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THORAD-AGENA PERFORMANCE FOR THE NIMBUS III MISSION

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APPENDIXES

THORAD-AGENA PERFORMANCE FOR THE NIMBUS III MISSION

by Lewis Research Center

I. SUMMARY

The Thorad-Agena launch vehicle with Nimbus III spacecraft was successfully launched on the third attempt (previous attempts were made on April 10 and 11) from the Space Launch Complex 2-East, Vandenberg Air Force Base, California, on April 13, 1969, at 2354:03.136 Pacific standard time. The Thorad boosted the Agena-spacecraft into a suborbital coast ellipse. After separation of the Agena-spacecraft from the Thorad, the Agena engine was started and the Agena-spacecraft was injected into a near-polar transfer orbit with a perigee altitude of approximately 157 kilometers and an apogee altitude of approximately 1112 kilometers. After a 46-minute coast (to near the apogee of the transfer orbit), the Agena engine was restarted, and the Agena-spacecraft orbit was circularized. The Nimbus III was then successfully separated from the Agena, in the near-polar orbit, with a perigee altitude of 1087 kilometers and an apogee altitude of 1143 kilometers. The Agena cold-gas retrothrust system was then operated for two programmed maneuvers intended to lower the Agena orbit and insure noninterference with the Nimbus III during subsequent orbits.

A secondary payload, an Engineers Geodetic Ranging Satellite - 13, (EGRS-13) was also carried by the Agena. The EGRS-13 was successfully separated from the Agena approximately 48 minutes after the Nimbus III separation.

The Thorad and Agena vehicle systems performed satisfactorily through initiation of the first operation of the Agena retrothrust system. At the start of the first retrothrust the Agena began an unexpected pitch-up and at the end of this intended retrothrust the Agena had stabilized with a 150° pitch attitude error. This resulted in the Agena (and therefore the intended retrothrust vector) being improperly oriented during both the programmed operations of the retrothrust system. Consequently, the final orbits of the Agena and the EGRS-13 were slightly different than planned; the resultant orbits, however, were satisfactory, and all mission requirements were achieved.

This report contains an evaluation of the performance of the Thorad-Agena systems in support of the Nimbus III mission.

II. INTRODUCTION

The primary purpose of the Nimbus III mission was to perform weather research and scientific experimentation which will improve long-range weather forecasting and enable a better understanding of meteorological phenomena. A secondary payload, the Engineers Geodetic Ranging Satellite-13 (EGRS-13), was also carried to obtain geodetic data for the U.S. Army. The objective of the Thorad-Agena launch vehicle was to place the Nimbus III and the EGRS-13 into their proper near-polar, near-circular orbits.

The launch vehicle and the spacecraft integration effort to support the Nimbus Project was under the direction of Lewis Research Center. Nimbus III was the third of a series of planned missions. Nimbus I and Nimbus II, launched in August 1964 and May 1966, respectively, were successfully placed into near-polar orbits. In May 1968, the Thorad for Nimbus B malfunctioned shortly after launch and was destroyed in flight. Nimbus III replaced the Nimbus B mission. A Thor-Agena B was used to launch Nimbus I, and a Thrust Augmented Thor-Agena B was used to launch Nimbus II. A Thorad-Agena D vehicle was used for the Nimbus B and the Nimbus III missions. The Thorad provided greater payload capability than the boosters used for previous Nimbus missions. All Nimbus missions were launched from the Western Test Range, Vandenberg Air Force Base, California.

The butterfly shaped weather-eye Nimbus III satellite weighed 576 kilograms (1269 lbm), a record weight for meteorological satellites. The EGRS-13 weighed 21 kilograms (47 lbm).

This report discusses and evaluates the performance of the Thorad-Agena launch vehicle systems for the Nimbus III mission from lift-off through the second operation of the Agena retrothrust system.

III. LAUNCH VEHICLE DESCRIPTION

by Eugene E. Coffey and Richard P. Geye

The Thorad-Agena is a two-stage launch vehicle consisting of a Thorad first-stage and an Agena second-stage, connected by a booster adapter. The composite vehicle (fig. III-1) including the shroud and the booster adapter is about 33 meters (109 ft) long. The total weight at lift-off is approximately 91 625 kilograms (202 000 lbm). Figure III-2 shows the Thorad-Agena lift-off with Nimbus III.

The Thorad (fig. III-3) stage consists of a long-tank Thor and three Castor II solid-propellant rocket motors located 120° apart and attached to the long-tank Thor near the aft end. The long-tank Thor is 21.4 meters (70.3 ft) long and is 2.4 meters (8 ft) in diameter, except for the conical forward section which tapers to a diameter of about 1.6 meters (5.3 ft). The Thorad is approximately 4.3 meters (14 ft) longer than Thors previously used resulting in approximately 50 percent greater propellant tank volume. The additional propellant provides for an increased main engine thrust duration of approximately 72 seconds. The solid-propellant rocket motors are each about 7 meters (24 ft) long and are 0.8 meter (2.5 ft) in diameter with a conical forward end. The Thorad is powered by a main engine with a sea-level rated thrust of 756×10^{3} newtons (170 000 lbf), two vernier engines with a total sea-level rated thrust of 89×10^{2} newtons (2000 lbf) and the three solid-propellant rocket motors with a total sea-level rated thrust of 696×10^{3} newtons (156 450 lbf). The total impulse of these solid-propellant rockets is approximately 3829×10^{3} newton-seconds (861×10^{3} lbf-sec) greater than the total impulse of the solid-propellant rocket motors previously used on the Thrust Augmented Thor vehicle. The propellants for the Thorad main engine and the vernier engines are liquid oxygen and high grade kerosene. The propellant for the solid-propellant rocket motors is basically a solid grain of polybutadiene acrylic acid and ammonium perchlorate. The vernier engines, the main engine, and the solid-propellant rocket motors are ignited in sequence prior to lift-off. The fixed-nozzle solid-propellant rocket motors burn for approximately 39 seconds. They are jettisoned at T + 102 seconds in order to insure that the cases for the solid propellant impact in a safe area (water impact). The Thorad main-engine thrusts until the desired velocity for the planned suborbital ellipse is achieved as determined by the radio-guidance system, or until propellant depletion. During powered flight, the Thorad main engine is gimbaled for pitch and yaw control, and the vernier engines are gimbaled for roll control. After Thorad main engine cutoff, the vernier engines continue to thrust for 9 seconds to provide for vehicle attitude control and for fine trajectory corrections. After vernier engine cutoff, the Thorad is severed from the Agena by the firing of a Mild Detonating Fuse system located on the forward end of the booster adapter. The firing of a retro-rocket system, mounted on the booster adapter, then separates the Thorad with booster adapter from the Agena.

The second stage Agena and the shroud protecting the Nimbus III spacecraft are shown in figure III-4. The diameter of the Agena is 1.52 meters (5 ft), and the length of the Agena and shroud is about 12 meters (40 ft). The Agena engine has a rated vacuum thrust of 71.17×10^3 newtons (16 000 lbf). This engine uses unsymmetrical dimethylhydrazine and inhibited red fuming nitric acid as propellants. During powered flight, pitch and yaw control are provided by gimbaling the Agena engine, and roll control is provided by a cold gas (mixture of nitrogen and tetrafluoromethane) attitude control system. During periods of nonpowered flight, pitch, yaw, and roll control are provided by the cold-gas system. The cold-gas attitude control system is also used to perform a 75° pitchup maneuver prior to Nimbus III separation and a coning (combined yaw and roll) maneuver subsequent to Nimbus III separation. For this mission a cold-gas retrothrust system provided the impulse (two separate thrust periods) required to perturb the Agena orbit after the coning maneuver. A fiber glass laminate clamshell shroud was used to provide environmental protection for the spacecraft during ascent. This shroud was jettisoned approximately 10 seconds after Agena engine first start. The Nimbus III is shown in figure III-5.

The Thorad used for the Nimbus III mission provided approximately 20 percent more payload capability than was provided by the Thrust Augmented Thor used for previous NASA missions.

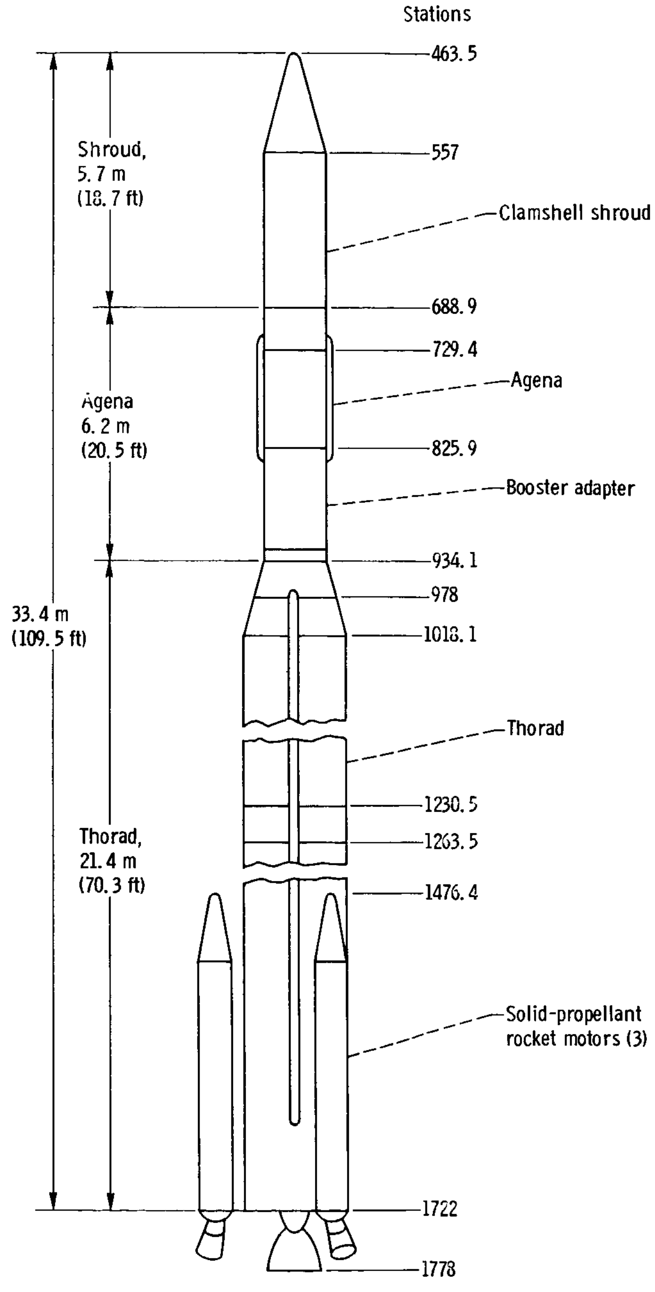


Figure III-1. - Composite space vehicle, Nimbus III.

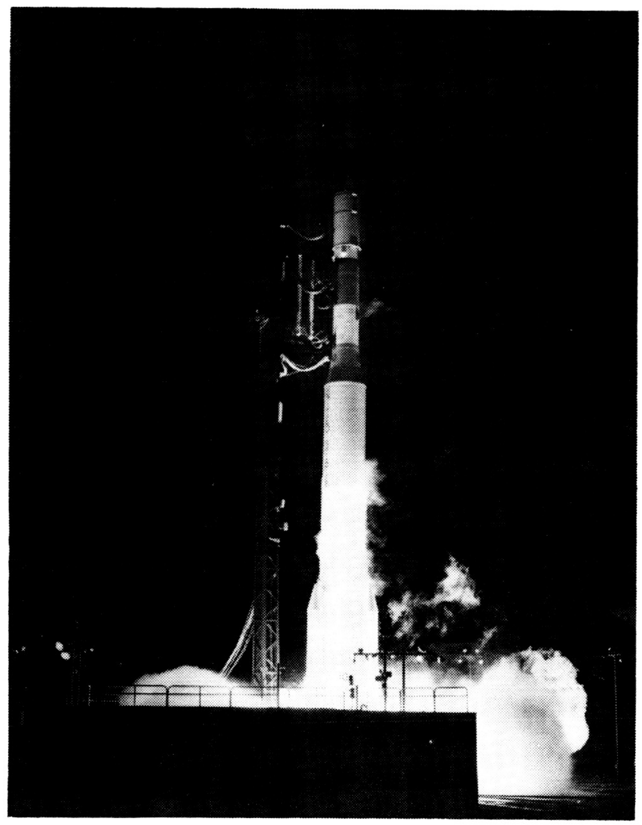


Figure III-2. - Thorad-Agena lift-off with nimbus III.

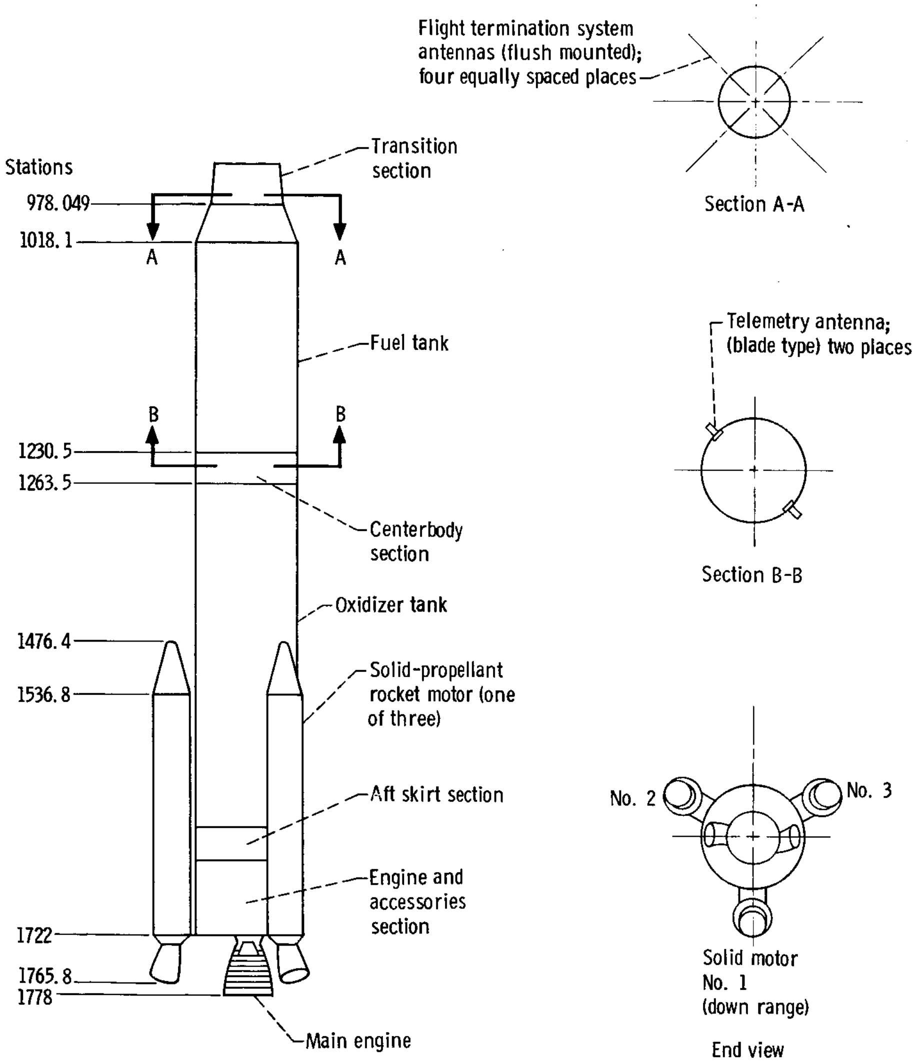


Figure III-3. - Thorad general configuration, Nimbus III.

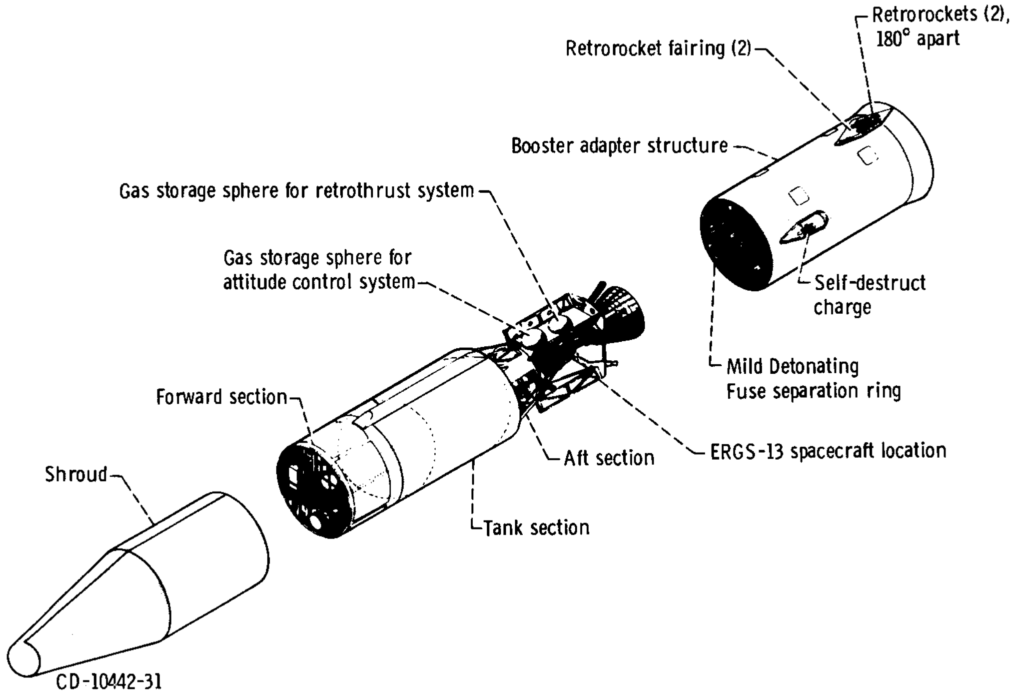


Figure III-4. - Agena, shroud, and booster adapter, Nimbus III.

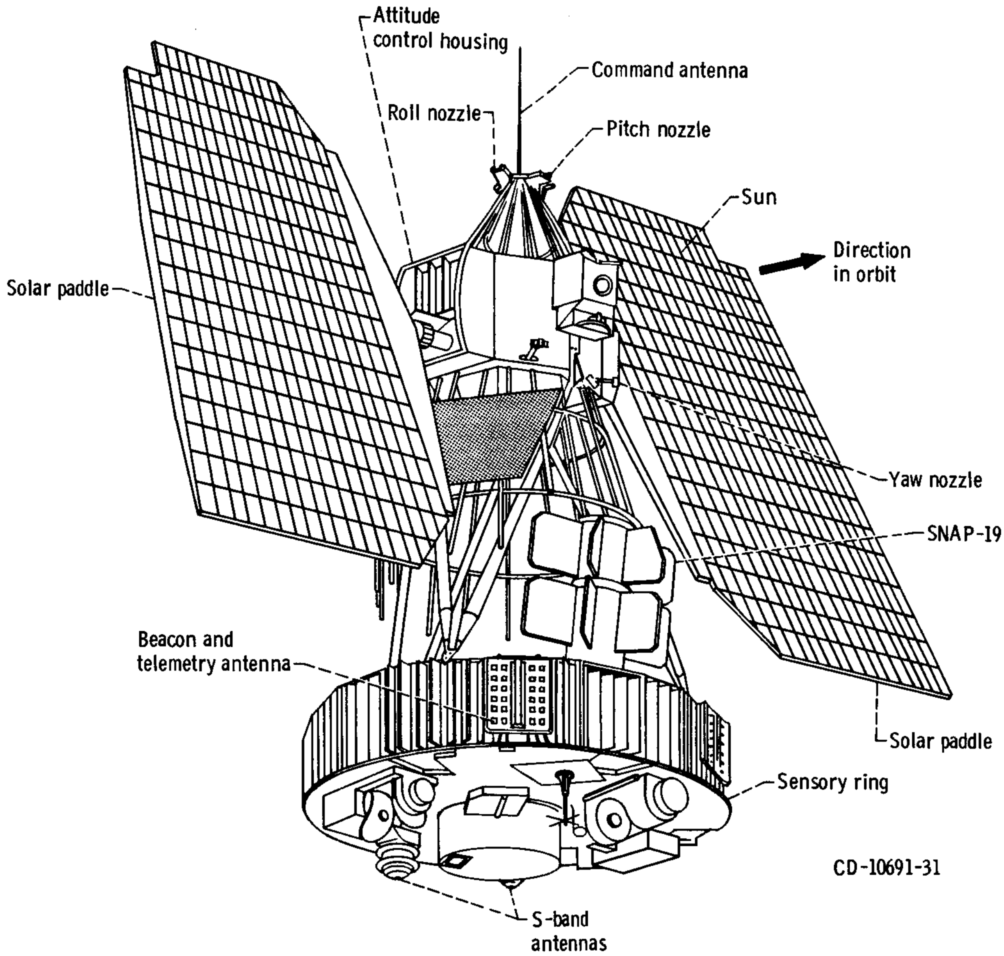


Figure III-5. - Nimbus III spacecraft in deployed configuration.

IV. TRAJECTORY AND PERFORMANCE

by James C. Stoll

Nimbus III was successfully launched from the Space Launch Complex 2-East Vandenberg Air Force Base on April 13, 1969, at 2354:03.136 Pacific standard time. Actual and expected times for major flight events are given in appendix A.

TRAJECTORY PLAN

The Thorad boosts the Agena - Nimbus III onto a prescribed subortital coast ellipse. Following Thorad-Agena separation, the Agena engine is started and thrusts until it places the Agena-Nimbus III onto a transfer ellipse of 153.72 kilometers (83 n mi) perigee altitude and 1111.20 kilometers (600 n mi) apogee altitude. The Agena-Nimbus III coasts to near the apogee of the transfer ellipse where the Agena engine is restarted. The Agena engine then thrusts for a second time to place the Agena-Nimbus III onto a near-polar circular orbit at an altitude of 1111.20 kilometers (600 n mi). The Nimbus III spacecraft is separated from the Agena about 250 seconds after the Agena engine second cutoff. After Nimbus III separation, the Agena retrothrust system is operated at two separate times. During the second operation of the retrothrust system, a secondary payload Engineers Geodetic Ranging Satellite (EGRS-13) is separated from the Agena. The two operations of the retrothrust system are performed to insure that the Agena and the EGRS-13 will not interfere with the Nimbus III or with each other during subsequent orbits. The planned ascent profile and orbit configurations for this mission are shown in figure IV-1.

TRAJECTORY RESULTS

Winds Aloft

The winds aloft at launch were light and predominately from the west with a peak velocity of 39.01 meters per second (128 ft/sec) occurring at an altitude of 12 192 meters (40 000 ft). Wind data are shown in figure IV-2. The wind shears produced by abrupt changes in wind velocity were not severe.

The T - 0 (lift-off) weather balloon data were used to predict the maximum vehicle bending response and the maximum gimbal angle. The maximum vehicle bending response was calculated to be 46.4 percent of the critical value at Thorad station 1229.89 and to occur at an altitude of 3688.4 meters (12 101 ft). The maximum booster gimbal angle was calculated to be 30.0 percent of the total available gimbal angle in the pitch plane and to occur at an altitude of 1261.0 meters (4137 ft).

Thorad Boost Phase

Lift-off occurred from a launch-pad azimuth of 259.5°. At T + 2.14 seconds, the vehicle began a programmed roll maneuver to achieve a launch azimuth of 193.8°. At T + 16.20 seconds the roll maneuver was terminated, and the vehicle started to pitch downrange at the programmed pitch rates.

