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ATLAS-AGENA FLIGHT PERFORMANCE FOR THE APPLICATIONS TECHNOLOGY SATELLITE ATS-1 MISSION

by Staff of the Lewis Research Center

Lewis Research Center
Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • MAY 1969

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ABSTRACT

The Atlas-Agena launch vehicle injected the Applications Technology Satellite ATS-1 into an elliptical transfer orbit with an apogee altitude of about 19 750 nautical miles (36 577 km) and a perigee altitude of about 100 nautical miles (185 km). Sixteen hours after separation of the spacecraft from the Agena, the spacecraft motor was started and the spacecraft was placed onto a circular orbit at a near-constant altitude of about 19 750 nautical miles (36 577 km). This report discusses the flight performance of the Atlas-Agena launch vehicle from lift-off through the Agena vehicle final attitude maneuver.

CONTENTS

I. SUMMARY

The Atlas-Agena launch vehicle, with Applications Technology Satellite ATS-1, was successfully launched from the Eastern Test Range Launch Complex 12, on December 6, 1966, at 2112:00.876 eastern standard time. The vehicle was launched on the first attempt within 2 seconds after the scheduled launch time. The Atlas placed the Agena-spacecraft in the proper suborbital coast ellipse. After separation of the Agena-spacecraft from the Atlas, the Agena engine was started and the Agena-spacecraft was injected onto the desired approximate 100-nautical-mile (185-km) circular parking orbit. Restart of the Agena engine after an 11-minute coast period resulted in injection of the Agena-spacecraft onto the desired elliptical transfer orbit with an apogee altitude of about 19 750 nautical miles (36 577 km) and a perigee altitude of about 100 nautical miles (185.2 km). Approximately 16 hours after separation of the Applications Technology Satellite ATS-1 from the Agena, the spacecraft apogee motor was started, and the spacecraft was placed in the planned near-synchronous Earth orbit at an altitude of about 19 750 nautical miles (36 577 km).

The Atlas-Agena vehicle systems performed satisfactorily throughout the flight. This flight was the first to use the Standard Agena Clamshell shroud. The shroud provided satisfactory aerodynamic shielding for spacecraft during ascent through the atmosphere.

All aerospace ground equipment operated properly during the countdown and launch except that the 2-inch- (5.08-cm-) motion switch failed at lift-off. Consequently, the time of first motion was derived from the 8-inch- (20.32-cm-) motion switch.

This report contains an evaluation of Atlas and Agena systems in support of the Applications Technology Satellite ATS-1 mission.

II. INTRODUCTION

The Atlas-Agena vehicle was first developed as a two-stage launch vehicle for Earth orbiting payloads. Twelve Atlas-Agena vehicles (including ATS-1) were launched under the direction of the Lewis Research Center to boost lunar probes, planetary probes, and various Earth-orbiting spacecraft.

Applications Technology Satellite ATS-1, launched in December 1966, was the first of a series of three ATS probes to be boosted into orbit by the Atlas-Agena vehicle. The Atlas vehicle was used to boost the combined Agena-spacecraft into a suborbital coast ellipse. The Agena then performed two separate burns to place the ATS-1 spacecraft into the proper transfer orbit. After separation from the Agena, an apogee motor aboard the ATS-1 spacecraft was used to position the spacecraft on a near-synchronous Earth orbit with an inclination of about zero degrees. The ATS-1 is a spin-stabilized spacecraft designed to perform experiments in communications, cloud cover photography, and space environment. The ATS-1 spacecraft, including the apogee motor, weighed 1550 pounds (703.06 kg).

This report presents an evaluation of the Atlas and Agena vehicle systems for the purpose of showing how the performance of the launch vehicle supported the objectives of ATS-1. This report does not evaluate the performance of the spacecraft.

III. LAUNCH VEHICLE DESCRIPTION

The Atlas-Agena is a two-stage launch vehicle consisting of an Atlas first stage and an Agena second stage connected by a booster adapter. The Atlas is 10 feet (3.05 m) in diameter except for the forward section of the tank which is conical and tapers to a diameter of about 6 feet (1.83 m). The booster adapter provides the transition from this diameter to the Agena diameter of 5 feet (1.5 m). The composite vehicle including the spacecraft shroud is 109 feet (33.2 m) in length. The vehicle weight at lift-off is approximately 279 000 pounds (127 000 kg). Figure III-1 shows the Atlas-Agena launch vehicle lifting off with ATS-1.

The first-stage Atlas SLV-3 (fig. III-2) is 70 feet (21.34 m) long and was propelled by a standard Rocketdyne MA-5 propulsion system consisting of a booster engine that has two thrust chambers with a total sea-level thrust of 330×10^3 pounds (1467.9×10^3 N); a sustainer engine with a sea-level thrust of 57×10^3 pounds (253.55×10^3 N); and two vernier engines, each with a sea-level thrust of 6.69×10^2 pounds (2.98×10^3 N). All engines used liquid oxygen and high grade kerosene propellants. The booster, sustainer, and vernier engines are ignited prior to lift-off for the booster phase of flight. The booster thrust chambers are gimbaled for pitch, yaw, and roll control during the booster phase of flight. This phase was terminated when the vehicle acceleration equaled about 6 g's and the booster engine was shut down. The booster engine section was jettisoned after booster engine shutdown. The sustainer and vernier engines continued to burn for the Atlas sustainer phase of flight. During this phase, the sustainer engine was gimbaled for pitch and yaw control, and vernier engines were gimbaled for roll control only. The sustainer engine burned until the vehicle achieved the desired suborbital coast ellipse. After sustainer engine shutdown, the vernier engines continued to burn for a short period of time. During this vernier "solo" phase, the vernier engines were gimbaled to provide vehicle attitude control. After vernier engine shutdown, the Atlas was severed from the Agena by the firing of an explosive severance system (Primacord) located on the booster adapter. The firing of a retrorocket system then separated the Atlas and the booster adapter from the Agena.

The second-stage Agena and the spacecraft shroud are shown in figure III-3. This stage including the shroud is about 34 feet (10.36 m) in length and was powered by a Bell Aerosystems Company Model 8096 engine which generates a thrust of 16×10^3 pounds (71.17×10^3 N). The Agena engine used unsymmetrical dimethylhydrazine (UDMH) and inhibited red fuming nitric acid (IRFNA) as propellants. The Agena propulsion system is designed so that the engine can be started a second time. The Agena engine was gimbaled for pitch and yaw control during powered flight, and roll control was provided by a cold gas (nitrogen) pneumatic system. During periods of nonpowered flight, pitch, yaw, and roll control were provided by the cold gas system. A fiberglass laminate shroud was used to provide an aerodynamic shield for the ATS-1 spacecraft during ascent. This shroud was jettisoned during the Agena first firing phase of flight. The Applications Technology Satellite is shown in figure III-4.

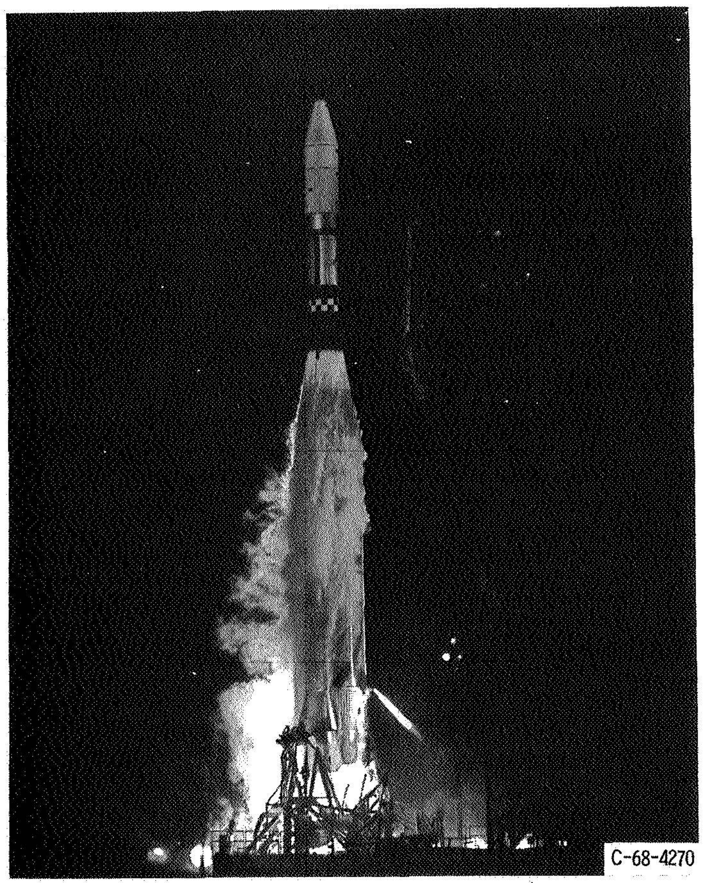


Figure III-1. - Atlas-Agena lifting off with ATS-1.

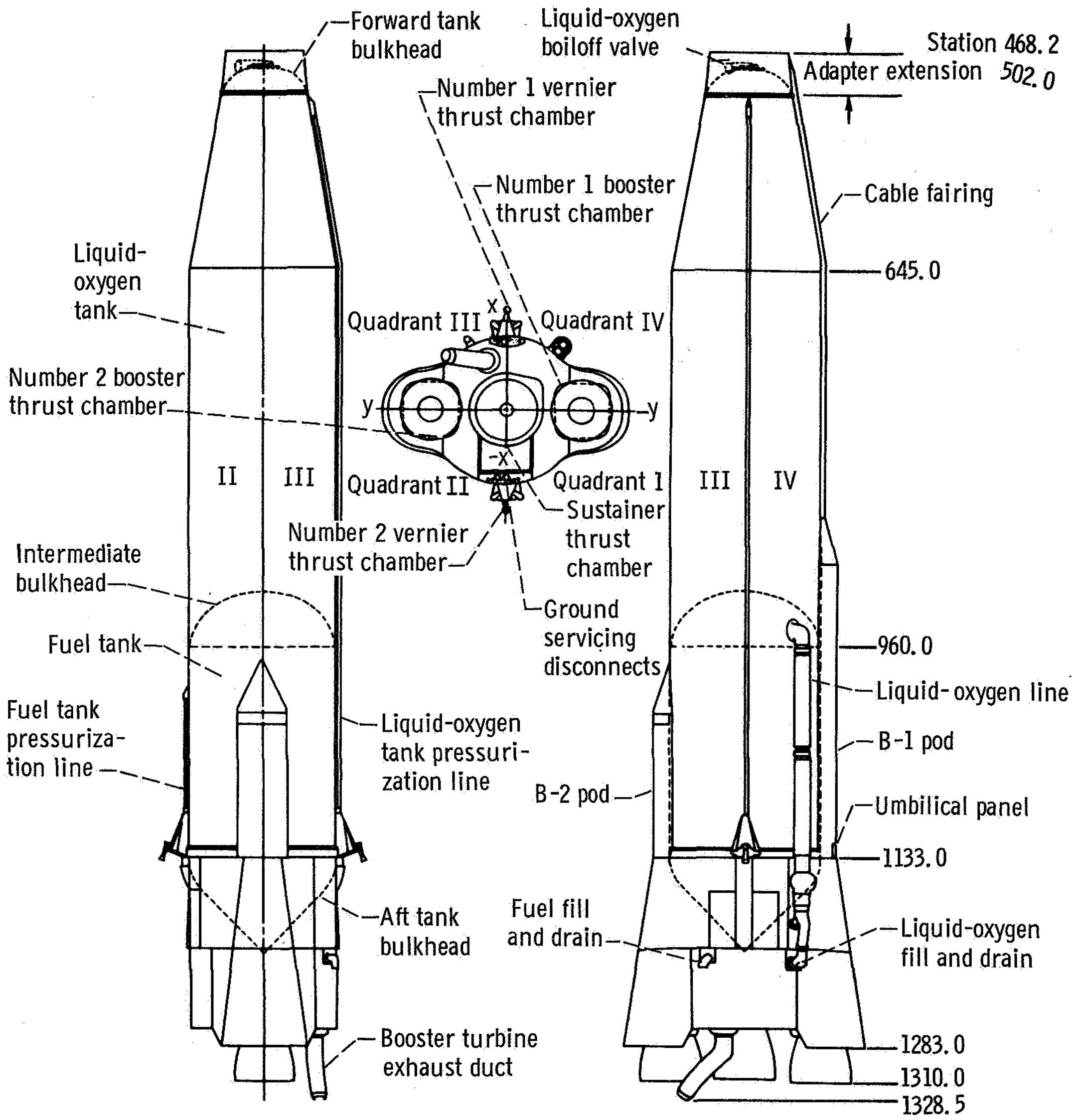


Figure III-2. - Atlas SLV-3 configuration, ATS-1.

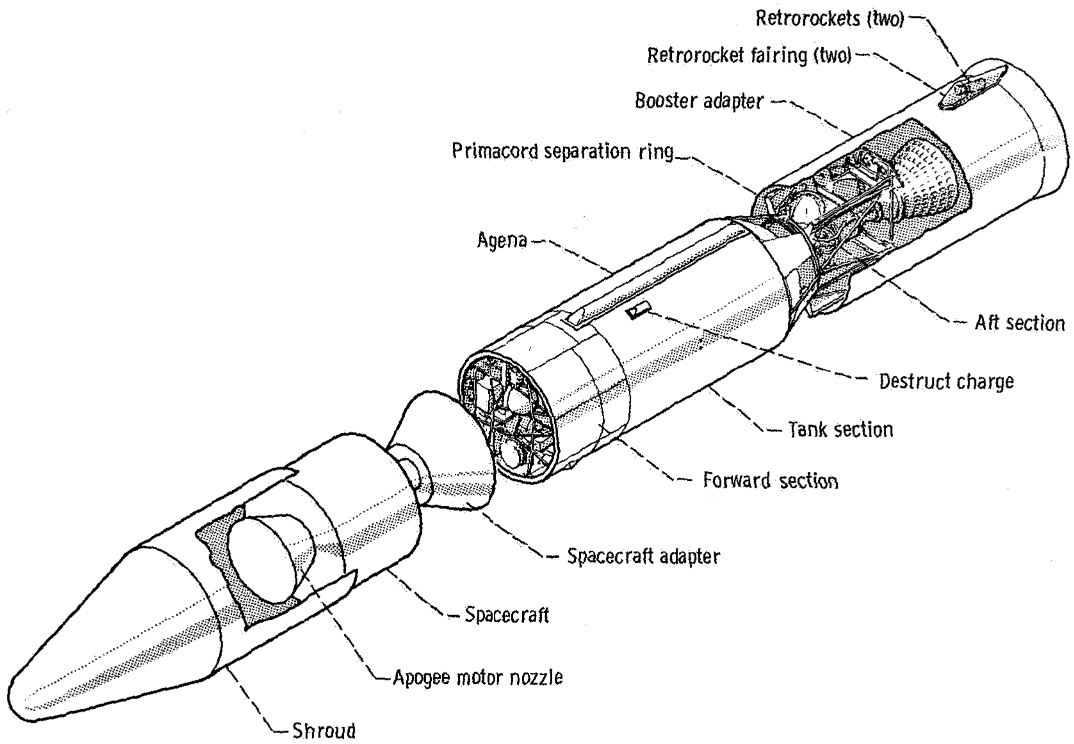


Figure III-3. - Agena-shroud-spacecraft, ATS-1.

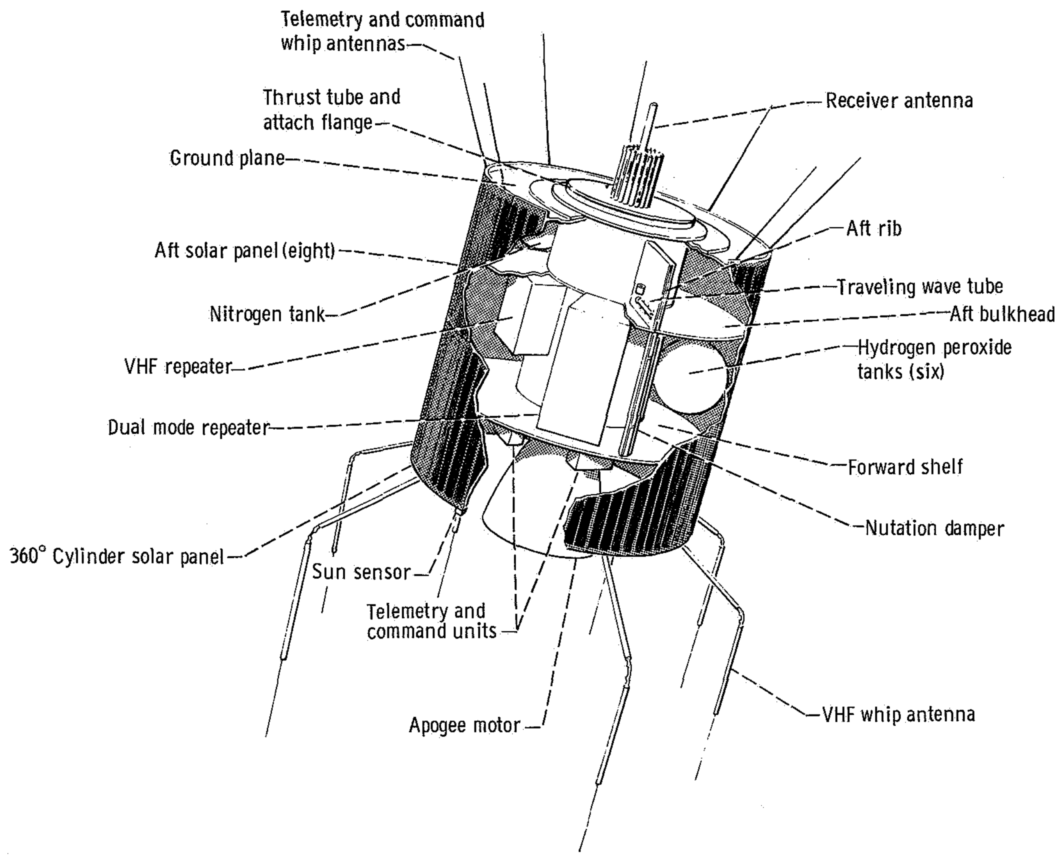


Figure III-4. - Applications technology satellite, ATS-1.

IV. TRAJECTORY AND PERFORMANCE

ATS-1 was launched from Eastern Test Range Complex 12 on December 6, 1966, at 2112:00.876 eastern standard time. Lift-off occurred 2 seconds after the time scheduled for this event. The spacecraft was injected into the desired orbital transfer ellipse. A comparison of nominal[1] and actual times for significant flight events is given in table IV-I. A detailed sequence of flight events is provided as appendix A.

TRAJECTORY PLAN

The Atlas boosts the Agena-spacecraft onto a prescribed suborbital coast ellipse. The Atlas flight consists of three powered phases: a booster engine phase, a sustainer engine phase, and a vernier engine phase. Atlas-Agena separation occurs after the vernier phase is completed. Following Atlas-Agena separation, the Agena engine is ignited to inject the Agena-spacecraft onto a 100-nautical-mile (185-km) circular parking orbit. The Agena-spacecraft coasts in the parking orbit to the first descending node where the Agena engine is restarted. The Agena engine second-burn phase places the Agena-spacecraft onto a transfer ellipse with an apogee (approximately synchronous) of 19 798 nautical miles (36 666 km). Prior to spacecraft separation, the Agena pitches up 9.36° and yaws left 57° to position the spacecraft motor thrust axis in the attitude required for the spacecraft motor burn. Three seconds after spacecraft separation, the Agena performes a programmed 237° yaw right. The purpose of this preprogrammed yaw maneuver is to place the Agena in the proper attitude for a retromaneuver; however, the retromaneuver was not planned for this mission. The preceding sequence is shown in figure IV-1.

TRAJECTORY RESULTS

Winds Aloft

Winds aloft at the time of launch were predominately from the West, attaining a

[1]The word "nominal," as used in this report, denotes a design, programmed, or expected value. Three-sigma (3σ) dispersions about nominal define acceptable limits for flight or hardware performance.

maximum speed of 110 feet per second (33.5 m/sec) at an altitude of 49 450 feet (15 072 m) as shown in figure IV-2. These winds were light and had only a minor effect on the vehicle flight path. Abrupt changes in wind velocity did produce strong wind shears at altitudes from 37 000 feet (11 278 m) to 46 000 feet (14 021 m).

As calculated from the T - 0 Jimsphere balloon data, the peak bending response of the vehicle was 49.7 percent of the critical value. This bending occurred at Agena station 492 when the vehicle was at an altitude of 11 981 feet (3651.8 m). (See fig. VI-1 for vehicle station location.) From the same data, the peak booster engine gimbal angle was calculated to be 41.0 percent of that available in the pitch plane and to occur at an altitude of 12 259 feet (3736.5 m).

Atlas Booster Phase

A comparison of the actual and nominal trajectories in the vertical plane is shown in figure IV-3. The actual trajectory was higher than the nominal trajectory because the total pitchover command by the flight control system was 1.4° less at T + 100 seconds than that required. At booster engine cutoff, the actual trajectory was 3650 feet (1113 m) higher in altitude and 4250 feet (1295 m) less in ground range than the nominal.

A projection of the nominal and actual trajectories in the horizontal plane is shown in figure IV-4. The actual trajectory is to the left of the nominal trajectory throughout the booster phase of the flight. As determined from reconstruction, this deviation was caused primarily by crossrange winds and an actual roll angle 0.22° greater than the programmed roll angle. The booster was programmed to begin a 1.28° roll to the right (clockwise looking forward) at T + 2 seconds to provide a launch azimuth of 103.9° from a launch pad azimuth of 105.18°. The actual booster roll maneuver was 1.50° right. The actual flight azimuth resulting from the roll maneuver was 103.68°, or 0.22° left of the programmed azimuth. Thrust misalinement and yaw gyro drift were small and had little effect on the trajectory.

Figure IV-5 shows the nominal and actual velocity histories measured with respect to the rotating Earth. The difference between the actual and nominal velocities during the booster phase did not exceed 10 feet per second (3.05 m/sec).

Radio guidance was enabled at T + 80 seconds. Booster steering was initiated at T + 103.2 seconds. At this time, a pitch-down command of about 0.2° was transmitted to correct for the higher than nominal trajectory. (Radio guidance yaw steering is not used during the booster phase.) Transmission of the radio guidance booster engine cutoff discrete occurred at T + 128.8 seconds. Booster engine cutoff occurred at T + 129.1 seconds at a vehicle longitudinal acceleration of 6.07 g's. This value was exactly the nominal acceleration cutoff level. Tracking data indicated that, at booster engine cutoff, the actual velocity was about 10 feet per second (3.05 m/sec) greater than the nominal. This deviation in velocity resulted from tailwinds and a slightly better-than-nominal engine performance. The Atlas booster engine section was jettisoned at T + 131.8 seconds.

Atlas Sustainer Phase

The actual trajectory remained higher than, and to the left of, the nominal trajectory from booster engine cutoff to Atlas sustainer engine cutoff, as illustrated in figures IV-3 and IV-4. The altitude of the vehicle at sustainer engine cutoff was 14 000 feet (4267 m) higher than the nominal sustainer engine cutoff altitude. The crossrange deviation at sustainer engine cutoff was about 12 750 feet (3886 m) left of the nominal trajectory.

The sustainer engine cutoff discrete was transmitted at T + 292.9 seconds by radio guidance command, and cutoff occurred at T + 293.0 seconds.

The velocity, relative to the rotating Earth, was 30 feet per second (9.14 m/sec) less than nominal at sustainer engine cutoff as the result of increased gravitational losses associated with a steeper-than-nominal ascent following booster staging. However, this velocity decrement was consistent with the higher-than-nominal spatial position. The energy (velocity and position) of the vehicle at sustainer engine cutoff resulted in a near-nominal coast ellipse. The actual and nominal Atlas suborbital coast ellipse parameters are given in table IV-II.

The radio guidance Start Agena Timer discrete started the primary Agena timer at T + 288.2 seconds, or 7.6 seconds before the nominal time for this event. This earlier-than-expected start occurred because the ground guidance system sensed that the Agena would be injected onto the coast ellipse at an altitude higher than nominal and, consequently, the Agena would reach apogee earlier than predicted. The 7.6-second adjustment was made so that the initiation of the Agena engine first burn, an Agena timer event, would occur at the proper altitude.

Atlas Vernier Phase

Vernier engine burn duration after sustainer engine cutoff was 19.9 seconds, 0.2 second longer than nominal. During the vernier phase, pitch-up and yaw-left steering commands were issued by radio guidance in order to make final adjustments to the velocity vector to meet the energy requirements of the final coast ellipse. These commands displaced the vehicle 0.96° up in pitch and 1.26° left in yaw. The Atlas insertion velocities at Atlas vernier engine cutoff are given in table IV-III.

Agena Engine First-Burn Phase

Atlas-Agena separation occurred by radio guidance command at T + 315.1 seconds, 0.1 second later than nominal. After Atlas-Agena separation, the Agena pitch-down maneuver placed the Agena-spacecraft in the proper attitude for Agena engine first burn.

Because the start Agena timer discrete was transmitted 7.6 seconds earlier than the nominal time, the actual times for all Agena timer events listed in table IV-I are earlier by this amount within the ±0.2-second timer tolerance.

Agena engine first start sequence was initiated at T + 359.2 seconds, and shroud separation occurred at T + 369.3 seconds. Agena engine first-burn duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 161.3 seconds, 1.2 seconds longer than nominal. The preset velocity increment was gained as evidenced by the velocity meter cutoff. At the end of the Agena engine first burn, the Agena-spacecraft was in a near-circular parking orbit. The actual parking orbit parameters are listed in table IV-IV.

Agena Engine Second-Burn Phase

The Agena timer was set so that the Agena-spacecraft would coast to the proper position for Agena engine second start. After a coast of 648.6 seconds to the first equatorial crossing, the Agena engine second burn sequence was initiated at T + 1170.3 seconds. The Agena second-burn duration (measured from 90 percent chamber pressure to velocity meter cutoff) of 77.3 seconds was 1.7 seconds shorter than nominal burn durations. The preset velocity increment was gained, as evidenced by the velocity meter cutoff. At the end of the Agena engine second burn, the Agena-spacecraft was in an orbital transfer ellipse with an apogee at approximately synchronous altitude. Actual transfer orbit parameters are listed in table IV-V.

