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ATLAS-AGENA FLIGHT PERFORMANCE FOR THE APPLICATIONS TECHNOLOGY SATELLITE ATS-2 MISSION

by Staff of the Lewis Research Center

Lewis Research Center
Cleveland, Ohio

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ABSTRACT

The Atlas-Agena launch vehicle boosted the Applications Technology Satellite ATS-2 into a transfer orbit with a perigee of 100 nautical miles (185 km) and an apogee of 6000 nautical miles (11 112 km). However, the Agena second burn did not occur, and the final circular orbit was not achieved. The Agena engine failed to restart for second burn because a propellant isolation valve malfunctioned. This report discusses the flight performance of the Atlas-Agena launch vehicle from lift-off through the Agena vehicle retromaneuver. A discussion of the propellant isolation valve failure is also included.

CONTENTS

I. SUMMARY

The NASA Applications Technology Satellite ATS-2 Atlas-Agena vehicle was launched from Complex 12, Eastern Test Range, on April 5, 1967, at 2223:01.901 eastern standard time, approximately 44 minutes after the scheduled lift-off time. The vehicle flight was satisfactory through the Atlas boost phase and the Agena first-burn phase. The Agena-spacecraft achieved the proper transfer orbit at the end of Agena first burn; however, the programmed Agena second burn did not occur. Consequently, the final spacecraft orbit was essentially the same as the transfer orbit, with a perigee of approximately 100 nautical miles (185 km) and an apogee of approximately 6000 nautical miles (11 112 km). The ATS-2 did not accomplish its primary mission in this orbit. However, the final orbit was satisfactory for the successful completion of the "heat pipe" experiment in Air Force Research Payload Module 481, a secondary payload carried on the Agena.

All aerospace ground equipment operated satisfactorily during the countdown and launch. A misinterpretation of facility water pressure data at T - 90 seconds resulted in an unscheduled countdown hold of 5.5 minutes and 5.5 minutes of additional count time. The Mod III Radio Guidance system performed satisfactorily during the flight. The tracking subsystem acquired the vehicle in the first acquisition cube as planned.

All Atlas systems performed satisfactorily throughout the flight. All Agena systems performed satisfactorily through the first-burn phase. At the end of the Agena first burn, the oxidizer propellant isolation valve on the Agena failed to fully close and as a result, the programmed Agena engine second burn did not occur.

The ATS-2 flight was the second to use the Standard Agena Clamshell shroud. The shroud performed successfully during flight.

This report contains an evaluation of the Atlas-Agena systems in support of the ATS-2 mission. The performance of each system is evaluated, and significant data are presented.

II. INTRODUCTION

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

Applications Technology Satellite ATS-2 was the second of a series of three ATS satellites to be launched with the Atlas-Agena vehicle. ATS-1 was launched in December 1966. This spin-stabilized spacecraft was placed in a synchronous altitude orbit and was designed for experimentation in communications, cloud cover photography, and spacecraft environment. The primary objective of the ATS-2 launch in April 1967 was to place the 704-pound (319-kg) satellite into a 6000-nautical-mile (11 112-km) circular orbit. An orbit of this kind is suitable for conducting planned experiments in gravity gradient stabilization, communications, and cloud cover photography. A secondary objective was to carry a United States Air Force Research Module 481 ("heat pipe" experiment) into a useful orbit (ref. 1).

The Atlas vehicle is used to boost the combined Agena-spacecraft into a suborbital coast ellipse. The Agena is designed so that the first burn places the Agena-spacecraft into a proper transfer orbit. The Agena second burn is then performed to circularize the final orbit, and the spacecraft is separated from the Agena.

This report presents an evaluation of the Atlas and Agena vehicle systems. The results of the ATS-2 launch are evaluated with the purpose of showing how the performance of the launch vehicle supported the objectives of ATS-2. The performance of the spacecraft and the Air Force Module is not evaluated.

III. VEHICLE DESCRIPTION

The launch vehicle for ATS-2 was a two-stage vehicle consisting of a standard Atlas (SLV-3) first stage and a standard Agena D second stage, along with a Standard Agena Clamshell shroud. The Atlas was 10 feet (3.048 m) in diameter, the Agena was 5 feet (1.52 m) in diameter, and the Standard Agena Clamshell shroud was about 5.5 feet (1.68 m) in diameter. The composite vehicle, including the shroud, was about 109 feet (33.22 m) long. The vehicle weight at lift-off was 278 051 pounds (126 121.2 kg). Figures III-1 to III-6 illustrate the arrangement of the composite vehicle: the Atlas, the composite Agena-shroud-spacecraft, the Atlas-Agena lifting off with ATS-2 the Standard Agena Clamshell shroud, and the ATS-2 spacecraft.

The first-stage Atlas (fig. III-2) was about 70 feet (21.34 m) long and was propelled by a standard Rocketdyne MA-5 propulsion system consisting of a booster engine having two thrust chambers with a total sea-level thrust of 330 000 pounds (1467.9×10^3 N); a sustainer engine with a sea-level thrust of 57 000 pounds (253.55×10^3 N); and two vernier engines, each with a sea-level thrust of 669 pounds (2.976×10^3 N). All engines used liquid oxygen and high-grade kerosene (RP-1) propellants. The booster, sustainer, and vernier engines were all ignited prior to lift-off for the booster phase of flight. During this phase, the booster engine thrust chambers were gimbaled for pitch, yaw, and roll control. At a predetermined acceleration level of about 6 g's, the booster phase of flight was terminated, the booster engine was shut down, and the booster engine section was jettisoned. The sustainer and vernier engines continued to burn for the 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. 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 to provide vehicle steering in preparation for Atlas-Agena separation. After vernier engine shutdown, the Agena was separated from the Atlas by firing a Mild Detonating Fuse severance system located around the booster adapter. The firing of two retrorockets attached to the booster adapter backed the Atlas away from the Agena and completed the separation of the stages.

The configuration of the Agena second stage is shown in figure III-3. This stage was about 20 feet (6.1 m) long and was propelled by an engine with a thrust of 16 000 pounds (71.171×10^3 N). This engine used unsymmetrical dimethylhydrazine (UDMH) and inhibited red fuming nitric acid (IRFNA) propellants. After Atlas-Agena separation, the Agena engine was ignited and burned until the vehicle was placed in the desired low-altitude orbit. During the burn period, the engine was gimbaled for pitch and yaw control, while a cold gas pneumatic system provided roll control. After engine shutdown, pitch, yaw, and roll control were provided by the pneumatic system. The Agena propulsion system was configured so that the engine could be started a second time. This second burn of the Agena engine was required to place the spacecraft in the proper circular orbit.

The 19-foot (5.79-m) Standard Agena Clamshell shroud (fig. III-5) was used to protect the spacecraft during ascent through the atmosphere. It was jettisoned during the Agena first-burn period. The ATS-2 spacecraft is shown in figure III-6.

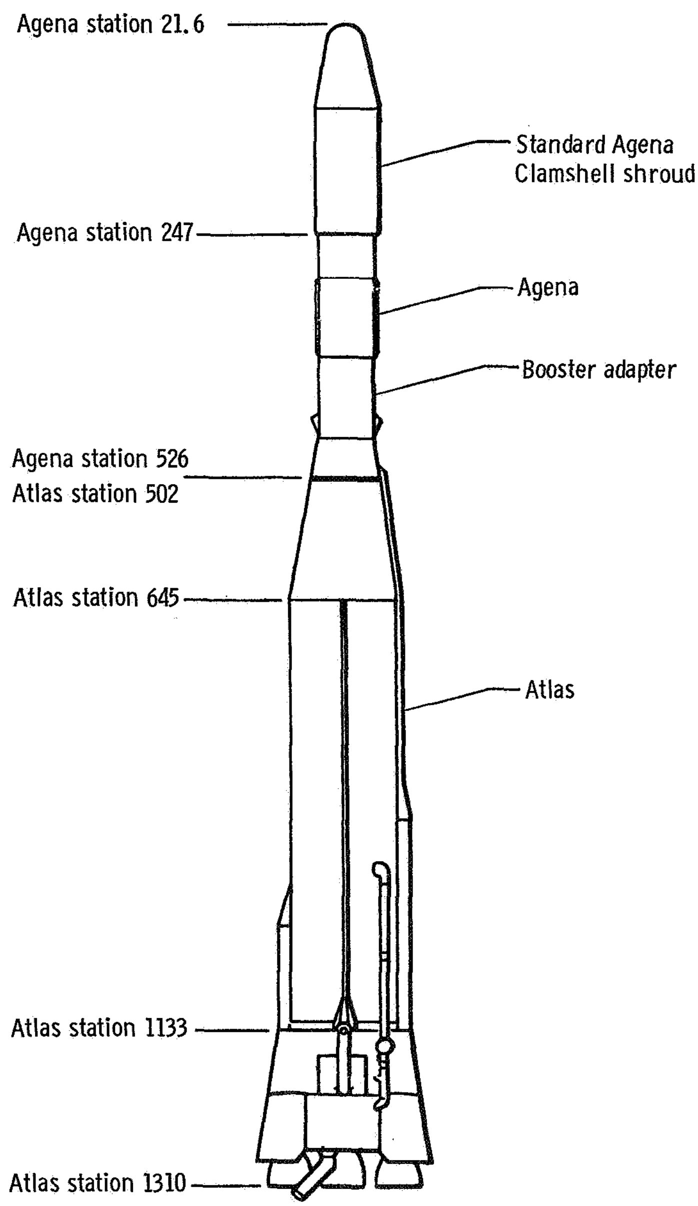


Figure III-1. - Space vehicle profile, ATS-2.

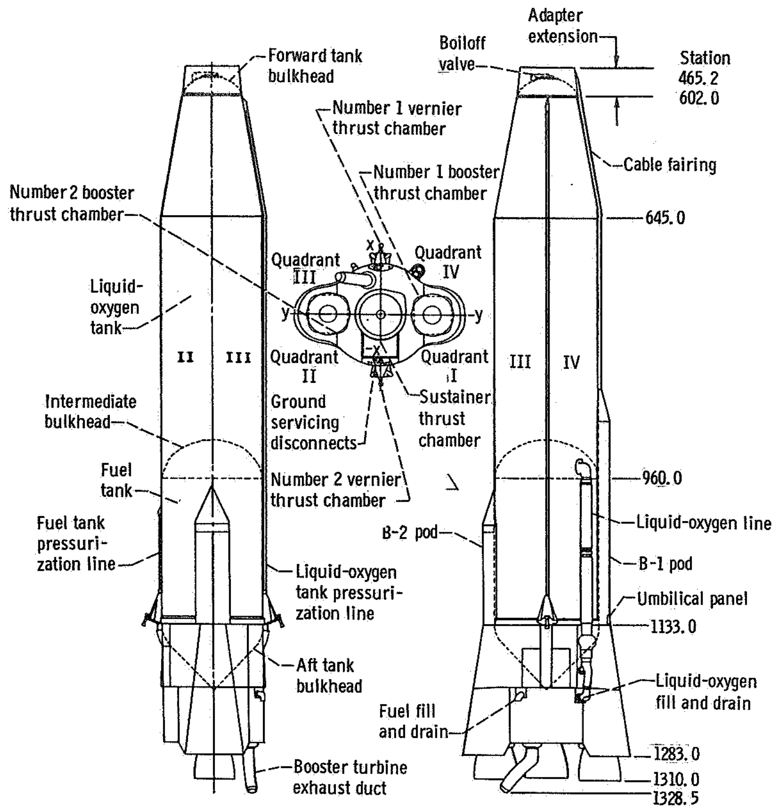


Figure III-2. - Atlas SLV-3 configuration, ATS-2.

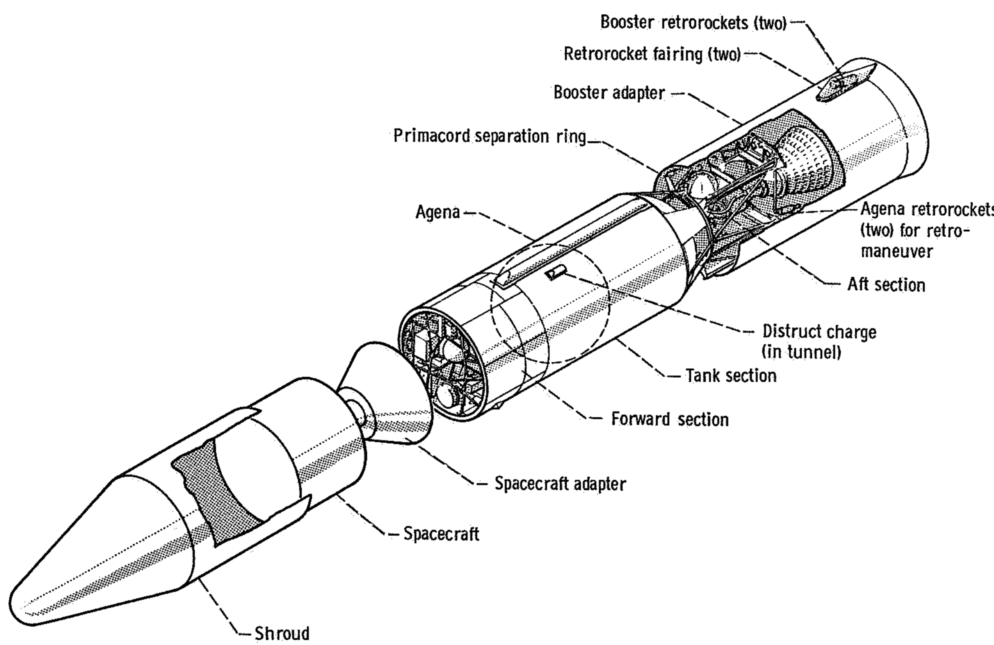


Figure III-3. - Agena-shroud-spacecraft, ATS-2.

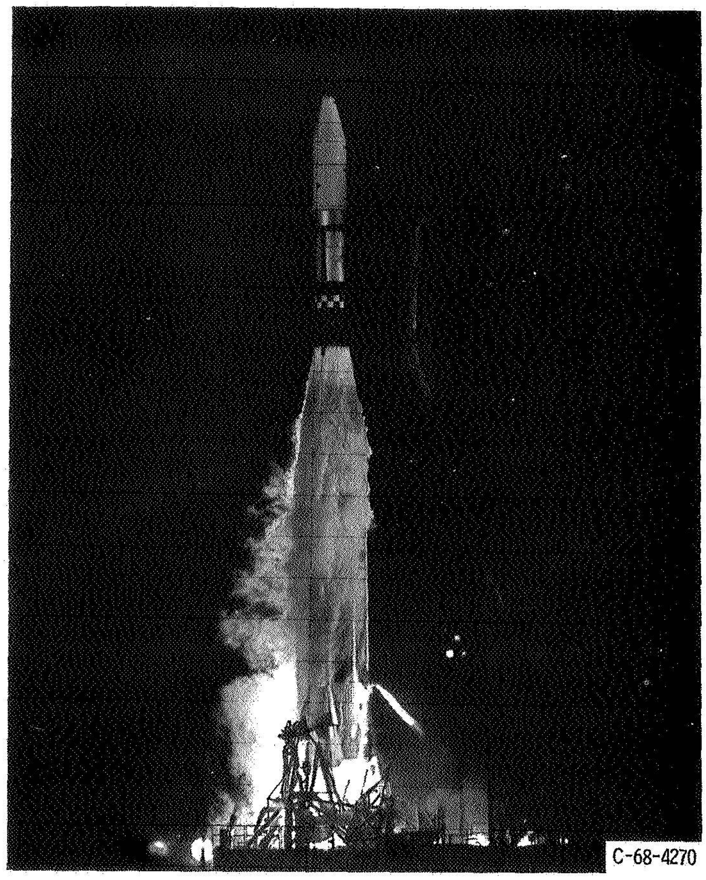


Figure III-4. - Atlas Agena lifting off with ATS-2.

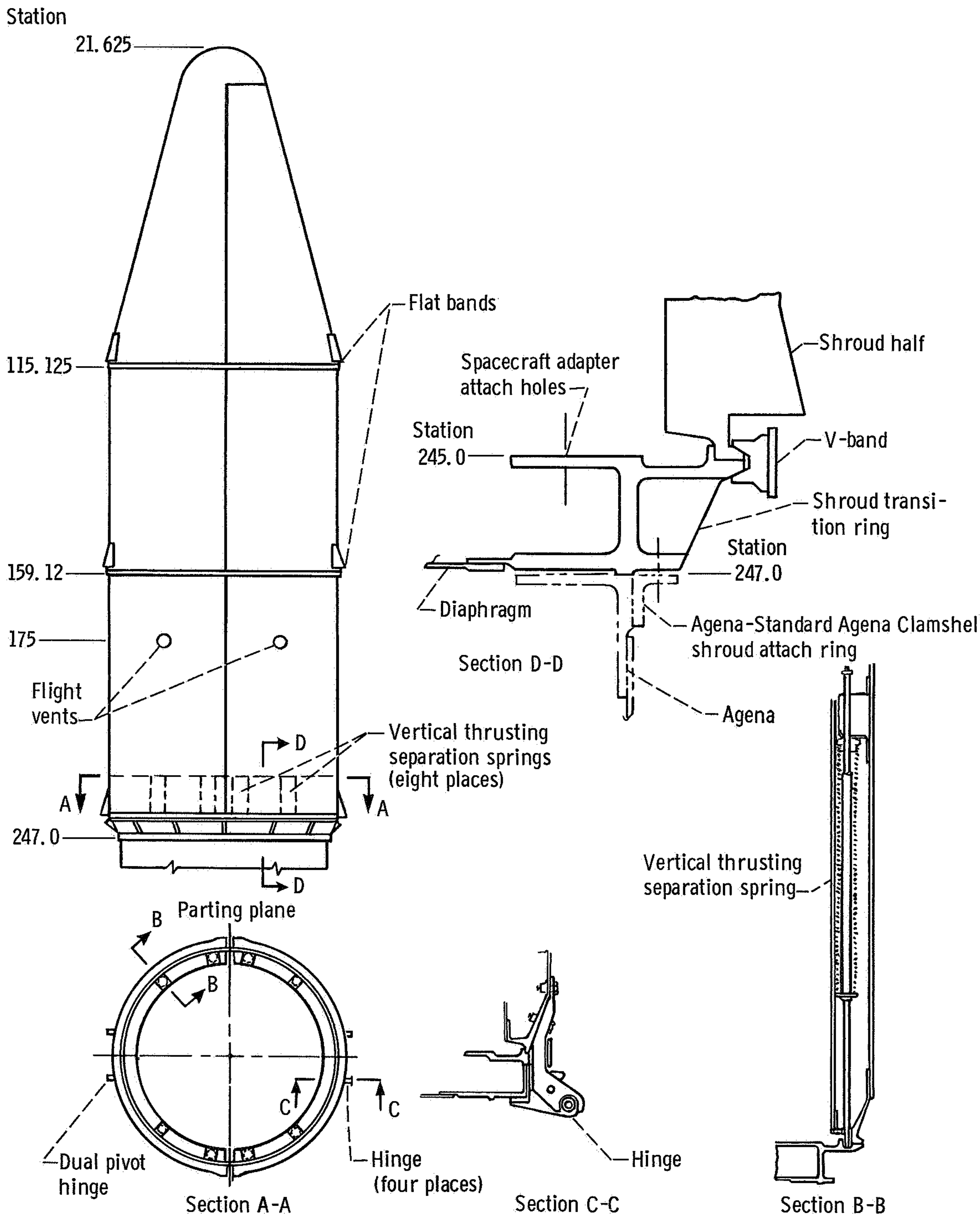


Figure III-5. - Standard Agena Clamshell shroud, ATS-2.

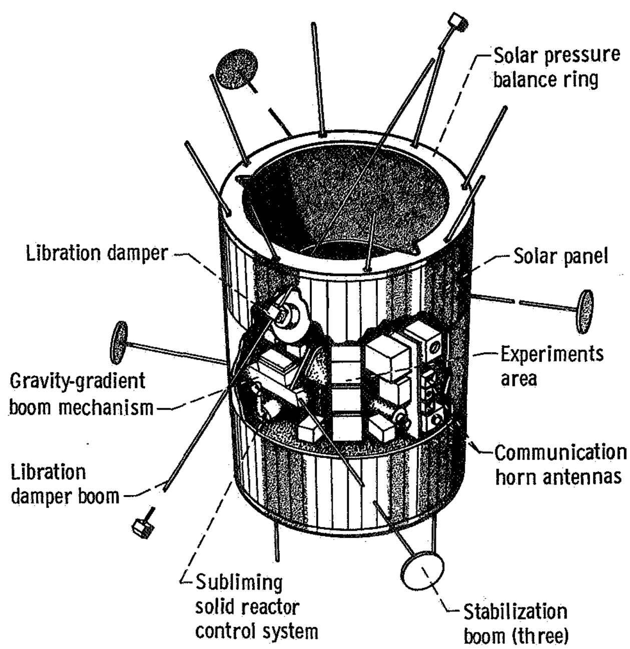


Figure III-6. - Spacecraft, ATS-2.

IV. TRAJECTORY AND PERFORMANCE

ATS-2 was launched from Complex 12, on April 5, 1967, at 2223:01.901 eastern standard time. Because the Agena failed to provide a second burn, the spacecraft was not injected into the desired 6000-nautical-mile (11 112 km) circular orbit, but remained in an elliptical orbit with a perigee of approximately 100 nautical miles (185.2 km) and an apogee of approximately 6000 nautical miles (11 112 km). A comparison of nominal[1] and actual times for the major flight events is given in table IV-I. A detailed sequence of flight events is provided in 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. Following Atlas-Agena separation and a coast period of approximately 56 seconds, the Agena engine is ignited and burns until it places the Agena-spacecraft onto a transfer orbit with an apogee altitude of approximately 6000 nautical miles (11 112 km). At the first apogee of the transfer orbit, the Agena engine is reignited. The Agena second burn injects the Agena-spacecraft onto a 6000-nautical-mile (11 112-km) circular orbit. Prior to spacecraft separation, the Agena performs an 81° yaw-right maneuver to place the spacecraft in the required attitude. Three seconds after spacecraft separation, the Agena performs a yaw-right maneuver of 99° to orient the Agena tail first in the direction of the inertial velocity vector. This maneuver places the Agena in the proper attitude for retrorocket firings that reduce the Agena orbit altitude and provide adequate separation distance between the Agena and the spacecraft.

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

TRAJECTORY RESULTS

Winds Aloft

The winds above 5000 feet (1524 m) at the time of launch were predominately from the northeast. These were crosswinds that tended to displace the trajectory to the right of nominal. The maximum wind velocity was 78 feet per second (23.77 m/sec) at an altitude of 46 260 feet (14 100 m) (see fig. IV-1).

Abrupt changes in wind velocity produced strong wind shears at altitudes from 30 000 feet (9144 m) to 50 000 feet (15 240 m). From atmospheric data taken at lift-off (T - 0), the peak bending response was calculated to be 51 percent of the critical value on the shroud (Agena station 160) and to occur at 31 134 feet (9489.6 m). These atmospheric data were obtained by a weather sounding balloon. The peak booster engine gimbal angle was calculated to be 60 percent of that available in the pitch plane and to occur at 31 368 feet (9560.9 m).

Atlas Booster Phase

The Atlas was programmed to roll the vehicle from a launch pad azimuth of 105.18° to a launch azimuth of 90.0° during the time interval of T + 2 to T + 15 seconds. The actual measured roll was approximately equal to the programmed roll. The actual pitch program is not available because the data were unreadable from T + 58 to T + 82 seconds. Yaw steering is not used during the booster phase.