The three solid-propellent rocket motors burned out at about T + 39.2 seconds but were not jettisoned until T + 102.2 because of range safety considerations. At the time of jettison the actual trajectory of the vehicle was about 152.4 meters (500 ft) higher than, 762.0 meters (2500 ft) downrange of, and 670.6 meters (2200 ft) left of the predicted trajectory. The deviations in altitude and range were insignificant. The deviation to the left resulted from both the Thorad control system tolerances and the crosswinds.

The radio-guidance system steered the Thorad from T + 124.3 seconds to T + 218.9 seconds. The trajectory was designed for Thorad engine cutoff to occur by either radio-guidance system command or propellant depletion. For this mission, cutoff occurred by propellant (oxidizer) depletion at T + 222.20 seconds which was 0.56 second later than the predicted cutoff time. At this time the actual trajectory was about 274.3 meters (900 ft) higher than, 2590.8 meters (8500 ft) downrange of, and 457.2 meters (1500 ft) left of the predicted trajectory. Also, at main engine cutoff the velocity of the vehicle was 37.8 meters per second (124 ft/sec) lower than expected; the residual fuel was 381.5 kilograms (841 lbm), and the propellant utilization was 99.43 percent. The vehicle performance at main engine cutoff was within allowable tolerances. The Thorad vernier engine thrust was terminated 9 seconds after main engine cutoff. The insertion parameters at vernier engine cutoff are listed in table IV-I, and the resulting coast ellipse parameters are listed in table IV-II. The Thorad-Agena separation was commanded by the radio-guidance system and occurred at T + 238.1 seconds. The total performance of the Thorad was satisfactory.

Agena First Powered Phase

The Agena timer was started by Thorad main engine cutoff at T + 222.2 seconds, 0.6 second later than planned; all subsequent Agena timer controlled events were 0.6 second later than planned but were within the ±0.2-second tolerance of the timer. After Thorad - Agena separation, the Agena performed a pitch-down maneuver to place the vehicle in the proper attitude for the Agena engine first start. The Agena engine was started by the Agena timer at T + 256.2 seconds and 90 percent chamber pressure was achieved 1.2 seconds later.

The radio-guidance system determined that the Agena trajectory should be lofted and the Agena thrust duration extended to compensate for the lower than expected orbit energy at Thorad main engine cutoff. Accordingly, the radio-guidance system commanded the Agena to pitch-up and fly a slightly lofted trajectory (the maximum resultant vehicle displacement was about 3° at T + 310 sec). The radio-guidance pitch steering commands started at T + 269.9 and were terminated at T + 389.1 seconds (Yaw steering was negligible).

During the early portion of the Agena first powered phase (prior to T + 389 sec) the radio-guidance system determined that the Agena engine thrust was slightly higher than expected. Accordingly, the radio-guidance system combined the corrections needed to compensate for the high thrust of the Agena, the low orbit energy at Thorad cutoff, and the 0.6-second late start of the Agena timer. Then the system issued the command to enable the velocity meter at T + 390.5 seconds. The Agena velocity meter then commanded engine cutoff when the proper velocity increment preset in the velocity meter had been gained at T + 488.0 seconds. Thrust decay added 7.99 meters per second (26.2 ft/sec) velocity compared with the predicted 6.43 meters per second (21.1 ft/sec) set in the velocity meter. The injection parameters at Agena engine first cutoff are given in table IV-III. The Agena-Nimbus III transfer orbit parameters are given in table IV-IV.

Agena Second Powered Phase

After the Agena engine first cutoff, the Agena-Nimbus III coasted for 2773.2 seconds to a point near the apogee of the transfer ellipse. At this point the Agena engine was restarted (at T + 3261.2 sec) and 90 percent chamber pressure was attained 1.1 seconds later. Thrust duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 5.1 seconds, 0.18 second less than nominal. Agena engine cutoff by the velocity meter indicated that the proper velocity increment had been gained. Thrust decay velocity increment was 7.99 meters per second (26.2 ft/sec) as compared with the predicted 9.42 meters per second (30.9 ft/sec) included in the velocity meter setting. Injection parameters at Agena engine second cutoff are given in table IV-V.

Post Second Burn Phase

At T + 3517.1 seconds the Nimbus III was separated from the Agena, and the first operation of the Agena retrothrust system was started at T + 3710.2 seconds. During this operation of the retrothrust system, unexpected torquing about the pitch axis exceeded the correction capability of the attitude control system. (See the AGENA PROPULSION SYSTEM section of this report for the cause of the torque, and the AGENA GUIDANCE AND FLIGHT CONTROL section for effects on vehicle orientation.) This resulted in the Agena (and therefore the intended retrothrust vector) being improperly oriented for both operations of the retrothrust system. The second operation of the retrothrust system was started at T + 6217.21. Because of the improper orientation of the Agena during the retrothrust system operation the Agena final orbit was above the Nimbus III orbit instead of below as planned. Nevertheless, there is essentially no possibility of the Agena colliding with the Nimbus III as can be seen by noting the apogee and perigee radii of the Agena and the Nimbus III in table IV-VI.

The EGRS-13 satellite was separated at T + 6384.6 seconds during the second operation of the Agena retrothrust system. The Nimbus III and the EGRS-13 were placed in satisfactory orbits. The final orbit parameters of the Agena, Nimbus III, and the EGRS-13 are given in table IV-VI.

TABLE IV-I. - INSERTION PARAMETERS AT VERNIER ENGINE CUTOFF, NIMBUS II

Altitude, km (n mi)	104.545 (56.45)
Range, km (n mi)	263.558 (142.31)
Velocity, m/sec (ft/sec)	3894.856 (12 778.400)
Inclination, deg	97.310
Azimuth, deg	188.655
Flight path angle, deg	12.855
Radius, km (n mi)	6476.598 (3497.083)

TABLE IV-II. - SUBORBITAL COAST ELLIPSE PARAMETERS AT APOGEE, NIMBUS III

Radius, km (n mi)	6528.581 (3525.152)
Altitude, km (n mi)	155.261 (83.834)
Velocity, m/sec (ft/sec)	3766.895 (12 358.58)
Inclination, deg	97.309
Eccentricity, dimensionless	0.76759

TABLE IV-III. - INJECTION PARAMETERS AT AGENA ENGINE FIRST CUTOFF, NIMBUS III

Altitude, km (n mi)	157.827 (85.22)
Range, km (n mi)	1597.443 (862.55)
Velocity, m/sec (ft/sec)	8072.427 (26 484.34)
Azimuth, deg	190.614
Inclination, deg	99.924
Flight path angle, deg	0.438
Radius, km (n mi)	6533.315 (3527.708)

TABLE IV-IV. - AGENA TRANSFER ORBIT PARAMETERS, NIMBUS III

Apogee altitude, km (n mi)	1112.102 (600.487)
Apogee radius, km (n mi)	7485.803 (4042.01)
Perigee altitude, km (n mi)	156.744 (84.635)
Perigee radius, km (n mi)	6530.30 (3526.08)
Period, min	97.325
Inclination, deg	99.924
Eccentricity, dimensionless	0.06817

TABLE IV-V. - INJECTION PARAMETERS AT AGENA ENGINE SECOND CUTOFF, NIMBUS III

Altitude, km (n mi)	1112.311 (600.60)
Velocity, m/sec (ft/sec)	7295.217 (23 934.44)
Azimuth, deg	-11.434
Inclination, deg	99.925
Flight path angle, deg	0.1937
Radius, km (n mi)	7485.224 (4041.697)

TABLE IV-VI. - FINAL ORBIT PARAMETERS,

NIMBUS III

Parameters and units	Nimbus III	EGRS-13	Agena
Apogee altitude			
km	1142.821	1140.697	1139.763
n mi	617.074	615.927	615.423
Apogee radius			
km	7510.164	7507.078	7512.566
n mi	4055.164	4053.498	4056.461
Perigee altitude			
km	1086.550	1084.613	1091.671
n mi	586.690	585.644	589.455
Perigee radius			
km	7450.233	7447.486	7460.441
n mi	4022.804	4021.321	4028.316
Period			
min	107.339	107.279	107.459
Inclination			
deg	99.922	99.911	99.903
Eccentricity			
dimensionless	0.004006	0.003985	0.003481

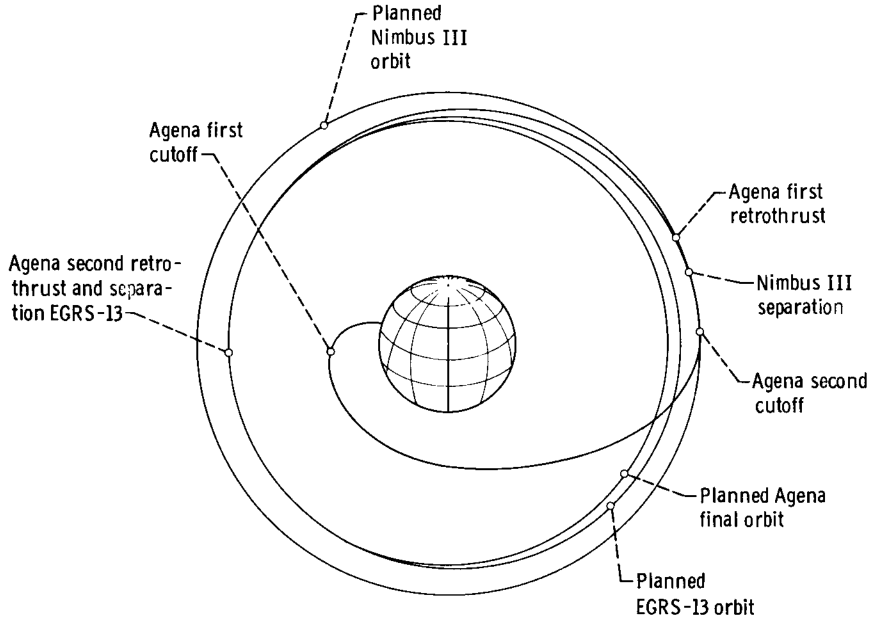


Figure IV-1. - Planned ascent profile and orbit configurations, Nimbus-III.

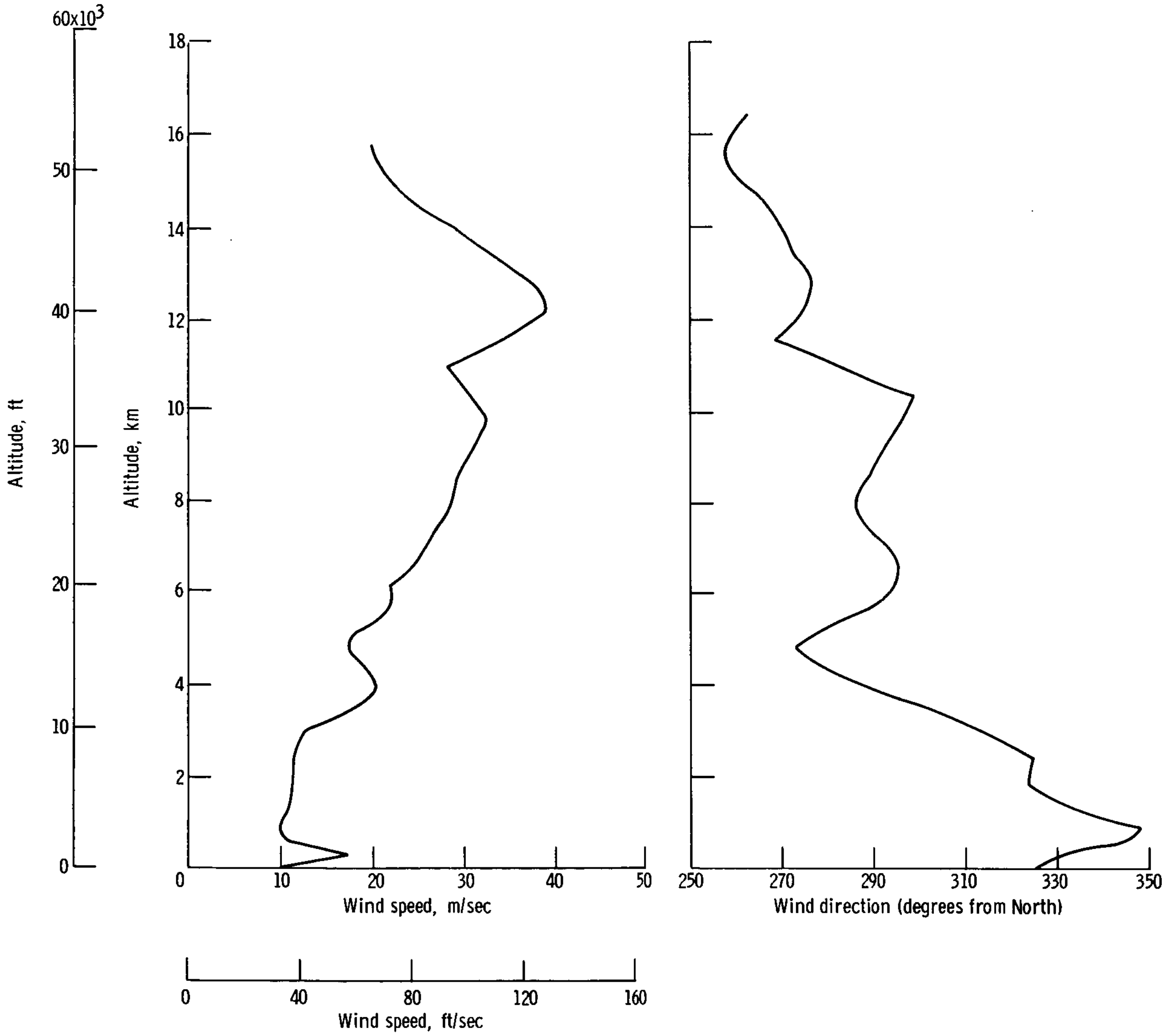


Figure IV-2. - Wind data, Numbus III.

V. THORAD VEHICLE SYSTEM PERFORMANCE

VEHICLE STRUCTURE SYSTEM

by Robert N. Reinberger

Description

The Thorad airframe structure (fig. V-1) consists of seven sections: the transition section, the adapter section, the fuel tank, the centerbody section, the oxidizer tank, the aft skirt section, and the engine and accessories section. The Thorad is 21.4 meters (70.3 ft) long and 2.4 meters (8 ft) in diameter, except for the conical forward section which tapers to a diameter of about 1.7 meters (5.5 ft).

The transition section at the forward end of the Thorad is 1.1 meters (3.7 ft) long and consists of a truncated cone of semimonocoque construction. The transition section houses the flight-control equipment, electrical power components, umbilical connection assembly, and flight termination equipment. Suitable access doors are provided to facilitate inspection and replacement of assembly mounted equipment. The adapter section, also a truncated cone, is 1.0 meter (1.3 ft) long and connects the transition section to the fuel tank.

The fuel tank assembly is 5.4 meters (17.7 ft) long. It is longitudinally butt welded to form a cylinder from three sheets of 0.63 centimeter (0.25 in.) aluminum, milled on the interior surface in a waffle-like pattern to obtain the maximum strength to weight ratio. It has convex domes at either end, intermediate frames, circumferential and antivortex baffles, and a fuel transfer tube and sump. The convex domes are bolted to the cylinder and have small welds to seal the joints.

The centerbody section, of semimonocoque construction, is 0.8 meter (2.8 ft) long and contains the Thorad telemetry equipment. Doors are provided for access to this section. The oxidizer tank assembly, 8.6 meters (28.2 ft) long, is similar in construction to the fuel tank assembly. The aft skirt section is 0.9 meter (2.8 ft) long and contains the nitrogen pressurization tanks and associated components and the oxidizer fill valve.

The engine and accessories section is 2.2 meters (7.1 ft) long and is of semimonocoque aluminum construction with stringers and ring frames. The main engine is attached through a gimbal block and tripod structure to three uniformly spaced thrust beams. These beams transmit the engine thrust loads to the Thorad structure. While the vehicle is on the launcher, the three thrust beams and three launcher beams support the vehicle. All the liquid propulsion support items such as the turbopump, lubrication unit, gas generator, hydraulic unit, engine relay boxes, integrated-start airborne system, and the fuel fill valve are housed in this section. Three solid-propellant rocket motors are attached to the thrust beams.

Performance

The structure system performance was satisfactory, and all flight loads were within the design limits. The peak longitudinal steady-state load of 6.15 g's occurred at Thorad main engine cutoff.

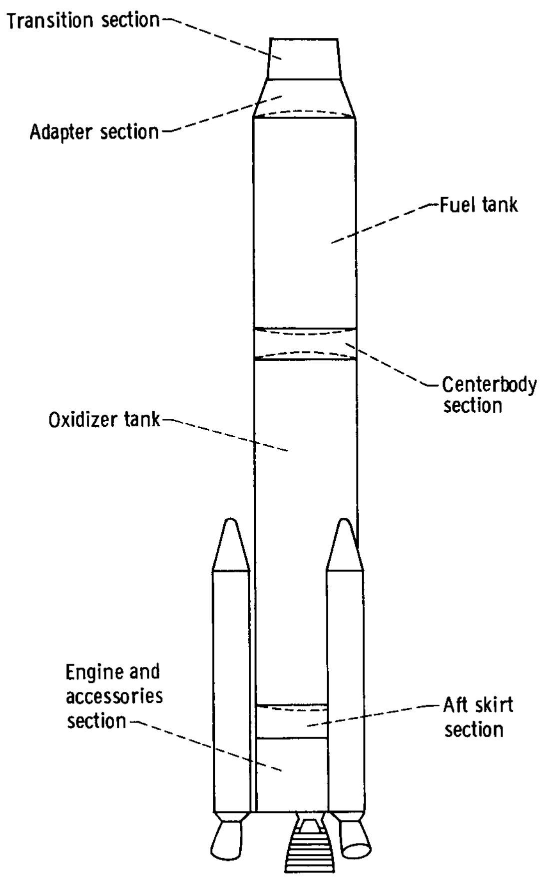


Figure V-1. - Thorad vehicle structure system, Nimbus III.

PROPULSION SYSTEM

by Charles H. Kerrigan

Description

The Thorad propulsion system is composed of a liquid-propellant engine system (fig. V-2) and three solid-propellant rocket motors.

The liquid-propellant engine system consists of a main engine, two vernier engines, and an engine start system. These engines use liquid oxygen and RJ-1 (kerosene) for propellants. During the engine start sequence, electrically initiated pyrotechnic igniters are used to ignite gas generator propellants for driving the turbopumps, and hypergolic igniters are used to ignite the propellants in the thrust chambers of the main and vernier engines. The pneumatic control of the liquid-propellant engine system is discussed in the section on the Thorad PNEUMATIC SYSTEM.

The Thorad main engine, rated at 756×10^3 newtons (170×10^3 lbf) thrust at sea level, consists of a gimbaled thrust chamber, propellant valves, an oxidizer and a fuel turbopump driven by a gas generator, a lubricating system, and a heat exchanger. Fixed-area orifices regulate the propellant flow to the gas generator. There is no thrust control system to compensate for changes in propellant head pressure to the turbopumps.

Each gimbaled vernier engine is rated at 4.45×10^3 newtons (1000 lbf) thrust at sea level with propellants supplied from the main engine turbopumps. Because the turbopumps do not operate after main engine cutoff, the vernier engines are supplied with propellants from the engine start tanks during the vernier solo phase of flight. For this phase, each vernier engine is rated at 3.68×10^3 newtons (830 lbf) thrust at sea level. The duration of the vernier engine solo phase is controlled by a time-delay relay that starts at main engine cutoff and provides the vernier engine cutoff command 9 seconds later.

The engine start system consists of two small propellant tanks and a pressurization system. These engine start tanks have a volume of approximately 0.028 cubic meter (1 ft^3) each and are filled and pressurized prior to launch to supply propellants for engine start. They remain pressurized and are refilled during flight to provide propellants for vernier engine operation after main engine cutoff.

The propellant grain for the three solid-propellant rocket motors is basically polybutadiene acrylic acid and ammonium perchlorate. Each solid motor is rated at 232×10^3 newtons (52 150 lbf) thrust at sea level. These motors have an improved grain configuration and contain more solid propellant than those previously used on the Thrust Augmented Thor (TAT) vehicle. This improvement results in about a 16-percent increase in total specific impulse. These motors are ignited by a signal from a pressure switch on the Thorad main engine thrust chamber. This switch actuates when the chamber pressure increases during the main engine start sequence. These solid motors provide thrust for about 39 seconds and are jettisoned 102 seconds after ignition. The jettison command is provided by a timer that starts at solid motor ignition. These motors are mounted 120° apart on the Thorad engine and accessories section and have an 11° nozzle cant angle (see fig. III-3).

Performance

The performance of the Thorad propulsion system for the Nimbus III mission was satisfactory. During the liquid-propellant engine start phase, engine valve opening times and starting sequence events were within tolerances. Performance parameters for the solid-propellant rocket motors and for the liquid-propellant engines were normal as indicated by a comparison of measured with expected values. These data are tabulated in table V-I. The solid motors burned for 39.2 seconds and were jettisoned at T + 102.2 seconds as planned.

Shortly before main engine cutoff, as on previous Thorad flights, there was a coupling of the propulsion system response characteristics with the first longitudinal mode of the vehicle structure (i.e., "POGO" effect). During this coupling, from T + 207 to T + 218 seconds, the main engine chamber pressure exhibited a 17.5-hertz, 13.8-newton-per-square-centimeter (20-psi) peak-to-peak oscillation. These values are typical for the POGO effect. (See Agena VEHICLE STRUCTURE SYSTEM for maximum POGO acceleration levels.)

Main engine cutoff was initiated by the oxidizer depletion switch. Vernier engine cutoff occurred 9 seconds later. Transients were normal at solid motor burnout and during the shut down of the main and vernier engines.

TABLE V-I. - THORAD PROPULSION SYSTEM PERFORMANCE, NIMBUS III

(a) Solid-propellant rocket motors

Combustion chamber pressure	Units	Flight values at -					
		T + 10 sec		T + 25 sec		T + 35 sec	
		Expected	Measured	Expected	Measured	Expected	Measured
Motor 1	N/cm^2	403	409	491	488	454	455
	psia	585	592	712	708	658	660
Motor 2	N/cm^2	403	405	491	485	454	452
	psia	585	587	712	704	658	655
Motor 3	N/cm^2	403	405	491	485	454	452
	psia	585	587	712	704	658	655

(b) Liquid-propellant engine

Performance parameter	Units	Flight values at -					
		T + 30 sec (stabilized operation)		Main engine cutoff		Vernier engine cutoff	
		Expected	Measured	Expected	Measured	Expected	Measured
Main engine thrust chamber pressure	N/cm^2	407	408	361	363	---	---
	psia	591	592	524	527	---	---
Turbopump speed	rpm	6229	6250	5888	5978	---	---
Vernier engine[a] No. 2 thrust chamber pressure	N/cm^2	268	266	246	247	---	---
when pump supplied	psia	389	385	356	358	---	---
Vernier engine[a] No. 2 thrust chamber pressure	N/cm^2	----	----	----	----	208	205
when tank supplied	psia	----	----	----	----	302	297

[a]Vernier engine No. 1 was not instrumented; however, proper vehicle roll control indicated satisfactory engine performance.

