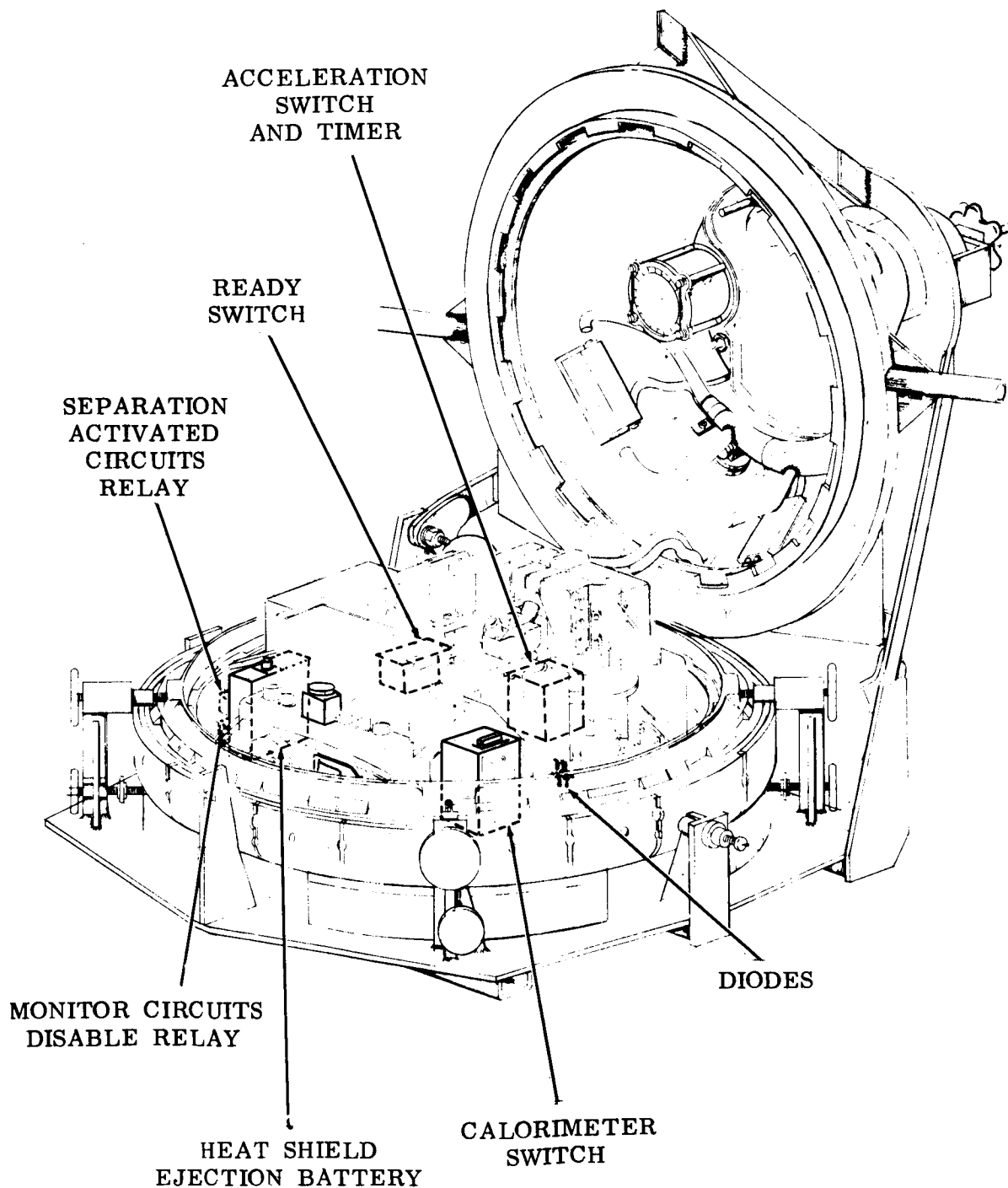
REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-7
INTEGRATED REPORT NO. GDC/BKF65-042
RAC REPORT NO. 499-51-II
DESCRIPTION

SCHEMATIC OF DATA ACQUISITION SYSTEM



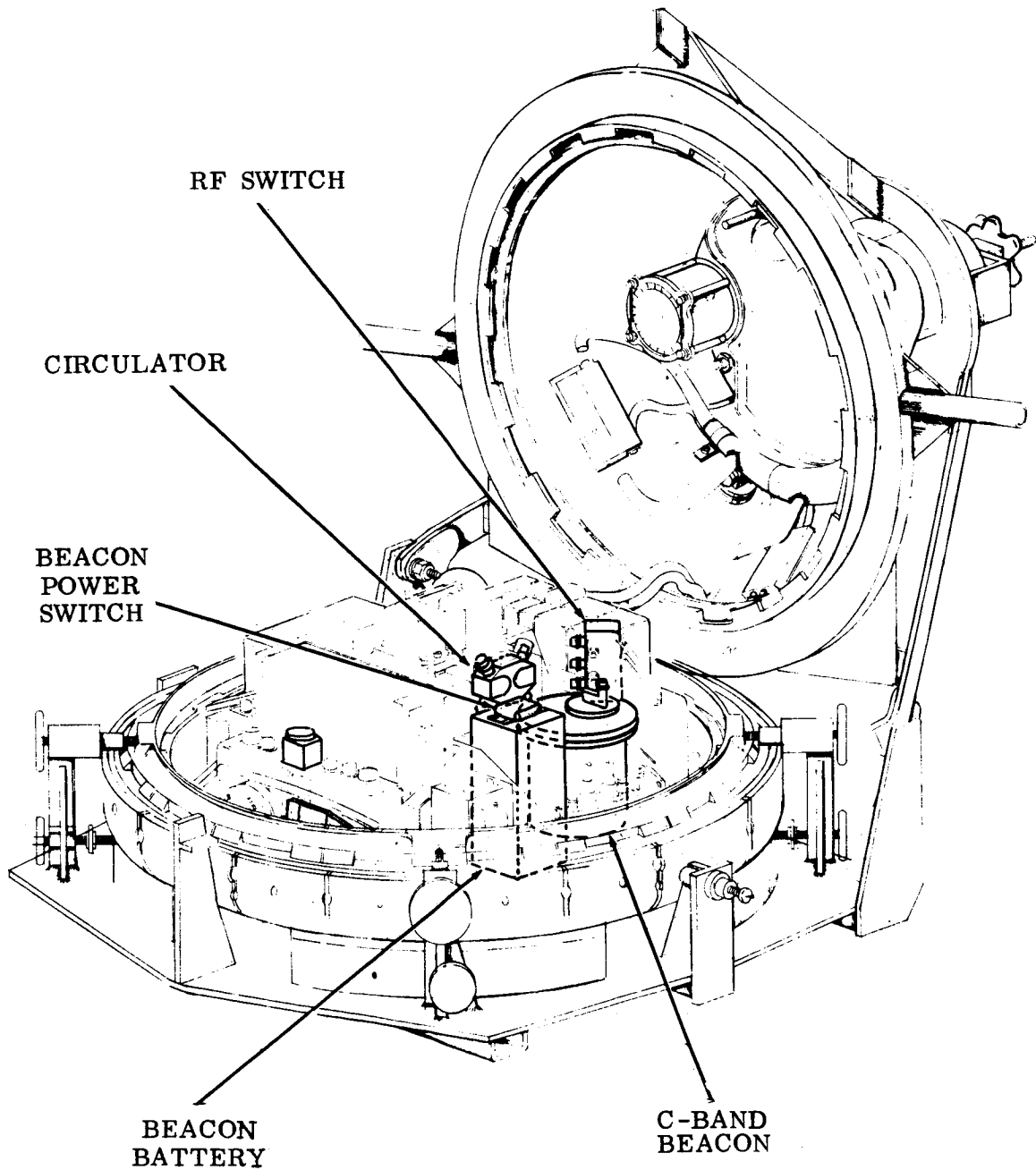
REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-8
INTEGRATED REPORT NO. GDC/BKF65-042
RAC REPORT NO. 499-51-II
DESCRIPTION

SCHEMATIC OF HEAT SHIELD EJECTION SYSTEM



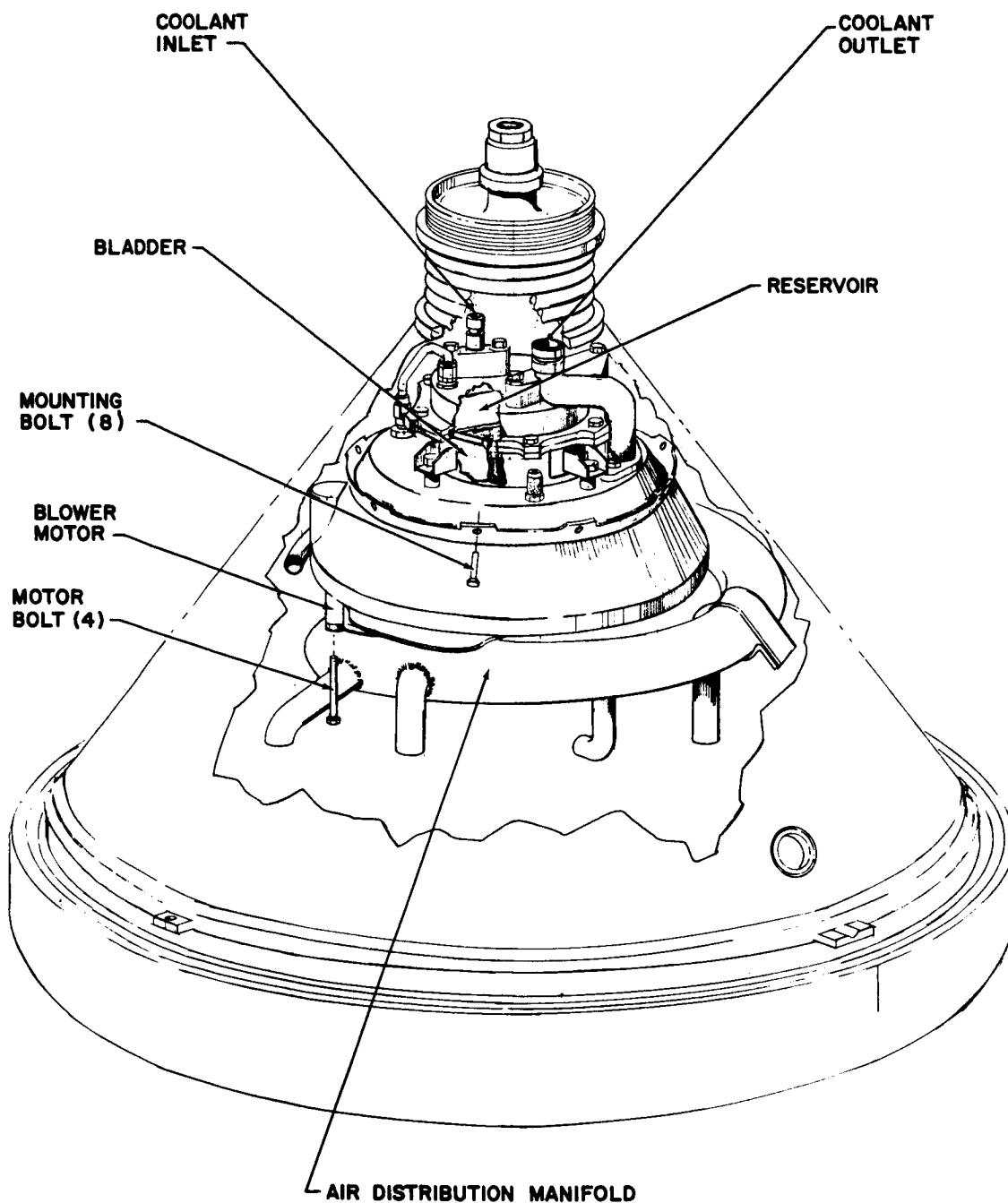
REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-9
INTEGRATED REPORT NO. GDC/BKF65-042
RAC REPORT NO. 499-51-II
DESCRIPTION

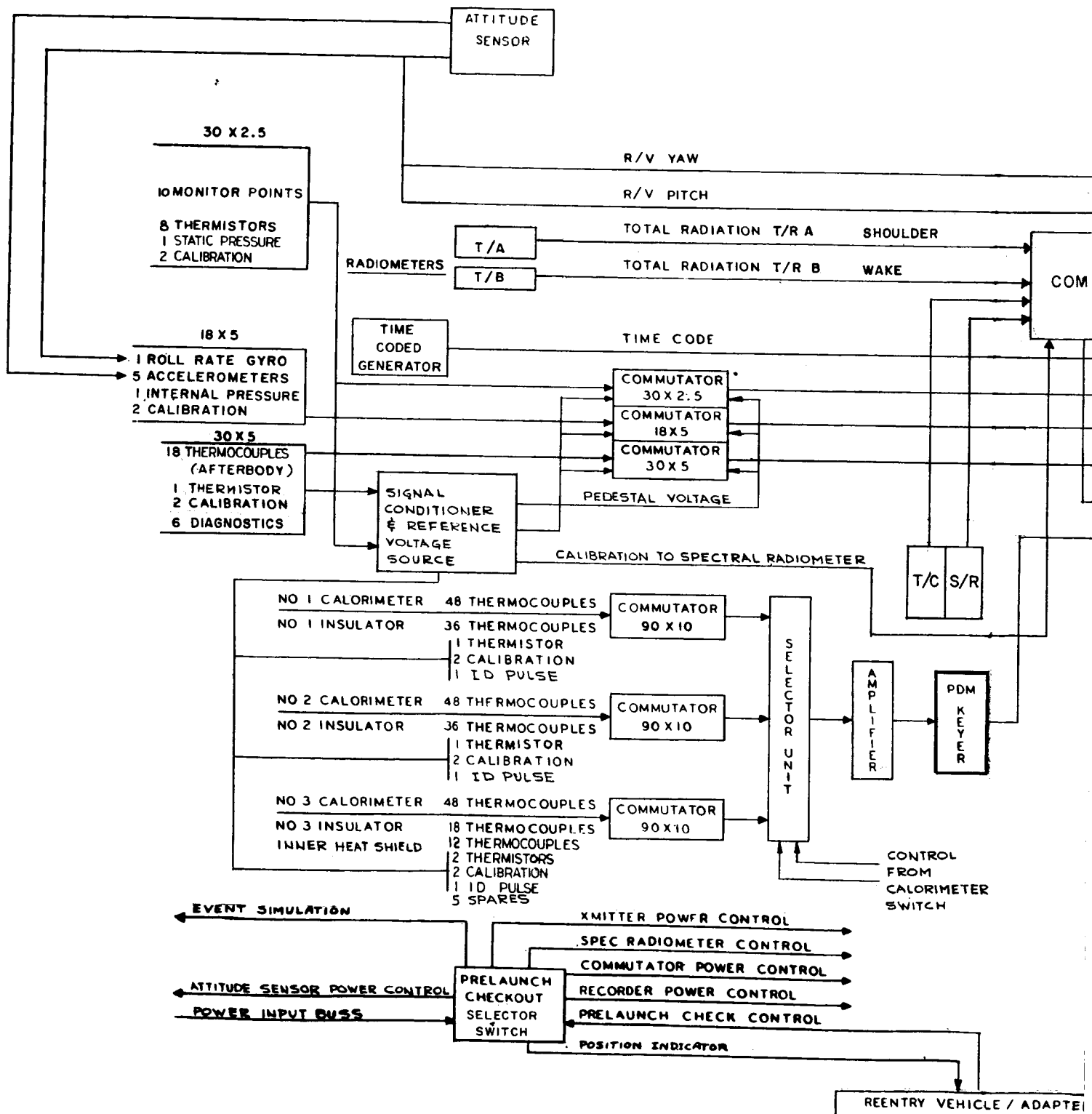
SCHEMATIC OF C-BAND BEACON SYSTEM



REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-10
INTEGRATED REPORT NO. GDC/BKF65-042
RAC REPORT NO. 499-51-II
DESCRIPTION

SCHEMATIC OF COOLING PACKAGE INSTALLATION

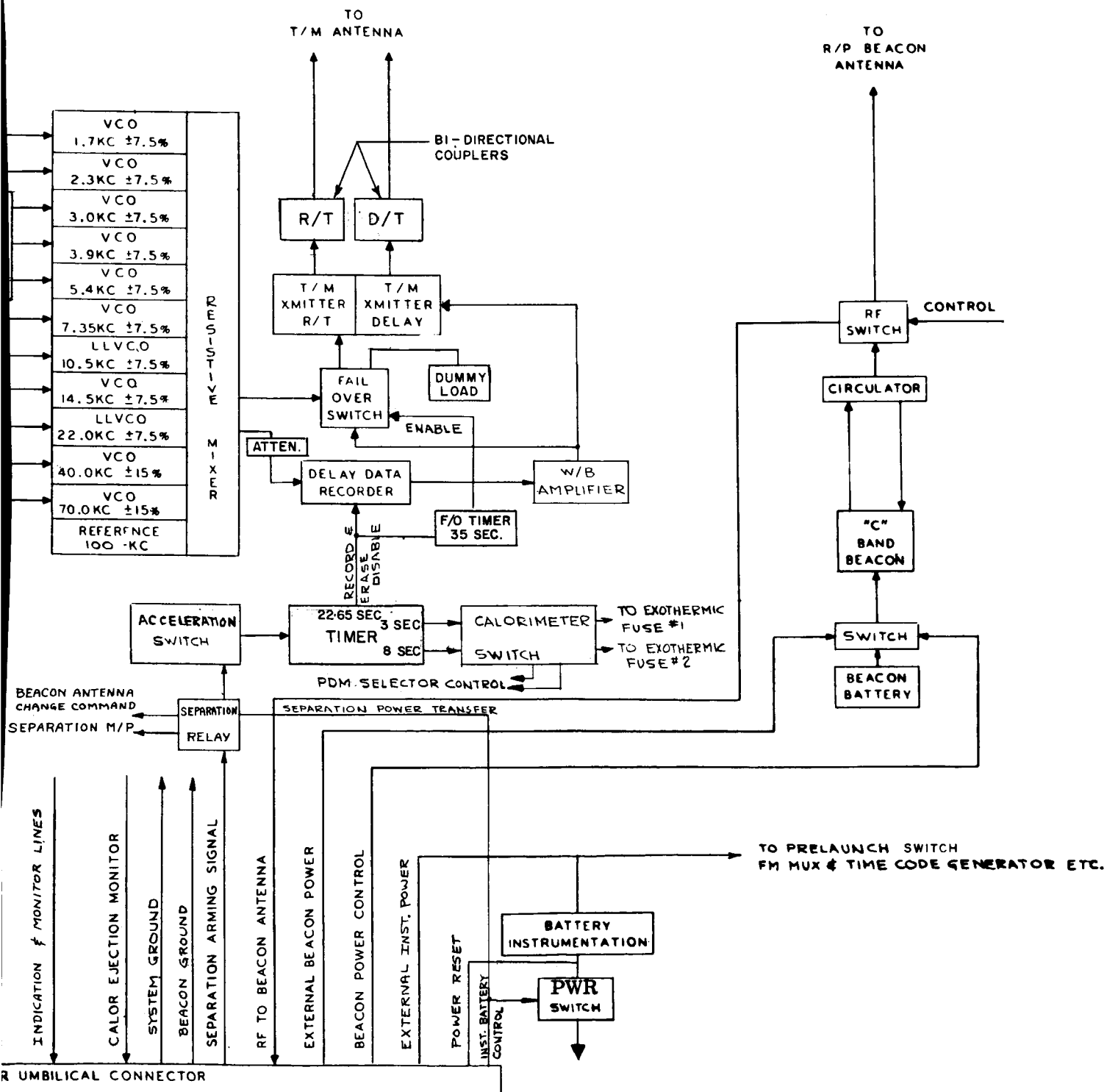




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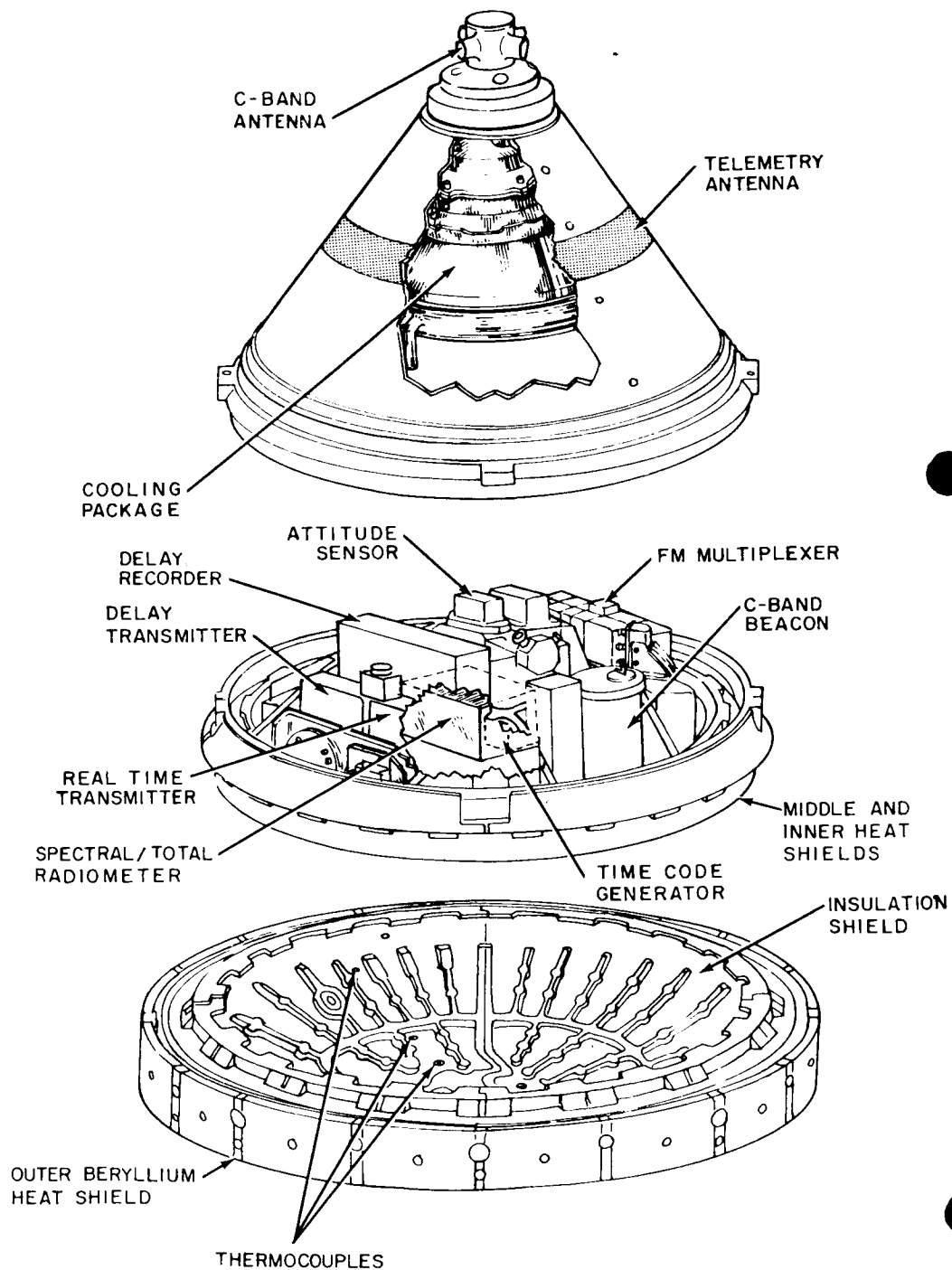
REENTRY PACKAGE PERFORMANCE
 FIGURE NO. 4-1-11
 INTEGRATED REPORT NO. GDC/BKF65-042
 RAC REPORT NO. 499-51-II
 DESCRIPTION