Comparison of the radar position data with the postflight nominal trajectory in the vertical plane during the booster phase shows that the vehicle flight profile was nearly nominal. However, since booster engine cutoff was slightly late and velocity during the booster phase was slightly high, the actual booster engine cutoff point was about 1500 feet (457.2 m) higher in altitude and 7300 feet (2225 m) farther down range compared with the nominal cutoff point.

Comparison of the actual and nominal trajectories in the horizontal plane shows that the actual trajectory was slightly to the right of nominal during the booster phase. This deviation resulted primarily from crosswinds. Thrust misalinement and yaw gyro drift were indicated from telemetry to be small and, therefore, had little effect on the trajectory. At booster engine cutoff, the actual trajectory was about 2900 feet (883.9 m) to the right of nominal.

Comparison of the actual and nominal velocity histories shows that the velocity of the vehicle, relative to the rotating Earth, remained greater than nominal during the boost phase. The actual velocity at booster engine cutoff was about 160 feet per second

(48.77 m/sec) greater than nominal. This deviation resulted primarily from the longer-than-nominal booster burn time.

Radio guidance was enabled at T + 80 seconds. Booster pitch steering, which is constrained to the period from T + 100 to T + 110 seconds, did not occur since the pitch plane dispersion was less than the ±1.0-sigma threshold. Transmission of the booster engine cutoff discrete occurred at T + 129.2 seconds and cutoff occurred at 129.4 seconds, 0.9 second later than nominal. The acceleration level was 6.13 g's, which is 0.03 g higher than nominal but well within the ±0.2-g limits. The booster engines were jettisoned at T + 132.2 seconds.

Atlas Sustainer Phase

At the start of sustainer steering, the vehicle was turned up about 6.0° to compensate for the slightly depressed trajectory at this time. No yaw corrections are made for the cross range displacement errors accumulated during the booster phase; therefore, the effects of these errors are present at sustainer engine cutoff.

The actual trajectory during the sustainer phase remained to the right and was slightly depressed with respect to the nominal trajectory. Comparison of tracking data with the nominal trajectory at sustainer engine cutoff indicates that the vehicle was about 29 000 feet (8839.2 m) up range, 8000 feet (2438.4 m) right, and 2600 feet (792.5 m) low. Sustainer engine cutoff occurred at T + 289.0 seconds, or 3.6 seconds earlier than nominal.

Comparison of the actual and nominal sustainer phase velocities, relative to the rotating Earth, shows that the actual velocity was higher than nominal throughout the sustainer phase. Because of the early occurrence of sustainer engine cutoff, the velocity at that time was reduced to about 10 feet per second (3.048 m/sec) higher than nominal. This increment was consistent with the depressed and up range spatial position of the vehicle, which provided an energy at cutoff compatible with the desired nominal coast ellipse. The Atlas suborbital coast ellipse parameters are given in table IV-II.

Atlas Vernier Phase

Vernier engine burn duration after sustainer engine cutoff was approximately 20.5 seconds, or 0.7 second longer than the nominal time. The actual trajectory remained below and to the right of the nominal. At vernier engine cutoff, the vehicle was about 17 000 feet (5181.6 m) up range, 8300 feet (2529.8 m) right, and 1700 feet (518.2 m) below the planned trajectory. A vernier phase steering yaw-right command was issued by radio guidance to place the vehicle in the proper attitude for Atlas-Agena separation. This command displaced the vehicle 0.4° right in yaw. Vernier engine cutoff occurred at T + 309. 5 seconds, or 2. 9 seconds earlier than nominal. Atlas insertion velocities at Vernier engine cutoff are shown in table IV-III.

The radio guidance start Agena primary timer discrete started the Agena timer at T + 295. 8 seconds, or 0. 9 second earlier than the nominal time. The guidance system adjusted the timer by this amount so that Agena first burn would occur at the proper point on the coast ellipse. Because the start Agena timer discrete was transmitted 0. 9 second earlier than the nominal time, the actual times for all timer events listed in table IV-I are earlier than nominal by this amount within the ±0. 2-second timer tolerance.

Agena First-Burn Phase

Atlas-Agena separation occurred at T + 312. 0 seconds, which was 2. 5 seconds earlier than the nominal time but was consistent with guidance equation requirements. After Atlas-Agena separation, the Agena pitch-down maneuver was initiated to place the vehicle in the proper attitude for Agena first burn.

Agena first-burn ignition and shroud separation occurred at T + 369. 8 and T + 379. 9 seconds, respectively. The duration of the firing period, measured from 90 percent chamber pressure to velocity meter cutoff, was 206. 7 seconds, or 1. 6 seconds shorter than the predicted firing period. Velocity meter shutdown indicated that the proper first-burn velocity increment was attained. The first-burn thrust decay velocity contributed 13. 6 feet per second (4. 15 m/sec) to the total velocity. The resultant Agena-spacecraft transfer orbit had an apogee that was slightly higher than nominal and a perigee that was slightly lower than nominal. The actual Agena-spacecraft transfer-orbit parameters, as determined from tracking data, are given in table IV-IV.

Agena Second-Burn Phase

Agena second burn failed to occur. Because of the failure to develop second-burn impulse, the final orbit remained nearly the same as the transfer orbit. Perigee altitude increased slightly because of the addition of a small amount of impulse associated with ignition of the start canisters. The final Agena-spacecraft orbit parameters at spacecraft separation are listed in table IV-V.

Post-Second-Burn Phase

Prior to spacecraft separation, the Agena performed the programmed 81° yaw-right-attitude maneuver. At T + 7242.0 seconds, the programmed Agena and spacecraft separation occurred. Three seconds later, the Agena performed the programmed 99° yaw-right-attitude maneuver.

In the period during which Agena second burn should have occurred there was no pitch or yaw control since there was no hydraulic pressure, engine thrust, or pneumatic control. The vehicle drifted in yaw, and yaw reference was lost. Therefore, even through programmed Agena yaw maneuvers were performed correctly after the time for Agena second burn, the Agena was not at the correct yaw attitude for spacecraft separation or the firing of the two retrorockets.

TABLE IV-I. - SIGNIFICANT FLIGHT EVENTS, ATS-2

Event description	Nominal time, sec	Actual time sec
Lift-off (2223:01.901 EST)	0.0	0.0
Booster engine cutoff	128.5	129.4
Booster engine jettison	131.5	132.2
Sustainer engine cutoff	292.6	289.0
Start Agena primary timer	296.7	295.8
Vernier engine cutoff	312.4	309.5
Atlas-Agena separation	314.5	312.0
Fire first-burn ignition squibs	370.7	369.8
Agena steady-state thrust (90 percent chamber pressure)	371.8	371.0
Shroud separation	380.7	379.9
Agena first-burn cutoff	580.2	577.7
Fire second-burn ignition squibs	7115.7	7114.9
Agena steady-state thrust (90 percent chamber pressure)	7116.8	(a)
Agena second-burn cutoff	7143.0	(a)
Start 81° yaw right	7205.7	7205.0
Stop 81° yaw right	7232.7	7232.0
Fire spacecraft separation squibs	7242.7	7242.0
Start 99° yaw right	7245.7	7245.0
Stop 99° yaw right	7278.7	7278.0
Fire first retrorocket	7942.7	7942.0
Fire second retrorocket	12 031.7	12 031.0

[a]Agena second burn did not occur.

TABLE IV-II. - ATLAS SUBORBITAL COAST ELLIPSE PARAMETERS, ATS-2

Parameter	Units	Nominal	Actual[a]
Semimajor axis	n mi	2 392.98	2 392.91
	km	4 431.8	4 431.67
Semiminor axis	n mi	2 095.86	2 095.86
	km	3 881.53	3 881.53
Radius vector magnitude at apogee	n mi	3 547.81	3 547.65
	km	6 570.54	6 570.24
Inertial velocity at apogee	ft/sec	18 381.4	18 382.1
	m/sec	5 602.7	5 602.9
Inclination	deg	28.33	28.30

[a]Calculated from guidance system data.

TABLE IV-III. - ATLAS INSERTION VELOCITIES AT VERNIER ENGINE CUTOFF, ATS-2

Parameter	Units	In-flight objective[a]	Actual[b]
Velocity magnitude	ft/sec	18 553.0	18 552.7
	m/sec	5 654.9	5 654.8
Altitude rate	ft/sec	1 751.5	1 751.0
	m/sec	533.9	533.7
Lateral velocity	ft/sec	0	1.3 (right)
	m/sec	0	0.4 (right)

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

TABLE IV-IV. - AGENA-SPACECRAFT TRANSFER ORBIT PARAMETERS, ATS-2

Parameter	Units	Actual
Apogee altitude	n mi	6 012.1
	km	11 134.4
Perigee altitude	n mi	99.8
	km	184.8
Period	min	218.8
Inclination	deg	28.42

TABLE IV-V. - FINAL AGENA-SPACECRAFT ORBIT PARAMETERS, ATS-2

Parameter	Units	Actual
Apogee altitude	n mi	6 011.3
	km	11 132.9
Perigee altitude	n mi	105.7
	km	195.8
Inclination	deg	28.34
Period	min	218.9
Eccentricity	----	0.4543

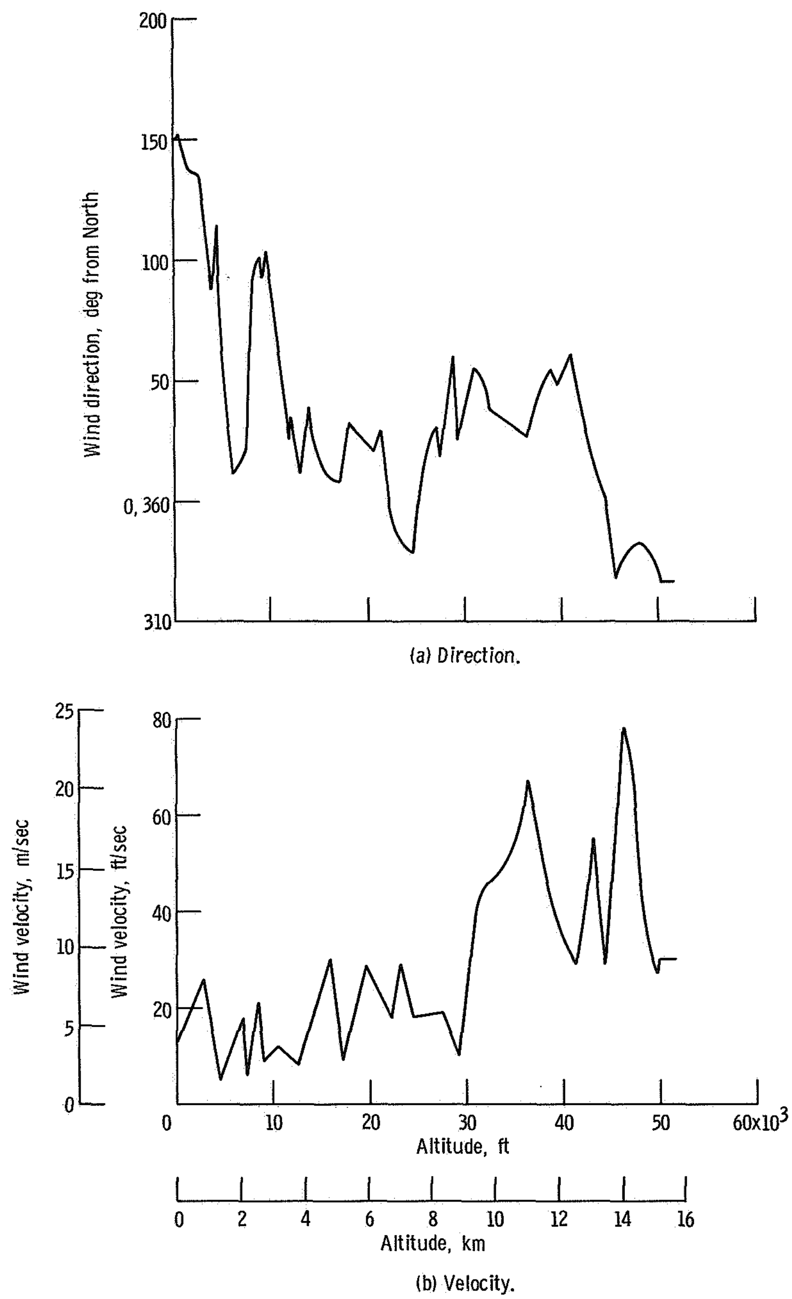


Figure IV-1. - Wind data, ATS-2.

V. ATLAS VEHICLE SYSTEM PERFORMANCE

AIRFRAME SYSTEM

The performance of the Atlas airframe system was satisfactory. No structural anomalies were encountered.

Description

The Atlas airframe structure consists of two sections: the tank section and the booster section.

The tank section consists of a thin-walled, all-welded, monocoque, stainless-steel cylinder which is divided into a kerosene (RP-1) compartment and a liquid-oxygen compartment by an intermediate bulkhead. The tank section is 10 feet (3.048 m) in diameter, with an ellipsoidal bulkhead enclosing the conical forward end and a thrust cone enclosing the aft end. The overall tank length is 60.9 feet (18.6 m). Tank structural rigidity is derived from internal pressurization. Fairings are provided on the tank to form equipment pods to protect the equipment against aerodynamic effects.

The booster section consists of a thrust structure, booster engines, nacelles, and a fairing installation. The booster engine section is attached to the thrust ring at the aft end of the tank section by a latching mechanism. Approximately 3 seconds after booster engine cutoff, this section is jettisoned.

Performance

The airframe system performance was satisfactory. All measured loads were within the expected limits. The peak longitudinal load factor during the boost phase of flight was 6.13 g's at booster engine cutoff. The control valve actuation command signal, which initiates high pressure pneumatics flow to the booster release latching mechanism, occurred at T + 132.2 seconds. The mechanism functioned properly and the booster separated satisfactorily.

This flight was the second NASA SLV-3 flight using a new engine boot design. In certain previous flights, with Atlas using the old type boots, telemetry data indicated that engine compartment temperatures were considerably higher than normal, possibly caused by insufficient shielding of the engine compartment and its components from engine radiation and hot gases. Because high temperatures could prove injurious to the engine compartment components and wiring, new boots were designed to provide improved shielding. The engine compartment temperatures on this mission were satisfactory and compared favorably with expected temperatures. The locations of the temperature measurements are shown in figure V-1, and the measured values are given in table V-1.

TABLE V-I. - ATLAS ENGINE COMPARTMENT

TEMPERATURES, ATS-2

Measurement number	Units	Flight values at-	
		T - 0	Booster engine cutoff
A743T	^{o}F	42	40
	K	279	277.7
A745T	^{o}F	70	94
	K	294.4	307.75
P15T	^{o}F	70	76
	K	294.4	297.75
P16T	^{o}F	60	80
	K	288.85	300
P671T	^{o}F	52	76
	K	284.4	297.75

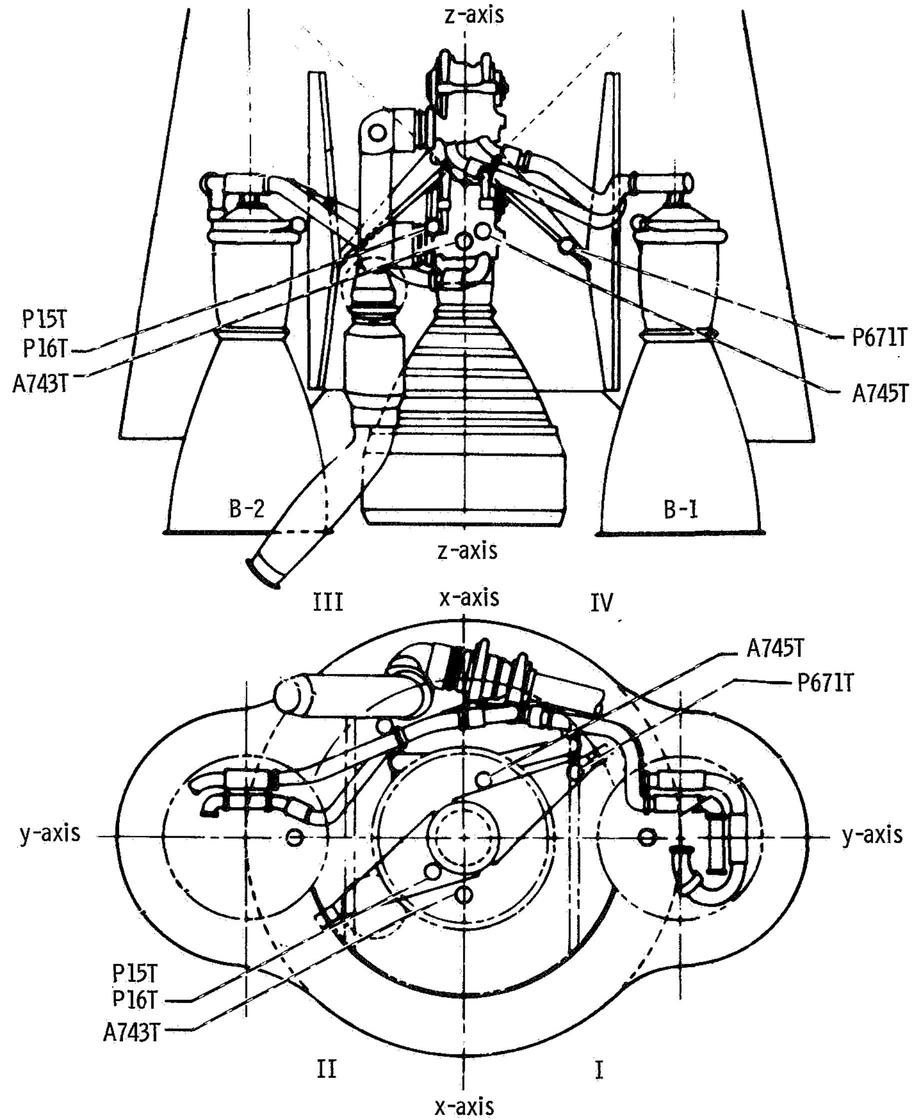


Figure V-I. - Atlas engine compartment temperature instrumentation, ATS-2.

PROPULSION SYSTEM

The performance of the Atlas propulsion system was satisfactory. Engine ignition and start sequence occurred as prescribed. In-flight telemetry data compared closely with expected values. All engine cutoff signals were generated by discrete guidance commands and were properly executed. Transients at booster engine, sustainer engine, and vernier engine shutdowns appeared normal.

Description

The Atlas engine system consists of a booster, a sustainer, and two vernier engine systems, an engine tank system, and an electrical control system. The engines are fixed thrust and of the single-start type. During engine start, electrically fired pyrotechnic igniters are used to ignite the gas generator propellants, and hypergolic igniters are used to ignite the booster, sustainer, and vernier engine thrust chamber propellants. The propellants are liquid oxygen (oxidizer) and liquid RP-1 (fuel).

The booster engine, rated at 330 000 pounds (1467.9×10^{3} N) thrust at sea level, is made up of two gimbaled thrust chambers, propellant valves, two liquid-oxygen and two fuel turbopumps driven by one gas generator, a lubricating oil system, and a heat exchanger. The sustainer engine, rated at 57 000 pounds (253.55×10^{3} N) thrust at sea level, consists of a thrust chamber, propellant valves, a gas-generator-driven liquid-oxygen and fuel turbopump, and lubricating oil system. The sustainer engine system is mounted so that it can gimbal.

Each vernier engine is rated at 669 pounds (2.976×10^{3} N) thrust at sea level when supplied with propellants from the sustainer pumps and 525 pounds (2.335×10^{3} N) thrust at sea level when supplied with propellants from the engine tank system. The thrust chambers are mounted so that they can gimbal.

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

Performance

The engine start sequence and operation for all engines were 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 (see table V-II). 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 the propellant utilization valve position and vehicle acceleration. (See PROPELLANT UTILIZATION SYSTEM section for details.)

Vernier engine operation throughout flight was satisfactory. Evaluation of in-flight performance was made by comparing expected and measured values of vernier thrust chamber pressures. The number 2 vernier chamber pressure data showed a gradual decrease of 20 psi (14 N/cm^2) beginning at T + 152 seconds; it abruptly recovered at T + 165 seconds. This indication of chamber pressure decay probably resulted from a buildup of carbon in the chamber pressure instrumentation sensing port. Engine performance did not reflect an actual pressure decay.

TABLE V-II. - ATLAS PROPULSION SYSTEM FLIGHT DATA, ATS-2

Performance parameters	Units	Sea-level design values	Flight values at -			
			T + 10 sec	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
B-1 booster engine chamber pressure (absolute)	psi N/cm^2	560 to 595 386 to 410	560 386	568 392	------ ------	------ ------
B-2 booster engine chamber pressure (absolute)	psi N/cm^2	560 to 595 386 to 410	568 392	576 397	------ ------	------ ------
Booster engine gas generator chamber pressure (absolute)	psi N/cm^2	510 to 555 352 to 383	523 361	516 356	------ ------	------ ------
B-1 pump speed	rpm	6225 to 6405	6282	6273	------	------
B-2 pump speed	rpm	6165 to 6345	6220	6250	------	------
Sustainer engine chamber pressure (absolute)	psi N/cm^2	680 to 715 469 to 493	715 493	700 483	695 479	------ ------
Sustainer engine gas generator discharge pressure (absolute)	psi N/cm^2	590 to 686 407 to 473	625 431	632 436	633 436	------ ------
Sustainer pump speed	rpm	10 025 to 10 445	10 344	10 390	10 381	------
V-1 chamber pressure (absolute):						
Pump supplied	psi N/cm^2	250 to 265 172 to 183	252 174	248 171	252 174	------ ------
Tank supplied	psi N/cm^2	210 to 225 145 to 155	------ ------	------ ------	------ ------	212 146
V-2 chamber pressure (absolute):						
Pump supplied	psi N/cm^2	250 to 265 172 to 183	------ 176	------ 176	------ 176	------ ------
Tank supplied	psi N/cm^2	210 to 225 145 to 155	------ ------	------ ------	------ ------	216 149

HYDRAULIC SYSTEM

The Atlas booster and sustainer hydraulic systems performance was satisfactory. Pressures were maintained stable and at the proper level throughout the Atlas phase of flight.

Description

Two independent hydraulic systems are used on the Atlas to supply fluid power to the booster engine system and the sustainer-vernier engine system. A variable displacement pump and an accumulator in the booster engine hydraulic system provides the hydraulic pressure to gimbal the booster engines a maximum of $\pm 5^{\circ}$. A variable displacement pump and three accumulators in the sustainer-vernier engine hydraulic system provide the hydraulic pressure to gimbal the sustainer engine a maximum of $\pm 3^{\circ}$ and the vernier engines a maximum of $\pm 70^{\circ}$. The sustainer-vernier system also provides hydraulic 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 engine gimbal actuators are provided with hydraulic pressure from two pressurized accumulators.

Performance

Hydraulic system pressure data for both the booster and sustainer hydraulic circuits are shown in table V-III. Pressures were stable throughout the boost phase. The transfer of fluid power from ground to airborne hydraulic systems was normal. The starting transient produced a normal overshoot of about 10 percent in the hydraulic pump discharge pressure. The pressure just after start, stabilized at an absolute value of 3045 psi (2099 N/cm^2) in the sustainer and 3115 psi (2148 N/cm^2) in the booster system.