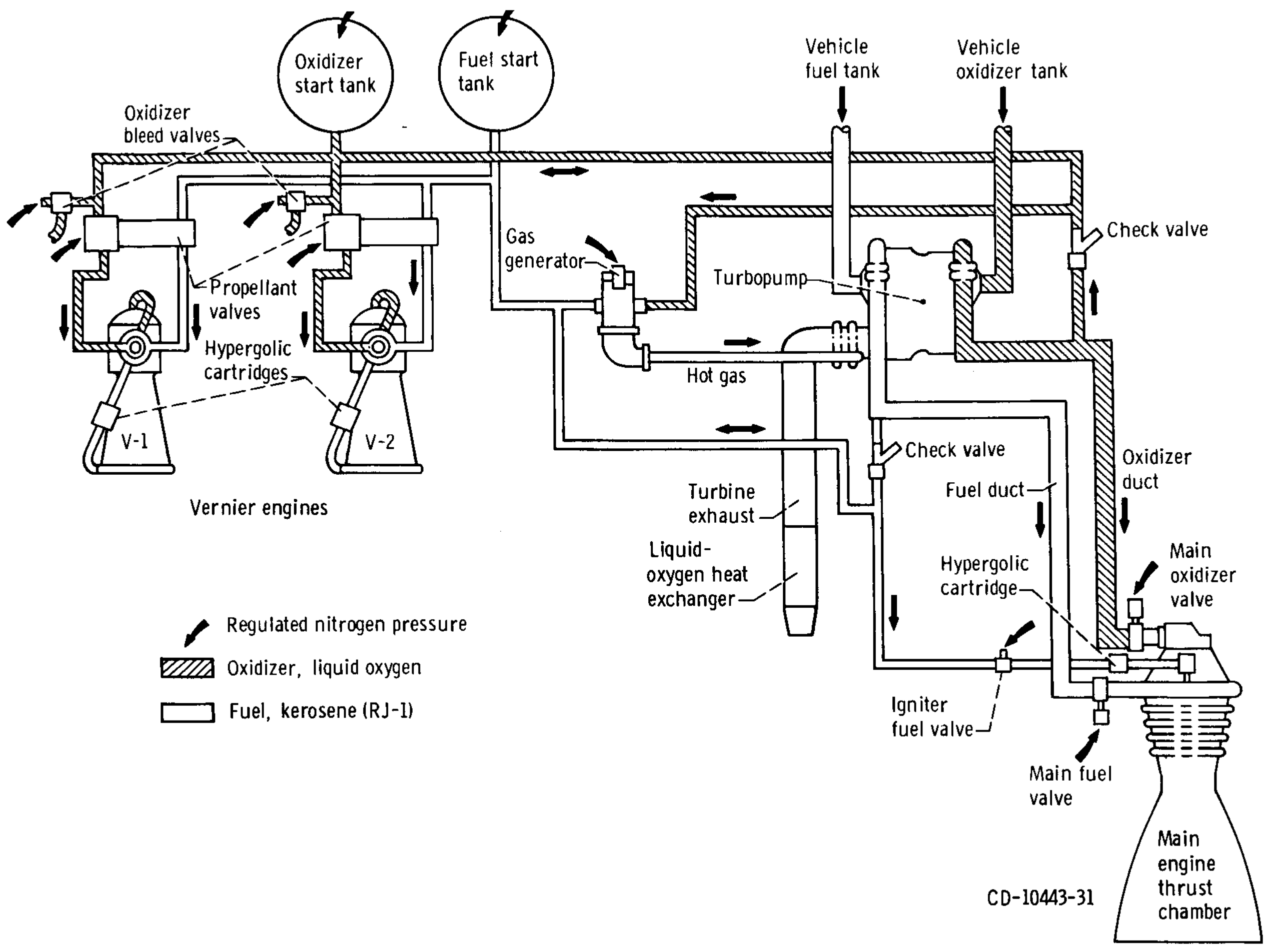


Figure V-2. - Liquid propellant engine system, Nimbus III.

HYDRAULIC SYSTEM

by Eugene J. Fourney

Description

The Thorad hydraulic system provides hydraulic fluid at the pressures and flow rates required for gimbaling the main and vernier engines during engine operation. The system consists of a pump, reservoir, accumulator, filters, check and pressure relief valves, six actuator assemblies, hydraulic fluid, and the necessary lines and fittings. The positive displacement pump, mounted on the turbopump accessory unit, provides the required flow rate and pressure of hydraulic fluid during the main engine thrust phase. The accumulator, which is precharged with nitrogen gas during ground operations, provides the required flow rate and pressure of hydraulic fluid for gimbaling the vernier engines during vernier engine solo operation. The reservoir provides hydraulic fluid to the pump inlet from the system lines. Two actuator assemblies are provided for each thrust chamber. Each actuator assembly consists of a servovalve for controlling the flow of hydraulic fluid for engine positioning and a feedback potentiometer.

Performance

The hydraulic system functioned satisfactorily. The hydraulic system flight performance data are presented in table V-II.

TABLE V-II. - THORAD HYDRAULIC SYSTEM PERFORMANCE, NIMBUS III

Performance parameter	Supply pressure		Return pressure	
	N/cm^2	psia	N/cm^2	psia
Preignition	2095	3020	68.9	100
Normal range during main engine operation	2065 - 2340	3000 - 3400	31 - 65.5	45 - 90
Flight time, sec				
T + 10	2205	3200	41.4	60
T + 25	2180	3160	41.4	60
T + 40	↓	↓	48.4	70
T + 60	↓	↓	55.1	80
T + 222.2 (main engine cutoff)	↓	↓	60.6	88
Flight time, sec				
T + 231.2 (vernier engine cutoff)	1790[a]	2600[a]	57.9	84
T + 246.2	1380[a]	2000[a]	57.9	84
T + 267.2	1020[a]	1480[a]	57.9	84

[a]Hydraulic supply pressure normally decays during vernier engine solo phase and continues to decay until pressure depletion.

PNEUMATIC SYSTEM

by Eugene J. Fourney

Description

The Thorad pneumatic system comprises the pneumatic control subsystem and the main fuel tank pressurization subsystem. High-pressure gaseous nitrogen is stored in four airborne spherical tanks to supply pressure for the airborne pneumatic system. A check valve in the system insures that one of these tanks will be restricted to providing nitrogen gas for the operation of the various functions of the pneumatic control subsystem. The three remaining tanks provide nitrogen gas for pressurizing the fuel tank and, if required, provides nitrogen gas for the operation of the pneumatic control system.

The pneumatic control subsystem regulates gaseous nitrogen pressure for pressurization of the engine start subsystem, the liquid-oxygen pump seal cavity, and the actuation of propellant valves. The system consists of a pneumatic control package, filter, four-way solenoid control valves, and the required fittings and connecting tubing. One of the solenoids controls pneumatic pressure to the main oxidizer valve; another controls pneumatic pressure to the main fuel valve and the gas generator blade valve.

The main fuel tank pressurization subsystem bleeds high-pressure gaseous nitrogen through a fixed-area orifice to maintain the fuel-tank ullage absolute pressure between 8.3 and 33.9 newtons per square centimeter (12 to 49 psi) during flight. A heat exchanger in the main engine gas generator exhaust system is used to convert liquid oxygen to gaseous oxygen to pressurize the oxidizer tank during flight.

Performance

The pneumatic system performed satisfactorily. All pneumatic system parameters observed were satisfactory and did not indicate any abnormal trends or characteristics. System flight performance data are presented in table V-III.

TABLE V-III. - THORAD PNEUMATIC SYSTEM AND TANK PRESSURIZATION

SYSTEM PERFORMANCE, NIMBUS III

Performance parameter	Main fuel tank ullage pressure		Main liquid oxygen tank ullage pressure		Pneumatic control bottle pressure	
	N/cm^2	psia	N/cm^2	psia	N/cm^2	psia
Normal range[a]	8.3 - 33.9	12 - 49	22.1 - 42.1	32 - 61	1665 - 2206	2400 - 3200
Flight time, sec						
T - 10	31	45	34.4	50	2070	3000
T - 0	30.3	44	27.5	40	1925	2800
T + 10	23.4	34	24.8	36	↓	↓
T + 60	15.2	22	26.2	38	↓	↓
T + 120	12.4	18	24.8	36	↓	↓
Main engine cutoff	8.9	13	23.4	34	1895	2750
Vernier engine cutoff	8.9	13	23.4	34	[b]965	[b]1400

[a]Normal range values apply only during main engine operation.

[b]Pressure change from main engine cutoff to vernier engine cutoff reflects use of nitrogen for pressurization of start tanks during vernier engine solo operation as well as pneumatic control system operation.

GUIDANCE AND FLIGHT CONTROL SYSTEM

by Howard D. Jackson and James L. Swavely

The Thorad flight path is controlled by two interrelated systems: The Thorad flight control system and the radio-guidance system. The flight control system directs the vehicle in a preprogrammed open-loop mode from lift-off through vernier engine cutoff. The radio-guidance system will provide, if needed, pitch and yaw steering commands during approximately the last half of the Thorad powered flight. It also provides discrete commands for Thorad main engine cutoff and Thorad-Agena separation. The radio-guidance system's use during the Agena phase of flight is discussed in the Agena GUIDANCE AND FLIGHT CONTROL SYSTEM (section VI).

Description

The major components of the Thorad flight-control system are the control electronic assembly, and three rate gyros. The control electronic assembly contains a programmer, three displacement gyros, and associated electronic circuitry. These displacement gyros are single-degree-of-freedom, floated, hermetically sealed rate integrating gyros. These gyros are mounted in an orthogonal configuration alining the input axis of each gyro to its respective vehicle axis of pitch, yaw, or roll. Each gyro provides an electrical output signal proportional to the difference in angular position of the measured axis from the gyro input (reference) axis.

The programmer supplies commands for starting and stopping the fixed roll and pitch programs, solid motor jettison arm and backup jettison, enabling radio-guidance system steering, enabling vernier engine yaw control, and enabling main engine cutoff. The programmer commands are generated by a motor-driven prepunched tape. Slots in the prepunched tape activate relay circuits for the commands. For this mission the capability of the Thorad flight control system to accept radio-guidance system pitch and yaw steering commands is enabled at T + 124 seconds. Between T + 124 seconds and vernier engine cutoff, all radio-guidance system pitch and yaw steering commands are routed to the Thorad flight control system to provide corrections for deviations from the programmed trajectory.

The three-rate gyros are of the single-degree-of-freedom, spring restrained type. The roll-rate gyro is mounted in the centerbody section with its input axis alined to the vehicle roll axis. The pitch and yaw rate gyros are mounted in the cable tunnel adjacent to the fuel tank with the input axes alined to the pitch and yaw vehicle axes. Each rate gyro provides an electrical output signal proportional to the angular rate of rotation of the vehicle about the gyro input (reference) axis.

The radio-guidance system includes airborne equipment located in the Agena (a radar transponder and command receiver, a control package, two antennas, a directional coupler, and connecting wave guide) and ground-based equipment (a radar tracking station and a computer). The major functions (fig. V-3) are described in the following paragraphs:

The radar tracking station transmits a composite message train containing an address code and the steering and discrete commands to the vehicle. The radar transponder and command receiver in the Agena receives the message train and transmits a return pulse to the ground each time the address code is correct. The radar tracking station determines vehicle position (range, azimuth, and elevation) from the return pulses. The computer processes the position information, computes trajectory corrections, and issues appropriate steering and discrete commands which are transmitted to the Agena by the radar tracking station. The steering and discrete commands are routed from the Agena to the Thorad through vehicle harnesses.

A dorsal and a ventral antenna are mounted on the forward section of the Agena and are connected through the waveguide and the directional coupler to the radar transponder and command receiver. The location of the radar tracking station antenna with respect to the launch site is such that for prelaunch testing and early ascent, the dorsal antenna provides the greater signal strength to the ground antenna. As the vehicle pitches over and moves downrange, the ventral antenna provides the greater signal strength. The directional coupler attenuates the signal from the dorsal antenna to minimize interference effects between the dorsal and ventral antenna. Mission trajectory information determines the antenna configuration, antenna orientation, and the type of directional coupler for each mission.

During the early portion of flight, multipath and ground clutter effects might cause the radar tracking station to acquire (lock on) a false vehicle position. To avoid this problem the following procedure is used for radar tracking station acquisition (lock-on) of the vehicle. Before lift-off the centerline of the ground radar antenna beam is manually pointed at the junction of the ground antenna horizon and the programmed trajectory. At lift-off a ground timer is started which will close the ground radar angle tracking loops at T + 6 seconds, the time at which the vehicle is predicted to fly through the radar beam. When the angle tracking loops are closed, the acquisition (lock-on) is complete, and the radar tracking station will track the actual vehicle position.

As a backup to the angle-loop timer, the radar tracking station operator manually closes the angle tracking loops 7 seconds after lift-off. If the radar tracking station still does not acquire the launch vehicle, the ground antenna is slewed to 20 mils (5.08 μm) elevation to acquire at T + 11 seconds, and then it is manually slewed through a planned series of pointing coordinates until acquisition is effected. These coordinates correspond to first a 40-mil elevation (T + 14.5 sec) and then to the expected vehicle positions at T + 20, T + 40, and T + 62 seconds. Range lock and frequency lock are accomplished before lift-off.

Performance

The Thorad flight control system performed satisfactorily throughout the flight. Main engine gimbal angles at lift-off were -0.29° in pitch and -0.15° in yaw. Lift-off transients in pitch, yaw, and roll were negligible.

The maximum main engine gimbal angles at the time of greatest wind shear (approx T + 55 sec) were -1.78° in pitch and $+1.00^{\circ}$ in yaw. Maximum angular displacements of the vehicle at T + 124.4 seconds (after radio-guidance system steering was enabled) were $+1.92^{\circ}$ in pitch and -1.24° in yaw; the corresponding rates were -1.16 and -1.90 degrees per second. Gimbal angles at main engine cutoff (T + 222.2 sec) were $+0.19^{\circ}$ in pitch and -0.05° in yaw. Angular displacements of the vehicle at vernier engine cutoff (T + 231.2 sec), when the Agean gyros were uncaged, were $+0.21^{\circ}$ in pitch, $+0.32^{\circ}$ in yaw, and $+0.14^{\circ}$ in roll. These angular displacements were within allowable limits and provided a satisfactory reference attitude for the Agena. Angular rates at Thorad-Agena separation (T + 238.1 sec) were -0.02 degree per second in pitch, +0.15 degree per second in yaw, and -0.25 degree per second in roll.

After solid motor jettison at T + 102.2 seconds, peak-to-peak oscillations were observed in roll (0.48 deg/sec at a frequency of 1.25 Hz) and in the pitch-roll vernier engine plane (1.23 deg/sec at a frequency of 1.25 Hz). These oscillations were due to servovalve nonlinearities and backlash in the linkage to the vernier engines. Longitudinal oscillations (POGO effect) were encountered between T + 207 and T + 218 seconds. None of these oscillations was detrimental to the control system performance.

The radio-guidance system, ground and airborne, performed satisfactorily throughout the guided portion of flight. The range and frequency loops of the radar tracking station were locked on the vehicle before lift-off. Signal strength at the radar tracking station before lift-off was -26 dBm (decibels referenced to 1 mW). The ground angle-loop timer started at lift-off and actuated at T + 6 seconds. The angle tracking loops were closed and vehicle acquisition (lock on) occurred at T + 6 seconds. The manual backup angle-loop was closed at T + 7 seconds. The signal strength at the radar tracking station was approximately -28 dBm during the first few seconds of flight, increased to -20 dBm by T + 110 seconds, then gradually decreased to -50 dBm by T + 430 seconds. Signal strength fluctuations occurred, as expected, during the two-antenna interference region from T + 40 and T + 90 seconds when the received signal strengths from the dorsal and ventral antennas were within 10 dB of each other. Radar tracking station data indicated that the actual vehicle position was continuously tracked throughout the flight (i.e., no radar coast), even during the two-antenna signal-interference region. The performance of the ground-based computer was satisfactory throughout the countdown and vehicle flight.

Prior to lift-off the airborne radio-guidance system equipment indicated a received signal strength of -12 dBm. The maximum received signal strength was -2 dBm at T + 100 seconds and decreased to -30 dBm by T + 430 seconds. The signal strength received by the vehicle was adequate throughout the operation of the radio-guidance system.

All radio-guidance system commands were satisfactorily generated by the computer, transmitted by the radar tracking station, and received and executed by the vehicle. Table V-IV shows planned and actual times of all radio-guidance system commands.

TABLE V-IV. - RADIO-GUIDANCE SYSTEM COMMANDS,

NIMBUS III

Command	Planned time from lift-off, sec	Actual time from lift-off, sec
Thorad steering (pitch and yaw)		
Commenced	124.6	124.4
Terminated	217.9	218.9
Thorad main engine cutoff	221.6	[a]222.6
Thorad-Agena separation	237.2	238.1
Agena steering (pitch and yaw)		
Commenced	268.2	269.9
Terminated	387.0	389.1
Agena velocity meter enable	388.6	390.5

[a]The planned mode of Thorad main engine cutoff is by either propellant depletion or radio-guidance system command. For Nimbus III, the Thorad main engine cutoff occurred from propellant depletion at T + 222.2 sec (0.4 sec before the radio-guidance system command).

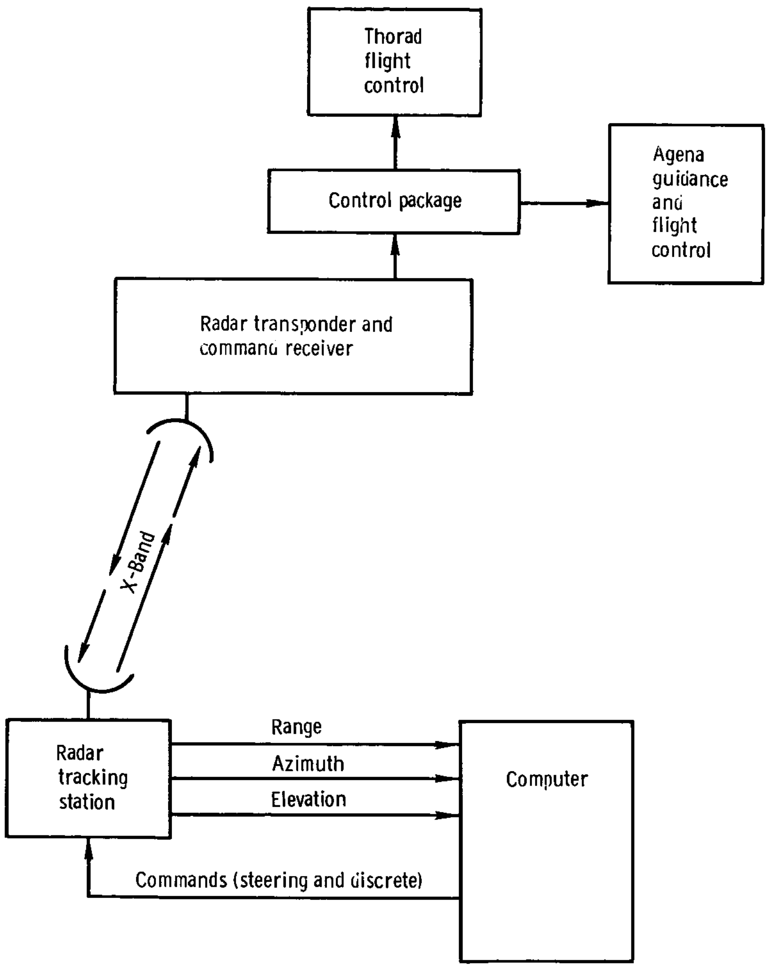


Figure V-3. - Block diagram of major functions of the radio-guidance system functions, Nimbus III.

ELECTRICAL SYSTEM

by Edwin R. Procasky

Description

The Thorad power requirements are supplied by three 28-volt silver-zinc alkaline batteries and a 400-hertz rotary inverter (see fig. V-4). Distribution boxes are located throughout the vehicle to facilitate interconnection and switching of electrical functions. Two tunnels located outside the propellant tanks are used to route cables between the transition, centerbody, and engine and accessory sections.

The main battery is rated at 20 ampere-hours and supplies all the vehicle power requirements except the telemetry system and the flight termination system. The power requirements for these systems are supplied by the two other batteries. The telemetry battery, rated at 3 ampere-hours, supplies the telemetry system and the flight termination subsystem No. 1 power requirements. The remaining battery, rated at 1 ampere-hour, supplies power to the flight termination subsystem No. 2.

The rotary inverter (a dc motor-driven ac alternator) provides the 400-hertz, 115/208-volt ac, three-phase power. The voltage output and frequency of the inverter are regulated to ±1.5 percent. The alternator is wye connected with a grounded neutral.

Performance

The main battery supplied the requirements of the dependent systems at normal voltage levels. The battery voltage at lift-off was 26.0 volts dc, increased to 29.1 volts dc by T + 120 seconds, and was 30.0 volts dc at Thorad main engine cutoff (T + 222.2 sec). The telemetry battery supplied power to the telemetry system and the flight termination subsystem No. 1 at 28.8 volts dc throughout the Thorad flight phase. The battery which supplies power to flight termination subsystem No. 2 was not monitored in flight. The rotary inverter operated within the ±1.5 percent tolerance for voltage and frequency throughout the Thorad flight. The inverter frequency was 403 hertz at lift-off and increased to 406 hertz by Thorad main engine cutoff (T + 222.2 sec). The inverter voltage was 114.0 volts dc throughout the Thorad powered flight.

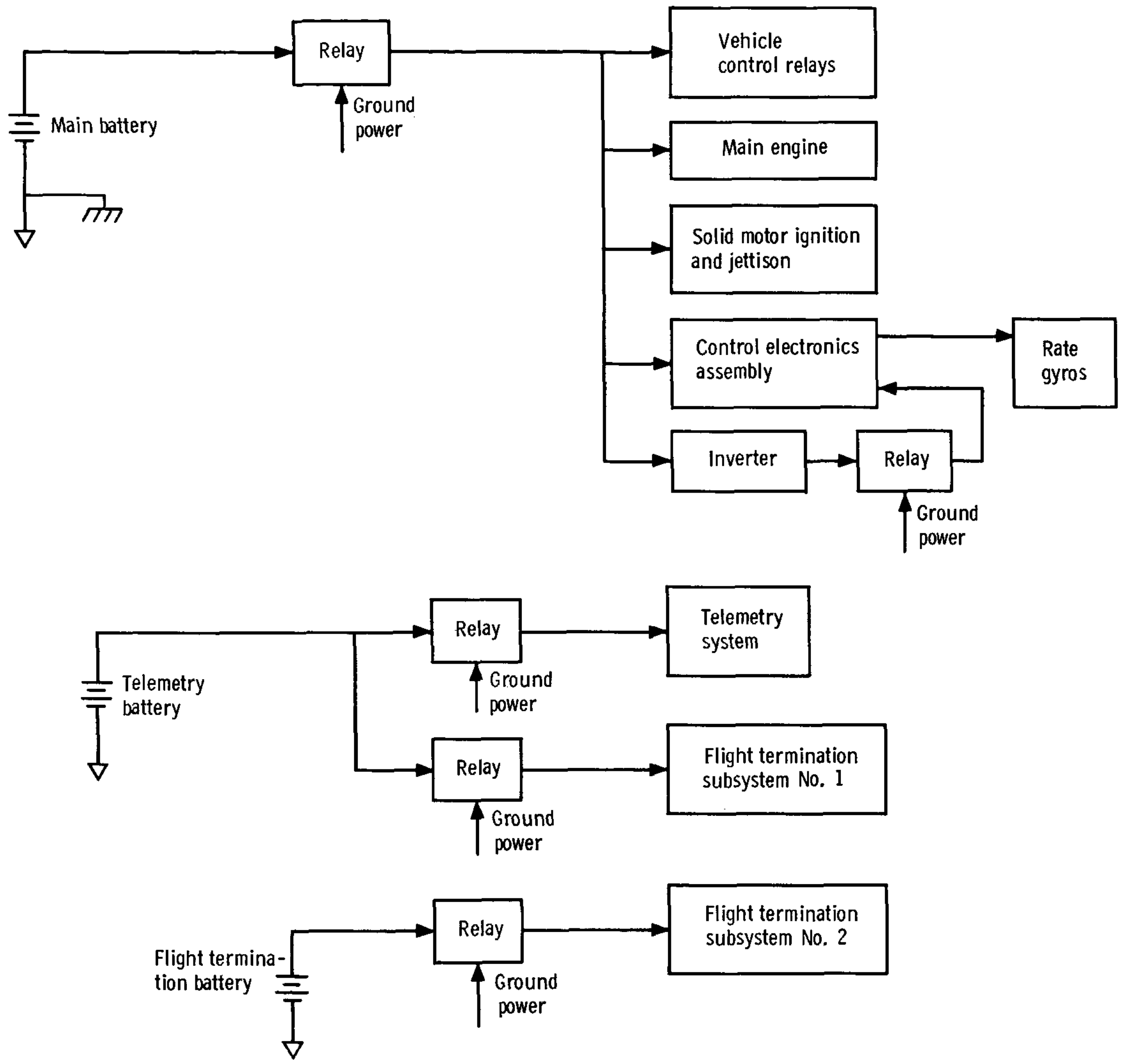


Figure V-4. - Thorad power distribution block diagram, Nimbus III.

