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FLIGHT PERFORMANCE OF THE ATLAS-AGENA LAUNCH VEHICLE IN SUPPORT OF THE APPLICATIONS TECHNOLOGY SATELLITE ATS-3

Lewis Research Center
Cleveland, Ohio

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ABSTRACT

Three Applications Technology Satellites were launched by Atlas-Agena vehicles for experimentation in communications, cloud cover photography, and spacecraft environment. This report discusses the flight performance of the Atlas-Agena vehicle (for ATS-3) from lift-off through the Agena reorientation maneuver. This flight (ATS-3) in November 1967 concluded the Applications Technology Satellite missions launched by the Atlas-Agena vehicle.

CONTENTS

APPENDIXES

I. SUMMARY

The Atlas-Agena launch vehicle, with Applications Technology Satellite - 3 (ATS-3), was successfully launched on the third attempt (previous attempts were made on Nov. 3 and 4) from Eastern Test Range Complex 12 on November 5, 1967 at 1837:00.257 eastern standard time. The Atlas placed the Agena-spacecraft onto the proper suborbital coast ellipse. After separation of the Agena-spacecraft from the Atlas, the Agena engine was started and the Agena-spacecraft was injected onto the desired approximate 185-kilometer (100-n mi) circular parking orbit. Restart of the Agena engine after a 16-minute coast resulted in injection of the Agena-spacecraft onto the desired elliptical transfer orbit having an apogee altitude of about 35 500 kilometers (19 170 n mi) and a perigee altitude of about 185 kilometers (100 n mi).

Approximately 16 hours after separation of the Applications Technology Satellite from the Agena, the spacecraft apogee motor was started, and the spacecraft was placed onto the planned near-synchronous Earth orbit at an altitude of about 35 500 kilometers (19 170 n mi).

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

This report contains an evaluation of Atlas-Agena systems in support of the Applications Technology Satellite ATS-3 mission.

II. INTRODUCTION

The Atlas-Agena vehicle was first developed as a two-stage launch vehicle for Earth-orbiting payloads. Eighteen Atlas-Agena vehicles (including Applications Technology Satellite - 3) were launched under the direction of the Lewis Research Center to boost lunar probes, planetary probes, and other various Earth-orbiting spacecraft.

Applications Technology Satellite - 3 (ATS-3) was the last of a series of three ATS satellites to be launch with the Atlas-Agena vehicle. ATS-1 was launched in December 1966 as a synchronous-altitude, spin-stabilized spacecraft designed for communications, cloud cover photography, and spacecraft environmental experiments. ATS-2 was launched in April 1967 but failed to achieve an orbit suitable for conducting the planned experiments in gravity-gradient stabilization, communications, and cloud cover photography. ATS-3 was then launched in November 1967 with the objective of placing the spacecraft onto a near-synchronous Earth orbit. This spin-stabilized spacecraft was designed to extend experimentation in communications and cloud cover photography. The ATS-3 spacecraft, including the apogee motor, weighed 713.96 kilograms (1574 lb).

The Atlas vehicle boosts the combined Agena-spacecraft onto a suborbital coast ellipse. The Agena then performs two separate burns to place the ATS-3 spacecraft into the proper transfer orbit. After separation from the Agena, the ATS-3 spacecraft (equipped with an apogee motor for thrust) achieves a near-synchronous Earth orbit with an inclination of about zero degrees.

This report evaluates the flight performance of the Atlas-Agena launch vehicle, for the ATS-3, from lift-off through the Agena reorientation maneuver after spacecraft release.

III. LAUNCH VEHICLE DESCRIPTION

by Eugene E. Coffey and Richard P. Geye

The Atlas-Agena is a two-stage vehicle consisting of an Atlas (SLV-3) first stage and an Agena D (SS-01B) second stage connected by a booster adapter. The composite vehicle, including the spacecraft shroud and booster adapter is 33 meters (109 ft) long. The vehicle weight at lift-off is approximately 126 500 kilograms (279 000 lb). Figures III-1 to III-5 illustrate the arrangement of the composite space vehicle, the Atlas, the composite Agena-shroud-spacecraft, the Standard Agena Clamshell shroud, and the ATS-3 spacecraft.

The Atlas SLV-3 (fig. III-2) is about 21 meters (70 ft) long and is 3.05 meters (10 ft) in diameter except for the forward section of the tank which is conical and tapers to a diameter of about 2 meters (6 ft). The Atlas is propelled by a standard Rocketdyne MA-5 propulsion system consisting of a booster engine, which has two thrust chambers, with a total thrust of 1467.9×10^3 newtons (330×10^3 lb); a sustainer engine with a thrust of 253.5×10^3 newtons (57×10^3 lb); and two vernier engines, each with a thrust of 2.98×10^3 newtons (669 lb). All Atlas thrust levels are ratings at sea level. All engines use liquid oxygen and high-grade kerosene propellants and are ignited prior to lift-off. The booster engine thrust chambers are gimbaled for pitch, yaw, and roll control during the booster phase of flight. This phase is completed when the vehicle acceleration level equals about 6 g's. The booster engine section is jettisoned about 3 seconds after booster engine shutdown. The sustainer and vernier engines continue to burn for the sustainer phase of flight. During this phase, the sustainer engine is gimbaled for pitch and yaw control, and vernier engines are gimbaled for roll control. The sustainer engine burns until the vehicle achieves the desired suborbital parameters as determined by the ground radio guidance. After sustainer engine shutdown, the vernier engines continue to burn for a short period of time prior to Atlas-Agena separation. During this phase, the vernier engines are gimbaled to provide vehicle attitude control and fine trajectory corrections. After vernier engine shutdown, the Atlas booster adapter is structuraly severed from the Agena by firing of a Mild Detonating Fuse system located on the booster adapter. The firing of a retrorocket system, mounted on the booster adapter, then separates the Atlas booster adapter from the Agena.