Post-Second-Burn Phase

After the Agena engine shutdown, the Agena performed the programmed pitch-up maneuver (nominal 9.36°), followed by the programmed yaw-left maneuver (nominal 57°). These maneuvers were performed to orient the spacecraft motor thrust axis in the attitude required for spacecraft motor burn. The Agena was separated from the spacecraft at T + 1387.3 seconds. Three seconds later, the Agena performed a programmed 237° yaw right. The purpose of this preprogrammed yaw maneuver is to place the Agena in the proper attitude for a retromaneuver; however, the retromaneuver was not planned for this mission, and consequently, the retrorockets had been removed prior to launch.

Spacecraft Apogee Motor Burn

When the spacecraft arrived at the transfer-orbit apogee for the second time, nearly 16 hours after being separated from the Agena, the apogee motor aboard the spacecraft was fired. As planned, the apogee motor burn reduced the spacecraft orbit inclination to about 0° and placed the spacecraft onto an almost circular orbit slightly above synchronous altitude.

TABLE IV-I. - SIGNIFICANT FLIGHT EVENTS, ATS-1

Event description	Nominal time, sec	Actual time, sec
Lift-off (2112:00.876 EST)	0	0
Booster engine cutoff	129.1	129.1
Booster engine jettison	132.0	131.8
Sustainer engine cutoff	293.1	293.0
Start Agena timer	295.8	288.2
Vernier engine cutoff	312.8	312.9
Atlas-Agena separation	315.0	315.1
Fire first-burn ignition squibs	366.8	359.2
Agena steady-state thrust (90 percent chamber pressure)	368.0	360.4
Fire shroud squibs	376.8	369.3
Agena engine first cutoff (velocity meter)	528.1	521.7
Fire second-burn ignition squibs	1177.8	1170.3
Agena steady-state thrust (90 percent chamber pressure)	1179.0	1171.4
Agena engine second cutoff (velocity meter)	1258.0	1248.7
Start 9.36° pitch-up maneuver	1334.8	1327.3
Stop 9.36° pitch-up maneuver	1354.8	1347.3
Start 57° yaw-left maneuver	1364.8	1357.2
Stop 57° yaw-left maneuver	1383.8	1376.4
Fire spacecraft separation squibs	1394.8	1387.4
Start 237° yaw-right maneuver	1397.8	1390.3
Stop 237° yaw-right maneuver	1476.8	1469.3

TABLE IV-II. - ATLAS SUBORBITAL COAST

ELLIPSE PARAMETERS, ATS-1

Parameter	Units	Nominal	Actual[a]
Semimajor axis	n mi km	2386.77 4420.3	2386.72 4420.2
Semiminor axis	n mi km	2088.71 3868.3	2088.66 3868.2
Radius vector magnitude at apogee	n mi km	3541.74 6559.3	3541.79 6559.4
Inertial velocity at apogee	ft/sec m/sec	18 374.7 5600.6	18 372.8 5600.0
Inclination	deg	30.96	30.97

[a]Calculated from guidance system data.

TABLE IV-III. - ATLAS INSERTION VELOCITIES AT

VERNIER ENGINE CUTOFF, ATS-1

Parameter	Units	In-flight objective[a]	Actual[b]
Velocity magnitude	ft/sec m/sec	18 448.6 5623.13	18 448.0 5622.95
Altitude rate	ft/sec m/sec	1439.5 438.76	1437.7 438.21
Lateral velocity	ft/sec m/sec	0.0	4.5 (left) 1.37 (left)

[a]Determined by guidance system during flight as velocity required to achieve nominal coast ellipse from actual position in space.
[b]Calculated from guidance system data.

TABLE IV-IV. - AGENA PARKING

ORBIT PARAMETERS, ATS-1

Parameter	Units	Actual
Apogee	n mi	105.0
	km	194.5
Perigee	n mi	100.0
	km	185.2
Period	min	88.2
Inclination	deg	31.08

TABLE IV-V. - FINAL AGENA-SPACECRAFT

TRANSFER ORBIT PARAMETERS, ATS-1

Parameter	Units	Actual
Apogee altitude	n mi	19 753.3
	km	36 583.1
Perigee altitude	n mi	99.2
	km	183.7
Inclination	deg	31.29
Eccentricity	---	0.7350
Period	min	644.4
Semimajor axis	n mi	13 370
	km	24 761.2

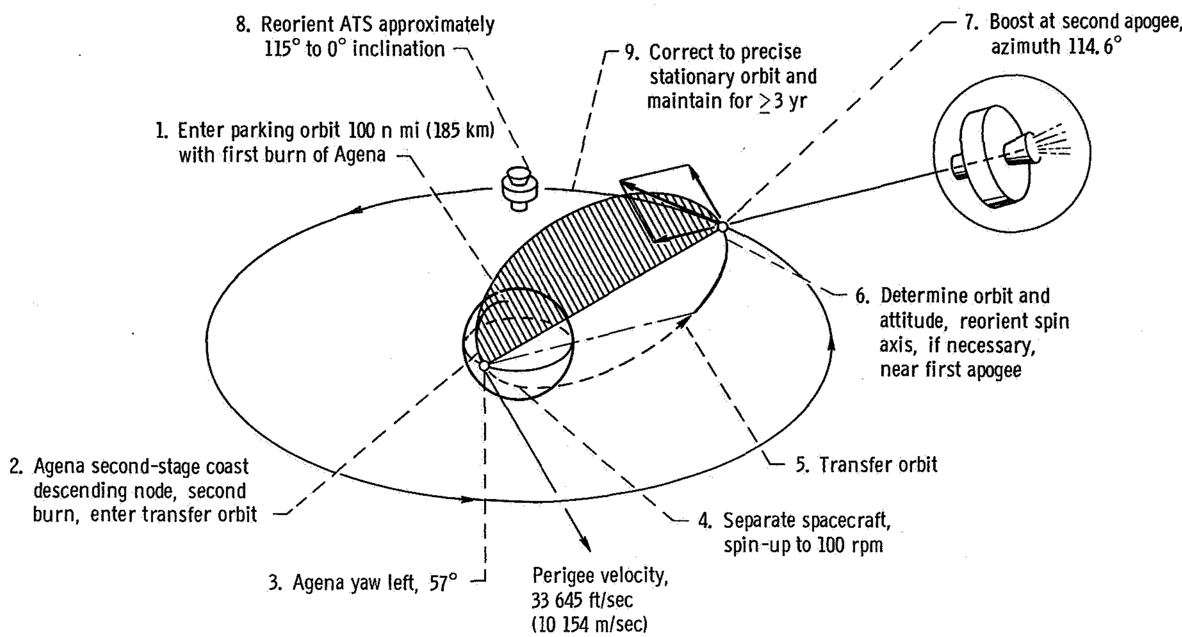


Figure IV-1. - Trajectory profile, ATS-1.

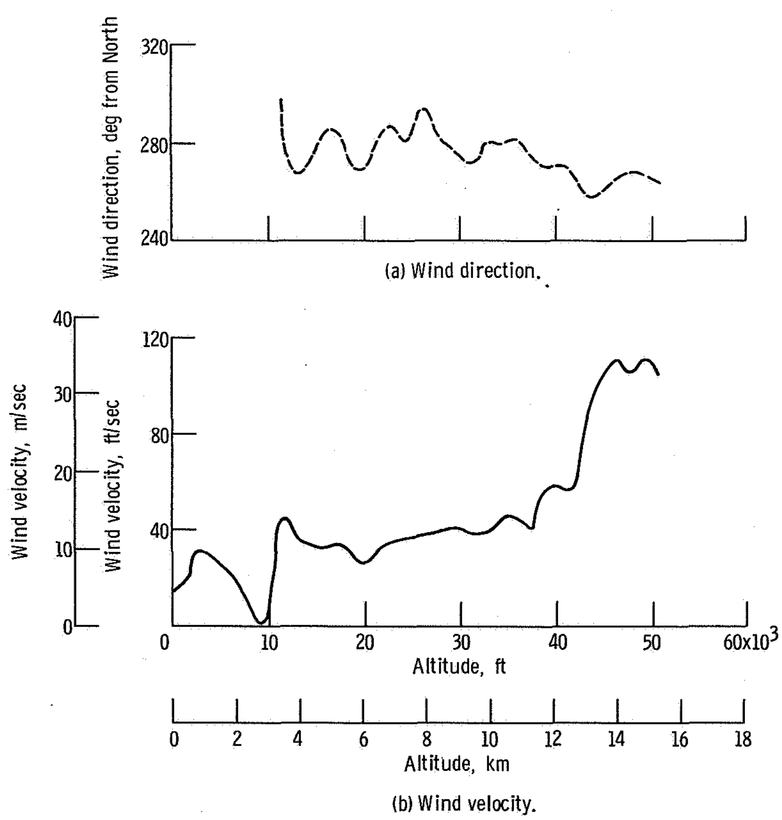


Figure IV-2. - Winds aloft at T - 0 balloon release, ATS-1.

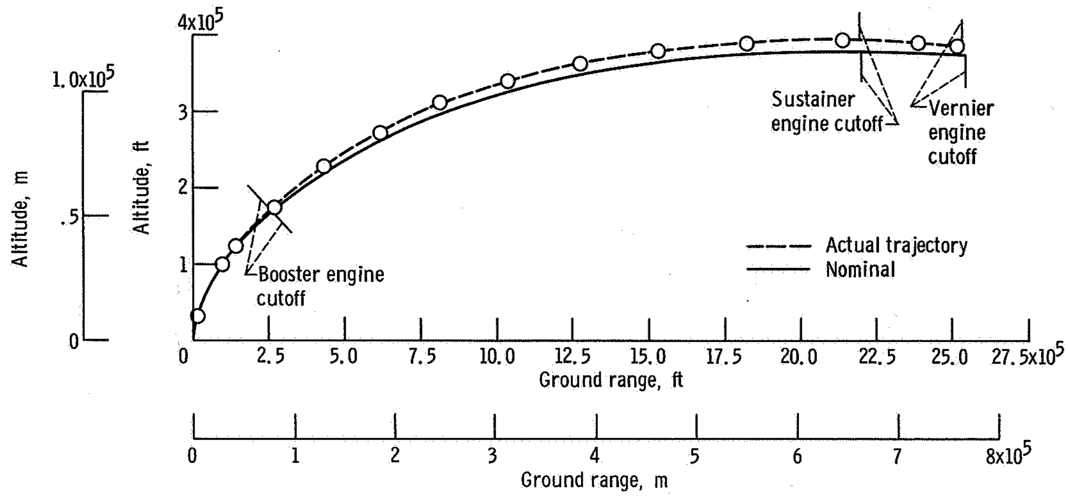


Figure IV-3. - Altitude and ground range displacement, ATS-1.

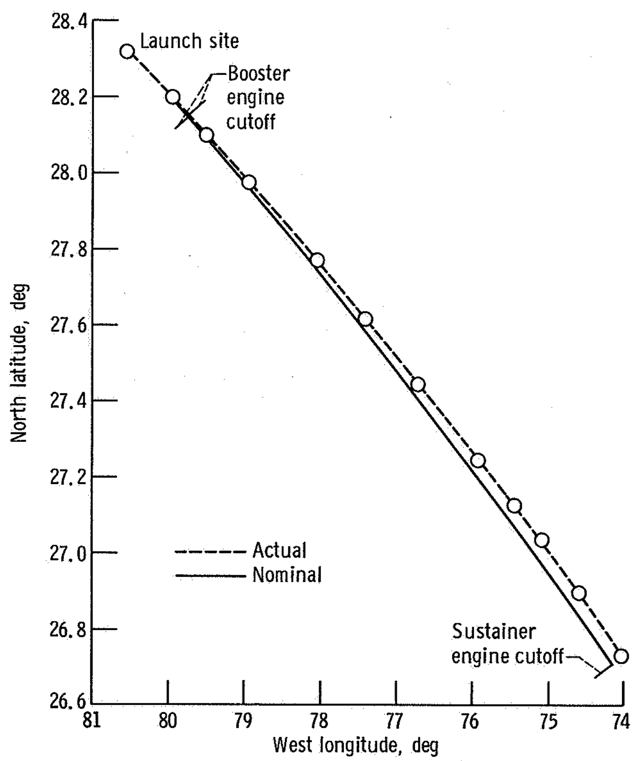


Figure IV-4. - Trajectory projection in horizontal plane, ATS-1.

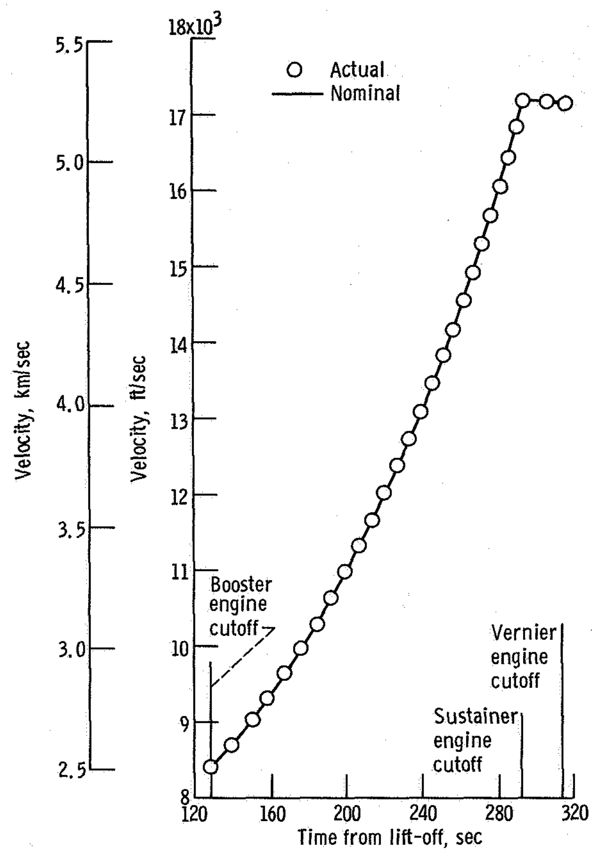


Figure IV-5. - Vehicle velocity measured with respect to rotating Earth, ATS-1.

V. ATLAS VEHICLE SYSTEM PERFORMANCE

PROPULSION SYSTEM

Description

The Atlas engine system consists of a booster, a sustainer, two vernier engines, an engine tank system (pressurization and auxiliary propellant), and an electrical control system (see fig. V-1). The engines are of the single-burn type. During engine start, electrically fired pyrotechnic igniters are used to ignite the gas generator propellants for driving the turbopumps, and hypergolic igniters are used to ignite the propellants in the thrust chambers of the booster, sustainer, and vernier engines. The propellants are liquid oxygen (oxidizer) and RP-1 (kerosene) fuel.

The booster engine, rated at 330×10^3 pounds (1467×10^3 N) thrust at sea level, is made up of two gimbaled thrust chambers, propellant valves, two oxidizer and two fuel turbopumps driven by one gas generator, a lubricating oil system, and a heat exchanger. The sustainer engine, rated at 57×10^3 pounds (253.5×10^3 N) thrust at sea level, consists of a thrust chamber, propellant valves, one oxidizer and one fuel turbopump driven by a gas generator, and a lubricating oil system. The entire sustainer engine system is gimbal mounted. Each vernier engine is rated at 669 pounds (2.98×10^3 N) thrust at sea level when supplied with propellants from the sustainer turbopumps during sustainer engine operation. In the "solo" phase of flight, each vernier engine is rated at 525 pounds (2.335×10^3 N) thrust at sea level. For this phase, the engines are supplied with propellants from the engine tank system because the sustainer turbopumps do not operate after sustainer engine cutoff.

The engine tank system is composed of two small propellant tanks and a pressurization system. This system supplies propellants for starting the engines and also for vernier "solo" operation after sustainer engine cutoff.

Performance

The engine start sequence for all engines was satisfactory. Valve opening times and starting sequence events were within tolerances. This was verified by landline-graphic recorders and by telemetry.

The flight performance of the booster engine was evaluated by comparing measured values of booster thrust chamber pressures, turbopump speeds, and gas generator chamber pressure with expected values. Favorable agreement verified that booster engine performance was satisfactory.

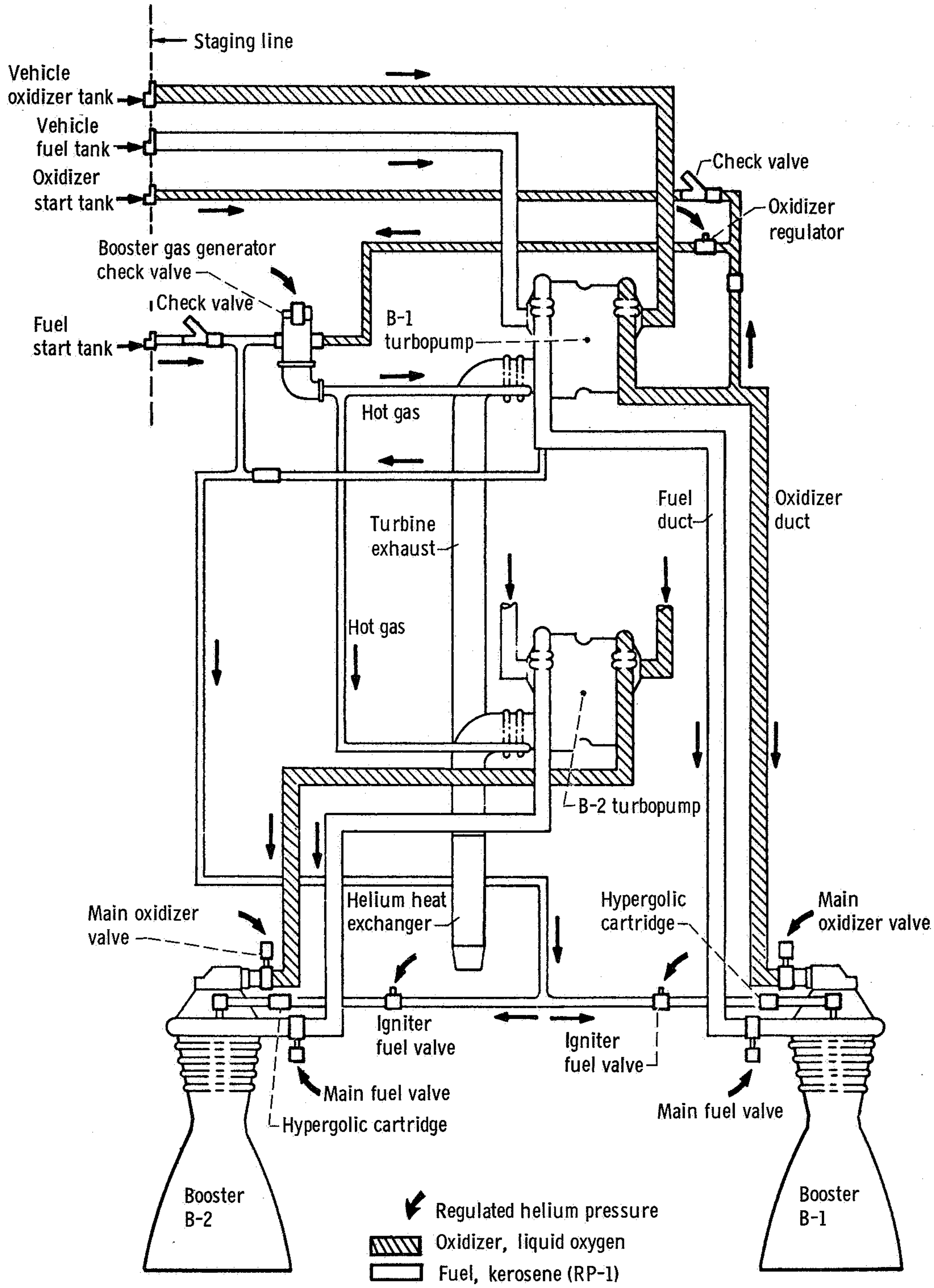
The flight performance of the sustainer engine was satisfactory. Measured values of thrust chamber pressure, turbopump speed, and gas generator discharge pressure compared favorably with expected values. The propellant utilization valve responded satisfactorily to signals generated at each Acoustica sensor station. The head-suppression valve properly responded to changes in propellant utilization valve position and vehicle acceleration. (See section PROPELLANT UTILIZATION SYSTEM for details.)

Vernier engine operation throughout flight was satisfactory. Evaluation of the flight performance was made by comparing expected and measured values of vernier thrust chamber pressures.

All engine cutoff signals were issued by guidance system commands and were properly executed. Transients at engine shutdown appeared normal. Atlas propulsion system performance data are presented in table V-1.

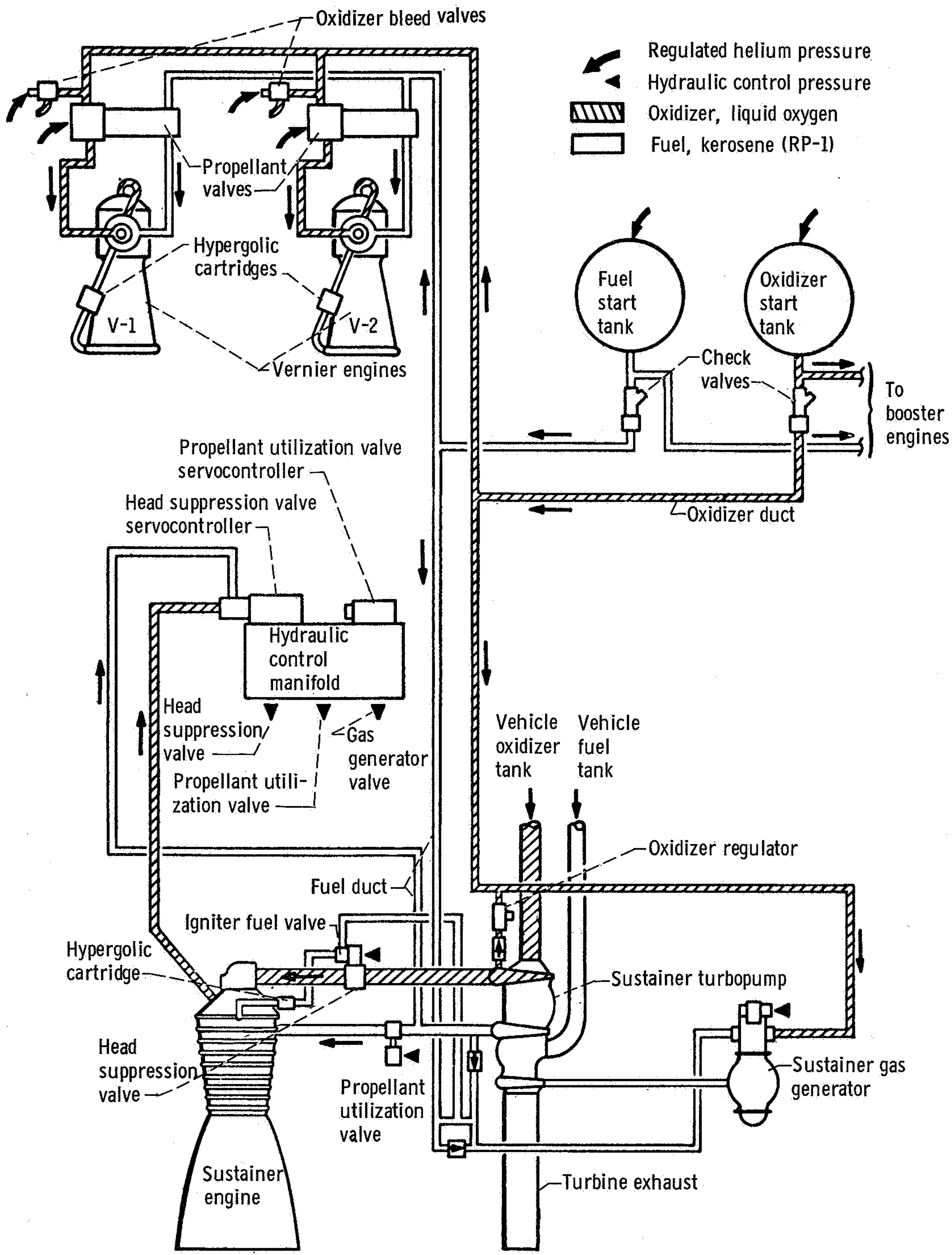
TABLE V-I. - ATLAS PROPULSION SYSTEM PERFORMANCE, ATS-1

Performance parameters	Units	Expected operating range	Flight values at -			
			T + 10 sec	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
B-1 thrust chamber pressure (absolute)	psi N/cm^2	560 to 595 386 to 410	576 397	576 397	----- -----	----- -----
B-2 thrust chamber pressure (absolute)	psi N/cm^2	560 to 595 386 to 410	580 400	584 403	----- -----	----- -----
Booster gas generator chamber pressure (absolute)	psi N/cm^2	510 to 555 351 to 382	528 364	528 364	----- -----	----- -----
B-1 pump speed	rpm	6225 to 6405	6306	6306	-----	-----
B-2 pump speed	rpm	6165 to 6345	6250	6280	-----	-----
Sustainer thrust chamber pressure (absolute)	psi N/cm^2	680 to 715 469 to 493	710 490	700 483	700 483	----- -----
Sustainer gas generator discharge pressure (absolute)	psi N/cm^2	590 to 686 407 to 473	664 458	656 452	656 452	----- -----
Sustainer pump speed	rpm	10 025 to 10 445	10 232	10 133	10 292	-----
V-1 thrust chamber pressure (absolute) pump supplied	psi N/cm^2	250 to 265 172 to 183	256 176	264 182	264 182	----- -----
V-1 thrust chamber pressure (absolute) tank supplied	psi N/cm^2	210 to 225 145 to 155	----- -----	----- -----	----- -----	220 152
V-2 thrust chamber pressure (absolute) pump supplied	psi N/cm^2	250 to 265 172 to 183	256 176	260 179	264 182	----- -----
V-2 thrust chamber pressure (absolute) tank supplied	psi N/cm^2	210 to 225 145 to 155	----- -----	----- -----	----- -----	220 152
Duration of booster engine burn	sec	129.0	-----	129.1	-----	-----
Duration of sustainer engine burn	sec	293.1	-----	-----	293.2	-----
Duration of vernier engine burn	sec	312.8	-----	-----	-----	312.9



(a) Booster.

Figure V-1. - Atlas propulsion system, ATS-1.



(b) Sustainer and vernier.

Figure V-1. - Concluded.