BLOCK DIAGRAM OF REENTRY PACKAGE INSTRUMENTATION



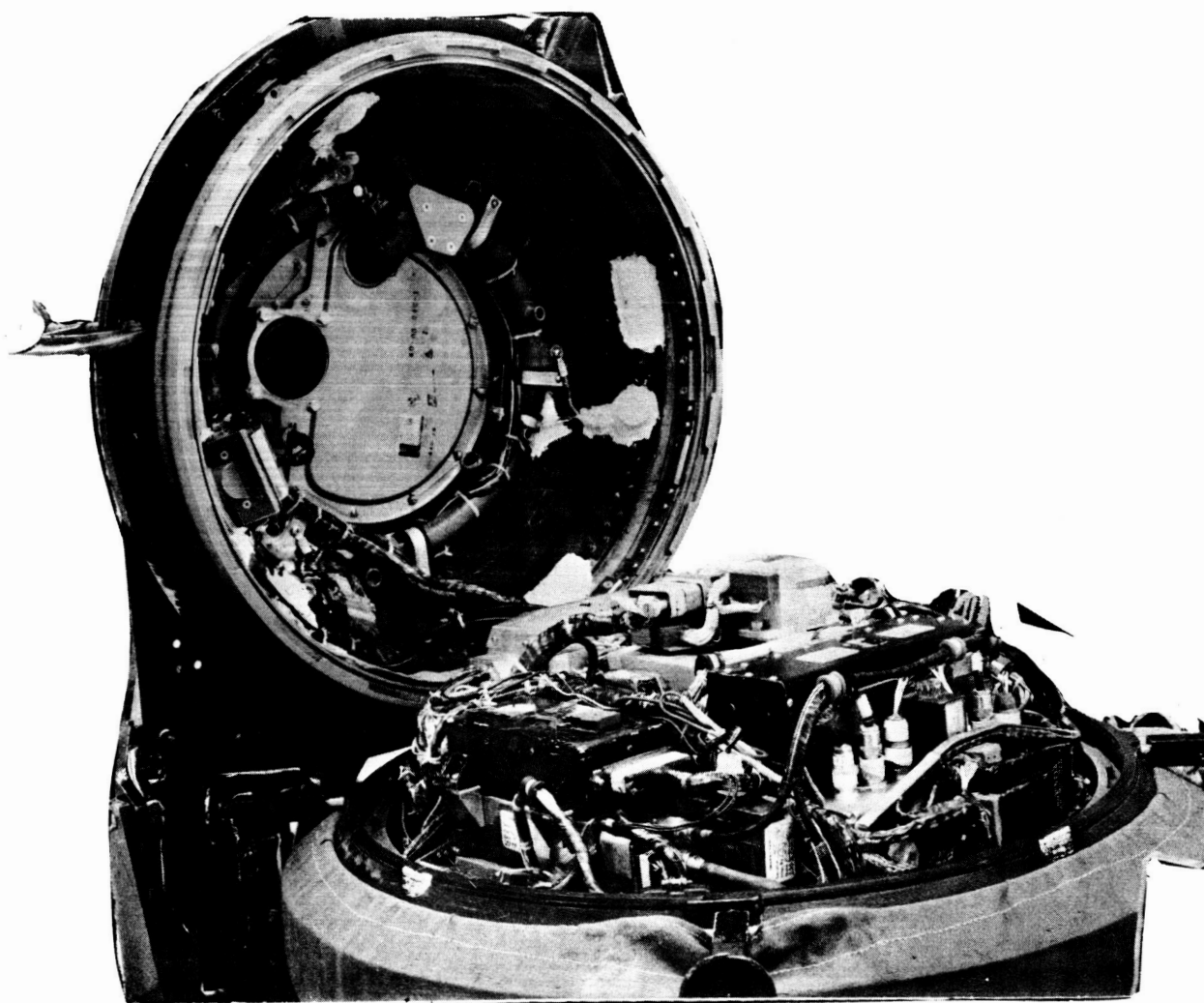
REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-12
INTEGRATED REPORT NO. GDC/BKF65-042
RAC REPORT NO. 499-51-II
DESCRIPTION

BUILD-UP OF THE REENTRY PACKAGE



REENTRY PACKAGE PERFORMANCE
FIGURE NO. 4-1-13
INTEGRATED REPORT NO. GDC/BKF65-042
RAC REPORT NO. 499-51-II
DESCRIPTION

REENTRY PACKAGE OPENED



REENTRY PACKAGE PERFORMANCE

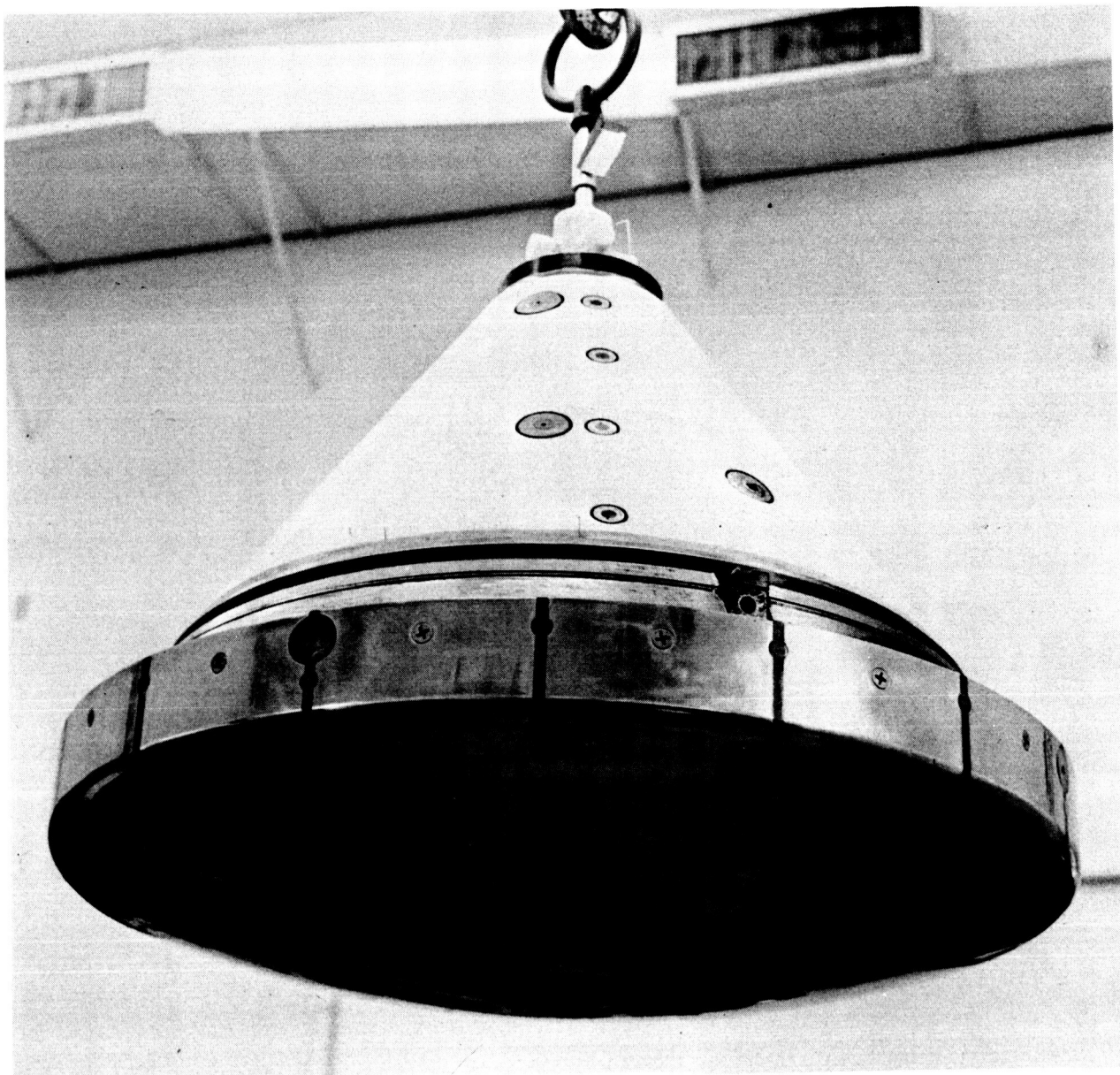
FIGURE NO. 4-1-14

INTEGRATED REPORT NO. GDC/BKF65-042

RAC REPORT NO. 499-51-II

DESCRIPTION

REENTRY PACKAGE IN THE REENTRY FLIGHT CONDITION



REENTRY PACKAGE PERFORMANCE
PAGE NO. 4-2-1
INTEGRATED REPORT NO. GDC/ BKF65-042
RAC REPORT NO. 499-51-II
ACCOMPLISHMENT OF FLIGHT OBJECTIVES

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.

A 100% data recovery was attained during the telemetry blackout. Minor deviations in performance that did occur are discussed later in this report.

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

Flight Objective 1 - Temperature data were obtained from approximately 80% of the thermocouples. However, thermocouple redundancy provided full data coverage.

Flight Objective 2 - All radiometers functioned throughout the flight.

Flight Objective 3 - The bi-directional couplers 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, the pitch rate gyro, and the roll rate gyro.

In summary, all primary flight objectives were met with excellent quality data that will allow completely automatic data processing as planned.

REENTRY PACKAGE PERFORMANCE
PAGE NO. 4-3-1
INTEGRATED REPORT NO. GDC/BKF65-042
RAC REPORT NO. 499-51-II
REENTRY PACKAGE FLIGHT SEQUENCE

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.

	<u>Event</u>	<u>Planned</u>	<u>Actual</u>
1.	Two-Inch Motion	T-0 sec	T-0 sec
2.	Start R/P Separation Timer	T+1538.6	T+1538.0
3.	V/P Spin-up	1545.6	1545.0
4.	Completion of Switchover to Internal Battery Power (V/P separation)	1548.6	1548.0
5.	X-259 Ignition	1551.7	1551.34
6.	X-259 Burnout	1584.4	1583.0
7.	R/P Separation from Adapter	1611.6	1610.43
8.	Start of T/M Blackout	1620.6	1624.7
9.	10g Reentry Deceleration Command	1639	1639.11
10.	Ejection of First Phenolic Heat Shield	1642	1642.12
11.	Ejection of Second Phenolic Heat Shield	1647	1647.53
12.	End of T/M Blackout	1652.6	1655.1
13.	Disable of Recorder Erase/Record	1661.6	1661.48
14.	Recorder Output Switched to Both Transmitters	1696.5	1696.11
15.	Splash	1940	1934.3
16.	Number of Delay Loop Cycles Played Back	3	4+

SECTION 4

PERFORMANCE DEVIATIONS

Reentry package operation throughout the flight was considered excellent with no significant performance deviations encountered. (A description of overall R/P systems performance is presented in Section 5.) Following is a brief summary of those deviations which were encountered prior to launch and during the actual flight.

Preflight Deviations

Although the R/P was in a "go" status at the time of launch, several anomalies were present which were not considered significant. These are listed below:

- Radiometer motor speed diagnostic intermittent - Not significant as motor speed could be determined from wave train period.
- Yaw accelerometer commutated pulses noisy - Noise would not preclude data reduction.
- Offset total radiometer telemetry zero loss - Telemetry zero occurred in two places in the wave train; the second telemetry zero was "solid."
- Afterbody total radiometer high bias - Would not preclude valid data reduction.
- Time code generator intermittent reset - Would not preclude correlation of real and delay time data.
- Beryllium thermocouple opens - Of the 48 thermocouples in each of the three beryllium shields, five in the first shield, and seven in each of the second and third shields were out at the time of launch. The number and the locations of the failed thermocouples were within the limits specified by launch condition criteria.

Inflight Deviations

The following deviations were encountered during flight:

- Delay recorder playback speed variation - The four playback loops varied in time up to 0.9 second for the fourth loop. However, this can be readily compensated for in the automatic data reduction program and will not preclude accurate time-correlation of data.

REENTRY PACKAGE PERFORMANCE
PAGE NO. 4-4-2
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RAC REPORT NO. 499-51-II
PERFORMANCE DEVIATIONS

- Beryllium thermocouple opens - In addition to the open thermocouples noted prior to launch, two opens in the first beryllium shield and four in each of the second and third shields were encountered during flight prior to the reentry period. These losses will not preclude attainment of the flight objective of total heating definition.
- R-F interference - Some internal R-F interference effects on the afterbody radiometer data were noted during blackout. However, these effects were minor and can be compensated for in the data analysis program.
- Signal noise - Some noise was encountered in the flight data tapes. However, this noise will not significantly affect the automatic data reduction program.

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. Following is a more detailed indication of performance by individual subsystems.

Power Supplies

The instrumentation battery was applied to the buss at T + 1538.0 (approximately the planned time). The battery performed well, providing power of about 26.3 vdc during reentry, which increased slowly to 26.8 vdc at the time of programmed failover (loss of real time modulation).

The beacon battery was turned on prior to launch. The battery performed well, maintaining its voltage above 28.5 vdc.

The heat shield and the separation and tumbling batteries worked well, as evidenced by the proper operation of the components powered by these batteries.

The auxiliary battery performed well throughout the flight, maintaining the buss voltage well above the minimum specification. The last voltage reading taken, prior to instrumentation battery transfer, was 26.8 vdc.

Data Transmission

Both real time and delay time transmitters functioned well throughout the flight, maintaining a good power level. The real time bi-directional coupler indicated 5 watts incident; the delay time bi-directional coupler indicated 3.4 watts incident. Also, the complete coverage of the reentry phase of the flight indicated proper R-F transmission.

The delay recorder functioned properly, disabling at the proper time and providing over four playbacks of reentry data.

The FM multiplexer functioned properly. All VCO's multiplexed data throughout the flight, with no evidence of frequency shift in the form of clipping or nonlinearities.

REENTRY PACKAGE PERFORMANCE
PAGE NO. 4-5-2
INTEGRATED REPORT NO. GDC/BKF65-042
RAC REPORT NO. 499-51-II
PERFORMANCE EVALUATION

Thermistors

All diagnostic thermistors functioned throughout the flight. All equipment temperatures during reentry were less than 95°F, indicating that turn-off of the blower upon instrumentation battery turn-on did not compromise the mission. The maximum temperatures encountered were on the delay time and real time transmitters, as expected.

Hastings-Raydist Pressure Transducer

The external pressure transducer functioned properly, saturating towards vacuum from just after the initial phase of flight until T + 1632. At T + 1632 the transducer indicated an increase in pressure, saturating at an indication of 20 mm of Hg at approximately T + 1643.

30 x 2.5 Diagnostics

All 30 x 2.5 diagnostics functioned properly throughout the flight.

30 x 5 Diagnostics

All 30 x 5 diagnostics functioned properly throughout the flight (with the exception of the radiometer motor speed diagnostic).

Radiometers

The radiometers all functioned well during the flight and complete data was obtained during the preplanned periods of interest. The resolution obtained on the spectral and total stagnation radiometers and the offset total radiometer was excellent; the maximum output recorded was 2 to 2-1/2 decades out of a 3-decade range.

Video information was lost for periods of 100 to 800 milliseconds during periods that coincided with the melting of the beryllium shields. This loss was apparently due to the shield material blocking the input light to the radiometer.

A slight downward shift occurred on the baseline of the afterbody radiometer coincident with blackout. Since blackout resulted in high VSWR on the real time and delay time transmitters, the shift in baseline was apparently caused by RFI. The shift can be compensated for and will not affect data analysis. The stagnation and offset total radiometers did not exhibit any apparent effect due to RFI.

Colvin Pressure Transducer

The Colvin pressure transducer operated properly throughout the flight. Minimum internal pressure monitored during reentry and just prior to programmed failover was approximately 10.4 lb/in.² (540 mm of Hg).

Attitude Sensor

All components of the attitude sensor system functioned properly to monitor the R/P body motions during the boost, coast, and reentry phases of the flight. Final values for the reentry phase will be determined during analysis of the results of the IBM data program. However, a preliminary analysis of oscillograph recordings indicates that the following values were achieved:

- Spin-up roll rate - 15.5 rad/sec
- Maximum roll rate - 16.6 rad/sec
- Maximum X-259 acceleration - 31.5 g
- Peak reentry g load - -83 g

In addition, the attitude sensor indicated that R/P-X-259 separation was smooth with no significant body motions induced. Slight R/P oscillations were encountered coincident with melting of the first and second beryllium shields. These oscillations were within expected orders of magnitude and did not affect attainment of flight objectives.

PDM Multicoder System

The PDM multicoder system performed satisfactorily with no anomalies encountered. Automatic commutator switching from 1 to 2 to 3 occurred on command at the time of phenolic heat shield ejection. PDM waveforms were excellent as all full scale, zero, and identification pulses were present.

Detailed data on thermocouple status during flight will be presented in the final data report. A brief discussion of the number of open thermocouples is presented in Section 4.

Cooling System

The R/P cooling system performed satisfactorily, as it reverted to the water cooling mode at lift-off and switched the blower motor off upon command at R/P separation. Maximum R/P component temperatures did not exceed 95°F.

C-Band Beacon

C-band beacon system performance was satisfactory. The beacon was interrogated through the adapter antenna prior to R/P separation and through the R/P apex antenna from separation to splash except for the blackout period.

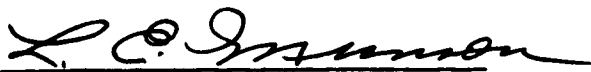
PART 5

ANTARES II A5 PERFORMANCE

GENERAL DYNAMICS CONVAIR

INTEGRATED REPORT NO. GDC/BKF65-042

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 re-entry stage from 20,705 feet per second to the desired re-entry 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. II.

SECTION 2

SUMMARY

The performance of the Antares II A5 solid rocket motor for the second 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 RE-entry Package provided a re-entry velocity which satisfied mission requirements.

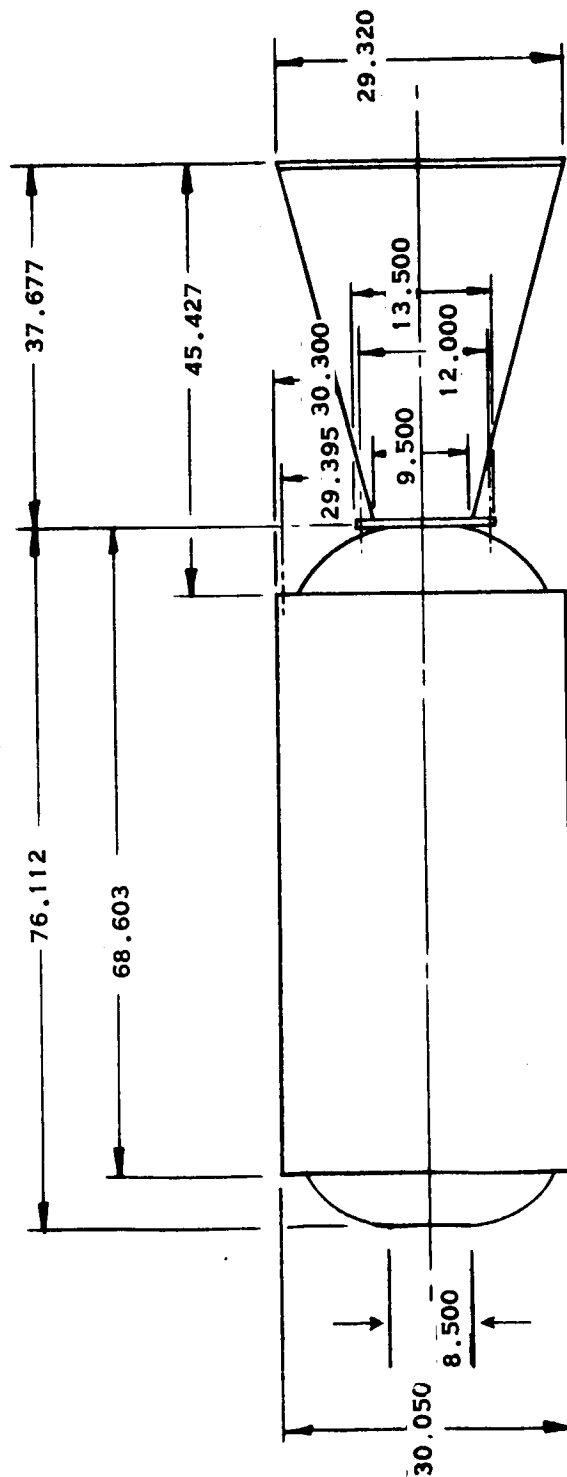
SECTION 3MOTOR DESCRIPTION

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 preignition weight (including the FIRE payload) was 3079 pounds, and the burnout weight was 492 pounds. The following weights were used in the derivation of the Antares II A5 performance:

R/P	191 lb
X259 Motor, Support and	
R/P Adapter	<u>301</u> lb
X259 Burnout	492 lb
X259 Expendable	<u>2587</u> lb
X259 Ignition	3079 lb

ANTARES PERFORMANCE
FIGURE NO. 5-3-2
INTEGRATED REPORT NO. GDC/BKF65-042
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 Re-entry Package. A zero to +45g accelerometer was mounted along the longitudinal axis (thrust axis), and three $\pm 6g$ accelerometers were mounted along the longitudinal, transverse, and normal axes. Data from these accelerometers were obtained by telemetry through a commutated channel in the Re-entry Package telemetry system which provided 5 data points per second. Radar tracking information was obtained from the Ascension island TPQ-18 radar.

PREFLIGHT PREDICTIONS

The predicted performance of the Antares II A5 motor was used to generate a preflight nominal trajectory. This predicted trajectory 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	984,206 Feet
Flight path angle	-15.283 Degrees
Velocity (earth relative)	20,705 Feet/Second
Pitch angle of attack	-5.860 Degrees

The parameters listed above (except angle of attack) are from the L/V (264D) post-flight trajectory, computed to the Antares ignition point. This trajectory also closely matches the TPQ-18 radar data from Ascension Island. The angle of attack at ignition was obtained after iterating to a V/P ignition attitude which produced time histories of altitude, latitude and longitude which match the radar data down to 400,000 feet altitude.

COMPARISON OF ACCELEROMETER AND RADAR DATA

Velocity data from the TPQ-18 radar on Ascension Island were plotted and differentiated to obtain a plot of acceleration versus time. Telemetered acceleration data, based on the 45-g accelerometer on board the R/P, were plotted on the same graph. A basic discrepancy was noted between these two acceleration curves in the areas of thrust build-up and thrust decay. It was assumed that the 45-g accelerometer data were correct since it more nearly described the expected performance of the Antares and because the techniques used to differentiate and smooth the raw radar data, from which the acceleration data were derived, were unknown.

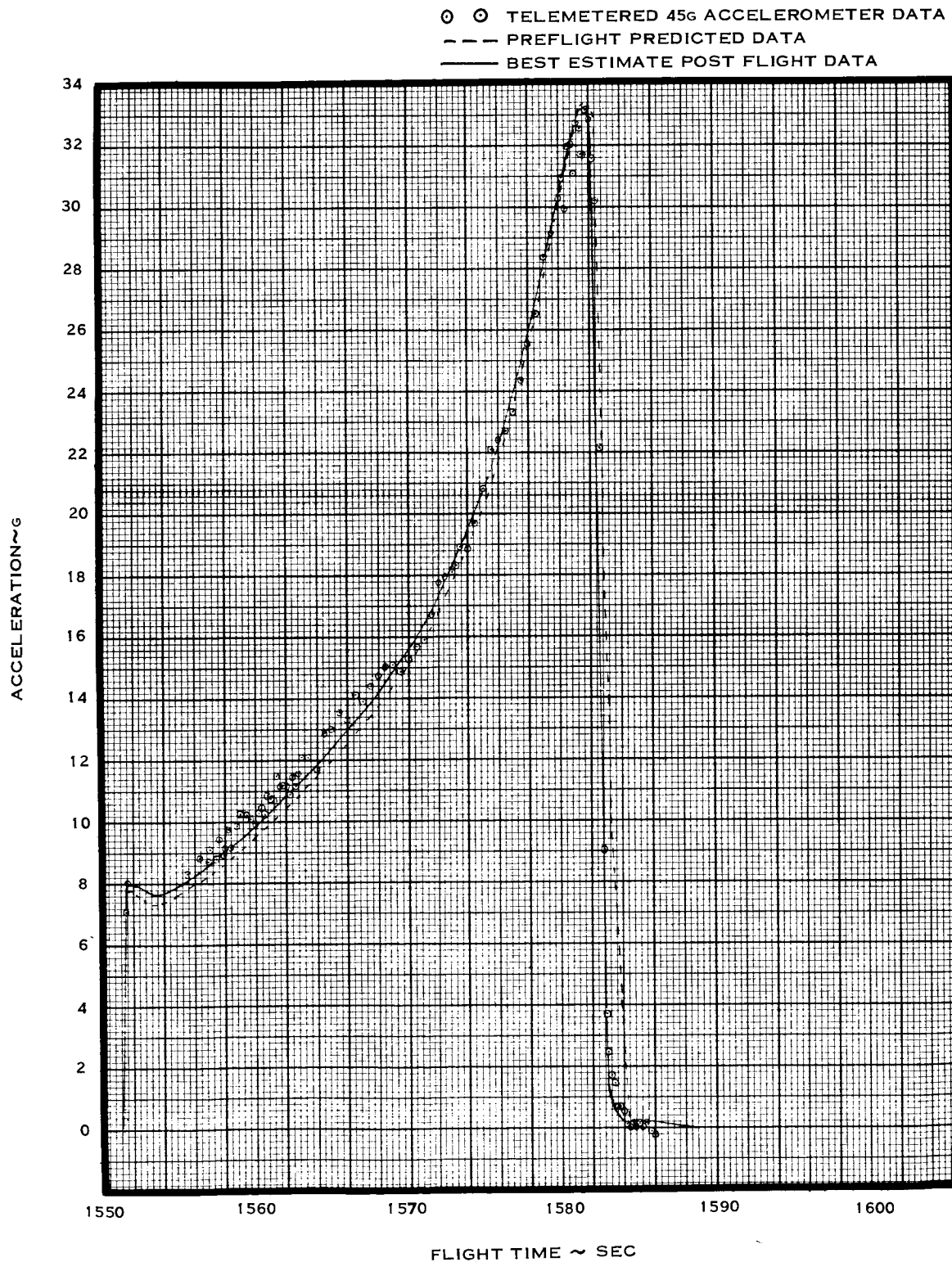
METHOD OF DERIVATION OF ANTARES PERFORMANCE PARAMETERS

Based on the plots described above, an acceleration versus time curve was constructed which closely matched the accelerometer data, and whose time integral matched the velocity gained according to TPQ-18 radar data. A time history of specific impulse, similar to preflight expected data, was assumed. Based on these, time histories of flow rate, weight and thrust were calculated. These data were tabulated and input to the trajectory simulation program. The resulting simulation required a multiplier on thrust to exactly match the velocity gained as shown by radar. It also required a reorientation in pitch at Antares ignition in order to match altitude versus time and latitude versus longitude versus time plots. The latter indicated an angular error of 0.778 degree (attitude vector too high) from planned at V/P ignition. Possible sources of this difference include the L/V attitude correction maneuver, gyro drift (L/V and V/P), V/P pitch program, V/P timer, and shift of the axis of coning during V/P spin stabilization.