The configuration of the Agena is shown in figure III-3. The Agena is 1.52 meters (5 ft) in diameter and 6.1 meters (20 ft) long. It is powered by a model 8096 Bell Aerosystems engine with a rated thrust of 71.17×10^3 newtons (16×10^3 lb) and has a two-burn capability. This engine uses unsymmetrical dimethylhydrazine (UDMH) and inhibited red fuming nitric acid (IRFNA) propellants. During powered flight, pitch and yaw control are provided by gimbaling the Agena engine, and roll control is provided by a cold gas (mixture of nitrogen and tetrafluoromethane) system. During periods of nonpowered flight, pitch, yaw, and roll control are provided by the cold-gas system. This cold-gas system is also used to complete the Agena reorientation maneuvers before and after spacecraft release.

A Standard Agena Clamshell shroud is used to protect the spacecraft during ascent through the atmosphere. This shroud is jettisoned during the Agena first firing period. The shroud (fig. III-4) is 5.7 meters (19 ft) long and about 1.6 meters (5.4 ft) in diameter. The Applications Technology Satellite is shown in figure III-5.

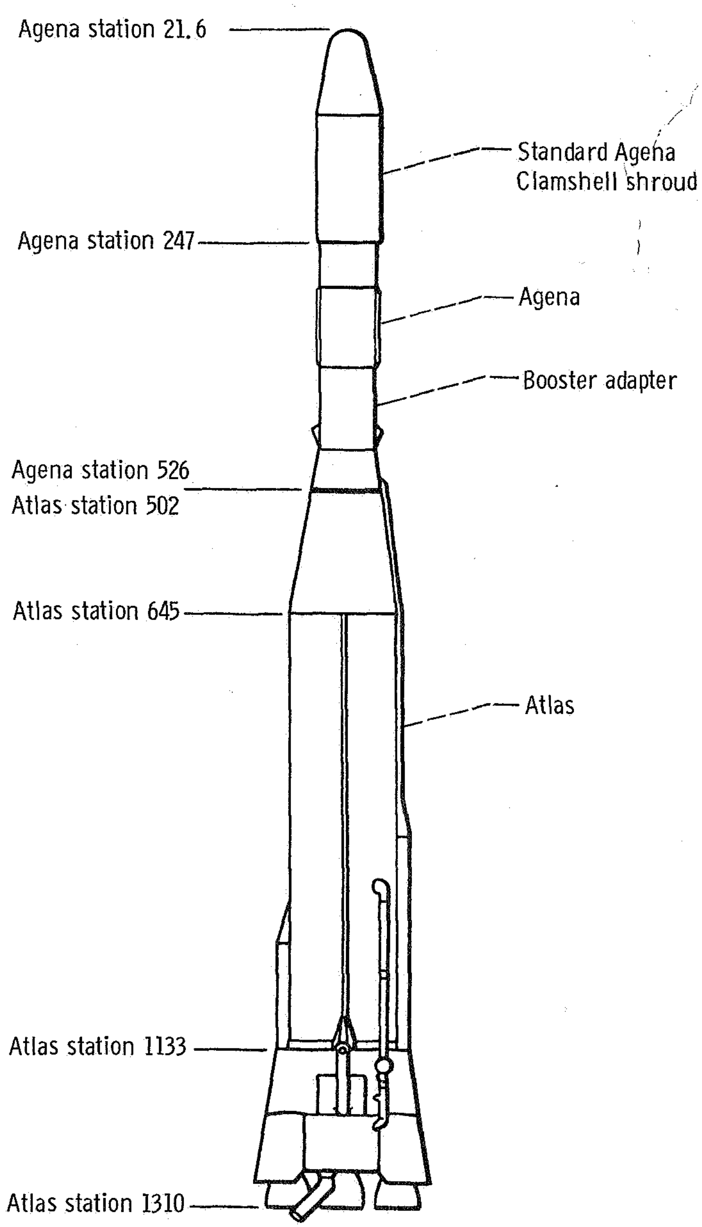


Figure III-1. - Space vehicle profile, ATS-3.