HYDRAULIC SYSTEM

Description

The Atlas hydraulic system consists of two independent systems: the booster system and the sustainer-vernier system (see fig. V-2). The booster system variable displacement pump and accumulator furnish the hydraulic pressure required for booster engine gimbaling. The sustainer-vernier system variable displacement pump and three accumulators furnish the hydraulic pressure required for sustainer and vernier engine gimbaling. The sustainer-vernier system also furnishes pressure for operating the propellant utilization valve, the head-suppression valve, and the gas generator blade valve. During vernier "solo" operation, after sustainer engine cutoff, the vernier engines are provided with hydraulic pressure from two accumulators previously pressurized during sustainer engine operation.

Two significant modifications to the Atlas hydraulic system were incorporated for this flight. The booster and sustainer accumulators were replaced by accumulators with a smaller volume, lower precharge pressure, and significantly different tubing arrangement. Improved hydraulic servoactuators were installed to correct servovalve "null shift" problems and actuator leakage problems. These modifications were incorporated to improve the performance and reliability of the system.

Performance

The hydraulic system performance was satisfactory. Steady-state pressures were maintained at nominal levels except for the usual transients which occur at engine start and booster engine cutoff. The values of steady-state hydraulic pressures monitored during flight are presented in table V-II.

TABLE V-II. - ATLAS HYDRAULIC SYSTEM FLIGHT DATA, ATS-1

Measurement pressure	Units	Lift-off	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Sustainer-vernier pressure (absolute)	psi	3080	3080	3080	1365
	N/cm^2	2124	2124	2124	941
Sustainer pump discharge pressure (absolute)	psi	3080	3080	3080	----
	N/cm^2	2124	2124	2124	----
Booster pump discharge pressure (absolute)	psi	3150	3045	----	----
	N/cm^2	2172	2100	----	----
B-1 engine accumulator pressure (absolute)	psi	3115	3080	----	----
	N/cm^2	2148	2124	----	----
Sustainer return line pressure (absolute)	psi	85	85	80	86
	N/cm^2	58	58	55	59
Booster system return pressure (absolute)	psi	85	85	----	----
	N/cm^2	58	58	----	----

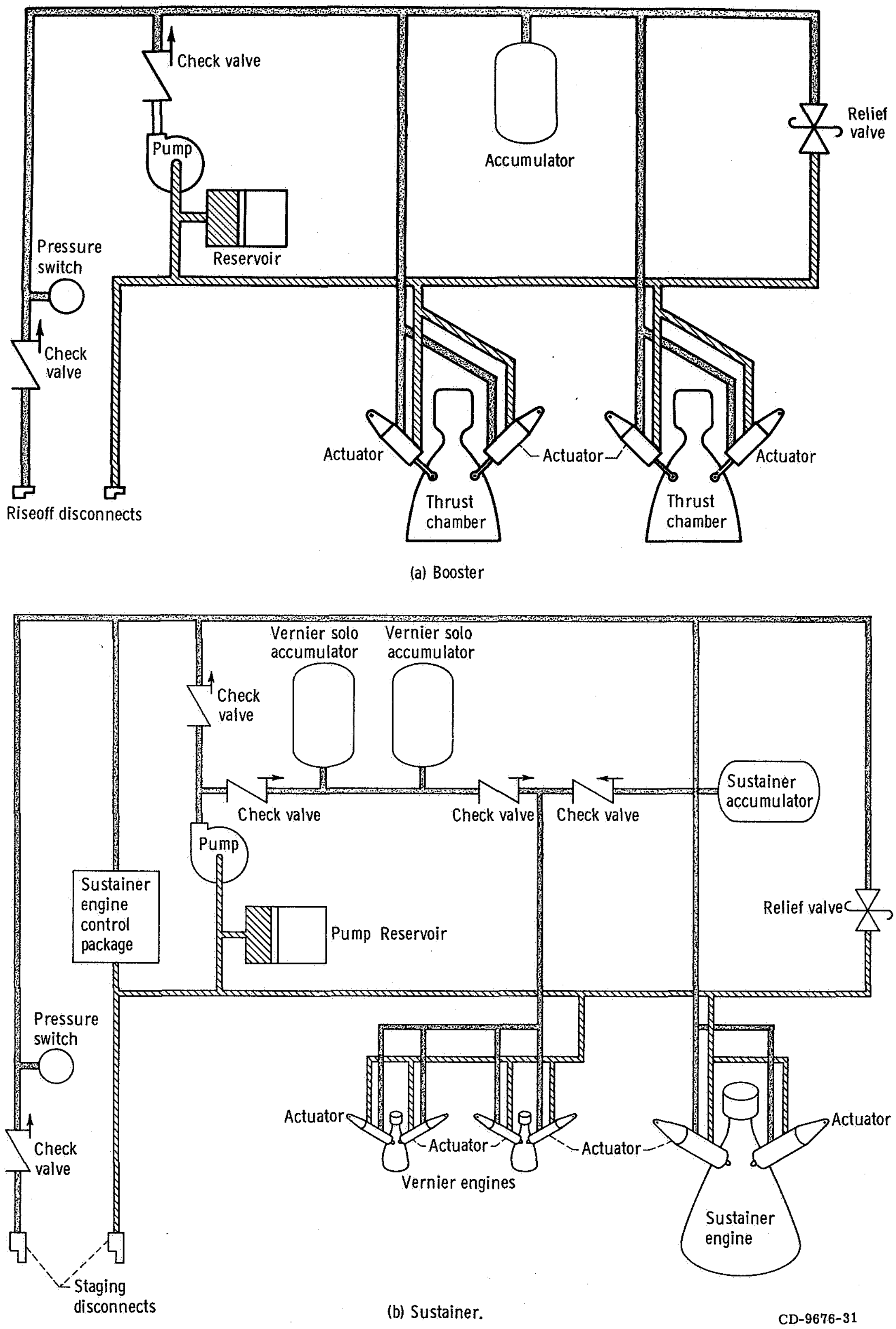


Figure V-2. - Hydraulic system, ATS-1.

PROPELLANT UTILIZATION SYSTEM

Description

The Atlas propellant utilization system (fig. V-3) is designed to cause near-simultaneous depletion of both propellants. This system is a digital type, which adjusts the operating mixture ratio of the sustainer engine by sampling the propellant volume ratio at six discrete points during flight. Six fuel (RP-1) and six oxidizer (liquid oxygen) level sensors are positioned in the propellant tanks so that both sensors will uncover simultaneously if the propellants are being consumed at the proper ratio. If the propellant usage ratio is incorrect, one sensor of a pair will uncover before the other sensor. The time difference in the uncovering of the sensors comprising a pair is directly proportional to the propellant usage ratio error. If this time difference is greater than the limit error times for each sensor pair, the propellant utilization valve is commanded to the full open or closed position, depending on which sensor uncovers first. If the actual error time is less than the limit error time, the valve would be commanded to something less than the fully open or closed position. This adjustment would theoretically result in a zero error time when the liquid level reaches the next sensor pair.

This uncovering time difference is measured and is transmitted to a hydraulic control unit. This hydraulic control unit controls the position of the propellant utilization (fuel) valve and indirectly controls the position of the liquid-oxygen valve. When an error signal is sent to the propellant utilization valve for an increase in fuel flow, the fuel pump discharge pressure will decrease as the valve moves open. The liquid-oxygen head-suppression servocontrol senses this decreasing pressure and causes the liquid-oxygen head-suppression valve to move to restrict the flow of the liquid oxygen to the thrust chamber, thus decreasing the liquid-oxygen injection pressure by approximately the same amount as the decrease in RP-1 (fuel) pump discharge pressure. The net effect of the combined liquid-oxygen head-suppression valve and propellant utilization system performance is a near-constant, because the total flow weight of propellants is supplied to the sustainer thrust chamber.

Performance

Propellant utilization system performance was satisfactory. The fuel and oxidizer sensing ports uncovered at 5 seconds, and the liquid-oxygen head-sensing port uncovered at 6 seconds prior to sustainer engine cutoff. Burnable propellant residuals at sustainer engine cutoff were calculated to be 306 pounds (138.8 kg) of fuel and 499 pounds (226.3 kg) of liquid oxygen. These residuals would have allowed 2.7 seconds more sustainer engine burn time, if required, to achieve the proper velocity. The fuel residual at theoretical liquid-oxygen depletion was calculated to be 71 pounds (32.2 kg).

Fuel and liquid-oxygen pressure-sensing ports provide the final propellant level data. The differential pressure between the ullage and the port pressure is measured. When the pressure differential indicates zero, the propellant level is below the port. As the sensing port uncovers, a time interval is calculated from the instant of port uncovering to sustainer engine cutoff. This time interval is used in determining the propellant residuals.

Table V-III contains the error times between the fuel and liquid-oxygen sensor uncovery for all six sensor pairs. The data show that the error times are all within the limit times; thus, at no time during the flight was full correction capability necessary.

TABLE V-III. - PROPELLANT UTILIZATION SYSTEM

LEVEL SENSOR ERROR TIMES, ATS-1

Station	Limit error time, sec	Actual error time, sec	First sensor uncovered
1	1.018	1.0	Liquid oxygen
2	.968	.4	Liquid oxygen
3	.78	.2	Fuel
4	1.89	.15	Fuel
5	8.4	1.6	Fuel
6	4.2	1.4	Liquid oxygen

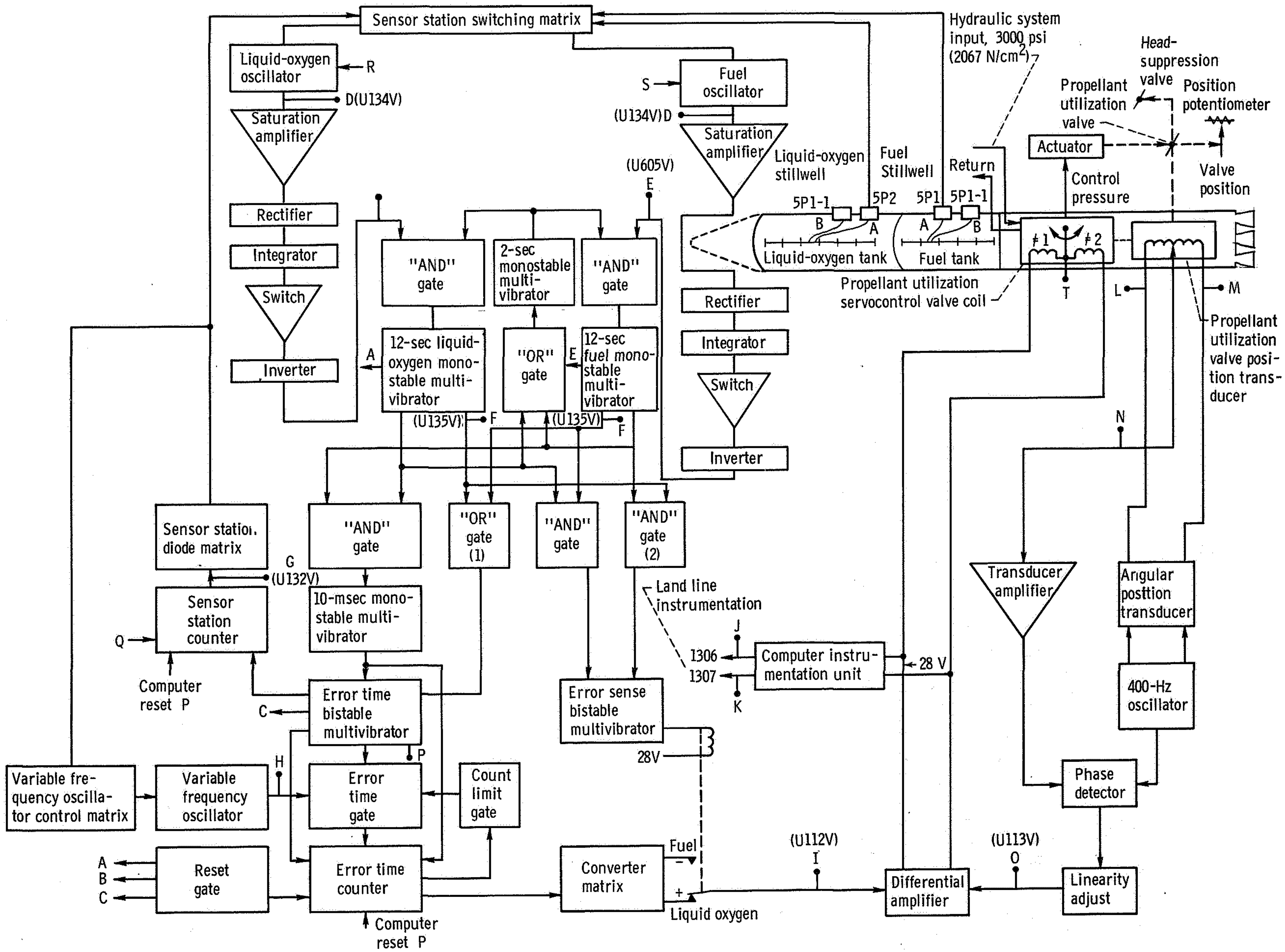


Figure V-3. - Atlas propellant utilization system, ATS-1.

PNEUMATICS SYSTEM

Description

The Atlas pneumatic system supplies helium gas for tank pressurization and for various vehicle control functions. The system comprises three independent subsystems: propellant tank pressurization, engine control, and booster section jettison. This system schematic is shown in figure V-4.

Propellant tank pressurization subsystem. - This subsystem is used to maintain propellant tank pressures at required levels to (1) support the pressure stabilized tank structure and (2) satisfy the inlet pressure requirements of the engine turbopumps. In addition, helium is supplied from the fuel tank pressurization line to pressurize the hydraulic reservoirs and turbopump lubricant storage tanks. The subsystem consists of six shrouded helium storage bottles, a heat exchanger, and fuel and oxidizer tank pressure regulators and relief valves.

The six shrouded helium storage bottles with a total capacity of 44 190 cubic inches (724 000 cm^3) are mounted in the jettisonable booster engine section. The bottle shrouds are filled with liquid nitrogen during prelaunch operations to chill the helium in order to provide a maximum storage capacity at about 3000 psia (2070 N/cm^2). The liquid nitrogen drains from the shrouds at lift-off. During flight, the cold helium from the supply spheres passes through a heat exchanger located in the booster engine turbine exhaust duct and is heated before being supplied to the tank pressure regulators. The propellant tank pressurization subsystem pressurization control is switched from the ground to the airborne system at about T - 60 seconds. Airborne regulators are set to control fuel tank gage pressure between 64 and 67 psi (44.1 and 46.2 N/cm^2) and the oxidizer tank pressure between 28.5 and 31.0 psi (19.5 and 21.4 N/cm^2).

Pneumatic regulation of tank pressure is terminated at booster staging. Thereafter, the fuel tank pressure decays slowly, but the oxidizer tank pressure is sustained by liquid-oxygen boiloff.

Engine controls subsystem. - This subsystem supplies helium pressure for actuation of engine control valves, for pressurization of the engine start tanks, for purging booster engine turbopump seals, and for the reference pressure to the regulators which controls the oxidizer flow to the gas generator. Control pressure in the system is maintained through Atlas-Agena separation. These pneumatic requirements are supplied from a 4650-cubic-inch (76 000-cm^3) storage bottle pressurized to a gage pressure of about 3000 psi (2070 N/cm^2) at lift-off.

Booster engine jettison subsystem. - This subsystem supplies pressure for release of the pneumatic staging latches to separate the booster engine package. A command from the Atlas flight control system opens two explosively actuated valves to supply helium pressure to the 10 piston-operated staging latches. Helium for the system is supplied by a single 870-cubic-inch (14 260-cm^3) bottle charged to a gage pressure of 3000 psi (2070 N/cm^2).

Performance

The pneumatic system performance was satisfactory. All tank pressures were satisfactory and all control functions were performed properly. Pneumatic system parameters during flight are shown in table V-IV. Liquid-oxygen tank ullage pressure oscillations were within the range experienced for previous flights. Prior to lift-off, oscillation frequencies of 3.25 hertz were measured. The oscillation amplitudes (differential pressure across the bulkhead) varied with a maximum peak-to-peak amplitude of 3.0 psi (2.01 N/cm^2). After lift-off, these oscillations increased in frequency to 5.25 hertz and increased in amplitude slightly. These oscillations damped out within the time span experienced in previous flights. These oscillations have been encountered on Atlas boosters that use this type of liquid-oxygen regulator and are considered normal.

TABLE V-IV. - ATLAS PNEUMATIC SYSTEM PERFORMANCE, ATS-1

Parameter	Units	Design range	Relief valve operates	Flight values at -			
				T - 0	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Oxidizer tank ullage pressure (gage)	psi N/cm^2	31.0 to 28.5 21.4 to 19.5	33.0 to 34.0 22.7 to 23.4	29.8 20.5	29.7 20.5	[a]29.0 20.0	29.0 20.0
Fuel tank ullage pressure pressure (gage)	psi N/cm^2	67.0 to 64.0 46.2 to 44.1	69.5 to 70.5 47.9 to 48.6	64.8 44.7	64.9 44.7	[a]49.9 34.4	49.9 34.4
Sustainer controls bottle pressure (gage)	psi N/cm^2	[b]3400 to 2900 2344 to 1999	----------- -----------	3067 2115	2770 1910	2480 1710	1400 965
Booster helium bottles pressure (absolute)	psi N/cm^2	[b]3400 to 2900 2344 to 1999	----------- -----------	3066 2114	630 434	---- ----	---- ----
Booster helium bottles temperature	^oF K	[b]-306 maximum 85.5	----------- -----------	-319 78.5	-382 43.5	---- ----	---- ----

[a]Helium supply bottles are jettisoned with booster engine section at booster engine cutoff + 3 sec. No additional helium is supplied to the propellant tanks.
[b]Required range for T - 0.

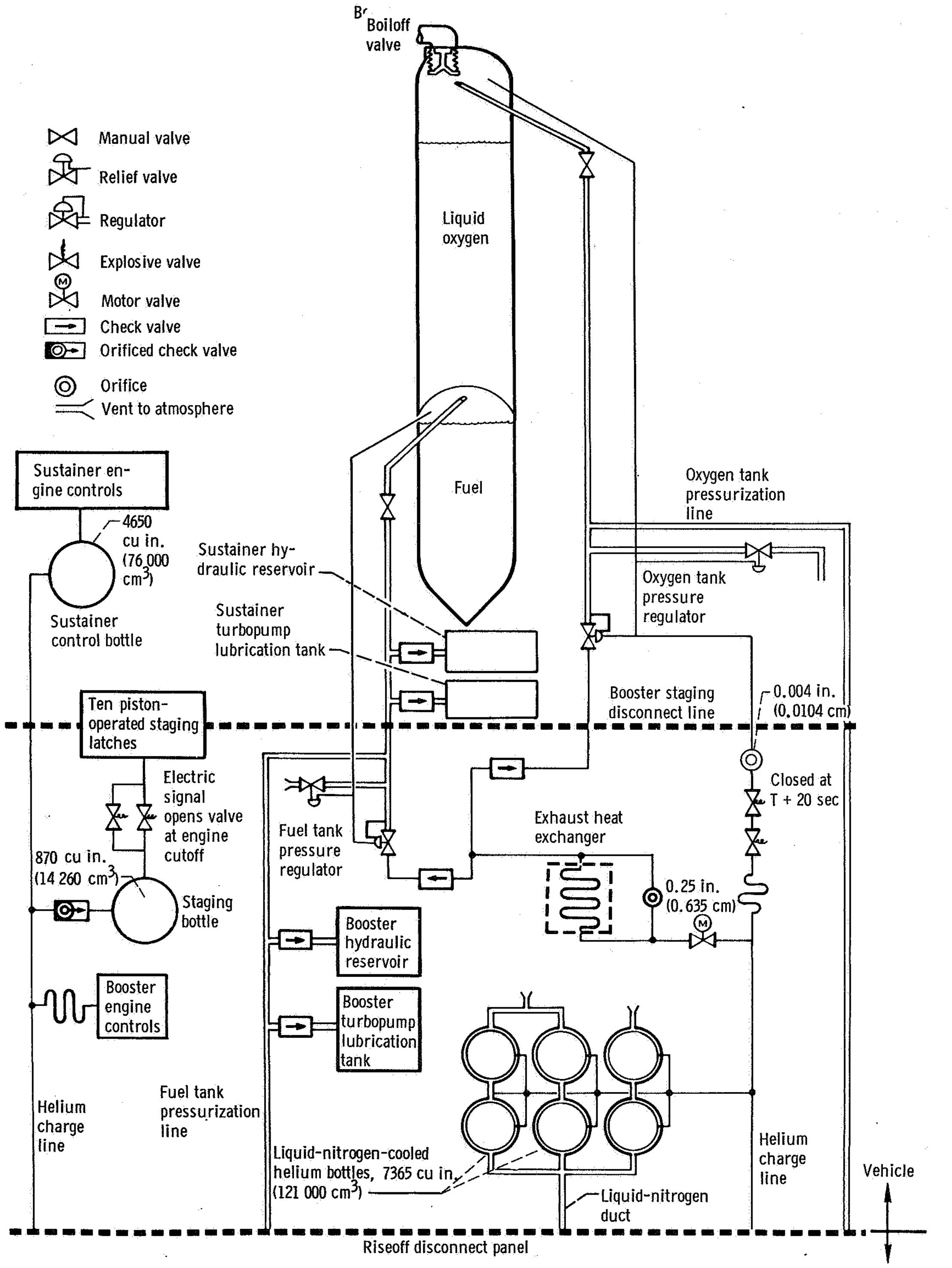


Figure V-4. - Atlas vehicle pneumatic system, ATS-1.

GUIDANCE AND FLIGHT CONTROL SYSTEM

Description

The Atlas flight path is controlled by two interrelated systems: the autopilot system and the Mod III Radio Guidance System. The autopilot directs the vehicle in a programmed open loop mode from lift-off through booster engine cutoff, except for the period from T + 100 to T + 110 seconds. During this 10-second period, the Mod III Radio Guidance System may generate and transmit pitch and yaw steering signals to the vehicle in order to provide any attitude corrections required as a result of vehicle deviations from the preplanned trajectory. After booster engine jettison, steering signals are issued, as required, during the sustainer and vernier phases of flight.

The Mod III Radio Guidance System is also the primary mode for initiating discrete commands for booster engine cutoff, sustainer engine cutoff, start Agena timer, and Atlas-Agena separation.

The Atlas flight control system (fig. V-5) consists of the following major units:

(1) The displacement gyro canister contains three single-degree-of-freedom, floated, rate-integrating gyros, one single-degree-of-freedom, floated, rate gyro, and associated electronic circuitry for gain selection and signal amplification. The displacement gyros are mounted in an orthogonal triad configuration alining the input axis of a gyro to its respective vehicle axis of pitch, yaw, or roll. The input axis of the rate gyro is alined with the vehicle roll axis. Each displacement gyro provides an electrical output signal proportional to the difference in angular position of the measured axis from the gyro reference axis. The rate gyro provides an electrical output signal proportional to the rate of rotation of the vehicle about the gyro input (reference) axis.

(2) The rate gyro canister contains two single-degree-of-freedom, floated, rate gyros and associated electronic circuitry. The input axes of the rate gyros are alined to their respective vehicle axis of pitch and yaw. Each gyro provides an electrical output signal proportional to the angular rate of rotation of the vehicle about the gyro input (reference) axis.

(3) The servoamplifier canister contains electronic circuitry to amplify, filter, integrate, and algebraically sum engine position feedback signals with position and rate signals. The electrical outputs of this unit direct the hydraulic actuators which, in turn, gimbal the engines to provide thrust vector control.

(4) The programmer canister contains an electronic timer, arm-safe switch, high, low, and medium power electronic switches, the fixed pitch program, and circuitry to set the roll program from launch ground equipment. The programmer issues discrete commands to other subsystems.

The Mod III Radio Guidance System includes the vehicle-borne pulse beacon, rate beacon, and decoder, and a ground station comprised of a monopulse X-band (radar) track subsystem, a continuous wave L-band rate subsystem, and a digital guidance computer subsystem.

The track subsystem, which measures range, aximuth angle, and elevation angle, transmits a composite message-train containing an address code and the coded steering and discrete commands. If the address code of the received signal is correct, the vehicle-borne pulse beacon transmits a return pulse to the ground station. The airborne decoder receives the ground-transmitted pulse-position coded message, decodes the message, and issues the pitch and yaw steering signals and discrete commands.

The rate subsystem transmits two continuous wave signals of different frequencies from a single ground antenna. The vehicle-borne rate beacon is interrogated by the signals from the ground subsystem. The rate beacon transmits a continuous wave signal at a frequency equal to the arithmetic average of the frequencies of the received signals. The signal is received by ground based stations which are designated the central rate station and two outlying rate leg stations. The two-way doppler shift and phase relation of the signals, as received at the three separate ground stations, are used to determine the vehicle range, azimuth, and elevation rates.

Acquisition of the vehicle is accomplished through use of an acquisition cube procedure, an optical tracking acquisition aid, or by using range, azimuth, and elevation data supplied by the Eastern Test Range. In the acquisition cube procedure, which is the primary method of acquisition, the antennas are directed to one of seven predetermined positions along the programmed trajectory. These positions represent cubes defined by range, azimuth angle, and elevation angle.

A conical scan antenna on the same mount as the X-band track subsystem antenna is used for initial acquisition. Once the vehicle is acquired by the conical scan antenna, tracking is automatically switched to the main track antenna. The rate subsystem antennas are slaved to the track subsystem antennas. Lock-on of the rate subsystem is normally accomplished before lock-on of the track subsystem due to differences in antenna gains, antenna beam widths, and to receiver sensitivities.

The position and rate information from the track and rate subsystems is sent to the ground computer where the steering signals are determined. The steering signals are transmitted to the vehicle and used to guide the vehicle along the desired flight path.

Performance

The flight control system autopilot performance was satisfactory. Lift-off transients were within acceptable limits. Before activation of the flight control system, an initial roll transient was observed with a maximum displacement of 0.5° at a peak rate of 2.8 degrees per second in a clockwise direction (viewing the vehicle from aft). The flight control system was activated by a 42-inch- (106.7-cm-) motion switch that quickly damped this transient. The resulting roll overshoot transient (following engine gimbal activation) was 0.5° counterclockwise at a peak rate of 0.4 degree per second. This transient was also quickly damped.

During the programmed roll maneuver, the roll rate gyro indicated an average roll rate of 0.115 degree per second for 13 seconds. This resulted in a total roll of 1.5°. The programmed launch azimuth required a total roll of 1.28°. The pitch program was initiated at T + 15 seconds, as planned. The actual compared with nominal times and amplitudes of each step are listed in table V-V. The actual roll and pitch maneuver dispersions were within acceptable limits.