Figure 5-5-3 presents a comparison of time histories of acceleration based on the pre-flight predicted performance, the telemetered 45-g accelerometer data, and the best estimate post flight trajectory. Figure 5-5-4 shows time histories of velocity based on the pre-flight predicted trajectory, the TPQ-18 radar data, and the best estimate post flight trajectory. These curves indicate that the Antares burned slightly faster than expected, with thrust decay occurring earlier in time.

ANTARES PERFORMANCE
FIGURE NO. 5-5-3
INTEGRATED REPORT NO. GDC/BKF65-042
METHOD OF PERFORMANCE EVALUATION

ANTARES AXIAL ACCELERATION DURING BURN



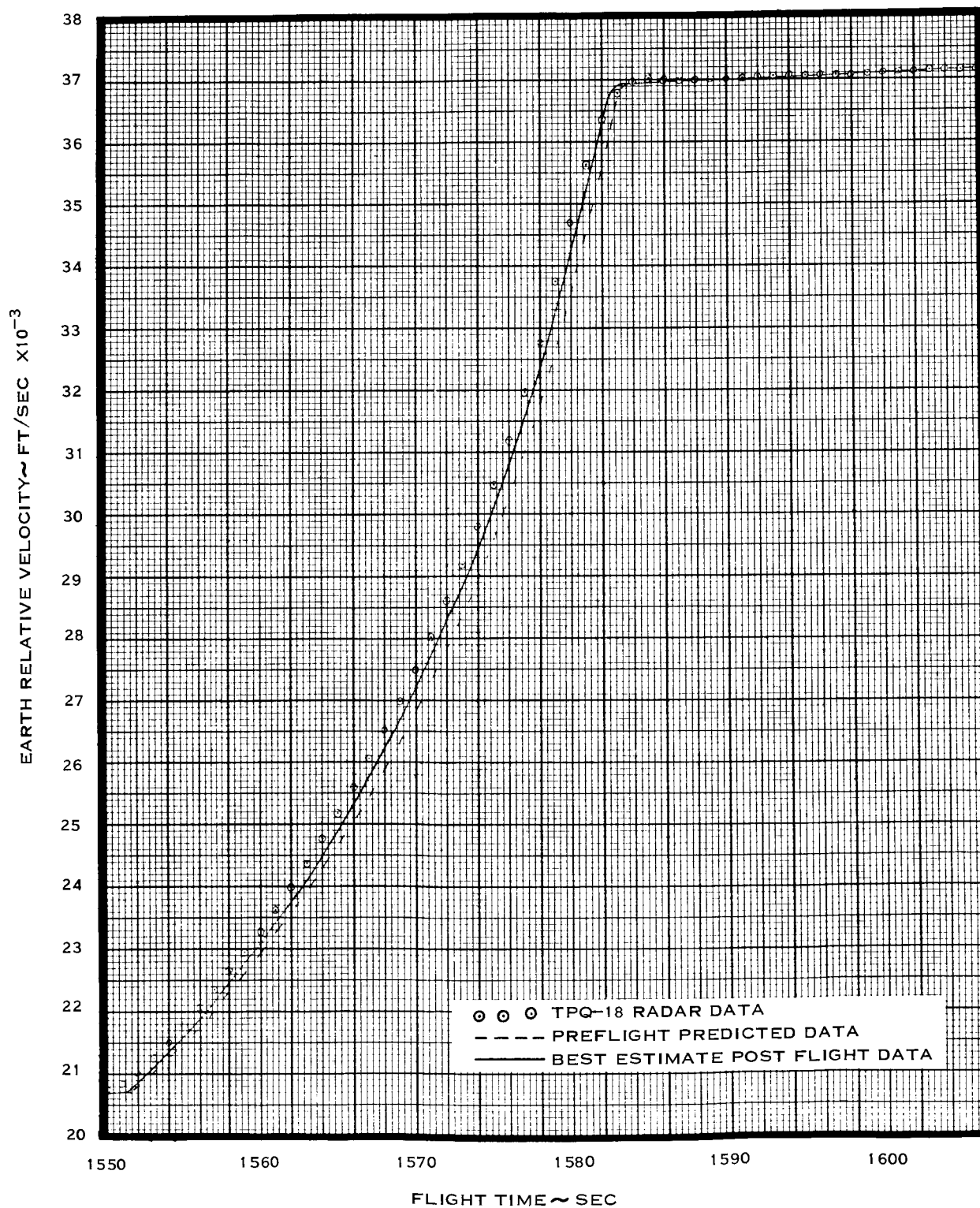
ANTARES PERFORMANCE

FIGURE NO. 5-5-4

INTEGRATED REPORT NO. GDC/BKF65-042

METHOD OF PERFORMANCE EVALUATION

COMPARISON OF BEST ESTIMATE AND RADAR VELOCITIES FOR ANTARES 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 713.956.9 pound-seconds as compared to an expected value of 714,171.7 pound-seconds. The consumed weight average specific impulse was determined to be 275.98 pounds of thrust per pound of mass per second.

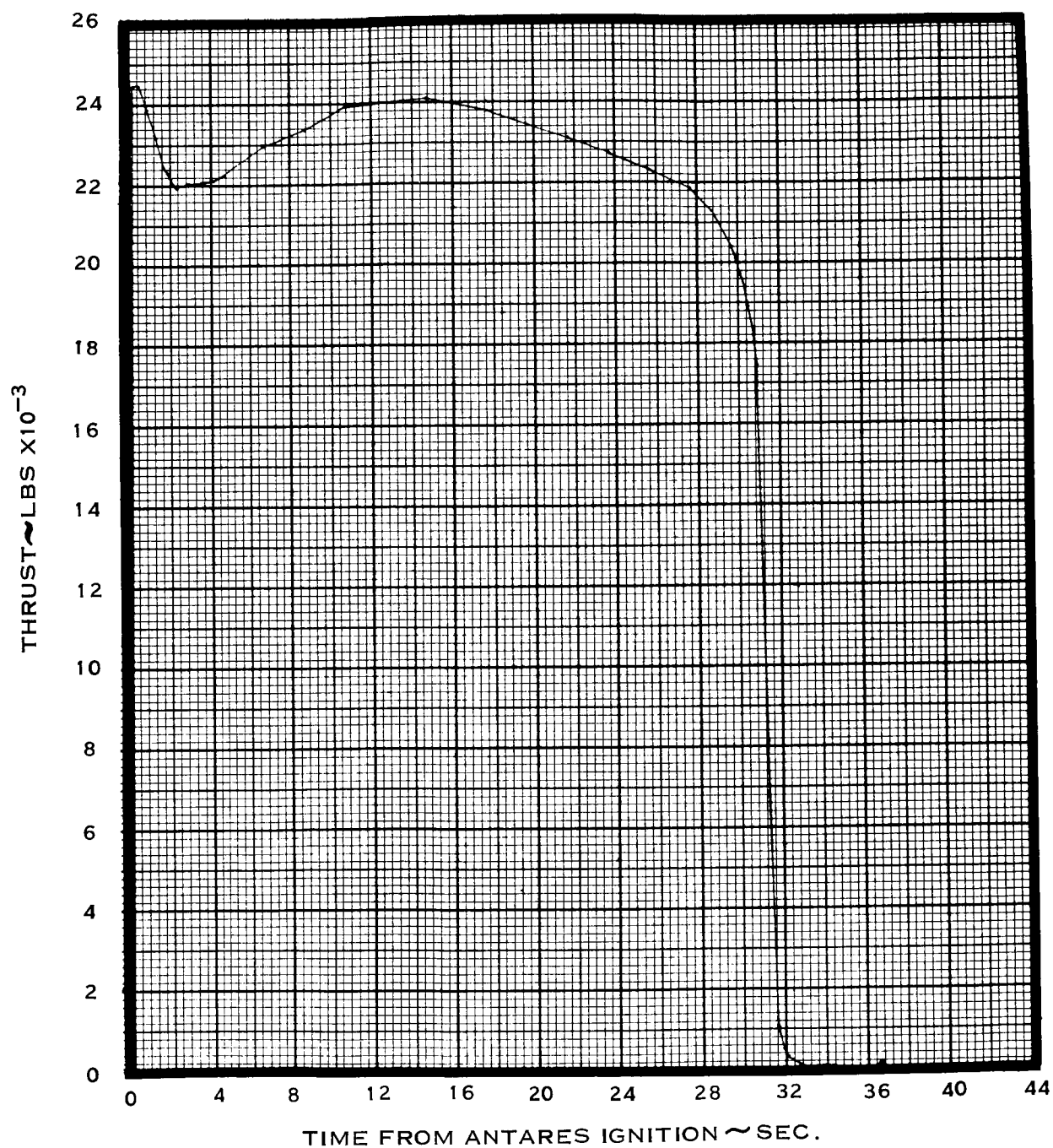
ANTARES PERFORMANCE

FIGURE NO. 5-6-2

INTEGRATED REPORT NO. GDC/BKF65-042

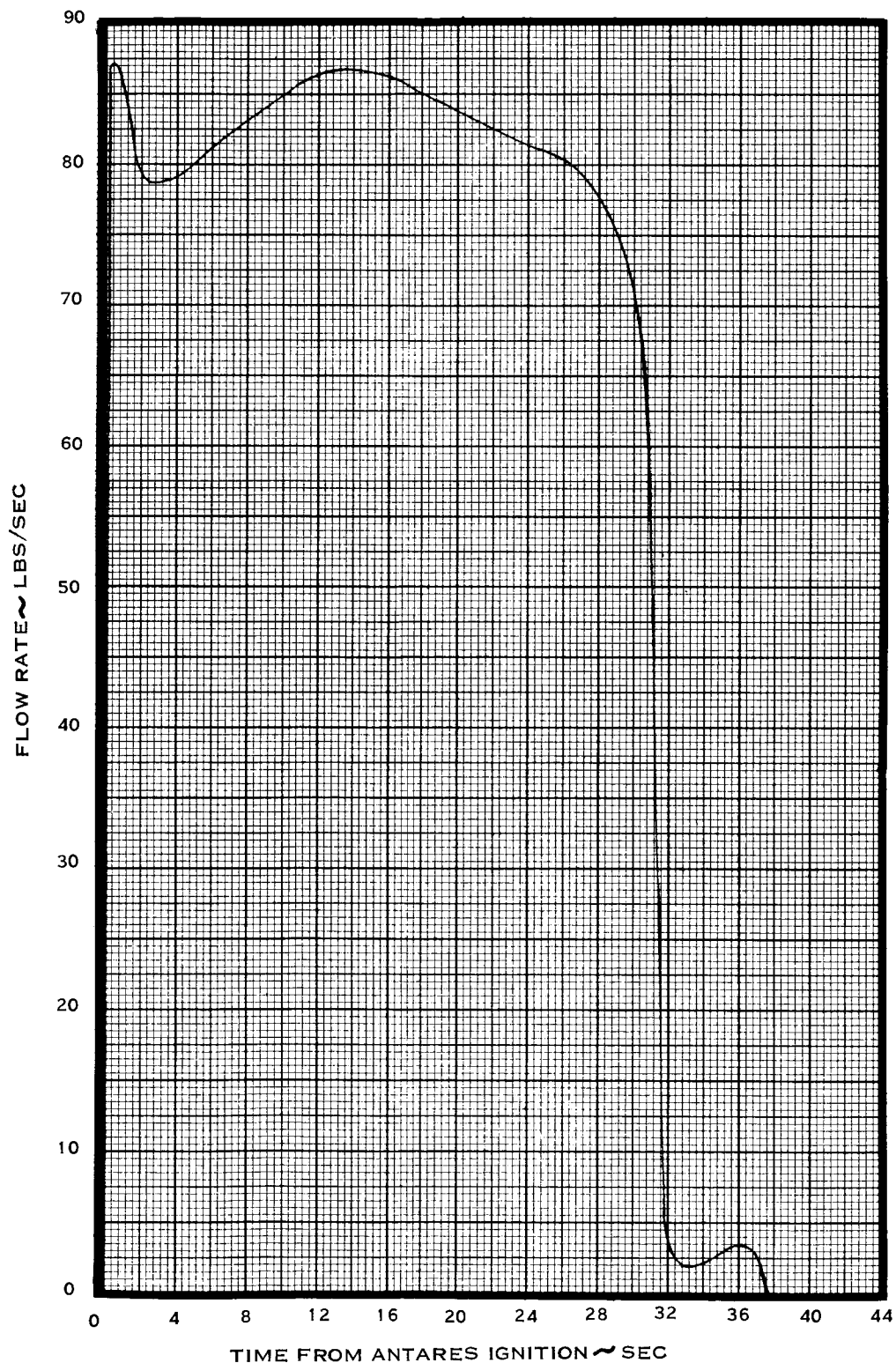
ANTARES II A5 PERFORMANCE

BEST ESTIMATE OF ANTARES THRUST



ANTARES PERFORMANCE
FIGURE NO. 5-6-3
INTEGRATED REPORT NO. GDC/BKF65-042
ANTARES II A5 PERFORMANCE

BEST ESTIMATE OF ANTARES WEIGHT FLOW RATE



ANTARES PERFORMANCE

PAGE NO. 5-6-4

INTEGRATED REPORT NO. GDC/BKF65-042

ANTARES II A5 PERFORMANCE

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	-
0.1	24418.5	87.0	280.7
0.5	24446.4	87.1	280.7
0.9	23866.2	86.1	277.2
1.3	23198.9	83.9	276.5
1.7	22451.9	80.5	278.9
2.3	21942.5	78.9	278.1
2.9	22013.2	78.7	279.7
3.5	22068.5	78.9	279.7
4.1	22108.8	79.2	279.2
4.7	22268.8	79.8	279.1
6.7	22974.6	81.9	280.5
8.7	23379.2	83.8	279.0
10.7	23924.9	85.5	279.8
14.7	24087.3	86.6	278.1
17.7	23791.8	85.2	279.2
19.7	23427.1	84.0	278.9
21.7	23101.3	82.7	279.3
23.7	22736.4	81.6	278.6
25.7	22297.1	80.5	277.0
27.7	21860.4	78.3	279.2
28.7	21299.2	76.0	280.2
29.7	20403.2	72.5	281.4
29.9	20083.1	71.3	281.7
30.1	19704.3	70.1	281.1
30.3	19384.3	68.9	281.3
30.5	18904.0	67.5	279.9
30.7	18293.8	65.8	277.8
30.9	17436.4	63.6	274.0
31.3	8130.8	33.5	242.7
31.7	1023.3	9.9	103.4
31.9	510.4	4.5	113.4
32.1	305.7	3.3	92.6
32.3	203.6	2.8	72.7
32.7	101.6	2.1	48.4
33.1	50.7	1.9	26.7
33.5	65.8	2.0	32.9
33.9	80.9	2.1	38.5
34.3	50.5	2.3	22.0
37.5	0	0	-

Average I_{sp} = 275.98

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

Antares Burn Time (sec)	Weight (lbs)	Consumed Weight (lbs)	Cumulative Impulse (lb - sec)
Ignition	3079.00	0	0
.96	2998.11	80.89	22,082.4
2.96	2837.21	241.79	67,057.6
4.96	2678.56	400.44	111,330.5
6.96	2516.01	562.99	156,754.2
8.96	2349.54	729.46	203,216.8
10.96	2179.60	899.40	250,656.3
12.96	2006.60	1072.40	298,608.6
14.96	1833.71	1245.29	346,718.4
16.96	1661.61	1417.39	394,644.1
18.96	1491.26	1587.74	442,109.0
20.96	1323.43	1755.57	488,882.8
22.96	1157.94	1921.06	534,984.0
24.96	995.00	2084.00	580,331.0
26.96	834.82	2244.18	624,810.7
28.96	679.19	2399.81	668,131.3
30.96	538.53	2540.47	707,520.4
32.96	504.41	2574.59	713,789.3
34.96	499.73	2579.27	713,906.3
36.96	493.64	2585.36	713,954.7
37.50	492.00	2587.00	713,956.9

PART 6
VELOCITY PACKAGE PERFORMANCE

GENERAL DYNAMICS/ASTRONAUTICS
INTEGRATED REPORT NO. GDC/BKF65-042
LTV/A REPORT NO. 3-30000/5R-30

APPROVED BY: *CJ Bitter*
C. J. BITTER
PROJECT MANAGER - PROJECT FIRE

SECTION 1

INTRODUCTION

The Project FIRE Flight 2 vehicle was successfully launched by the National Aeronautics and Space Administration from Cape Kennedy, Florida, at approximately 1655 EST on 22 May 1965. This was the second 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 the LTV Astronautics Division of LTV Aerospace Corporation (LTV/A)** and a Reentry Package (R/P) manufactured by Republic Aviation Corporation (RAC), into a ballistic trajectory along the Air Force Eastern Test 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 Company, 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.

VELOCITY PACKAGE PERFORMANCE

PAGE NO. 6-1-2

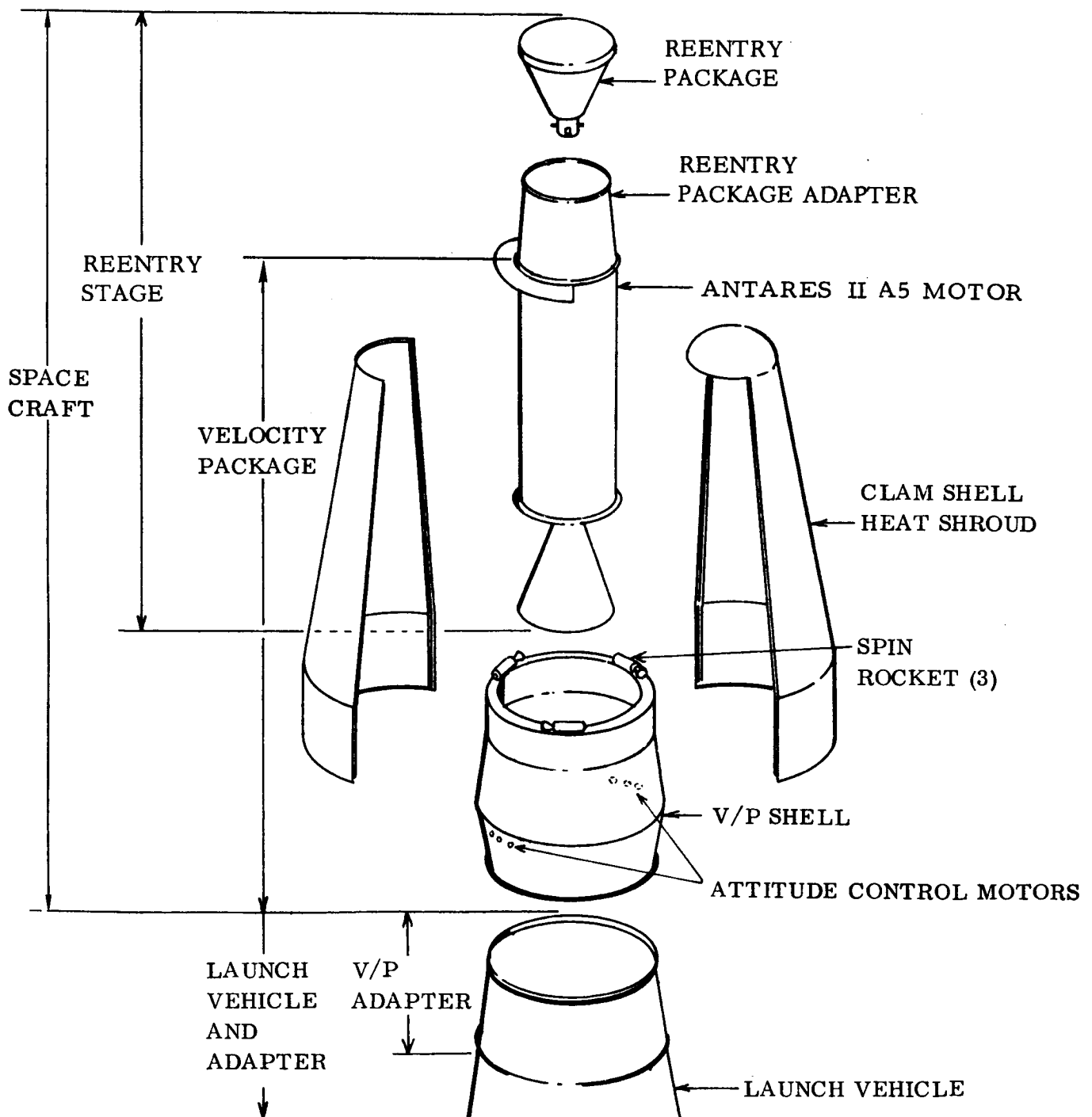
INTEGRATED REPORT NO. GDC/BKF65-042

LTV/A REPORT NO. 3-30000/5R-30

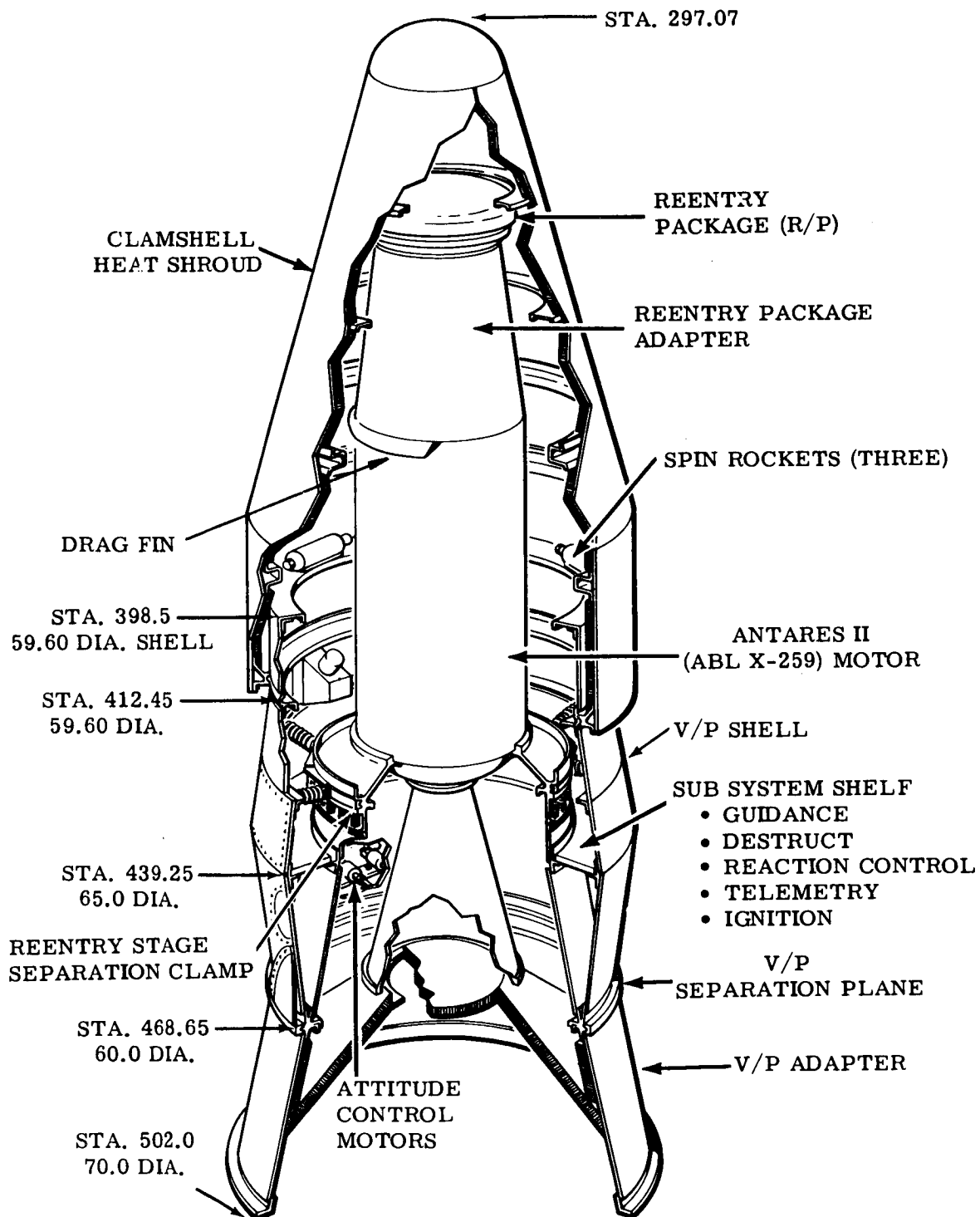
INTRODUCTION

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 2 as related to the Velocity Package only. The flight trajectory evaluation, the vibrometer analysis, and the ANTARES II A5 motor performance evaluation are not included in Part 6.