TABLE V-III. - ATLAS HYDRAULIC SYSTEM FLIGHT DATA, ATS-2

Measurement	Units	Expected value at time interval	Flight values at -			
			T - 0	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Sustainer-vernier pressure (absolute)		T - 0 to sustainer engine cutoff:				
	psi	2950 to 3150	3040	3040	3010	
	N/cm^2	2034 to 2172	2096	2096	2075	
		Sustainer engine cutoff to vernier engine cutoff:				
	psi	3150 to 1000	----	----	----	1155
	N/cm^2	2172 to 689	----	----	----	796
Sustainer pump discharge pressure (absolute)		T - 0 to sustainer engine cutoff:				
	psi	2950 to 3150	3045	3045	3010	----
	N/cm^2	2034 to 2172	2099	2099	2075	----
Booster pump discharge pressure (absolute)		T - 0 to booster engine cutoff:				
	psi	2950 to 3150	3115	3080	----	----
	N/cm^2	2034 to 2172	2148	2124	----	----
B-1 booster engine accumulator pressure (absolute)		T - 0 to booster engine cutoff:				
	psi	2950 to 3150	3150	3080	----	----
	N/cm^2	2034 to 2172	2172	2124	----	----

PROPELLANT UTILIZATION SYSTEM

The performance of the Atlas propellant utilization system was satisfactory. The burnable propellant residuals at sustainer engine cutoff were satisfactory for this flight. The propellant utilization valve responded properly to the error counter output at each sensor station.

Description

The Atlas propellant utilization system is designed to cause nearly simultaneous depletion of both propellants. This system is a digital type that adjusts the operating mixture ratio of the sustainer engine by sampling the propellant volume ratio at six discrete points during flight. Six fuel and six oxidizer 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 fully 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 fully open or closed. This adjustment would theoretically result in a zero error time when the liquid level reached the next sensor pair.

This uncovery time difference is measured and then is transmitted to a hydraulic control unit that 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. It 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

The burnable propellant residuals in the propellant tanks at sustainer engine cutoff were 646 pounds (293 kg) of fuel and 585 pounds (265 kg) of oxidizer. There residuals would have allowed the sustainer engine to burn an additional 3 seconds if required; however, the proper velocity had been attained and the guidance system shut down the engine. If the flight had continued to theoretical liquid-oxygen depletion, 393 pounds (178 kg) of RP-1 (fuel) would have remained in the tank. The RP-1 and liquid-oxygen sensing ports were uncovered at 1.2 and 5.4 seconds, respectively, prior to sustainer engine cutoff.

Fuel and oxidizer pressure 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 uncovery to sustainer engine cutoff. This time interval is used in determing the propellant residuals.

Table V-IV 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-IV. - ATLAS PROPELLANT LEVEL SENSOR

ERROR TIMES, ATS-2

Station	Limit error time, sec	Actual error time, sec	First sensor uncovered
1	1.018	0.6	Liquid oxygen
2	.968	.5	Liquid oxygen
3	.78	.05	Fuel
4	1.89	.1	Fuel
5	8.4	2.4	Fuel
6	4.2	1.6	Fuel

PNEUMATIC SYSTEM

The performance of the Atlas pneumatic system was satisfactory. Both the fuel and the oxidizer tank ullage pressure regulators operated within the design range.

Description

The pneumatic system performs two principal functions: the system pressurizes the tanks to supply the propellant pumps with sufficient inlet pressure to prevent cavitation, and it provides pressure to support the monocoque tank structure. The system also performs several service functions. Pressurized gas is bled from the fuel tank pressurization line and is used to pressurize the hydraulic reservoirs and the lubrication tanks. Pressure is also used to activate the booster separation mechanism, purge seals, pressurize vernier engine start tanks, and for various engine control functions.

The helium pressurization gas is stored in eight spheres which are located in the booster and sustainer engine sections. Six of these spheres supply helium gas for tank pressurization during the booster phase of flight; they are located in the booster engine compartment and are attached to the jettisonable thrust barrel. The spheres are surrounded by metal shrouds and are cooled, prior to launch, by liquid nitrogen between each sphere and shroud. The cooling liquid drains from the shrouds at lift-off.

During booster stage, cold gas leaves the six storage containers and flows through the helium shutoff valve into the heat exchanger where it is heated and expanded. Hot gas from the heat exchanger flows through tank regulators and enters the propellant tanks at proper pressure levels. The relief valves relieve excessive pressure in the propellant tanks if required. The tank pressurization helium storage containers are jettisoned at booster engine staging, and pneumatic regulation of tank pressure is terminated. Thereafter, the pressure in the propellant tanks decays slowly except that the oxidizer tank pressure is partly sustained by liquid-oxygen boiloff.

Gas is stored in the remaining two spheres at ambient temperature. The engine control sphere is located on the sustainer section and provides gas for both booster and sustainer engine controls. The other helium supply sphere, located in the booster engine section, provides the pressure to actuate the booster section release mechanism.

Performance

All tank pressures were maintained satisfactorily, and all control functions were performed properly. Pneumatic system parameters during flight are shown in table V-V.

Liquid-oxygen tank ullage pressure oscillations were present. This condition is considered normal with the type of liquid-oxygen regulator employed. Prior to lift-off, these oscillations, at a frequency of 3.3 hertz, caused the amplitude of the differential pressure across the liquid-oxygen fuel tank bulkhead to vary a maximum of 3.5 psi (2.4 N/cm^2) peak to peak. After lift-off, the frequency of these oscillations increased to 5.3 hertz and caused the amplitude of differential pressure across the liquid-oxygen - fuel tank bulkhead to vary a maximum of 6.0 psi (4.1 N/cm^2). The liquid-oxygen tank ullage pressure oscillations began to damp at approximately T + 7 seconds and were completely damped by T + 29 seconds. All pressures were within specifications.

TABLE V-V. - ATLAS PNEUMATIC SYSTEM FLIGHT DATA, ATS-2

Parameter	Units	Design range	Relief pressure	Flight values at -				
				T - 10 sec	T - 0	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Oxidizer tank ullage pressure (gage)	psi N/cm^2	28.5 to 31.0 19.6 to 21.4	33.0 to 34.0 22.8 to 23.4	31.5 21.7	30.3 20.9	29.7 20.5	[a]27.4 18.9	27.4 18.9
Fuel tank ullage pressure (gage)	psi N/cm^2	64.0 to 67.0 44.1 to 46.2	69.5 to 70.5 47.9 to 48.6	67.2 46.3	65.3 45.0	65.7 45.3	[a]48.9 33.7	48.9 33.7
Sustainer controls sphere pressure (absolute)	psi N/cm^2	3400 (maximum) 2344 (maximum)	----------- -----------	3080 2124	2990 2062	2660 1834	2450 1689	1575 1086
Tank pressurization helium spheres pressure (absolute)	psi N/cm^2	3400 (maximum) 2344 (maximum)	----------- -----------	3080 2124	2985 2058	630 434	[a]---- -----	---- ----
Tank pressurization helium spheres temperature	^{o}F K	-305 to -320 85.5 to 77.5 (prior to T - 0)	----------- -----------	-320 77.5	-320 77.5	-383 43.3	[a]---- -----	---- ----

[a]Helium supply bottles for propellant tank pressurization were jettisoned at booster engine cutoff + 3 sec.

GUIDANCE AND FLIGHT CONTROL SYSTEM

The flight control and radio guidance system performance was satisfactory. The Atlas placed the Agena-spacecraft into a suborbital coast ellipse with the correct energy. All radio guidance discrete commands were properly transmitted and executed.

Description

The Atlas flight path is controlled by two interrelated systems: the autopilot system and the Mod III Radio Guidance system. The Atlas is flown in an open-loop mode along a desired trajectory by the autopilot system. Deviations from the desired trajectory occur, however, because of wind forces on the vehicle and other causes. These deviations are detected by the Mod III Radio Guidance ground based radars which are tracking the vehicle. When deviations from the desired trajectory are detected, the Mod III Radio Guidance command link is used to provide corrective steering commands to the Atlas. These corrective steering commands are determined in the ground computer by comparing the radar track of the vehicle with the desired track. The corrective steering capability of the Mod III Radio Guidance system is only employed for a short period during the booster phase (i.e., from T + 100 to T + 110 sec). It also provides continuous steering correction from booster engine cutoff to termination of Atlas engine thrust at vernier engine cutoff. In addition to providing steering corrections, the Mod III Radio Guidance system is also used to transmit discrete commands to the vehicle. The discrete commands (for which the Mod III Radio Guidance system is prime) transmitted are booster, sustainer, and vernier engine cutoffs, Atlas-Agena separation, and start Agena primary timer. The timing of these commands is determined in the ground computer based on the actual trajectory being flown by the vehicle. The commands are timed to assure that the Agena-spacecraft will be placed in the proper suborbital coast ellipse by the Atlas, and also that the Agena first burn will occur at the proper time.

The major elements of the autopilot subsystem are the flight programmer, gyro reference packages, servocontrol electronics, and hydraulic controllers. Timing and switching functions are performed by the flight programmer using command inputs from the radio guidance for certain events.

Steering commands from either the flight programmer or radio guidance are transmitted to the gyro package, which monitors the instantaneous difference between actual and desired vehicle attitude. In each of the three axes (pitch, roll, and yaw) single-degree-of-freedom rate-integrating displacement gyros form a prime reference. Steering commands are effected by torquing these gyros. Signals proportional to the difference between actual and desired vehicle attitude, as measured by the gyro gimbal angle, are transmitted to the respective gyro signal amplifier for input to the servocontrol units. Rate damping is provided by signals generated by three rate gyros. These gyros sense vehicle angular rates in pitch, yaw, and roll and introduce corrective signals into the gyro signal amplifiers.

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 track subsystem, a continuous wave L-band rate subsystem, and a digital guidance computer subsystem.

The track subsystem, which measures range, azimuth, and elevation, 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 and relays the pitch and yaw steering commands and discrete command outputs to the autopilot system.

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. It transmits a continuous wave signal at a frequency equal to the arithmetic average of the frequencies of the received signals. The return signal is received by the central rate station and two outlying rate leg receiving stations. The two-way doppler shift and phase relation of the signals, as received at the three separate ground antennas, 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 acquistion aid, or by slaving to range, 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, and elevation. The first cube on the programmed trajectory is selected at a point where good lock can be obtained as the vehicle passes through the cube.

A conical scan antenna on the same mount as the main 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 antenna; however, during initial acquisition, rate subsystem lock is normally accomplished before track subsystem lock due to differences in antenna gains and beamwidths and to receiver sensitivities.

The position and velocity information from the track and rate subsystem is transmitted to the data processing circuitry in the ground computer where the commands are generated. The steering commands are received by the pulse beacon and processed by the decoder. The resulting outputs from the decoder are continously variable voltages which are transmitted to the autopilot to torque the appropriate gyro. The gyro torque rate is proportional to the decoder output voltage and varies from zero to a maximum of 2 degrees per second. Gyro torque rates from the decoder are summed with other torque rates generated within the autopilot, and the resulting gyro gimbal angles are sensed. Signals proportional to the gyro gimbal angles are amplified and transmitted to the servo-control units that gimbal the engines as required. Engine gimbal angles and vehicle attitude changes then are a function of the autopilot open-loop guidance and control as well as the magnitude and duration of the radio guidance steering commands. The radio guidance thus provides vernier adjustments to the autopilot open-loop guidance to guide the vehicle onto the desired flight path.

The booster engine cutoff discrete terminates booster engine thrust at a specified value of longitudinal acceleration. The sustainer and vernier engine cutoff discretes are transmitted when the vehicle energy (position and velocity) is such that the desired coast ellipse will be achieved. The start Agena timer discrete is timed so that the initiation of Agena first burn will occur at a fixed time prior to the coast ellipse apogee.

Performance

The performance of the Atlas autopilot was satisfactory. Lift-off transients in pitch, yaw, and roll were within acceptable limits. The maximum roll displacement during this lift-off transient was 0.4° in a counterclockwise direction (viewing the vehicle from aft). The maximum pitch displacement was 0.5° up. The maximum yaw displacement was 0.22° to the right.

During the programmed roll maneuver, the roll rate gyro indicated that the vehicle rolled approximately 15°. The length of the roll period was 13 seconds. The programmed launch azimuth required a total roll of 15.18°.

The pitch program was initiated at T + 15 seconds as planned. The programmed and actual times and pitch rates for each step of the Atlas pitch program are listed in table V-VI.

Maximum dynamic pressure occurred at T + 71.0 seconds. Dynamic disturbances during the period of maximum dynamic pressure were small and required the booster engines to gimbal a maximum of 1.5° in pitch and 0.71° in yaw to correct for these disturbances. This represents 30.0 percent of the engine gimbal capability in pitch and 14.2 percent of the engine gimbal capability in yaw. These gimbal angles compare favorably with the maximum gimbal angle prediction based on the atmospheric data (wind soundings) taken at T - 0. The vehicle was offset by 0.07° pitch up and 0.6° yaw left at the time the booster engines were commanded to null (0.9 sec after engine shutdown).

The booster engine jettison sequence was normal, and the resulting small pitch and yaw transients were quickly damped. The sustainer engine returned the vehicle to the correct attitude.

After completion of sustainer burn and vernier solo, the command to separate Agena was initiated by radio guidance. At the time, the vehicle was stable in attitude and separation was successfully completed.

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

The track subsystem conical scan antenna acquired the vehicle in the first cube at T + 59.5 seconds. The automatic switch to monopulse tracking with the main antenna occurred at T + 64.5 seconds, and good data were presented to the computer by T + 67.6 seconds.

Track lock was continuous from acquisition until T + 419.9 seconds, 108.3 seconds after the Atlas-Agena separation discrete command. Track lock was then intermittent until final loss of lock at the ground station occurred at T + 433.9 seconds. At this time, the Atlas was at an elevation angle of 2.17^{o} above the horizon. The signal received by the ground track subsystem was within 4 decibels of the theoretically expected level.

All rate antennas were locked on the vehicle by T + 58.0 seconds, and good data were presented to the computer by T + 59.9 seconds. Rate lock was continuous until T + 392.1 seconds. It then became intermittent and was finally lost at T + 433.9 seconds, when the Atlas was 2.17^{o} above the horizon. The signals received by the central rate antenna were within 5 decibels of the expected (calculated) levels, and the signals received at the two rate-leg antennas were within 5 decibels of those received at the central rate antenna.

The computer subsystem performance was satisfactory throughout the vehicle flight. A countdown difficulty was encountered during the built in hold at T - 60 minutes when the temperature of the erasable memory exceeded the high temperature limit. Additional cooling was provided to the unit, and normal readings were restored without interruption to the countdown.

Following the flight, the guidance program was verified before removal of the program from the computer. A simulated rerun of the flight indicated that no transient errors occurred during the flight.

The pulse beacon automatic gain control monitor indicated a received signal strength of -52 dBm (decibels referenced to 1 mW) at the time of acquisition, with a signal increase to approximately -22 dBm at T + 65 seconds. The received signal strength reached a maximum of -10 dBm at T + 75 seconds and gradually decreased to -35 dBm at Agena separation. The signal strength continued to decrease until T + 394 seconds when the received signal strength was less than -78 dBm.

Pulses were missing as expected between T + 81.5 and T + 82.4 seconds and between T + 91.8 and T + 94.7 seconds because of momentary loss of signal when the look angles from the vehicle antenna to the ground station coincided with the nulls in the vehicle-borne antenna radiation pattern. Momentary loss of pulses occurred, as expected, during the booster engine staging sequence because of signal attenuation when the booster engines separated from the Atlas.

The pulse beacon magnetron current monitor indicated intermittent beacon response during acquisition until T + 58 seconds. Except for the normal momentary telemetry dropouts during periods of missing pulses, the magnetron current monitor indicated good beacon response from T + 58 to T + 433 seconds and intermittent beacon response until the magnetron current decreased to zero at T + 438 seconds.

The rate beacon automatic gain control monitors 1 and 2 indicated that the received signal strengths of the two carrier frequencies were intermittent during acquisition until T + 55.5 seconds. At T + 55.5 seconds, they reached a signal level greater than -75 dBm, and remained so until approximately T + 383.5 seconds. Both signal strengths gradually decayed to the threshold sensitivity of the receiver, -85.4 dBm, at approximately T + 390 seconds. The rate beacon phase detector and power output monitors indicated that the received signal was processed and that the return signal was properly transmitted to the ground station during the period from T + 55.5 to T + 390 seconds.

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

False pitch and yaw commands were observed, as on prior flights, during the periods of intermittent pulse beacon lock between T + 50.5 and T + 57.5 seconds during cube acquisition. The false commands were less than ±18 percent of maximum command during the preceding interval and were inhibited by the vehicle-borne autopilot subsystem. Gyro torque rates resulting from radio guidance steering commands are proportional to the magnitude of the steering command; the maximum torque rate is 2 degrees per second (see pp. 32 and 33 of Description).

Guidance steering was enabled within the vehicle-borne autopilot at T + 80 seconds. No steering commands were transmitted during the booster phase, since the pitch attitude errors were within the ±1 sigma error band. Steering commands are transmitted between T + 100 and T + 110 seconds; in this interval they are sent only if required.

Sustainer steering was initiated at T + 138.1 seconds. The largest pitch steering commands were a pitch down of 37 percent of maximum steering command at T + 139.4 seconds followed by a pitch up of 93 percent of maximum steering command at T + 141 seconds. Pitch steering commands were reduced to within ±10 percent of the maximum steering command by T + 145 seconds, and remained so until sustainer engine cutoff. Yaw steering commands from the decoder were within ±10 percent throughout the sustainer phase of flight. The amplitude and duration of steering commands indicated normal steering by radio guidance.

Vernier steering commands to correct vehicle attitude resulted in an 18-percent yaw-right command from the decoder which was reduced to within ±10 percent in 0.5 second. Pitch commands remained within ±10 percent during this period. These commands were within the acceptable limits.

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

TABLE V-VI. - ATLAS PITCH PROGRAM, ATS-2

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	.980
35 to 45	35 to 45.3	.848	.810
45 to 58	45.3 to 58.1	.509	.485
58 to 70	58.1 to 70.3	.678	(a)
70 to 82	70.3 to 82.3	.806	(a)
82 to 91	82.3 to 91.3	.678	.645
91 to 105	91.3 to 105.2	.551	.525
105 to 120	105.2 to 120.2	.382	.323
120 to 139.2	120.2 to sometime during sustainer pitch steering	.254	.242
139.2 to sustainer engine cutoff	Sometime during sustainer pitch steering to sustainer engine cutoff	.042	(b)

[a]Attitude control pitch activity prevents determination of programmed pitch rate.
[b]Data cannot be read because of telemetry accuracy.

TABLE V-VII. - MOD III RADIO GUIDANCE DISCRETE TIMES AND COORDINATES, ATS-2

Flight event and trajectory function	Units	Actual[a] discrete generation time after lift-off, sec	Vehicle coordinates at time of discretes	Discrete duration, sec
Booster engine cutoff:	sec	T + 129.150		0.497
Range	ft		326 935	
	m		99 650	
Azimuth	deg		88.115	
Elevation	deg		31.211	
Sustainer engine cutoff:	sec	T + 288.910		0.737
Range	ft		2 212 333	
	m		674 319	
Azimuth	deg		90.296	
Elevation	deg		10.325	
Start Agena timer	sec	T + 295.713	(b)	0.933
Vernier engine cutoff:	sec	T + 309.497		0.650
Range	ft		2 561 918	
	m		780 873	
Azimuth	deg		90.359	
Elevation	deg		9.917	
Atlas-Agena separation	sec	T + 311.650	(b)	Until final loss of lock

[a]Time discrete was initiated by ground guidance system.
[b]Not available.

ELECTRICAL SYSTEM

The performance of the Atlas electrical system was satisfactory. The inverter output frequency remained essentially constant at 400 hertz throughout the flight and the phase voltages were approximately 116 volts ac. The main missile battery voltage was essentially constant at 28. 0 volts dc.

Description

The electrical system supplies and distributes power to user systems. The electrical system consists of four 28-volt dc manually activated batteries; a 115-volt ac, 3-phase, 400-hertz inverter; a power changeover switch; a distribution box; two junction boxes; and related electrical harnesses. The main missile 28-volt dc battery supplies power to the autopilot system, the radio guidance system, the propellant utilization system, the propulsion system, and the inverter. One 28-volt dc battery supplies power to the telemetry system and two 28-volt dc batteries supply power to the flight termination system. The inverter supplies power to the autopilot, the propellant utilization system, and the radio guidance system. Phase A of the inverter is used as a reference in the autopilot 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 internal power.

Performance

All measured electrical system parameters were within specifications at all times. The inverter frequency was 398. 8 hertz at T - 0 and increased normally to 400. 3 hertz at Atlas-Agena separation. The inverter phase A output voltage remained constant at 115. 9 volts during the period between lift-off and Atlas-Agena separation. The main missile 28-volt dc battery voltage varied from 27. 9 volts dc at T - 0 to 28. 3 volts dc at Atlas-Agena separation. Atlas electrical system flight data are shown in table V-VIII.

TABLE V-VIII. - ATLAS ELECTRICAL MEASUREMENTS, ATS-2

Measurement	Lift-off	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff	Atlas-Agena separation
Main battery voltage	27.9	28.0	28.3	28.3	28.3
Inverter frequency, Hz	398.8	399.6	400.3	400.3	400.3
Phase A voltage, V ac	115.9	115.9	115.9	115.9	115.9
Phase B voltage, V ac	116.1	116.0	116.0	116.0	116.0
Phase C voltage, V ac	116.2	116.1	116.0	116.0	116.0

TELEMETRY AND INSTRUMENTATION SYSTEM

The Atlas telemetry and instrumentation system performance was satisfactory. No prelaunch or flight failure or launch delays resulted from telemetry and instrumentation system operation. No measurements of telemetry system flight performance were made.

Description

The telemetry and instrumentation system consists of one telemetry package, a manually activated 28-volt dc battery, and associated 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 better 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 level of 3.5 to 6 watts and requires 28 volts dc for operation. The transmitter for the Atlas is designed to use standard Interrange Instrumentation Group subcarrier channels 1 to 18 (see table V-IX). The output frequencies of all subcarrier channels modulate the 249.9-megahertz carrier frequency.

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

Telemetry and instrumentation system performance was satisfactory. All 114 measurements attempted yielded data. Adequate signal strength was maintained throughout the Atlas flight, except for the expected momentary loss of signal (less than 1 sec) at booster engine staging. Commutator speeds were stable and decommutation was satisfactory. The stations used to record the Atlas telemetry signals are shown in appendix C (fig. C-2).

TABLE V-IX. - ATLAS TRANSMITTER

SUBCARRIER CHANNELS, ATS-2

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

The Atlas flight termination system performance was satisfactory. Telemetry measurement on receiver number 1 indicated that the signal strength at the vehicle was sufficient to provide flight termination capability during the powered flight.

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 independently 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 can be commanded by the range safety officer.

Performance

Flight termination system performance during countdown and flight was normal. Battery voltage was satisfactory and receiver number 1 automatic gain control was adequate prior to launch. Receiver number 2 was not instrumented. Transfer to internal power and arming were accomplished successfully. No commands were required nor were any commands inadvertently generated during the flight by the flight termination or other electrical systems. Manual fuel cutoff commands, destruct commands, and receiver number 1 automatic gain control were the only measurements made of flight termination system performance. The automatic gain control dropped to zero for 0. 3 second at the time of booster engine jettison (T + 132. 2 sec), as expected. The minimum automatic gain control level during the flight except at booster engine jettison was 15 microvolts. The minimum level required for proper receiver operation is 3 microvolts.