The vehicle was displaced by less than 0.1° pitch up and a yaw left of less than 0.6° at the time that booster engines were commanded to zero. Following booster engine staging, the sustainer engine returned the vehicle to the attitude it had before booster engine cutoff. Flight control system gyro data indicated only small transients at staging.

After sustainer burn and vernier solo, Atlas-Agena separation was initiated by radio guidance. The vehicle was in a stable attitude at separation.

Postflight evaluation of ground and vehicle data indicates that both the ground station and the vehicle-borne guidance equipment performed satisfactorily.

The track subsystem acquired the vehicle in the first cube at T + 60.7 seconds using the conical scan antenna. The track subsystem was automatically switched to the main antenna at T + 63.6 seconds, and good data were presented to the computer by T + 66.7 seconds.

Tracking was continuous from time of vehicle acquisition until T + 400.8 seconds (this was 86.0 sec after Atlas-Agena separation). Tracking was then intermittent until final loss of tracking occurred at T + 411.2 seconds when the Atlas was at an elevation angle of 2.82° above the horizon. The signal received by the track subsystem throughout the flight was within 3 decibels of the theoretically expected level.

Lock-on of the rate subsystem was accomplished by T + 59.5 seconds, and good data were presented to the computer by T + 63.2 seconds. Lock-on was continuous thereafter until T + 393.1 seconds. Rate data were lost at T + 411.2 seconds coincident with the loss of tracking. The signals received by the central rate antenna were within 4 decibels of the expected (calculated) levels, and the signals received at the two rate leg antennas were within 2 decibels of those received at the central rate antenna.

The computer subsystem performance was satisfactory throughout the countdown and vehicle flight. A postflight simulation of the flight successfully verified the guidance program and indicated that no transient errors occurred during the flight.

The pulse beacon automatic gain control monitor indicated a received signal strength of -54 dBm (decibels referenced to 1 mW) at acquisition, with a signal increase to approximately -18 dBm at T + 64 seconds. The received signal strength reached a maximum of -8 dBm at T + 79 seconds and gradually decayed to -33 dBm at Agena separation. Then it continued to decay until T + 400 seconds when the received signal strength was less than -70 dBm.

From T + 59 to T + 402 seconds, the magnetron current monitor indicated good pulse beacon response except for a normal momentary loss of telemetry data during booster staging.

The rate beacon automatic gain control monitors indicated that the received signal strength of the two carrier frequencies reached a level in excess of -75 dBm by T + 57 seconds and remained above -75 dBm until approximately T + 383 seconds. The signal strength of the two carrier frequencies gradually decayed to the threshold sensitivity of the receiver, -85 dBm, by approximately T + 393 seconds. The rate beacon phase detector and power output monitors indicated that the signals received by the vehicle were processed and that the return signal was properly transmitted to the ground station during the period from T + 57 to T + 393 seconds.

The steering and discrete commands transmitted from the ground station were properly processed by the decoder.

Spurious pitch and yaw commands were observed, as on prior flights, during the periods of intermittent pulse beacon lock between T + 50.6 and T + 59.0 seconds during cube acquisition. The spurious commands were less than ±20 percent of maximum command during the preceding interval. Since guidance steering was not enabled at this time, the vehicle did not respond to these spurious commands.

Guidance steering was enabled by the flight control system at T + 80 seconds but was constrained except for the period T + 100 to T + 110 seconds. Booster steering was commanded by the ground guidance station at T + 103.2 seconds, with an initial pitch-down command of 20 percent of maximum for a duration of one computer cycle (0.5 sec). This command was followed 4 seconds later by another pitch-down command of 20 percent for one computer cycle. These commands resulted in a vehicle pitch-down rate of 0.25 degree per second. The resultant pitch-down rate indicated that only minor booster steering by radio guidance was required.

Sustainer steering was initiated at T + 137.8 seconds. The largest steering command outputs from the decoder were a yaw-right command of 60 percent of maximum steering and a pitch-down command of 75 percent. This was followed by a pitch-up command of 100 percent. Both pitch and yaw steering commands were reduced to within ±10 percent of maximum steering by T + 144.0 seconds, and remained so until sustainer engine cutoff. The amplitude and duration of steering commands indicated normal steering by radio guidance.

Vernier steering commands to correct vehicle attitude resulted in a 100-percent pitch-up and a 90-percent yaw-left command, both for a duration of 1 second. These were within the acceptable limits. These commands caused the vehicle to pitch up 0.96° and to yaw left 1.26°.

Table V-VI gives the times at which the actual booster engine cutoff start Agena timer, sustainer engine cutoff, vernier engine cutoff, and Atlas-Agena separation discretes were generated by the guidance computer.

TABLE V-V. - ATLAS PITCH PROGRAM, ATS-1

Time interval, sec		Step level, deg/sec	
Programmed	Actual	Programmed	Actual
0 to 15	0 to 15	0	0
15 to 35	15 to 35	1.018	1.000
35 to 45	35 to 45	.848	.850
45 to 58	45 to 58	.509	.500
58 to 70	58 to 70	.678	.650
70 to 82	70 to 82	.806	.800
82 to 91	82 to 91	.678	.650
91 to 105	91 to 105	.551	.550
105 to 120	105 to 120	.382	.400
120 to 138.9	120 -----	.254	.275
138.9 to sustainer engine cutoff	---------	.042	----

TABLE V-VI. - MOD III RADIO GUIDANCE CONTROLLED EVENTS, ATS-1

Flight event and trajectory function	Units	Actual discrete generation time	Vehicle location at time of discrete	Vehicle velocity at time of discrete	Discrete duration
Booster engine cutoff	sec	T + 128.837			0.497
Range	ft		319 241		
	m		97 305		
Azimuth	deg		100.257		
Elevation	deg		32.576		
Range rate	ft/sec			8089	
	m/sec			2466	
Start Agena timer	sec	T + 288.197			0.637
Sustainer engine cutoff	sec	T + 292.914			0.920
Range	ft		2 233 581		
	m		680 785		
Azimuth	deg		103.580		
Elevation	deg		10.256		
Range rate	ft/sec			16 876	
	m/sec			5144	
Vernier engine cutoff	sec	T + 312.740			0.594
Range	ft		2 567 899		
	m		782 686		
Azimuth	deg		103.707		
Elevation	deg		8.787		
Range rate	ft/sec			16 858	
	m/sec			5138	
Shroud separation[a]	sec	T + 313.837			0.497
Atlas-Agena separation	sec	T + 314.837			To end

[a]This discrete is normally used for over-the-nose shroud separations; however, since ATS-1 used the Standard Agena Clamshell shroud, this discrete was not used by the vehicle even though it was sent by the ground station.

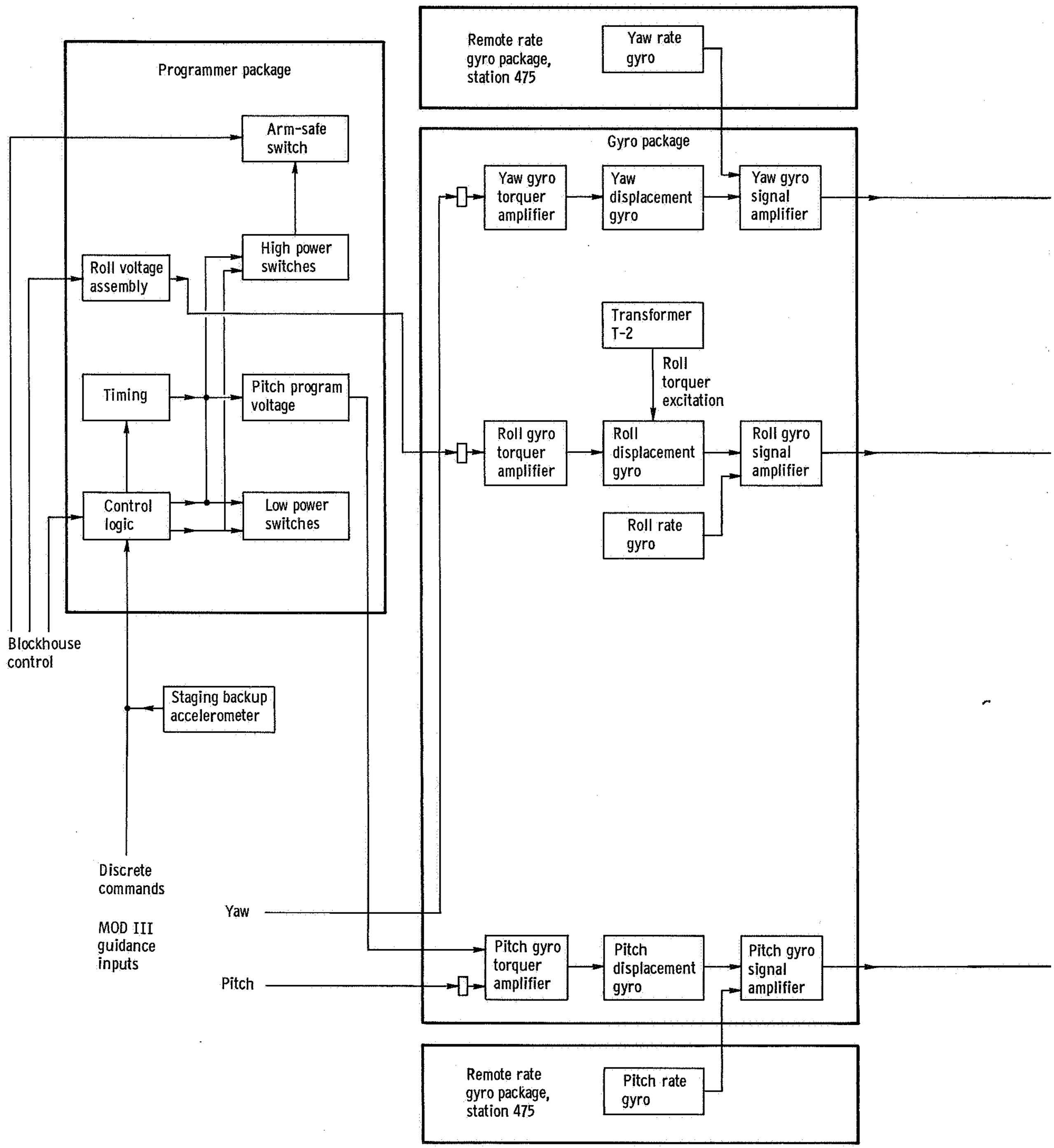
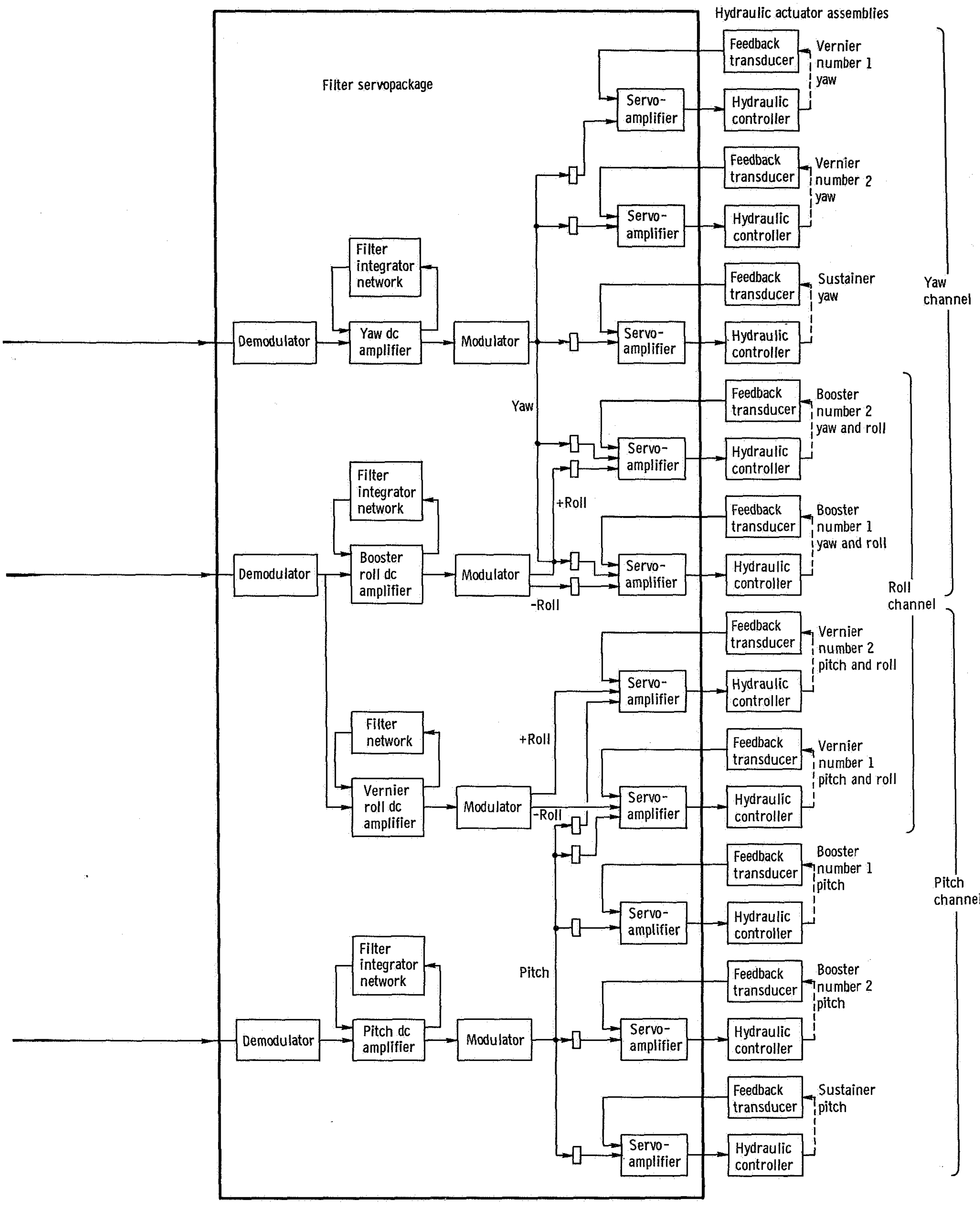


Figure V-5. - Atlas flight control



system block diagram, ATS-I.

ELECTRICAL SYSTEM

Description

The Atlas electrical system consists of four 28-volt dc batteries, a 400-hertz inverter, a power changeover switch, a distribution box, two junction boxes, and related electrical harnesses. The main battery (28 A-hr) supplies 28 volts dc power to the flight control system, the radio guidance system, the propellant utilization system, the propulsion system, and the inverter. One battery supplies 28 volts dc power to the telemetry system, and two batteries supply 28 volts dc power to the flight termination system. The inverter also supplies 115 volts ac, 3 phases, and 400 hertz to the flight control system, the propellant utilization system, and the radio guidance system. Phase A of the inverter is used as a phase reference in the flight control system and the radio guidance system.

The vehicle electrical system operates from ground power sources until 2 minutes prior to lift-off. At this time, the power changeover switch is used to transfer the electrical loads to vehicle power.

Performance

All measured electrical system parameters were within specifications at all times. The inverter frequency was 399.7 hertz at T - 0 seconds and increased as expected to 400.6 hertz at Atlas-Agena separation. The inverter phase A output voltage was 116.2 volts at T - 0 and decreased as expected to 115.2 volts at Atlas-Agena separation. The main 28-volt dc battery voltage varied from 28.0 volts dc at T - 0 seconds to 28.1 volts dc at Atlas-Agena separation. Atlas electrical system flight data are shown in table V-VII.

TABLE V-VII. - ATLAS ELECTRICAL SYSTEM PERFORMANCE DATA, ATS-1

Parameters	Units	Countdown limits	Flight events			
			Lift-off	Booster engine cutoff	Sustainer engine cutoff	Atlas-Agena separation
Main battery voltage	V dc	26.6 minimum	28.0	27.9	28.1	28.1
Inverter frequency	Hz	394.0 minimum 404.0 maximum	399.7	400.0	400.6	400.6
Phase A voltage	V ac	113.6 minimum 117.0 maximum	116.2	115.5	115.3	115.2
Phase B voltage	V ac	---------	115.9	115.8	115.8	115.7
Phase C voltage	V ac	---------	116.0	115.9	115.7	115.5

TELEMETRY AND INSTRUMENTATION SYSTEM

Description

The Atlas telemetry and instrumentation system consists of one telemetry package, a 28-volt dc battery, transducers, wiring, and antennas.

The 18 channel PAM/FM/FM telemetry package consists of a transmitter, commutator assemblies, signal conditioning components, and the subcarrier oscillators. The letter designation PAM refers to Pulse Amplitude Modulation, a technique of sampling data to allow utilization of the data handling capacity of the telemetry system. The letter designation FM/FM refers to Frequency Modulation/Frequency Modulation, a technique of frequency modulating a transmitter with the output of several subcarrier oscillators which, in turn, have been modulated by data signals.

The telemetry transmitter has an output power of 3.5 to 6 watts and requires 28 volts dc for operation. The transmitter for the Atlas is designed to use standard subcarrier channels 1 to 18 (see table V-VIII). The subcarrier oscillators frequency modulate the 249.9-megahertz carrier wave.

The vehicle instrumentation system is used to monitor specific parameters and functions of the Atlas systems. A complete list of the instrumentation flown on the Atlas vehicle is given in appendix B.

Performance

The performance of the telemetry and instrumentation system was satisfactory. No measurement failures were noted; however, the data playback from station 4 located on Merritt Island was slightly noisy. At T + 249 seconds, a momentary loss of data occurred on channels 15, 16, and 18. This momentary loss of data was noted at all Cape Kennedy stations and was the result of the antenna look angle. No momentary loss of data occurred at downrange stations. Commutator speeds were stable during Atlas phase of flight. The stations used to record the Atlas telemetry data are presented in appendix C (fig. C-2).

TABLE V-VIII. - ATLAS TRANSMITTER

SUBCARRIER CHANNELS, ATS-1

Channel	Type
1	Not used for this program
2	Not used for this program
3	Continuous direct (no subcarrier oscillator)
4	Continuous direct (no subcarrier oscillator)
5	Continuous
6	
7	
8	
9	
10	
11	Commutated at 2.5 revolutions/sec
12	Continuous direct (no subcarrier oscillator)
13	Commutated at 5 revolutions/sec
14	Not used for this program
15	Commutated at 10 revolutions/sec
16	Commutated at 10 revolutions/sec
17	Not used for this program
18	Commutated at 30 revolutions/sec

FLIGHT TERMINATION SYSTEM

Description

The Atlas contains a vehicle-borne flight termination system which is designed to function on receipt of command signals from the ground stations. This system includes redundant receivers, a power control unit, an electrical arming unit, a destructor, and two batteries which operate entirely independent of the main vehicle power system.

The Atlas flight termination system provides a highly reliable means of shutting down the engines only, or shutting down the engines and destroying the vehicle. When the vehicle is destroyed in the event of a flight malfunction, the tank is ruptured with a shaped charge, and the liquid propellants are dispersed. The operation of the flight termination system is commanded by the range safety officer only.

Performance

Performance of the flight termination system was satisfactory. Prelaunch checks were completed without incident. Telemetry automatic gain control measurements indicated that the capability to terminate flight was maintained throughout powered flight. No commands were required, nor were any commands inadvertently generated by any vehicle system.

VI. AGENA VEHICLE SYSTEM PERFORMANCE

VEHICLE STRUCTURE SYSTEM

Description

The Agena vehicle structure system consists of four major sections: the forward section, tank section, aft section, and the booster adapter assembly. The forward section is an aluminum structure covered with magnesium panels (fig. VI-1). This section encloses most of the electrical, guidance, and communication equipment and provides the mechanical interface for the spacecraft and shroud. The 5-foot- (1.52-m-) diameter tank section consists of two integral aluminum propellant tanks with a sump below each tank for engine start and propellant feed. The aft section consists of an engine mounting cone structure and an equipment mounting rack. The magnesium alloy booster adapter supports the Agena and remains with the Atlas after Atlas-Agena separation.

Performance

The measured dynamic environment of the structure system was within design limitations. The measured data are presented in appendix D.

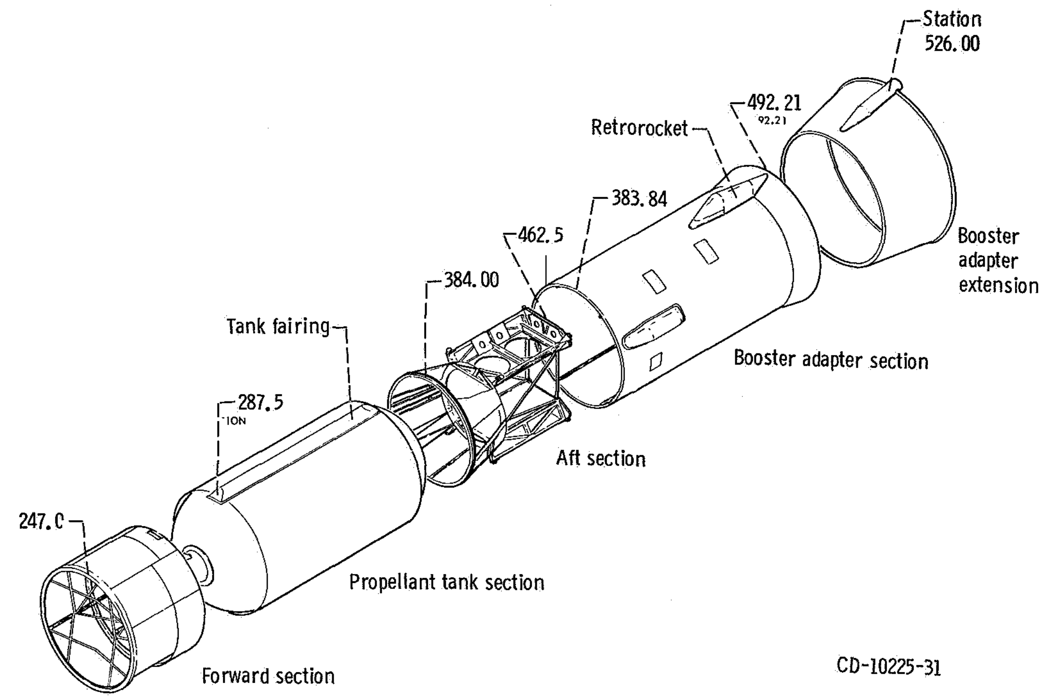


Figure VI-1. - Agena vehicle structure system, ATS-1.

SHROUD SYSTEM

Description

The Standard Agena Clamshell shroud system was developed by the Lewis Research Center to provide a standard payload shroud for Agena missions. As shown in figure VI-2, the Standard Agena Clamshell shroud system flown on ATS-1 includes a transition ring and two shroud halves that form a fairing consisting of a cylindrical section, a conical section, and a hemispherical dome. The weight for the ATS-1 system was 723 pounds (327.9 kg). The two longitudinal halves are made of fiberglass strengthened by internal aluminum longerons and ribs. The halves are held together by a nose bolt, two flat bands around the cylindrical section, and a V-band at the base of the cylinder. The V-band clamps the shroud halves to the aluminum transition ring that is bolted to the Agena forward section. The top and middle flat bands and the bottom V-band are tensioned to 5000 pounds (22 241 N), 2600 pounds (11 565 N), and 8000 pounds (35 586 N), respectively. The spacecraft adapter is mounted to the top surface of the transition ring (Agena station 245). A metal diaphragm across the bottom of the transition ring is used to isolate the shroud compartment from the Agena.

The shroud compartment is vented through four vent ports during ascent (see fig. VI-2). The vent ports are designed to permit venting only in an outward direction. The Agena is vented independently of the shroud.

Figure VI-3 shows an ATS composite spacecraft mounted on the transition ring with one shroud half installed. The ATS-1 spacecraft was encapsulated in the shroud in an environmentally controlled explosive safe area and then transported as a unit to the launch pad and mated to the Agena.

Shroud jettison was initiated approximately 10 seconds after Agena engine first start. At this time, two pyrotechnic bolt cutters were fired in the nose bolt assembly and two explosive bolts in each of the three shroud bands were also fired. Springs at the base of the shroud then forced the halves to rotate about hinges mounted on the transition ring. In the 1-g acceleration field provided by the Agena at the time of shroud separation, each shroud half rotated approximately 75° and then fell free of the vehicle.

Performance

The data from two temperature transducers mounted on the shroud inner skin at Agena vehicle station 176 are shown in figure VI-4. The peak temperature measured was 192° F (362 K), which was well within the worst case maximum predicted temperature of 280° F (468.5 K).

The differential pressure measured across the shroud diaphragm in flight is shown in figure VI-5. The differential pressure was essentially zero during the early portion of the flight. During the transonic phase, the differential pressure was -0.8 psi (-0.55 N/cm^2). A negative pressure differential is defined as a shroud cavity pressure less than the pressure in the Agena forward equipment section. After the transonic phase, the differential pressure became slightly positive for a short period and then returned nearly to zero at T + 85 seconds, as shown in figure VI-5. It remained at essentially this level for the remainder of the flight. The differential pressure of -0.8 psi (-0.55 N/cm^2) was the result of the development of shock waves which raised the pressure in the Agena forward equipment section.

Shroud separation pyrotechnics were fired at T + 369.3 seconds, and shroud jettison started 21 milliseconds later. The start of shroud jettison was determined by the cessation of data transmitted through electrical connectors on the shroud halves. These connectors separate after first motion of the shroud halves. Total shroud jettison time after the firing of shroud pyrotechnics was approximately 1.65 seconds. No measurable Agena roll, pitch, or yaw rates developed as a result of shroud jettison.