MAJOR COMPONENTS

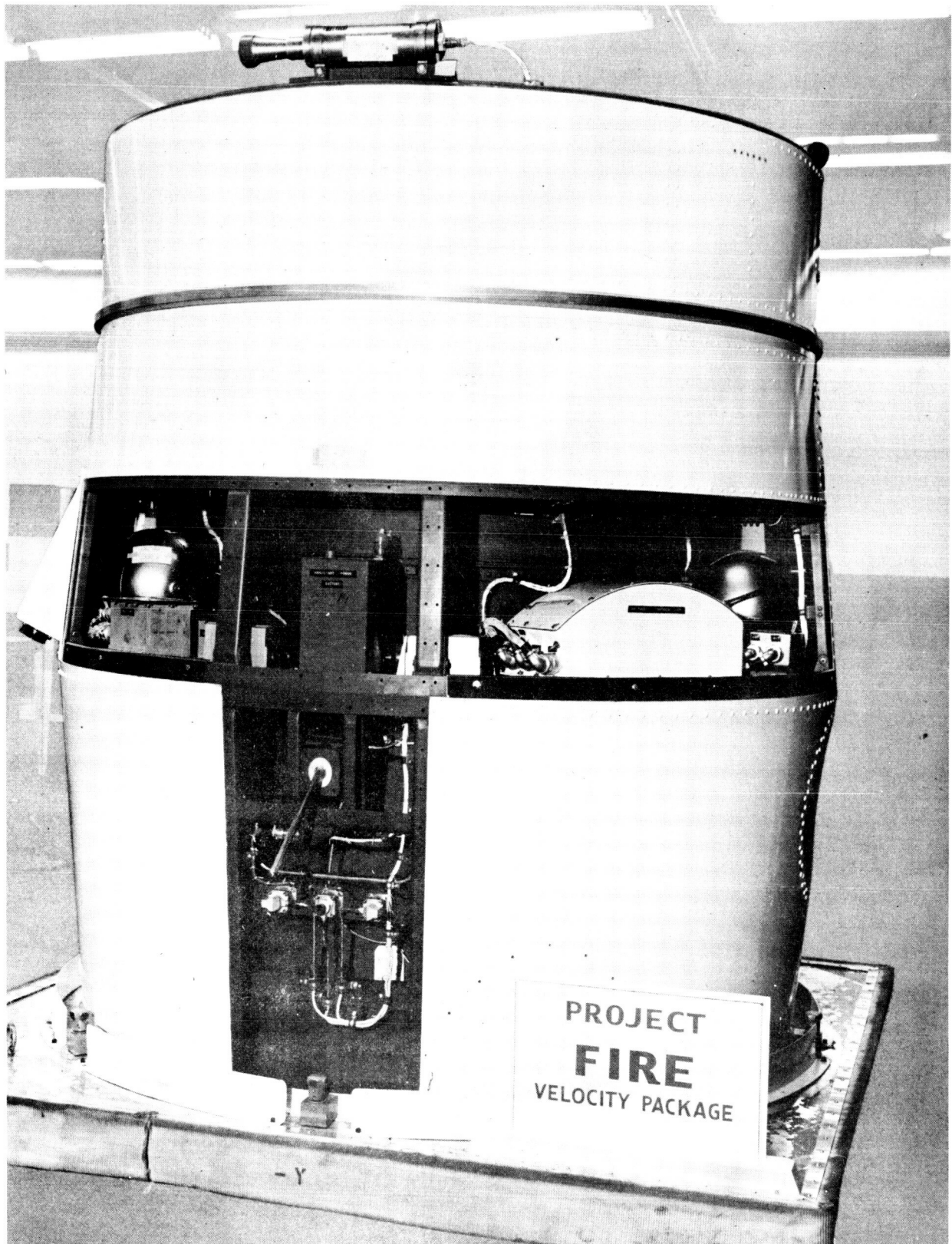


SPACECRAFT



VELOCITY PACKAGE PERFORMANCE
FIGURE NO. 6-1-5
INTEGRATED REPORT NO. GDC/BKF65-042
LTV/A REPORT NO. 3-30000/5R-30
INTRODUCTION

VELOCITY PACKAGE SHELL ASSEMBLY



VELOCITY PACKAGE PERFORMANCE

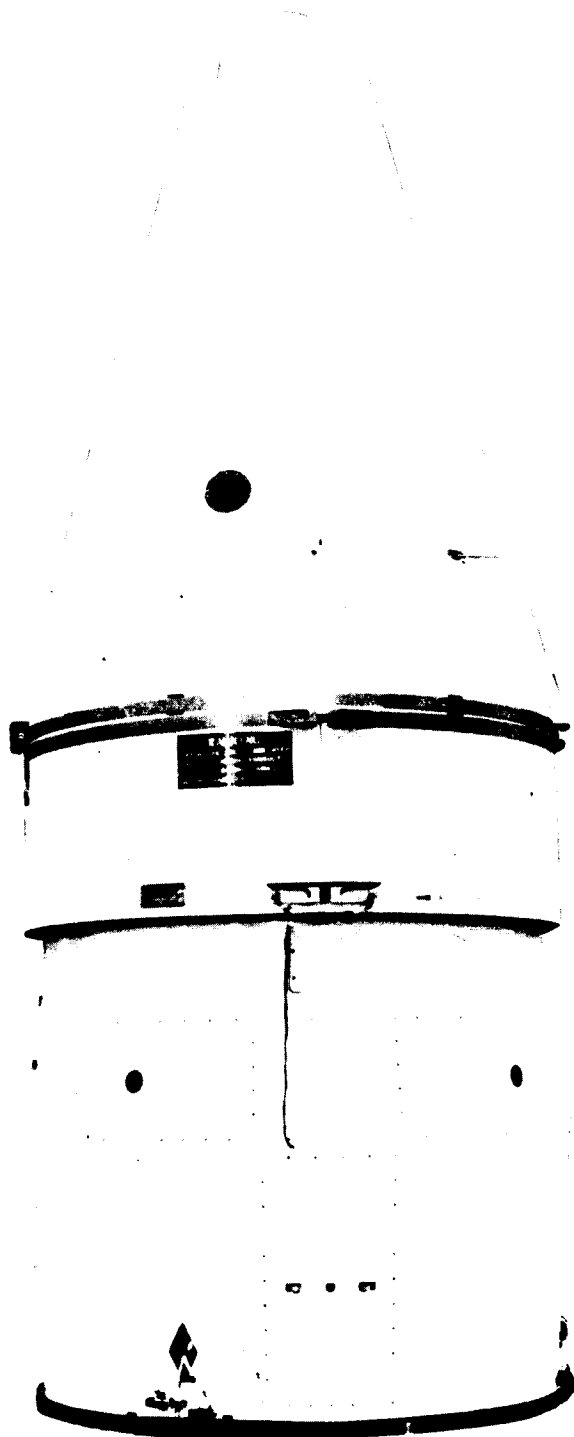
FIGURE NO. 6-1-6

INTEGRATED REPORT NO. GDC/BKF65-042

LTV/A REPORT NO. 3-30000/5R-30

INTRODUCTION

VELOCITY PACKAGE



SECTION 2

SUMMARY

The Project FIRE Flight 2 vehicle (ETR Test 0501) was successfully launched from ETR Complex 12, Cape Kennedy, Florida, at 1654: 59.703 EST on 22 May 1965. 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	REMARKS
Determine V/P interlock activation	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.
Determine that the V/P timer starts	The V/P timer was started at the proper time by the L/V timer start discrete signal.
Determine that the V/P gyros uncage	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.
Determine that the V/P guidance and control system functions	The V/P guidance and control system stabilized the spacecraft and oriented it to the correct attitude for reentry.

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SUMMARY

FLIGHT OBJECTIVE

REMARKS

Determine control system unregulated pressure

The unregulated nitrogen pressure remained at a constant pressure (2990 psia) during boost. During the coast phase, the nitrogen pressure decreased approximately 190 psi.

Determine control system regulated pressure

A nitrogen regulated pressure of 345 psia was maintained during the flight.

Determine that the heat shroud separates

A clean heat shroud separation occurred at the proper time. The L/V provided a backup signal, which was not required on this flight.

Determine that the V/P separates from the L/V

A clean L/V-V/P separation occurred at the proper time as a result of receipt of the back-up signal rather than the L/V discrete signal which occurred approximately 1.7 seconds later.

Determine that the spin motors ignite and function properly

All three spin motor nozzle temperatures increased suddenly over 250°F thus verifying that all three spin motors had ignited. An initial spin rate of 158.6 rpm was achieved in 0.5 second.

Determine that the R/S separates

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.

Determine that the ANTARES motor ignites

The R/P telemetry data verified that the ANTARES motor ignited at the proper time.

Determine adequacy of the thermal protection system

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

FLIGHT OBJECTIVE

REMARKS

Determine structural capability of the V/P

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.

NOTE: The last two items are considered secondary and tertiary objectives, respectively; all others are considered primary objectives for the Velocity Package.

All V/P events occurred within allowable limits of their expected times. The events determined by the V/P guidance timer occurred within 0.014% of the expected times. 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 V/P sequence of events is presented in the following table.

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SUMMARY

V/P SEQUENCE OF EVENTS

EVENT	SIGNAL SOURCE	NOMINAL TIME, SEC.	ACTUAL TIME, SEC.
V/P Timer Start Discrete	L/V	294.83	294.40
V/P Timer Start Indication	V/P	294.93	294.50
V/P Shroud Jettison	L/V	295.50	295.29
Uncage V/P Gyros and V/P Timer Start Backup	L/V	302.87	304.64
L/V-V/P Separation	L/V	308.37	308.73
Start Pitch Program	V/P	335.33	334.89
End Pitch Program	V/P	435.50	435.05
R/P Separation Timer Start Signal	V/P	1538.63	1538.05
Spin Motor Ignition	V/P	1545.63	1545.04
V/P-R/S Separation	V/P	1548.63	1548.03
ANTARES Ignition	V/P	1551.63	1551.34

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.

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SUMMARY

The V/P telemetry system operated satisfactorily and data were recovered from all telemetry functions for the entire flight, except for a short period between T+1097 and T+1220 seconds during which intermittent drop-out occurred. This anomaly was probably due to atmospheric conditions and/or marginal ground receiving equipment capability. The V/P telemetry system ceased transmitting as programmed at T+1548 seconds when the V/P separated from the Reentry Stage.

Accuracy of the attitude reference and the programmer was not independently determinable, however, the reentry angle error of approximately 0.6 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 158.6 rpm compared to a predicted rate of 158.5 ± 11 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, yaw and roll 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 1.6 deg/sec pitch up, well within the predicted limits. The V/P separated from the Reentry Stage at the proper time and within the limitations of the R/P instrumentation no coning could be detected.

SECTION 3

TELEMETRY SYSTEM ANALYSIS

The V/P telemetry system operated satisfactorily and data were recovered from all telemetry functions for the entire flight except for a short period between T+1097 and T+1220 seconds during which intermittent drop-out occurred. Ground Station 1 (TEL-3 located at Kennedy Space Center) received useable data through T+733 seconds. Overlapping coverage was obtained between Station 1 and Station 9.1 (located at Antigua). Overlapping coverage was also obtained between Station 9.1 and Station 12 (Ascension Island); however, intermittent drop-out occurred at the end of the Station 9.1 tape and at the beginning of the Station 12 tape. This anomaly was probably due to atmospheric conditions and/or marginal ground receiving equipment capability.

The V/P telemetry system ceased transmitting as programmed at T+1548 seconds when the V/P separated from the Reentry Stage.

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

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TELEMETRY

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 Torque Rate	-2.5 to 1.5 deg/sec
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	

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VELOCITY PACKAGE
TELEMETRY INSTRUMENTATION SUMMARY

IRIG CHANNEL	SCO FREQUENCY KC	COMMUTATOR SEGMENT	MEASUREMENT CODE	MEASUREMENT	NOMINAL MEASUREMENT RANGE
10	5.40	N/A	(V-4)	Lower Left Valve	On-Off
			(PS-3)	Lower Right Pressure	
			(PS-4)	Lower Left Pressure	
			M-3	Pitch Matrix	
			(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	± 10 deg/sec
12	10.50	N/A	GS-2	Pitch Rate	± 10 deg/sec
13	14.50	N/A	GS-1	Roll Rate	± 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

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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 Nozzle Temperature	0-350°F
15	30.00	13	T-5	Spin Motor Nozzle Temperature	0-350°F
15	30.00	14	T-6	Spin Motor Nozzle 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 2 (X-axis-Lateral)	± 15 "g"
17	52.80	N/A	A-5	Vibration System 3 (Y-axis-Normal)	± 15 "g"
18	70.00	N/A	A-6	Vibration System 1 (Z-axis-Long.)	± 15 "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 0.014% 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, the normal control sequence for the roll-yaw motor was modified by the payload water boiler cooling system as in Flight 1. Allowances and expenditures for normal operations are summarized in the following table:

<u>MISSION PHASE</u>	<u>ALLOWANCE IMPULSE, POUND-SEC.</u>	<u>FLIGHT (CALCULATED) IMPULSE, POUND-SEC.</u>
Capture	125.0	11.7
Pitch Program	7.0	7.1
Coast	128.8	66.1
Contingency	339.2	—

The actual capture maneuver was mild, with the maximum attitude error approximately 0.7 degree and the induced rate at capture not exceeding 1.6 degrees 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.

During the period between 450 and 550 seconds after launch the duty cycle for the pitch down motor was 0.16%, the upper right motor was 0.54%, and the lower right motor was

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GUIDANCE

0.49%. To the extent that this is representative of the coast average, the total impulse expenditure would be 66.1 lb-sec.

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 ($\cos 23^\circ$) about the yaw axis, and a small translational velocity increment. The general characteristics of the pitch limit cycle indicate that the probable primary source of the reduced fuel consumption was asymmetrical 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 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	57.2 lb-sec.
Mission Time (active control), t_m	1236 sec.

Assuming that additional moment sources were not present, the yaw angular impulse provided by the roll-yaw jets was approximately 265 ft lb-sec., which should equal the payload cooling system angular impulse. It is further assumed that the output from the cooling system was constant; then for the 1236-second mission the yaw component of cooling system thrust (T_c) would be:

$$\begin{aligned} T_c &= l_y I_y / t_m l_c \\ &= 0.056 \text{ lbs} \end{aligned}$$

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 0.6 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. Comparisons of valve command to chamber pressure switch closures, show that response times of the motors were low. Flight data indicates the valve response times were less than 15 milliseconds which is well within the performance requirements of the system. There is no positive indication that the left roll motors were commanded to fire at any time during the mission.

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.

Reaction Controls Nitrogen Consumption

The amount of nitrogen consumed during the flight was calculated to be 0.84 pounds. This represents approximately eight percent of the 11 pounds of nitrogen available. The consumed weight was calculated from the initial and final unregulated nitrogen tank pressure

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and assuming an isothermal process. A plot of the unregulated nitrogen pressure versus flight time is presented in Figure 6-4-5. There was no significant leakage from the system during pre-launch activities or during flight.

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 the low instrumentation sensitivity, displacement rates could not be determined during deadband cycling. Therefore, the only useable data for determining the thrust levels was during the pitch maneuver when a known displacement rate existed. 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 below. The thrust levels and system regulated pressure during flight agree closely with the predicted.

<u>MOTOR</u>	<u>CONDITION</u>	<u>THRUST- POUNDS</u>
Pitch Up	Flight (Calc.)	5.02
	Predicted	5.20
Pitch Down	Flight (Calc.)	4.75
	Predicted	5.09

The regulated system pressure (no flow) plotted versus flight time in Figure 6-4-5, averages about 345 psia. This compares favorably with the predicted pressure of 349 psia.

VELOCITY PACKAGE PERFORMANCE

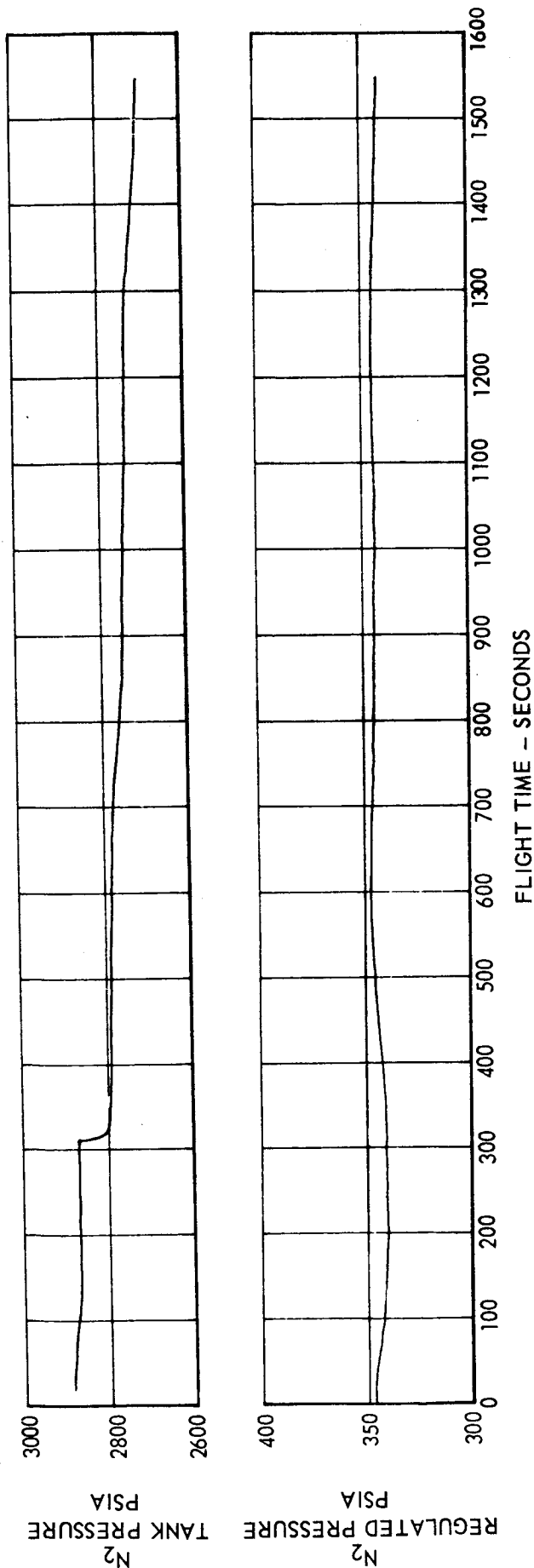
FIGURE NO. 6-4-5

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NITROGEN PRESSURES



SECTION 5

PYROTECHNICS ANALYSIS

The ANTARES II A5 rocket motor ignited 6.30 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 158.6 rpm. An immediate increase in the spin motor nozzle 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.

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 from the L/V programmer, except for the L/V-V/P separation which occurred from receipt of the back-up signal. This would confirm that V/P ignition systems No. 1 and No. 2 functioned properly since No. 1 operates from L/V guidance discrete signals and system No. 2 operates from L/V programmer 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 both 31.0 volts.

Destruct System

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

SECTION 7

STRUCTURAL SYSTEMS ANALYSIS

General

The V/P did not have any specific instrumentation to verify structural integrity. However, as with Flight 1, the success of the flight 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)	239.27	479.3	100.2	100.2	50.4	30.9	29.9
V/P Shell and Dynamic Balance Weights	789.33	435.5	100.0	100.0	124.7	111.2	109.6
ANTARES Ring Adapter and Dynamic Balance Weights	28.78	424.3	100.0	100.0	1.86	0.94	0.94
V/P Heat Shroud (S/N 00002)	294.68	368.4	100.9	99.9	42.5	88.9	83.1

Heat Shroud Separation

Heat shroud separation appeared to be clean with minor disturbances being noted on all three vibrometer traces at the time the separation bolts were fired; these disturbances

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decayed in approximately 0.1 second. Similar disturbances were also noted on the roll, pitch, and yaw rate traces and are considered normal. These disturbances are very similar to those noted on Flight 1.

V/P Separation From the V/P-Adapter

Separation of the Velocity Package from the Launch Vehicle was very clean with virtually no tip-off effect. The maximum angular rate imparted to the V/P was 1.6 deg/sec pitch up. No disturbances were noted on the yaw or roll rate traces.

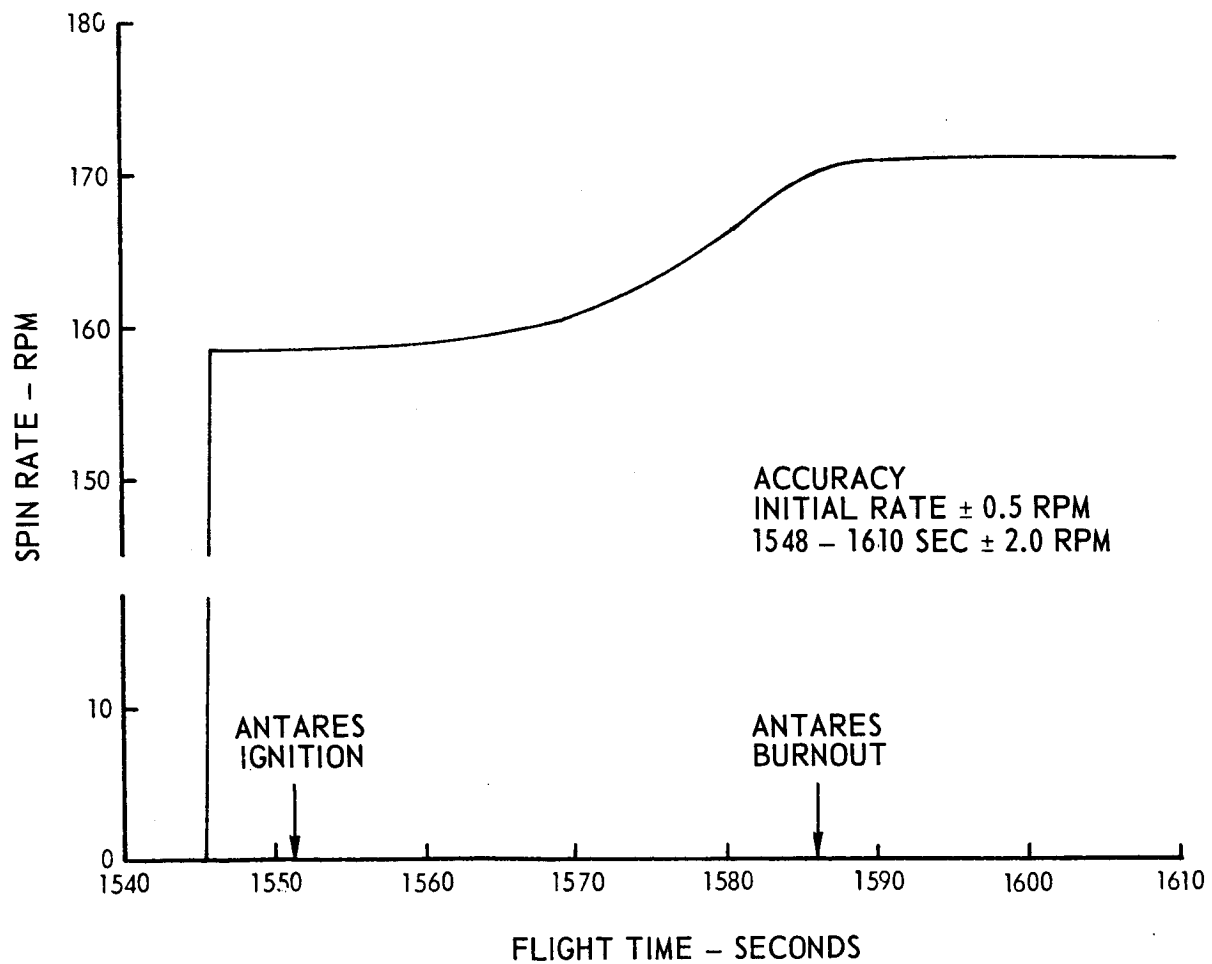
Spin-Up

Small disturbances were noted on all three vibrometer traces at the time that the spin motors fired. The V/P angular rate traces did not show any evidence of coning during or after spin-up, thus indicating good thrust balance between the motor and good dynamic balance of the Velocity Package. The initial rate was 158.6 rpm as determined from variations in the telemetry signal strength caused by antenna rotation. This spin rate was almost exactly as predicted. The spin rate increased during burning of the ANTARES motor to approximately 171 rpm at burn-out due to jet damping effects. A plot of the spin rate versus flight time is presented in Figure 6-7-3. The slight increase in spin rate after ANTARES motor burn-out is apparently due to accuracy limitations in data reduction or 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, trace readability corresponded to approximately 5 deg/sec and would preclude the detection of very small angular rates induced at separation. There was no evidence of disturbances on these traces during or following ANTARES motor ignition.

SPIN RATE



SECTION 8

THERMAL ENVIRONMENT ANALYSIS

General

The FIRE Velocity Package thermal control system assures 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 protection design provisions for Flight 2 were the same as for Flight 1. The following components were considered sufficiently marginal to warrant instrumenting for Flight 2:

<u>ITEM</u>	<u>TEMPERATURE, °F</u>	
	<u>Allowable</u>	<u>Calculated</u>
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)	384 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 selected components.

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TEMPERATURE MEASUREMENTS MONITORED DURING FLIGHT

<u>Code No.</u>	<u>Type of Measurement</u>	<u>Flight Sensor Location</u>
T ₁	ANTARES Case	Located within a sector of $\pm 10^\circ$ from the V/P +Y-Axis between stations 360 and 365.
T ₂	ANTARES Case	Located within a sector of $\pm 10^\circ$ from the V/P - Y-Axis between stations 360 and 365.
T ₃	ANTARES Base	Located at the ANTARES motor base on the electrical mounting pad at Station 420 (Quad III).
T ₄	Spin Motor Nozzle	Located on the Spin Motor nozzle (Quad IV) 1.5 inches from the nozzle exit.
T ₅	Spin Motor Nozzle	Located on the Spin Motor nozzle (- Y Axis) 1.5 inches from the nozzle exit.
T ₆	Spin Motor Nozzle	Located on the Spin Motor nozzle (Quad I) 1.5 inches from the nozzle exit.
T ₇	Equipment Operating Temperature	Located on the rate gyro base.
T ₈	Equipment Operating Temperature	Located on the base of the PVE unit.
T ₉	Equipment Operating Temperature	Located on the base of the telemetry transmitter.
T ₁₀	Equipment Operating Temperature	Located on the MIG (Miniature Integrating Gyros) block (Note: Vendor installed)

All temperature instrumented components operated at approximately the same temperature levels as they did on Flight 1, considering the fact that starting temperatures were slightly different. As in the case of Flight 1, the components operated within their respective temperature limits, and in no way impaired the operation and/or performance of any other V/P system or component.