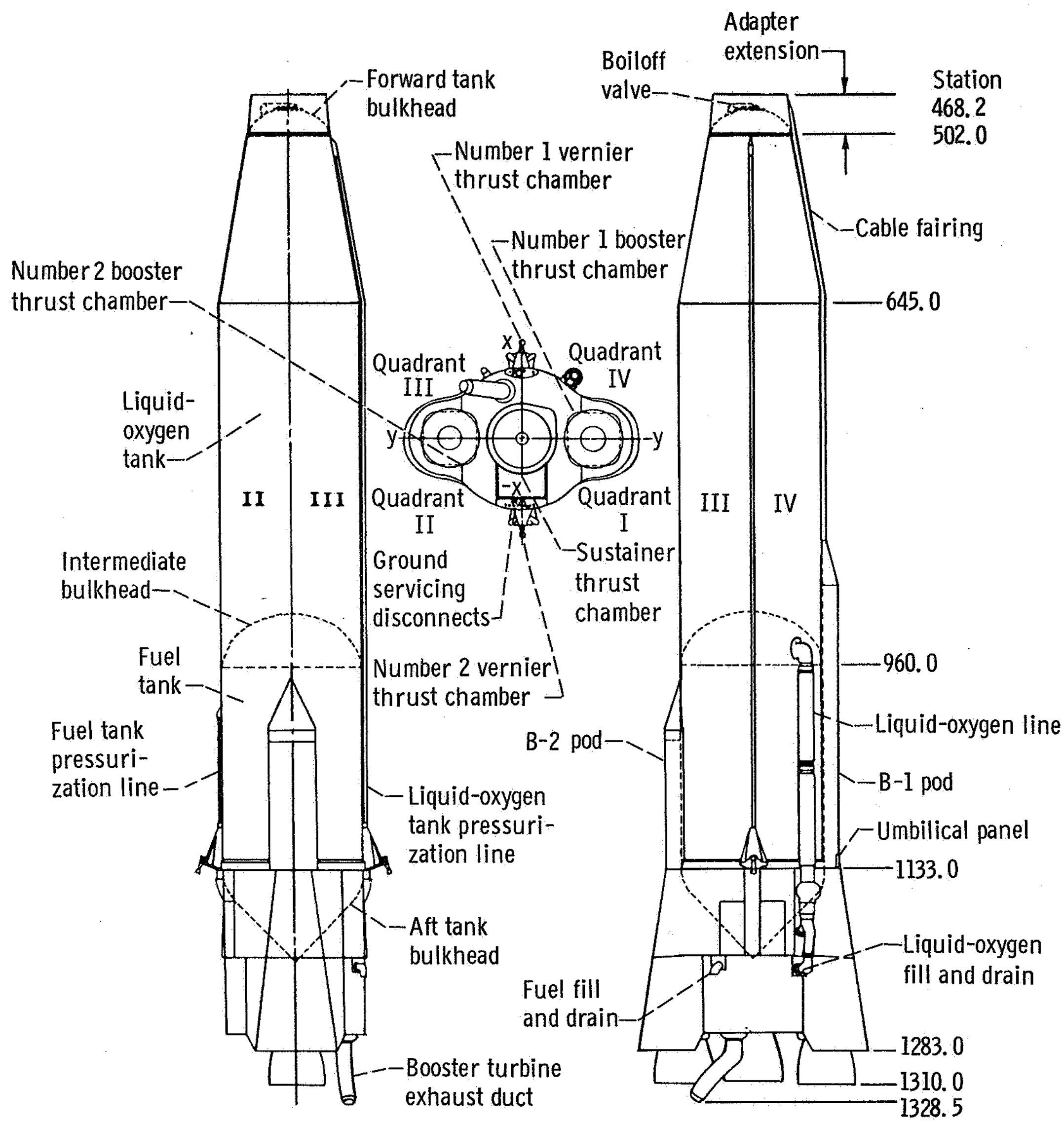


Figure III-2. - Atlas SLV-3 configuration, ATS-3.