TELEMETRY SYSTEM

by Richard E. Orzechowski

Description

The Thorad telemetry system consists of two antennas, an FM transmitter, signal conditioning networks, associated transducers, a 28-volt battery, and a multicoder. The telemetry system is located in the centerbody section. The telemetry system uses a combination of two types of transmitter modulation techniques, frequency modulation/ frequency modulation (FM/FM) and pulse duration modulation/frequency modulation/ frequency modulation (PDM/FM/FM), with a common radiofrequency link.

The FM/FM section of the PDM/FM/FM telemetry system provides 10 channels of continuous data. Voltage controlled subcarrier oscillators are modulated by a single transducer or by a series sequence of signals, to produce an FM output. The FM outputs of the voltage controlled oscillators are combined and applied to the wideband amplifier. The composite FM output of the wideband amplifier modulates the FM transmitter output producing FM/FM.

In the PDM/FM section of the telemetry system transducer data are time multiplexed by a multicoder. The PDM output frequency modulates a single voltage controlled oscillator. The voltage controlled oscillator output is applied to the wideband amplifier and modulates the FM transmitter producing PDM/FM/FM. The multicoder provides 43 channels of commutated data.

Performance

Thorad telemetry performance was satisfactory on Nimbus III. Fifty-three measurements (appendix B) were flown, and all yielded usable data. No telemetry problems occurred during the countdown or during the flight. Radiofrequency signal strength was adequate during flight as evidenced by good quality data. Carrier frequency was stable and no data reduction difficulties were encountered. No direct measurements of telemetry system performance or environment were made. Appendix C (fig. C-2) shows the specific coverage provided by the supporting telemetry stations.

FLIGHT TERMINATION SYSTEM

by Richard E. Orzechowski

Description

The Thorad flight termination system (fig. V-5) consists of two identical and redundant subsystems designed to destroy the vehicle on receipt of ground command signals. Each subsystem includes two antennas (located on opposite sides of the Thorad), a command receiver, a safe/arm mechanism, and destructor cords. The antenna locations are shown on figure III-3. The safe/arm mechanisms are armed by lanyards at lift-off. After lift-off the range safety officer can command destruction, if required, by transmitting a coded signal to the command receivers. Each command receiver will supply an electrical signal to two detonators in a Thorad safe/arm mechanism and, prior to Thorad-Agena separation, to a detonator in the Agena destruct initiator. Either detonator on a safe/arm mechanism will initiate the two destructor cords (one on each side of the Thorad propellant tanks) and, through other destructor cords, will initiate a shaped charge on the forward end of each solid-propellant rocket motor. A 0.1-second time delay in the Thorad safe/arm mechanisms insures that the Agena destruct initiator receives the destruct signal before the Thorad is destroyed.

The Agena destruct components are discussed in the Agena COMMUNICATION AND CONTROL section of this report.

Performance

Both receivers in the Thorad flight termination system functioned satisfactorily during flight. The data indicated that the vehicle received adequate signal strength for the operation of each flight termination subsystem and that the signal level remained essentially constant throughout the period in which destruct capability was required. No flight termination commands were required, nor were any commands inadvertently generated by any vehicle system.

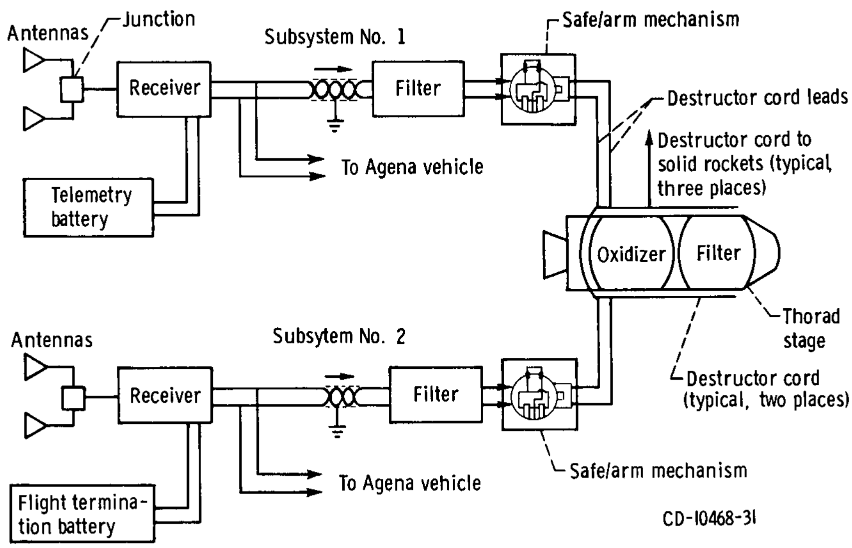


Figure V-5. - Thorad flight termination system, Nimbus III.

VI. AGENA VEHICLE SYSTEM PERFORMANCE

VEHICLE STRUCTURE SYSTEM

by Robert N. Reinberger

Description

The Agena vehicle structure system (fig. VI-1) consists of four major sections: the forward section, propellant tank section, aft section, and the booster adapter assembly. Together they provide the aerodynamic shape, structural support, and environmental protection for the vehicle. The forward section is basically an aluminum structure with beryllium and magnesium panels. This section encloses most of the electrical, guidance, and communication equipment and provides the mechanical and electrical interface for the spacecraft adapter and shroud. The propellant tank section consists of two integral aluminum tanks, with a sump below each tank to insure the supply of propellants for engine starts in space. The aft section consists of an engine mounting cone structure and an equipment mounting rack. The magnesium alloy booster adapter section supports the Agena and remains with the Thorad after Thorad-Agena separation.

Performance

The measured dynamic environment of the structure system was within design limits. The maximum longitudinal oscillation (POGO effect at T + 209.14 sec), measured at Agena station 226 on the spacecraft adapter, was 3.91 g's (zero-to-peak). Longitudinal oscillation (POGO effect) data and other significant dynamic data are presented in appendix D.

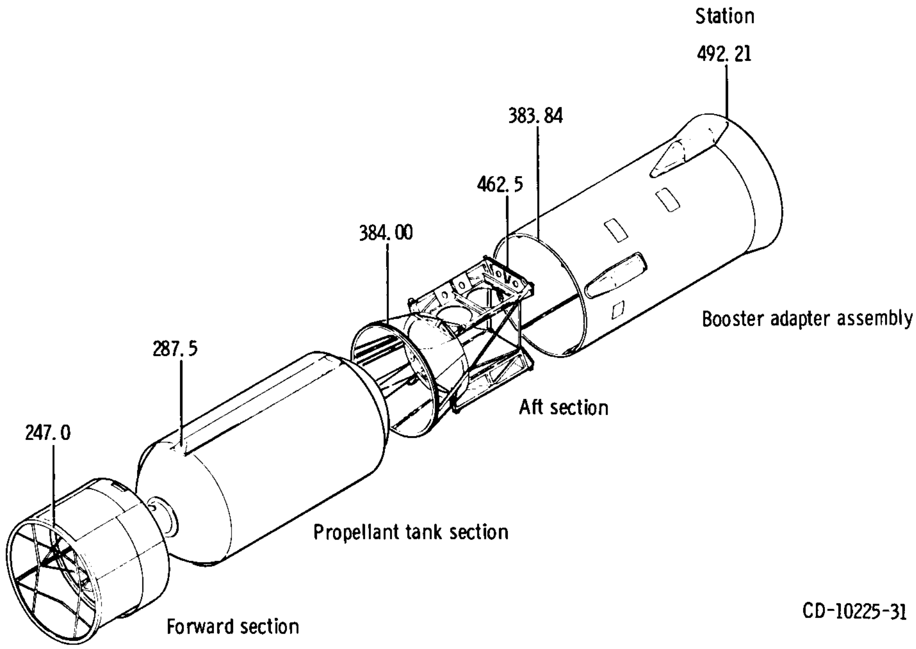


Figure VI-1. - Agena vehicle structure system, Nimbus III.

SHROUD SYSTEM

by Robert N. Reinberger

Description

The shroud system used for the Nimbus III flight was a standard Agena clamshell (SAC) shroud with minor modifications incorporated for this mission. It provides environmental protection for the spacecraft before launch and during ascent. The SAC shroud shown in figure VI-2 is 5.72 meters (18.78 ft) long and weighs 328 kilograms (723 lbm). It consists of an aluminum transition ring and two shroud halves. The two shroud halves form a fairing with a 1.65-meter (5.42-ft) diameter cylindrical section, a 15° half-angle conical section, and a 0.61-meter (2-ft) diameter, hemispherical nose cap. The shroud halves are constructed of laminated fiberglass strengthened by internal aluminum longerons at the split lines and semicircular frames. Microquartz thermal insulation blankets in the cylindrical section of each shroud half and foil covering in the conical section of each shroud half provide thermal protection for the spacecraft. The shroud halves are held together by a nose latch, two flat bands around the cylindrical section, and a V-band around the base of the cylindrical section. The top, middle, and bottom bands are tensioned to 22 250, 11 570, and 35 600 newtons (5000, 2600, and 8000 lbf), respectively.

The V-band clamps the shroud to the transition ring, which is approximately 0.051 meter (0.17 ft) high and is bolted to the forward end of the Agena. Both the shroud and the spacecraft adapter are attached to the transition ring. A metal diaphragm attached to the transition ring isolates the shroud cavity from the Agena. During ascent this cavity is vented through four ports, equipped with flappers, in the cylindrical section of the shroud.

Shroud jettison is commanded by the Agena timer 10 seconds after Agena engine first start. At this time, Agena electrical power is used to fire two pyrotechnic bolt cutters in the nose latch assembly and two explosive bolts in each of the three bands. The firing of one bolt cutter in the nose latch and one bolt in each of the bands will effect shroud separation. When the nose latch and the bands are released, two pairs of springs in each shroud half thrust against the transition ring and provide the energy to rotate each shroud half about hinges mounted on the transition ring. At the time of shroud separation, the Agena has a longitudinal acceleration of approximately 1 g. At this acceleration level, each shroud half rotates through an angle of about 75° before it leaves the hinges and falls free. The shroud separation springs provide sufficient energy to jettison the shroud halves at vehicle (Agena) longitudinal acceleration levels up to 3.5 g's.

The shroud system is instrumented with a single pressure transducer, which measures the differential pressure across the shroud diaphragm. This pressure transducer is located at approximately Agena station 247.0.

Performance

The history of the shroud diaphragm differential pressure ($\Delta P = P_{shroud} - P_{Agena\ forward\ section}$) is presented in figure VI-3. Shortly after lift-off the differential pressure became positive and increased to a value of 0.26 newton per square centimeter (0.37 psi) at the onset of the transonic phase of flight. During the transonic phase, the differential pressure became negative and decreased to a value of -0.61 newton per square centimeter (-0.89 psi). After the transonic phase, the differential pressure again became slightly positive, decreased to essentially zero at T + 177 seconds, and remained at that level for the remainder of the flight.

Shroud pyrotechnics were fired at T + 266.3 seconds, and the shroud was satisfactorily jettisoned. At this time the Agena roll and yaw rates were very nearly zero, and the pitch rate was at the programmed value. No measurable Agena roll or yaw rates or change in pitch rate developed as a result of shroud jettison.

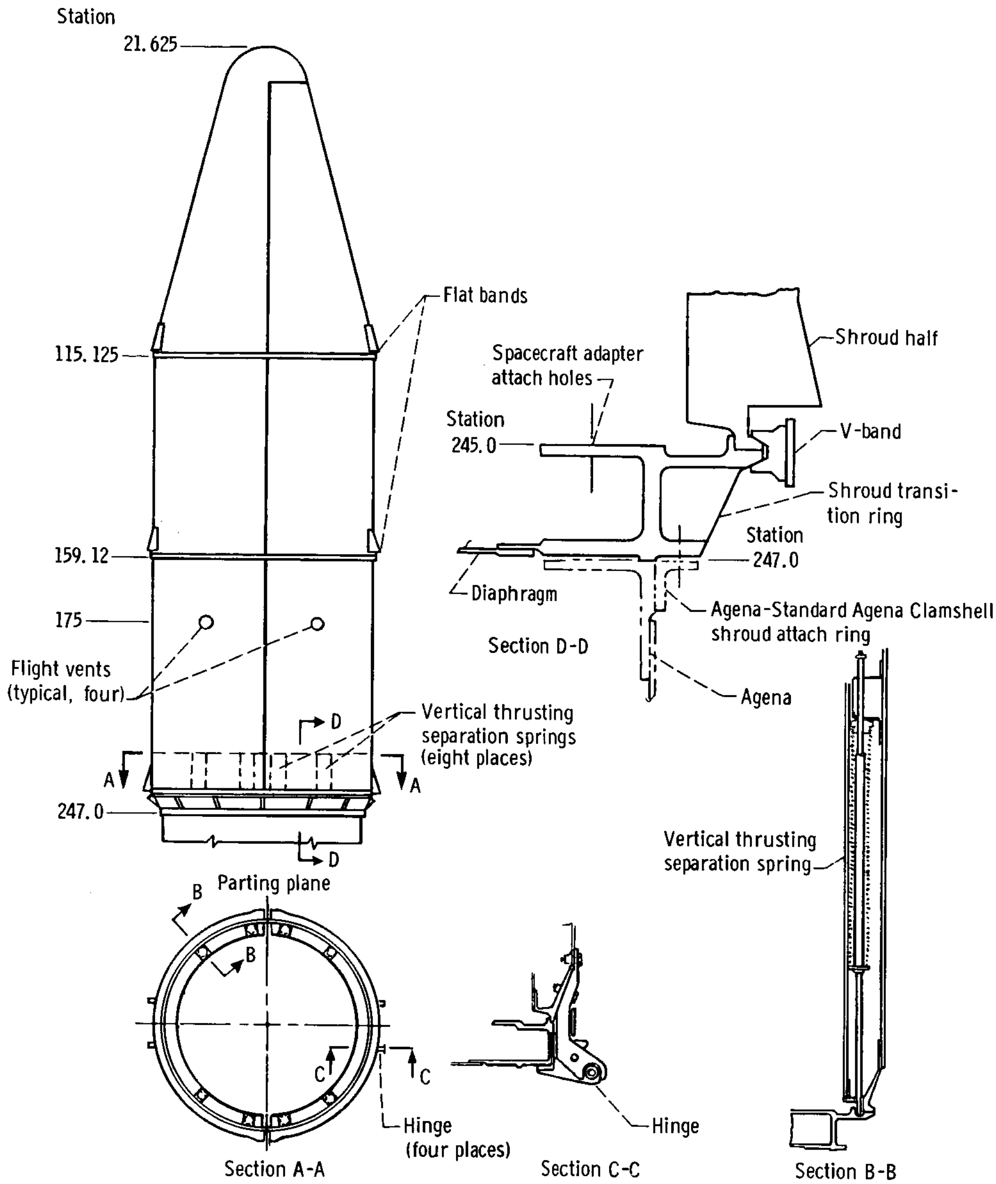


Figure VI-2. - Standard Agena Clamshell shroud, Nimbus III.

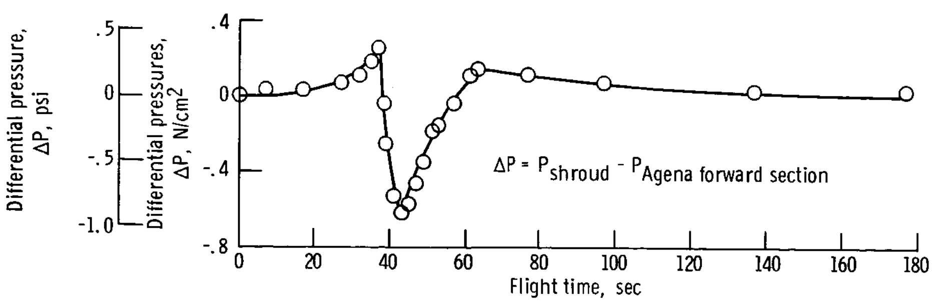


Figure VI-3. - Diaphragm differential pressure, Numbus III.

PROPULSION SYSTEM

by Robert J. Schroeder

Description

The Agena propulsion system (fig. VI-4) consists of a propellant tank pressurization system, a propellant management system and an engine system. Also considered part of the propulsion system are the Agena cold gas retrothrust system, the Thorad-Agena separation system, and the Agena vehicle pyrotechnic devices.

The propellant tank pressurization system provides the required propellant tank pressures and consists of a helium supply tank and a pyrotechnically operated helium control valve. Before lift-off, the ullage volume in the propellant tanks is pressurized with helium from a ground supply source. The helium control valve is activated 1.5 seconds after initiation of the Agena engine first start to permit helium gas to flow from the supply tank through fixed-area flow orifices to each propellant tank. After the Agena engine first cutoff, the helium control valve is again activated to isolate the oxidizer tank from the helium supply. This prevents the mixing of oxidizer and fuel vapors that could occur if pressures in the propellant tanks were permitted to reach the same level. Pressurization during the second powered phase of the Agena engine is provided by the residual pressure in the propellant tanks.

The propellant management system consists of the following major items: propellant fill disconnects to permit the loading of fuel and oxidizer, feed lines from the propellant tanks to the engine pumps, tank sumps to retain a sufficient amount of propellants for engine restart in a zero-gravity environment, and an electric motor driven propellant isolation valve in each feed line. The propellant isolation valves are open at lift-off, closed after the Agena engine first cutoff, and opened 2 seconds before the Agena engine second start. When closed, these valves isolate the propellants in the tanks from the engine pump inlets and provide an overboard vent for propellants trapped in the engine pumps.

The Agena engine system consists of a liquid bipropellant engine which uses unsymmetrical dimethylhydrazine as fuel and inhibited red fuming nitric acid as oxidizer. Rated thrust in a vacuum is 71 172 newtons (16 000 lbf) with a nozzle expansion area ratio of 45. The engine has a regeneratively cooled thrust chamber, a radiation-cooled nozzle extension, and a turbopump propellant flow system. Turbine rotation is initiated for each engine start by igniting a solid-propellant start charge. The turbine is driven during steady-state operation by hot gas produced in a gas generator. Propellants to the gas generator are supplied by the turbopump. An oxidizer fast-shutdown system, which consists of a pyrotechnically operated valve and a high-pressure nitrogen storage cylinder, is used to rapidly close the main oxidizer valve at engine first cutoff. Engine thrust vector control in pitch and yaw is provided by the gimbal mounted thrust chamber in response to signals produced by the Agena guidance and flight control system.

The Agena retrothrust system (fig. VI-5) provides the impulse to lower the altitude of the Agena orbit after Nimbus III separation. This is accomplished with two operations of the retrothrust system. The impulse for the first operation of the retrothrust system is provided by 3.97 kilograms (8.75 lbm) of gas in the retrothrust storage sphere. The impulse for the second operation of the retrothrust system is provided by the gas remaining in the attitude-control gas storage sphere. Based on a nominal gas consumption of the attitude control system during the ascent phase of the flight, approximately 8.48 kilograms (18.7 lbm) of residual gas are available for the second operation of the retrothrust system. The first operation of the retrothrust system is initiated 193 seconds after Nimbus III separation, and the second operation of the retrothrust system occurs 2700 seconds after Nimbus III separation. Each operation of the retrothrust system is initiated by activating a normally closed pyrotechnic valve which permits a regulated flow of gas from the appropriate storage sphere to a fixed retrothrust nozzle. The retrothrust nozzle is alined through the vehicle center of gravity to minimize disturbing torques. An electrically operated valve in the fill line to each storage sphere permits both spheres to be loaded through a common fill coupling and provides isolation between the spheres during flight.

The Thorad-Agena separation is accomplished by firing a Mild Detonating Fuse which severs the booster adapter circumferentially near the forward end. The Thorad with booster adapter is then separated from the Agena by firing two solid-propellant retrorockets mounted on the booster adapter. Rated average sea-level thrust of each retrorocket is 2180 newtons (490 lbf) with an action time of 0.93 second. Rollers on the Agena aft section are guided by rails on the booster adapter to maintain clearance and alinement during separation.

Pyrotechnic devices are used to perform a number of functions on the Agena. These devices include squibs, igniters, detonators, and explosive bolt cartridges. Squibs are used to open and close the helium control valve, to eject the horizon sensor fairings, to activate the oxidizer fast shutdown system, to activate the retrothrust system, and to activate shroud bolt cutters. Igniters are used for the main engine solid-propellant start charges, and the retrorockets. Detonators are used for the self-destruct charge and the Mild Detonating Fuse separation charge. Explosive volt cartridges are used to rupture release devices for shroud jettison.

Performance

The Thorad-Agena separation system performance was normal. Separation was commanded by the radio-guidance system at T + 238.1 seconds. The command resulted in the ignition of the Mild Detonating Fuse and the two retrorockets. The booster adapter guide rails cleared the last rollers on the Agena aft rack at T + 240.7 seconds.

Agena engine first start was initiated by the Agena timer at T + 256.2 seconds. The engine switch group monitor data indicated a normal start sequence of the engine control valves. Ninety-percent combustion chamber pressure was attained at T + 257.4 seconds. The average steady-state thrust produced by the Agena engine was 72 177 newtons (16 226 lbf) compared with an expected value of 71 857 newtons (16 154 lbf). Agena engine cutoff was commanded by the velocity meter at T + 488.0 seconds. The engine thrust duration, measured from 90-percent chamber pressure to velocity meter cutoff command, was 230.6 seconds. This was 0.22 second less than the expected value of 230.82 seconds. The actual thrust duration and thrust level indicate that engine performance was within the allowable limits.

The propellant tank pressurization system supplied the required tank pressures. The fuel and oxidizer pump inlet pressures were within 1.4 newtons per square centimeter (2 psi) of the expected values during the Agena engine first powered phase.

The propellant isolation valves closed normally after Agena engine first cutoff and opened normally prior to engine second start, as evidenced by the valve position measurement and the changes in pump inlet pressures.