Prior to Flight 1, the Prototype V/P, with installed operational systems, was tested in the LTV Space Environment Simulator (SES) as part of the V/P qualification program. These tests yielded thermal data for component temperature profiles, which with the transient analysis, make up the predicted performance of the V/P thermal control system and are compared with the flight data from Flight 2. On Flight 2, spin motor nozzle temperatures were measured, rather than spin motor case temperatures as on Flight 1. Since it is reasonable to assume that the nozzle and case temperatures are essentially the same up to the point of spin motor ignition, the data measured during Flight 2 for the spin motor nozzles is compared with predicted data for the motor cases.

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 somewhat different from values predicted from analysis and SES testing, in most cases. The thermal environment encountered after launch can be ascertained from the component temperature profiles as shown in Figures 6-8-7 and 6-8-8. Several points taken from temperature measurements made on Flight 1 are also shown for comparison. In general, the measured temperatures for Flight 2 were slightly higher than for Flight 1, due to the fact that the starting temperatures were slightly higher for Flight 2.

During the first phase of the flight, with the heat shroud in place, aerodynamic heating of the shroud occurred, but the transfer of this heat to the V/P interior components was blocked by insulation and aluminized tape on the shroud inner surface. 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, up to the point of shroud separation (at approximately 300 seconds flight time). Also, significant convective heating of interior components during the phase immediately after shroud jettison is apparently absent (relatively flat temperature profile slopes). Therefore, the thermal environment encountered by interior components not exposed directly to the space environment upon heat shroud jettison was almost entirely due to component internal heat generation.

After heat shroud ejection, the environment of the ANTARES II-A5 rocket motor and the spin motors was influenced by deep space conditions on the +Y side (See Figure 6-8-7) and earth radiation on the -Y side. During the flight, the vehicle passed out of the sunlight,

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THERMAL

and into an area of night. This is shown by the diverging temperature profiles for the spin motors. The spin motor located on the -Y axis (T5) receives heat radiated from the earth, and hence increased in temperature above the values for the other two spin motors (T4 and T6) which radiate to deep space, and receive essentially no radiant energy. The same effect is evident in the three temperature profiles measured on the ANTARES motor case.

Spin Motors

The temperatures of all three spin motor nozzles just prior to the time of heat shroud ejection were approximately 78°F. After shroud ejection, the temperatures diverged due to the fact that two of the motors radiated to deep space conditions, while the other received radiant energy from the earth. The motor receiving energy from the earth reached a maximum temperature of approximately 91°F, about 15°F above the predicted maximum at that point in time, and about 25 to 30°F above the temperature of the other two spin motor nozzles. The probable reason for exceeding the predicted upper limit is that the analysis upon which the predicted upper limit is based did not include radiation received by the motors from other parts of the vehicle. All spin motor nozzle temperatures remained in the allowable range shown in the table on page 6-8-1. Points from Flight 1 plotted for comparison show that a greater spread of temperatures for the three motors existed during Flight 2.

ANTARES Rocket Motor

The ANTARES motor case temperature profiles show that the motor remained well within the range of the predicted upper and lower limit temperatures, and well within the required range shown on page 6-8-1. In general, the ANTARES motor temperatures were very similar to those measured on Flight 1, as shown by the points on Figure 6-8-7.

Rate Gyro Unit

The rate gyro unit is subjected primarily to internal heat generation. As seen from Figure 6-8-8, the temperature profile falls between that predicted from analysis and from SES testing. The slope of the temperature gradient is approximately the same as that for Flight 1, but the magnitude is consistently about 10 to 12°F higher due to the higher starting temperature. Again, temperatures remain well within the limits shown in the table on page 6-8-1.

PVE Unit (Poppet Valve Electronics)

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 temperature sensor for the PVE unit 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 14°F of that predicted from the SES test. Flight 2 temperatures ran slightly lower than those recorded in Flight 1, despite similar starting temperatures. When corrected for starting temperatures, the flight temperatures are approximately the same as the SES predictions. The highest flight temperature (135°F) was well within the allowable limits for the PVE unit.

Telemetry Transmitter

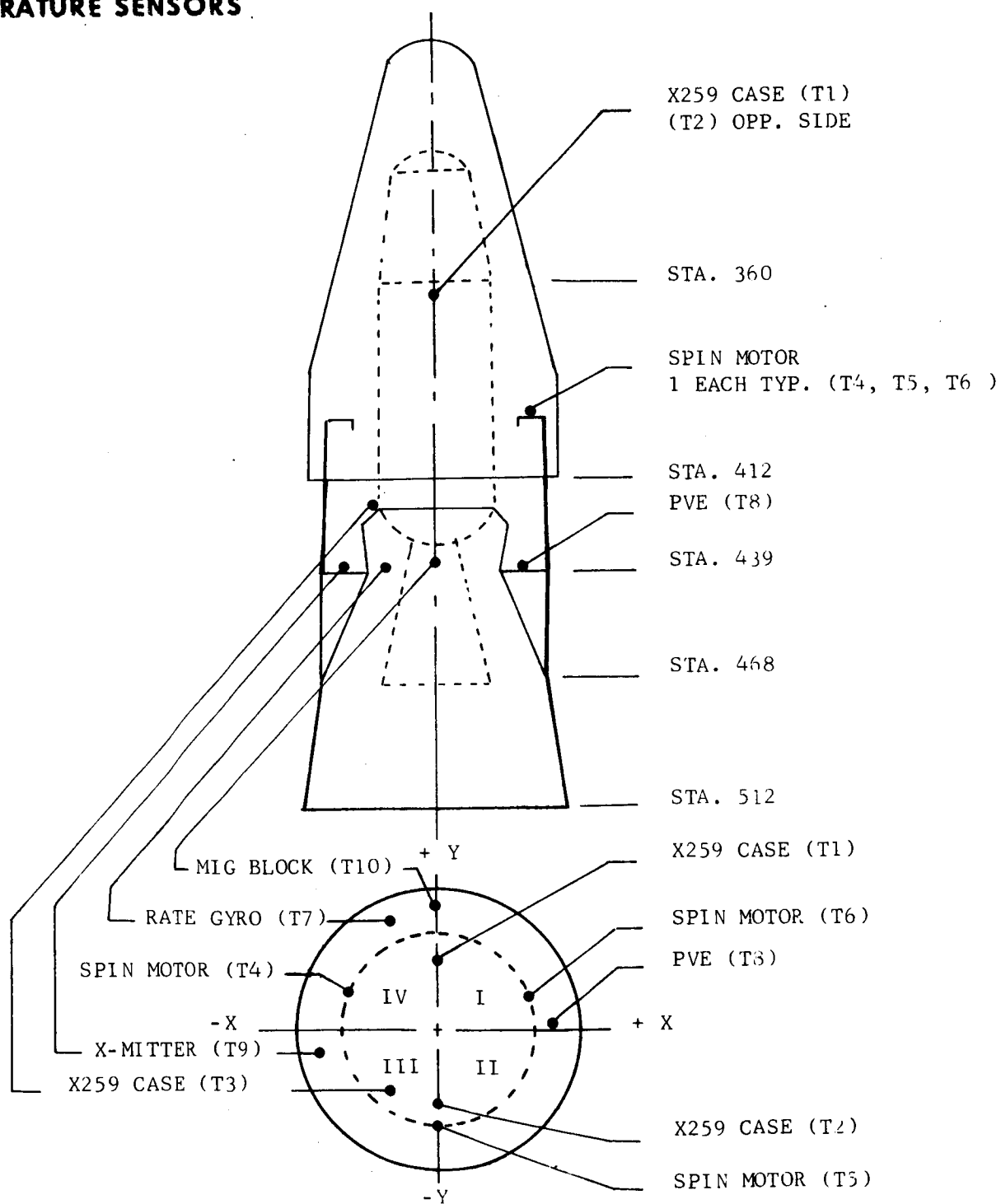
The telemetry transmitter in-flight temperature profile very nearly coincides with the profile predicted from analysis. Transmitter temperature about midway between tracking stations was approximately 110°F which is below the desired maximum of 120°F at this point in time. The temperature remained above that for Flight 1 due to the higher initial temperature on Flight 2. Final temperature was about 126°F, still well below the maximum allowable of 160°F.

Guidance Unit Assembly

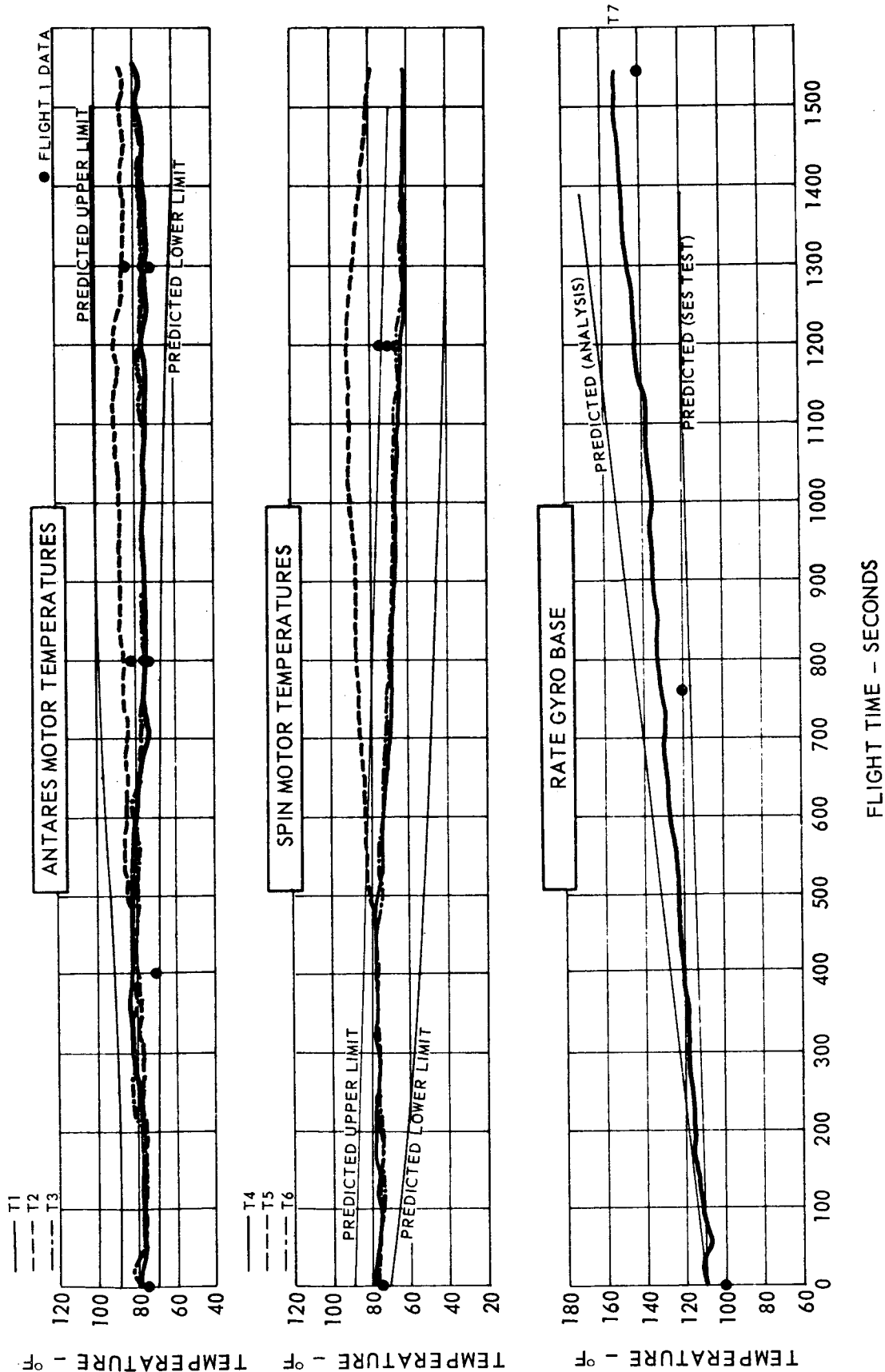
The flight temperature profile shows that the MIG block temperature cycled about its control level of 180°F, which verified that the heaters maintained control of the MIG block temperature. Although the temperature of the guidance unit assembly itself was not recorded, the cycling of the MIG block temperature 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. GDC/BKF65-042
 LTV/A REPORT NO. 3-30000/5R-30
 THERMAL

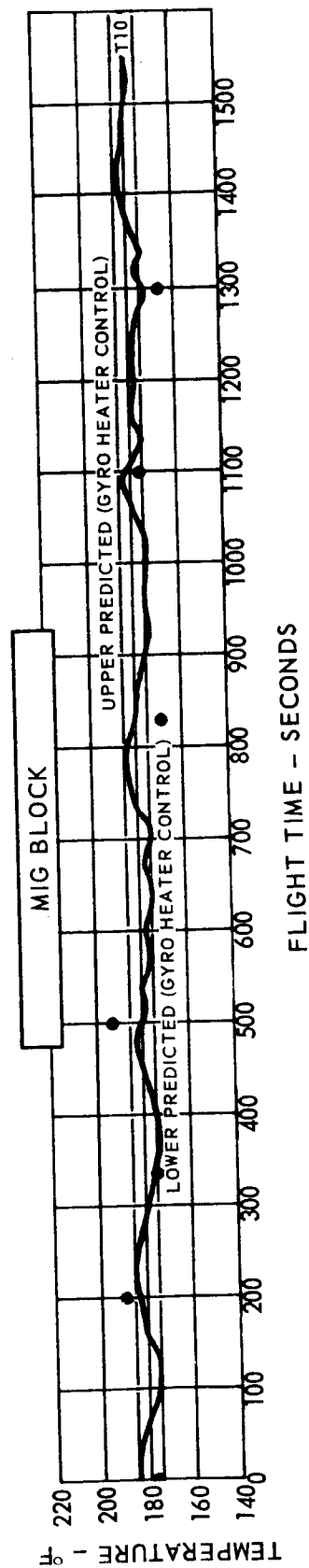
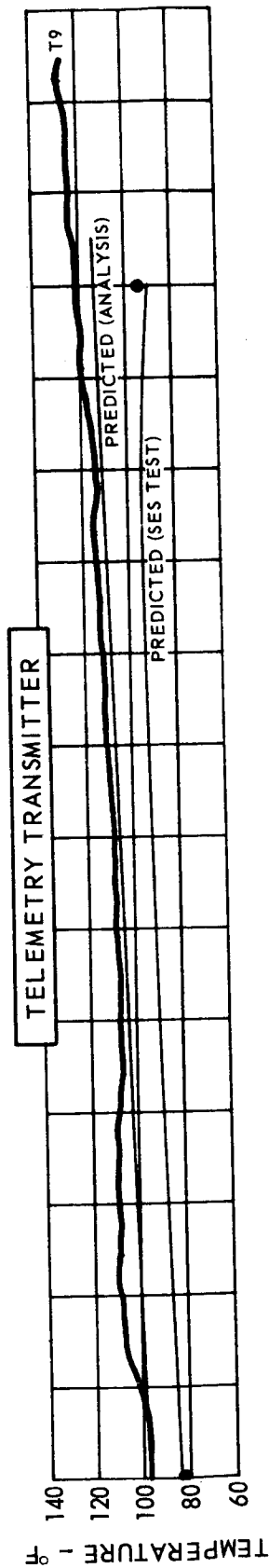
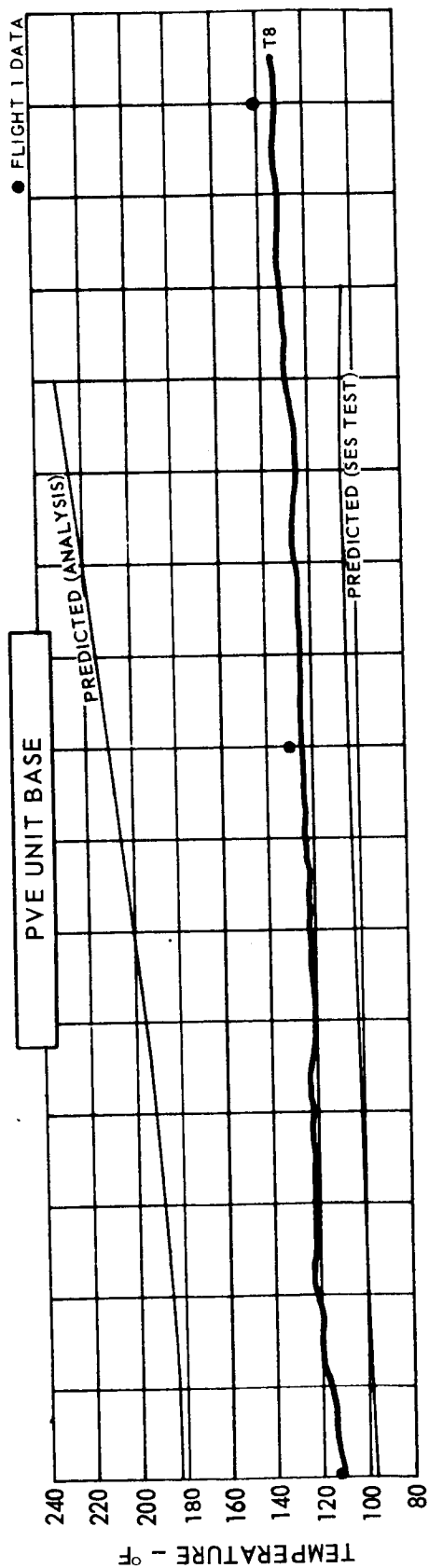
TEMPERATURE SENSORS



VELOCITY PACKAGE PERFORMANCE
 FIGURE NO. 6-8-7
 INTEGRATED REPORT NO. GDC/BKF65-042
 LTV/A REPORT NO. 3-30000/5R-30
 THERMAL
FLIGHT TEMPERATURES



VELOCITY PACKAGE PERFORMANCE
 FIGURE NO. 6-8-8
 INTEGRATED REPORT NO. GDC/BKF65-042
 LTV/A REPORT NO. 3-30000/5R-30
 THERMAL
FLIGHT TEMPERATURES



PART 7
GUIDANCE SYSTEM PERFORMANCE

GENERAL DYNAMICS/CONVAIR
INTEGRATED REPORT NO. GDC/BKF65-042

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

SECTION 1

INTRODUCTION

Launch Vehicle (L/V) 264D 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/Convair. 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 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 measured. Sustainer thrust was terminated at the proper time to achieve the velocity required to satisfy the target conditions. Because of the relatively large propellant 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 measured 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. A block diagram of the back-up sustainer cutoff system is shown in Figure 7-2-4.

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

GUIDANCE SYSTEM PERFORMANCE

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DISCUSSION

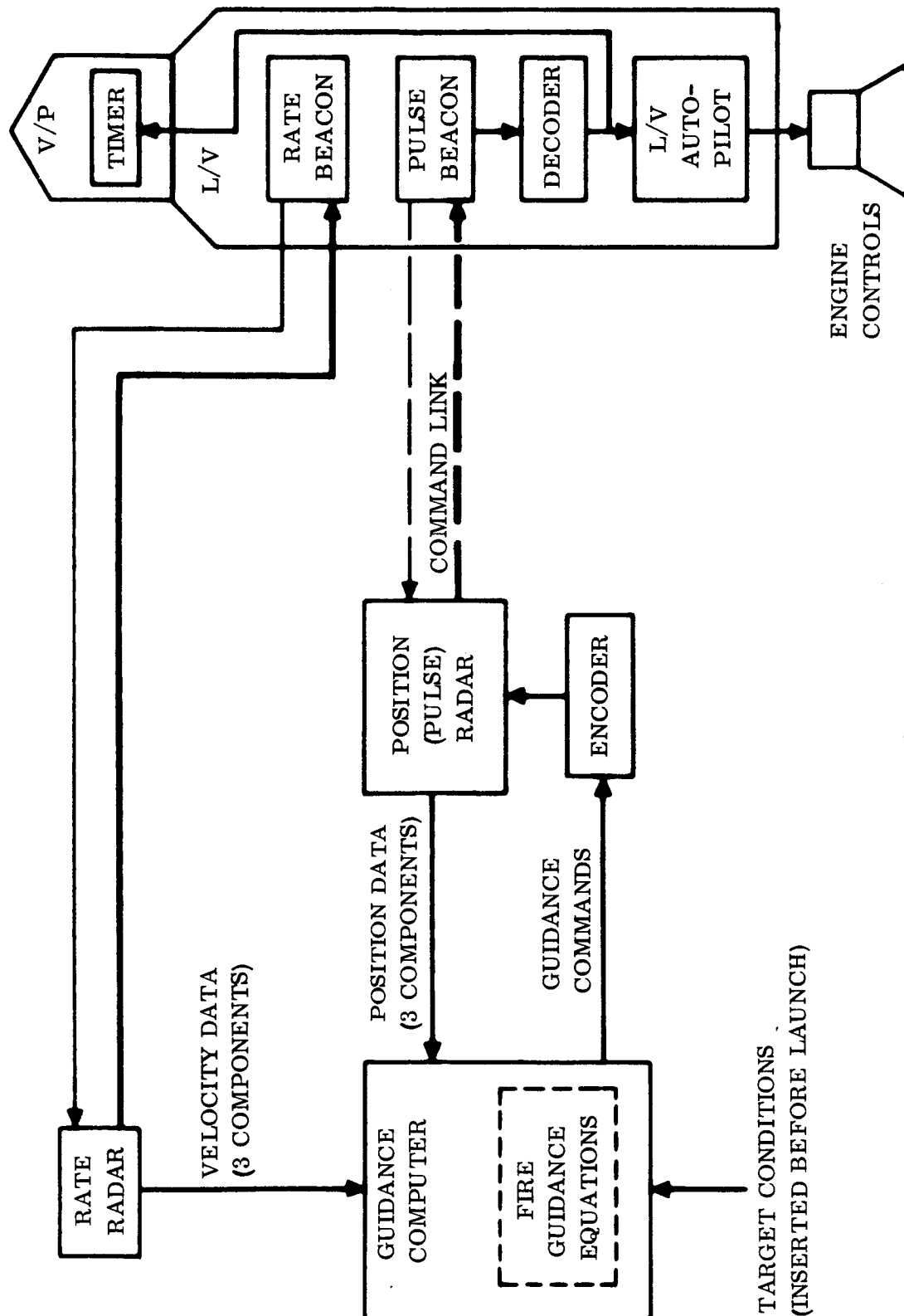
the spacecraft separation discrete.

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 as a function of time from rise off.

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.

Because of the explicit nature of the guidance equations, the only required change for flight No. 2 was an adjustment of two targeting constants which was necessary because of the trajectory changes for flight No. 2. Two additional changes were incorporated into the guidance equations for this flight to improve performances. The ignition interlock command logic was changed to permit better synchronization with the flight programmer, and the vernier attitude steering logic was changed so that the steering period could be reduced to allow more time for vehicle damping prior to V/P gyro enable.

GUIDANCE SYSTEM PERFORMANCE
 FIGURE NO. 7-2-3
 INTEGRATED REPORT NO. GDC/BKF65-042
 DISCUSSION
 SIMPLIFIED BLOCK DIAGRAM
 OF RADIO GUIDANCE SYSTEM

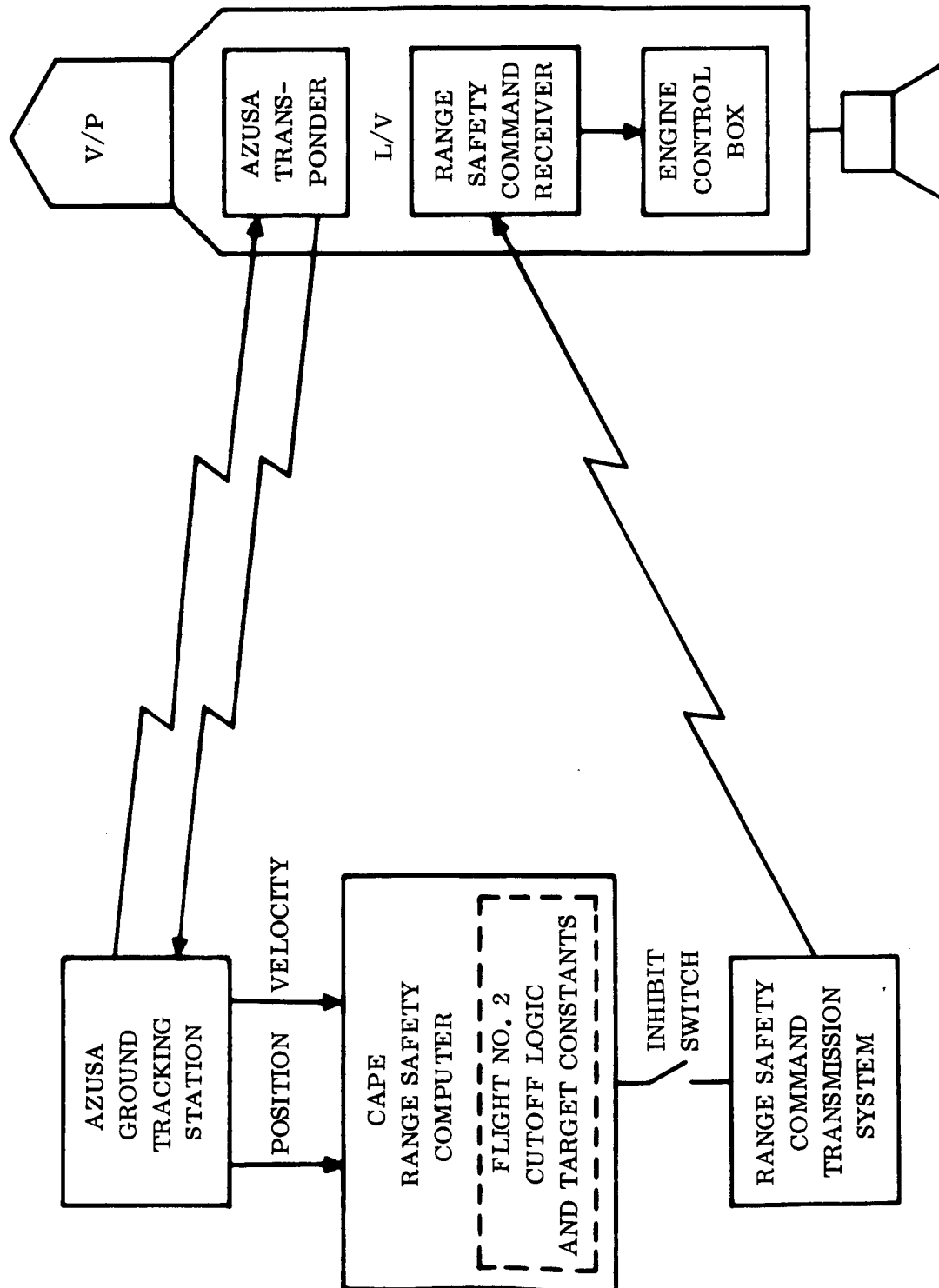


GUIDANCE SYSTEM PERFORMANCE

FIGURE NO. 7-2-4

INTEGRATED REPORT NO. GDC/BKF65-042

SIMPLIFIED BLOCK DIAGRAM OF
BACKUP SUSTAINER CUTOFF SYSTEM



SECTION 3

CONCLUSIONS

All L/V guidance objectives for FIRE Flight No. 2 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 CONVAIR
INTEGRATED REPORT NO. GDC/BKF 65-042

APPROVED BY: L. E. Munson

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SECTION 1

INTRODUCTION

The second Project FIRE launch vehicle, Atlas 264D, was successfully launched from the ETR, Complex 12, at 1655 EST on 22 May 1965. The Atlas space launch vehicle, produced by General Dynamics Convair (GDC), 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 GDC test objectives were satisfactorily accomplished.