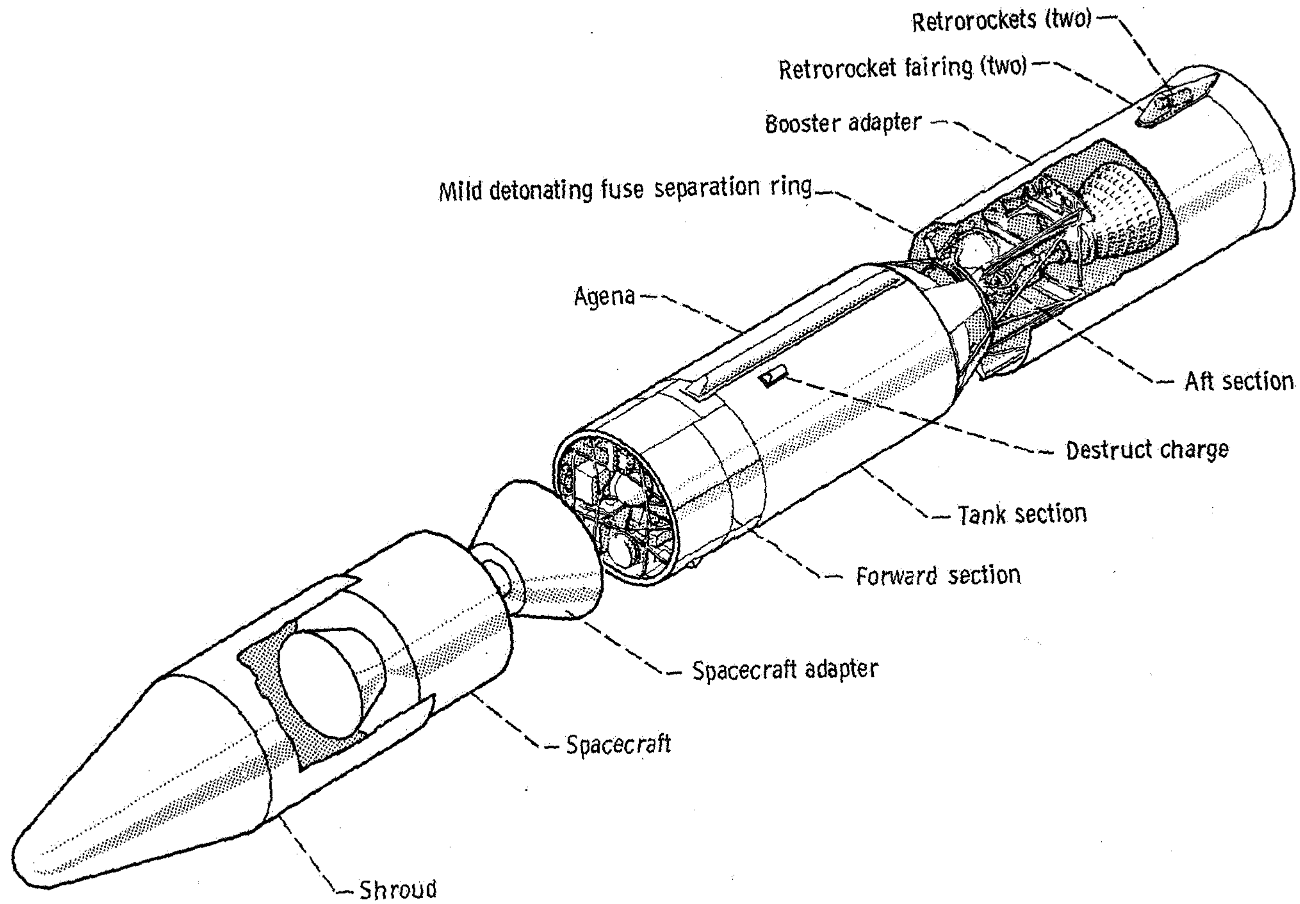


Figure III-3. - Agena-shroud-spacecraft, ATS-3.

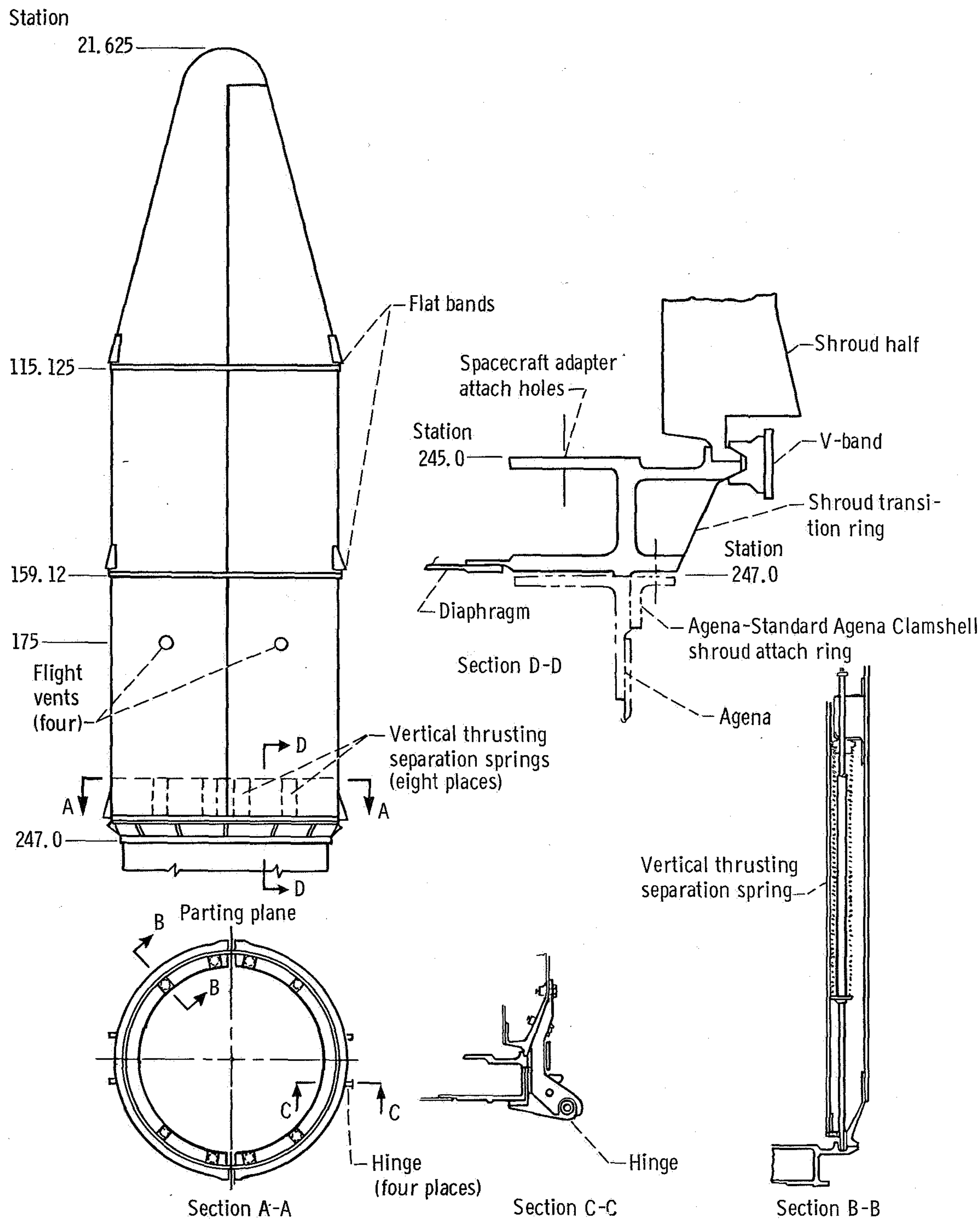


Figure III-4. - Standard Agena Clamshell shroud, ATS-3.