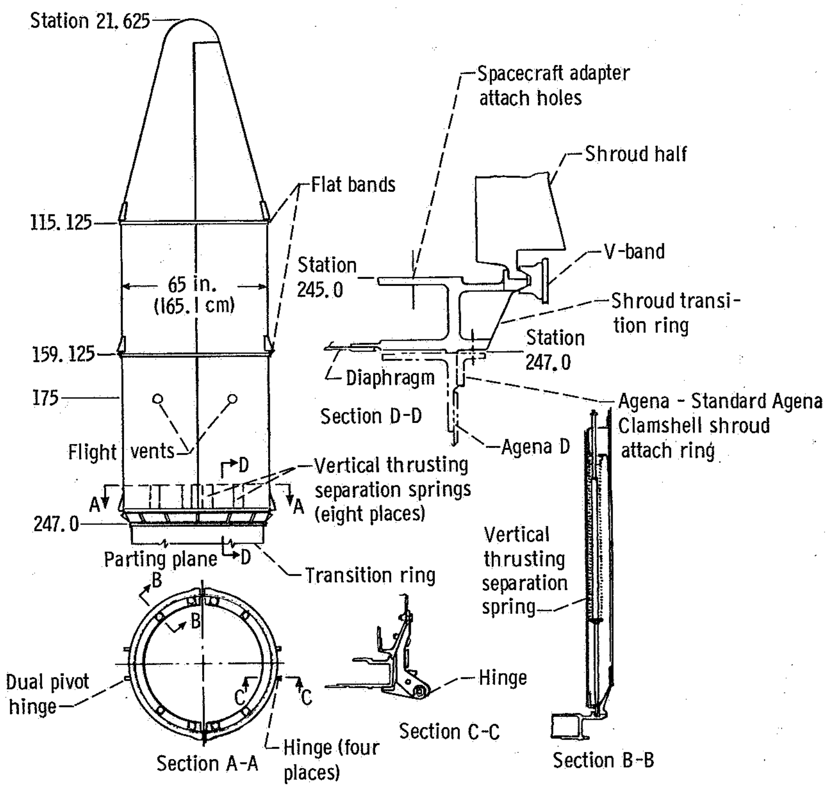


Figure VI-2. - Standard Agena clamshell shroud system, ATS-1.

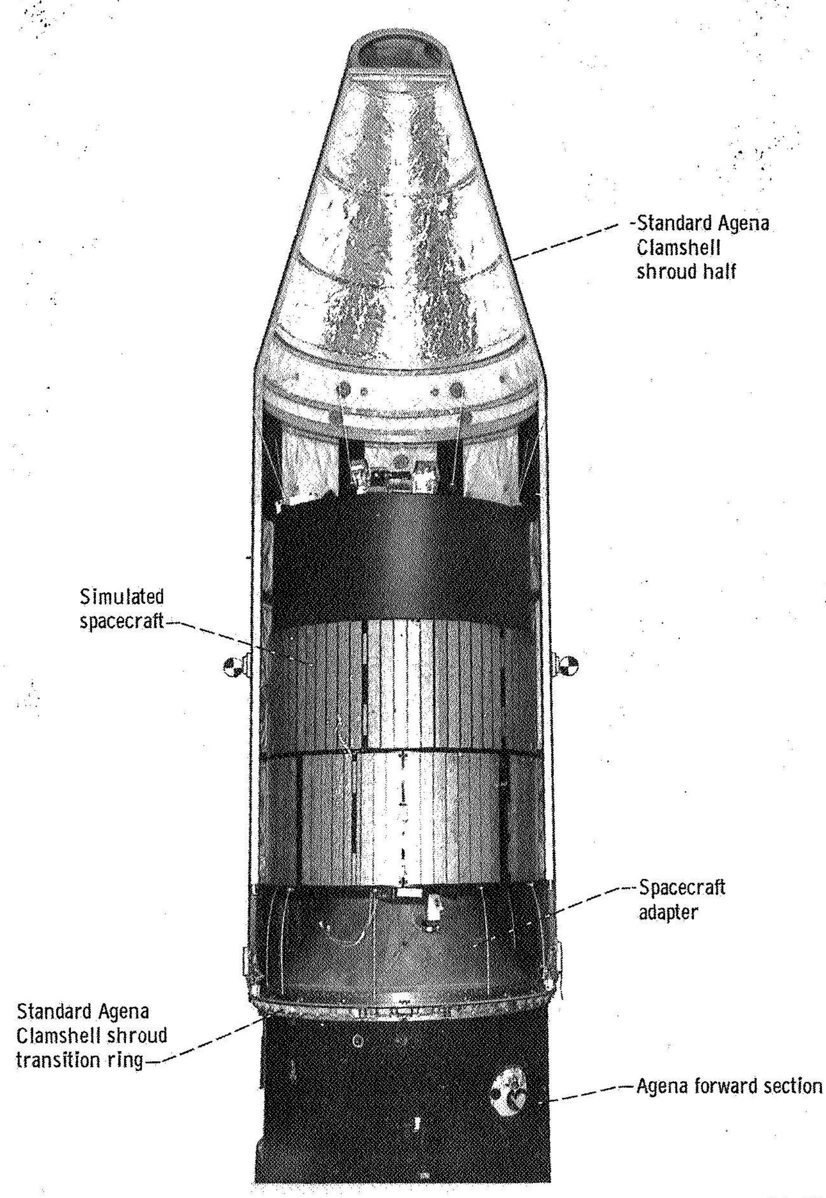


Figure VI-3. - Shroud - simulated spacecraft, ATS-1.

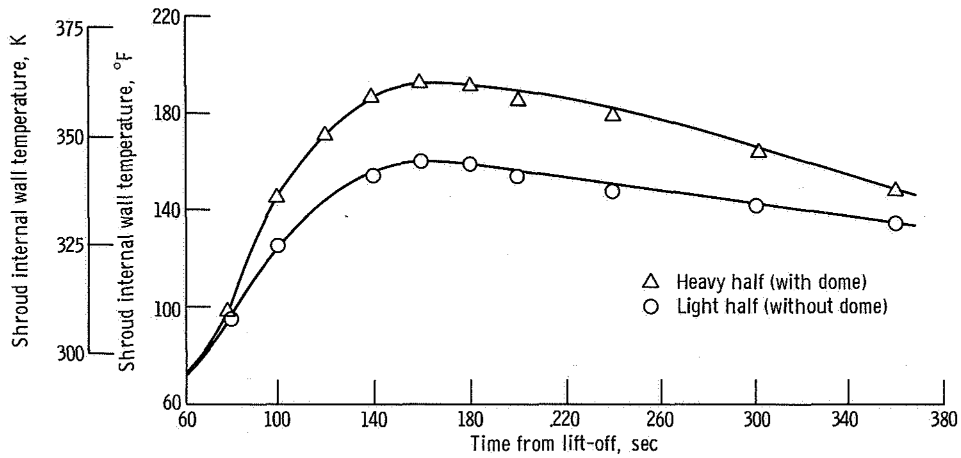


Figure VI-4. - Shroud internal wall temperature as function of time from lift-off, ATS-1.

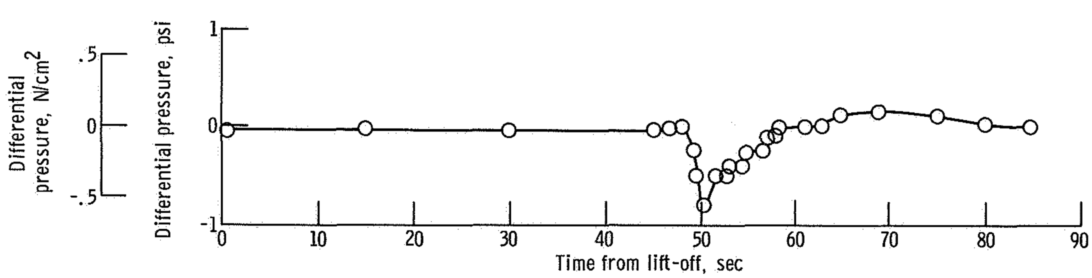


Figure VI-5. - Differential pressure across diaphragm as function of time from lift-off, ATS-1.

PROPULSION SYSTEM

Description

The Agena propulsion system, as shown in figure VI-6, consists of a propellant tank pressurization system, a propellant management system, and an engine system. It also includes (not shown) the Atlas-Agena separation system and vehicle pyrotechnic devices.

The propellant pressurization system consists of a helium supply tank and a pyrotechnically operated helium control valve to provide the required propellant tank pressures. Prior to lift-off, the ullage volume in the propellant tanks is pressurized with helium. A pyrotechnically operated helium control valve is actuated after Agena first ignition to permit helium gas to flow from the storage sphere through fixed-area flow orifices to the propellant tanks. At a predetermined time after the Agena first cutoff, the pyrotechnically operated helium control valve is again activated to isolate the oxidizer tank. Pressurization for the Agena engine second burn is provided by residual propellant tank pressures.

The propellant management system is used to load propellants, distribute propellants to the engine, and retain propellants in the tank sumps for engine restart after a zero-gravity coast. A propellant isolation valve in each propellant feed line is open at lift-off, closed after the end of Agena first burn for venting, and opened prior to the start of Agena second burn. These valves when in the closed position vent propellants trapped in the engine pumps and engine feed lines.

The Agena engine is a Bell Aerosystems Company Model 8096 liquid bipropellant engine which uses unsymmetrical dimethylhydrazine (UDMH) fuel and inhibited red fuming nitric acid (IRFNA) oxidizer. Rated thrust in a vacuum is 16 000 pounds (71 168 N) with a nozzle expansion area ratio of 45. The engine has a regeneratively cooled thrust chamber and a turbopump-fed propellant flow system. Turbine rotation is initiated for each engine firing by 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. Engine thrust vector control is provided by the thrust chamber which is gimbal mounted. A pair of hydraulic actuators provide the force for thrust chamber movement in response to signals produced by the vehicle guidance system.

An oxidizer fast-shutdown system is used to provide a rapid closure of the main oxidizer valve at first-burn cutoff. This oxidizer fast-shutdown system consists of a pyrotechnically operated valve and a high pressure nitrogen storage cylinder.

After Atlas vernier engine cutoff, Atlas-Agena separation is accomplished by a Mild Detonating Fuse which cuts the interstage adapter circumferentially near its forward end. The Atlas and interstage adapter are then separated from the Agena by firing two retrorockets mounted on the interstage adapter.

Electrically initiated pyrotechnic devices are used to perform a number of functions on the Agena. These devices include squibs, igniters, detonators, explosive bolt cutters, and explosive bolt cartridges. Squibs are used to open and close the helium control valve, to eject the horizon sensor fairings, and to activate the fast closing oxidizer valve system. Igniters are used to initiate the solid propellant starter charges and solid propellant retrorockets. Detonators are used to ignite the command destruct charge and the Mild Detonating Fuse separation charge. Explosive bolt cartridges are used to rupture the shroud band bolts. Explosive bolt cutters are used to sever the shroud nose bolts (see VI. SHROUD SYSTEM).

Performance

The propellant tank pressurization system performance was satisfactory. Adequate fuel and oxidizer pump inlet pressures were provided during the Agena first and second firings.

The Agena engine performance was satisfactory except for a momentary drop in chamber pressure during the second burn. The Agena engine first-burn start sequence was initiated by the primary sequence timer at T + 359.2 seconds. Telemetered data of the engine switch group monitor indicated a normal start sequence. Ninety percent chamber pressure was attained 1.2 seconds later, indicating a normal start transient. Engine shutdown was commanded by the velocity meter at T + 521.7 seconds. Engine first-burn duration (90 percent chamber pressure to velocity meter cutoff signal) was 161.3 seconds. This was 1.2 seconds longer than predicted but within the three-sigma variation of ±3.96 seconds. The propellant isolation valves were closed after Agena engine cutoff at T + 527.2 seconds to vent propellants trapped in the engine pumps and feed lines. This was verified by the decay in fuel and oxidizer pump inlet pressures.

Two seconds prior to initiation of the Agena second burn, the propellant isolation valves were opened, as evidenced by the increase in pump inlet pressures. The Agena engine second-burn ignition sequence occurred at T + 1170.3 seconds. Telemetered data of the engine switch group monitor indicated a normal start sequence. Ninety percent chamber pressure was achieved 1.1 seconds later. Engine performance was normal for approximately 7.5 seconds after initiation of the engine start sequence. The chamber absolute pressure at T + 1177.8 seconds was 506.5 psi (349.2 N/cm^2). However, at this time (T + 1177.8 sec), the chamber absolute pressure started to drop reaching a low value of 479.5 psi (330.6 N/cm^2) at T + 1178.5 seconds. At T + 1179.0 seconds, the chamber absolute pressure recovered to 505.0 psi (348.2 N/cm^2) and increased to 525.2 psi (362.1 N/cm^2) by T + 1180.5 seconds. For the remainder of the Agena second burn, the chamber absolute pressure averaged 524.5 psi (361.6 N/cm^2). This was 15.5 psi (10.7 N/cm^2) above the predicted value of 509 psi (350.9 N/cm^2). This anomaly was confirmed by the turbine speed and the oxidizer and fuel venturi inlet pressure data. Engine shutdown was commanded by the velocity meter at T + 1248.7 seconds, resulting in a second-burn duration (90 percent chamber pressure to velocity meter cutoff signal) of 77.3 seconds. This was 1.7 seconds shorter than the predicted burn duration and was the result of higher-than-predicted chamber pressure. However, burn duration was within the three-sigma variation of ±2.2 seconds.

This was the seventh Agena flight to experience a momentary decay in chamber pressure out of approximately 140 flights. The exact cause of this anomaly is unknown at this time, but the primary suspect area is the turbopump on the Agena main engine. A comprehensive historical data search and turbopump test program has been initiated to determine the exact cause of this phenomenon.

The Atlas-Agena separation system performance was normal. Separation was initiated at T + 315.1 seconds by igniting the Mild Detonating Fuse and firing the two retrorockets on the interstage adapter. Complete separation of the Atlas and Agena was accomplished in 2.3 seconds.

All the Agena pyrotechnic devices performed their intended functions. Momentary electrical shorting of some squibs occurred after firing but this did not produce any adverse effect on vehicle performance (see VI. ELECTRICAL SYSTEM). The Agena command destruct charge and detonator were not activated during the flight since the Atlas and Agena flight trajectory was within the allowable flight corridor.

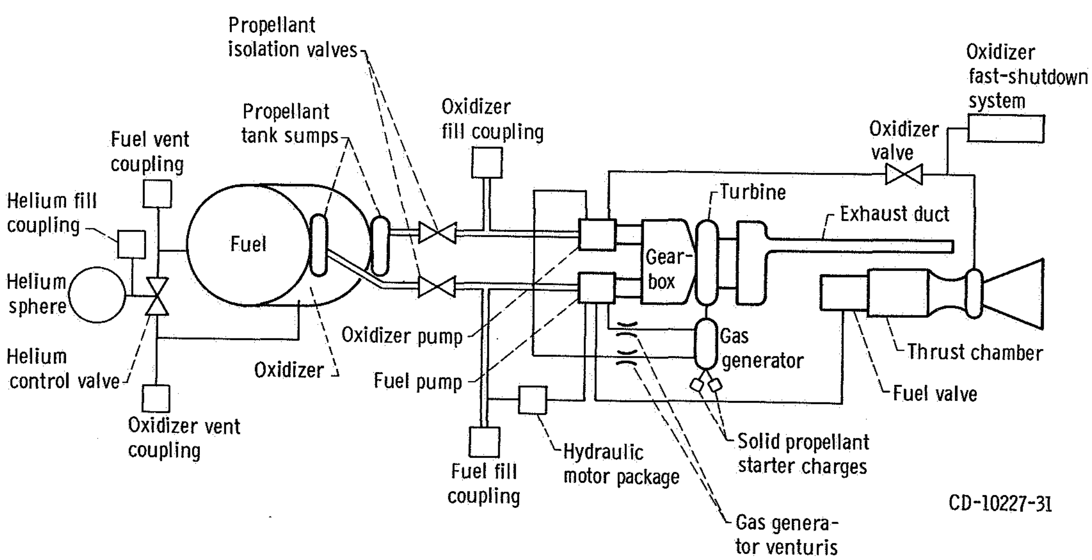


Figure VI-6. - Agena main engine propellant system schematic, ATS-1.

ELECTRICAL SYSTEM

Description

The Agena electrical system consists of batteries, power conversion equipment, and a distribution network. Figure VI-7 is a block diagram of the Agena electrical system. The electrical system supplies all power for the pyrotechnic, propulsion, flight termination, guidance, and telemetry systems.

Four batteries are used on the Agena vehicle. Two silver-zinc primary batteries (type VI-A, minimum design rating of 966 W-hr) supply the unregulated dc power; one of the primary batteries is used for the main vehicle power, the other primary battery is used for the pyrotechnic system, but can support the main vehicle system through a diode which isolates pyrotechnic transients from the main vehicle power. Two nickel-cadmium secondary batteries are used for the flight termination system.

The power conversion equipment converts unregulated 28 volts dc power to regulated ac and regulated dc power. The power conversion equipment consists of one inverter and two dc-dc converters. The inverter supplies 115 volts rms (±1 percent) at 400 hertz (±0.02 percent) to the guidance system. One dc-dc converter supplies ±28.3 volts dc (±1 percent) to the guidance system. The other converter supplies 28.3 volts dc (±1 percent) to the telemetry system.

Performance

The electrical system voltages and currents were normal at lift-off, and met the demands of the using systems throughout flight.

The main voltage was 25.5 volts dc at lift-off, and varied between 25.0 and 25.5 volts in flight. The main unregulated current was 15 amperes at lift-off, and varied between 10 and 16 amperes in flight. The pyrotechnic voltage was 27.0 volts dc at lift-off, and varied between 26.2 and 27.0 volts in flight.

The plus regulated dc voltage was constant at 27.6 volts, and the minus regulated dc voltage was constant at -28.3 volts throughout the flight. The inverter phase voltages were approximately 114.1 volts rms throughout the flight.

The ground current through the airframe was zero except during the two Agena engine burn periods when the current was 3 amperes, as expected, and during the momentary shorting of some squibs. The squib shorts were also observed on the unregulated current monitor coincident with the squib firing of the oxidizer fast-shutdown valve and spacecraft separation squib firings. Neither the current during Agena burns nor the momentary shorting of the squibs affected the vehicle performance.

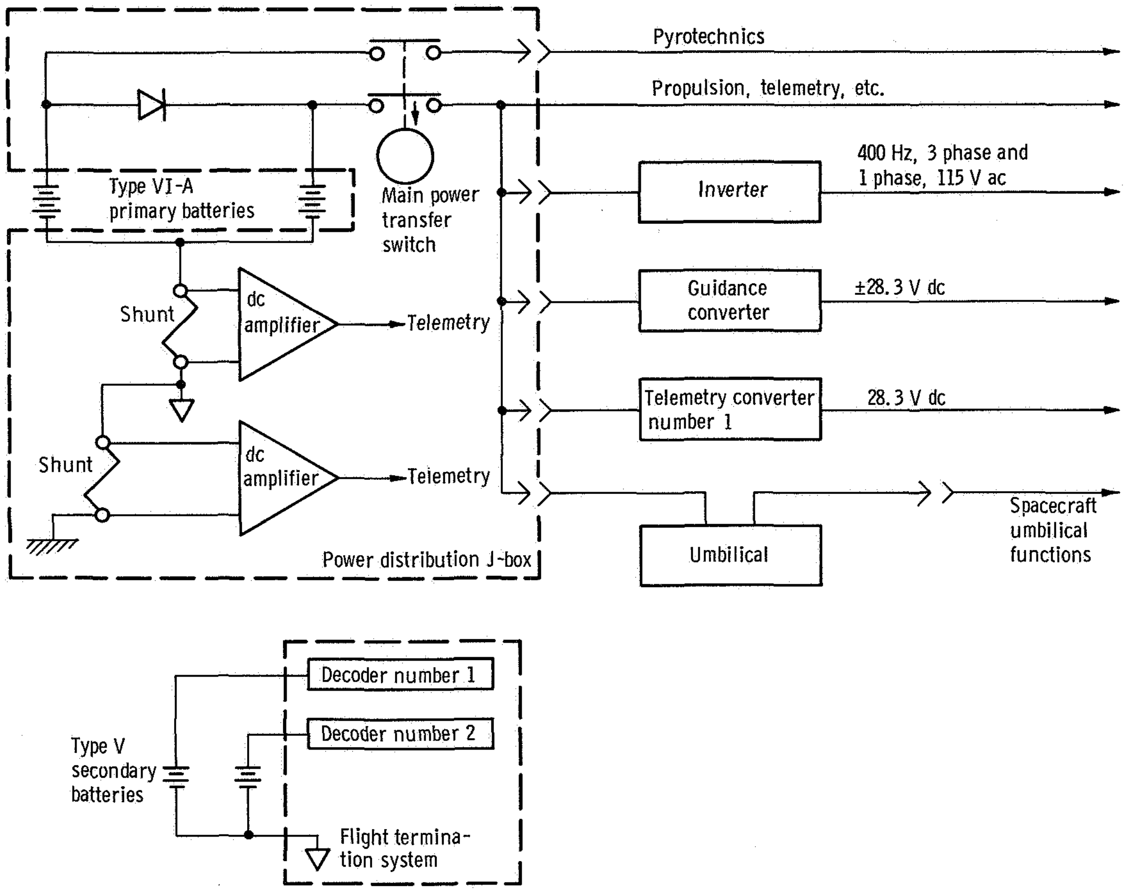


Figure VI-7. - Agena electrical system, ATS-1.

GUIDANCE AND FLIGHT CONTROL

Description

The Agena guidance and flight control system consists of three subsystems: guidance, flight control, and flight programmer. This system functions after Atlas-Agena separation.

The guidance subsystem consists of an inertial reference package, horizon sensors, and velocity meter. Primary attitude reference is provided by three orthogonal, rate-integrating gyroscopes in the inertial reference package. The infrared horizon sensor provides continuous corrections in pitch and roll to the inertial reference package. Yaw attitude reference is obtained from the booster and after separation is corrected by gyrocompassing techniques during the coast period after Agena engine first burn. Vehicle longitudinal acceleration is sensed by the accelerometer in the velocity meter system. The velocity meter counter generates a signal to terminate engine thrust for both the first and second burn when the vehicle has increased its velocity by predetermined increments.

The flight control subsystem, consists of a flight control electronics unit, a pneumatic control system, a hydraulic control system, and a flight control junction box. Attitude error signals from the inertial reference package are conditioned and amplified by the flight control electronics to operate the control systems. During Agena coast periods, the pneumatic control system provides roll, pitch, and yaw control by use of six thrusters operating on a mixture of nitrogen and Freon. During Agena engine burn, pitch and yaw control is provided by two hydraulic actuators that gimbal the Agena engine thrust chamber; roll control is provided by the pneumatic system. A patch panel in the flight control junction box provides the means for interconnecting the desired functions of the guidance and flight control subsystems to suit specific mission requirements.

The ATS-1 flight programmer subsystem used two sequence timers to program the Agena flight events. The primary sequence timer is started by the radio guidance start Agena timer discrete command. The second (auxiliary) sequence timer is started by the primary timer.

Performance

Analysis of flight data shows that Atlas-Agena separation induced low rates in the Agena. These vehicle rates were 0.1 degree per second yaw left, 0.075 degree per second roll clockwise, and 0.95 degree per second pitch up. (Clockwise and counterclockwise roll reference applies when looking forward along the Agena longitudinal axis, see fig. VI-8). At the time of pneumatic control system activation, the indicated required corrections for the vehicle were 0.4^{o} yaw left, 0.05^{o} roll counterclockwise, and 1.5^{o} pitch up. Within 4.5 seconds, these indicated attitude corrections were reduced by the pneumatic system to within the threshold limits of $\pm 0.2^{o}$ in pitch, $\pm 0.18^{o}$ in yaw, and $\pm 0.6^{o}$ in roll. The vehicle then completed the programmed pitch down of 10^{o}, after which the programmed geocentric rate of 3.21 degrees per minute pitch down was applied. The pitch horizon sensor was set at a pitch bias angle of 5.10^{o} nose up required for Agena first burn. The vehicle was in the process of stabilizing in pitch at the time of Agena engine first start. Vehicle rates of 0.18 degree per second pitch up and 0.80 degree per second roll clockwise existed at the time of engine first start. No yaw errors were discernible from either rate integrating gyro or pneumatic thruster data.

At Agena engine first start, the gas generator turbine spin-up, coupled with the existing low clockwise roll rate, produced a roll rate that reached a maximum of 0.95 degree per second clockwise. This rate induced a maximum roll displacement 1.6^{o} clockwise. Pneumatic thrusters reduced the clockwise roll rate to zero in 1.6 seconds and attempted to return the vehicle to the roll reference attitude. The vehicle overshot in roll 1.5^{o} counterclockwise due to a counterclockwise torque induced by thrust through the misalined turbine exhaust duct. The vehicle subsequently returned to the edge of the roll threshold limit.

Hydraulic pressure buildup and off-center engine gimbal position induced vehicle transient responses about the pitch and yaw axes. Peak overshoots of 0.3^{o} yaw left and 0.5^{o} pitch down were evidenced approximately 4 seconds after engine start; however, the vehicle was stabilized within 8 seconds after Agena engine first start.

Shroud jettison (initiated 10 sec after engine first-start ignition) occurred during the portion of the 1.5^{o} counterclockwise roll displacement when the vehicle roll rate had reached a minimum (approximately zero). Gyro data in roll and pitch indicate that little or no attitude error was introduced by shroud jettison. At the same time the horizon sensors indicated a slight disturbance which was attributable to the shroud halves passing through the field of view of the horizon sensors.

Agena engine first shutdown was commanded by the velocity meter after the vehicle attained the required velocity increment of 7231.49 feet per second (2204.2 m/sec). The additional velocity increment due to engine decay was 6.9 feet per second (2.1 m/sec). An expected roll transient, caused by Agena engine first shut down resulted in a 2.4^{o} roll counterclockwise excursion, which was reduced to within the pneumatic control system threshold limits in 5 seconds.

The pneumatic control system was transferred to the low pressure mode; the programmed geocentric pitch rate was set to -4.5 degrees per minute; and the horizon sensor pitch bias angle was decreased to zero. The horizon sensor and pneumatics data show that the vehicle maintained the proper attitude during the coast phase with minimum pneumatic thruster activity.

Agena engine second start induced roll rates that resulted in a maximum roll attitude displacement of 1.8° clockwise. These rates were satisfactorily damped by pneumatic control. The vehicle assumed a counterclockwise roll offset of 0.2° in the same manner as that of first burn. The Agena engine start transient resulted in a yaw actuator peak overshoot of 0.1° yaw left, which returned to zero in a normal time span.

Agena engine second shutdown was commanded by the velocity meter after the vehicle attained the required velocity increment of 8078.4 feet per second (2462.3 m/sec). Thrust decay increased the velocity 27 feet per second (8.2 m/sec) prior to velocity meter counter disable. As expected, second burn shutdown transient of 2.8° roll counterclockwise occurred and was reduced to within the threshold limits in 11 seconds.

The horizon sensors were disconnected from the gyros, and the vehicle responded to 9.36° pitch-up and a 57° yaw-left commands, as programmed. The Agena was stable within the threshold limits of the pneumatic control system at the time of spacecraft separation.