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

SECTION 2

LAUNCH VEHICLE CONFIGURATION

A brief description of the Atlas 264D 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. The booster engine used hypergolic ignition, the sustainer and vernier engines used pyrotechnic 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 five 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. 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.

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CONFIGURATION

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 Convair 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. 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 vehicle systems operation and areas of interest.

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CONFIGURATION

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

GDC TEST OBJECTIVES

The following table presents the list of flight objectives which were scheduled for Atlas LV-3A 264D and against which data were obtained and evaluated.

<u>Description</u>	<u>Priority</u>	<u>Satisfied</u>
Demonstrate the ability of the LV-3A 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 discretizes and steering commands to achieve the trajectory defined by the guidance equations.	1	Yes
Determine LV-3A systems flight performance utilizing telemetry data.	1	Yes
Demonstrate the structural integrity, during flight, of the LV-3A portion of the assembled vehicle.	2	Yes
Obtain data on the LV-3A 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 Eastern Test Range support equipment to obtain external telemetry and tracking data throughout the vehicle powered flight.	2	Yes
Demonstrate that the LV-3A flight programmer and MOD IIIG General Electric/Burroughs guidance system provided the correct commands for flight operations peculiar to the FIRE program.	2	Yes
Demonstrate that the LV-3A flight control system has the ability to stabilize and control the LV-3A vehicle in proper response to guidance commands generated by the GE/Burroughs guidance system to achieve the desired trajectory.	2	Yes
Demonstrate that the LV-3A flight control system has the ability to stabilize and control the LV-3A 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 that 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 operation of the propulsion system was satisfactory. The ISS pneumatic regulator outlet pressure and the engine fuel tank pressure exhibited abnormal characteristics between booster jettison and SECO, but system operation was not affected.

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

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

<u>Parameter</u>	<u>Units</u>	<u>Redline Limit</u>	<u>Engine Start Value</u>
Booster Lox Regulator Reference Pressure	psig	561 to 581	575
Sustainer Lox Regulator Reference Pressure	psig	819 to 859	832
B2 Turbine Inlet Temperature	° F	0 Minimum	96
Sustainer Turbine Inlet Temperature	° F	0 Minimum	78
Lox Temperature at Breakway Valve	° F	-283 Maximum	-293.5
Sustainer Lube Oil Temperature	° F	>45	85

Inflight Booster Engine Performance

Operation 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.

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SUMMARY

TABLE 8-4-2. BOOSTER ENGINE FLIGHT DATA

<u>Measurement</u>	<u>Units</u>	<u>+10 seconds</u>	<u>BECO</u>
B1 Chamber Pressure	psia	540	552
B1 Pump Speed	rpm	(1)	(1)
B1 Fuel Pump Inlet Pressure	psia	67	57
B1 Lox Pump Inlet Pressure	psia	61	98
B2 Chamber Pressure	psia	537	546
B2 Pump Speed	rpm	6,025	5,995
B2 Fuel Pump Inlet Pressure	psia	73	64
B2 Lox Pump Inlet Pressure	psia	69	>100
BGG Chamber Pressure	psia	483	480
BGG Lox Reg. Ref. Pressure	psia	580	565

NOTE: (1) Data Invalid.

Inflight Sustainer Engine Performance

Sustainer engine operation 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

<u>Measurement</u>	<u>Units</u>	<u>Liftoff</u>	<u>BECO</u>	<u>SECO</u>
Chamber Pressure	psia	680	670	655
Pump Speed	rpm	10,415	10,355	10,355
Fuel Pump Discharge Pressure	psia	893	885	895
Fuel Pump Inlet Pressure	psia	72	71	44
Lox Pump Inlet Pressure	psia	68	110	78
GG Discharge Pressure	psia	624	624	632
Lox Regulator Reference Pressure	psia	840	825	825

Inflight Vernier Engine Performance

Operation of the vernier engine system was satisfactory. However, the ISS pneumatic regulator outlet pressure and the engine fuel tank pressure exhibited anomalous characteristics between jettison and SECO. Between jettison (137 seconds) and 158 seconds these pressures exhibited abnormal increases (from 580 psia and 590 psia, respectively) to 670 psia. This abnormal pressure increase was identical in character to the normal pressure rise of the

engine lox tank pressure as it made the characteristic increase from the pneumatic regulator outlet pressure level to lox charge line pressure level. Upon reaching the 670-psia level, the regulator outlet pressure initiated a slow decrease (at normal manifold bleed rate) until it stabilized at a level of 560 psia at 260 seconds. However, the engine fuel tank pressure maintained the 670-psia level until vernier solo was initiated (at SECO).

It has been concluded that these two anomalies were both probably caused by a malfunction of the isolation check valve in the engine lox tank pressurization line. If this check valve failed to close when the engine lox tank pressure rose above regulator outlet pressure (as a normal consequence of the lox charge to the tank) the higher pressure would enter the pneumatic manifold and, therefore, be reflected by the regulator outlet pressure and the engine fuel tank pressure. On the flight of Vehicle 264D, the subsequent decrease of regulator outlet pressure apparently occurred because of closure of the lox isolation check valve which allowed the excess pressure to bleed down to the regulator control level. The engine fuel tank pressure was prevented from bleeding down because of the isolation check valve in the engine fuel tank pressurization line.

As a result of a similar occurrence on Vehicle 301D, Field Engineering Bulletin (FEB) R65-19 had been issued which required a reverse flow leak check on the lox isolation check valve prior to flight. This procedure was performed on Vehicle 264D, but apparently failed to prevent a recurrence of this malfunction. More effective remedial action has been taken by issuance of ECP MA5-146. This ECP will accomplish the re-design of the isolation check valve poppet and seat to prevent leakage.

TABLE 8-4-4. VERNIER ENGINE FLIGHT DATA

<u>Measurements</u>	<u>Units</u>	<u>10 seconds</u>	<u>BECO</u>	<u>SECO</u>	<u>VECO</u>
V1 Chamber Pressure	psia	360	362	368	320
V2 Chamber Pressure	psia	360	358	362	318
Engine Lox Tank Pressure	psia	-	-	705	580
Engine Fuel Tank Pressure	psia	-	265	670	580

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.

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

	<u>Liftoff</u>	<u>BECO</u>	<u>SECO</u>	<u>VECO</u>
<u>Booster Engine Thrust (pounds)</u>				
Actual	301,070	359,500	---	---
Predicted	308,585	365,525	---	---
<u>Sustainer Engine Thrust (pounds)</u>				
Actual	57,850	78,350	77,150	---
Predicted	56,304	78,968	79,114	---
<u>Vernier Engine Thrust (pounds)</u>				
Actual	1,745	1,822	1,676	1,460
Predicted	1,718	1,980	1,616	1,524

FLIGHT CONTROL SYSTEM

Flight control system operation 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 15 seconds after liftoff and maintained through the end of the booster engine phase (refer to Table 8-4-6 below for Atlas 264D nominal pitch program). The actual pitchover angle at 130 seconds, as evaluated by using ETR tracking data, was -64.35 degrees. Comparison of this value with the nominal angle of -64.85 degrees indicates the vehicle attitude was 0.50 degree high.

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

BOOSTER PHASE				
Time (sec)	Programmer Output (volts)	Programmer Output Integral (volts-sec)	Rate (deg/sec)	Vehicle Angle (degrees)
15	1.6	0	-0.602	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
133.881	0.0	174.829	0	-65.74

SUSTAINER PHASE				
143.881	0.3	0	-0.1128	-65.74
285.682	0	42.54	0	-81.74

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

The booster pitch program ends 0.1 seconds after the BECO discrete (S236X) or the "staging backup" acceleration switch signal (S359X) 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 0.14 degree at a peak rate of 0.88 degree per second. Atlas 264D employed the 0.108-degree booster thrust chamber roll offset to reduce the roll magnitude at liftoff. Maximum aerodynamic loading occurred at approximately 70 seconds, requiring booster No. 1 and No. 2 thrust chamber deflections of 1.7 and 1.6 degrees, respectively, to maintain vehicle stability. Propellant slosh amplitudes were small with complete damping prior to BECO. Vehicle transients associated with booster cutoff were normal and quickly damped by the autopilot. Due to the mission constraints and a 2.1-second longer-than-nominal vernier solo, spacecraft separation resulted from Atlas programmer backup at SECO + 23 seconds

rather than by guidance discrete at VECO + 5.5 seconds. The possibility of this occurrence had been anticipated by GDC design, occurred during the separation sequence of the FIRE I vehicle (Atlas 263D), and does not represent a problem.

Retrorocket Firing Sequence

A retrorocket exhaust gas deflector, previously utilized on Vehicle 263D, was not installed on this vehicle (refer to Airframe Section). This deflector, on Vehicle 263D, contributed an additional component of pitch angular acceleration to the normally-expected pitch acceleration at retrofire. This additional component of pitch acceleration could conceivably cause the vehicle to hit the Velocity Package during separation. With removal of the exhaust gas deflector, the wiring to the forward rate gyro package in the vicinity of the retrorockets was exposed to the exhaust gases. It was determined, when the decision was made to remove the deflector, that its removal might cause some wiring damage after retrofire.

As anticipated, rate gyro data was completely lost at 0.7 second after retrofire began. At this same time, displacement gyro data indicated reductions in gain followed by a complete loss of data 18.7 seconds after retrofire. The loss of rate gyro data at retrofire, the reduction in displacement gyro gains, and the subsequent loss of displacement gyro data are all indicative of shorting of the Phase A excitation to the rate gyro signal generators and loading down of the T2 transformer, in the U1 gyro package, which is common to both the rate and displacement gyro signal generators. The loss of displacement gyro data indicates that the loading of the T2 transformer eventually caused a complete burn-out. Loss of the excitation voltage, and subsequent abnormal system conditions, did not compromise the mission.

Removal of the retrorocket exhaust gas deflector was successful in reducing the angular accelerations at retrofire. Table 8-4-7 summarizes the angular rates and accelerations at retrofire for 263D and 264D.

TABLE 8-4-7. ATLAS ANGULAR ACCELERATIONS AT RETROFIRE

	<u>Roll</u>	<u>Pitch</u>	<u>Yaw</u>
263D			
Peak Rate	3.7 deg/sec (CW)	2.3 deg/sec (DN)	0.3 deg/sec (RT)
Acceleration	5.5 deg/sec/sec (CW)	2.8 deg/sec/sec (DN)	0.3 deg/sec/sec (RT)
264D			
Peak Rate	2.1 deg/sec (CW)	1.1 deg/sec (DN)	0.3 deg/sec
Acceleration	4.5 deg/sec/sec (CW)	1.6 deg/sec/sec (DN)	Negligible

NOTES: CW = Clockwise
DN = Nose down
RT = Nose right

Special Instrumentation

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

GUIDANCE SYSTEM

Operation of the MOD III guidance system was satisfactory. Actual insertion parameters compared very closely to desired values. The vehicle was acquired by the cube-in-space method as planned, and conical radar tracking was established at 60.1 seconds. Reliable track data (as indicated by the track data flag) was continuously presented to the ground guidance computer from 67.9 seconds until 344.2 seconds. The rate flags (range and lateral rates) were indicating valid rate data continuously from 66.9 seconds until 332.2 seconds, including the interval during booster package jettison.

Corrective pitch/yaw steering commands were small in magnitude, and the planned discrete commands were correctly generated and received, as presented in Table 8-4-8. No booster steering was scheduled for this flight.

TABLE 8-4-8. DISCRETE COMMAND SUMMARY

<u>Discrete Command</u>	<u>Computer Output</u>	<u>Decoder Output Nominal/ Actual (1)</u>	<u>Engine Relay Activation (1)</u>	<u>Axial Accelerometer Indication</u>
V/P Ign Interlock	126.750	126.538/ 126.779 $\pm$ 0.023	NA	NA
BECO/Staging	133.750	134.714/ 133.773 $\pm$ 0.009	133.937	134.007
V/P Ign Interlock Backup	144.250	145.038/ 144.262 $\pm$ 0.024	NA	NA
SECO	285.650	285.841/ 285.703 $\pm$ 0.025	285.694 $\pm$ 0.020	285.727
Start V/P Timer	294.347	294.768/ 295.387 $\pm$ 0.025	NA	NA
V/P Heat Shield Jettison	295.250	295.538/ 295.286 $\pm$ 0.025	NA	295.298
VECO	304.594	302.658/ 304.634 $\pm$ 0.011	304.664 $\pm$ 0.049	304.742
V/P Separation (2)	310.250	308.687/ 310.442 $\pm$ 0.025	NA	308.733

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NOTES: All times are in seconds from Atlas 2-inch motion, 1654:59.703 hours EST.

- (1) Uncertainties result from commutated data.
 - (2) Actual V/P separation was accomplished by Atlas programmer switch 22 backup at SECO + 23.0 seconds (308.729 ± 0.025 seconds).
- N/A Not applicable.

PNEUMATIC SYSTEM

Operation of the pneumatic system was satisfactory. As evidenced by telemetry data, propellant tanks pressurization and pneumatic control functions were properly accomplished throughout powered flight.

Table 8-4-9 presents pressures tabulated at significant times during the flight.

TABLE 8-4-9. PROPELLANT TANK FLIGHT PRESSURE DATA (psig)

<u>Measurement</u>		<u>Units</u>	-10	Steady	<u>BECO</u>	<u>SECO</u>	<u>VECO</u>
			<u>Seconds</u>	<u>After</u> <u>Liftoff</u>			
F1P	Lox Tank Press.	psig	25.0	24.0	24.5	25.0	25.0
F3P	Fuel Tank Press.	psig	58.5	57.0	58.0	41.0	41.0
F116P	Bulkhead Diff. Pr.	psid	15.6	9.0	12.3	13.8	14.7
F125P	Boost. Control Reg.	psig	750	750	750	-	-
F246P	Boost. Btls. Press.	psig	3085	2995	665	-	-
F288P	ISS Reg. Press.	psig	592	592	584	560	576
F291P	Sust. Btl. Press.	psig	3065	3000	2660	2450	1400
F247T	Boost. Btls. Temp.	DGF	-314	-316	-382	-	-

HYDRAULIC SYSTEM

Operation of the hydraulic system was satisfactory. As evidenced by telemetry data, hydraulic pressures were adequate to support all user systems throughout the flight.

Normal booster and sustainer hydraulic evacuations were initiated at minus 23.4 and 23.6 seconds, respectively. Normal transients were noted at engine ignition, after which the system pressures stabilized and remained stable at nominal flight pressure levels. The dual vernier solo accumulators bottomed out 68.2 seconds after SECO at 875 psia.

Table 8-4-10 presents hydraulic system data at significant times.

TABLE 8-9-10. HYDRAULIC SYSTEM PRESSURES (psia)

<u>Measurement</u>	<u>Before Oil Evacu- ation</u>	<u>After Oil Evacu- ation</u>	<u>(-10 Seconds)</u>	<u>Steady After Liftoff</u>	<u>BECO</u>	<u>SECO</u>	<u>VECO</u>
B1 Hydraulic Accum- ulator Pressure (H33P) (1)	1890	1855	1855	3045	3030	--	--
Booster Hydraulic Pump Discharge Pres- sure (H3P)	--	--		DATA INVALID			
Booster System Return Pressure (H224P) (2)	126	78	78	78	66	--	--
Sustainer/Vernier System Pressure (H140P) (1)	1890	1855	1855	3045	3045	3030	3030
Sustainer Hydraulic Pump Discharge Pressure (H130P)	--	--	--	3045	3045	3045	--
Sustainer/Vernier System Return Pres- sure (H601P) (2)	105	72	69	66	72	72	72

NOTES: (1) Redline pressure tolerances measured at HPU: 2000 plus 250 psig prior to engine start.

(2) Minimum redline pressure measured at HPU: 25 psig prior to engine start.

-- Not applicable

ELECTRICAL SYSTEM

Operation of the L/V electrical system was satisfactory. The main vehicle battery voltage and rotary inverter frequency and voltage were within specification throughout Atlas powered flight.

TABLE 8-4-11. ELECTRICAL SYSTEM PARAMETERS AT SELECTED TIMES

<u>Description</u>	<u>Liftoff</u>	<u>BECO</u>	<u>SECO</u>	<u>VECO</u>
E28V Vehicle DC Bus, VDC	26.9	27.3	27.3	27.3
E50Q Inverter Freq., CPS	398.8	398.8	398.8	398.8
E51V Phase "A" Voltage, VAC	114.5	114.5	114.6	114.6

PROPELLANT UTILIZATION SYSTEM

Operation of the GDC propellant utilization (PU) and propellant loading systems was satisfactory. The predicted and actual propellant residuals at sustainer engine cutoff (SECO) are presented in Table 8-4-12.

TABLE 8-4-12. PROPELLANT RESIDUALS AT SECO

TOTAL PROPELLANTS ABOVE PUMP

	<u>Lox (lb)</u>	<u>Fuel (lb)</u>	<u>Total (lb)</u>
Predicted (1)	2408	1213	3621
Actual	2525	1158	3683

USEABLE PROPELLANTS (2)

	<u>Lox (lb)</u>	<u>Fuel (lb)</u>	<u>Time to Depletion (Sec.)</u>	<u>Outage (lb)</u>
Predicted (1)	2338	1149	12.64	127 Fuel
Actual	2455	1094	12.85	132 Fuel

NOTES: (1) The predicted values are based on the preflight trajectory simulation.

(2) The lox depletion level is 70 pounds above the pump inlet. The fuel depletion level is the Station 1198 anti-vortex web, 64 pounds above the pump inlet.

AZUSA SYSTEM

Operation of the AZUSA system was satisfactory. The angle cosines were switched to "fine" at 5 seconds and automatic track was established at 6.2 seconds. Azusa data was selected by the range IBM 7094 computer for impact predictions from 14.2 to 135.2 seconds, 135.5 to 310 seconds, 311 to 327.9 seconds, and from 330.2 to 340.6 seconds.

RANGE SAFETY COMMAND SYSTEM

Operation 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 285.733 ± 0.103 seconds. The manual fuel cutoff (MFCO) and destruct command signals were not required nor transmitted. The ASCO signal based on AZUSA data and generated by the Real Time Computer Facility was recorded at Central Control at 285.647 seconds. Transmission of this signal was inhibited.

AIRFRAME SYSTEM

Vehicle structural integrity was maintained throughout powered flight and beyond spacecraft separation. The normal 5-cps longitudinal vehicle oscillations following liftoff were normal and attained a maximum amplitude of 0.92g (p-p) at one second. Peak accelerations at BECO and SECO were 7.23g and 5.41g, respectively. Planned accelerations for these functions were 7.14g and 5.48g, respectively. Axial acceleration data indicated that booster section jettison and payload separation were satisfactorily accomplished and occurred at 136.909 and 308.733 seconds, respectively. Integration of the "fine" axial acceleration data over the firing period of the retrorockets yielded a 0.060g-second level, which corresponded to a decrease in velocity of 1.94 feet per second for the Atlas second-stage tank section.

Environmental conditions in the Atlas thrust section were satisfactory throughout flight with a maximum of 92° F recorded at BECO in the Quadrant II "A" frame area.

Retrorocket Configuration Change

As a result of the excessive vehicle pitch-down acceleration created by the retrorockets' flame impingement upon the retrorockets-to-pod baffle plates during the flight of Atlas 263D, the baffle plates were removed from vehicle 264D by ECP 7933. As a consequence, it was anticipated that wire harnesses to the forward rate gyro package, the inter-stage adapter, and the J-106 interface plug would be damaged at the time of retrorocket firing. An evaluation of all possible conditions which could arise from retrorocket gas impingement in the pod area was conducted. The results of this analysis indicated that no structural damage would

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occur and that even if severe cable damage were incurred, no detrimental effects to the mission should arise, since the booster mission is completed with the generation of the retrorocket fire signal.

Analysis of 264D flight control and electrical systems data showed indications of cable damage beginning during the period of retrorocket firing. Cable damage was attributed to flame impingement from the retrorockets.

The two vehicles of this program (263D and 264D), originally had a retrorocket installation which was different from all previous Atlas space launch vehicles. The installation was in the forward end of the B1 pod and consisted of Rocket Power Inc. retrorockets and retro-rocket-to-pod baffle plates. During the countdown of 263D, the baffle plates had to be modified so that the retrorockets could be installed. The plates were subsequently removed from 264D. The only other Atlas vehicles, other than 263D, to use the retrorocket-to-pod baffle plates were the Nike Target Group A vehicles which flew after 159D, but before the introduction of the HIRS.

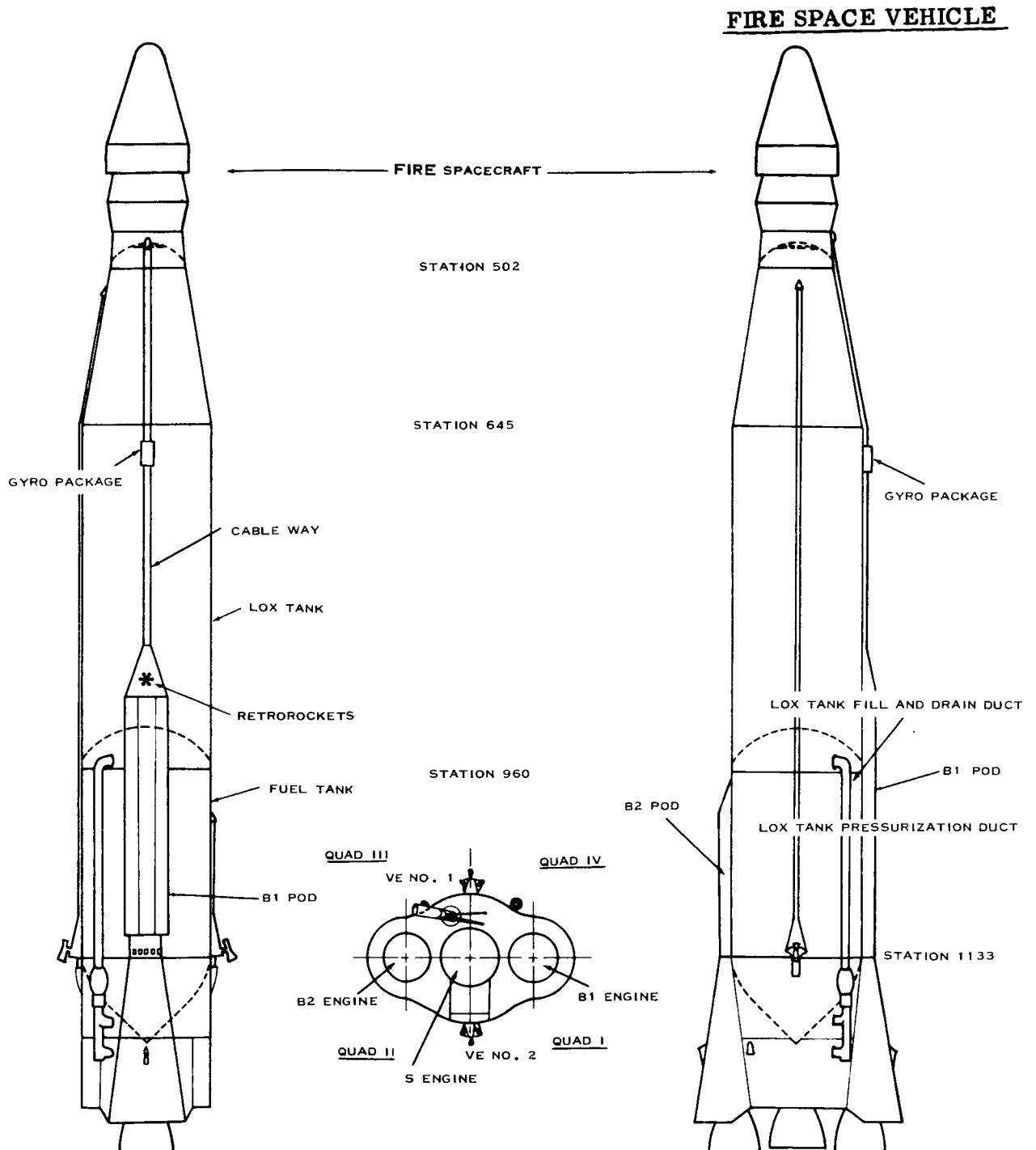
TELEMETRY SYSTEM

Performance of the telemetry system was satisfactory. Valid data was received from pre-liftoff through the end of powered flight. Telemetry RF signal received at the Tel II ground receiving station was of high strength and good quality throughout flight. From liftoff to 60 seconds the signal averaged above 10,000 microvolts, thereafter it gradually decayed and reached the 500-microvolt level at velocity-package separation (308.7 seconds).