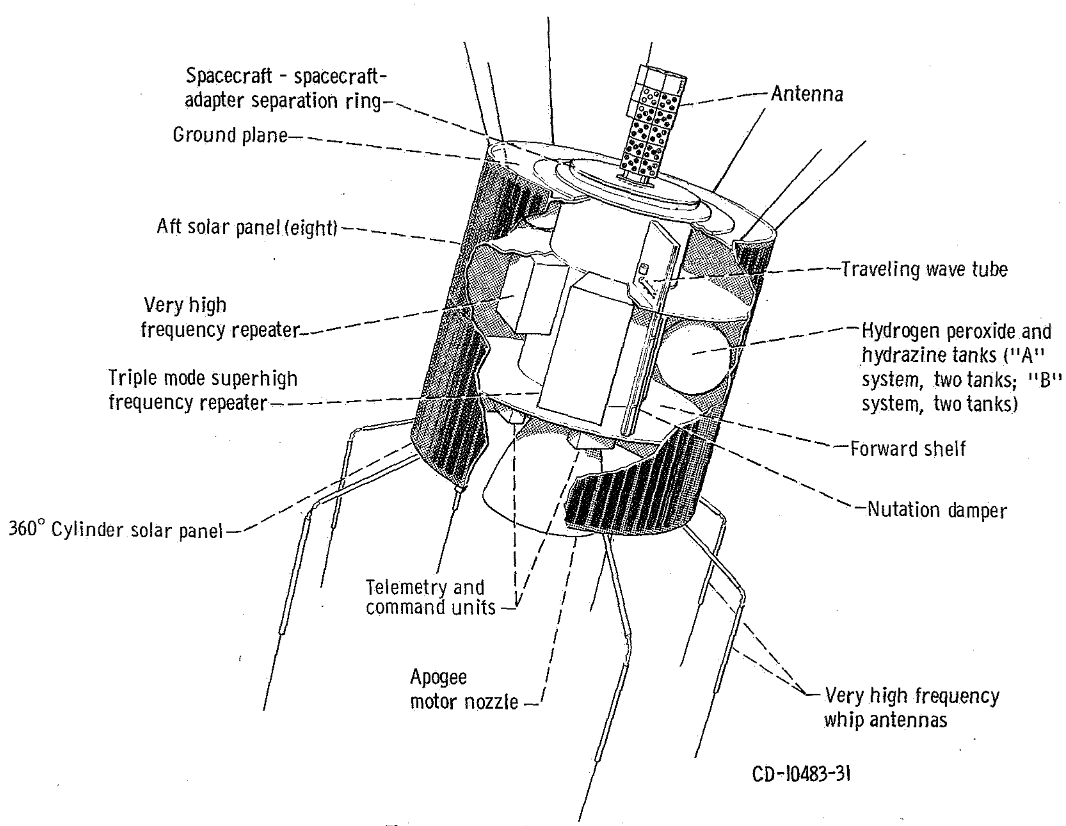


Figure III-5. - Spacecraft, ATS-3.

IV. TRAJECTORY AND PERFORMANCE

by George J. Schaefer and James C. Stoll

TRAJECTORY PLAN

The Atlas boosts the Agena - ATS-3 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, the Agena engine is ignited and burns until it places the Agena - ATS-3 onto a 185-kilometer (100-n mi) circular parking orbit. After coasting in the parking orbit to the first equatorial crossing, the Agena engine is reignited. The Agena second-burn phase injects the Agena - ATS-3 onto a transfer ellipse with a perigee altitude of 183 kilometers (99 n mi) and an apogee altitude of 35 503 kilometers (19 170 n mi). After Agena second-burn cutoff, the Agena pitches up 10.10° and yaws left 52.85° to place the spacecraft motor thrust axis in the required attitude. Then the spacecraft is separated from the Agena. Three seconds after spacecraft separation, the Agena performs a yaw-right maneuver of 211.41° to orient the Agena tail first in a direction approximately 21° left of the inertial velocity vector. This maneuver ensures that any Agena thrust resulting from the expulsion of residual propellants and gases will lower the Agena orbit and thereby reduce the probability of the Agena impacting the spacecraft. When the ATS-3 arrives at the apogee of the transfer ellipse the second time, the spacecraft motor is fired to place the ATS-3 onto a nearly circular orbit at synchronous altitude.

TRAJECTORY RESULTS

Lift-Off Through Atlas Booster Phase

ATS-3 was successfully launched from Complex 12, Eastern Test Range on November 5, 1967 at 1837:00.257 eastern standard time. Actual and expected times for significant flight events are given in table IV-I. A detailed sequence of flight events is provided in appendix A. The programmed booster roll was set at 15.18° clockwise looking forward to effect a vehicle roll from the launch pad azimuth of 105.18° to the intended flight azimuth of 90.0°.