Agena engine second start was initiated at T + 3261.2 seconds. The engine switch group monitor data indicated a normal start sequence of the engine control valves. Ninety-percent combustion chamber pressure was attained at T + 3262.3 seconds. The average steady-state thrust produced by the Agena engine was 73 580 newtons (16 544 lbf) compared with an expected value of 72 167 newtons (16 224 lbf). Agena engine second cutoff was commanded by the velocity meter at T + 3267.4 seconds. The engine thrust duration, measured from 90-percent combustion chamber pressure to velocity meter cutoff command, was 5.1 seconds. This was 0.18 second less than the expected value of 5.28 seconds. The actual thrust duration and thrust level indicate that engine performance was within the allowable limits.

The propellant tank pressures during the second burn were adequate for satisfactory engine operation. The pump inlet pressure measurements were within 1.4 newtons per square centimeter (2 psi) of the expected values.

After Nimbus III separation, the Agena performed a conical turn which placed the vehicle in a nose-aft position with respect to the trajectory vector. The first operation of the retrothrust system was initiated at T + 3710.2 seconds. Immediately, the Agena began an unexpected pitch-up (positive pitch) which continued until the gas in the retrothrust sphere had been nearly depleted. A postflight evaluation indicated that the unexpected pitch-up was the result of a disturbing torque, which was most probably caused by the exhaust gas from the retrothrust nozzle impinging on the Agena engine nozzle extension. This disturbing torque exceeded the correction capability of the attitude control system which was in the low thrust mode during the first operation of the retrothrust system. (See the Agena GUIDANCE AND FLIGHT CONTROL section for discussion of the vehicle attitude during the first retrothrust.)

The second operation of the retrothrust system was initiated at T + 6217.2 seconds. At the same time the attitude control system was transferred from the low thrust mode to the high thrust mode. Approximately 8.94 kilograms (19.7 lbm) of gas was in the attitude control sphere at the start of the second operation of the retrothrust system. During this operation of the retrothrust system, the attitude control system was able to compensate for the disturbing torque produced by the exhaust gas from the retrothrust nozzle impinging on the Agena engine extension.

In addition to the unexpected disturbance torque produced by the retrothrust system, the duration of each operation of the retrothrust system was approximately twice as long as predicted. Correspondingly, the actual weight flow rates for each operation of the retrothrust system were approximately one-half of the expected values. This change in the retrothrust system operation had no adverse effect on the Agena vehicle since the actual total impulse supplied during each operation of the retrothrust system was the same as predicted.

A postflight evaluation to determine the cause of the increased thrust duration revealed that the design calculations did not include allowance for system line losses due to friction. Subsequent incorporation of these friction losses into the analysis resulted in weight flow rates which were consistent with those experienced in flight.

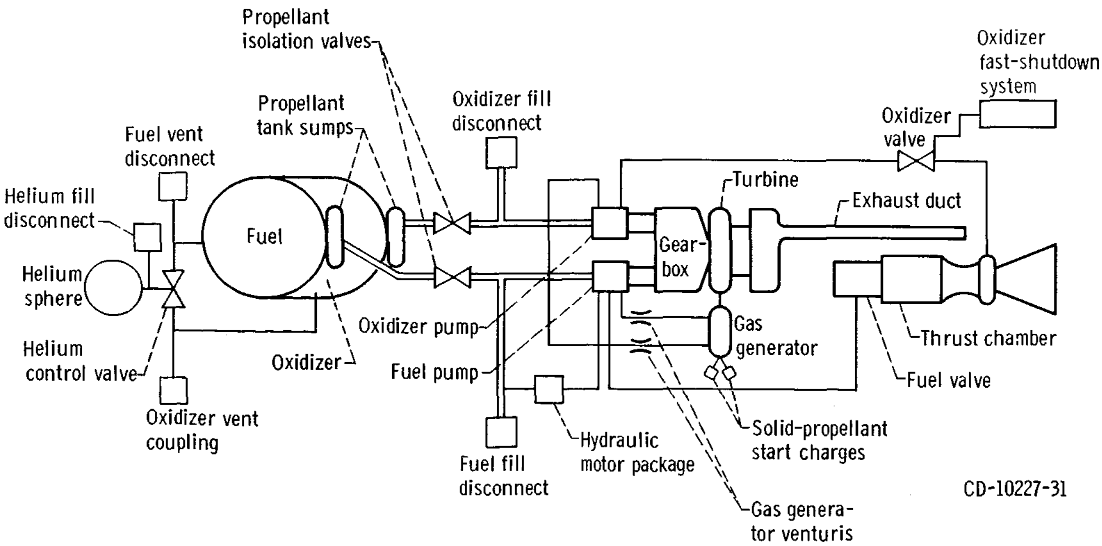


Figure VI-4. - Agena propulsion system, Nimbus III.

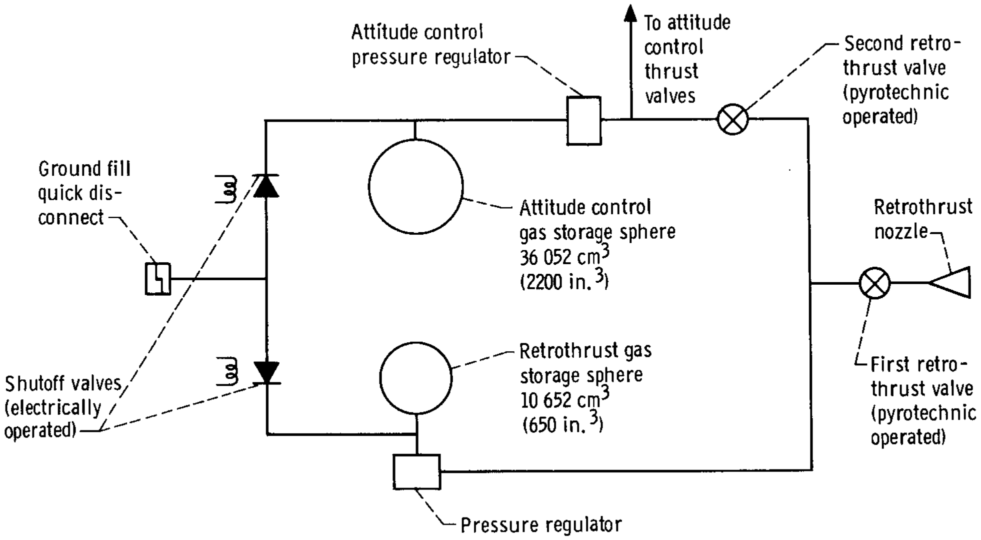


Figure VI-5. - Agena retrothrust system, Nimbus III.

GUIDANCE AND FLIGHT CONTROL SYSTEM

by Howard D. Jackson

The Agena flight path is controlled by two interrelated systems: the Agena guidance and flight control system and the radio-guidance system. The Agena guidance and flight control system directs the Agena, after Thorad-Agena separation, in a preprogrammed open-loop mode. The radio-guidance system will supply, if needed, pitch and yaw steering commands during a major portion of the Agena first powered phase. The radio-guidance system also issues a discrete command to enable the Agena velocity meter about midway through the Agena first powered phase. The radio-guidance system description, location of components, and use during the Thorad phase of flight is provided in the Thorad GUIDANCE AND FLIGHT CONTROL SYSTEM (section V).

Description

The Agena guidance and flight control system consists of three subsystems: a guidance subsystem, a control subsystem, and a timer to provide flight programming. A block diagram of the system is shown in figure VI-6.

The Agena guidance subsystem consists of an inertial reference package (IRP), horizon sensors, a velocity meter, and a guidance junction box. All components of the guidance subsystem are located in the guidance module in the Agena forward section. Primary attitude reference is provided by three orthogonal rate-integrating gyros in the IRP. (These gyros are uncaged at shroud vernier engine cutoff.) The infrared horizon sensors, consisting of a left and right optical sensor (head) and a mixer box, provide pitch and roll error signals to the IRP. For this mission the pitch horizon sensor signal is inhibited until after Agena engine first cutoff. The Agena yaw attitude is referenced to the attitude of the vehicle at the time of Thorad vernier engine cutoff and is then corrected by gyro-compassing techniques during long (greater than 10 min) coast periods. The velocity meter consists of an accelerometer, an electronics package, and a counter. The velocity meter accelerometer senses vehicle longitudinal acceleration. The velocity meter electronics processes the acceleration information and produces an output pulse each time the velocity increases by a known increment. The velocity meter counter generates an engine cutoff command when a predetermined number of pulses (i.e., when the sum of the velocity increments equals the total velocity to be gained) have been received. The guidance junction box serves as a center for guidance signals and contains relays for control of operating modes and gains.

The Agena flight control subsystem, which controls vehicle attitude, consists of a flight control electronics unit, a cold-gas attitude control system, a hydraulic attitude control system, and a flight control junction box. Attitude error signals from the IRP are conditioned and amplified by the flight control electronics to operate the cold-gas and hydraulic systems. During Agena coast periods, the cold-gas system consisting of six thrusters provides roll, pitch, and yaw control. These thrusters are located in the Agena aft section and operate on a mixture of nitrogen and tetrafluoromethane. During the Agena powered flight, the hydraulic system provides pitch and yaw control by means of two hydraulic actuators which gimbal the Agena engine thrust chamber; and the cold-gas system provides roll control. A patch panel in the flight control junction box provides the means for preprogramming the interconnections of the guidance and flight control system to meet mission requirements.

The Agena timer programs the flight events. This timer provides 22 usable discrete event times with multiple switch closure capability and has a maximum running time of 6000 seconds. The Agena timer is started at Thorad main engine cutoff.

The radio-guidance system airborne control package transfers steering control from the Thorad to the Agena at Thorad-Agena separation. The capability of the Agena guidance and flight control system to accept radio-guidance system pitch and yaw steering commands is enabled 8 seconds before and is disabled 141 seconds after Agena engine first start, by the Agena timer. During this 149-second period of Agena flight all radio-guidance system pitch and yaw steering commands (generated by the ground-based computer and transmitted to the Agena) are routed to the Agena guidance and flight control system to provide corrections for deviations from the programmed trajectory.

The radio-guidance system also provides the discrete command to the Agena for enabling the Agena velocity meter. The ground-based computer determines the time for this discrete command based on inflight performance of the Thorad and the Agena. With nominal Thorad and Agena performance, this discrete command occurs 133 seconds after Agena engine first start.

After the radio-guidance system has completed its planned period of operation, the airborne components are turned off to conserve Agena power. The Agena timer performs this function 264 seconds after Agena engine first start (about 32 sec after first cutoff).

Performance

The guidance and flight control system performance was satisfactory through both operations of the Agena engine, Nimbus III separation, and the conical turn prior to the first retrothrust. During the first operation of the retrothrust system, the cold-gas attitude control system was preprogrammed to operate in a low thrust mode and was not capable of counteracting the torque introduced by the retrothrust system. (See the Agena

PROPULSION section for discussion on the cause of this torque.) The torque caused by the operation of the retrothrust system acted primarily in the pitch plane and caused the vehicle to pitch-up approximately 160° with respect to the desired attitude. At the end of first retrothrust the vehicle was stable with about 150° pitch-up error.

All events initiated by the Agena timer were within tolerance. A comparison of the expected and actual times of programmed events is given in appendix A. The rates imparted to the Agena at Thorad-Agena separation (T + 238.1 sec) and the attitude errors at cold-gas activation (T + 240.7 sec) were within the range of values experienced on previous flights and are shown as follows:

Rates imparted to Agena at separation, deg/sec			Attitude errors at cold-gas activation, deg		
Yaw	Roll	Pitch	Yaw	Roll	Pitch
0.154 right	0.328 CCW[a]	0.263 down	1.46 right	0.53 CCW	2.06 down

[a]Counterclockwise (CCW). See fig. VI-7 for reference orientation.

The cold-gas attitude control system reduced these errors to within the deadband limits of ±0.2° pitch, ±0.18° yaw, and ±0.6° roll in 6 seconds.

At T + 248.2 seconds, the Agena initiated a programmed pitch down of 10.4° at a rate of 90 degrees per minute. The pitch-down rate was then decreased to 0.955 degree per minute at Agena engine first start. For the Agena first powered phase the radio-guidance steering was enabled in pitch and yaw with the horizon sensors controlling only the roll gyro. At the time of Agena engine first start (T + 256.2 sec), the pitch attitude was approaching null with an error of 1.08° pitch-up, and the vehicle was stable in the roll and yaw.

Gas generator turbine spin-up at Agena engine first start resulted in a roll rate and induced a maximum roll error as follows:

Roll rate, deg/sec	1.74 [a]CW
Maximum roll error, deg	2.64 CW
Time to reverse initial rate, sec	1.7

[a]Clockwise. See fig. VI-7 for reference orientation.

Minimal attitude control was required during the Agena engine first powered phase, and the vehicle attitude remained very close to gyro null positions. The attitude control activity (hydraulic and cold gas) was normal throughout this phase. Radio-guidance system steering commands were slight in pitch and negligible in yaw during the period programmed for use.

The velocity meter was enabled by the radio-guidance system at T + 390.5 seconds and commanded Agena engine first cutoff at T + 488.0 seconds, when the vehicle had attained the required velocity increment. The roll transients caused by engine cutoff (i.e., turbine spindown and turbine exhaust decay) were as experienced on previous flights. The time required to reduce the roll excursion to within the attitude control deadbands was 15 seconds.

Approximately 5.5 seconds after Agena engine first cutoff, a programmed geocentric pitch rate of 3.69 degrees per minute was applied, and the pitch horizon sensor was connected to the pitch gyro. The airborne components of the radio-guidance system were turned off (at T + 520.2 sec) about 32 seconds after first cutoff. Horizon sensor, gyro, and attitude control data showed that the vehicle maintained the proper attitude during the 46-minute coast.

Gas generator turbine spin-up at Agena engine second start (T + 3261.2 sec) resulted in a roll rate and induced a maximum roll error as follows:

Roll rate, deg/sec	2.31 [a]CW
Maximum roll error, deg	1.69 CW
Time to reverse initial rate, sec	1.5

[a]Clockwise. See fig. VI-7 for orientation reference.

Minimal attitude control was required during the Agena engine second powered phase. Engine cutoff was commanded by the velocity meter at T + 3267.4 seconds when the vehicle had attained the required velocity increment.

At T + 3421.2 seconds the Agena was commanded to pitch-up 75° from local horizontal at a rate of 56 degrees per minute to position the vehicle for the Nimbus III separation. The horizon sensors were disconnected from the pitch and roll gyros at the start of this pitch maneuver. The vehicle was stable for Nimbus III separation, which occurred at T + 3517.1 seconds. Two seconds after Nimbus III separation, a conical turn maneuver (simultaneous rates of 40 deg/min left yaw and 40° deg/min clockwise roll) was initiated. This maneuver oriented the vehicle in a nose-aft attitude, with respect to the velocity vector, with the horizon sensors viewing the Earth.

At T + 3710.2 seconds, the conical turn rates were stopped, the horizon sensors were reconnected to the pitch and roll gyros, and the first operation of the retrothrust system was initiated. However, during this operation of the retrothrust system, the unexpected torque occurred which resulted in the Agena pitching up. This unexpected torque was about 56.5 newtons-centimeter (5 in.-lb) greater than the pitch correction capability of the attitude control system. The attitude control system is designed to operate either a low-pressure or a high-pressure mode. It was preprogrammed to operate in the low-pressure mode during the first operation of the retrothrust system. The following table presents the pitch force level and the corresponding pitch torque level for the low and high modes of operation of the attitude control system.

Type mode	Pitch force level		Pitch torque level[a]	
	N	lbf	N-cm	in.-lb
Low pressure mode	2.22	0.5	678	60
High pressure mode	44.5	10	13 560	1200

[a]Distance from attitude control thruster nozzle to c.g., of Agena is approximately 304.8 cm (120 in.).

Shortly after the Agena began the unexpected pitch-up, the pitch horizon sensor and the pitch gyro error signals reached their maximum telemetry outputs (5° for the pitch horizon sensor and 10° for the pitch gyro). The pitch thrust valve was on full time in response to these error signals. The horizon sensor telemetry output remained at 5° for about 104 seconds. At this time the Agena had pitched up about 105° and both horizon sensor heads lost their Earth reference. Consequently, the horizon sensors could no longer detect any attitude error and provide correction signals. The pitch gyro error remained at 10°, and the pitch thrust valve remained on full time until near the end of the first operation of the retrothrust system.

At this time the pressure in the retrothrust sphere had decreased sufficiently so that the corresponding torque was within the correction capability of the attitude control system, and the Agena had been pitched up about 160°. The 10° pitch gyro error was then corrected. At the end of the first operation of the retrothrust system, the Agena was stable in attitude at an approximately 150° pitch error, the pitch gyro was at null, and the horizon sensors were viewing space.

The second operation of the retrothrust system was initiated by the Agena timer at T + 6217.2 seconds. At this time the attitude of the Agena was essentially the same as its attitude at the end of the first operation of the retrothrust system. The attitude control system was preprogrammed to operate in the high-pressure mode for the second operation of the retrothrust system. In this mode the attitude control system correction capability exceeded the unexpected torque produced by the retrothrust system. Consequently, the attitude of the Agena was stable during the second intended retrothrust. Since the Agena was improperly oriented (due to the 150° pitch-up error) at the start of the second operation of the retrothrust system, the velocity of the Agena and the EGRS-13 was increased; whereas, according to mission plan, the velocity should have been decreased. The EGRS-13 was successfully separated from the Agena at T + 6384.6 seconds during the second operation of the retrothrust system.

The effects of the intended retrothrusts on the Agena orbit and the EGRS-13 orbit are discussed in the TRAJECTORY AND PERFORMANCE (section IV).

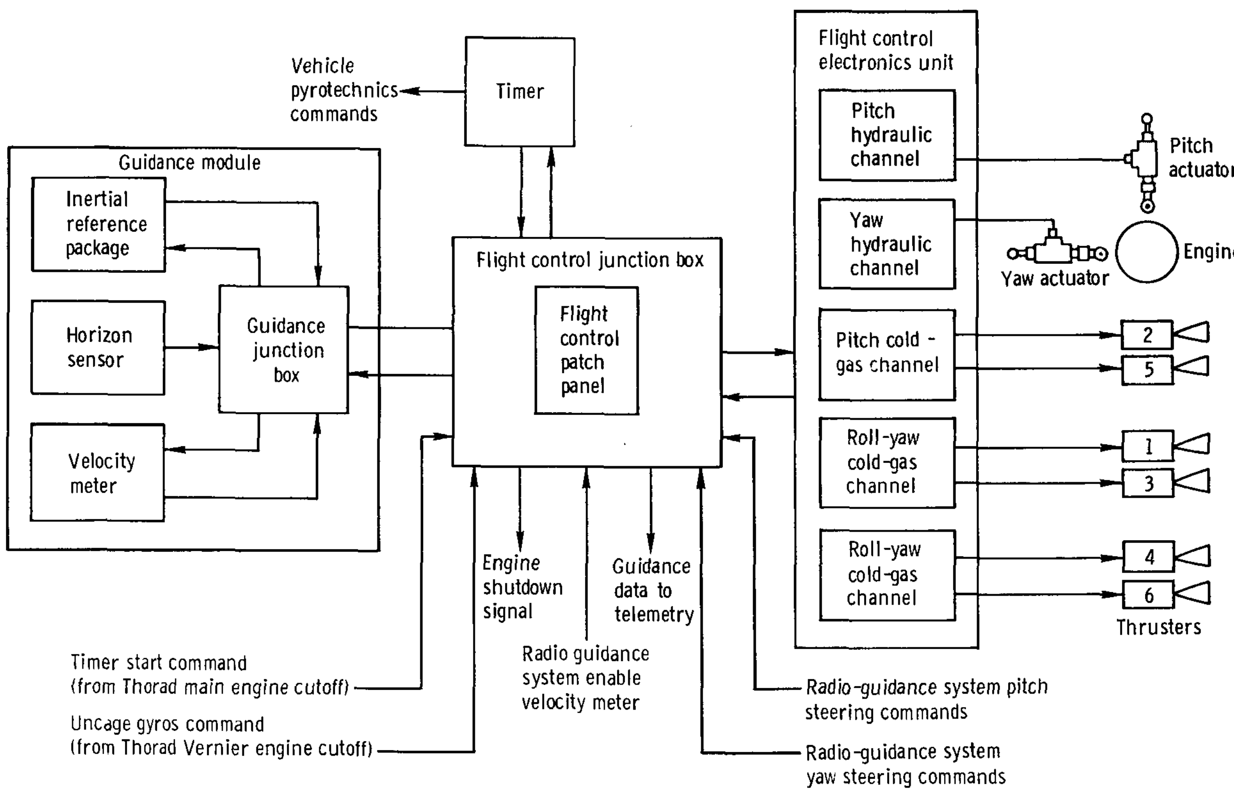


Figure VI-6. - Agena guidance and flight control system block diagram and radio-guidance system functions, Nimbus III.

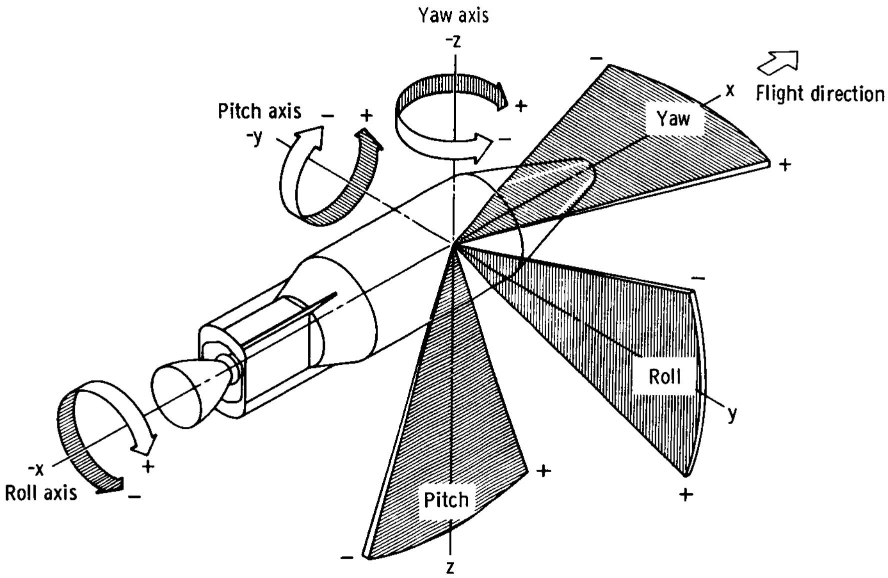


Figure VI-7. - Agena vehicle axes and vehicle movement designations, Nimbus III. Clockwise (CW) and counter clockwise (CCW) roll reference applies when looking forward along Agena lontitudinal axis.

ELECTRICAL SYSTEM

by Edwin R. Procasky

Description

The Agena electrical system (fig. VI-8) supplies all power, frequency, and voltage requirements for the pyrotechnics, propulsion, flight termination, inertial guidance, radio guidance, and telemetry systems. The electrical system consists of the power source equipment, power conversion equipment, and the distribution network.

The power source equipment consists of two silver-zinc primary batteries (minimum design rating of 966 W-hr each) and two nickel-cadmium secondary type batteries. One primary type battery (the main battery) supplies power to the vehicle loads that use unregulated power and to the power conversion equipment. The other primary type battery (the pyrotechnic battery) supplies power to all Agena vehicle pyrotechnics except the destruct charges in the flight termination system. The pyrotechnic battery is also connected to the main battery through a diode so that it can support the load on the main battery. However, the diode isolates the main battery loads from pyrotechnic transients and from pyrotechnic loads. The two secondary type batteries are used with the flight termination system.