The normally-encountered signal strength dip at booster jettison occurred at 136.9 seconds, however, on this flight the signal attenuation was much less than usual and did not cause loss of telemetry data.

Eleven channels of continuous data and six commutated channels were transmitted on one RF carrier. A total of 103 measurements were instrumented. One measurement was considered unsatisfactory for flight analysis, resulting in a telemetry system data recovery of 99.0 percent. One other measurement provided qualitative data only.

LAUNCH VEHICLE PERFORMANCE
FIGURE NO. 8-4-13
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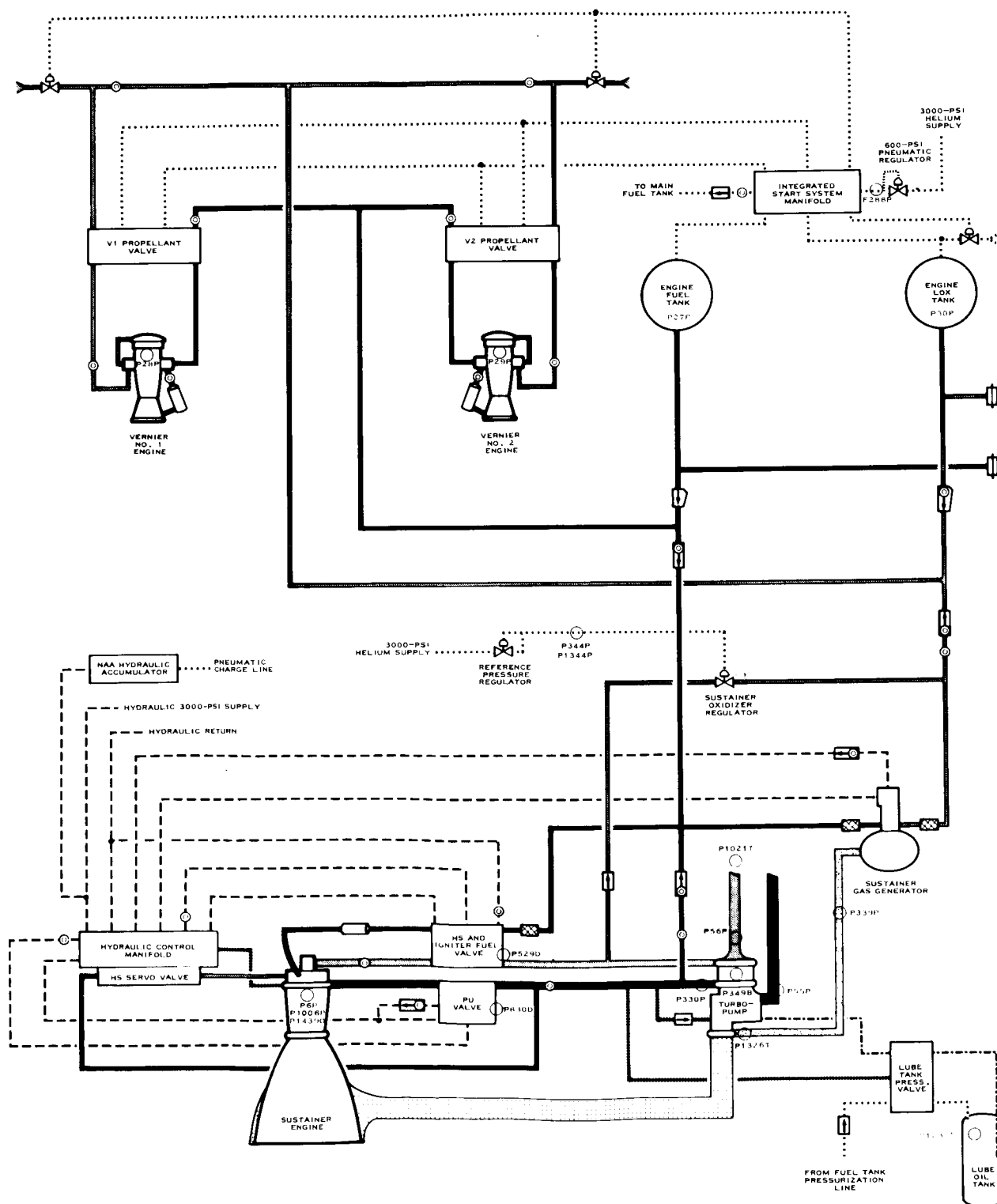
LAUNCH VEHICLE PERFORMANCE

FIGURE NO. 8-4-14

INTEGRATED REPORT NO. GDC/BKF65-042

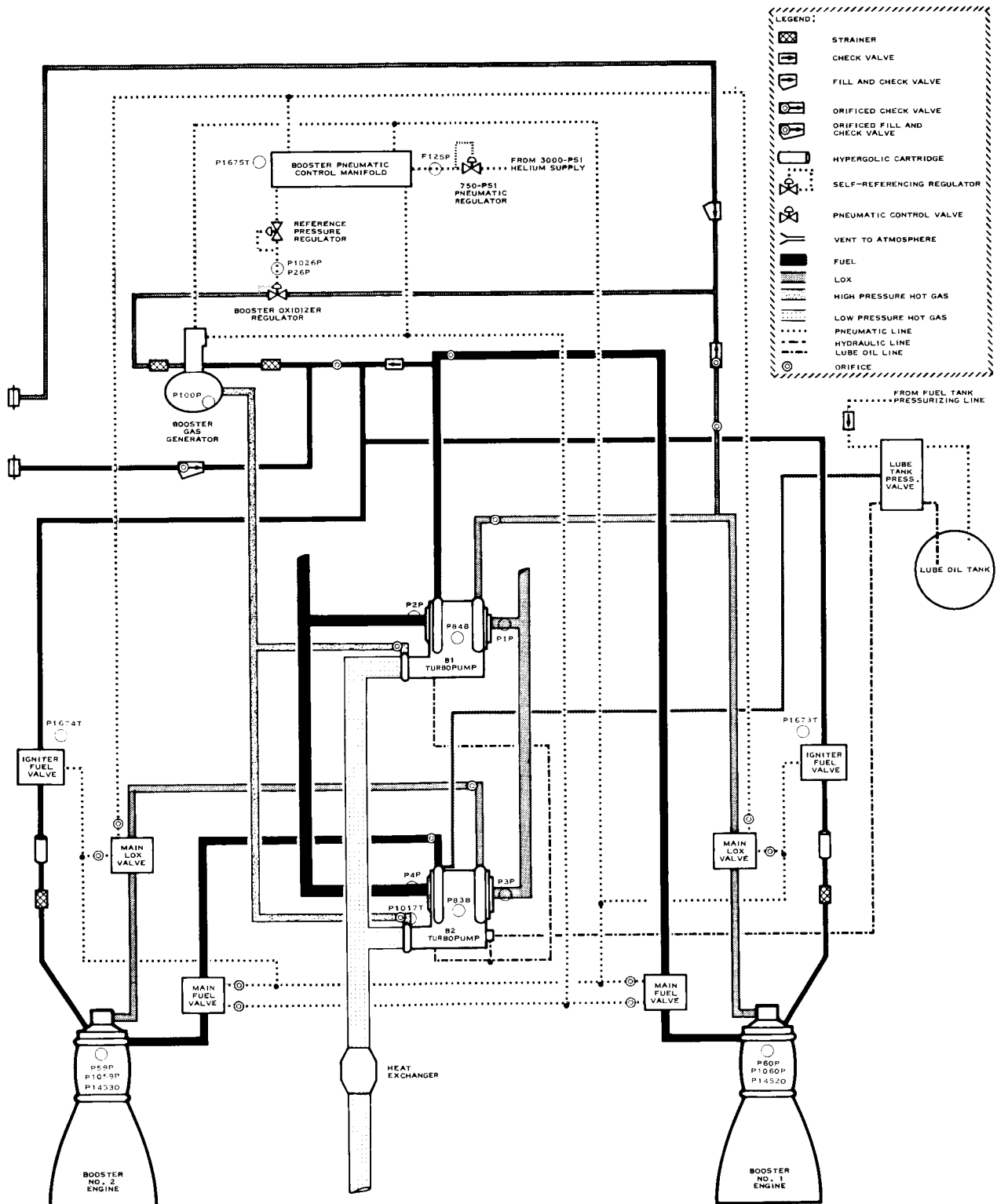
SUMMARY

FIRE L/V SUSTAINER/VERNIER PROPULSION SYSTEM



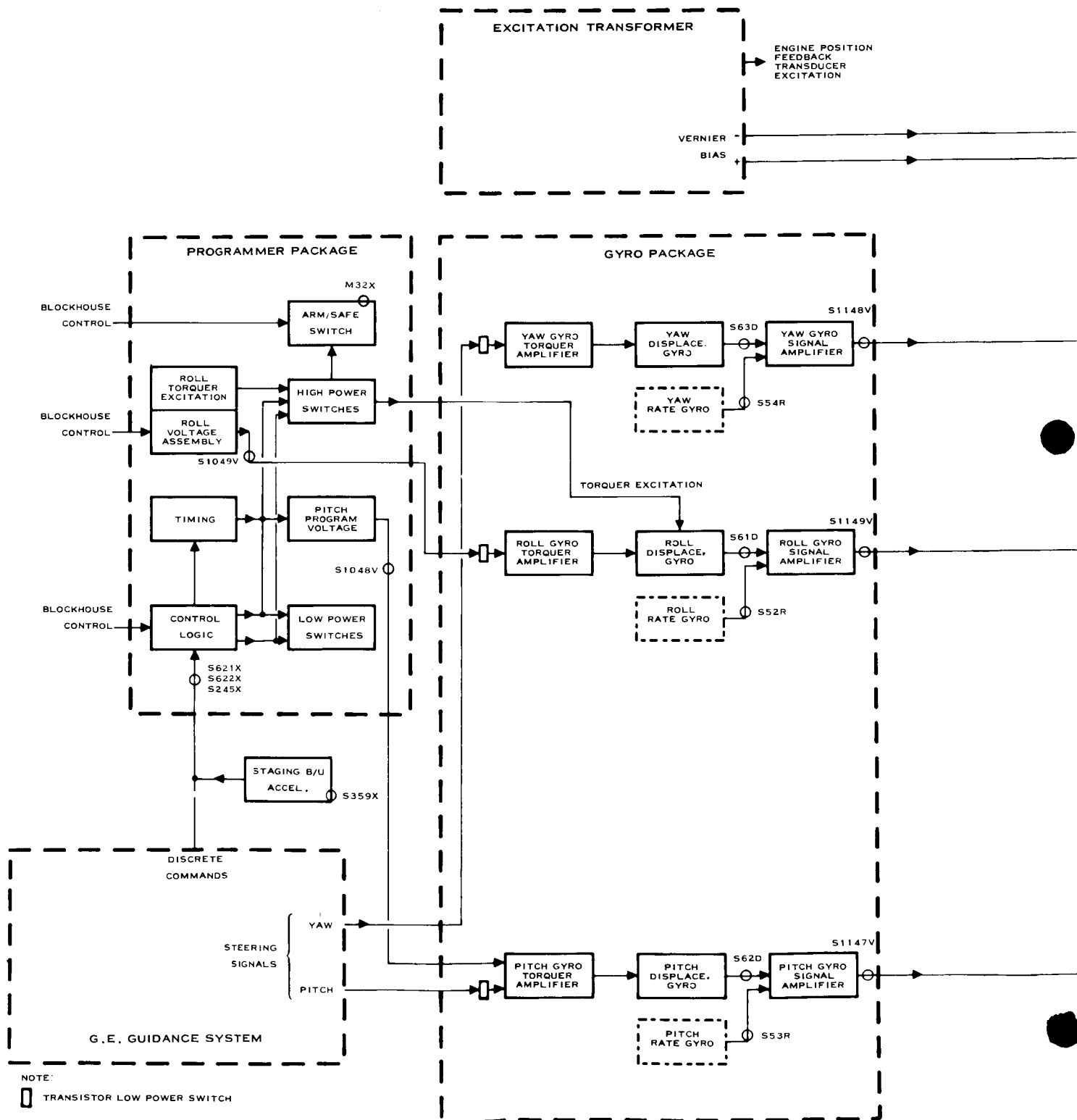
LAUNCH VEHICLE PERFORMANCE
 FIGURE NO. 8-4-15
 INTEGRATED REPORT NO. GDC/BKF65-042
 SUMMARY

FIRE L/V BOOSTER PROPULSION SYSTEM



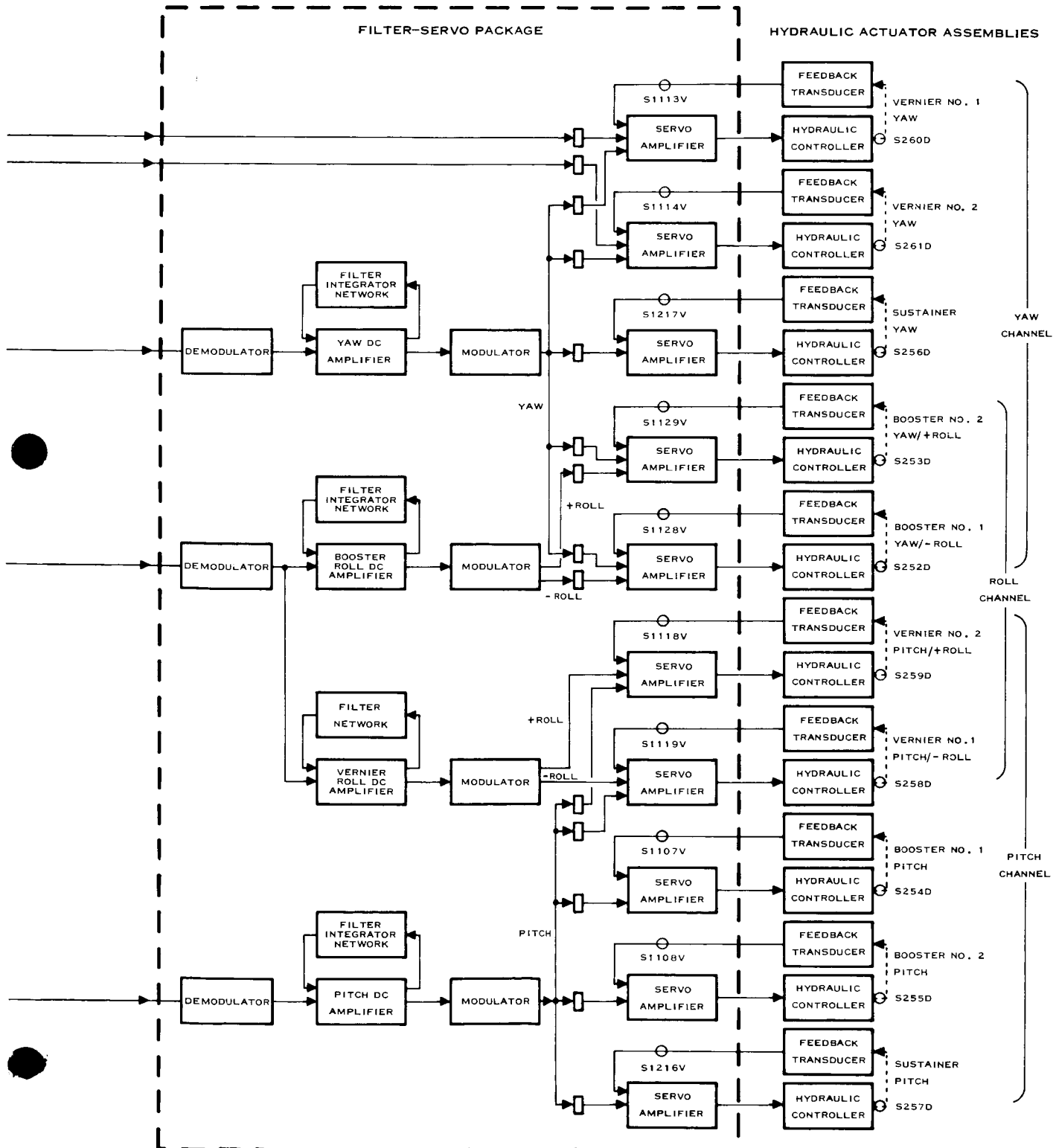
LAUNCH VEHICLE PERFORMANCE
 FIGURE NO. 8-4-16
 INTEGRATED REPORT NO. GDC/BKF65-042
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FIRE L/V FLIGHT CONTROL SYSTEM