VI. AGENA VEHICLE SYSTEM PERFORMANCE

STRUCTURE SYSTEM

The Agena structure system performance was satisfactory. The measured flight loads on the structure were within their expected range.

Description

The structure system consists of four major sections: the forward section, the tank section, the aft section, and the booster adapter. The forward section, constructed of beryllium, magnesium, and aluminum, contains most of the electrical, guidance, and communication equipment, and provides the mechanical and electrical interface for the spacecraft and shroud. The 5-foot- (1.52-m-) diameter tank section is composed of two integral aluminum propellant tanks. The aft section structure, constructed of magnesium and aluminum, provides structural continuity from the tank section to the booster adapter and supports the rocket engine and associated equipment. The magnesium alloy booster adapter supports the Agena and remains with the Atlas at separation.

Performance

The measured flight loads on the structure system were within their expected range. The peak steady-state longitudinal load during flight was 6.13 g's at booster engine cutoff.

The measured flight environment did not show any unexpected excitation, and the shock levels experienced were within expected ranges. Flight dynamics data are presented in appendix D.

SHROUD SYSTEM

ATS-2 was the second flight to use the Standard Agena Clamshell shroud system. The shroud separated successfully after its pyrotechnics were fired at T + 379.9 seconds. All shroud system temperatures and pressures measured during the flight were within acceptable limits.

Description

The shroud system consists of the Standard Agena Clamshell shroud with minor mission peculiar modifications incorporated. As shown in figure III-5, this system includes a transition ring and two shroud halves that form a fairing with a cylindrical section 65 inches (165.1 cm) in diameter, a 15° half-angle cone, and a 12-inch- (30.48-cm-) radius hemispherical dome. The total shroud system weight for ATS-2 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 latch, two flat bands around the cylindrical section, and a V-band at the base of the cylinder.

The V-band clamps the shroud to a 2-inch- (5.08-cm-) high aluminum transition ring bolted to the Agena forward section. The top, middle, and bottom bands 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 at 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 is vented through four ascent vent ports located in the cylindrical portion of the shroud (see fig. III-5). The ascent vent ports are designed to permit venting only in an outward direction. The Agena is vented through holes in its aft equipment area which permit inward as well as outward flow.

Thermal protection for the spacecraft is provided by microquartz thermal insulation blankets in the cylindrical portion of each shroud half and a foil covering on the inner surface of the conical portion of each shroud half. Figure VI-1 shows a simulated-ATS spacecraft mounted on the transition ring with one half-shroud installed. The shroud is instrumented with two temperature transducers on the inner surface of the shroud fiberglass skin at Agena station 176, and with one transducer to measure the differential pressure across the diaphragm.

Approximately 10 seconds after the initiation of Agena first burn, shroud jettison is initiated. At this time, the Agena fires two pyrotechnic bolt cutters in the nose latch assembly and two explosive bolts in each of the bands. Springs at the base of the shroud then force the halves to rotate about hinges mounted on the transition ring. After the center of gravity of each shroud half has rotated over its hinge point, the Agena vehicle acceleration increases the rotation rate of the shroud halves. In the 1-g acceleration field provided by the Agena at the time of shroud separation, each shroud half leaves its hinge and falls free of the vehicle after having rotated approximately 75°. The shroud separation springs provide sufficient energy to jettison the shroud halves at vehicle longitundinal acceleration levels up to 3.5 g's.

The spacecraft "encapsulation" concept was used on ATS-2. The complete shroud system and spacecraft were mated together in a clean room area. The spacecraft-shroud assembly (encapsulated spacecraft) was then transported to the launch pad and mated to the Agena as a unit.

Performance

The internal skin temperature history is shown in figure VI-2. The peak temperature measured was 207° F (370.5 K) which was well below the maximum predicted temperature of 380° F (466.5 K) for a three-sigma depressed trajectory.

The differential pressure measured across the shroud diaphragm in flight is shown in figure VI-3. The differential pressure was essentially zero during the early portion of the flight. During the transonic phase, a differential pressure of -0.7 psi (-0.48 N/cm^2) developed. (A shroud cavity pressure less than the pressure in the Agena forward equipment section is considered a negative pressure differential.) After the transonic phase, the differential pressure became slightly positive for a short period and then returned nearly to zero for the remainder of the flight. The differential pressure of -0.7 psi (-0.48 N/cm^2) was caused by the development of shock waves on the vehicle.

Shroud pyrotechnics were fired at T + 379.9 seconds. At this time, the vehicle roll and yaw rates were nearly zero, and the pitch rate was at the programmed value of -3.21 degrees per second. No measurable Agena roll, pitch, or yaw rates developed as a result of shroud jettison.

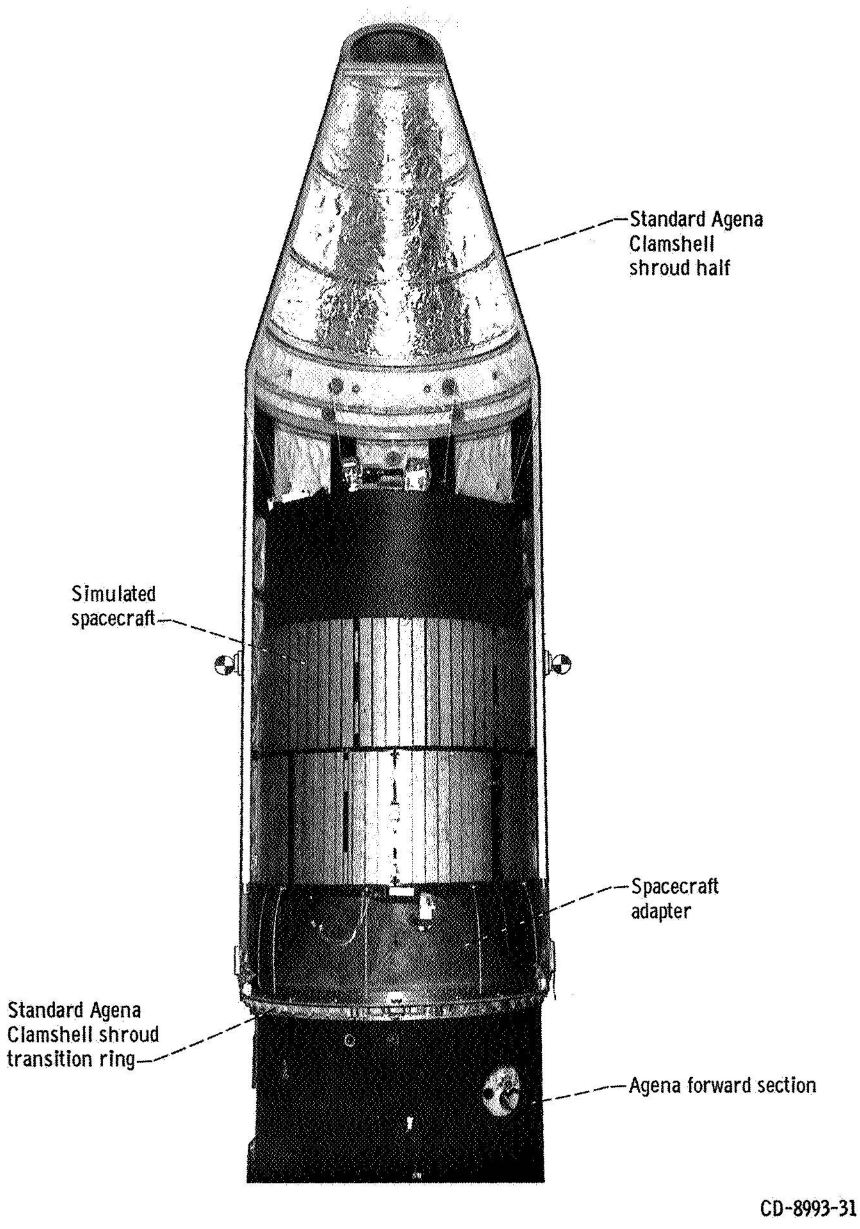


Figure VI-1. - Shroud simulated spacecraft, ATS-2.

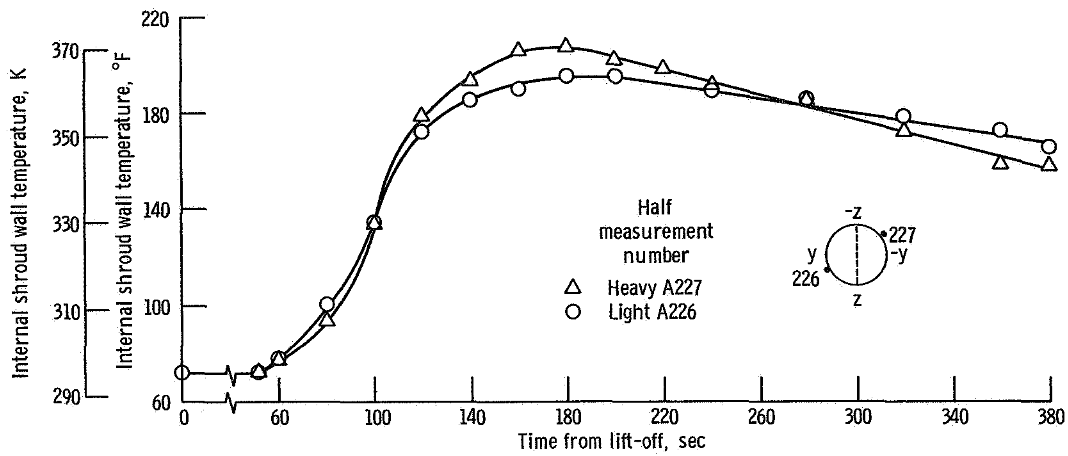


Figure VI-2. - Internal shroud wall temperature history, ATS-2. Agena station 176.

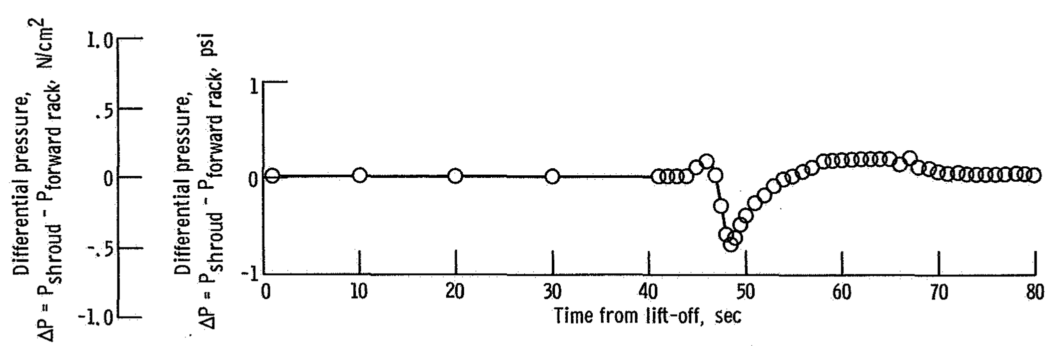


Figure VI-3. - Diaphragm differential pressure history, ATS-2. Measurement number A519.

PROPULSION SYSTEM

The Agena propulsion system performance was satisfactory through the first-burn period. The programmed second burn did not occur because the oxidizer propellant isolation valve failed to fully close. A discussion of the oxidizer propellant isolation valve failure and the resultant effect on the propulsion system performance is presented in appendix E.

Description

The propulsion system consists of a rocket engine, propellant pressurization system, booster adapter retrorockets, and vehicle pyrotechnics. The engine has a regeneratively cooled combustion chamber mounted in a gimbal ring. Two hydraulic actuators are used to move the thrust chamber in response to guidance signals. The engine, which burns unsymmetrical dimethylhydrazine (UDMH) fuel and inhibited red fuming nitric acid (IRFNA) as an oxidizer is designed to generate a thrust of 16 000 pounds (71.171×10^3 N) in vacuum.

Propellant containment sumps are attached to the bottom of each propellant tank. These sumps hold propellants for engine start after the zero-gravity coast is completed. Located between these sumps and the fuel and oxidizer pump inlets are the propellant isolation valves (see appendix E, fig. E-1). These valves are closed between engine firings to isolate the propellant tanks from the engine system and to vent the propellants downstream of the valves. This venting of propellants prevents the formation of vapor in the propellant pumps and engine feed lines during the coast period.

The propellant pressurization system provides sufficient helium gas pressure in both propellant tanks to ensure satisfactory engine operation. Tank pressurization is achieved by allowing helium to flow from a high pressure storage sphere to each propellant tank through a pair of fixed-area flow control orifices.

The booster adapter is separated from the Agena by the cutting action of a length of Primacord around the circumference of the adapter. Two retrorockets mounted on the booster adapter are used to separate the Atlas and booster adapter from the Agena. For this mission, the propulsion system was configured for two burns. Two retrorockets were mounted on the Agena aft section to reduce the Agena orbit altitude after spacecraft separation.

Performance

The Agena first-burn engine ignition sequence was initiated by the primary sequence timer at T + 369.8 seconds. Voltage levels on the engine switch group monitor indicated a normal start sequence. Ninety percent combustion chamber pressure was reached 1.19 seconds later, indicating a normal start transient. The calculated, average, steady-state thrust derived from combustion chamber pressure measurements was 16 426 pounds (73.066×10^3 N). This was slightly higher than the predicted nominal thrust of 16 280 pounds (72.417×10^3 N). First burn was terminated by velocity meter cutoff at T + 577.71 seconds. The actual thrust duration measured from 90 percent chamber pressure to velocity meter cutoff was 206.71 seconds. This was 1.64 seconds shorter than the predicted time of 208.37 seconds. The difference between actual and predicted burn times is consistent with the higher than nominal average thrust level.

Following completion of the Agena first burn, the propellant isolation valves were programmed by the primary timer to close at T + 586.79 seconds. Analysis of propulsion system data indicated that the oxidizer propellant isolation value did not fully close. (Refer to appendix E for further discussion of the propellant isolation valve malfunction.) As a consequence, when the second-burn ignition sequence was initiated at T + 7114.93 seconds, vapor lock occurred within the oxidizer pump. After the gas generator solid propellant starter charge was expended, there was no oxidizer to support combustion in the gas generator and sustain turbopump operation, and combustion in the engine thrust chamber was not initiated. The only impulse added to the Agena vehicle during the period when second burn should have occurred was from the solid propellant starter charge.

The propellant tank pressurization system performance was satisfactory. Propellant tank pressures were normal during the engine first burn and were adequate for second burn if it had occurred.

The Agena pyrotechnics and both retrorockets performed their functions satisfactorily.

ELECTRICAL SYSTEM

The Agena electrical system performance was satisfactory. All electrical system components performed within specification. There was one minor anomaly during this flight. The squibs which actuate the oxidizer fast-shutdown valve indicated short circuits after firing. However, these short circuits were cleared quickly, and they did not affect vehicle performance.

Description

The electrical system supplies all power, frequency, and voltage requirements for the pyrotechnic, propulsion, flight termination, guidance, and telemetry systems. The system consists of the power source equipment, power conversion equipment, and the distribution network.

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

The power conversion equipment accepts unregulated dc power and converts it to regulated ac or 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.

The distribution network distributes, controls, and monitors vehicle power. The wire harnesses, power distribution J-box, etc., make up this network.

Performance

The electrical system voltages and currents were normal at lift-off and met the demands of the using systems throughout flight. A summary of electrical system parameters is shown in table VI-I.

The battery current was approximately 14 amperes dc at lift-off and remained at this value except during Agena engine burn periods when it increased to approximately 17 amperes. The battery current fluctuated approximately 3 amperes as the main gyro heaters cycle on and off. All currents quoted are with the main gyro heaters on.

The guidance converter outputs were essentially constant throughout the flight. The plus output reading averaged 28. 5 volts dc, and the minus output averaged 28. 7 volts dc. The telemetry dc-dc converter output was constant at 28. 2 volts dc throughout the flight. The inverter outputs, phase AB and phase BC, were a constant 116. 1 volts rms throughout the flight.

Components exhibited very little variation in voltage output. The pyrotechnic battery voltage was 25. 3 volts dc at lift-off and remained almost constant throughout the flight. The main vehicle unregulated voltage was 24. 6 volts dc at lift-off and dropped to 24. 4 volts dc at Agena first ignition. The voltage remained at this level throughout the remainder of the flight. As noted on the battery current, the unregulated voltage also fluctuated approximately 0. 5 volt as the main gyro heaters cycle on and off. All unregulated voltages quoted are with the main gyro heaters on.

TABLE VI-I. - AGENA ELECTRICAL SYSTEM FLIGHT DATA, ATS-2

Measurement	Measurement number	Flight values at -						
		Lift-off	Agena first burn		Agena second burn		Spacecraft separation	Second retrorocket ignition
			Ignition	Cutoff	Ignition	Cutoff		
Pyrotechnic battery voltage, V dc	C141	25.3	25.3	25.3	25.3	25.3	25.3	25.3
Main battery voltage,[a] V dc	C1	24.6	24.4	24.4	24.4	24.4	24.4	24.4
Main battery current,[a] A dc	C4	14	17	14	17	14	14	14
Structure current, A dc	C38	0	0	0	0	0	0	0
Guidance and control converter 28.3-V dc regulator	C3	28.5	28.6	28.5	28.4	28.4	28.4	28.4
Guidance and control converter -28.3-V dc regulator	C5	-28.7	-28.7	-28.7	-28.7	-28.7	-28.7	-28.7
Guidance and control inverter phase AB, V rms	C31	116.1	116.1	116.1	116.1	116.1	116.1	116.1
Guidance and control inverter phase BC, V rms	C32	116.1	116.1	116.1	116.1	116.1	116.1	116.1
Telemetry converter 28.3-V dc regulator	H-204	28.2	28.2	28.2	28.2	28.2	28.2	28.2

[a]C1 and C4 are listed with main gyro heaters cycled on; C1 is approximately 0.4 to 0.6 V higher with main gyro heaters off; C4 is approximately 3 A less with main gyro heaters off.

GUIDANCE AND FLIGHT CONTROL SYSTEM

The Agena guidance and flight control system performance was satisfactory throughout flight. All preprogrammed flight events were initiated within tolerance by the sequence timers. The primary sequence timer was started 0.9 second earlier than the nominal time by the radio guidance start Agena timer discrete command to compensate for trajectory dispersions. Therefore, all subsequent timer events were approximately 0.9 second earlier than nominal. A comparison of the nominal and actual times of programmed events is given in appendix A.

Description

The guidance and flight control system performs the vehicle guidance, control, and flight programming functions necessary to accomplish the vehicle mission after Atlas-Agena separation. The system consists of three subsystems corresponding to the guidance, control, and flight programming functions.

The guidance subsystem consists of an inertial reference package, horizon sensors, velocity meter, and guidance junction box. Primary attitude reference is provided by three orthogonal, rate integrating gyroscopes in the inertial reference package. The infrared horizon sensors provide continuous corrections in pitch and roll to the inertial reference package. Yaw attitude reference is obtained from the booster and is corrected by gyrocompassing techniques during long coast periods. The velocity meter counter generates a signal to terminate engine thrust when the vehicle has increased its velocity by a predetermined increment. Longitudinal acceleration is sensed by the velocity meter accelerometer, which counts down the first-burn velocity-to-be-gained binary number in the counting register. The second-burn velocity-to-be-gained number is transferred from a storage register to the counting register after first-burn cutoff.

The flight control subsystem, which controls vehicle attitude, 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 pneumatic and hydraulic systems. During Agena coast periods, the pneumatic system provides roll, pitch, and yaw control by use of six thrusters operating on a mixture of nitrogen and Freon. The hydraulic system provides pitch and yaw control during Agena engine burn. Two hydraulic actuators gimbal the rocket engine thrust chamber. Roll control during engine burn is provided by the pneumatic system. A patch panel in the flight control junction box provides the means for varying the interconnections of the guidance and flight control system to suit mission requirements.

The flight programming subsystem uses sequence timers to program Agena flight events. A sequence timer provides 22 usable discrete event times with multiple switch closure capability and has a maximum running time of 6000 seconds. Two timers (a primary and an auxiliary timer) are required if the Agena mission duration exceeds 6000 seconds or if more than 22 discrete events are required. The primary sequence timer is started by a radio guidance discrete command at a time determined by the trajectory of the booster. The auxiliary sequence timer is started by the primary timer.

Since the ATS-2 Agena mission is longer than 6000 seconds and requires more than 22 discrete events, two sequence timers were used.

Performance

Analysis of postflight data shows that the Atlas-Agena separation induced low body rates to the Agena of 0.25 degree per second pitch up, 0.20 degree per second roll clockwise, and 0.20 degree per second yaw left. (Clockwise and counterclockwise roll reference applies when looking forward along the Agena longitudinal axis, see fig. IV-4.) These body rates, coupled with Atlas attitude errors, resulted in Agena attitude errors of 0.3° pitch up, 0.4° roll clockwise, and 0.4° yaw left at pneumatics activation.

The vehicle stabilized in pitch with a pitch-down-attitude error of 0.5° at the initiation of the first pitch program. The roll attitude remained within the deadband limits of $\pm 0.6^{\circ}$ until initiation of first burn. The yaw-attitude error was reduced to the deadband limits of $\pm 0.18^{\circ}$ within 5 seconds and remained there until initiation of first burn.

The vehicle completed the programmed pitch down of 16°, and the programmed geocentric rate of 3.21 degrees per minute pitch down was applied. The pitch horizon sensor, set at a pitch bias angle of 0.48° (nose up) for first burn, was connected to the pitch gyro, and the vehicle was stabilized in all axes by the time of first-burn ignition.

Gas generator turbine spin-up at Agena first-burn ignition, coupled with the existing low clockwise roll rate, produced a roll rate that reached a maximum of 2.3 degrees per second clockwise. Roll pneumatic activity reduced the clockwise roll rate to zero in 1.2 seconds and started to return the vehicle to the zero degree position. The vehicle overshot in roll 2.6 degrees counterclockwise due to turbine exhaust duct misalinement supplying a counterclockwise torque. The vehicle subsequently returned to the edge of the roll deadband with a 0.2° counterclockwise roll offset.

Hydraulic pressure buildup and off-null engine position at the start of first burn introduced pitch and yaw displacements. Peak displacements of 0.8° pitch down and 0.4° yaw left were evidenced approximately 2.1 seconds after engine start. These displacements were removed by the hydraulic controls, which had stabilized 5.0 seconds after ignition. Hydraulic and pneumatic activity were normal throughout the burn period.

Shroud jettison (initiated 10 sec after first-burn ignition) occurred during the 2.6° counterclockwise roll displacement, but when the vehicle roll rate was approximately zero. The horizon sensors indicated a slight disturbance approximately 2.1 seconds after the shroud separation pyrotechnics fired; however, gyro data in roll and pitch indicate that little or no attitude error was introduced by shroud jettison. These disturbances are attributable to the shroud halves passing through the fields of view of the horizon sensors.

The Agena engine first shutdown was commanded by the velocity meter after the vehicle had attained the required first-burn velocity increment of 12 408 feet per second (3782 m/sec). The additional velocity increment due to engine tailoff thrust was 13.6 feet per second (4.14 m/sec). A normal roll transient, caused by first-burn engine shutdown (turbine spin and turbine exhaust decay), initiated a pneumatic system response resulting in a 1.1° roll counterclockwise excursion which was reduced to within the pneumatic control system deadband limits in 17 seconds.