After spacecraft separation, the vehicle completed a programmed yaw-right maneuver (nominal 237°). There was essentially no pneumatic activity required after the yaw maneuver.

The pneumatic control system gas usage was less than anticipated. A comparison of the predicted and actual gas usage is given in table VI-I.

TABLE VI-I. - AGENA PNEUMATIC CONTROL SYSTEM, ATS-1

Flight sequence	Units	Predicted	Actual
Lift-off	lb	29.7 load	29.3 load
	kg	13.5	13.3
Agena first coast	lb	5.7 usage	3.2 used
	kg	2.6	1.4
Agena engine first burn	lb	4.0 usage	4.2 used
	kg	1.8	1.9
Parking orbit coast	lb	0.6 usage	0.4 used
	kg	0.3	0.18
Agena engine second burn	lb	2.4 usage	2.1 used
	kg	1.1	0.95
Prespacecraft separation maneuver	lb	1.1 usage	0.9 used
	kg	0.5	0.4
Postspacecraft separation maneuver	lb	0.3 usage	0.3 used
	kg	0.14	0.14
Loss of telemetry signal	lb	15.6 surplus	18.2 surplus
	kg	7.1	8.3

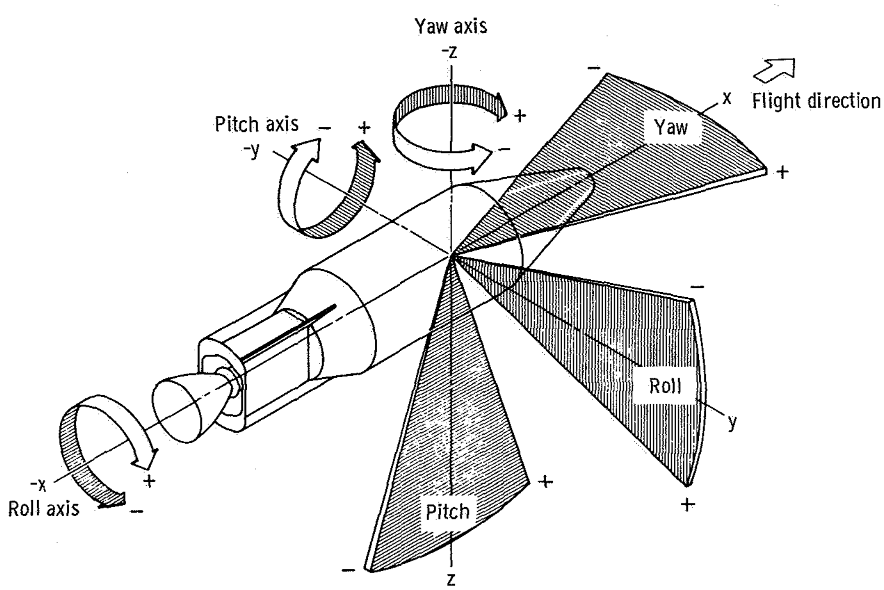


Figure VI-8. - Agena vehicle axes and vehicle movement designations, ATS-1.

COMMUNICATIONS AND CONTROL

Description

The Agena communication and control system consists of telemetry, tracking, and flight termination subsystems.

The telemetry subsystem is mounted in the forward section. It monitors and transmits the Agena functional and environment condition measurements during ascent. The Pulse Amplitude Modulation/Frequency Modulation/Frequency Modulation (PAM/FM/FM) telemetry unit contains a very high frequency (VHF) transmitter, voltage controlled oscillators, a commutator, a switch and calibrate unit, and a dc-dc converter. The transmitter operates at a frequency of 244.3 megahertz at a power output of 10 watts. The telemetry system has the capacity for 18 subcarriers (using the standard Interrange Instrumentation Group (IRIG) subcarrier channels) but only 9 were used. Channels 15 and 16 are commutated at 5 revolutions per second with 60 segments on each channel. Figure VI-9 is a block diagram of the Agena telemetry system.

The telemetry instrumentation consists of 58 transducers and event monitors. Five accelerometers use individual continuous subcarrier channels. The input current to the attitude control system thruster valves is monitored on one continuous subcarrier channel. The velocity meter accelerometer and velocity meter counter time share one continuous subcarrier channel. The turbine speed signal does not require a subcarrier oscillator in the telemetry package because it directly modulates the transmitter during engine operation. The remaining 49 measurements are monitored on the two commutated subcarrier channels. The transducers are powered by a regulated 28-volt dc power supply located in the telemetry unit. A summary of the instrumentation flown is given in appendix B.

The tracking subsystem includes a C-band beacon transponder, radiofrequency switch, and antenna. The transponder received coded signals from the tracking radar on a carrier frequency of 5690 megahertz. It transmits coded responses on a carrier frequency of 5765 megahertz at a minimum peak 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 interrogating stations and the number of stations interrogating the vehicle at any one time. The radiofrequency switch connects the output of the C-band beacon transponder to either the umbilical for ground checkout or the antenna for flight.

The flight termination subsystem provides a range safety flight termination capability for the Agena from lift-off through Agena engine first cutoff. This subsystem consists of two receiver-decoders which are coupled to two antennas by a multicoupler, two batteries, two destruct initiators, and a destruct charge. These units, except for the multicoupler and the destruct charge are connected to provide redundant flight termination capability. Flight termination, if necessary, is initiated by a series of commands from the range safety transmitter located on the ground. The destruct charge, located near the fuel-oxidizer bulkhead, ruptures both propellant tanks. The resultant mixing of the hypergolic propellants destroys the vehicle.

Performance

The telemetry subsystem performance was satisfactory throughout the flight. Telemetry stations at Cape Kennedy and Antigua monitored the Agena first-burn phase and the Range Instrumentation Ship Twin Falls and Ascension monitored the Agena second-burn phase. All stations recorded a good telemetry signal strength of at least 80 microvolts throughout these phases. The Antigua signal strength data verified that telemetry was turned off at T + 753.3 seconds, as programmed. The ship Twin Falls signal strength data verified that telemetry was turned back on at T + 1110.3 seconds. A description of the tracking and data acquisition network used in support of the ATS-1 flight is given in appendix C.

The in-flight operating period of the telemetry subsystem was 1820.2 seconds. During flight operation, the temperature of the telemetry subsystem transmitter rose from a lift-off temperature of 85° F (302.5 K) to 117° F (320 K). Figure VI-10 shows that the highest rate of temperature rise during flight was approximately 0.0223° F per second (0.01228 K/sec). If this rate of increase is assumed to be a worst case condition, the transmitter could have operated for a period of 3578 seconds before reaching its maximum specification temperature of 160° F (344.5 K).

Usable data were obtained from all vehicle measurements. Flight data of channels 17 and 18 indicate that the wire harnesses for Longitudinal Vibrometer Measurement A520 and Radial Vibrometer Measurement A524 were interchanged. The flight dynamic data shown in appendix D reflect the "as-flown" configuration. The longitudinal vibrometer and the radial vibrometer were located near a spacecraft separation pyrotechnic device. The shock resulting from the firing of these pyrotechnics caused both transducers to saturate their amplifiers. Approximately 10 seconds were required to damp the resulting 2-hertz sinusoidal transient. Subsequent data indicated that the vibrometers were not damaged.

The tracking subsystem performance was satisfactory throughout the flight. The C-band beacon was turned off, as programmed, for a period of 351 seconds during the Agena coast phase between first and second burn.

The flight termination subsystem functioned satisfactorily during prelaunch tests and flight. The signal strength at the airborne receiver remained well above the threshold of 2 microvolts until the flight termination subsystem was disabled at T + 544.3 seconds.

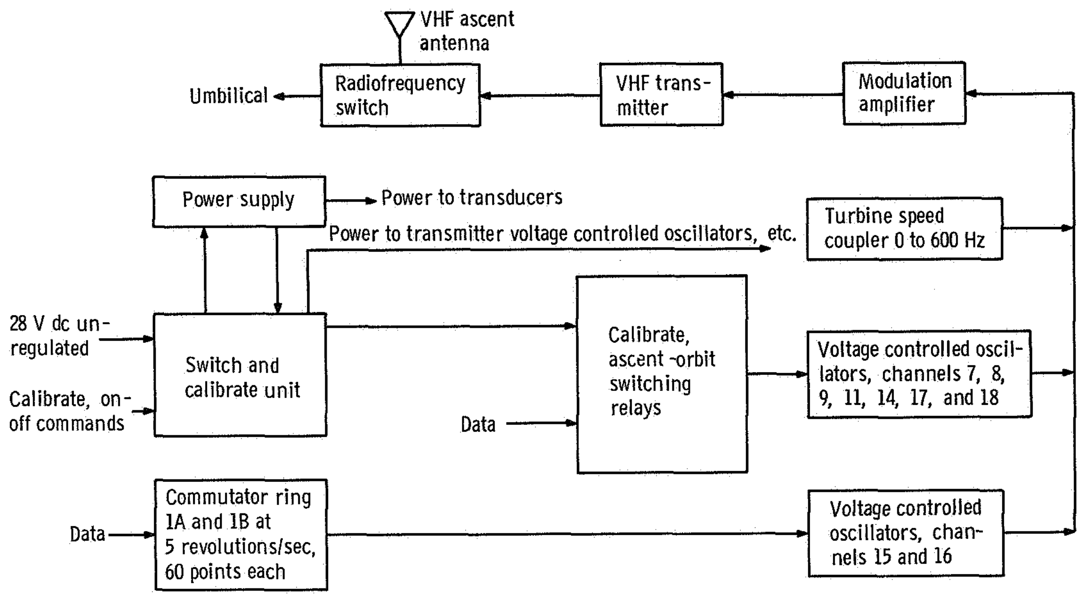


Figure VI-9. - Agena telemetry system, ATS-1.

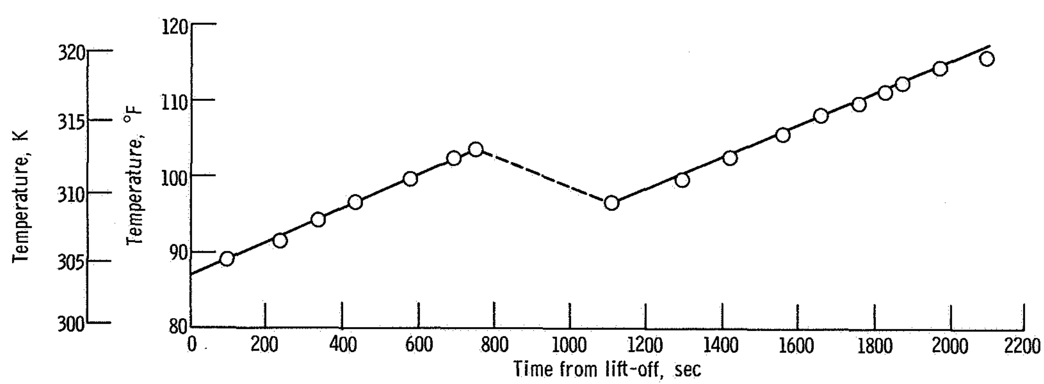


Figure VI-10. - Agena telemetry transmitter temperature, ATS-1. Agena measurement number H-218.

VII. LAUNCH OPERATIONS

PRELAUNCH ACTIVITIES

A calendar of major activities at the Eastern Test Range is shown in table VII-I. All prelaunch tests were completed satisfactorily. Only minor operational and ground equipment problems were encountered while conducting these tests. These problems were as follows:

(1) Atlas erection was delayed 1 day due to a misalinement of the vehicle and the launcher.

(2) Atlas dual propellant loading was attempted November 7 with the following problems occurring:

(a) A leak developed in the fuel pump inlet transducer boss on the B-1 Atlas engine. This problem was corrected by replacing the transducer boss seal.

(b) A leak occurred at the aft flange of the fuel staging valve requiring the replacement of the flange seal.

(c) The fuel drain disconnect on the B-2 booster engine developed a leak and was replaced.

After the preceding discrepancies were corrected, the test was successfully conducted on November 8.

(3) During the radiofrequency interference testing November 28, high current readouts were noted in the Launch Operations Building immediately after power application to the Agena. The excessive currents resulted from an electrical short caused by a ground equipment wiring error. This caused a diode to be damaged in the Agena flight control junction box necessitating the replacement of the junction box.

(4) During the Joint Flight Acceptance Test (J-FACT) December 1, the transmission of C-band beacon signals through the new umbilical hardline connections to the Agena resulted in low signal strength readout in Hangar E. Because of this low signal readout, the C-band beacon signal transmission was performed using the old "hat" coupler system during countdown for launch. Replacement of a portion of the umbilical tower coaxial cable after the launch corrected the problem. The removed cable had a low resistance between the shield and the primary conductor. The low resistance caused an excess signal loss and the low signal strength readout.

COUNTDOWN AND LAUNCH

The Atlas-Agena vehicle was successfully launched on the first attempt. The launch vehicle countdown started at T - 425 minutes, about 1307 hours, eastern standard time, December 6, 1966. The only holds were 55 minutes at T - 60 minutes and 5 minutes at T - 7 minutes, as planned.

Lift-off occurred at 2112:00.876 eastern standard time. The 2-inch- (5.08-cm-) motion switch malfunctioned at lift-off. Consequently, the 8-inch- (20.32-cm-) motion switch was used to calculate the 2-inch (5.08-cm) lift-off T - 0 time.

All other aerospace ground equipment systems operated satisfactorily during countdown and lift-off.

TABLE VII-I. - PRELAUNCH ACTIVITIES, ATS-1

Date, 1966	Event
September 21	Atlas arrival
October 12	Atlas erection
October 17	Agena arrival
October 27	Booster Flight Acceptance Test (B-FACT) Number 1
October 31	Spacecraft arrival
November 8	Atlas Dual Propellant Loading (DPL) Test
November 16	Booster Flight Acceptance Test (B-FACT) Number 2
November 23	Vehicle systems and aerospace ground equipment (AGE) compatibility checks completed
November 28	Radio Frequency Interference (RFI) Test
November 29	Joint Flight Acceptance Composite Test (J-FACT)
December 1	Joint Flight Acceptance Composite Test (J-FACT) - without spacecraft
December 2	Simulated Launch Demonstration (SLD)

VIII. CONCLUDING REMARKS

Launch on time was demonstrated when the Atlas-Agena vehicle lifted off within 2 seconds after the optimum scheduled time within the launch window. All Atlas and Agena systems performed satisfactorily. The Standard Agena Clamshell shroud, which was flown for the first time, demonstrated satisfactory aerodynamic shielding for the spacecraft during ascent through the atmosphere. The final Agena-spacecraft transfer orbit parameters were well within the requirements established for the ATS-1 mission.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, November 8, 1968,
491-05-00-02-22.

APPENDIX A

SEQUENCE OF EVENTS

Nominal time, sec	Actual time, sec	Event description	Source	Event monitor (a)
0	0	Lift-off (2112:00.876 EST)		2-in.- (5.08-cm-) motion switch (M3CX)[b]
129	129.1	Atlas booster cutoff	Atlas guidance	Longitudinal accelerometer (U101A)
293.1	293.0	Atlas sustainer cutoff		Longitudinal accelerometer (U101A)
295.8	288.2	Start Agena timer		Guidance and control monitor (D14)
312.8	312.9	Atlas vernier cutoff		Longitudinal accelerometer (U101A)
		Uncage Agena gyros		
		Jettison horizon sensor fairing		
		Arm Atlas-Agena separation		
315	315.1	Atlas-Agena separation		Guidance and control monitor (D14)
317.5	317.4	Activate pneumatics	Separation switch	
		Connect roll horizon sensor to roll gyro	Separation switch	
341.8	334.2	Initiate -120 deg/min pitch rate	Primary timer	Pitch torque rate (D73)
346.8	339.1	Transfer to -3.21 deg/min pitch rate		
		Connect pitch horizon sensor to pitch gyro		
		Connect velocity meter accelerometer output to telemetry		
366.8	359.2	Fire first-burn ignition squibs		Switch group Z (B31)
		Deactivate pitch and yaw pneumatics		
		Enable velocity meter		
368	360.4	Agena steady-state thrust		Chamber pressure (B91)
368.3	360.7	Fire helium pressure squibs		Chamber pressure (B91)
376.8	369.3	Fire shroud squibs		Shroud separation (A52)
511.8	504.3	Arm engine shutdown		(c)
		Arm command destruct disarm		(c)

[a]The first, second, third, and fifth events are monitored on Atlas telemetry; the remaining on Agena telemetry. The designation in parentheses is the monitor measurement designation. (See Atlas and Agena telemetry schedules, appendix B, for measurement range and channel assignment.)

[b]Lift-off occurred at 2112:01.046 EST as indicated by 8-in.- (20.32-cm-) motion switch. The 2-in.- (5.08-cm-) motion switch malfunctioned and a nominal time of 0.170 sec between 2- and 8-in. (5.08- and 20.32-cm) motion was used throughout this report for a computed 2-in. (5.08-cm) lift-off time of 2112:00.876 EST. All nominal trajectory times are referenced to 2-in.- (5.08-cm-) motion lift-off.

[c]No direct measurement for these events.

Nominal time, sec	Actual time, sec	Event description	Source	Event monitor (a)
528.1	521.7	Agena engine cutoff	Velocity meter	Velocity meter accelerometer-counter (D83/D88)
		Activate pitch and yaw pneumatics		
		Fire liquid-oxygen fast-shutdown squib		
534.8	527.2	Remove 28 V from pitch and yaw pneumatics	Primary timer	
		Start telemetry calibration		
		Connect velocity meter counter output to telemetry		
		Close propellant isolation valve		
551.8	544.3	Stop telemetry calibrate and dis-arm command destruct		Pitch torque rate (D73)
		Pneumatics to low pressure		
		Attenuate horizon sensor for pitch and roll		
		Start gyro compassing		
		Transfer to -4.46 deg/min pitch rate		
556.8	549.3	Fire horizon sensor zero degree squib		Velocity meter accelerometer-counter (D83/D88)
		Transfer to second burn		
		Velocity increment		
758.8	751.3	Fire close helium valve squib		(c)
		Start auxiliary timer		(c)
760.8	753.3	Stop primary timer	Auxiliary timer	Telemetry signal strength
		Turn telemetry off		
		Beacon power off		
1111.8	1104.3	Start primary timer		(c)
		Beacon power on		(c)
1117.8	1110.3	Turn telemetry on	Primary timer	Telemetry signal strength
1118.8	1111.3	Pneumatics to high pressure		(c)
		Remove gyro compassing		(c)
1175.8	1168.3	Connect accelerometer output to telemetry		Velocity meter accelerometer-counter (D83/D88)
		Enable velocity meter		
		Open propellant isolation valve		
1177.8	1170.3	Deactivate pitch and yaw pneumatics		Switch group Z (B13)
		Fire second-burn ignition squibs		Switch group Z (B13)

[a]The first, second, third, and fifth events are monitored on Atlas telemetry; the remaining on Agena telemetry. The designation in parentheses is the monitor measurement designation. (See Atlas and Agena telemetry schedules, appendix B, for measurement range and channel assignment.)

[c]No direct measurement for these events.

Nominal time, sec	Actual time, sec	Event description	Source	Event monitor (a)
1179	1171.4	Agena steady-state thrust	Primary timer	Chamber pressure (B91)
1253.8	1246.3	Arm engine shutdown	Primary timer	(c)
1258	1248.7	Agena engine cutoff	Velocity meter	Velocity meter accelerometer-counter (D83/D88)
		Pitch and yaw pneumatics on	Velocity meter	
1261.8	1254.3	Connect velocity meter counter output to telemetry	Primary timer	
		Start telemetry calibrate		
1266.8	1259.3	Stop telemetry calibrate		
		Connect accelerometer output to telemetry		
		Start auxiliary timer		
1334.8	1327.3	Transfer to 28.1 deg/min pitch rate	Auxiliary timer	Pitch torque rate (D73)
		Disconnect horizon sensor inputs from pitch and roll gyro		
1354.8	1347.3	Stop 28.1 deg/min pitch rate		
1364.8	1357.2	Initiate -180 deg/min yaw rate		Yaw torque rate (D51)
1383.8	1376.4	Stop -180 deg/min yaw rate		Yaw torque rate (D51)
1394.8	1387.4	Fire spacecraft separation squibs	Primary timer	Longitudinal vibration (A520)
1397.8	1390.3	Initiate 180 deg/min yaw rate		Yaw torque rate (D51)
1476.8	1469.3	Stop 180 deg/min yaw rate		Yaw torque rate (D51)
1476.8	1469.3	Initiate 3.7 deg/min pitch rate	Auxiliary timer	Pitch torque rate (D73)
		Connect horizon sensor inputs to pitch and roll gyro		Pitch torque rate (D73)
1496.8	1489.3	Pneumatics to low pressure		Yaw torque rate (D51)
		Attenuate horizon sensor gains in pitch and roll gyro compassing		
2124.8	2117.3	Pneumatics to high pressure		
		Remove gyro compassing		
2184.8	2177.3	Remove all power from vehicle except C-band beacon		Telemetry signal strength

[a]The first, second, third, and fifth events are monitored on Atlas telemetry; the remaining on Agena telemetry. The designation in parentheses is the monitor measurement designation. (See Atlas and Agena telemetry schedules, appendix B, for measurement range and channel assighment.)

[c]No direct measurement for these events.

APPENDIX B

LAUNCH VEHICLE INSTRUMENTATION SCHEDULE

TABLE B-I. - ATLAS TELEMETRY, ATS-1

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
A743T	Ambient temperature at sustainer instrumentation panel	11-41	-50° to 550° F	227.5 to 561 K
A745T	Ambient temperature at sustainer fuel pump	11-45	-50° to 550° F	227.5 to 561 K
D1V	Range safety command cutoff output	5-S	(b)	
D1V	Range safety command cutoff output	15-1	0 to 5 V dc	
D7V	Number 1 range safety command radiofrequency input automatic gain control	15-1	0 to 10 000 μV	
D3X	Range safety command destruct output	16-S	0 to 6 V dc	
E28V	Main dc voltage	18-1/31	20 to 35 V dc	
E51V	400 Hz ac phase A	18-11	105 to 125 V dc	
E52V	400 Hz ac phase B	18-29	105 to 125 V ac	
E53V	400 Hz ac phase C	18-41	105 to 125 V ac	
E95V	28 V dc guidance power input	13-15	20 to 35 V dc	
E96V	115 V ac 400 Hz phase A to guidance	13-37	105 to 125 V ac	
E151V	400 Hz phase A waveform	10	0 to 150 V ac	
F1P	Liquid-oxygen tank helium pressure (absolute)	15-9	0 to 50 psi	0 to 34.5 N/cm^2
F3P	Fuel tank helium pressure (absolute)	15-11	0 to 100 psi	0 to 68.9 N/cm^2
F116P	Differential pressure across bulkhead	18-13/43	0 to 25 psi	0 to 17.2 N/cm^2
F125P	Booster control pneumatic regulator output pressure (absolute)	13-21	0 to 1000 psi	0 to 689 N/cm^2
F246P	Booster tank helium bottle pressure (absolute)	13-55	0 to 3500 psi	0 to 2413 N/cm^2
F288P	Start pneumatic regulator output	13-1	0 to 800 psi	0 to 551.5 N/cm^2
F291P	Sustainer control helium bottle	13-3	0 to 3500 psi	0 to 2413 N/cm^2
F247T	Booster tank helium bottle temperature	11-31	-400° to 250° F	33.5 to 116.5 K
G4C	Pulse beacon magnetron average current	15-15	0 to 5 V dc	
G82E	Rate beacon radiofrequency output	15-17	0 to 5 V dc	
G3V	Pulse beacon automatic gain control	15-19	0 to 5 V dc	
G279V	Rate beacon automatic gain control number 1	15-21	0 to 5 V dc	

[a]First number indicates the Interrange Instrumentation Group subcarrier channel used; second number indicates commutated position for measurement. If no second number is indicated, channel was used continuously for designated measurement.

[b]Items determined from a step change in voltage.

TABLE B-I. - Continued. ATLAS TELEMETRY, ATS-1

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
G280V	Rate beacon automatic gain control number 2	15-13	0 to 5 V dc	
G282V	Rate beacon phase detector number 1	15-45	0 to 5 V dc	
G287V	Decoder pitch output	15-47	0 to 5 V dc	
G288V	Decoder yaw output	15-49	0 to 5 V dc	
G296V	Pulse beacon 15 V dc power supply	13-9	0 to 5 V dc	
G298V	Decoder 10 V dc power supply	13-13	0 to 5 V dc	
G354V	Rate beacon 25 to 30 V dc power supply	13-11	0 to 5 V dc	
G590V	Discrete binary 1	16-33	0 to 5 V dc	
G591V	Discrete binary 2	16-35	0 to 5 V dc	
G592V	Discrete binary 4	16-37	0 to 5 V dc	
G593V	Discrete binary 8	16-39	0 to 5 V dc	
H3P	Booster hydraulic pump discharge pressure (absolute)	13-41	0 to 3500 psi	0 to 2413 N/cm^2
H33P	B1 hydraulic accumulator pressure (absolute)	15-31	0 to 3500 psi	0 to 2413 N/cm^2
H130P	Sustainer hydraulic pump discharge pressure (absolute)	15-33	0 to 3500 psi	0 to 2413 N/cm^2
H140P	Sustainer-vernier hydraulic pressure (absolute)	15-35	0 to 3500 psi	0 to 2413 N/cm^2
H224P	Booster hydraulic system low pressure (absolute)	15-7	0 to 600 psi	0 to 413.6 N/cm^2
H601P	Sustainer hydraulic return line	18-7/37	0 to 600 psi	0 to 413.6 N/cm^2
M79A	Vehicle axial accelerometer fine	7	-0.5 to 0.5 g's	
M30X	Vehicle 2-in. (5.08-cm) motion	7-S	(b)	
M32X	Jettison system command	5-S	(b)	
P83B	Booster 2 pump speed	15-41	4000 to 7000 rpm	
P84B	Booster 1 pump speed	4	6000 to 6950 rpm	
P349B	Sustainer pump speed	3	9.9 to 11.2 kpm	
P529D	Sustainer main liquid-oxygen valve	13-43	0^o to 90^o	
P830D	Sustainer fuel valve position	13-35	0^o to 30^o	
P1P	Booster 1 liquid-oxygen pump inlet pressure (absolute)	18-9	0 to 150 psi	0 to 103.4 N/cm^2
P2P	Booster 1 fuel pump inlet pressure (absolute)	13-31	0 to 100 psi	0 to 68.9 N/cm^2
P6P	Sustainer thrust chamber pressure (absolute)	18-3/33	0 to 1000 psi	0 to 689 N/cm^2
P26P	Booster liquid-oxygen regulator reference pressure (absolute)	13-17	500 to 1000 psi	344.7 to 689 N/cm^2
P27P	Vernier fuel tank pressure (absolute)	13-39	0 to 1000 psi	0 to 689 N/cm^2
P28P	Vernier 1 thrust chamber pressure (absolute)	18-15	0 to 400 psi	0 to 275.8 N/cm^2
P29P	Vernier 2 thrust chamber pressure (absolute)	18-17	0 to 400 psi	0 to 275.8 N/cm^2

[a]First number indicates the Interrange Instrumentation Group subcarrier channel used; second number indicates commutated position for measurement. If no second number is indicated, channel was used continuously for designated measurement.