The power conversion equipment consists of one static inverter and two dc-dc converters and converts unregulated dc power to regulated ac and regulated dc power. The inverter supplies 115 volts (rms) at 400 hertz (± 0.02 percent) to the guidance and flight control system. One dc-dc converter supplies regulated ± 28 volts dc to the guidance and flight control system. The second dc-dc converter, which has two regulated outputs, supplies 28 volts dc to the radio-guidance system and the telemetry system.

Performance

The Agena electrical system voltages and currents were as expected at lift-off, and the system satisfactorily supplied power to all electrical loads throughout the flight.

The battery (main and pyrotechnic) load profile was as expected for this mission. The inverter and converter voltages were within specification at lift-off and remained essentially constant throughout flight.

The inverter frequency was not monitored on the Agena; however, performance of the guidance and flight control system indicated that the inverter frequency was normal and stable. The electrical system performance data are summarized in table VI-I.

TABLE VI-I. - AGENA ELECTRICAL SYSTEM FLIGHT PERFORMANCE SUMMARY, NIMBUS III

Measurement	Range	Measurement number	Lift-off	At first ignition	At first shutdown	At second ignition	At second shutdown	First orbital pass
Pyrotechnic battery voltage	22.5 - 29.5 V	C-141	26.5 V	25.9 V	25.9 V	25.9 V	25.9 V	29.9 V
Main battery voltage	22.5 - 29.5 V	C-1	25.9 V	25.4 V	25.7 V	25.2 V	25.3 V	25.3 V
Battery current	-----------------	C-4	12 A	15 A	12 A	15 A	12 A	10 A
Converter output (guidance and flight control)								
+28.3 V dc regulated	27.7 - 28.9 V dc	C-3	28.2 V dc	28.2 V dc	28.2 V dc	28.2 V dc	28.2 V dc	28.2 V dc
-28.3 V dc regulated	-27.7 - -28.9 V dc	C-5	-28.5 V dc	-28.5 V dc	-28.5 V dc	-28.5 V dc	-28.5 V dc	-28.5 V dc
Inverter output								
Phase AB rms	112.7 - 117.3 V ac	C-31	114.6 V ac	114.6 V ac	114.6 V ac	114.6 V ac	114.6 V ac	114.6 V ac
Phase BC rms	112.7 - 117.3 V ac	C-32	114.6 V ac	114.6 V ac	114.6 V ac	114.6 V ac	114.6 V ac	114.6 V ac
Converter output, +28.3 V dc regulated								
Telemetry	27.7 - 28.9 V dc	C-2	27.8 V dc	27.8 V dc	27.8 V dc	27.8 V dc	27.8 V dc	27.8 V dc
Radio guidance	27.7 - 28.9 V dc	BTL-6	28.2 V dc	28.2 V dc	28.2 V dc	28.2 V dc	28.2 V dc	28.2 V dc

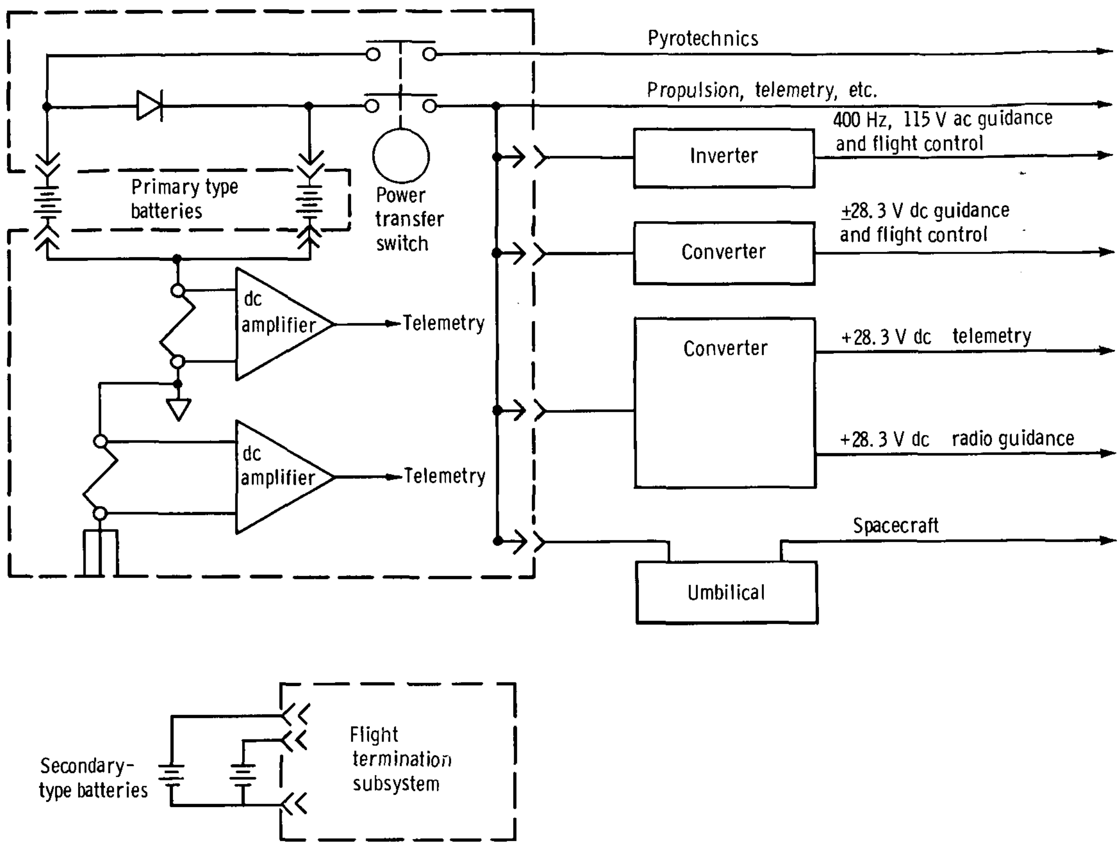


Figure VI-8. - Agena electrical system, Nimbus III.

COMMUNICATION AND CONTROL SYSTEM

by Richard L. Greene

Description

The Agena communication and control system consists of telemetry, tracking, and flight termination subsystems with associated power supplies and cabling.

The telemetry subsystem is mounted in the Agena forward section. It monitors and transmits the Agena functional and environmental measurements during flight. The frequency modulation/frequency modulation (FM/FM) telemetry unit contains a very high frequency (VHF) transmitter, voltage controlled oscillators, a commutator, a switch and calibrate unit, a radiofrequency switch, and an antenna. Regulated 28 volts dc power for telemetry is supplied from a dc-dc converter. The radiofrequency switch connects the telemetry output to either the umbilical for ground checkout or the antenna for flight. The transmitter operates on an assigned frequency of 244.3 megahertz at a power output of 2 watts. The telemetry subsystem consists of nine continuous subcarrier channels and two commutated subcarrier channels.

Fifty-nine measurements are telemetered from the Agena vehicle. Appendix B summarizes the launch vehicle instrumentation by measurement description. Four continuous subcarrier channels are used for monitoring acceleration and vibration data at the spacecraft adapter; three continuous channels are used for radio-guidance system measurements; one continuous channel monitors the gas thruster valve activity; and one continuous channel is time shared by the velocity meter accelerometer and the velocity meter counter. The turbine speed signal does not use a subcarrier channel but directly modulates the transmitter during engine operation. The remaining 48 measurements are monitored on the two commutated subcarrier channels. These channels are commutated at 5 revolutions per second with 60 segments on each channel.

The airborne tracking subsystem includes a C-band radar transponder, a radiofrequency switch, and an antenna. The transponder receives coded signals from the tracking radar on a carrier frequency of 5630 megahertz and transmits coded responses on a carrier frequency of 5555 megahertz at a minimum pulsed-power output of 200 watts at the input terminals of the antenna. The coded responses are at pulse rates (pulse repetition frequency) from 0 to 1600 pulses per second. The pulse rate is dependent on the rates transmitted from the ground tracking stations and the number of stations simultaneously interrogating the transponder. The radiofrequency switch connects the output of the transponder to either the umbilical for ground checkout or the antenna for flight.

The Agena flight termination subsystem (located on the booster adapter) provides a range safety flight termination capability for the Agena from lift-off until Thorad-Agena separation. This subsystem is composed of two batteries, interconnecting wiring assemblies, two separation switches, a destruct initiator with two detonators, and a destruct charge. Flight termination can be initiated by a signal from either of the Thorad command receivers prior to Thorad-Agena separation, or automatically if Thorad-Agena separation occurs before Thorad main engine cutoff (i.e., premature). The automatic portion of the system is disabled at Thorad main engine cutoff to permit a normal Thorad-Agena separation.

A time delay circuit in the Thorad safe/arm mechanisms insures destruction of both stages by delaying Thorad destruct initiation until 0.1 second after Agena destruct initiation. Agena destruct is effected by ignition of a shaped mounted on the booster adapter which ruptures the propellant tanks causing mixing of the hypergolic propellants.

Performance

The telemetry subsystem performance was satisfactory throughout the flight. Signal strength data from all participating ground telemetry stations indicated an adequate and continuous signal level from the vehicle telemetry transmitters from lift-off through the completion of the Agena second retrothrust. Analysis of the telemetry data indicated that the performance of the voltage controlled oscillators, switch and calibrate unit, dc-dc converters, and the commutator were satisfactory. Usable data were obtained from all Agena telemetered instrumentation. Appendix C (fig. C-2) presents the coverage provided by the supporting telemetry stations.

The tracking subsystem performance was satisfactory throughout the flight. The C-band transponder transmitted a continuous response to received interrogation during all periods of radar tracking. Analysis of ground radar signal strength records indicated that received signal levels were lower than expected during the first orbital pass over radar stations at Vandenberg Air Force Base and Hawaii. These low signal levels were caused by the Agena C-band antenna being pointed in a direction which lowered the effective antenna gain as viewed by the ground tracking stations. The incorrect pointing direction of the C-band antenna resulted from the Agena being in an improper attitude during the first orbital pass over the radar stations at Vandenberg Air Force Base and Hawaii (see Agena GUIDANCE AND FLIGHT CONTROL (section VI) for a discussion on the Agena attitude anomaly). Appendix C (fig. C-3) presents the coverage provided by the supporting radar tracking station.

The Agena flight termination subsystem was not monitored during flight. However, because of the system redundancy, it is assumed that the system was capable of destructing the Agena throughout the Thorad powered phase

VII. LAUNCH OPERATIONS

by Alvin C. Hahn

PRELAUNCH ACTIVITIES

The major prelaunch activities at the Western Test Range (WTR) are shown in table VII-I. During the tests problems were satisfactorily resolved as discussed in the following paragraphs.

Thorad electrical connector X-rays. - NASA directed that an X-ray examination be made of 99 of the readily accessible and flight critical connectors on the Thorad. The results of this inspection indicated that 23 of the connectors were suspect, and these were opened for further inspection, cleaning, or rework.

Thorad turbopump torque test. - An out of specification variation in the running torque was observed while the turbine in the turbopump was being manually rotated. The second stage nozzle of the turbine assembly was repositioned and the turbopump was retested satisfactorily.

Thorad single-propellant flow. - The liquid-oxygen propellant valve for vernier engine No. 1 indicated leakage and was replaced. The fuel depletion float switch stuck in the high position and was replaced. A regulation circuit in the ground power supply for the ground inverter malfunctioned and was replaced.

Thorad dual-propellant flow. - The fuel depletion float switch stuck in the high position and was replaced. An expansion joint in the liquid-oxygen transfer line to the booster leaked and was retorqued.

COUNTDOWN AND LAUNCH

The countdown for the first launch attempt of the Thorad-Agena-Nimbus III began on April 10, 1069, concurrent with the countdown of a United States Air Force mission which had launch priority. The Nimbus III countdown started at 1123 Pacific standard time (PST) and proceeded as planned to the start of Agena propellant tanking. At this time the status of the Air Force mission was reevaluated. The results of the evaluation indicated that the countdown for the Air Force mission was continuing successfully toward its scheduled launch. Consequently, the Nimbus III countdown was terminated.

The countdown for the second launch attempt began at 1123 PST on April 11, 1969.

The countdown proceeded without significant problems until 2234 PST when the fuel leak detector (Aerospace Ground Equipment) for the Agena showed the presence of fuel vapors. The cause of the vapors could not be determined and the launch attempt was terminated at 0022 PST on April 12, 1969. Pressure checks after countdown termination and propellant unloading showed a small leak through the vent port of the fuel propellant isolation valve. The gas leakage rate observed during the pressure check was determined to be within specification limits. A plastic tube and bottle was attached to the vent line on the fuel propellant isolation valve so that the actual fluid leakage rate of fuel could be checked during the next scheduled countdown.

The final countdown for launch was initiated at 1124 Pacific standard time on April 13, 1969. Early completion of the launch vehicle tasks through mobile service tower removal permitted early tanking of the Agena and provided a 2-hour check on the leakage rate through the vent port of the fuel propellant isolation valve. Only a trace of fuel had collected in the bottle after the 2-hour period, and the plastic tubing and bottle were removed. No significant problems occurred during the countdown. Lift-off, 5.08-centimeter (2-in.) motion, occurred at 2354:03.136 Pacific standard time on April 13, 1969.

TABLE VII-I. - MAJOR PRELAUNCH

ACTIVITIES FOR NIMBUS III

Date	Event
1/17/69	Thorad Arrival at Vandenberg AFB
2/19/69	Agena Arrival at Vandenberg AFB
3/19/69	Spacecraft Arrival at Vandenberg AFB
3/30/69	Thorad-Agena Mate
3/30/69	Thorad-Agena Erection
4/01/69	Agena-Spacecraft Mate
4/03/69	Simulated Launch
4/10/69	First Launch Attempt
4/11/69	Second Launch Attempt
4/13/69	Launch

VIII. CONCLUDING REMARKS

The Thorad-Agena launch vehicle, carrying the Nimbus III as a primary payload and the Engineers Geodetic Ranging Satellite-13 (EGRS-13) as a secondary payload, was launched on the third attempt (previous attempts were made on April 10 and April 11) at 2354:03.136 Pacific standard time on April 13, 1969. The 576-kilogram Nimbus III meteorological satellite was successfully separated from the Agena in the desired near-polar orbit with a perigee altitude of 1087 kilometers and an apogee altitude of 1143 kilometers. The 21-kilogram EGRS-13 attached to the Agena aft section was separated as planned, approximately 48 minutes after Nimbus III separation.

During the first operation of the Agena retrothrust system, shortly after the Nimbus III separated from the Agena, an Agena attitude anomaly occurred. This anomaly resulted in the Agena (and therefore the intended retrothrust vector) being misoriented during the first and second operation of the retrothrust system. Although the final Agena orbit and EGRS-13 orbit were perturbed by the Agena misorientation during the intended retrothrust periods, there was no effect on the mission success of either the Nimbus III or the EGRS-13.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, February 24, 1970,
493-01.

APPENDIX A

SEQUENCE OF MAJOR FLIGHT EVENTS

by Richard L. Greene

Nominal time, sec	Actual time, sec	Event description	Initiated by	Event monitor (a)
0	0	Lift-off (2354:03.136PST) (April 13, 1969)	------------------	Lift-off switch
38.61	39.2	Solid motor burnout	------------------	Solid motor chamber pressure[b]
102.0	102.2	Solid motor jettison	Thorad timer	Sequence 3
124.0	124.3	Enable Thorad radio guidance steering	Thorad programmer	--------------------------------
221.64	222.2	Thorad main engine cutoff	Thorad propellant depletion	Main engine chamber pressure[b]
221.64	222.2	Start Agena timer		Guidance and control monitor (D-14)
230.64	231.2	Thorad vernier engine cutoff and Agena horizon sensor fairing jettison		Vernier engine chamber pressure[b]
237.22	238.1	Thorad-Agena separation	Radio guidance	Radial vibration (AA006)
237.92	238.7	Transfer radio-guidance steering to Agena	Pullaway plug	BTL relay transfer (BTL-5)
239.78	240.7	Activate pneumatic attitude control system	Separation switch	Guidance and control monitor (D-14)
247.64	248.2	Initiate -90 deg/min pitch rate	Agena timer	Pitch torque rate (D-73)
		Enable Agena radio-guidance steering		Estimated[c]
255.64	256.2	Transfer to -0.955 deg/min pitch rate		Pitch torque rate (D-73)
		Agena engine first start		Switch group Z (B-13)
256.79	266.3	Agena engine at 90 percent chamber pressure		Chamber pressure (B-91)
265.64	257.4	Shroud separation		Shroud separation monitor (A52)
388.62	390.5	Enable Agena velocity meter	Radio guidance	Velocity meter accelerator (D-83)
487.61	488.0	Agena engine first cutoff	Velocity meter	Chamber pressure (B-91)
3260.64	3261.2	Agena engine second start	Agena timer	Switch group Z (B-13)
3261.77	3262.3	Agena engine at 90 percent chamber pressure	------------------	Chamber pressure (B-91)
3267.05	3267.4	Agena engine second cutoff	Velocity meter	Chamber pressure (B-91)
3420.64	3421.2	Transfer to +56 deg/min pitch rate	Agena timer	Pitch torque rate (D-73)
3500.64	3501.2	Transfer to 0 deg/min pitch rate		Pitch torque rate (D-73)
3516.64	3517.1	Agena-Nimbus III separation		Spacecraft separation monitor (PL60)
3518.64	3519.2	Initiate 40 deg/min yaw-roll rate		Yaw torque rate (D-51) Roll torque rate (D-66)
3709.64	3710.2	Stop yaw-roll rate		Yaw torque rate (D-51) Roll torque rate (D-66)
3709.64	3710.2	Start first retrothrust		Estimated[c]
6216.64	6217.2	Start second retrothrust		Estimated[c]
6385.0	6384.6	Agena-EGRS-13 separation		Radial vibration (AA006)

[a]All events except as noted were monitored on Agena telemetry. The designation in parenthesis is the Monitor Measurement designation. See the Launch Vehicle Instrumenation Summary (appendix B) for the measurement range and channel assignment.

[b]These events were identified from the Thorad telemetry data.

[c]No direct measurement to identify the event.

APPENDIX B

LAUNCH VEHICLE INSTRUMENTATION SUMMARY

by Richard L. Greene and Richard E. Orzechowski

Thorad Telemetry

Measurement number	Measurement title	Channel number	Measurement range	
			SI units	U.S. customary units
FM-1-01	Inverter frequency	1	370.0 - 340.0 Hz	
FM-1-07	Sequence 3	7	0 - 5.00 V	
	Solid motor No. 1 jettisoned			
	Solid motor No. 2 jettisoned			
	Solid motor No. 3 jettisoned			
FM-1-08	Sequence 1	8	0 - 5.00 V	
	Programmer start			
	Liquid-oxygen tank float switch			
	Fuel tank float switch			
	Main engine cutoff			
	Vernier engine cutoff			
FM-1-09	Vernier engine No. 2 chamber pressure (absolute)	9	0 - 344.5 N/cm^2	0 - 500 psi
FM-1-10	Sequence 2	10	0 - 5.00 V	
	Solid motor ignition arm			
	Solid motor ignition			
	Solid motor jettison arm			
	Solid motor jettison command			
FM-1-11	Main engine chamber pressure (absolute)	11	0 - 551.5 N/cm^2	0 - 800 psi
FM-1-12	Turbopump speed	12	0 - 8000 rpm	
FM-1-13	Solid motor No. 1 chamber pressure (absolute)	13	0 - 551.5 N/cm^2	0 - 800 psi
FM-1-A	Solid motor No. 3 chamber pressure (absolute)	A	0 - 551.5 N/cm^2	0 - 800 psi
FM-1-C	Solid motor No. 2 chamber pressure (absolute)	C	0 - 551.5 N/cm^2	0 - 800 psi
PDM-1-01	5-V transducer calibrations voltage	Ch. E, seg. 1	0 - 5.00 V	
PDM-1-02	Instrumentation ground	Ch. E, seg. 2	0 - 5.00 V	
PDM-1-03	Main engine pitch position	Ch. E, seg. 3	-5.00° - 5.00°	
PDM-1-04	Main engine yaw position	Ch. E, seg. 4	-5.00° - 5.00°	
PDM-1-05	Vernier engine No. 1 pitch-roll position	Ch. E, seg. 5	-45.0° - 45.0°	
PDM-1-06	Vernier engine No. 1 yaw position	Ch. E, seg. 6	-28.0° - -8.0°	
PDM-1-07	Vernier engine No. 2 pitch-roll position	Ch. E, seg. 7	-45.0° - 45.0°	
PDM-1-08	Vernier engine No. 2 yaw position	Ch. E, seg. 8	8.0° - 28.0°	
PDM-1-09	Pitch attitude error	Ch. E, seg. 9	-5.00° - 5.00°	
PDM-1-10	Yaw attitude control	Ch. E, seg. 10	-5.00° - 5.00°	
PDM-1-11	Roll attitude error	Ch. E, seg. 11	-7.00° - 7.00°	

Thorad Telemetry (concluded)

Measurement number	Measurement title	Channel number	Measurement range	
			SI units	U.S. customary units
PDM-1-12	Pitch rate	Ch. E, seg. 12	-5.00 - 5.00 deg/sec	
PDM-1-13	Yaw rate	Ch. E, seg. 13	-5.00 - 5.00 deg/sec	
PDM-1-14	Roll rate	Ch. E, seg. 14	-8.00 - 8.00 deg/sec	
PDM-1-15	Pitch command	Ch. E, seg. 15	-4.00 - 4.00 deg/sec	
PDM-1-16	Yaw command	Ch. E, seg. 16	-4.00 - 4.00 deg/sec	
PDM-1-17	Actuator potentiometer positive voltage	Ch. E, seg. 17	0 - 30.0 V	
PDM-1-18	Actuator potentiometer negative voltage	Ch. E, seg. 18	-30.0 - -13.0 V	
PDM-1-19	400-Hz phase A inverter voltage	Ch. E, seg. 19	109.0 - 121.0 V	
PDM-1-20	5-V potentiometer excitation voltage	Ch. E, seg. 20	0 - 5.00 V	
PDM-1-21	Control electronic amplifier + 165 V dc	Ch. E, seg. 21	0 - 200 V	
PDM-1-22	Main engine chamber pressure	Ch. E, seg. 22	0 - 551.5 N/cm^2	0 - 800 psia
PDM-1-23	Main battery voltage	Ch. E, seg. 23	0 - 32.0 V	
PDM-1-24	Telemetry battery voltage	Ch. E, seg. 24	0 - 32.0 V	
PDM-1-25	Hydraulic supply pressure	Ch. E, seg. 25	0 - 2756 N/cm^2	0 - 4000 psia
PDM-1-26	Hydraulic return pressure	Ch. E, seg. 26	0 - 137.8 N/cm^2	0 - 200 psia
PDM-1-27	Roll command	Ch. E, seg. 27	-8.00 - 8.00 deg/sec	
PDM-1-28	Turbine inlet temperature	Ch. E, seg. 28	255 - 1255.4 K	-200^{o} - 1800^{o} F
PDM-1-29	Fuel pump inlet pressure	Ch. E, seg. 29	0 - 137.8 N/cm^2	0 - 200 psia
PDM-1-30	Flight termination receiver No. 1 automatic gain control voltage	Ch. E, seg. 30	0 - 5000 μV	
PDM-1-31	Vernier engine No. 1 housing temperature (left)	Ch. E, seg. 31	255 - 810.9 K	0^{o} - 1000^{o} F
PDM-1-32	Vernier engine No. 2 housing temperature (right)	Ch. E, seg. 32	255 - 810.9 K	0^{o} - 1000^{o} F
PDM-1-33	Engine pneumatic bottle pressure	Ch. E, seg. 33	0 - 3447 N/cm^2	0 - 5000 psia
PDM-1-34	Control electronics amplifier - 165 V dc	Ch. E, seg. 34	-200 - 0.0 V	
PDM-1-35	Main engine pitch-yaw actuator temperature	Ch. E, seg. 35	255 - 810.9 K	0^{o} - 1000^{o} F
PDM-1-36	Flight termination receiver No. 2 automatic gain control voltage	Ch. E, seg. 36	0 - 5000 μV	
PDM-1-37	Air conditioning duct inlet temperature	Ch. E, seg. 37	255 - 810.9 K	0^{o} - 1000^{o} F
PDM-1-38	Skirt section temperature	Ch. E, seg. 38	255 - 810.9 K	0^{o} - 1000^{o} F
PDM-1-39	Liquid-oxygen pump inlet pressure	Ch. E, seg. 39	0 - 68.9 N/cm^2	0 - 100 psia
PDM-1-40	Main fuel tank top pressure	Ch. E, seg. 40	0 - 68.9 N/cm^2	0 - 100 psia
PDM-1-41	Gas generator liquid-oxygen injector pressure	Ch. E, seg. 41	0 - 551.5 N/cm^2	0 - 800 psia
PDM-1-42	Liquid-oxygen tank top pressure	Ch. E, seg. 42	0 - 68.9 N/cm^2	0 - 100 psia
PDM-1-43	Liquid-oxygen pump inlet temperature	Ch. E, seg. 43	88 - 102.6 K	-300.0^{o} - -275.0^{o} F