After the Agena engine first shutdown, the pneumatic attitude control system was transferred from the ascent mode to the orbit mode, the programmed geocentric pitch rate was reduced to 1.66 degrees per minute pitch down, and the horizon sensor bias angle was decreased to 0.32°. Transfer of the required second-burn velocity-to-be-gained number, 4052 feet per second (1235 m/sec), from the storage register to the counting register was normal. Horizon sensor and pneumatics data show that the vehicle maintained the proper attitude in the coast phase with roll limit cycle operation.

Gas generator turbine spin-up at Agena second ignition produced a roll rate that reached a maximum of 2.28 degrees per second clockwise. This rate induced a maximum roll displacement of 2.8 degrees clockwise, before being satisfactorily damped by pneumatic control.

During the engine burn periods, pitch and yaw attitude control is provided by hydraulic gimbaling of the engine. Since the engine failed to attain steady-state operation for second burn, there was neither hydraulic pressure to gimbal the engine (the hydraulic pump is driven by high pressure UDMH from the engine fuel pump) nor engine thrust to provide a control force in pitch and yaw. Consequently, the pitch and yaw rates induced by the attempted second-burn ignition could not be damped.

Undamped rates of 0.57 degree per second pitch down and 1.87 degree per second yaw right were observed from T + 7114.93 sec, when the second-burn ignition squibs were fired, until data were lost when the pitch and yaw displacement gyros moved against their mechanical stops at approximately 10° displacement. If these rates were assumed to remain constant until pitch and yaw pneumatic control was reactivated at T + 7145 sec seconds, the vehicle attitude errors were 17.1° pitch down and 52° yaw right at the time of pneumatic activation.

After pneumatic activation, pitch attitude reference was reestablished by the horizon sensors and the pneumatic control returned the vehicle to a normal pitch attitude within

11 seconds. Pneumatic control moved the vehicle toward its normal yaw attitude, until the yaw displacement gyro returned to null. This allowed for recovery of approximately 10° of the yaw attitude error. Thus, the vehicle stabilized with a 42° yaw-right attitude error, as a result of the loss of yaw reference during the period for second burn. This yaw attitude error existed through the remainder of the flight.

Velocity meter performance during the attempted second burn was normal under the circumstances. From the time of velocity meter enable to cessation of pulsing, the velocity meter indicated that a total velocity increment of 1.82 feet per second (0.55 m/sec) was imparted to the vehicle by the engine start sequence.

Pneumatic control was normal subsequent to reestablishment of the pitch and yaw attitude references. The geocentric pitch rate of 1.66 degrees per minute pitch down was removed and the vehicle yawed right 81° prior to spacecraft separation. However, the yaw attitude at spacecraft separation was 123° right instead of 81° right because of the 42° yaw-right attitude error. The Agena was stable within the pneumatic deadbands at the time of spacecraft separation. Subsequent to spacecraft separation, the vehicle completed a yaw-right maneuver of 99°. Thereafter, minimal pneumatic activity was required to maintain the programmed geocentric pitch rate.

The Agena attitude control gas usage up to telemetry shutdown, 160 seconds after the first retromaneuver, was less than predicted. Data on gas usage during the final coast and second retromaneuver were not available; however, the remaining gas was more than adequate for this period. A comparison of predicted and actual gas usage is given in table VI-II. The gas usage during Agena first burn was slightly greater than the predicted nominal but less than the predicted worst case usage of 11.1 pounds (5.03 kg). Gas consumption during the engine second burn was also greater than nominal; however, the actual usage includes the gas used to reorient and stabilize the Agena when pitch and yaw pneumatics were reactivated.

TABLE VI-II. - AGENA GUIDANCE GAS LOADING

REQUIREMENTS AND USAGE, ATS-2

Flight sequence	Unit	Predicted	Actual
Lift-off	lb	29.7 load	29.15 loaded
	kg	13.5	13.22
Agena first coast	lb	5.0 usage	4.05 used
	kg	2.3	1.84
Agena first firing	lb	5.0 usage	6.90 used
	kg	2.3	3.13
Orbital coast	lb	1.1 usage	0.75 used
	kg	0.5	0.34
Agena second firing	lb	1.4 usage	3.60 used
	kg	0.6	1.63
Prespacecraft separation maneuver	lb	1.3 usage	0.80 used
	kg	0.59	0.36
Postspacecraft separation maneuver	lb	2.7 usage	1.55 used
	kg	1.2	0.70
First retromaneuver	lb	0.4 usage	0.80 used
	kg	0.18	0.36
End of maneuvers	lb	12.8 surplus	10.70 available
	kg	5.8	4.85

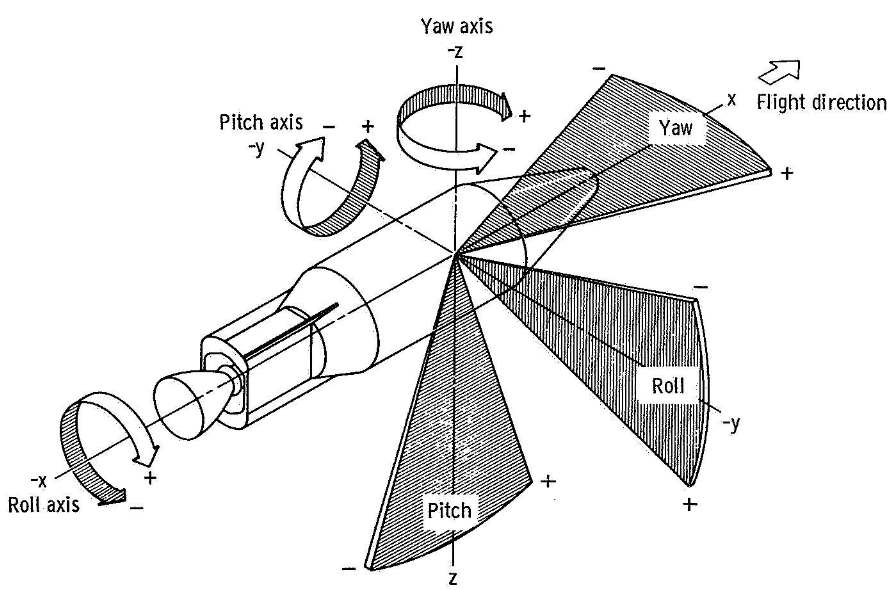


Figure VI-4. - Agena vehicle axes and vehicle movement designations, ATS-2.

COMMUNICATION AND CONTROL SYSTEM

The Agena communication and control systems performance was satisfactory. All subsystem parameters measured during the flight were at the expected levels except for commutated measurement D46 (gas valve cluster temperature 1), which was inoperative prior to lift-off.

Description

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

The telemetry subsystem is mounted in the forward equipment section. It monitors and transmits the Agena functional and environmental 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 on an assigned frequency of 244.3 megahertz at a power output of 10 watts. The telemetry system uses the standard Interrange Instrumentation Group (IRIG) subcarrier channels and has the capacity for 18 subcarriers. Channels 15 and 16 are commutated at 5 revolutions per second with 60 data segments on each channel.

The telemetry was turned off during the coast periods between first and second burn and between the first and second Agena retromaneuvers in order to conserve power and prevent the transmitter from overheating.

The instrumentation subsystem consists of 58 transducers and event monitors. Five continuous subcarrier channels are used for accelerometer data; one carries the gas valve current signals, and another is time shared by the velocity meter accelerometer and velocity meter counter.

The turbine speed signal does not use a subcarrier oscillator but 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 telemeter 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 receives coded signals from the tracking radar on a carrier frequency of 5690 megahertz, and transmits coded responses on a carrier frequency of 5765 megahertz at a minimum pulsed power 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 transponder to either the umbilical for ground checkout or the antenna for flight.

The C-band beacon was turned off for 1065 seconds between first and second burn and for 3700 seconds between the first and second retrorocket firings in order to conserve power.

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 secondary batteries, two destruct initiators, and a destruct charge. These units are connected to provide redundant flight termination capability with the exception of the multicoupler and destruct charge. Flight termination, if necessary, is initiated by a series of commands from the range safety transmitter. The first sequence of commands shuts down the engine, and the second sequence of commands fires the destruct charge. The resultant explosive mixture of hypergolic propellants destroys the vehicle.

Performance

The Agena telemetry subsystem performance was satisfactory throughout the flight. Telemetry stations at Cape Kennedy and Antigua combined to give complete telemetry coverage of the first-burn phase of the mission. Telemetry data for the second-burn phase, spacecraft separation, and the first and second Agena retrorocket firings were provided by the Air Force Satellite Control Facility station at Guam. Signal strength data from all stations showed an adequate and continuous signal level from the vehicle telemetry transmitter. The telemetry data indicated that the performances of the voltage controlled oscillators, switch and calibrate unit, dc-dc converters, and commutator were satisfactory. A description of the tracking and data acquisition network used in support of the ATS-2 flight is given in appendix C.

The telemetry transmitter, was prevented from overheating by being turned off by the onboard timer shortly after the Agena engine first cutoff for a period of 6200 seconds and shortly after the Agena first retromaneuver for a period of 3700 seconds. The telemetry transmitter temperature rose to 104° F (313.3 K) just prior to the first transmitter shutdown and to 117° F (320.5 K) just prior to the second transmitter shutdown. These temperatures are well within the allowable 160° F (344.5 K) operating temperature of the transmitter.

The Agena vehicle instrumentation subsystem performed satisfactorily during the vehicle flight. Useful data were obtained from all transducers and event monitors except gas valve cluster temperature 1 (measurement number D46). This transducer was known to be inoperative prior to lift-off but its replacement was not feasible.

While useful data were obtained from the low-frequency accelerometers (measurement numbers A4, A5, and A9), recurrent step changes of 2 to 3 percent of the full-scale acceleration level were observed in the flight data. These step changes, believed to be accelerometer amplifier bias shifts, occurred during the transonic period of the boost phase (T + 50 sec), at booster engine jettison (T + 132.2 sec), and during the sustainer phase (T + 154 sec). Similar behavior of this type of accelerometer (unbonded strain gage) had been noted on several previous flights, but as yet no suitable explanation for this anomaly has been found.

The tracking subsystem performance was satisfactory throughout the flight. The C-band transponder transmitted a continuous response to received interrogations for the required tracking periods.

Both flight termination receivers functioned satisfactorily during prelaunch tests and flight. The received signal strength remained well above the airborne receiver threshold of 2 microvolts through first burn. At lift-off, both receivers indicated a received signal strength greater than 10 microvolts. The signal level remained essentially constant throughout the period in which destruct capability was required. The data indicated that the vehicle was receiving adequate signal strength for the operation of the flight termination subsystem.

VII. LAUNCH OPERATIONS

PRELAUNCH ACTIVITIES

The Agena, Atlas, and spacecraft arrived at Eastern Test Range January 27, February 10, and March 16, 1967, respectively. A calendar of the major prelaunch activities at Launch Complex 12 is shown in table VII-I. All launch vehicle prelaunch tests were completed satisfactorily. Only minor operational and ground equipment problems were encountered.

During the Booster Flight Acceptance Test (B-Fact) number 2, the Guided Missile Control Facility did not receive the first motion pulse. A broken wire was found in the electrical distribution box (incoming) at the facility. The wire was repaired and the first motion pulse was received satisfactorily.

During the simulated launch demonstration, the following problems were encountered:

(1) A near redline condition (low pressure) was observed on the holddown head pressure recorders in the blockhouse. A check of the instrumentation circuit revealed that the recorders had drifted, giving a low reading. The recorder setting was adjusted to correct the problem.

(2) The Agena propellant isolation valve position monitor meter in the blockhouse gave an erroneous position indication. It indicated that only the oxidizer propellant isolation valve was open, when both the fuel and the oxidizer propellant isolation valves had been opened. An aerospace ground equipment amplifier in the circuitry to the position monitor meter was replaced to correct this problem. After the amplifier was installed, its gain was adjusted so that the position monitor meter indicated that both propellant isolation valves were in an open position.

COUNTDOWN AND LAUNCH

ATS-2 was successfully launched on April 5, 1967, approximately 44 minutes after the scheduled lift-off time. Since the aerospace ground equipment amplifier in the propellant isolation valve position monitor circuitry was replaced after the simulated launch demonstration, a special test was required to verify the position of the Agena propellant isolation valves and the proper operation of the propellant isolation valve position monitor meter in the blockhouse. This test was conducted during the launch countdown. The propellant isolation valves were cycled closed and then open. The propellant isolation valves and position monitor meter functioned properly.

During the countdown there were two problems which resulted in unscheduled holds:

(1) The scheduled hold at T - 7 minutes was extended by 33 minutes because of an erratic change in the aerospace ground equipment level of a rate receiver, a part of the Mod III ground guidance station. The received signal level from one of the outlying rate stations dropped from -85 to -95 dBm (decibels referenced to a 1-mW power load) for a 1.3-second period. A coaxial cable at the remote station was thought to be responsible for the anomaly. The cable was replaced, the system was reverified, and the countdown was resumed. Subsequent to the flight, analysis of the suspect cable revealed that it was not faulty. The cause of the problem was then determined to have been a diode which exhibited intermittent operation due to excessive tension on its leads. This diode was located, in the aerospace ground equipment panel of the receiver, at the north rate-leg receiver station.

(2) At approximately T - 90 seconds, a redline was called when the water pressure, supplying the Complex 12 flame bucket, indicated low pressure. The count was recycled to T - 7 minutes. At this time it was determined that a low pressure condition is normal when the water, which supplies the flame bucket, is first turned on. The countdown was then resumed at T - 7 minutes, after a hold of 5.5 minutes, and proceeded to lift-off without further incident.

During the launch countdown, aerospace ground equipment performance was satisfactory.

TABLE VII-I. - PRELAUNCH ACTIVITIES AT LAUNCH COMPLEX 12, ATS-2

Date	Event
2/14/67	Atlas erection
3/2/67	Booster Flight Acceptance Test (B-Fact) Number 1
3/8-9/67	Atlas dual propellant tanking test
3/21/67	Booster Flight Acceptance Test (B-Fact) Number 2
3/21/67	Agena mate
3/28/67	Joint Flight Acceptance Composite (J-Fact) Test
3/30/67	Spacecraft mate
3/31/67	Simulated launch demonstration
4/3/67	Spacecraft demate
4/5/67	Spacecraft mate
4/5/67	Launch

VIII. CONCLUDING REMARKS

The Atlas-Agena flight operation was successful in supporting its secondary objective, the "heat pipe" experiment (Air Force Research Payload Module 481) but failed to achieve the primary objective to place the Applications Technology Satellite ATS-2 in a proper 6000-nautical-mile (11 112-km) circular orbit. The Agena vehicle remained in its transfer orbit (perigee, 185 km; apogee, 11 112 km) when the Agena engine failed to restart. The Standard Agena Clamshell shroud, which was flown for the second time, again demonstrated satisfactory performance. The Mod III Radio Guidance System performed satisfactorily, and the tracking subsystem acquired the vehicle in the first acquisition cube as planned.

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

APPENDIX A

SEQUENCE OF FLIGHT EVENTS

Nominal time, sec	Actual time,[a] sec	Event description	Source	Event monitor[b]
0	0	Lift-off (2223:01.901 EST)		2-in. (5.08-cm) motion switch
128.5	129.4	Atlas booster cutoff	Atlas guidance	Longitudinal accelerometer (A9)
292.6	289.0	Atlas sustainer cutoff	↓	Longitudinal accelerometer (A9)
296.7	295.8	Start Agena timer (primary)	↓	Guidance and control (D14)
312.4	309.5	Atlas vernier cutoff	↓	↓
		Uncage Agena gyros	↓	↓
		Jettison horizon sensor fairings	↓	↓
		Arm Atlas Agena Separation	↓	↓
314.5	312.0	Atlas-Agena Separation	↓	↓
317.0	314.0	Activate pneumatic attitude control	Separation switch	↓
		Connect horizon sensor roll to roll gyro	Separation switch	(c)
343.7	342.9	Initiate -120 deg/min pitch rate	Primary timer	Pitch torque rate (D73)
351.7	350.9	Transfer to -3.21 deg/min pitch rate	↓	Pitch torque rate (D73)
		Connect pitch horizon sensor to pitch gyro	↓	(c)
		Transfer telemetry to velocity meter accel-erometer output	↓	Velocity meter accelerometer (D83)
370.7	369.7	Arm engine control and deactivate pitch and yaw pneumatics	↓	Switch group Z (B13)
		Fire first-burn ignition squibs	↓	Velocity meter accelerometer (D83)
		Enable velocity meter	↓	Velocity meter accelerometer (D83)
371.8	371.0	Agena steady-state thrust (90 percent chamber pressure)	↓	Chamber pressure (B91)

[a]Event times monitored on A52, D14, D51, and D73 are based on data sampled every 0.2 sec. Event times monitored on B13 are based on data sampled every 0.5 sec. Event times monitored on A9, A520, and D83/D88 are based on continuous data.

[b]All events except lift-off and those noted are monitored on Agena telemetry. The designation in parenthesis is the monitor measurement designation. See Agena telemetry schedule (appendix B) for measurement range and channel assignment.

[c]No direct measurement for these events.

Nominal time, sec	Actual time,[a] sec	Event description	Source	Event monitor[b]
372.2	371.2	Fire helium pressure squibs	Primary timer	(c)
380.7	379.9	Fire shroud separation squibs		Shroud separation (A52)
568.7	567.9	Arm engine shutdown		(c)
		Arm Agena command destruct disarm		(c)
580.2	577.7	Agena engine shutdown	Velocity meter	Velocity meter accelerometer (D83)
		Activate pitch and yaw pneumatics		(c)
		Fire oxidizer fast shut-down squibs		(c)
587.7	586.79	Transfer telemetry to velocity meter counter output	Primary timer	Velocity meter accelerometer (D83)
		Start telemetry cali-bration		All continuous channels
		Remove 28 volts from pitch and yaw pneu-matics and engine con-trol circuit		(c)
		Close propellant isolation valves[d]		(c)
		Disable velocity meter		Velocity meter accelerometer (D83)
616.7	616.0	Stop telemetry calibrate and disarm command destruct circuit		Pitch torque rate (D73)
		Transfer pneumatics to low pressure		(c)
		Transfer flight control to orbit mode and start gyrocompassing		
		Activate decoupling loops		
		Transfer to -1.66 deg/min pitch rate		Pitch torque rate (D73)
		Transfer telemetry gyro signal conditioners to low range		Pitch torque rate (D73)

[a]Event times monitored on A52, D14, D51, and D73 are based on data sampled every 0.2 sec. Event times monitored on B13 are based on data sampled every 0.5 sec. Event times monitored on A9, A520, and D83/D88 are based on continuous data.

[b]All events except lift-off and those noted are monitored on Agena telemetry. The designation in parenthesis is the monitor measurement designation. See Agena telemetry schedule (appendix B) for measurement range and channel assignment.

[c]No direct measurement for these events.

[d]Oxidizer propellant isolation valve did not close.

Nominal time, sec	Actual time,[a] sec	Event description	Source	Event monitor[b]
619.7	619.0	Transfer to second-burn ΔV	Primary timer	Velocity meter counter (D88)
		Fire horizon sensor 0.32° position squibs		Velocity meter counter (D88)
749.7	749.0	Fire close helium valve squib		(e)
		Start auxiliary timer		
751.7	751.0	Stop primary timer	Auxiliary timer	
		Turn telemetry and beacon off		
1816.7	1816.0	Start primary timer and beacon on		
1818.7	1818.0	Stop auxiliary timer	Primary timer	
6951.7	6951.0	Turn on telemetry		Telemetry signal strength
7062.7	7062.0	Flight control to ascent mode and remove gyrocompassing		Pitch torque rate (D73)
		Deactivate decoupling loops; transfer pneumatics to high pressure		(c) (c)
		Telemetry gyro signal conditioner to high range		
7113.7	7112.93	Enable velocity meter		Velocity meter accelerometer (D83)
		Transfer telemetry to velocity meter accelerometer		Velocity meter accelerometer (D83)
		Open propellant isolation valves[f]		(c)
7115.7	7114.93	Arm engine control and deactivate pitch and yaw pneumatics		Switch group Z (B13)
		Fire second-burn ignition squibs		Velocity meter accelerometer (D83)
7116.8	(g)	Agena steady-state thrust (90 percent chamber pressure)		Chamber pressure (B91)

[a]Event times monitored on A52, D14, D51, and D73 are based on data sampled every 0.2 sec. Event times monitored on B13 are based on data sampled every 0.5 sec. Event times monitored on A9, A520, and D83/D88 are based on continuous data.

[b]All events except lift-off and those noted are monitored on Agena telemetry. The designation in parenthesis is the monitor measurement designation. See Agena telemetry schedule (appendix B) for measurement range and channel assignment.

[c]No direct measurement for these events.

[e]No telemetry station coverage for these events.

[f]Oxidizer propellant isolation valve already in partly open position.

[g]Agena second burn did not occur.

Nominal time, sec	Actual time,[a] sec	Event description	Source	Event monitor[b]
7139.7	(g)	Arm engine shutdown	Primary timer	(c)
7143.0	(g)	Engine cutoff	Velocity meter	Velocity meter accelerometer (D83)
		Pitch and yaw pneumatics on	Velocity meter	
7145.7	7145.0	Engine cutoff (backup)	Primary timer	
		Disable velocity meter		
		Transfer telemetry to velocity meter counter output		
		Start telemetry calibrate		All continuous channels
		Remove power from pitch and yaw pneumatics control and engine control circuits		(c)
7161.7	7161.0	Restart auxiliary timer		
		Stop telemetry calibrate		
		Switch telemetry to velocity meter accelerometer output		Velocity meter accelerometer (D83)
		Transfer telemetry gyro signal conditioning to low range		(c)
7204.7	7204.0	Remove -1.66 deg/min geocentric rate	Auxiliary timer	Pitch torque rate (D73)
7205.7	7205.0	Start 180 deg/min yaw rate		Yaw torque rate (D51)
7232.7	7232.0	Stop 180 deg/min yaw rate		Yaw torque rate (D51)
7242.7	7242.0	Fire spacecraft separation squibs	Primary timer	Accelerometer spacecraft adapter (A520)
7245.7	7245.0	Start 180 deg/min yaw rate		Yaw torque rate (D51)
7278.7	7278.0	Stop 180 deg/min yaw rate		Yaw torque rate (D51)
7352.7	7352.0	Stop primary timer	Auxiliary timer	(c)
7942.7	7942.0	Fire retrorocket squib number 1	Auxiliary timer	Velocity meter accelerometer (D83)

[a]Event times monitored on A52, D14, D51, and D73 are based on data sampled every 0.2 sec. Event times monitored on B13 are based on data sampled every 0.5 sec. Event times monitored on A9, A520, and D83/D88 are based on continuous data.

[b]All events except lift-off and those noted are monitored on Agena telemetry. The designation in parenthesis is the monitor measurement designation. See Agena telemetry schedule (appendix B) for measurement range and channel assignment.

[c]No direct measurement for these events.

[g]Agena second burn did not occur.