[b]Items determined from a step change in voltage.

TABLE B-I. - Continued. ATLAS TELEMETRY, ATS-1

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
P30P	Vernier liquid-oxygen tank pressure (absolute)	13-53	0 to 1000 psi	0 to 689 N/cm^2
P47P	Vernier 1 liquid-oxygen inlet pressure (absolute)	13-45	0 to 600 psi	0 to 413.7 N/cm^2
P49P	Vernier 1 fuel inlet pressure (absolute)	13-49	0 to 600 psi	0 to 413.7 N/cm^2
P55P	Sustainer fuel pump inlet pressure (absolute)	13-5	0 to 100 psi	0 to 68.9 N/cm^2
P56P	Sustainer liquid-oxygen pump inlet pressure (absolute)	18-5	0 to 150 psi	0 to 103.4 N/cm^2
P59P	Booster 2 thrust chamber pressure (absolute)	18-19	0 to 800 psi	0 to 551.6 N/cm^2
P60P	Booster 1 thrust chamber pressure (absolute)	18-21	0 to 800 psi	0 to 551.6 N/cm^2
P100P	Gas generator combustion chamber pressure (absolute)	15-51	0 to 600 psi	0 to 413.7 N/cm^2
P330P	Sustainer fuel pump discharge pressure (absolute)	15-55	0 to 1500 psi	0 to 1034.2 N/cm^2
P339P	Sustainer gas generator discharge pressure (absolute)	18-55	0 to 800 psi	0 to 551.6 N/cm^2
P344P	Sustainer liquid-oxygen regulator reference pressure (absolute)	13-19	500 to 1000 psi	344.7 to 689 N/cm^2
P15T	Engine compartment air temperature	11-35	-50° to 550° F	227.5 to 561 K
P16T	Engine compartment component temperature	11-55	0° to 400° F	255.5 to 477.5 K
P117T	Booster 2 fuel pump inlet temperature	11-53	0° to 100° F	255.5 to 311 K
P530T	Sustainer liquid-oxygen pump inlet temperature	11-1	-300° to 270° F	89 to 105.5 K
P671T	Thrust section ambient temperature, Quadrant IV	11-15	-50° to 550° F	227.5 to 561 K
P77X	Vernier cutoff relay	8-S	(b)	
P347X	System cutoff relay	8-S	(b)	
P616X	Booster flight lock-in	16-19	(b)	
S61D	Roll displacement gyro signal	15-29	-3° to 3°	
S62D	Pitch displacement gyro signal	15-37	-3° to 3°	
S63D	Yaw displacement gyro signal	15-39	-3° to 3°	
S252D	Booster 1 yaw roll	16-15	-6° to 6°	
S253D	Booster 2 yaw roll	16-55	-6° to 6°	
S254D	Booster 1 pitch	7	-6° to 6°	
S255D	Booster 2 pitch	16-1	-6° to 6°	
S256D	Sustainer yaw	16-41	-4° to 4°	
S257D	Sustainer pitch	16-45	-4° to 4°	
S258D	Vernier 1 pitch roll	16-3	-70° to 70°	
S259D	Vernier 2 pitch roll	16-5	-70° to 70°	

[a]First number indicates the Interrange Instrumentation Group subcarrier channel used; second number indicates commutated position for measurement. If no second number is indicated, channel was used continuously for designated measurement.

[b]Items determined from a step change in voltage.

TABLE B-I. - Concluded. ATLAS TELEMETRY, ATS-1

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
S260D	Vernier 1 yaw	16-7	-5° to 55°	
S261D	Vernier 2 yaw	16-9	-5° to 55°	
S52R	Roll rate gyro signal	9	-8 to 8 deg/sec	
S53R	Pitch rate gyro signal	8	-6 to 6 deg/sec	
S54R	Yaw rate gyro signal	5	-6 to 6 deg/sec	
S190V	Pitch gyro torque amplifier	15-43	-1 to 1 V ac	
S209V	Programmer 28 V dc test	6	20 to 35 V dc	
S236X	Booster cutoff discrete	9-S	(b)	
S241X	Sustainer cutoff discrete	9-S	(b)	
S245X	Vernier cutoff discrete	9-S	(b)	
S248X	Release payload discrete	9-S	(b)	
S290X	Programmer output	16-29	0 to 28 V dc	
	Spare			
	Booster jettison			
	Enable discretes			
S291X	Programmer output	16-31	0 to 28 V dc	
	Booster engine cutoff			
	Sustainer engine cutoff			
	Vernier engine cutoff			
S359X	Booster staging backup	5-S	(b)	
S384X	Spin motor test output	15-5	0 to 5 V dc	
U101A	Axial acceleration	12	0 to 10 g's	
U80P	Liquid-oxygen tank head pressure (differential)	16-11	0 to 5 psi	0 to 3.4 N/cm^2
U81P	Fuel tank head pressure (differential)	16-13	0 to 2.5 psi	0 to 1.7 N/cm^2
U112V	Acoustica counter output	15-23/53	0 to 5 V dc	
U113V	Acoustica valve position feedback	13-33	0 to 5 V dc	
U132V	Acoustica station counter output	13-7	0 to 5 V dc	
U134V	Acoustica time shared oscillator output	18-23/53	0 to 5 V dc	
U135V	Acoustica sensor signal	18-39	0 to 5 V dc	
U605V	Acoustica time shared integrator switch	18-35	0 to 5 V dc	
Y44P	Interstage adapter pressure (absolute)	13-23	0 to 15 psi	0 to 10.3 N/cm^2
Y45T	Interstage adapter temperature	11-5	-200° to 200° F	144 to 366.5 K
Y41X	Start D timer	5-S	(b)	
H103	Safe-arm-fire destruct number 2	16-4	(b)	
H204	dc-dc converter number 1	15-50	22 to 30 V dc	
H218	Telemetry transmitter temperature	16-49	50° to 170° F	283 to 350 K
H354	Destruct receiver number 1 signal level	16-6	0 to 40 μV	
H364	Destruct receiver number 2 signal level	16-8	0 to 40 μV	

[a] First number indicates the Interrange Instrumentation Group subcarrier channel used; second number indicates commutated position for measurement. If no second number is indicated, channel was used continuously for designated measurement.

[b] Items determined from a step change in voltage.

TABLE B-II. - AGENA TELEMETRY, ATS-1

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
A4	Tangential accelerometer	9	-10 to 10 g's	
A5	Tangential accelerometer	11	-10 to 10 g's	
A9	Longitudinal accelerometer	8	-4 to 12 g's	
A52	Shroud separation	15-44	(b)	
A226	Shroud inside temperature	16-31	32° to 500° F	273 to 533 K
A227	Shroud inside temperature	16-33	32° to 500° F	273 to 533 K
A519	Shroud cavity pressure (differential	16-12/23/34/53	-5 to 5 psi	-3.4 to 3.4 N/cm^2
A520	Spacecraft adapter longitudinal vibration	[c]17	-20 to 20 g's	
A524	Spacecraft adapter radial vibration	[c]18	-20 to 20 g's	
B1	Fuel pump inlet pressure (gage)	15-15	0 to 100 psi	0 to 68.9 N/cm^2
B2	Oxidizer pump inlet pressure (gage)	15-17	0 to 100 psi	0 to 68.9 N/cm^2
B11	Oxidizer venturi inlet pressure (absolute)	15-19/49	0 to 1500 psi	0 to 1034.2 N/cm^2
B12	Fuel venturi inlet pressure (absolute)	15-23/53	0 to 1500 psi	0 to 1034.2 N/cm^2
B13	Switch group Z	15-7/22/37/52	(b)	
B31	Fuel pump inlet temperature	15-6	0° to 100° F	255 to 311 K
B32	Oxidizer pump inlet temperature	15-8	0° to 100° F	255 to 311 K
B35	Turbine speed	(d)	(d)	
B91	Combustion chamber pressure number 3 (gage)	15-4/34	475 to 550 psi	327.5 to 379.2 N/cm^2
C1	28 V dc unregulated supply	16-40	22 to 30 V dc	
C3	28 V dc regulator	15-12	22 to 30 V dc	
C4	28 V dc unregulated current	16-13/44	0 to 100 A	
C5	-28 V dc regulator	15-30	-30 to -22 V dc	
C21	Inverter temperature	15-14	0° to 200° F	255 to 367 K
C31	Inverter phase AB	15-18	90 to 128 V ac	
C32	Inverter phase BC	15-20	90 to 128 V ac	
C38	Structure current monitor	15-10/25/40/55	0 to 50 A	
C141	Pyrotechnic bus voltage	15-5/35	22 to 30 V dc	
D14	Guidance and control monitor	16-27	(b)	
D41	Horizon sensor pitch	16-45	-5 to 5 deg	
D42	Horizon sensor roll	16-46	-5 to 5 deg	

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

[b]Items determined from a step change in voltage.

[c]Preflight plan had intended channels 17 and 18 to be switched from those shown here; however, flight data indicate that A524 was transmitted over channel 18 and A520 was transmitted over channel 17.

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

TABLE B-II. - Concluded. AGENA TELEMETRY, ATS-1

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
D46	Gas valve cluster temperature 1	15-39	-50° to 150° F	228 to 339 K
D47	Gas valve cluster temperature 2	15-36	-50° to 150° F	228 to 339 K
D51	Yaw torque rate (ascent mode)	16-38	-200 to 200 deg/min	
D51	Yaw torque rate (orbital mode)	16-38	-10 to 10 deg/min	
D54	Horizon sensor head temperature (right head)	15-47	-50° to 200° F	228 to 366.5 K
D55	Horizon sensor head temperature (left head)	15-46	-50° to 200° F	227.7 to 366.5 K
D59	Control gas supply high pressure (absolute)	16-47	0 to 4000 psi	0 to 275.8 N/cm^2
D60	Hydraulic oil pressure (gage)	15-21	0 to 4000 psi	0 to 275.8 N/cm^2
D66	Roll torque rate (ascent mode)	16-41	-50 to 50 deg/min	
D66	Roll torque rate (orbital mode)	16-41	-4 to 4 deg/min	
D68	Pitch actuator position	15-3	-2.5 to 2.5 deg	
D69	Yaw actuator position	15-24	-2.5 to 2.5 deg	
D70	Control gas supply temperature	15-42	-50° to 200° F	227.7 to 366.5 K
D72	Pitch gyro output (ascent mode)	16-36	-10 to 10 deg	
D72	Pitch gyro output (orbital mode)	16-36	-5 to 5 deg	
D73	Pitch torque rate (ascent mode)	16-35	-200 to 200 deg/min	
D73	Pitch torque rate (orbital mode)	16-35	-10 to 10 deg/min	
D74	Yaw gyro output (ascent mode)	16-39	-10 to 10 deg	
D74	Yaw gyro output (orbital mode)	16-39	-5 to 5 deg	
D75	Roll gyro output (ascent mode)	16-42	-10 to 10 deg	
D75	Roll gyro output (orbital mode)	16-42	-5 to 5 deg	
D83	Velocity meter acceleration	14	Binary code	
D86	Velocity meter cutoff switch	16-28	(b)	
D88	Velocity meter counter	14	Binary code	
D129	Inertial reference package internal case temperature	15-54	0° to 155° F	255.5 to 341.5 K
D149	Gas valves 1 to 6 current	7	(e)	
H47	Beacon receiver pulse repetition frequency	15-27	0 to 1600 pulses/sec	
H48	Beacon transmitter pulse repetition frequency	15-28	0 to 1600 pulses/sec	
H101	Safe-arm-fire destruct number 1	16-2	(b)	

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

[b] Items determined from a step change in voltage.

[e] A unique voltage level is associated with any one or combination of gas jet activity.

APPENDIX C

TRACKING AND DATA ACQUISITION

The ground trace of the launch vehicle trajectory is shown in figure C-1. Seven stations were used to obtain radar and telemetry data during launch vehicle operation. The Eastern Test Range stations at Cape Kennedy, Grand Bahama Island, Grand Turk Island, Antigua, Ascension, Pretoria, and the Range Instrumentation Ship Twin Falls were the stations supporting this flight.

Telemetry Data

The telemetry coverage provided by the Eastern Test Range station is presented in figure C-2. Real-time telemetry monitoring provided verification of Atlas and Agena significant flight events through Agena engine first burn. This was accomplished by utilizing the submarine cable which linked the Eastern Test Range uprange station to Cape Kennedy. The subsequent flight events were monitored by the downrange stations, and the time of occurrence reported back to Cape Kennedy in "near" real time by single side band radio links.

Radar Data

The radar coverage provided by the Eastern Test Range stations is presented in figure C-3. Real-time radar data were used for tracking the launch vehicle for range safety purposes and to assist the downrange stations in acquiring track of the vehicle. These data were also used for computation of parking-orbit parameters and injection conditions at Agena engine first cutoff and for transfer-orbit parameters and injection conditions at Agena engine second cutoff.

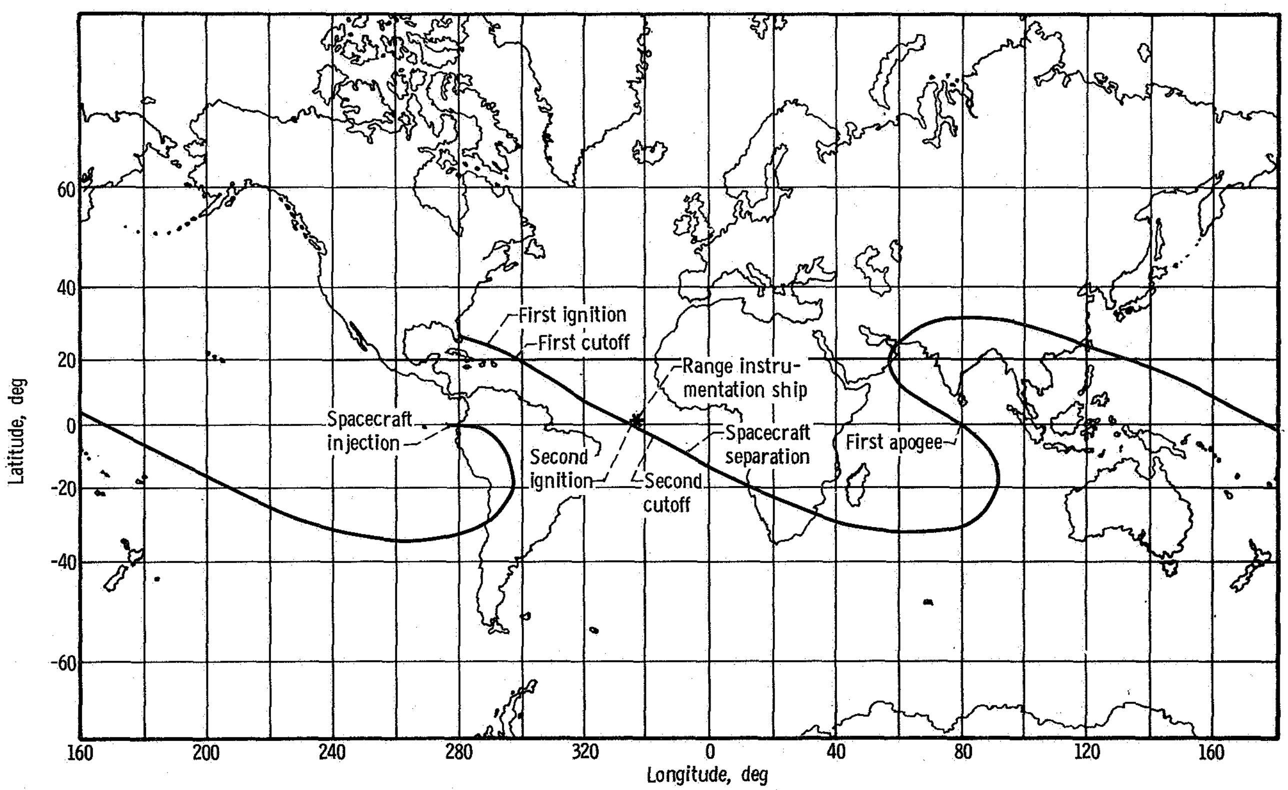


Figure C-1. - Ground trace for ATS-1.

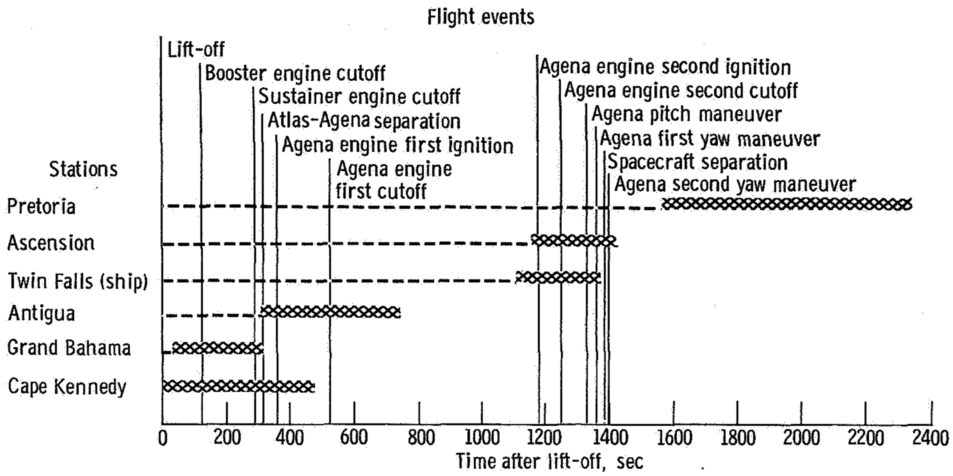


Figure C-2. - Launch vehicle telemetry converge, ATS-1.

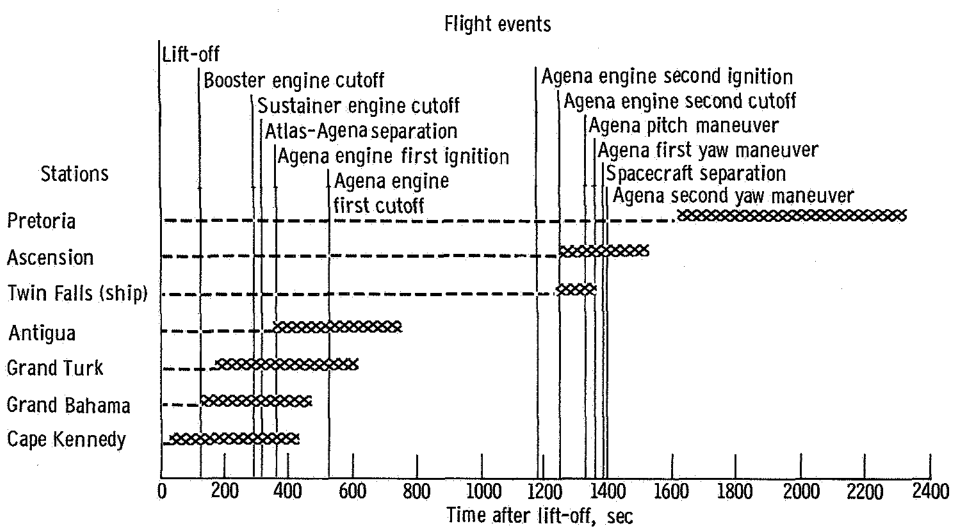


Figure C-3. - Launch vehicle radar coverage, ATS-1.

APPENDIX D

VEHICLE FLIGHT DYNAMICS

Flight dynamics data were obtained from three accelerometers installed in the Agena forward equipment section and two vibration transducers on the spacecraft adapter. A summary of dynamic instrumentation locations and characteristics is shown in figure D-1.

The flight data on channels 17 and 18 indicate that the wire harnesses for the vibrometer measurements were interchanged. For this flight, the longitudinal vibrometer data were transmitted over channel 17, and the radial vibrometer data were transmitted over channel 18.

Table D-I presents the actual flight times at which significant dynamic disturbances were recorded.

The measured flight environment did not show any unexpected excitation, and the experienced shock levels were within expected ranges. A summary of the flight dynamic data measurements is presented in table D-II. Dynamic data recorded for these flight events are shown in figures D-2 to D-12.

TABLE D-I. - SUMMARY OF DYNAMIC

DISTURBANCES, ATS-1

Event causing disturbance	Time of dynamic disturbance, sec
Lift-off	0
Transonic	T + 39 to T + 59
Booster engine cutoff	T + 129.3
Sustainer engine cutoff	T + 292.8
Horizon sensor fairing jettison	T + 312.67
Atlas-Agena separation	T + 314.9
Agena engine first burn	T + 360.4
Shroud separation	T + 369.13
Agena engine first cutoff	T + 521.6
Agena engine second burn	T + 1171.2
Agena engine second cutoff	T + 1248.6

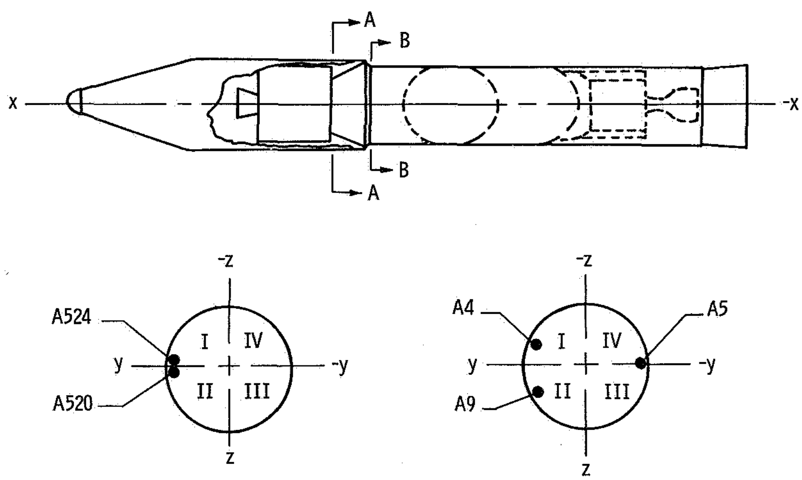
TABLE D-II. - SUMMARY OF FLIGHT DYNAMIC DATA, ATS-1

Event	Time of dynamic disturbance, sec	Accelerometer						Vibrometer			
		Channel 8		Channel 9		Channel 11		Channel 17		Channel 18	
		Measurement									
		A-9 Longitudinal		A-4 Tangential		A-5 Tangential		A-520 Longitudinal		A-524 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)	Frequency, Hz	g's (zero to peak)
Lift-off (2112:00.876 EST)	T - 0	5	0.2	6	0.3	6	0.3	1000	9.0	1200	6.0
Transonic	T + 39 to T + 59	100	.4	150	.5	200 to 400	.2	1000	14.0	1000	9.6
Booster engine cutoff	T + 129.3	12.0	.5	130	.5	120	.4	(a)	(a)	50	.3
Sustainer engine cutoff	T + 292.8	20	.3	(a)	(a)	(a)	(a)	(a)	(a)	50	.4
Horizon sensor fairing jettison	T + 312.67	0.040 sec pulse	6.0	100	0.6	200	2.1	2000	18.0	2000	13.0
Atlas-Agena separation	T + 314.9	0.020 sec pulse	3.0	120	.2	180	.4	900	22	900	19
Atlas engine first burn	T + 360.4	80	.4	55	.2	50	.3	700	1.8	55	.6
Shroud separation	T + 369.13	40	.4	80	.2	550	.4	1000	22	1000	23
Agena engine first cutoff	T + 521.6	70	.4	80	.5	[b]40 320	[b].3 .3	30 to 40	1.0	50	1.5
Agena engine second burn	T + 1171.2	75	.4	80	.5	130 to 240	.5	1200	2.0	50	1.8
Agena engine second cutoff	T + 1248.6	65	.4	45 to 120	.5	140	.7	900	3.0	50	2.0

[a]No measurable response.

[b]Double entries indicate frequencies and acceleration levels of two superimposed vibrations.

Channel	Measurement description	Measurement number	Frequency response, Hz	Range, g's	Transducer orientation
8	Longitudinal acceleration	A9 at station 247	0 to 35	-4 to 12	x-direction, quadrant II
9	Tangential acceleration	A4 at station 247	0 to 110	±10	z-direction, quadrant I
11	Tangential acceleration	A5 at station 247	0 to 160	±10	z-direction, quadrant III
18	Longitudinal vibrator	A520 at station 223	20 to 2000	±20	±x, + y-axis
17	Radial vibrator	A524 at station 223	20 to 1500	±20	±y, + y-axis



Section A-A, station 223

Section B-B, station 247

Figure D-1. - Flight instrumentation, ATS-1.