Agena Telemetry

Measurement number	Measurement title	Channel number (a)	Measurement range	
			SI units	U. S. customary units
A52	Shroud separation monitor	15-44	(b)	
A219	Diaphragm differential pressure	16-12/23/34/54	-3.4 - 3.4 N/cm^2	-5 - 5 psi
AA006	Radial vibration of Nimbus III adapter	17	-15 - 15 g's	
B1	Fuel pump inlet pressure (gage)	15-15	0 - 68.9 N/cm^2	0 - 100 psi
B2	Oxidizer pump inlet pressure (gage)	15-17	0 - 68.9 N/cm^2	0 - 100 psi
B11	Oxidizer venturi inlet pressure (absolute)	15-19/49	0 - 1034 N/cm^2	0 - 1500 psi
B12	Fuel venturi inlet pressure (absolute)	15-23/53	0 - 1034 N/cm^2	0 - 1500 psi
B13	Switch group Z (propulsion system monitor)	15-7/22/37/52	(b)	
B31	Fuel pump inlet temperature	15-6	255 - 311 K	0° - 100° F
B32	Oxidizer pump inlet temperature	15-8	255 - 311 K	0° - 100° F
B35	Turbine speed	(c)	0 - 600 Hz	
B91	Combustion chamber pressure No. 3 (gage)	15-4/34	328 - 379 N/cm^2	475 - 550 psi
B130	Propellant isolation valve monitor	15-41	(b)	
C1	28-V dc unregulated supply voltage	16-40	22 - 30 V dc	
C2	28-V dc regulated supply No. 3	16-33	22 - 30 V dc	
C3	28-V dc regulator No. 1 (guidance and control)	15-12	22 - 30 V dc	
C4	28-V dc unregulated current	16-13/44	0 - 100 A	
C5	-28-V dc regulator No. 1 (guidance and control)	15-30	-30 - -22 V dc	
C21	400-Hz inverter temperature	15-14	255 - 367 K	0° - 200° F
C31	400-Hz, three phase, phase AB voltage	15-18	90 - 130 V ac	
C32	400-Hz, three phase, phase BC voltage	15-20	90 - 130 V ac	
C38	Structure current	15-10/25/40/55	0 - 50 A	
C141	Pyrotechnic bus voltage	15-5/25	22 - 30 V dc	
D14	Guidance and control monitor	16-27	(b)	
D41	Horizon sensor pitch output	16-45	-5° - 5°	
D42	Horizon sensor roll output	16-46	-5° - 5°	
D46	Gas valve cluster No. 1 temperature	15-39	228 - 339 K	-50° - 150° F
D47	Gas valve cluster No. 2 temperature	15-36	228 - 339 K	-50° - 150° F
D51	Yaw torque rate	16-38		
	Ascent mode		-200 - 200 deg/min	
	Orbit mode		-10 - 10 deg/min	
D54	Horizon sensor head temperature			
	Right hand	15-47	228 - 367 K	-50° - 200° F
	Left hand	15-46	228 - 367 K	-50° - 200° F
B59	Control gas supply pressure (absolute)	16-47	0 - 2758 N/cm^2	0 - 4000 psi
B60	Hydraulic oil pressure (gage)	15-21	0 - 2758 N/cm^2	0 - 4000 psi
D66	Roll torque rate	16-41		
	Ascent mode		-50 - 50 deg/min	
	Orbit mode		-4 - 4 deg/min	

[a]The first number indicates the Interrange Instrumentation Group (IRIG) subcarrier channel used; the second, the commutated position for the measurement. If no second number is indicated, the channel was used continuously for the designated transducer.

[b]Events are determined by a step change in voltage.

[c]The turbine speed signal does not use a subcarrier channel, but directly modulates the transmitter during engine operation.

Agena Telemetry (concluded)

Measurement number	Measurement title	Channel number (a)	Measurement range	
			SI units	U.S. customary units
D68	Pitch actuator position	15-3	-2.5° - 2.5°	------------
D69	Yaw actuator position	15-24	-2.5° - 2.5°	------------
D70	Control gas supply temperature	15-42	228 - 367 K	-50° - 200° F
D72	Pitch gyro output	16-36		
	Ascent mode		-10° - 10°	
	Orbit mode		-5° - 5°	
D73	Pitch torque rate	16-35		
	Ascent mode		-200 - 200 deg/min	
	Orbit mode		-10 - 10 deg/min	
D74	Yaw gyro output	16-39		
	Ascent mode		-10° - 10°	
	Orbit mode		-5° - 5°	
D75	Roll gyro output	16-42		
	Ascent mode		-10° - 10°	
	Orbit mode		-5° - 5°	
D76	Retrothrust gas supply pressure (absolute)	16-21	0 - 2758 N/cm^2	0 - 4000 psi
D77	Retrothrust gas supply temperature	16-20	228 - 367 K	-50° - 200° F
D83	Velocity meter accelerometer	14	0 - 2000 pulses/sec	
D86	Velocity meter cutoff switch	16-28	(b)	
D88	Velocity meter counter	14	Binary code (50 bits/sec)	
D129	Inertial reference package internal case temperature	15-54	228 - 367 K	-50° - 200° F
D149	Gas valves 1 to 6 monitor	7	(d)	
H47	Beacon receiver pulse repetition rate	15-27	0 - 1600 pulses/sec	
H48	Beacon transmitter pulse repetition rate	15-28	0 - 1600 pulses/sec	
PL36	Nimbus III adapter y-axis acceleration	12	-5 - 5 g	
PL34	Nimbus III adapter x-axis acceleration	13	-5 - 5 g	
PL32	Nimbus III adapter z-axis acceleration	11	-15 - 15 g	
PL60	Nimbus III separation monitor	16-18	(b)	
BTL 1	Radio-guidance magnetron monitor	9	(b)	
BTL 2	Radio-guidance combined events monitor	10	(b)	
BTL 4	Radio-guidance automatic gain control monitor	8	-70 - 0 dBm	
BTL 5	Radio-guidance steering relay monitor	16-30	(b)	
BTL 6	Radio-guidance regulated voltage monitor	16-19/49	22 - 30 V dc	

[a]The first number indicates the Interrange Instrumentation Group (IRIG) subcarrier channel used; the second, the commutated position for the measurement. If no second number is indicated, the channel was used continuously for the designated transducer.

[b]Events are determined by a step change in voltage.

[d]A unique voltage level is associated with only one or a combination of several gas valve firings.

APPENDIX C

TRACKING AND DATA ACQUISITION

by Richard L. Greene

The launch vehicle trajectory from lift-off through the final operation of the retrothrust system is projected on a world map in figure C-1. Five tracking and data acquisition networks, Western Test Range (WTR), Pacific Missile Range (PMR), Eastern Test Range (ETR), Air Force Satellite Control Facility (AFSCF), and the NASA Manned Space Flight Network (MSFN), plus the NASA telemetry station at Vandenberg Air Force Base, provided telemetry and radar coverage of the launch vehicle for the Nimbus III mission. From these facilities 10 radar stations, eight telemetry stations, three real-time computer facilities, and two tracking cinetheodolites supported the launch vehicle requirements. The general quality of all collected data was excellent.

Telemetry Data

Telemetry signals from the Thorad-Agena launch vehicle were recorded on magnetic tape at supporting ground stations during all Thorad and Agena thrust periods, Agena attitude maneuvers, Agena-Nimbus III separation, Agena Geodetic Ranging Satellite-13 (AGRS-13) separation, and the Agena retrothrust periods. The data recorded on magnetic tape at each telemetry station were used for postflight analysis of the launch vehicle performance. Real-time monitoring at the launch site of all Thorad and Agena telemetry measurements permitted verification of all flight events and quick-look evaluation of the launch vehicle performance from lift-off through Agena engine first cutoff. Real-time retransmission of selected Agena telemetry measurements from Pretoria, South Africa, and Tananarive, Malagasy Republic, to the NASA telemetry station at Vandenberg Air Force Base, permitted verification of Agena engine second burn, Agena-Nimbus III separation, Agena attitude maneuvers, and the first operation of the Agena retrothrust system. Real-time monitoring of Agena telemetry measurements by the NASA telemetry station at Vandenberg Air Force Base during the first orbital pass permitted identification of the second operation of the Agena retrothrust system and the Agena-EGRS-13 separation. The time of occurrence of all flight events was identified by the supporting remote telemetry sites and voice reported to Vandenberg Air Force Base via radio and cable communication circuits.

All telemetry stations supporting the launch vehicle for the Nimbus III mission were operational, and received and recorded Thorad and Agena telemetry data. Figure C-2 shows the telemetry coverage provided by each station.

Radar Data

C-band radar data (time, elevation, azimuth, and range) were provided for both real-time operations and postflight analysis. The real-time radar data were provided for monitoring the launch vehicle flight performance for range safety purposes and for assisting the downrange stations in acquiring track of the vehicle. These data were also used for computation of orbital elements and insertion conditions at Agena first and second cutoff and at the completion of the second operation of the Agena retrothrust system.

Three real-time computers (at WTR, ETR, and MSFN) provided calculations for range safety, downrange station acquisition information, and descriptive parameters of the Agena intermediate orbits and the final orbit. Computer support is presented in table C-I. The radar coverage provided by the supporting tracking stations is presented in figure C-3.

TABLE C-I. - LAUNCH VEHICLE COMPUTER SUPPORT FOR

ORBITAL CALCULATIONS, NIMBUS III

Flight interval	Data source	Computer location	Type of computer support
Lift-off through Agena engine first cutoff	WTR Radars Ship "Wheeling" Radar	WTR	Range safety displays
		ETR, MSFN	Agena transfer orbit calculation
		ETR, MSFN	Acquisition information for downrange tracking stations
Agena engine second start through Agena retrothrust system first operation	Pretoria Radar Tananarive Radar	ETR, MSFN	Agena circular orbit calculation Agena orbit calculation after first retrothrust system operation Acquisition information for downrange tracking stations
Agena retrothrust system second operation	WTR Radars Hawaii Radar (MSFN)	MSFN	Agena orbit calculation after second retrothrust system operation

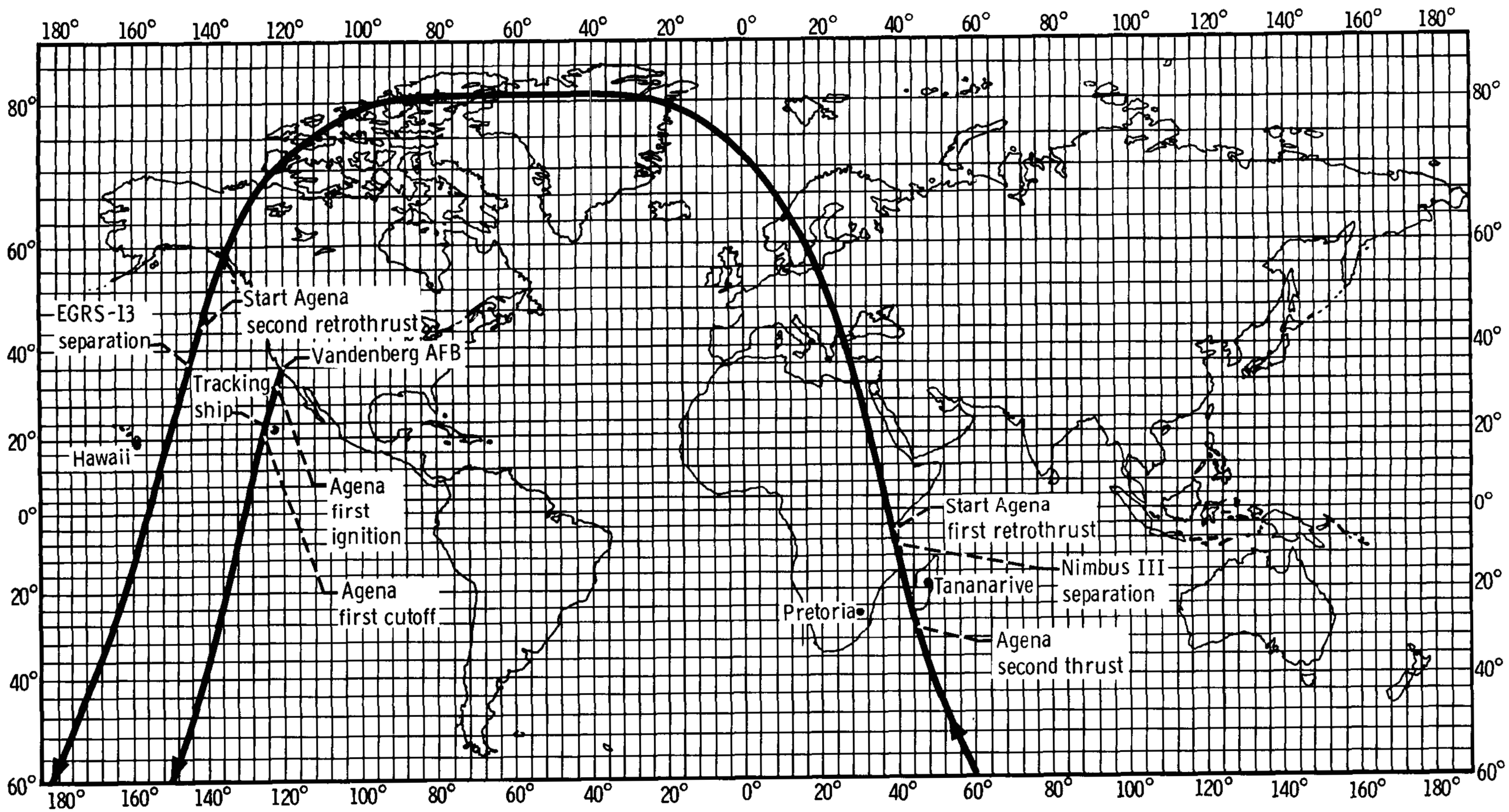


Figure C-1. - Ground trace of Nimbus III launch trajectory.

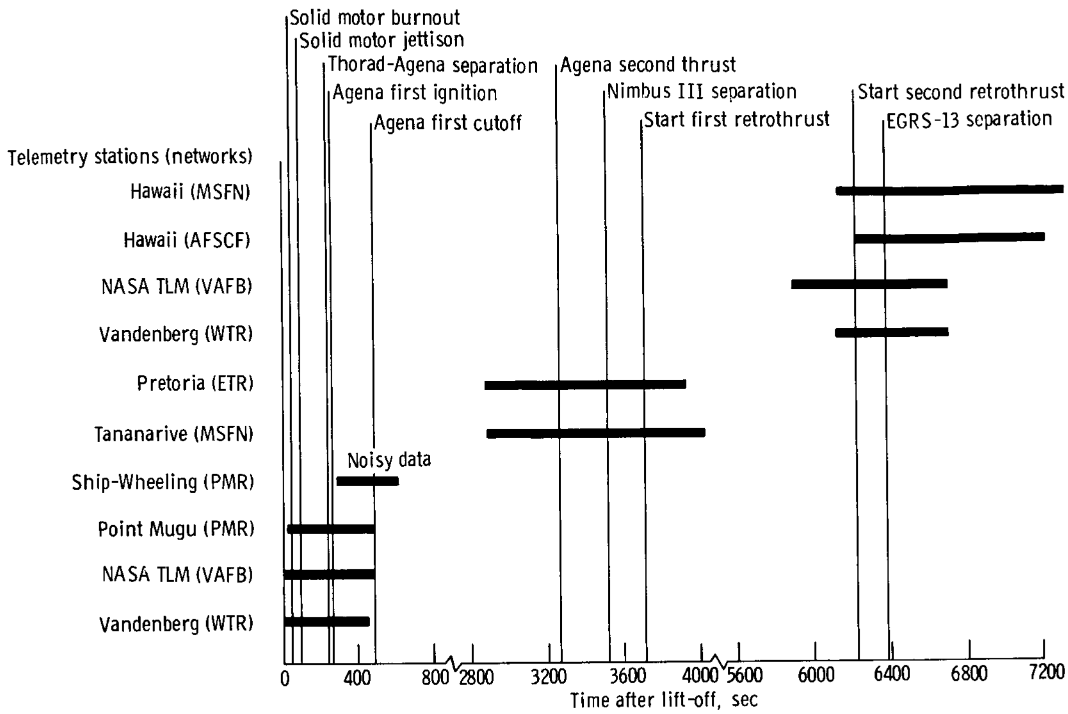


Figure C-2. - Launch vehicle telemetry coverage, Numbus III.

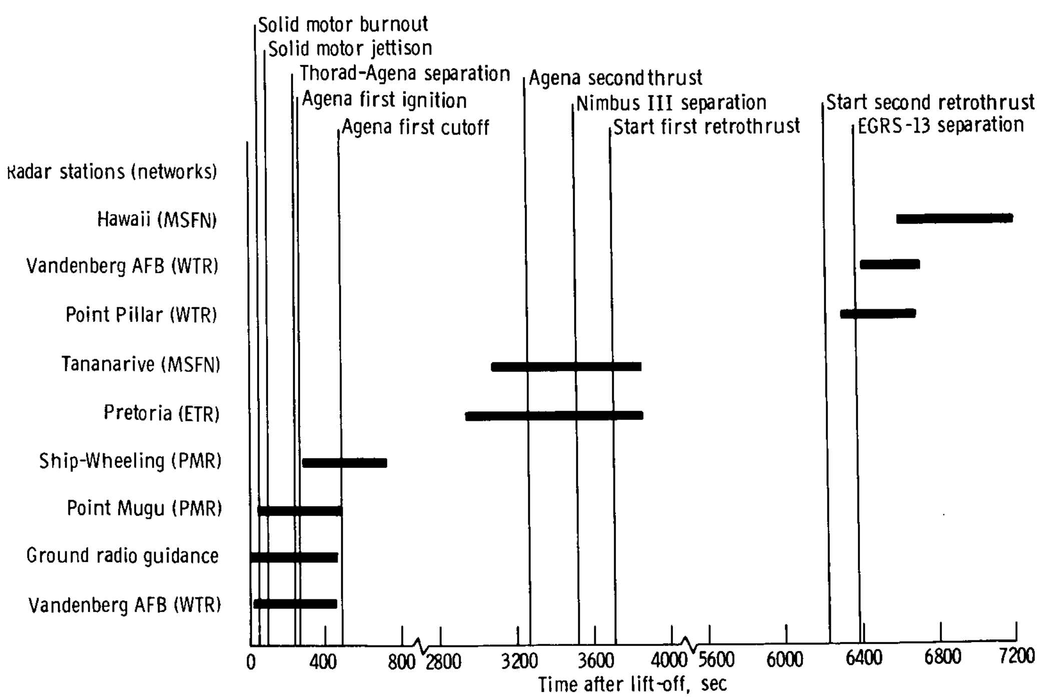


Figure C-3. - Launch vehicle radar coverage, Numbus III.

APPENDIX D

VEHICLE FLIGHT DYNAMICS

by Dana Benjamin

Flight dynamics data were obtained from three accelerometers and one vibrometer mounted on the spacecraft adapter. A summary of dynamic instrumentation locations and characteristics is presented in figure D-1.

Table D-I presents the actual flight times at which significant dynamic disturbances were recorded. Table D-II shows the maximum acceleration levels and corresponding frequencies recorded at times of significant dynamic disturbances during flight (All acceleration levels are shown in g's zero to peak). Data traces of the dynamic environment recorded by all four instruments for the events in table D-I are presented in figures D-2 to D-14.

TABLE D-I. - SUMMARY OF DYNAMIC

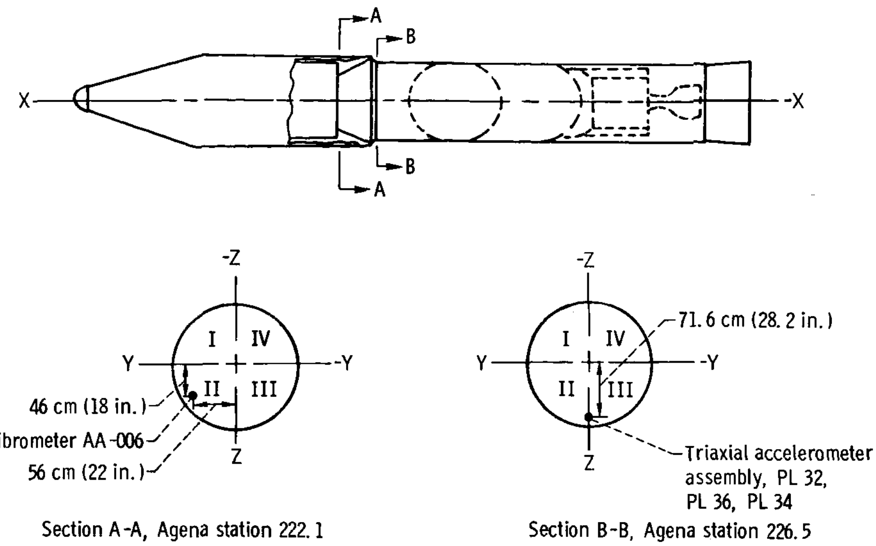
DISTURBANCES, NIMBUS III

Event causing disturbance	Time of dynamic disturbance, sec after lift-off
Lift-off	0.14
Solid motor, burnout	36.33
Solid motor jettison	102.19
Peak longitudinal oscillation POGO	209.14
Thorad main engine cutoff	222.24
Horizon sensor fairing jettison	231.18
Thorad-Agena separation	238.03
Agena engine first start	256.96
Shroud separation	266.55
Agena engine first cutoff	488.07
Agena engine second start	3262.30
Agena engine second cutoff	3267.44
Agena-Nimbus III separation	3517.16

TABLE D-II. - SUMMARY OF DYNAMIC ENVIRONMENT, NIMBUS III

Event causing disturbance	Time of dynamic disturbance, sec after lift-off	Accelerometer						Vibrometer	
		Channel 11		Channel 12		Channel 13		Channel 17	
		Measurement							
		PL-32 (longitudinal x-axis)		PL-36 (lateral y-axis)		PL-34 (lateral z-axis)		AA-006 (radial)	
		Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)
Lift-off	0.14	(a)	(a)	160	0.52	180	0.47	800	3.8
Solid motor burnout	36.33	180	0.78	170	1.0	190	.7	710	2.5
Solid motor jettison	102.19	(a)	(a)	95	.37	(a)	(a)	(a)	(a)
Peak longitudinal oscillation POGO	209.14	17.5	3.91	50	.47	35	.37	(a)	(a)
Thorad main engine cutoff	222.24	(a)	(a)	110	.57	(a)	(a)	(a)	(a)
Horizon sensor fairing jettison	231.18	160	1.8	210	1.0	90	2.2	860	6.9
Thorad-Agena separation	238.03	(a)	(a)	300	.42	(a)	(a)	1650	13
Agena engine first start	256.96	110	1.0	90	.42	95	.42	1500 - 1600	2.0
Shroud separation	266.55	280	1.8	360	3.5	330	3.0	1900	11.1
Agena engine first cutoff	488.07	[b]180	[b]6.8	160	1.2	(c)	.47	390	1.3
Agena engine second start	3262.30	100	.9	120	.45	51	.37	52	.9
Agena engine second cutoff	3267.44	75	1.3	145	.52	185	.47	800	1.2
Agena-Nimbus III separation	3517.16	160	2.6	130	5.3	73	5.0	520	(d)

[a]No detectable response.
[b]Questionable data.
[c]No predominant frequency.
[d]Channel saturated beyond band edge.