Nominal time, sec	Actual time,[a] sec	Event description	Source	Event monitor[b]
8 061.7	8 061.0	Transfer flight control to orbit mode	Auxiliary timer	(c)
		Start gyrocompassing and activate decoupling loops		
		Switch to nose-aft gyro-compassing		
		Transfer to 1.0 deg/min geocentric rate		Pitch torque rate (D73)
		Pneumatics to low pressure		(c)
8 101.7	8 101.0	Shutdown telemetry and beacon		Telemetry signal strength
11 801.7	11 801.0	Turn telemetry and beacon on		Telemetry signal strength
12 001.7	12 001.0	Transfer flight control to ascent mode and remove gyrocompassing		Pitch torque rate (D73)
		Remove decoupling and switch to nose forward polarity		(c)
		Pneumatics to high pressure		(c)
12 031.7	12 031.0	Fire retrorocket squib number 2		Velocity meter accelerometer (D83)
12 081.7	12 081.0	Remove all power from vehicle except C-band beacon		Telemetry signal strength

[a]Event times monitored on A52, D14, D51, and D73 are based on data sampled every 0.2 sec. Event times monitored on B13 are based on data sampled every 0.5 sec. Event times monitored on A9, A520, and D83/D88 are based on continuous data.

[b]All events except lift-off and those noted are monitored on Agena telemetry. The designation in parenthesis is the monitor measurement designation. See Agena telemetry schedule (appendix B) for measurement range and channel assignment.

[c]No direct measurement for these events.

APPENDIX B

LAUNCH VEHICLE TELEMETRY INSTRUMENTATION SCHEDULE

TABLE B-I. - ATLAS INSTRUMENTATION SCHEDULE, ATS-2

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
A743T	Ambient temperature at sustainer instrument panel	11-41	-50° to 550° F	227.7 to 561 K
A745T	Ambient temperature at sustainer fuel pump	11-45	-50° to 550° F	227.7 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	15-3	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 cycle, ac, phase A	18-11	105 to 125 V dc	
E52V	400 cycle, ac, phase B	18-29	105 to 125 V ac	
E53V	400 cycle, ac, phase C	18-41	105 to 125 V ac	
E95V	28 V dc guidance power in	13-15	20 to 35 V dc	
E96V	115 V ac, phase A reference to guidance system	13-37	105 to 125 V ac	
E151V	400 cycle, phase A waveform	10	0 to 150 V ac	
F1P	Liquid-oxygen tank helium pressure (absolute)	15-9	0 to 50 psi	0 to 35 N/cm^2
F3P	Fuel tank helium pressure (absolute)	15-11	0 to 100 psi	0 to 69 N/cm^2
F116P	Differential pressure across bulkhead	18-13/43	0 to 25 psi	0 to 17 N/cm^2
F125P	Booster control pneumatic regulator output pressure (absolute)	13-21	0 to 1000 psi	0 to 690 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 pressure (absolute)	13-1	0 to 800 psi	0 to 552 N/cm^2
F291P	Sustainer control helium bottle pressure (absolute)	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	

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

[b]Items determined from step change in voltage.

TABLE B-I. - Continued. ATLAS INSTRUMENTATION SCHEDULE, ATS-2

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
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	
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	Booster 1 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 414 N/cm^2
H601P	Sustainer hydraulic return line pressure (absolute)	18-7/37	0 to 600 psi	0 to 414 N/cm^2
M79A	Missile axial acceleration fine	7	-0.5 to 0.5 g	
M30X	Missile 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	9900 to 11 200 rpm	
P529D	Sustainer main liquid-oxygen valve	13-43	0 to 90 deg	
P830D	Sustainer fuel valve position	13-35	22.3 to 53.7 deg	
P1P	Booster 1 liquid-oxygen pump inlet pressure (absolute)	18-9	0 to 150 psi	0 to 103 N/cm^2
P2P	Booster 1 fuel pump inlet pressure (absolute)	13-31	0 to 100 psi	0 to 69 N/cm^2

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

[b] Items determined from step change in voltage.

TABLE B-I. - Continued. ATLAS INSTRUMENTATION SCHEDULE, ATS-2

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI units
P6P	Sustainer thrust chamber pressure (absolute)	18-3/33	0 to 1000 psi	0 to 690 N/cm^2
P26P	Booster liquid-oxygen regulator reference pressure (absolute)	13-17	500 to 1000 psi	345 to 690 N/cm^2
P27P	Vernier fuel tank pressure (absolute)	13-39	0 to 1000 psi	0 to 690 N/cm^2
P28P	Vernier 1 thrust chamber pressure (absolute)	18-15	0 to 400 psi	0 to 276 N/cm^2
P29P	Vernier 2 thrust chamber pressure (absolute)	18-17	0 to 400 psi	0 to 276 N/cm^2
P30P	Vernier liquid-oxygen tank pressure (absolute)	13-53	0 to 1000 psi	0 to 690 N/cm^2
P47P	Vernier 1 liquid-oxygen inlet pressure (absolute)	13-45	0 to 600 psi	0 to 414 N/cm^2
P49P	Vernier 1 fuel inlet pressure (absolute)	13-49	0 to 600 psi	0 to 414 N/cm^2
P55P	Sustainer fuel pump inlet pressure (absolute)	13-5	0 to 100 psi	0 to 69 N/cm^2
P56P	Sustainer liquid-oxygen pump inlet pressure (absolute)	18-5	0 to 150 psi	0 to 103 N/cm^2
P59P	Booster 2 thrust chamber pressure (absolute)	18-19	0 to 800 psi	0 to 552 N/cm^2
P60P	Booster 1 thrust chamber pressure (absolute)	18-21	0 to 800 psi	0 to 552 N/cm^2
P100P	Booster gas generator combustion chamber pressure (absolute)	15-51	0 to 600 psi	0 to 414 N/cm^2
P330P	Sustainer fuel pump discharge pressure (absolute)	15-55	0 to 1500 psi	0 to 1034 N/cm^2
P339P	Sustainer gas generator discharge pressure (absolute)	18-55	0 to 800 psi	0 to 1552 N/cm^2
P344P	Sustainer liquid-oxygen regulator reference pressure (absolute)	13-19	500 to 1000 psi	345 to 690 N/cm^2
P15T	Engine compartment air temperature	11-35	-50° to 550° F	227.7 to 561 K
P16T	Engine component temperature	11-55	0° to 400° F	255.6 to 477.5 K
P117T	Booster 2 fuel pump inlet temperature	11-53	0° to 100° F	255.6 to 311 K
P530T	Sustainer cutoff relay	11-1	-300° to -270° F	89 to 105.5 K
P671T	Thrust section ambient temperature, Quadrant IV	11-15	-50 to 550° F	227.7 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°	

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

[b]Items determined from step change in voltage.

TABLE B-I. - Continued. ATLAS INSTRUMENTATION SCHEDULE, ATS-2

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
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 yaw	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°	
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 dc	
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 Spare Booster jettison Enable discretes	16-29	0 to 28 V dc	
S291X	Programmer output Booster engine cutoff Sustainer engine cutoff Vernier engine cutoff	16-31	0 to 28 V dc	
S359X	Booster staging backup	5-S	(c)	
S384X	Spin motor test output	15-5	0 to 5 V dc	
U101A	Axial acceleration	12	0 to 10 g's	
U30P	Liquid-oxygen tank head differential pressure	16-11	0 to 5 psi	0 to 3.4 N/cm^2

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

[b]Items determined from step change in voltage.

[c]Turbine speed signal does not utilize subcarrier channel, but directly modulates transmitter during engine operation.

TABLE B-I. - Concluded. ATLAS INSTRUMENTATION SCHEDULE, ATS-2

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
U81P	Fuel tank head differential pressure	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)	

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

[b]Items determined from step change in voltage.

TABLE B-II. - AGENA INSTRUMENTATION SCHEDULE, ATS-2

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^{o} to 500^{o} F	273.5 to 533.5 K
A227	Shroud inside temperature	16-33	32^{o} to 500^{o} F	273.5 to 533.5 K
A519	Shroud cavity differential pressure	16-12/23/34/53	-5 to 5 psi	-3.4 to 3.4 N/cm^2
A520	Spacecraft adapter longitudinal vibration	18	-20 to 20 g's	
A524	Spacecraft adapter radial vibration	17	-20 to 20 g's	
B1	Fuel pump inlet pressure (gage)	15-15	0 to 100 psi	0 to 69 N/cm^2
B2	Oxidizer pump inlet pressure (gage)	15-17	0 to 100 psi	0 to 69 N/cm^2
B11	Oxidizer venturi inlet pressure (absolute)	15-19/49	0 to 1500 psi	0 to 1034 N/cm^2
B12	Fuel venturi inlet pressure (absolute)	15-23/53	0 to 1500 psi	0 to 1034 N/cm^2
B13	Switch group Z	15-7/22/37/52	(b)	
B31	Fuel pump inlet temperature	15-6	0^{o} to 100^{o} F	255.5 to 311 K
B32	Oxidizer pump inlet temperature	15-8	0^{o} to 100^{o} F	255.5 to 311 K
B35	Turbine speed	(c)	(c)	
B91	Combustion chamber pressure number 3 (gage)	15-4/34	475 to 550 psi	327 to 379 N/cm^2
C1	28 V dc unregulated supply	16-40	22 to 30 V dc	
C3	28 V dc regulator (guidance and control)	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 (guidance and control)	15-30	-30 to -22 V dc	
C21	400 Hz, three phase; inverter temperature	15-14	0^{o} to 200^{o} F	255.5 to 366.5 K

[a] First number indicates 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 step change in voltage.

[c] Turbine speed signal does not utilize subcarrier channel, but directly modulates transmitter during engine operation.

TABLE B-II. - Continued. AGENA INSTRUMENTATION SCHEDULE, ATS-2

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
C31	400 Hz, three phase; bus phase AB voltage	15-18	90 to 128 V ac	
C32	400 Hz, three phase; bus phase BC voltage	15-20	90 to 128 V ac	
C38	Structure current monitor	15-10/25/40/55	0 to 50 A	
C141	Pyrotechnic battery 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	
D46	Gas valve cluster temperature 1	15-39	-50° to 150° F	227.7 to 339 K
D47	Gas valve cluster temperature 2	15-36	-50° to 150° F	227.7 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	227.7 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 2758 N/cm^2
D60	Hydraulic oil pressure (gage)	15-21	0 to 4000 psi	0 to 2758 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	

[a]First number indicates 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 step change in voltage.

TABLE B-II. - Concluded. AGENA INSTRUMENTATION SCHEDULE, ATS-2

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			U.S. Customary Units	SI Units
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^{o} to 155^{o} F	255.5 to 341 K
D149	Gas valves 1 to 6 current	7	(d)	
H47	Beacon receiver pulse repetition frequency	15-27	0 to 1600 pulse/sec	
H48	Beacon transmitter pulse repetition frequency	15-28	0 to 1600 pulse/sec	
H101	Safe-arm-fire destruct number 1	16-2	(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^{o} to 170^{o} F	283.3 to 350 K
H354	Destruct receiver number 1 signal level	16-6	0 to 40 V	
H364	Destruct receiver number 2 signal level	16-8	0 to 40 V	

[a]First number indicates 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 step change in voltage.

[d]Unique voltage level is associated with any one or combination of gas jet activity.

APPENDIX C

TRACKING AND DATA ACQUISITION

The launch vehicle trajectory, as projected on a world map, is shown in figure C-1. Eastern Test Range (ETR), Manned Space Flight Network (MSFN), and Air Force Satellite Control Facility (AF/SCF) data acquisition facilities provided radar data and telemetry data coverage of the launch vehicle. Data coverage from lift-off through Agena engine first cutoff was provided by the ETR stations at Cape Kennedy, Grand Bahama Island, Grand Turk Island, Antigua, and the MSFN station at Bermuda. Data coverage for the planned Agena engine second-burn interval, spacecraft separation and Agena first retromaneuver was provided by the AF/SCF station at Guam and the MSFN stations at Tananarive, Malagasy Republic; Carnarvon, Australia; Guam; and Kauai Island, Hawaii. The MSFN stations at Guam, Hawaii, and Carnarvon, and the AF/SCF station at Guam, provided data coverage for the Agena second retromaneuver.

The AF/SCF was called on to support this mission because signal strength calculations indicated only marginal reception of the Agena telemetry link during the second-burn interval at the MSFN sites. However, postflight analysis of the telemetry data received by the MSFN sites during this flight interval revealed the data to be of usable quality. The AF/SCF Indian Ocean Station (IOS), located at Mahe (Seychelles Islands), was also scheduled to provide telemetry data coverage support; however, technical problems at the site precluded the station from receiving adequate signal strength for valid data recording.

Telemetry Data

Telemetry signals from the Atlas-Agena launch vehicle were recorded on magnetic tape at the supporting tracking stations during all engine operations, Agena preseparation yaw maneuver, spacecraft separation, Agena pre-retro-yaw maneuver, and the two Agena retromaneuvers. These recorded data were used for postflight analysis of the launch vehicle performance.

Real-time monitoring of specific Atlas and Agena parameters was required for verification of the occurrence of significant flight events. A submarine cable linking the ETR uprange stations to Cape Kennedy permitted real-time monitoring of the vehicle telemetered signals through Agena engine first cutoff. The subsequent flight events were monitored by the downrange ETR, MSFN, and AF/SCF stations and the time of occurrence reported to Cape Kennedy in "near" real time by voice communication circuits.

In addition, selected critical Agena telemetered data were retransmitted from the AF/SCF Guam tracking station to Cape Kennedy immediately following the Agena first retromaneuver, for "quick-look" evaluation of vehicle performance. These data confirmed the voice report that the Agena engine second burn did not occur.

Figure C-2 shows the specific telemetry coverage provided by the ETR, MSFN, and AF/SCF tracking stations.

Radar Data

C-band radar metric data (time, elevation, azimuth, and range) were required for real-time operations and postflight analysis. Real-time radar data were required for monitoring the launch vehicle flight performance for range safety purposes, and for assisting the downrange stations in acquiring track of the vehicle. These data were also used for computation of injection conditions at Agena engine first cutoff transfer orbital parameters, and the final Agena orbital parameters following the Agena final retromaneuver.

The specific radar coverage provided by the ETR and MSFN stations is presented in figure C-3.

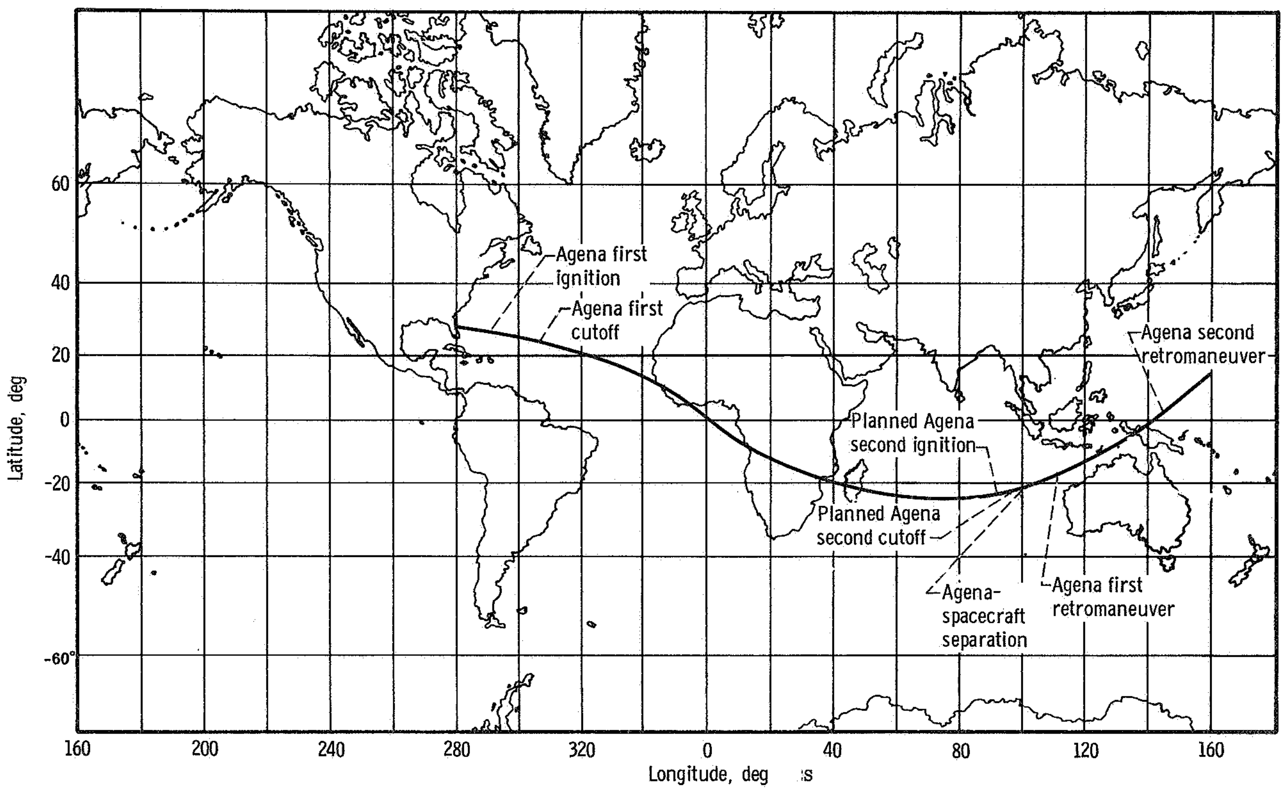


Figure C-1. - Planned ground trace for ATS-2.

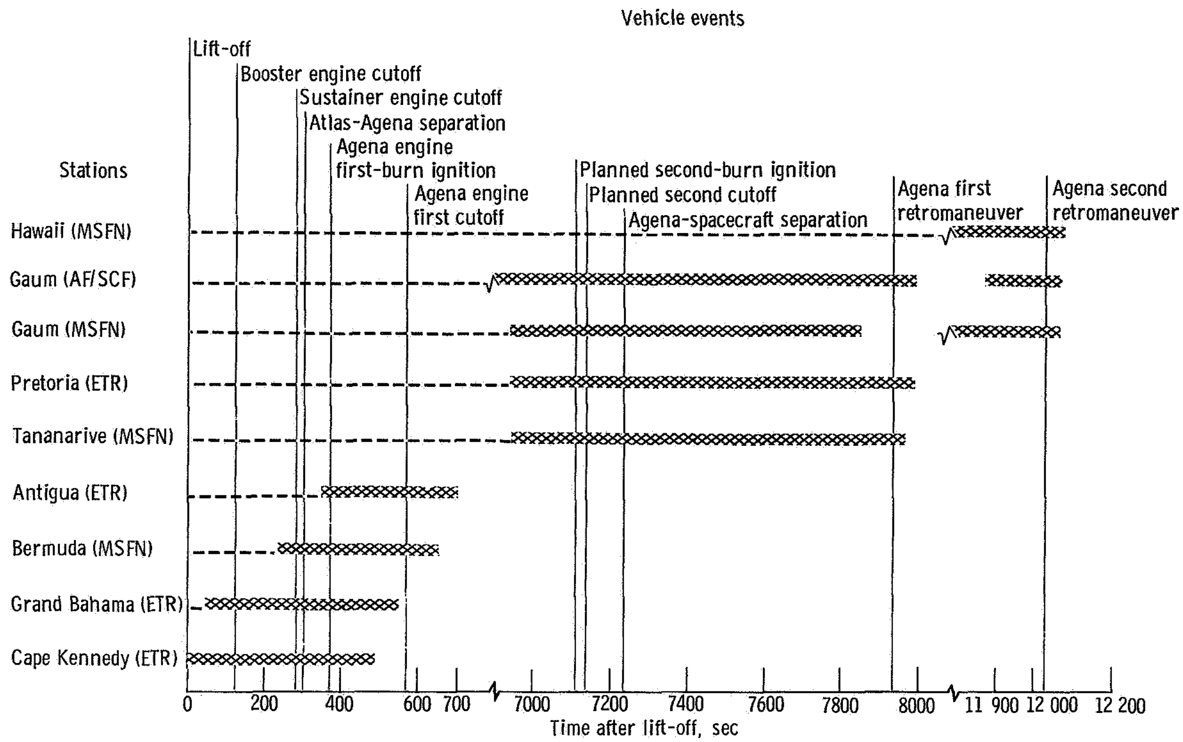


Figure C-2. - Telemetry coverage, ATS-2.

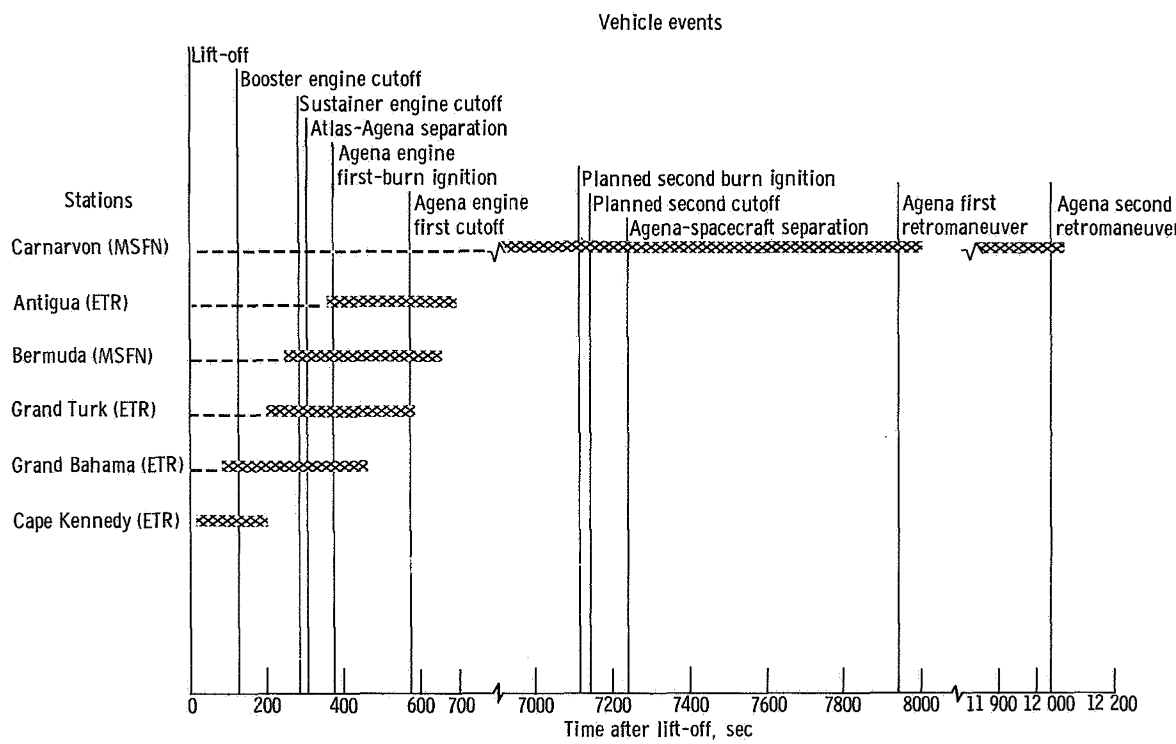


Figure C-3. - Radar coverage, ATS-2.

APPENDIX D

VEHICLE FLIGHT DYNAMICS

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

Table D-I presents the actual flight times at which significant dynamic disturbances were recorded. Table D-II shows the maximum acceleration levels and corresponding frequencies recorded at times of significant dynamic disturbances during flight. All acceleration levels are shown in g's zero to peak.

Dynamic environment data recorded for the preceding flight times are presented in figures D-2 to D-10.