Winds at launch were predominately tail winds from the west and tended to depress the trajectory. Wind data are shown in figure IV-1. The maximum wind velocity was 47.2 meters per second (155 ft/sec) at an altitude of 11 186 meters (36 700 ft). The wind shears produced by abrupt changes in wind velocity were not severe.

The maximum vehicle bending response was calculated to be 41.1 percent of the critical value on the booster adapter (Agena station 492) and to occur at an altitude of 5165.5 meters (16 947 ft). The maximum booster engine gimbal angle was calculated to be 41.9 percent of the available gimbal angle in the pitch plane and to occur at an altitude of 3079.7 meters (10 104 ft). The data used for these calculations were obtained from the T - 0 (lift-off) weather balloon.

Radar tracking data showed that the vehicle trajectory was slightly lower than the expected trajectory during the Atlas booster phase of the flight. This deviation resulted from tail winds. At booster engine cutoff, the actual trajectory was about 1981.2 meters (6500 ft) lower in altitude and 731.5 meters (2400 ft) uprange compared with the expected trajectory.

The trajectory deviated in the horizontal plane only slightly from the expected trajectory, resulting in a position 30.5 meters (100 ft) to the right at booster engine cutoff. This effect was primarily the result of an actual roll maneuver of 0.2° less than the programmed roll maneuver.

During the booster phase of flight, the capability of the flight control system to accept Mod III Radio Guidance commands was enabled at T + 80 seconds; however, pitch steering commands could only be transmitted from the ground guidance system between T + 100 and T + 110 seconds. Pitch steering commands were not transmitted during this interval since the velocity vector angle dispersion in the pitch plane was less than the predetermined threshold for booster steering. Mod III Radio Guidance yaw steering was not programmed to be used during the booster phase of flight. Booster engine cutoff occurred at T + 128.4 seconds (1.1 sec earlier than expected) by ground Mod III Radio Guidance command at a vehicle longitudinal acceleration of 5.99 g's. The acceleration level at booster engine cutoff was 0.07 g lower than expected but within the guidance equation requirement of 6.06 ± 0.2 g's. The booster engine section was jettisoned at T + 131.1 seconds.

Atlas Sustainer Phase

The trajectory remained depressed and to the right with respect to the expected trajectory during the sustainer phase of flight. A comparison of the tracking data with the expected trajectory at sustainer engine cutoff indicates that the vehicle position was about 5181.6 meters (17 000 ft) uprange, 487.7 meters (1600 ft) to the right, and 3352.8 meters (11 000 ft) lower than expected.

The Mod III Radio Guidance sustainer steering commands caused the vehicle to pitch up approximately 7.4° and yaw right less than 1°. These maneuvers were made to compensate for the low trajectory and to correct for a yaw-left displacement which occurred during the staging sequence. No corrections were made for the crossrange displacements accumulated during the booster phase. Therefore, the effects of these errors were present at sustainer engine cutoff. Sustainer engine cutoff was commanded by ground Mod III Radio Guidance at T + 289.1 seconds, 2.4 seconds earlier than expected.

The vehicle velocity (relative to the rotating Earth) at sustainer engine cutoff was about 6.1 meters per second (20 ft/sec) higher than expected. The total Atlas performance (including this velocity increment and the depressed trajectory of the vehicle) provided the desired energy for achieving the planned suborbital coast ellipse. The suborbital ellipse parameters are given in table IV-II.

Atlas Vernier Phase

Vernier engine thrust duration after sustainer engine cutoff was 20.1 seconds. During the vernier phase, pitch-down and yaw-left commands were issued by ground Mod III Radio Guidance to place the vehicle in the proper attitude for Atlas-Agena separation. These commands displaced the vehicle 0.54° left in yaw and 2.5° down in pitch. Vernier engine cutoff occurred by ground radio guidance at T + 309.2 seconds, 2.2 seconds earlier than expected. Atlas insertion velocities at vernier engine cutoff are shown in table IV-III.

The Atlas-Agena separation command from the Mod III Radio Guidance occurred at T + 313.5 seconds, 2.0 seconds earlier than expected but was consistent with the guidance equation requirement that the discrete must occur during the ground computer cycle following vernier engine cutoff plus 4.0 seconds.

Agena Engine First-Burn Phase

After Atlas-Agena separation, the Agena pitch-down maneuver was initiated to place the vehicle in the proper attitude for Agena engine first burn. The start Agena primary timer discrete had been transmitted during the Atlas phase by the ground Mod III Radio Guidance System at T + 300.1 seconds, 4.4 seconds later than expected. The ground Mod III Radio Guidance System had determined that the Agena would be injected onto the suborbital coast ellipse at an altitude lower than expected and, therefore, the Agena would reach first-burn altitude later than expected. The guidance system adjusted the start Agena primary timer by the 4.4 seconds so that Agena engine first burn would occur at the proper altitude.