LAUNCH VEHICLE PERFORMANCE
 FIGURE NO. 8-4-17
 INTEGRATED REPORT NO. GDC/BKF65-042
 SUMMARY

FIRE L/V FLIGHT CONTROL SYSTEM



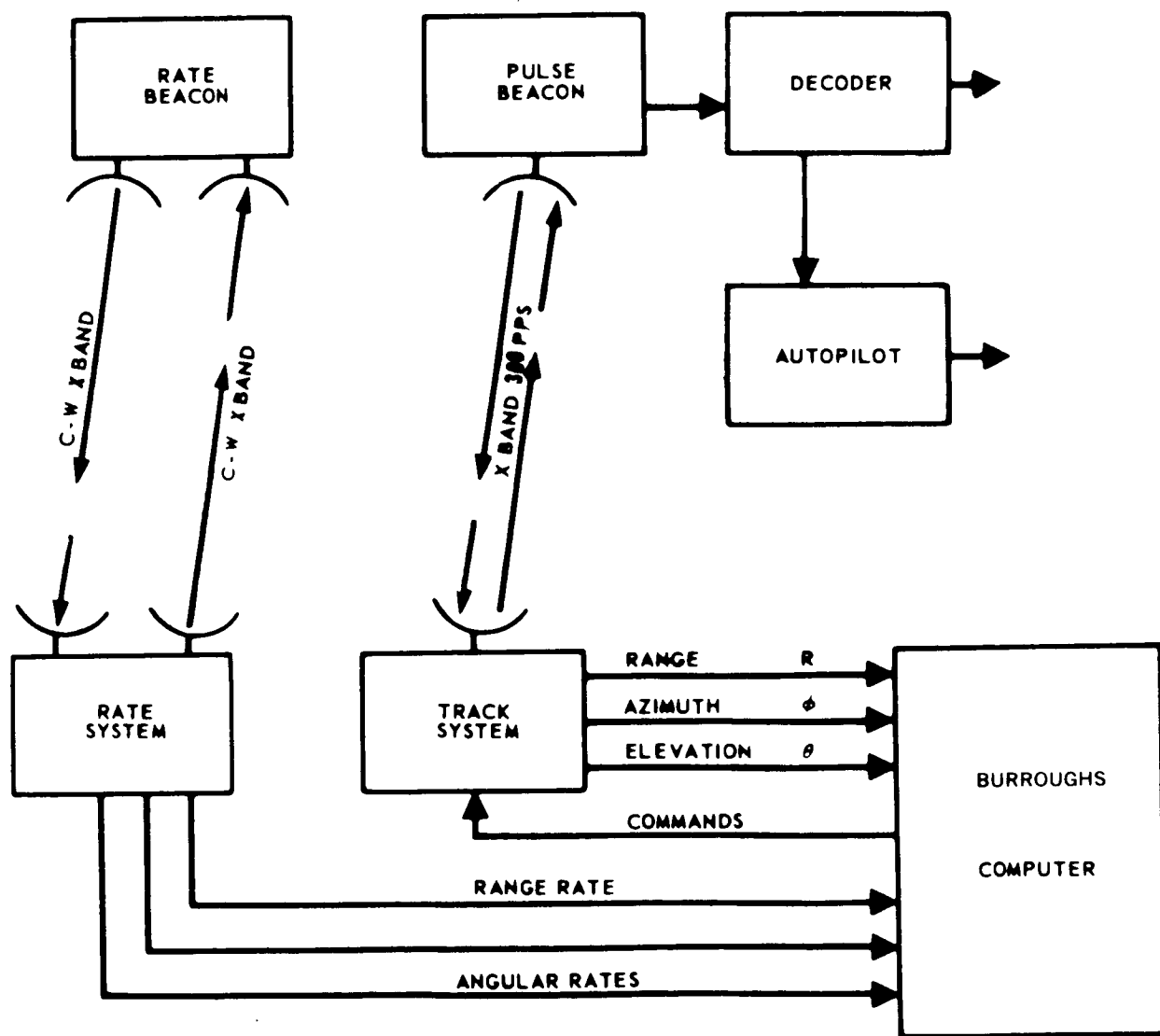
LAUNCH VEHICLE PERFORMANCE

FIGURE NO. 8-4-18

INTEGRATED REPORT NO. GDC/BKF65-042

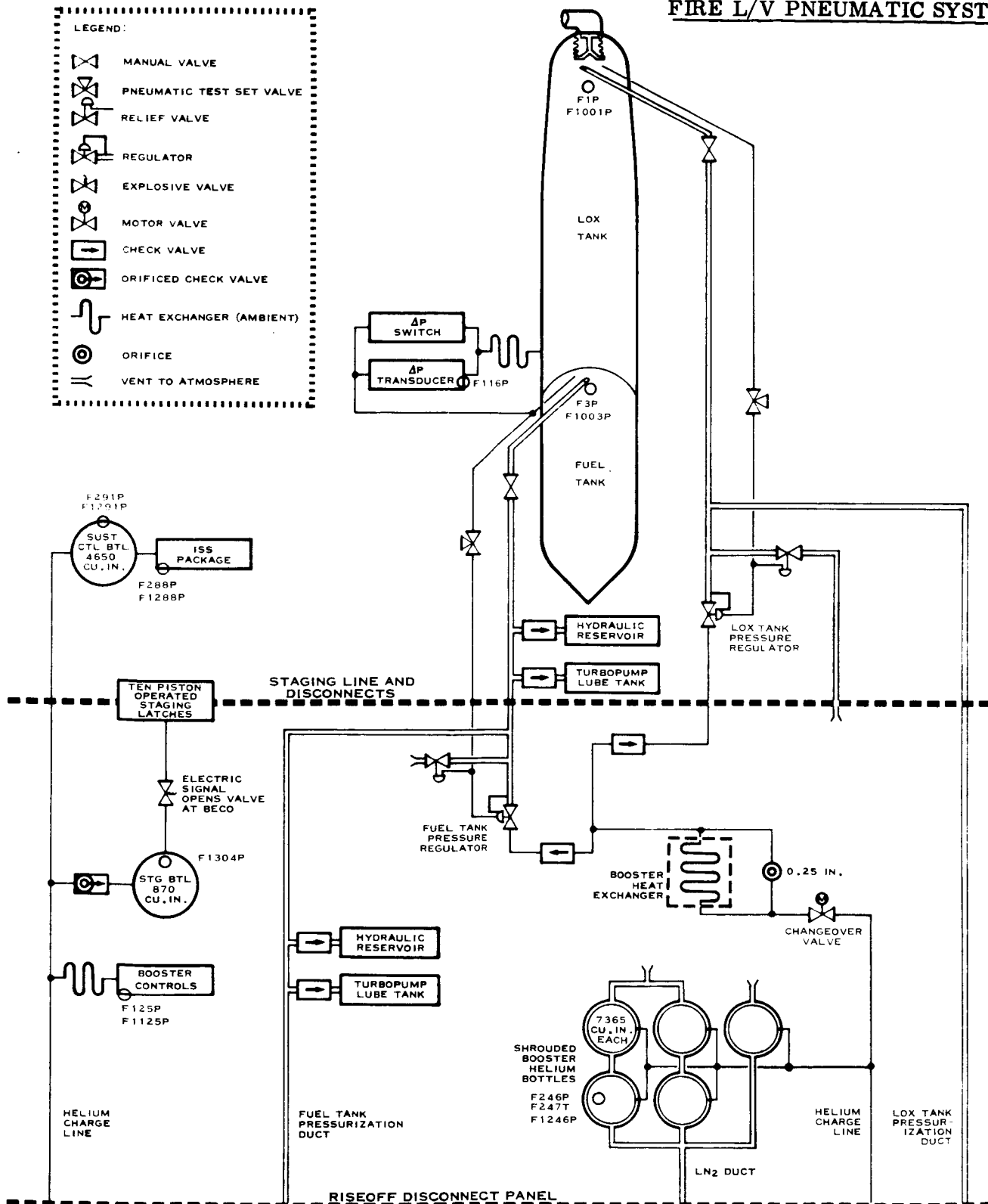
SUMMARY

MOD III GUIDANCE SYSTEM



LAUNCH VEHICLE PERFORMANCE
 FIGURE NO. 8-4-19
 INTEGRATED REPORT NO. GDC/BKF65-042
 SUMMARY

FIRE L/V PNEUMATIC SYSTEM



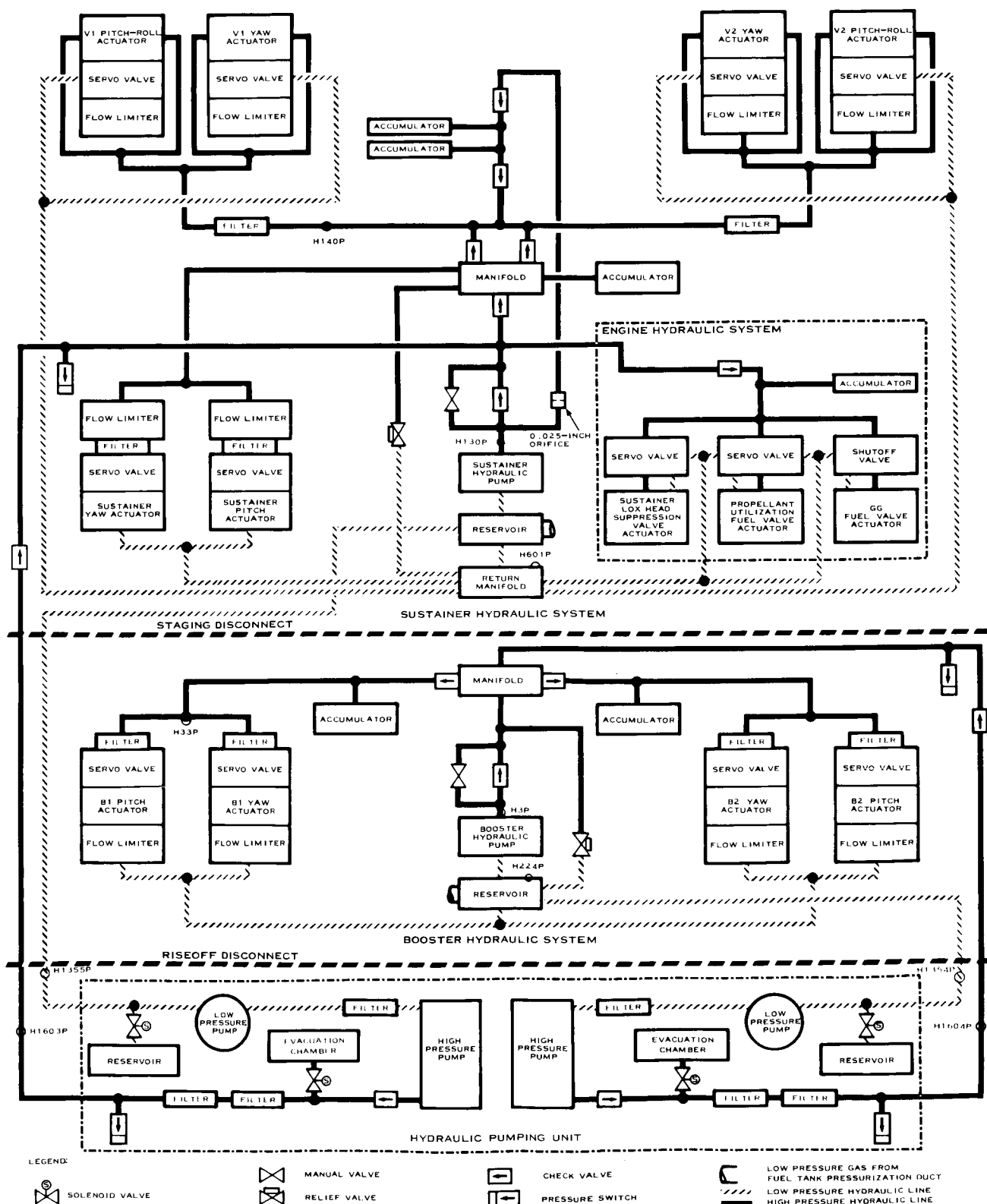
LAUNCH VEHICLE PERFORMANCE

FIGURE NO. 8-4-20

INTEGRATED REPORT NO. GDC/BKF65-042

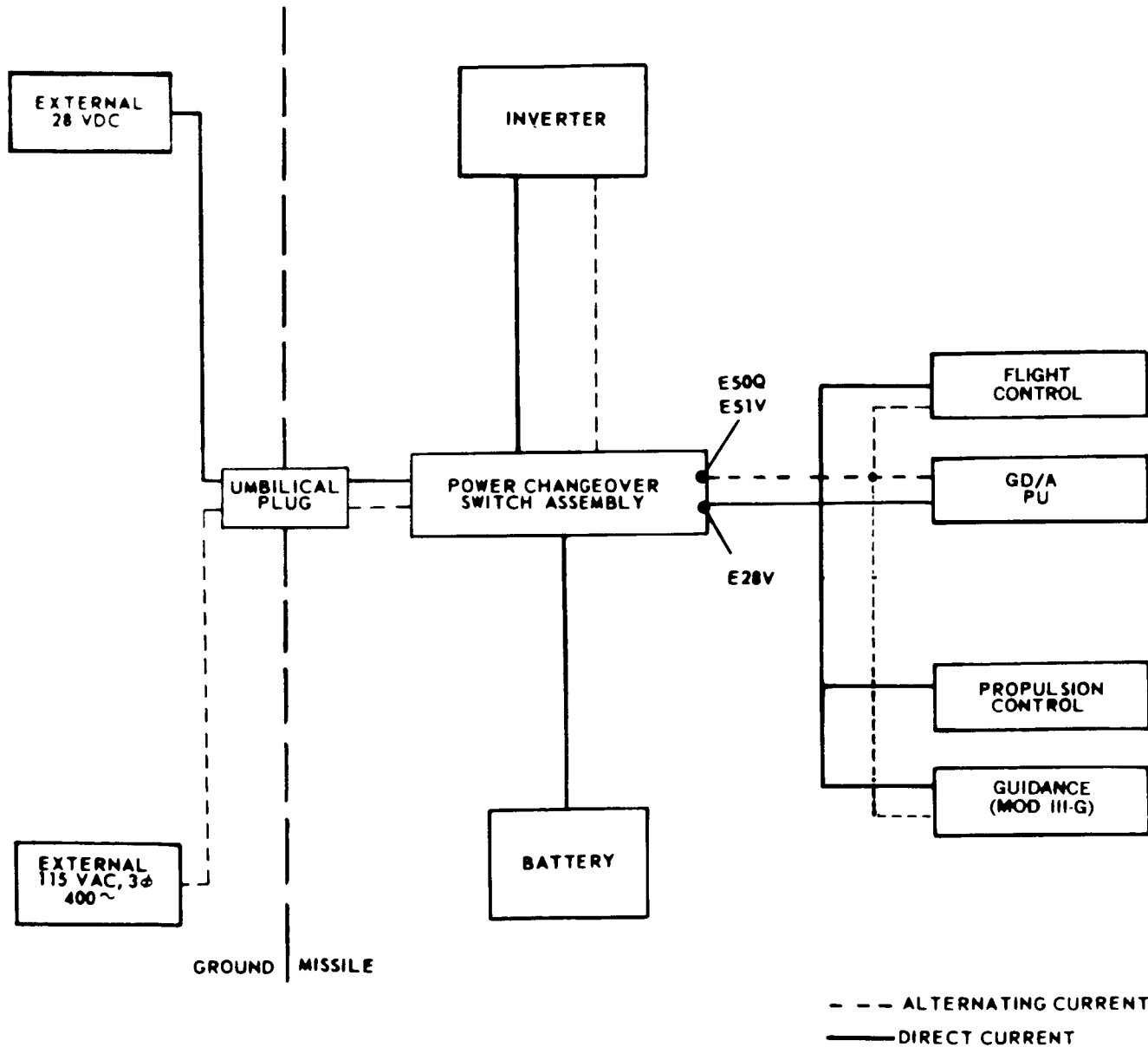
SUMMARY

FIRE L/V HYDRAULIC SYSTEM



LAUNCH VEHICLE PERFORMANCE
FIGURE NO. 8-4-21
INTEGRATED REPORT NO. GDC/BKF65-042
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FIRE L/V ELECTRICAL SYSTEM



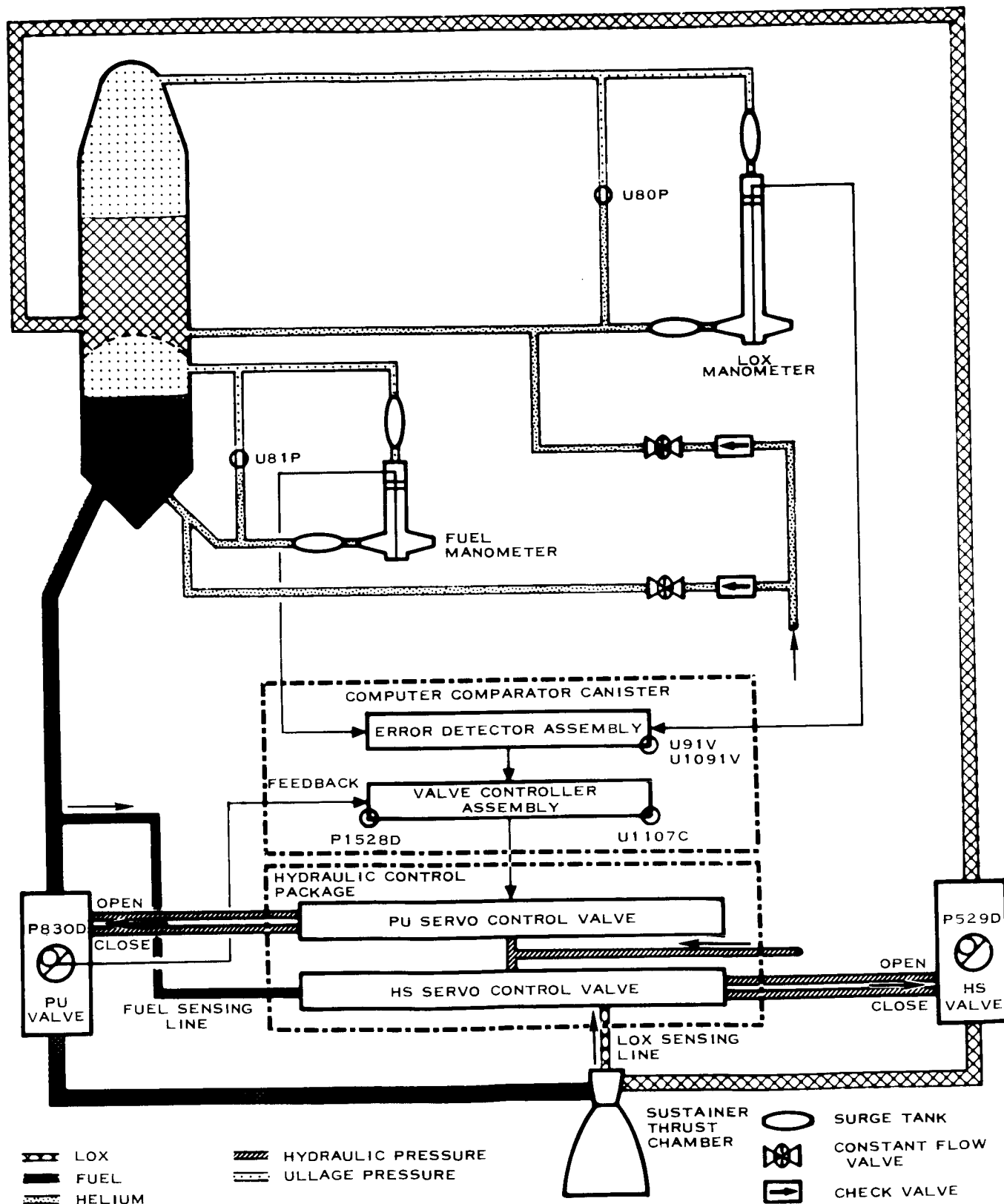
LAUNCH VEHICLE PERFORMANCE

FIGURE NO. 8-4-22

INTEGRATED REPORT NO. GDC/BKF65-042

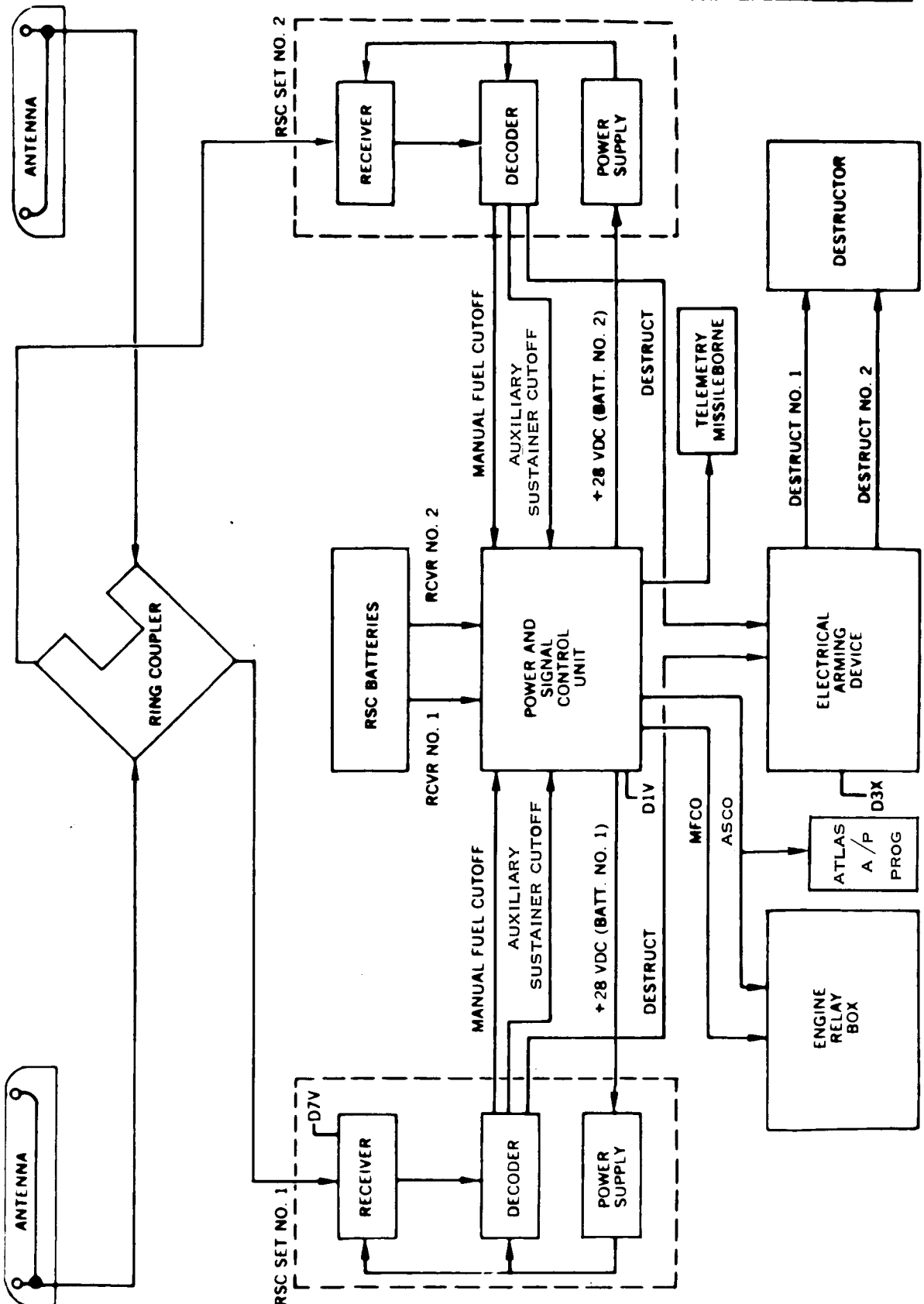
SUMMARY

PROPELLANT UTILIZATION SYSTEM



LAUNCH VEHICLE PERFORMANCE
 FIGURE NO. 8-4-23
 INTEGRATED REPORT NO. GDC/BKF65-042
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RANGE SAFETY COMMAND SYSTEM



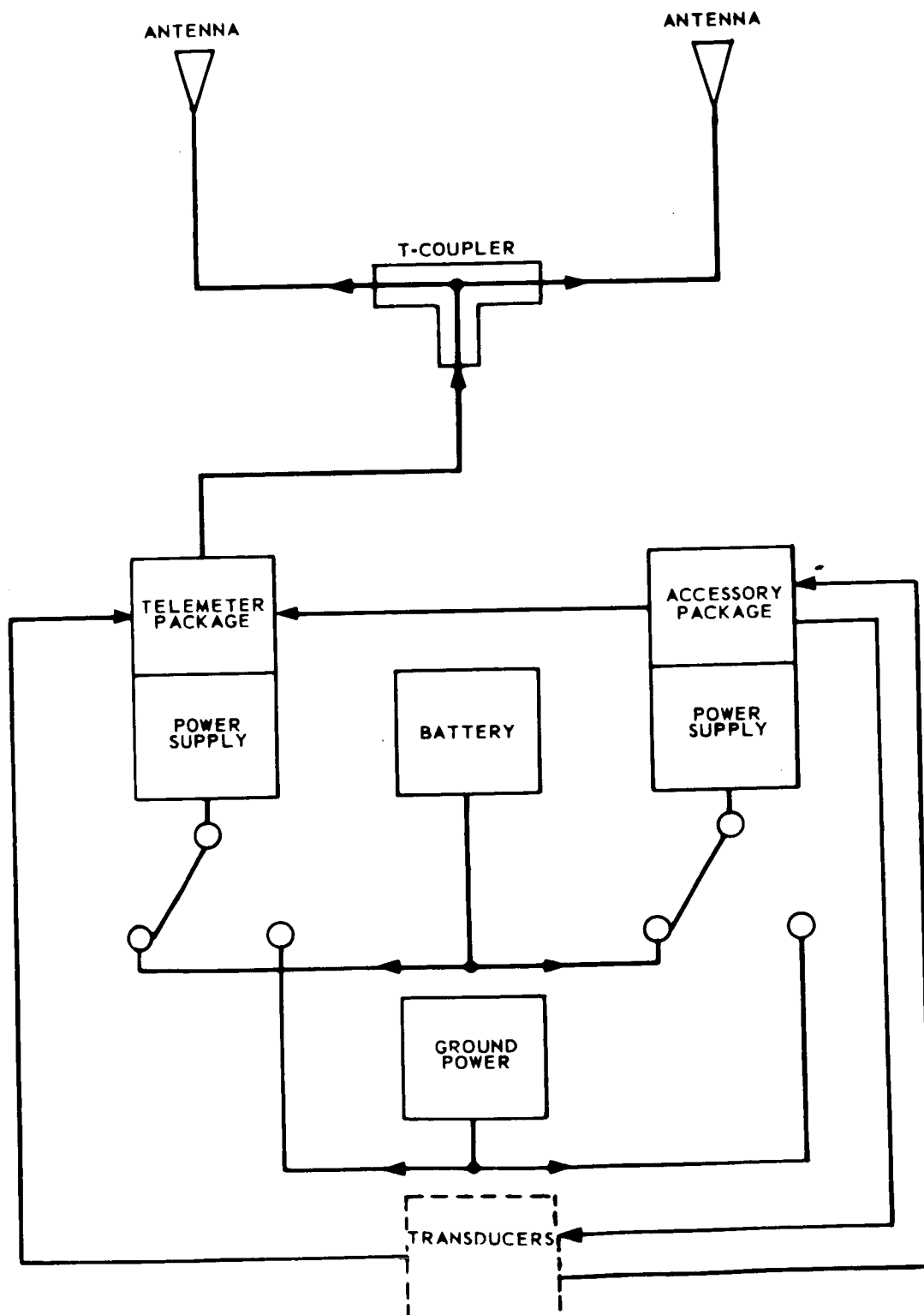
LAUNCH VEHICLE PERFORMANCE

FIGURE NO. 8-4-24

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TELEMETRY SYSTEM



PART 9

PREFLIGHT EVENTS

GENERAL DYNAMICS CONVAIR

INTEGRATED REPORT NO. GDC/BKF65-042

APPROVED BY: L. E. Munson

L. E. MUNSON

ASSISTANT PROGRAM DIRECTOR

FIRE PROGRAM OFFICE

SECTION 1

SPACE VEHICLE COUNTDOWN

Vehicle 264D was launched on the fourth attempt. The launch countdown (GDC Test P2-404-00-264) was scheduled for a 240-minute range countdown with a planned hold of 60 minutes at T-45 minutes, for a total countdown duration of 300 minutes. The countdown was initiated at 1022 EST on 22 May 1965 and required 393 minutes through vehicle liftoff. The 93 minutes of additional countdown time was due to an extension of the planned hold at T-45 minutes caused by "No Go" downrange weather. There were no GDC holds during the countdown and vehicle liftoff (two-inch-motion) occurred at 1654:59.703 hours EST.

FIRST LAUNCH ATTEMPT

The first launch attempt (GDC Test P2-401-00-264) was initiated at 1220 hours EST on 4 May 1965. After 10 minutes of countdown, a 100-minute hold was called by GDC to replace the sustainer hydraulic pump. The countdown was then resumed and proceeded to T-45 minutes where the planned 1-hour hold was initiated (at 1715 hours EST). After a total hold of 69 minutes, the launch attempt was aborted due to bad downrange weather.

SECOND LAUNCH ATTEMPT

The second launch attempt (GDC Test P2-402-00-264) was initiated at 1321 hours EST on 5 May 1965. At T-27 minutes (1654 EST), an estimated 15-minute hold was called to evaluate the downrange weather. The hold was extended several times until 1810 EST. At this time an estimated 90-minute extension of the hold was called and the count was re-cycled to T-40 minutes. The extended hold and re-cycled count was required because a velocity package environmental air access door prop had fallen out. Re-installation of the door prop required bringing the service tower back to the launcher and detanking LO₂ from the LV-3A booster. Beginning at 1948 EST, this hold was extended several times in order to evaluate downrange weather conditions. The countdown was resumed at 2032 EST after a hold duration of 218 minutes.

At T-22 minutes (2050 EST), a hold was called in order to verify the lox storage tank pressure. This necessitated sending a man to the storage tank area to read the pressure gauge. The countdown was resumed at 2059 EST.

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COUNTDOWN

At T-12 minutes (2109 EST), an estimated 15-minute hold was called due to downrange weather conditions. The count was re-cycled to T-27 minutes and holding at 2115 EST. After a hold of 19 minutes, the test was aborted at 2128 hours EST due to bad downrange weather conditions.

THIRD LAUNCH ATTEMPT

The third launch attempt (GDC Test P2-403-00-264) was initiated at 1022 hours EST on 21 May 1965. This test proceeded normally to T-45 minutes, when the planned 60-minute hold was initiated. The launch attempt was aborted after 34 minutes of hold due to bad downrange weather conditions.

SECTION 2L/V PREFLIGHT ACTIVITIES

The following tabulation presents Vehicle 264D milestones from the time of arrival at ETR until launch.

<u>Date (1965)</u>	<u>Event</u>
2 March	Vehicle arrived at Cape Kennedy, ETR.
6 April	Vehicle erected at Complex 12, ETR.
19 April	LV-3A tanking
21 April	Booster FACT.
27 April	RFI testing
28 April	RFI testing rerun to support spacecraft.
29 April	Joint FACT
4 May	Attempted launch. Aborted due to downrange weather.
5 May	Attempted launch. Aborted due to downrange weather.
14 May	Booster FACT.
17 May	Joint FACT.
21 May	Third Launch attempt. Aborted due to downrange weather.
22 May	Fourth launch attempt. Successful.

PART 10

APPENDIX

GENERAL DYNAMICS CONVAIR

INTEGRATED REPORT NO. GDC/BKF65-042

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})

APPENDIX

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GLOSSARY

J-FACT	- Joint Flight Acceptance Composite Test
KC	- Kilocycle (10^3)
KSC	- Kennedy Space Center
Liftoff	- Vehicle two-inch motion (Atlas)
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
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	- Re-entry Package
rpm	- revolutions per minute
R/S	- Re-entry 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

SECTION 1

GLOSSARY

•	
A	- Angstrom unit
ABL	- Allegheny Ballistics Laboratory
AGC	- Automatic Gain Control
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
ETR	- Eastern Test Range
° F	- Degrees Fahrenheit
FM	- Frequency Modulation
g	- unit acceleration of 32 ft/sec ²
GDC	- General Dynamics Convair
GE	- General Electric
HIRS	- High Impulse Retrorocket System
HSU	- Hydraulic Supply Unit
IBM	- International Business Machine
IBW	- Information Band Width
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

SECTION 2

REFERENCES

1. General Dynamics Convair Report GDA/BKF64-018, "Project FIRE Integrated Post Flight Evaluation Report, Flight I, (U)", dated 30 October 1964. (NASA CR 57017)

SECTION 2

REFERENCES

1. General Dynamics Convair Report GDA/BKF64-018, "Project FIRE Integrated Post Flight Evaluation Report, Flight I, (U)", dated 30 October 1964. (NASA CR 57017)

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ERRATA

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PROJECT FIRE INTEGRATED POST FLIGHT
EVALUATION REPORT

FLIGHT II

24 SEPTEMBER 1965

- Page No. vii : The last sentence of paragraph one should be changed to read as follows: The specific purpose of this flight was to obtain data on total and radiative heating and on radio signal attenuation during re-entry into the earth's atmosphere at a velocity near 37,000 feet per second.
- Page No. 8-4-5 : The title listed Vehicle 263D Nominal Pitch Program should be Vehicle 264D Nominal Pitch Program.
- Page No. 8-4-11 : Under Azusa System change IBM 7094 to read CDC 3600.

Issue date: 11-4-65

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