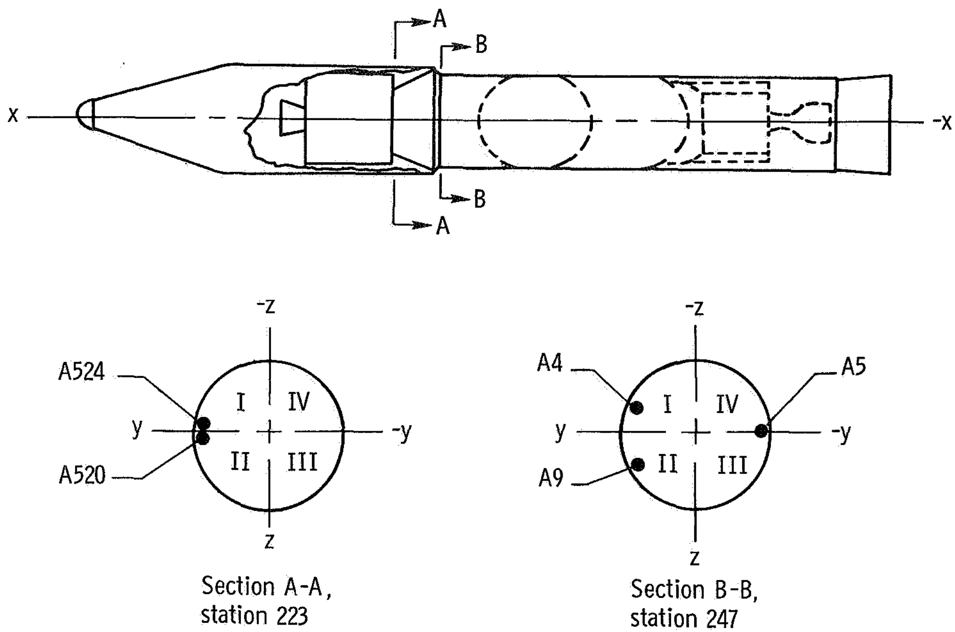
TABLE D-I. - SUMMARY OF DYNAMIC DISTURBANCES, ATS-2

Event causing disturbance	Time of dynamic distrubance, sec after lift-off
Lift-off	0
Transonic	49.60
Booster engine cutoff	129.63
Sustainer engine cutoff	289.10
Horizon sensor fairing jettison	309.54
Atlas-Agena separation	311.82
Agena engine ignition	370.95
Shroud separation	379.67
Agena engine cutoff	577.74

TABLE D-II. - SUMMARY OF DYNAMIC ENVIRONMENT, ATS-2

Event causing disturbance	Time of dynamic disturbance, sec after lift-off	Accelerometer						Vibrometer			
		Channel 8		Channel 9		Channel 11		Channel 17		Channel 18	
		Measurement									
		A9 Longitudinal		A4 Tangential		A5 Tangential		A524 Radial		A520 Longitudinal	
		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 (2223:01.901 EST)	0	[a]70 5	[a]0.3 .3	[a]114 5.5	[a]0.2 .5	[a]180 6	[a]0.3 .5	540	12.0	High	10.4
Transonic	49.60	54	.2	200	.2	200	.5	540	19.2	760	18.0
Booster engine cutoff	129.63	58	.3	65	1.0	65	1.3	64	.4	56	.4
Sustainer engine cutoff	289.10	56	.1	28	.2	28	.1	84	.1	64	.4
Horizon sensor fairing jettison	309.54	0.05-sec pulse	4.0	80	.2	220	2.0	500	Over 20	740	Over 20
Atlas-Agena separation	311.82	0.03-sec pulse	1.2	104	.1	220	.3	660	Over 20	High	Over 20
Agena engine ignition	370.95	56	.5	85	.2	40	.2	540	2.0	High[a] 28	[a]4.0 1.0
Shroud separation	379.67	0.03-sec pulse	5.6	72	.2	220	.2	560	Over 20	High	Over 20
Agena engine cutoff	577.74	62	.1	46	.6	46	.4	520	5.0	38	1.6

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



Chan-nel	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 vibration	A520 at station 223	20 to 2000	±20	±x, + y-axis
17	Radial vibration	A524 at station 223	20 to 1500	±20	±y, + y-axis

Figure D-1. - Flight instrumentation, ATS-2.

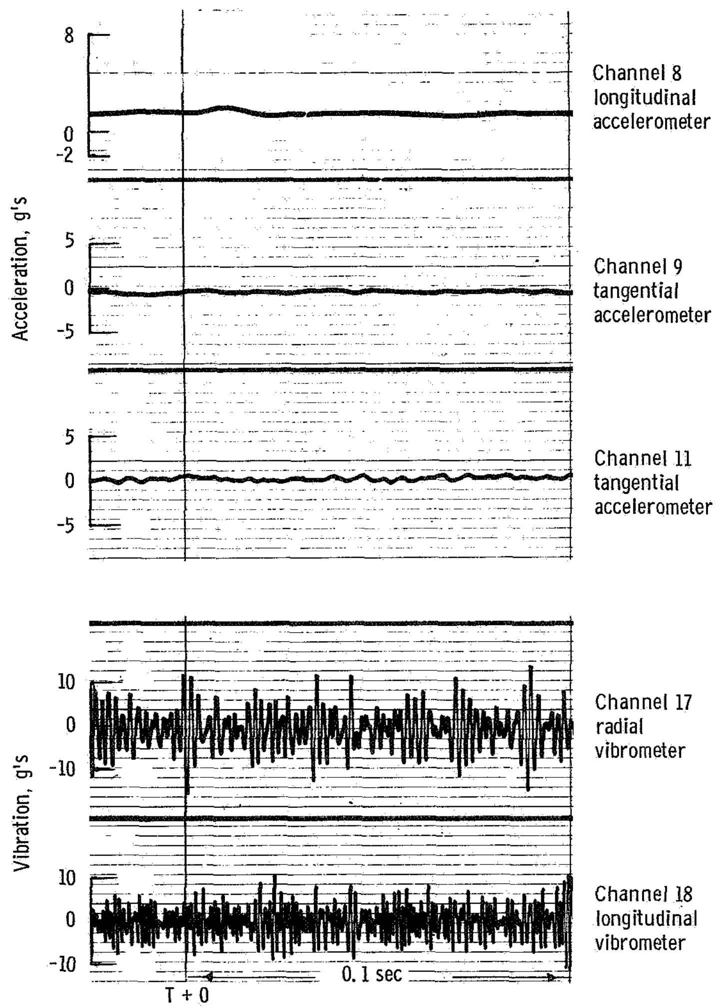


Figure D-2. - Dynamic data at lift-off, ATS-2.

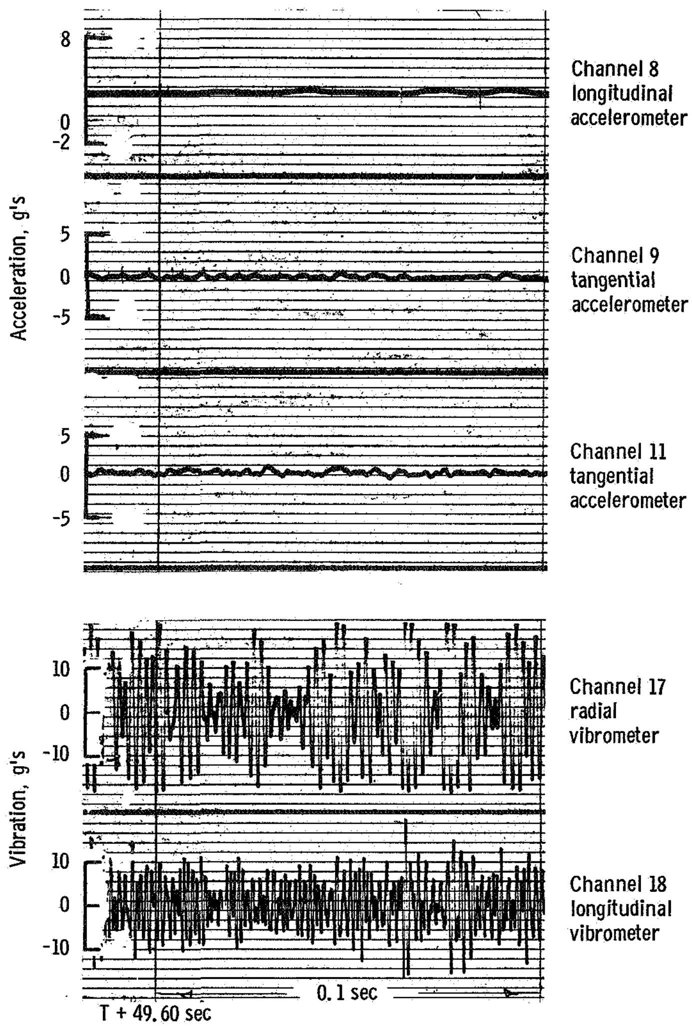


Figure D-3. - Dynamic data during transonic phase, ATS-2.

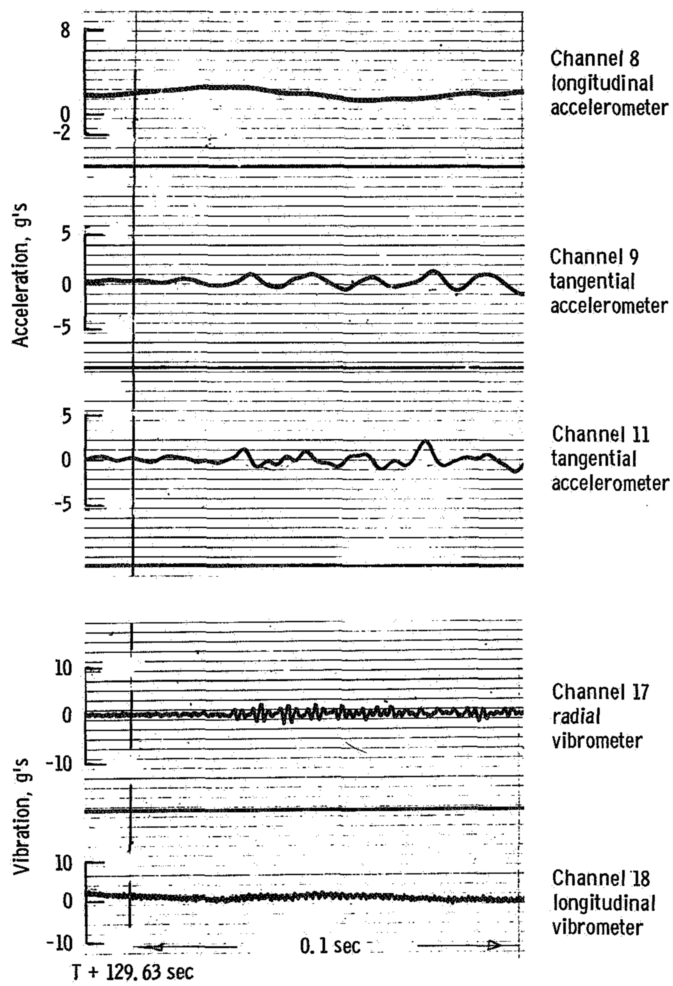


Figure D-4. - Dynamic data near time of booster engine cutoff, ATS-2.

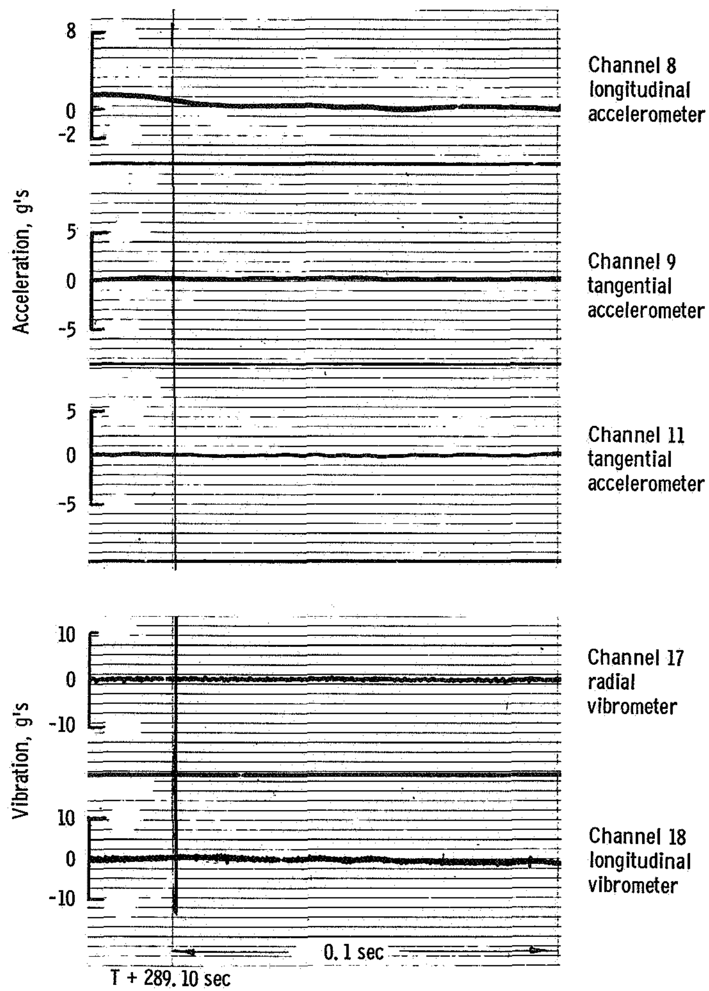


Figure D-5. - Dynamic data near time of sustainer engine cutoff, ATS-2.

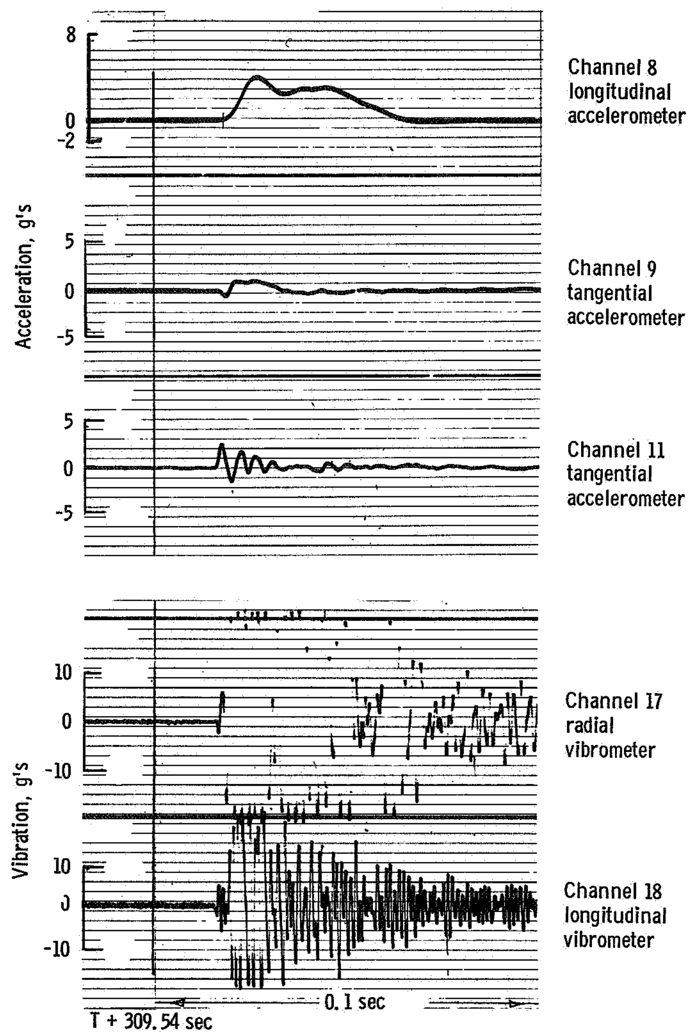


Figure D-6. - Dynamic data near time of horizon sensor fairing jettison, ATS-2.

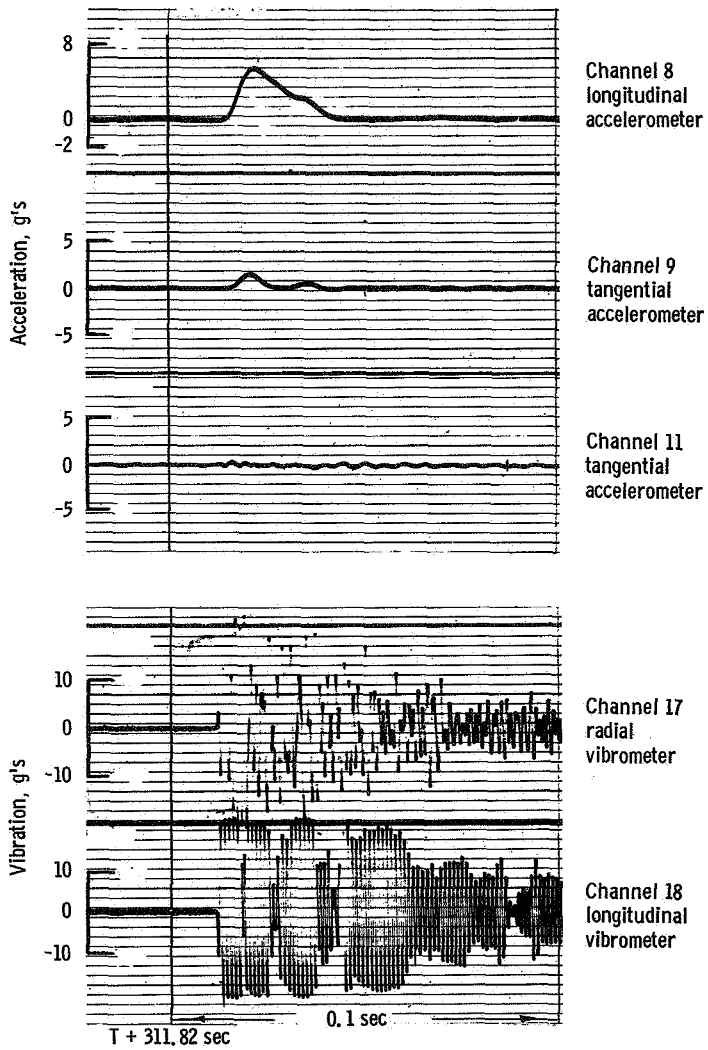


Figure D-7. - Dynamic data near time of Atlas-Agena separation, ATS-2.

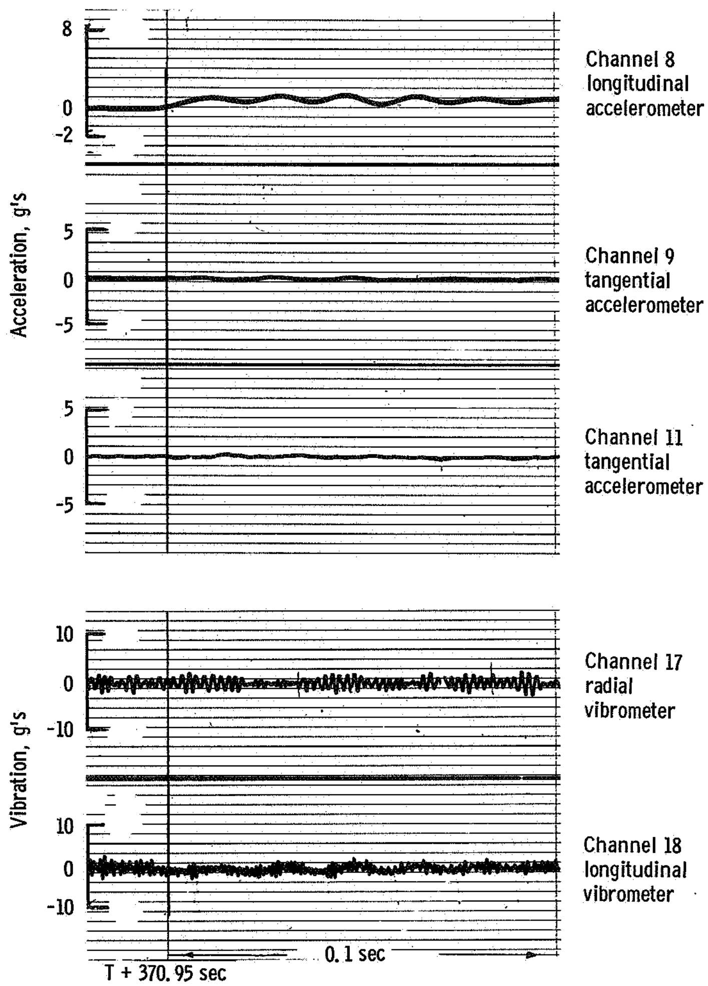


Figure D-8. - Dynamic data near time of Agena engine ignition, ATS-2.

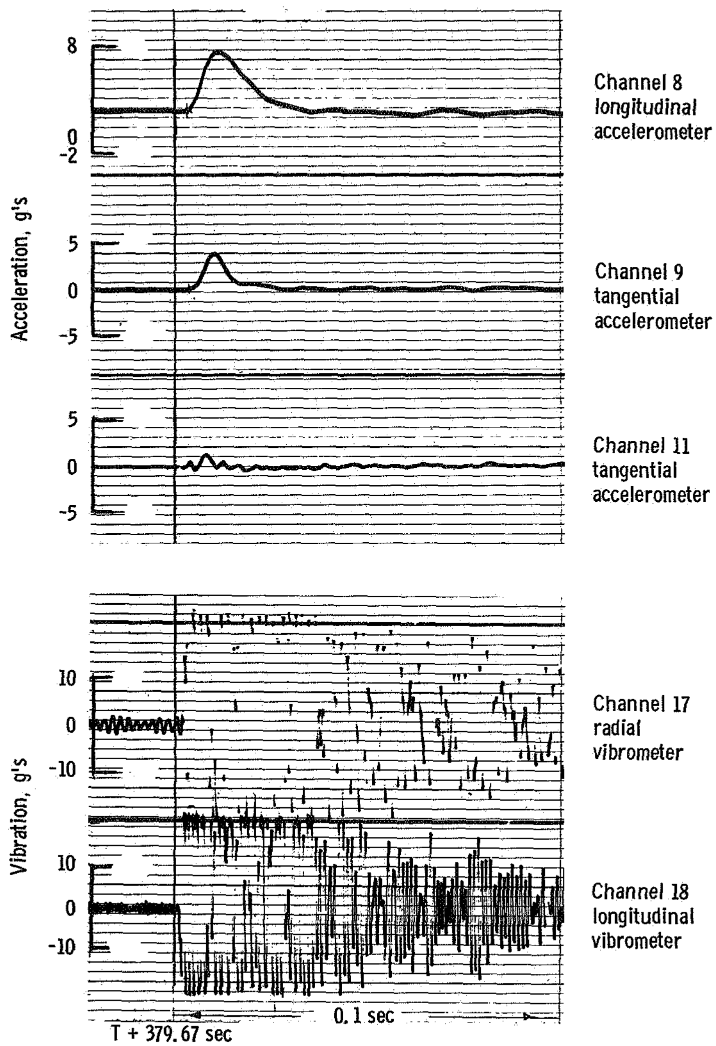


Figure D-9. - Dynamic data near time of shroud separation, ATS-2.