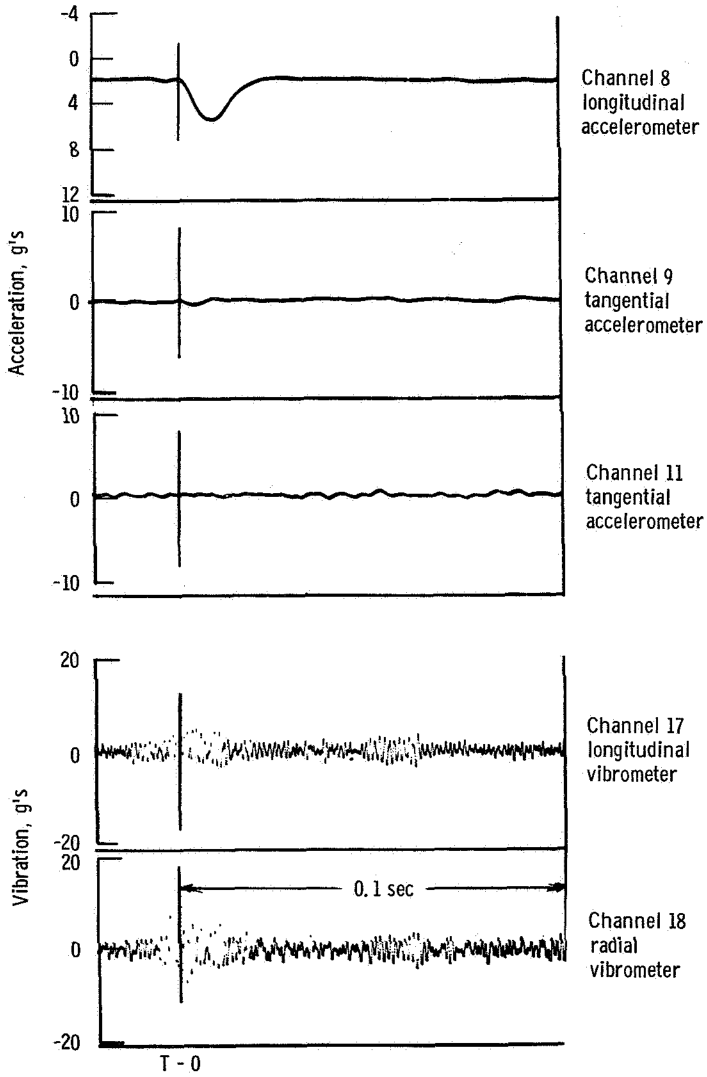


Figure D-2. - Dynamic data at lift-off, ATS-1.

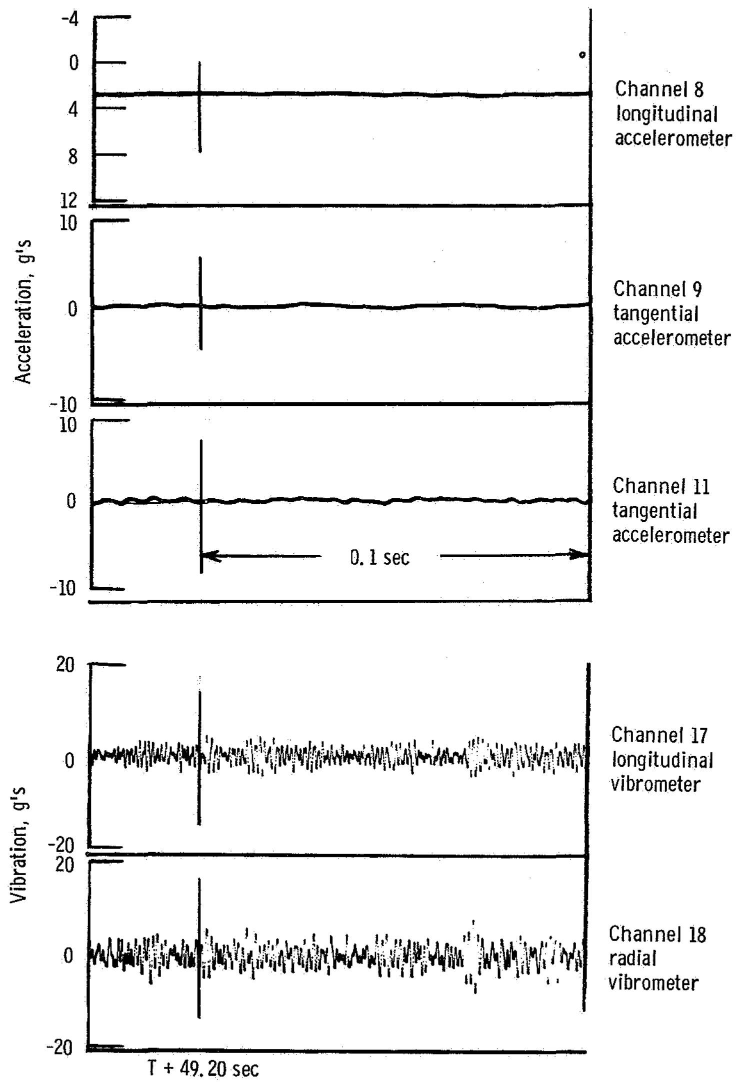


Figure D-3. - Dynamic data during transonic period, ATS-1.

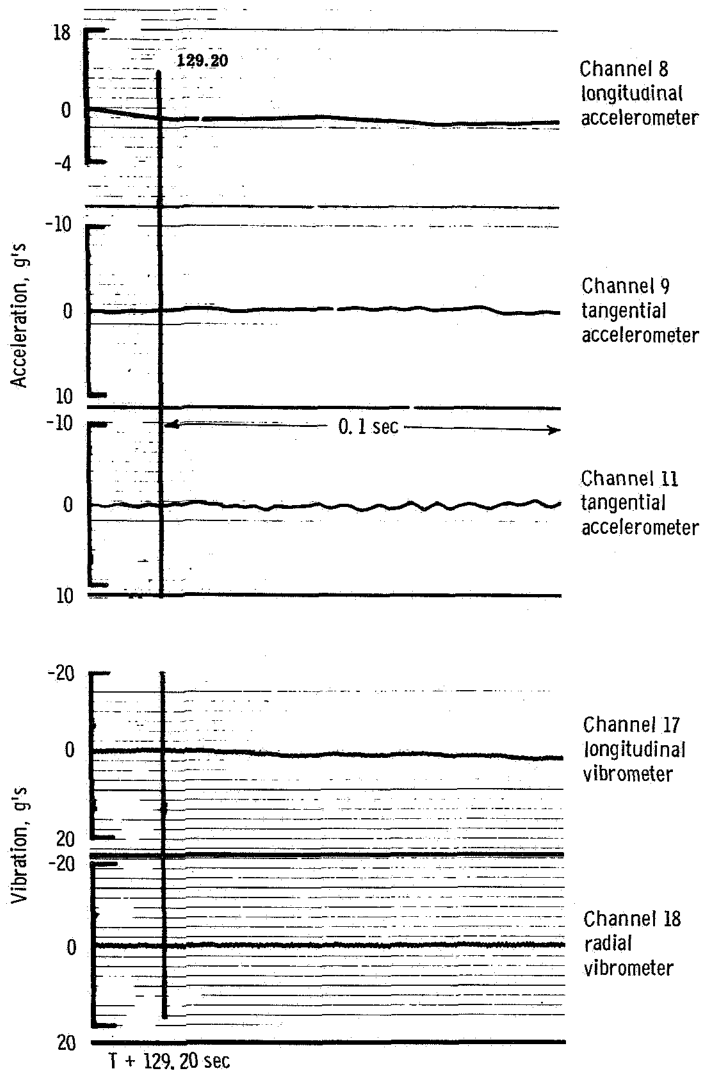


Figure D-4. - Dynamic data near time of booster engine cutoff, ATS-1.

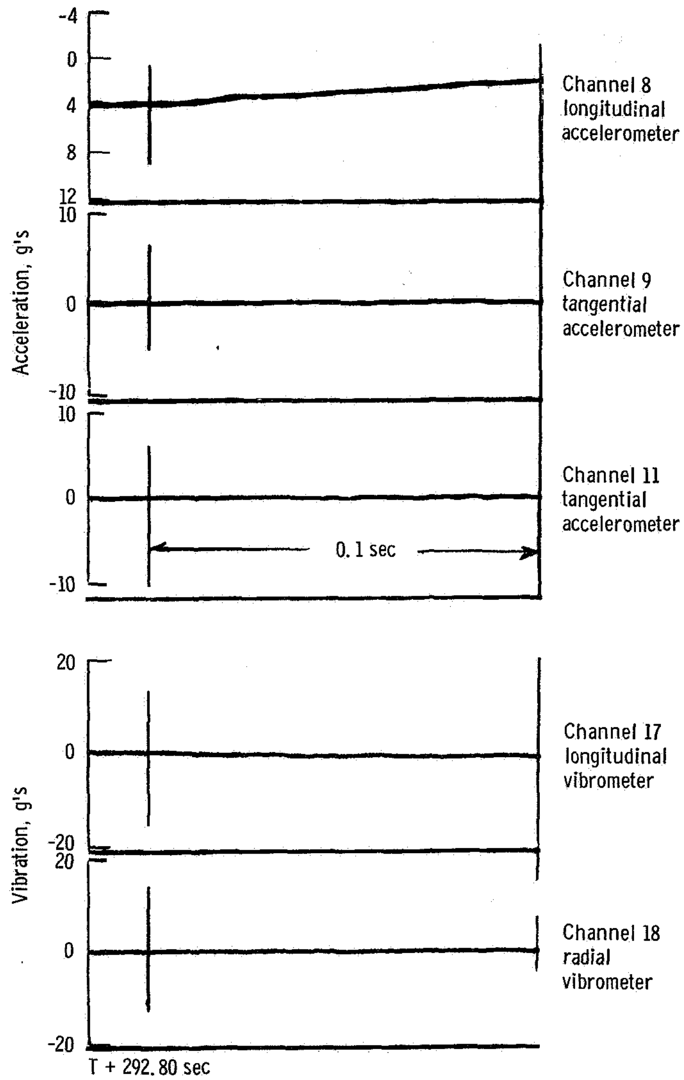


Figure D-5. - Dynamic data near time of sustainer engine cutoff, ATS-1.

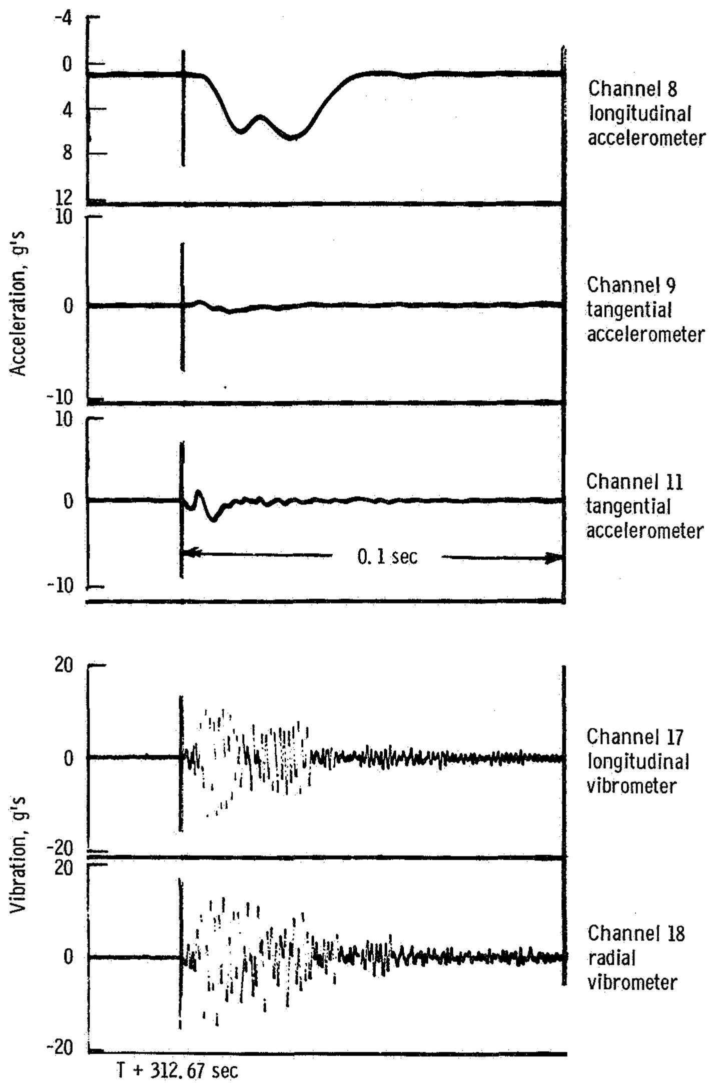


Figure D-6. - Dynamic data near time of horizon sensor fairing jettison, ATS-1.

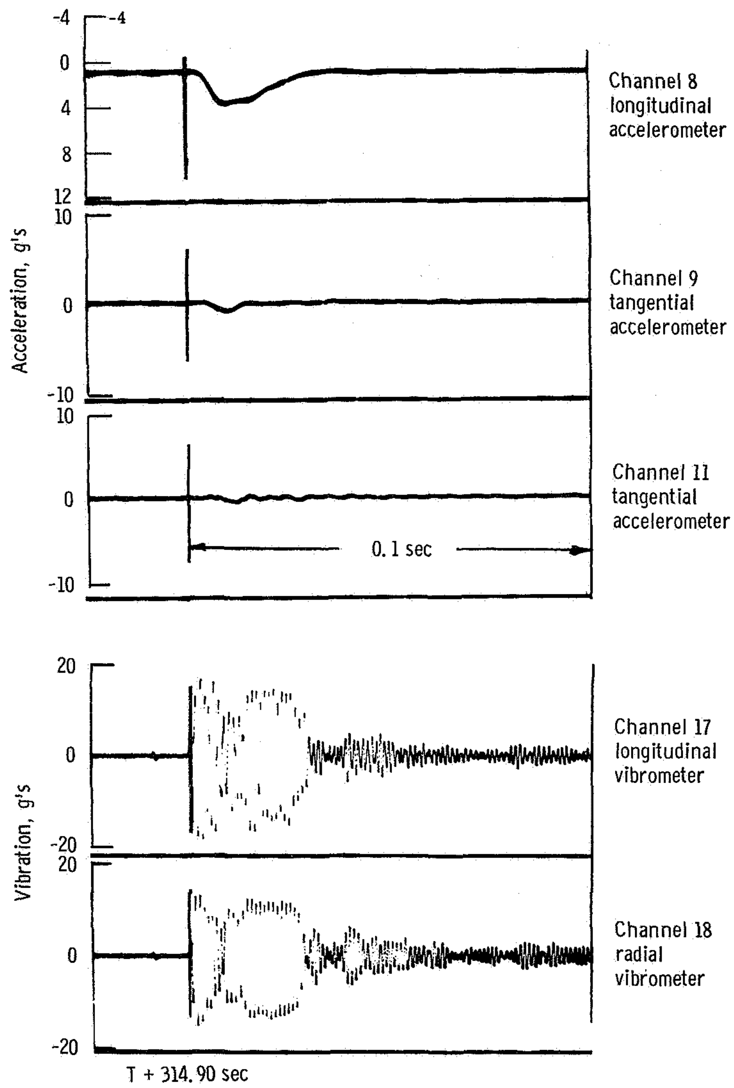


Figure D-7. - Dynamic data near time of Atlas-Agena separation, ATS-1.

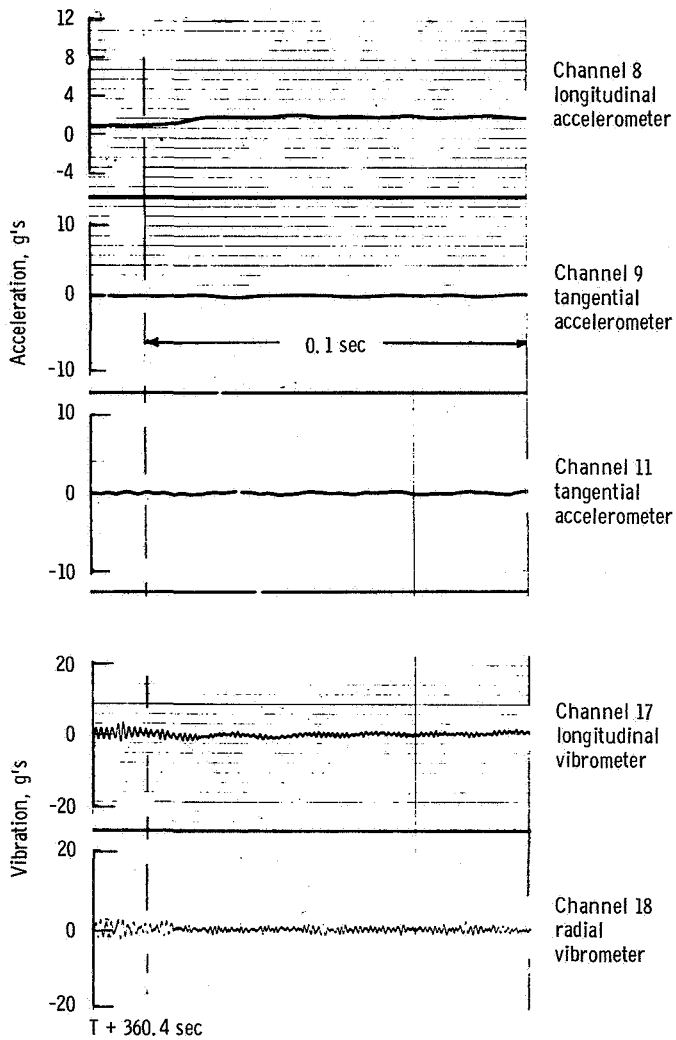


Figure D-8. - Dynamic data near time of Agena engine first burn, ATS-1.

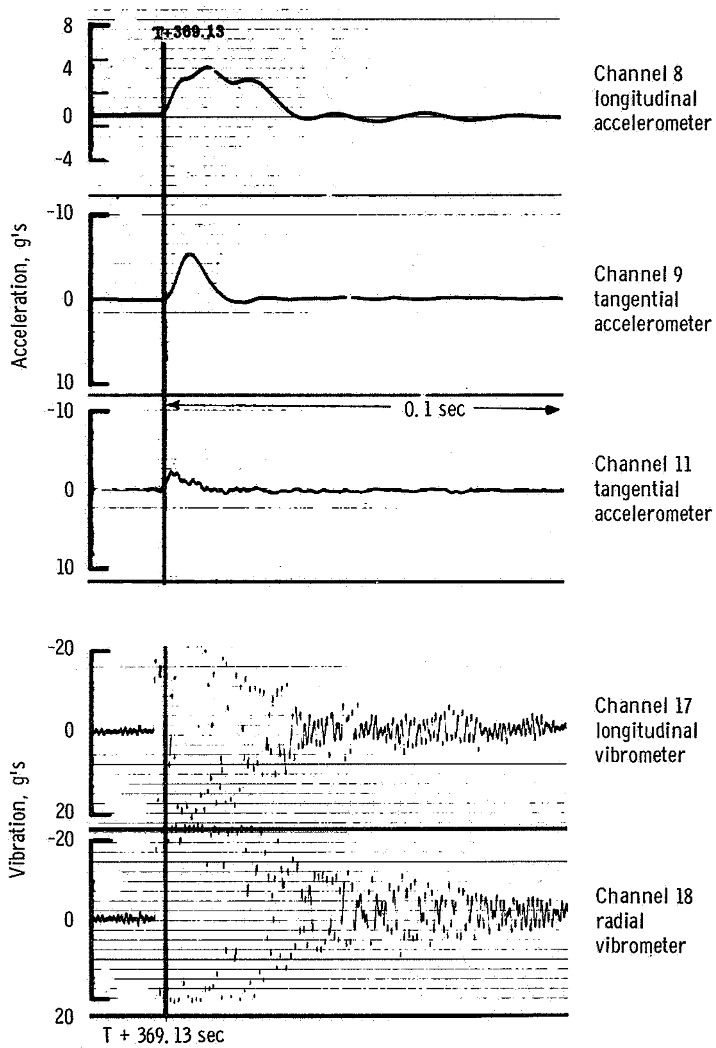


Figure D-9. - Dynamic data at shroud separation, ATS-1.

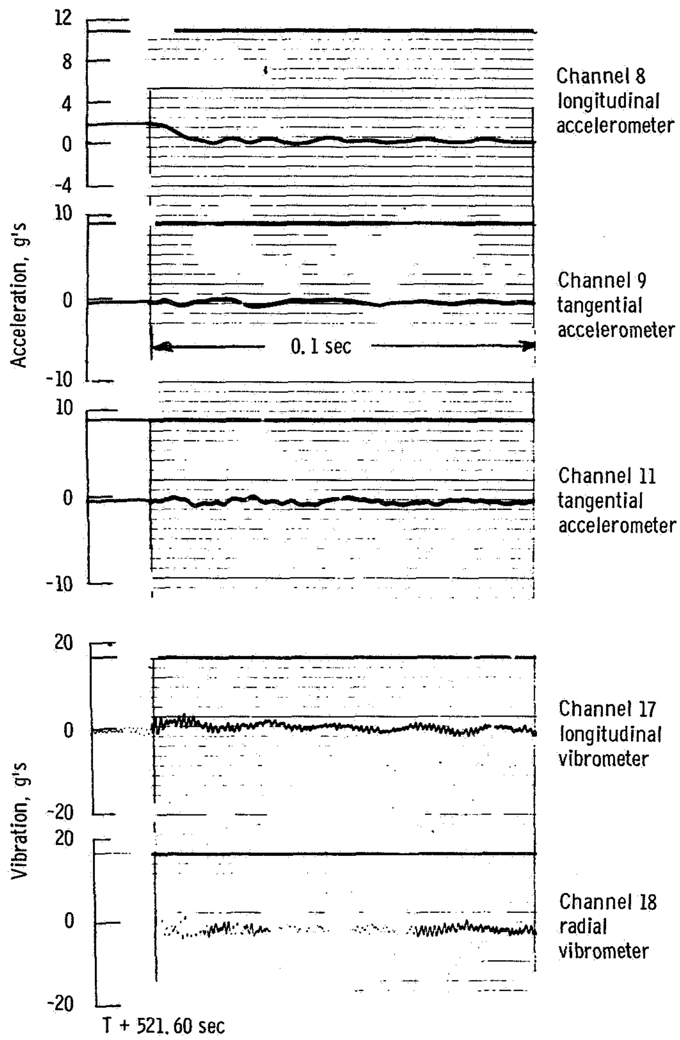


Figure D-10. - Dynamic data near time of Agena engine first cutoff, ATS-1.

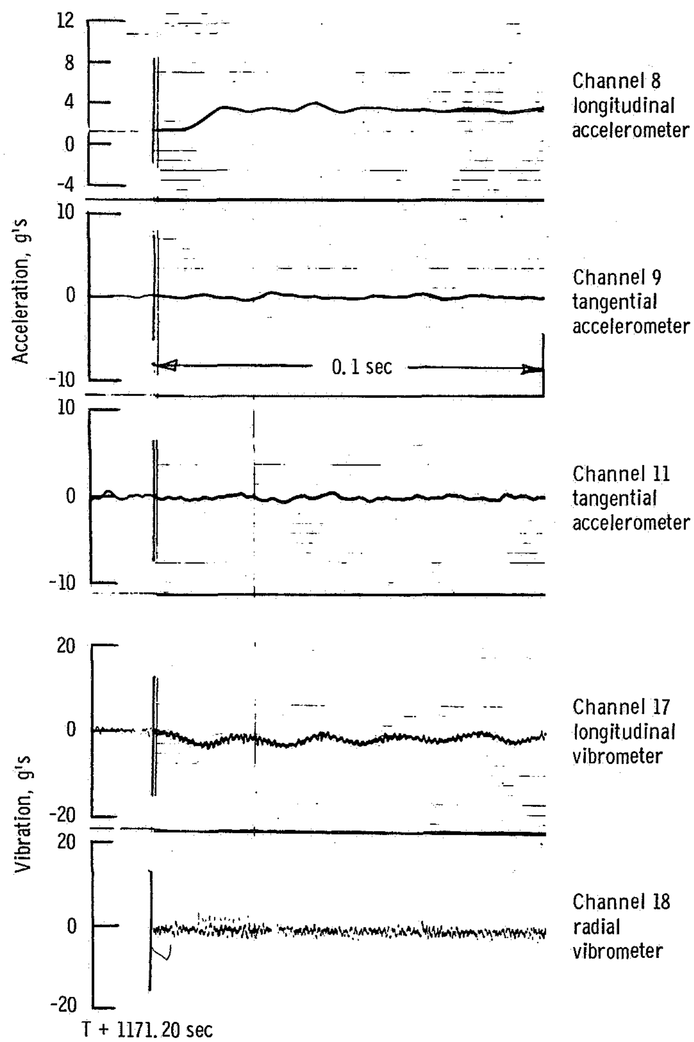


Figure D-11. - Dynamic data near time of Agena engine second burn, ATS-1.

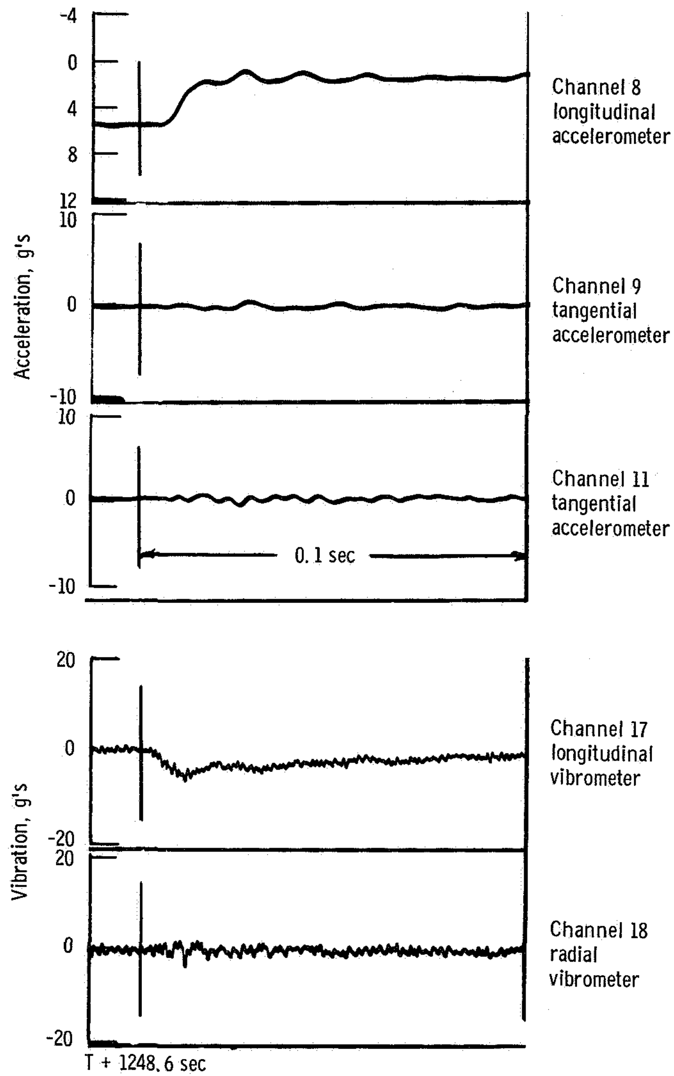


Figure D-12. - Dynamic data near time of Agena engine second cutoff, ATS-1.

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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