Telemetry channel	Measurement description	Measurement number	Frequency response	Range, g's	Accelerometer and vibrometer orientation
11	Longitudinal acceleration	PL 32 at station 226	0 to 110 Hz	±15	X-direction
12	Lateral acceleration	PL 36 at station 226	0 to 160 Hz	±5	Y-direction
13	Lateral acceleration	PL 34 at station 226	0 to 220 Hz	±5	Z-direction
17	Radial vibration	AA006 at station 222	20 to 1000 Hz	±15	Radial direction, quadrant II

Figure D-1. - Flight instrumentation, Nimbus III.

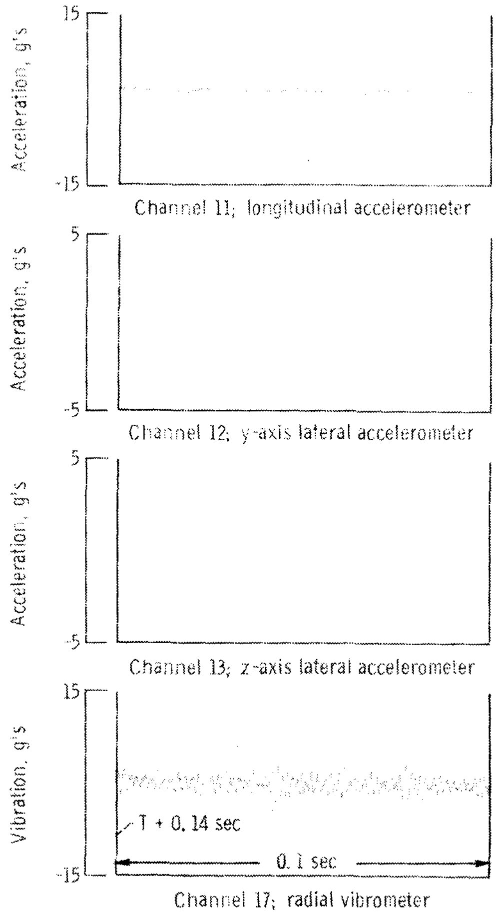


Figure D-2. - Dynamic data near time of lift-off, Numbus III.

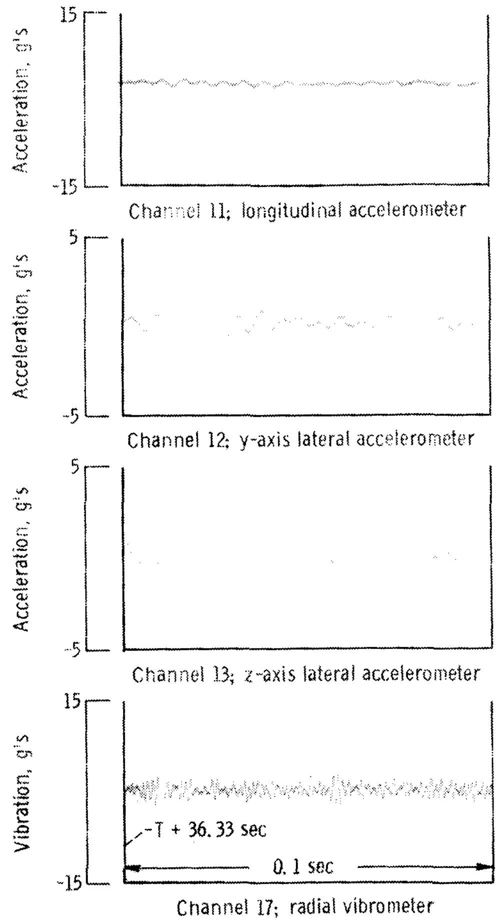


Figure D-3. - Dynamic data near time of solid motor burnout, Numbus III.

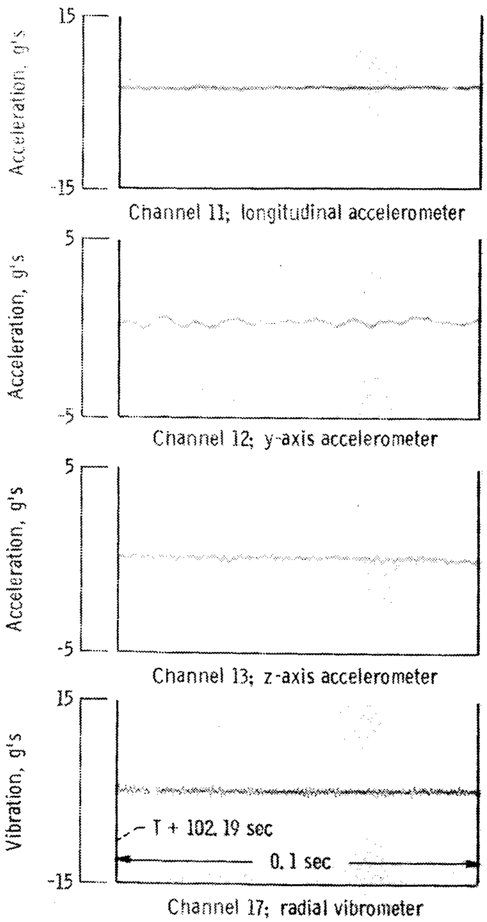


Figure D-4. - Dynamic data near time of solid motor jettison, Numbus III.

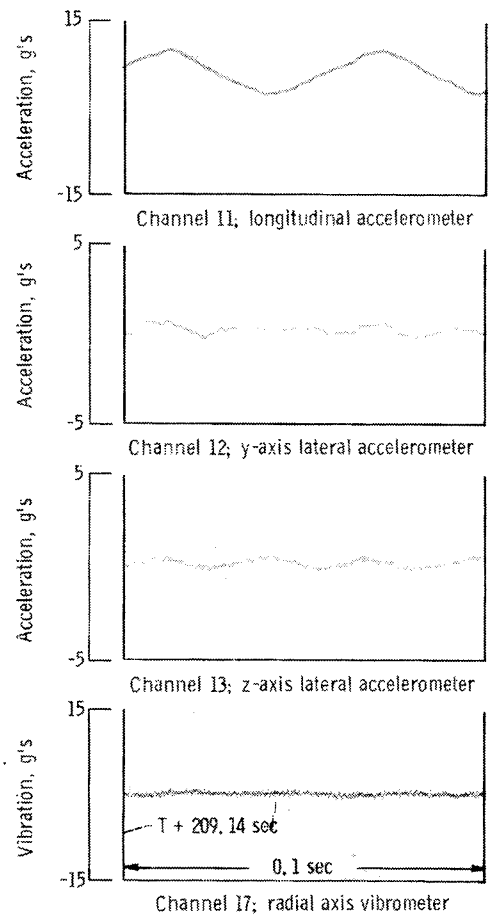


Figure D-5. - Dynamic data at peak longitudinal oscillation (POGO), Numbus III.

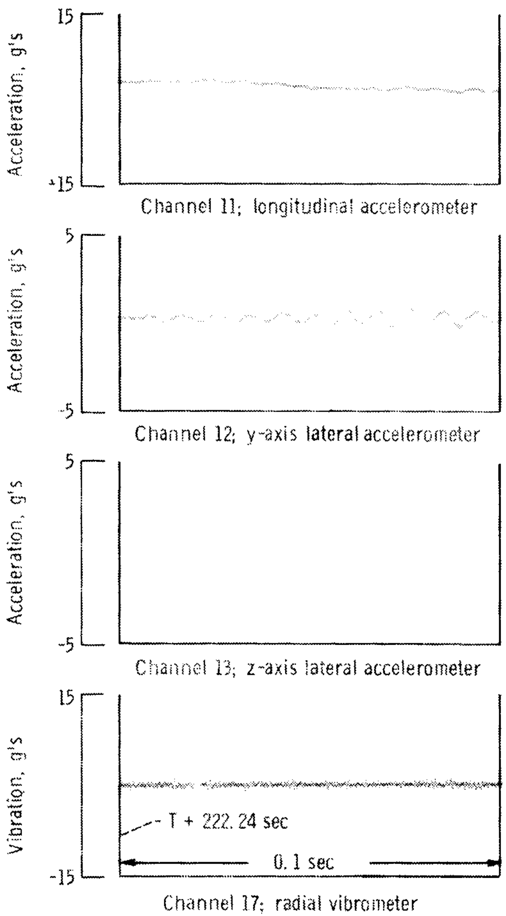


Figure D-6. - Dynamic data near time of Thorad main engine cutoff, Nimbus III.

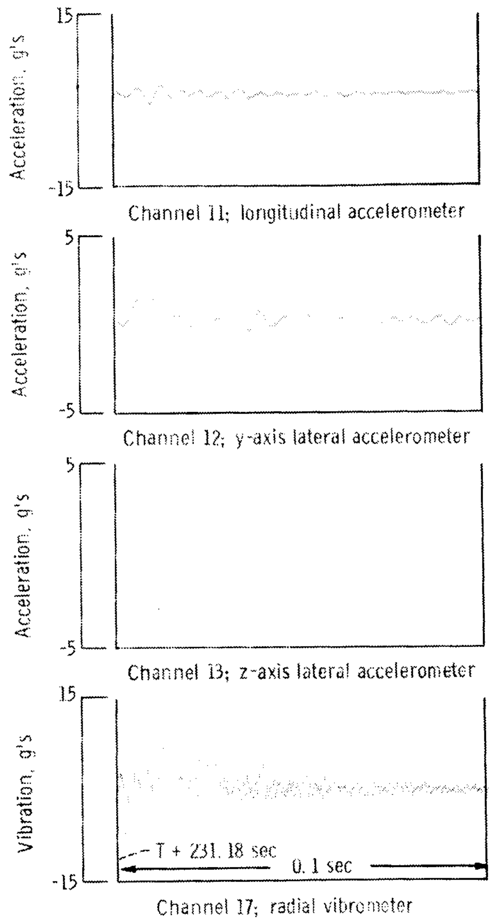


Figure D-7. - Dynamic data at time of Agena horizon sensor fairing jettison.

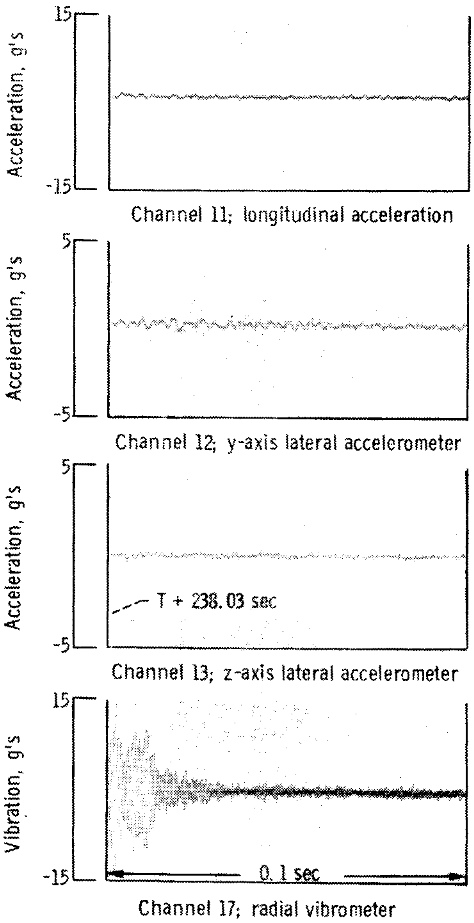


Figure D-8. - Dynamic data near time of Thorad-Agena separation, Nimbus III.

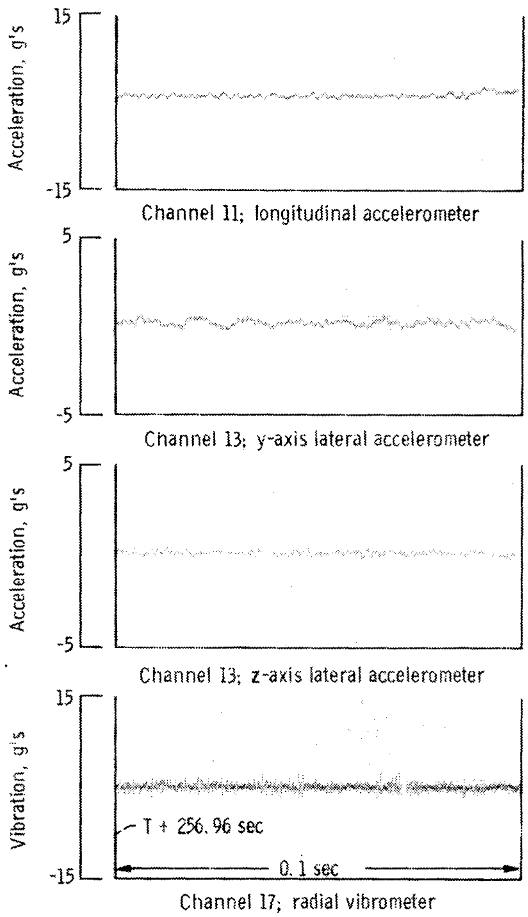


Figure D-9. - Dynamic data near Agena engine first start, Nimbus III.

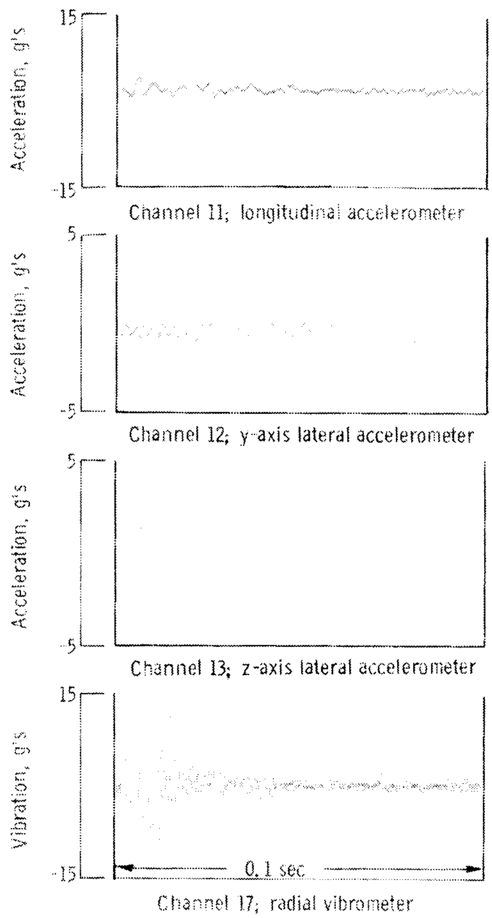


Figure D-10. - Dynamic data near time of shroud separation, Nimbus III.

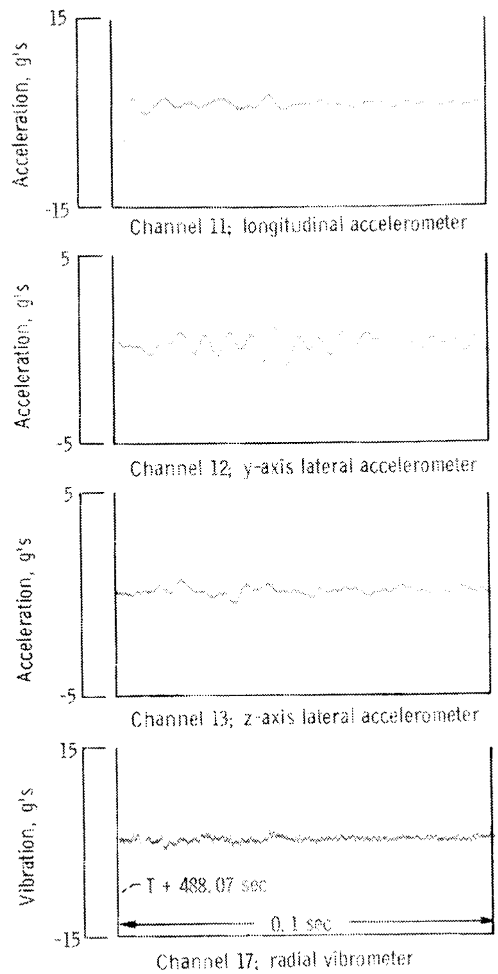


Figure D-11. - Dynamic data near time of Agena engine first cutoff, Nimbus III.

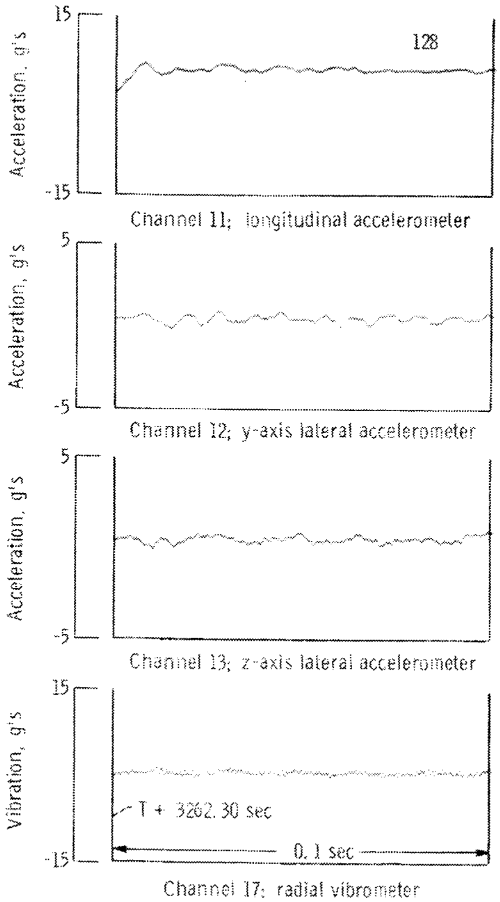


Figure D-12. - Dynamic data near time of Agena engine second start, Nimbus III.

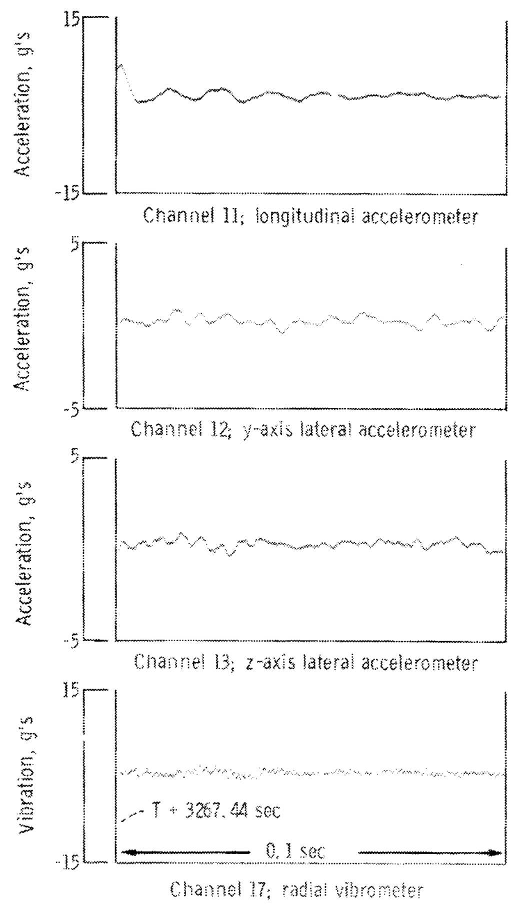


Figure D-13. - Dynamic data near time of Agena engine second cutoff, Nimbus III.

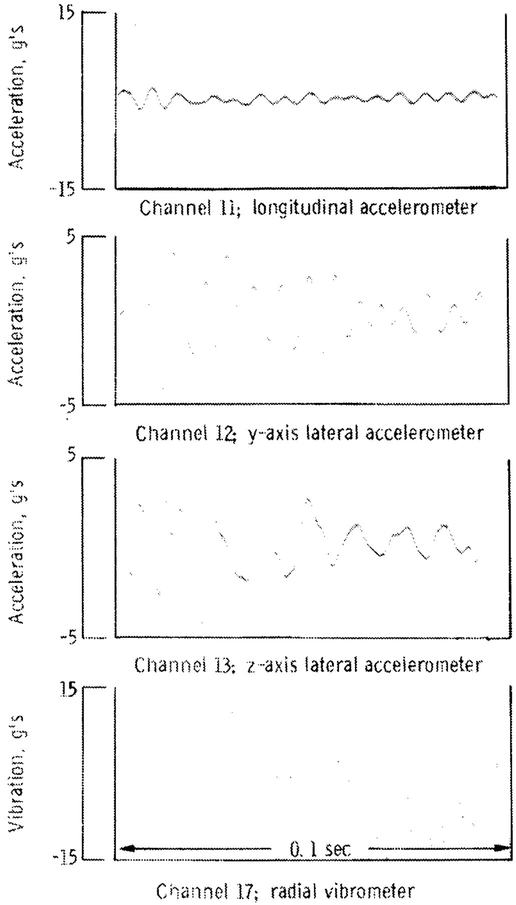


Figure D-14. - Dynamic data near time of Agena-Nimbus III separation, Nimbus III.

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16. Abstract

The Thorad-Agena launch vehicle successfully placed the Nimbus III onto a near-circular, near-polar orbit with a perigee altitude of 1087 km and an apogee altitude of 1143 km. The Nimbus III weighed 576 kg, a record for meteorological satellites and was launched from Vandenberg Air Force Base, California, at 2354:03 Pacific standard time on April 13, 1969. In addition to the Nimbus III, the Agena also successfully placed a secondary payload, an Engineers Geodetic Ranging Satellite-13 (EGRS-13), in orbit with orbital parameters similar to those of the Nimbus III. This report contains an evaluation of the performance of the Thorad-Agena systems in support of the Nimbus III mission.

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