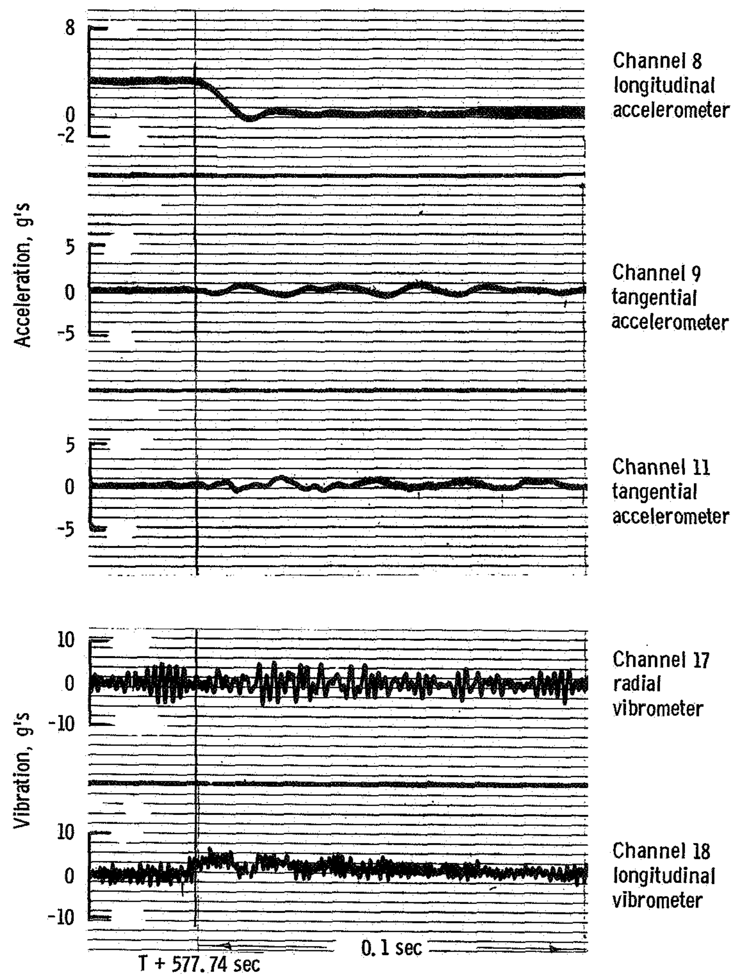


Figure D-10. - Dynamic data near time of Agena engine cutoff, ATS-2.

APPENDIX E

OXIDIZER PROPELLANT ISOLATION VALVE FAILURE

A schematic diagram of the Agena SS-01B propulsion system is shown in figure E-1. Propellants are provided to the engine from tanks with a common bulkhead by feed lines emanating from propellant containment sumps. These sumps are passive zero-gravity containment devices designed to assure the presence of propellants at the pump inlet after an orbital coast period. The propellant tanks are isolated from the engine system by a pair of propellant isolation valves (PIV's) located in the lines between the sumps and the fuel and oxidizer pump inlets. These valves are closed between burns not only to isolate the tanks from the engine but also to vent the volumes downstream to preclude the formation of vapor pockets in the pumps and propellant lines due to heat feedback from the hot engine. The tanks are pressurized by helium from a storage sphere to provide head pressure at the pump inlets. The tank pressurization system operates in a blowdown mode. Helium is admitted to the propellant tanks through a pyrotechnically operated valve with fixed area orifices.

The engine is started by pyrotechnic initiation of a solid-propellant start charge whose products of combustion spin-up the turbopump assembly which provides high pressure propellants to the engine. The start charge provides enough combustion products for about 3 seconds of turbopump operation. A portion of the propellants from the pump discharges is fed to the gas generator wherein the propellants react to provide the working fluid for steady-state turbopump operation. The engine, thus, "bootstraps" into steady-state operation. The gas generator combustion products pass through the turbine and then out the turbine exhaust duct located adjacent to the engine thrust chamber.

A portion of the flow from the fuel pump discharge is metered to a hydraulic power package. This fuel supplies power to a hydraulic motor which drives a hydraulic pump. The pump provides high pressure hydraulic fluid to operate the engine gimbal actuators.

A schematic flow diagram of the model 8096 Agena engine is shown in figure E-2. The brief description of the engine valve and switch sequence that follows is presented so that the flight data may be more easily understood. When electric power is applied to the engine (an event termed main power relay), the start charge is ignited and the propellant pumps generate pressure. The gas-generator solenoid valve is also actuated to permit propellants to enter the gas generator. The propellants are prevented from entering the thrust chamber by two valves: the main oxidizer valve and the main fuel valve. When the oxidizer pump discharge pressure reaches approximately 90 psi (62 N/cm^2), it overcomes the spring force of the main oxidizer valve and admits the oxidizer to the thrust chamber cooling passages and thence to the oxidizer manifold in the injector dome. When the oxidizer manifold gage pressure reaches approximately 33 psi

(22.76 N/cm^2), the oxidizer manifold pressure switch closes. The closure of the switch permits the application of power to the solenoid of the pilot-operated solenoid valve to shuttle the pilot valve which, in turn, allows the main fuel valve to admit fuel to the thrust chamber. Thus, the oxidizer manifold pressure switch ensures that an oxidizer lead prevails during engine start.

When the engine is commanded to shut down, all electrical power is removed from the engine circuits and the gas generator valves close, the combustion process in the gas generator decays, the turbopump decelerates, and the pump outlet pressures decay. The spring loaded main propellant valves then close, and propellants no longer enter the thrust chamber. One other event occurs at engine first-burn shutdown command and that is the firing of a pyrotechnic valve in the oxidizer fast-shutdown device. The latter ports high pressure nitrogen from a small reservoir to the back face of the main oxidizer valve poppet to assist the spring in closing the valve. This is done to minimize the oxidizer postflow and thus preserve propellants.

Post-First-Burn Data

Following completion of the Agena first burn, the Agena primary timer issued the command to close the PIV's at T + 586.79 seconds. Satisfactory operation of the fuel PIV was indicated by the decrease of fuel pump inlet pressure from 21 psi (14.48 N/cm^2) to 6 psi (4.14 N/cm^2) within 2 seconds as the volume between the fuel PIV and the main fuel valve was vented to ambient (see fig. E-3). The same was not true for the oxidizer PIV. The oxidizer pump inlet pressure remained at its post-engine-shutdown value of approximately 16 psi (11.03 N/cm^2) after the PIV should have vented the oxidizer system downstream of the PIV (fig. E-3).

The failure of the oxidizer PIV to vent properly is further confirmed by the oxidizer pump inlet temperature data following the PIV closure command. A comparison of the oxidizer pump inlet temperature on the ATS-1 flight with the ATS-2 flight is shown in figure E-4. On the ATS-1 flight, the oxidizer PIV functioned normally, and the oxidizer pump inlet temperature showed a rise of 6.6° F (3.67 K) within the first 30 seconds following the command to close the oxidizer PIV. This rise in temperature is caused by heat soak-back from the turbopump assembly following engine shutdown. However, the oxidizer pump inlet temperature on the ATS-2 flight showed only a 1.0° F (0.56 K) temperature rise during the same 30-second time period. Since there was only a small difference in first-burn duration between the two flights; the amount of heat soak-back and pump inlet temperature rise should be almost the same. The lower pump inlet temperature rise on the ATS-2 flight results from an additional heat path that existed for dissipating the residual heat. Such a heat path can exist through the liquid in the engine feed line if the oxidizer valve is not fully closed.

Attempted Second-Burn Data

At the conclusion of the coast to the apogee of the transfer orbit, the Agena engine was scheduled to reignite and circularize the orbit. Data acquired prior to the initiation of the engine start sequence for second burn indicate that the oxidizer pump inlet pressure had risen to 20 psi (13.79 N/cm^2), while the fuel pump inlet pressure was at essentially zero psi (fig. E-5). The oxidizer pump inlet temperature was 174° F (352.2 K) and the fuel pump inlet temperature had reached 195° F (363.85 K) (fig. E-6). This combination of pressure and temperature at the oxidizer pump inlet indicates that the oxidizer downstream of the PIV had been vaporized by the heat that soaked back from the hot turbopump assembly. The PIV open command was generated on schedule at T + 7112.93 seconds. Shortly thereafter, the fuel pump inlet pressure rose to 28 psi (19.3 N/cm^2), indicating normal fuel PIV operation (fig. E-5). At the same time, the fuel pump inlet temperature started to decrease rapidly (fig. E-6). This decrease in temperature is indicative of the entrance of a sizable quantity of liquid from the fuel sump into the pump inlet line.

The oxidizer pump inlet pressure remained constant during this same interval of time (fig. E-5). The oxidizer pump inlet temperature indicates only a slight decrease (fig. E-6). This decrease can be attributed to a small amount of liquid wetting the walls of the piping under the action of the small acceleration provided by the discharge of the start-charge combustion products through the turbine exhaust duct.

The main power relay was commanded on time, 2 seconds after the PIV open command, and the second start charge was ignited. A normal start sequence was evidenced by the rapid rise in fuel venturi inlet pressure (equivalent to pump discharge pressure, fig. E-7) indicating that the turbopump had been accelerated and the fuel pump was generating discharge pressure. However, the oxidizer venturi inlet pressure rose only to approximately 60 psi (41.37 N/cm^2) for about 1/3 second during the turbine spin-up period. The start of the oxidizer pressure rise also lagged the start of the fuel pressure rise by about 1 second. Normally, the oxidizer venturi inlet pressure buildup history would closely follow both in time and character that of the fuel venturi inlet pressure buildup history. The small pressure rise generated by the oxidizer pump was insufficient to open the main oxidizer valve and admit oxidizer to the manifold. Consequently, the oxidizer manifold pressure switch did not close to electrically actuate the pilot-operated solenoid valve, and no fuel was admitted to the engine thrust chamber. Thus, neither propellant reached the thrust chamber, and second burn was not obtained. This is substantiated by the data from switch group Z which indicate that the oxidizer manifold pressure switch did not close, nor did the thrust chamber pressure switch activate (fig. E-8).

Such turbopump behavior is characteristic of what may be termed a "vapor-lock" in the oxidizer pump and is a consequence of not venting the oxidizer lines downstream of the PIV after first-burn shutdown. The fact that the oxidizer pump discharge pressure did rise slightly for a short time indicated the presence of a small amount of liquid oxidizer at the pump; however, the quantity of propellant was insufficient to prime the pump. The only impulse imparted to the Agena was that from the discharge of the start-charge combustion products through the turbine exhaust duct. Once this energy source was depleted, the turbopump coasted to a stop.

From these data, it may be concluded that the oxidizer PIV did not close sufficiently to perform its isolation and venting function and that the consequence was the failure of the engine to start for the Agena second burn.

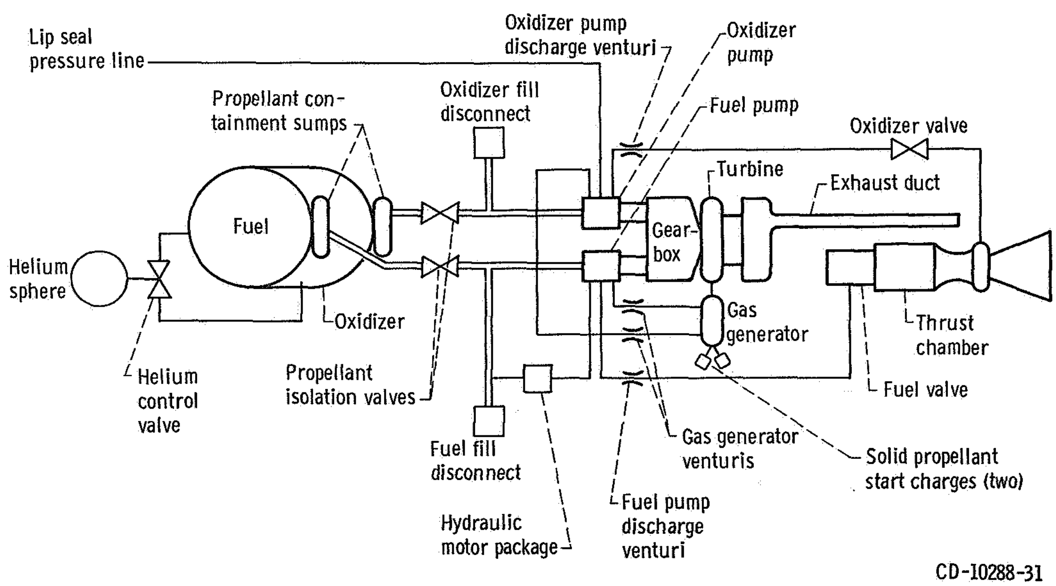


Figure E-1. - Agena propulsion system schematic drawing, ATS-2.

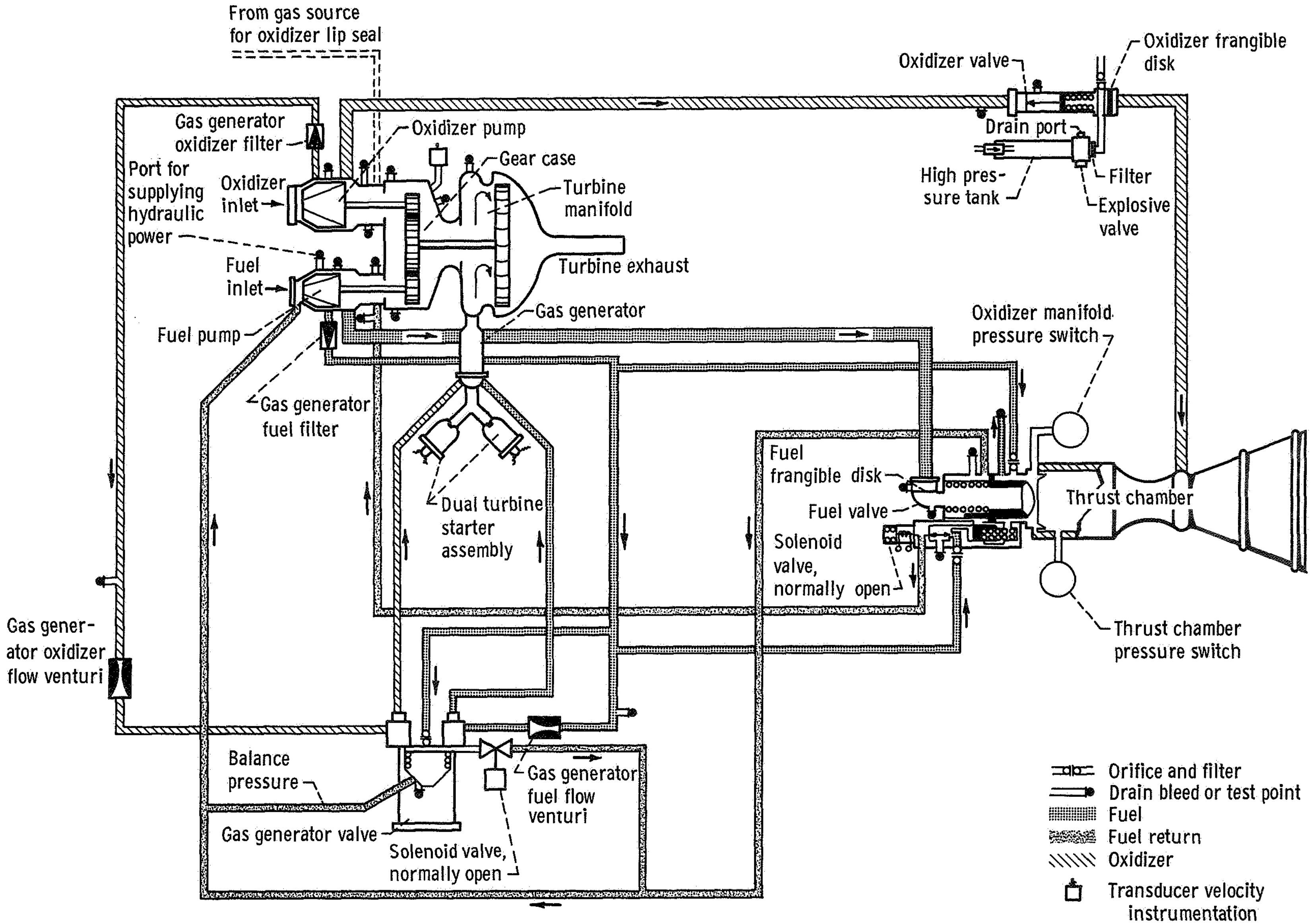


Figure E-2. - Schematic flow diagram model 8096 engine, ATS-2.

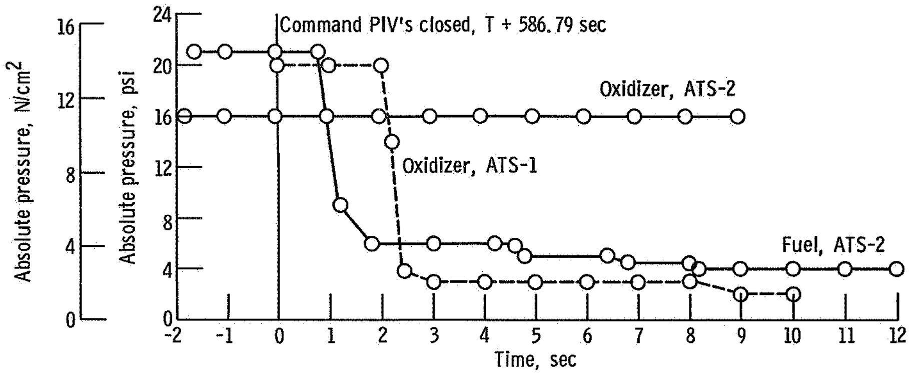


Figure E-3. - Fuel and oxidizer pump inlet pressures after first-burn shutdown, ATS-2.

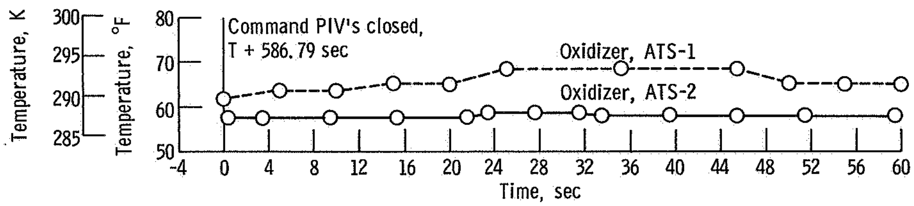


Figure E-4. - Comparison of oxidizer pump inlet temperatures after first-burn shutdown, ATS-2.

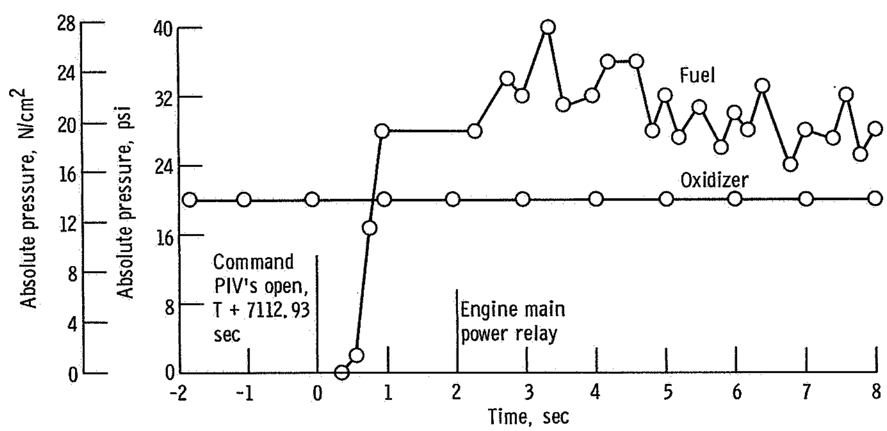


Figure E-5. - Fuel and oxidizer pump inlet pressures at start of second-burn ignition, ATS-2.

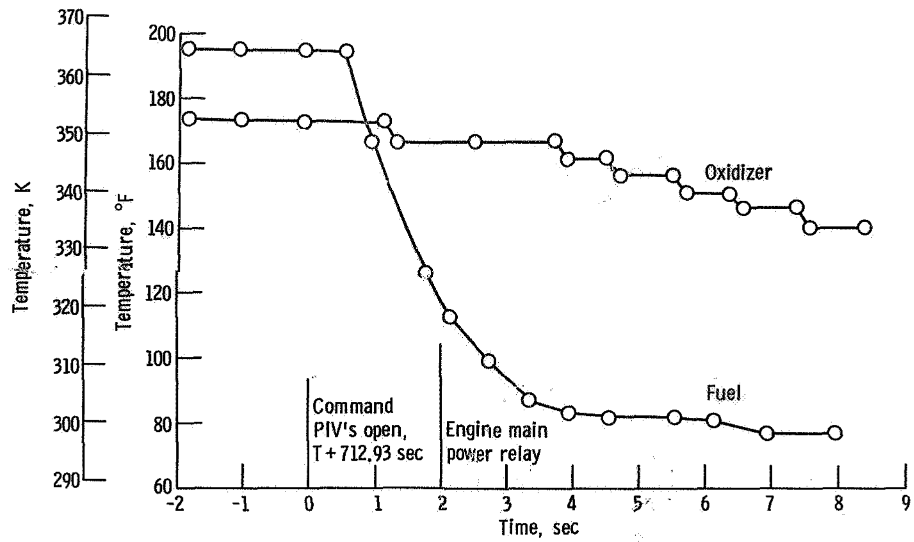


Figure E-6. - Fuel and oxidizer pump inlet temperatures at start of second-burn ignition, ATS-2.

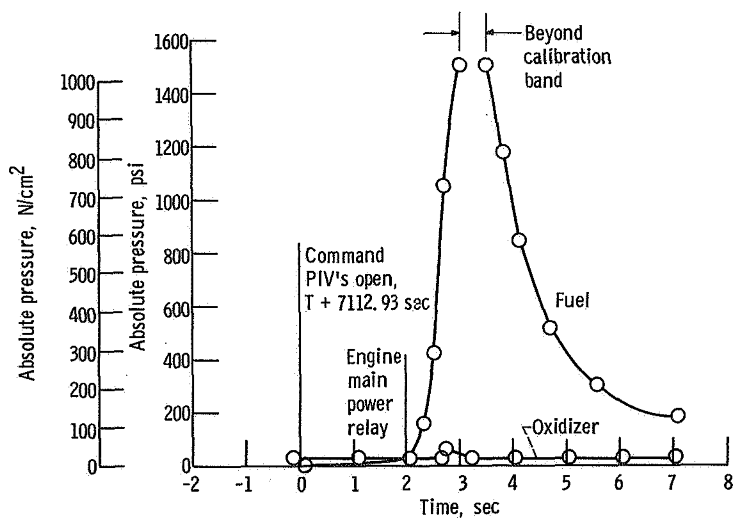


Figure E-7. - Fuel and oxidizer venturi inlet pressures at start of second-burn ignition, ATS-2.

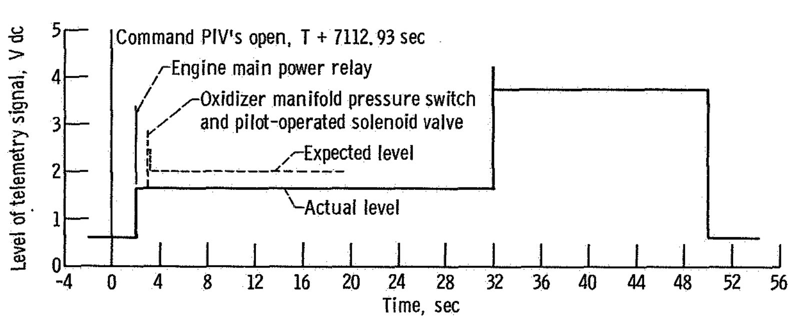


Figure E-8. - Switch group Z during second-burn attempt, ATS-2.

REFERENCE

1. Deverall, J. E.; Salmi, E. W.; and Knapp, R. J.: Orbital Heat Pipe Experiment. Rep. LA-3714, Los Alamos Scientific Lab., June 22, 1967.

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