Agena engine first ignition occurred at T + 371.1 seconds, and shroud separation occurred at T + 381.1 seconds. First ignition and all other timer events are consistent with the 4.4-second adjusted start Agena primary timer within the ±0.2-second timer tolerance. Thrust duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 161.3 seconds, 1.1 seconds longer than expected. Velocity meter shutdown at T + 533.6 seconds indicated that the proper velocity increment had been gained. Thrust decay added 2.47 meters per second (8.1 ft/sec) velocity compared with a predicted 2.96 meters per second (9.7 ft/sec). This predicted velocity had been compensated for by the velocity meter settings. Analysis of the tracking data indicated that the resultant parking orbit of the Agena-ATS at Agena engine first shutdown was lower than expected. The actual parking orbit parameters are listed in table IV-IV.

Agena Engine Second-Burn Phase

After the Agena engine first burn, the Agena - ATS-3 coasted for 970.5 seconds to the first equatorial crossing, the proper spatial position for Agena second burn. Agena engine second ignition occurred at T + 1504.1 seconds. The engine second-burn duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 78.3 seconds, 0.1 second longer than expected. Velocity meter shutdown indicated that the proper velocity increment had been gained. Thrust decay velocity was 7.74 meters per second (25.4 ft/sec) compared with a predicted decay velocity of 9.88 meters per second (32.4 ft/sec).

The resultant Agena - ATS-3 transfer orbit had an apogee that was slightly higher than predicted and a perigee that was slightly lower than predicted. The actual transfer orbit parameters are given in table IV-V.

Post-Second-Burn Phase

After Agena engine second-burn shutdown, the Agena performed the programmed pitch-up maneuver of an expected 10.10°, followed by the programmed yaw-left maneuver of an expected 52.85°. These maneuvers were performed to place the ATS motor thrust axis in the required attitude for the spacecraft motor burn. At T + 1748.1 seconds, separation of the Agena and the ATS-3 occurred. Three seconds later the Agena performed the programmed yaw-right maneuver of an expected 211.41° to ensure that the Agena would not interfere with the spacecraft.

ATS Apogee Motor Burn

When the spacecraft arrived at the transfer orbit apogee for the second time, nearly 16 hours after separation of the ATS-3 from the Agena, the apogee motor aboard the spacecraft was fired. As planned, the apogee motor burn reduced the spacecraft orbit inclination to about zero degrees and placed the spacecraft onto an almost circular orbit (35 527 km altitude) slightly below the synchronous altitude. The spacecraft performed satisfactorily.

TABLE IV-I. - SIGNIFICANT FLIGHT EVENTS, ATS-3

Event description	Expected time, sec	Actual time, sec
Lift-off (1837:00.257 EST)	0	0
Booster engine cutoff	129.5	128.4
Booster engine jettison	132.5	131.1
Sustainer engine cutoff	291.5	289.1
Start Agena primary timer	295.7	300.1
Vernier engine cutoff	311.4	309.2
Atlas-Agena separation	315.5	313.5
Fire first-burn ignition squibs	366.7	371.1
Agena steady-state thrust (90 percent of chamber pressure)	367.9	372.3
Shroud separation	376.7	381.1
Agena first-burn cutoff	528.0	533.6
Fire second-burn ignition squibs	1499.7	1504.1
Agena steady-state thrust (90 percent of chamber pressure)	1500.9	1505.2
Agena second-burn cutoff	1579.0	1583.5
Start 10.1^{o} pitch up	1666.7	1671.1
Stop 10.1^{o} pitch up	1688.7	1693.1
Start 52.9^{o} yaw left	1705.7	1710.0
Stop 52.9^{o} yaw left	1723.7	1728.0
Fire spacecraft separation squibs	1743.7	1748.1
Start 211.4^{o} yaw right	1746.7	1751.0
Stop 211.4^{o} yaw right	1818.7	1823.0

TABLE IV-II. - ATLAS SUBORITAL COAST

ELLIPSE PARAMETERS, ATS-3

Parameter	Units	Expected value	Actual value
Semimajor axis	km	4 413. 11	4 413. 02
	n mi	2 382. 89	2 382. 84
Semiminor axis	km	3 856. 60	3 856. 36
	n mi	2 082. 40	2 082. 27
Radius vector magnitude at	km	6 558. 38	6 558. 47
apogee	n mi	3 541. 24	3 541. 29
Inertial velocity at apogee	m/sec	5 588. 75	5 588. 33
	ft/sec	18 335. 8	18 334. 4
Inclination	deg	28. 34	28. 33

TABLE IV-III. - ATLAS INSERTION VELOCITIES AT

VERNIER ENGINE CUTOFF, ATS-3

Parameter	Units	In-flight objective	Acutal value
Velocity	m/sec	5 634. 75	5 634. 56
magnitude	ft/sec	18 486. 7	18 486. 1
Altitude	m/sec	501. 7	502. 0
rate	ft/sec	1 646. 0	1 647. 0
Lateral	m/sec	0	1. 25 (right)
velocity	ft/sec	0	4. 1 (right)

TABLE IV-IV. - AGENA PARKING ORBIT PARAMETERS,[a] ATS-3

Parameter	Units	Actual value
Apogee altitude	km	192.42
	n mi	103.9
Perigee altitude	km	184.27
	n mi	99.5
Period	min	88.2
Inclination	deg	28.4

[a]Security classification regulations preclude the listing together of actual and expected parameters of the Agena.

TABLE IV-V. - FINAL AGENA-SPACECRAFT TRANSFER ORBIT PARAMETERS,[a] ATS-3

Parameter	Units	Actual value
Apogee altitude	km	35 628.78
	n mi	19 238.0
Perigee altitude	km	177.05
	n mi	95.6
Inclination	deg	28.41
Eccentricity	----	.7300
Period	min	625.6

[a]Security classification regulations preclude the listing together of actual and expected parameters of the Agena.

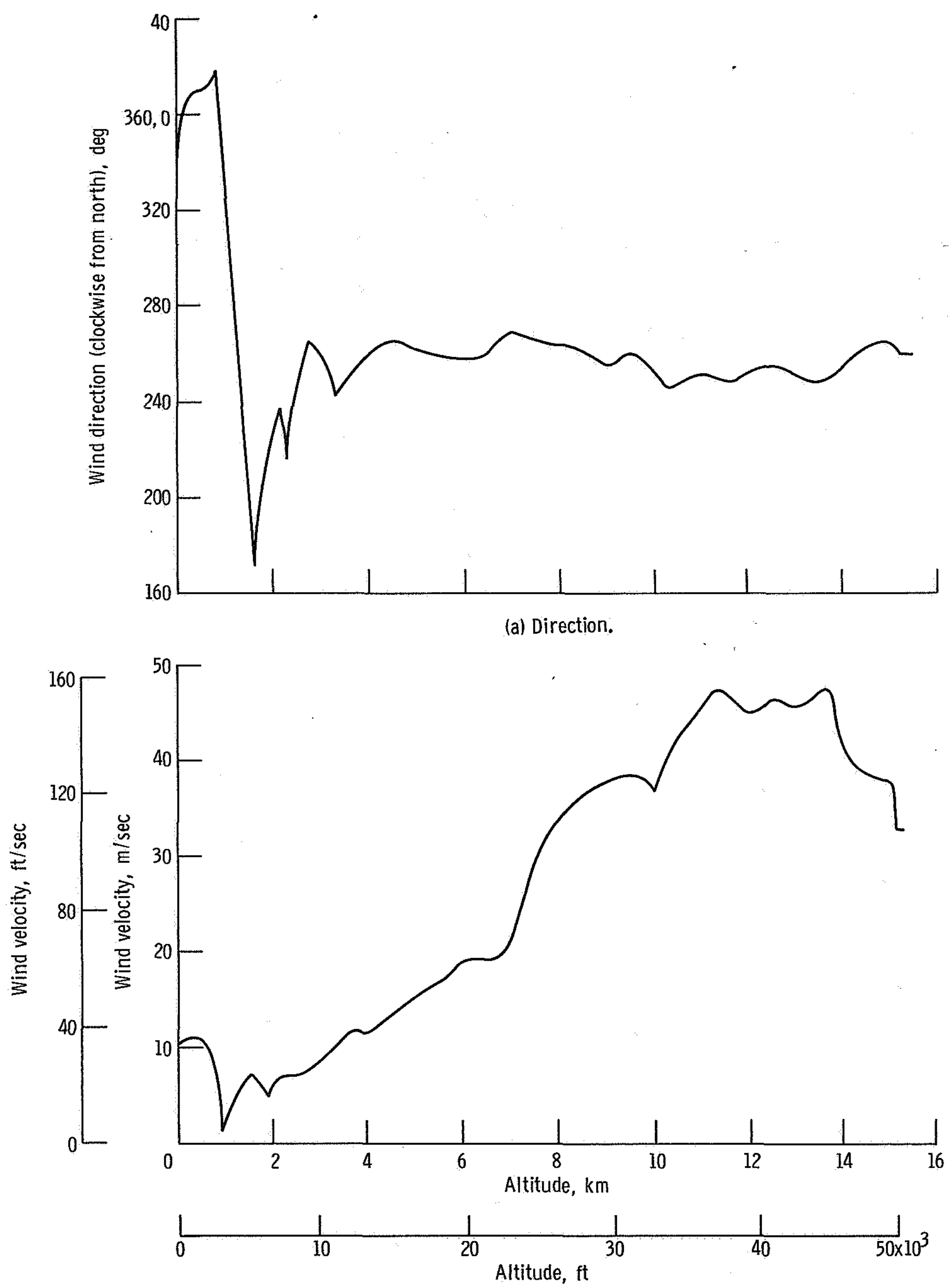


Figure IV-1. - Wind data at lift-off, ATS-3.

V. ATLAS VEHICLE SYSTEM PERFORMANCE

VEHICLE STRUCTURE SYSTEM

by Richard T. Barrett

Description

The Atlas 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 an RP-1 (kerosene) compartment and a liquid-oxygen compartment by an intermediate bulkhead. The tank section is 3.05 meters (10 ft) in diameter with an ellipsoidal bulkhead enclosing the conical forward end and a thrust cone enclosing the aft end. The sustainer engine is mounted on the thrust cone. The overall tank length is 18.6 meters (60.9 ft). 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 pneumatic latch mechanism that releases to allow the booster engine section to jettison after booster engine cutoff.

Performance

The vehicle structure system performance was satisfactory. All measured loads were within the expected limits. The peak longitudinal load factor during the Atlas boost phase of flight was 5.99 g's at booster engine cutoff. The command signal to initiate the high pressure pneumatics latch release mechanism occurred at T + 131.1 seconds. The mechanism functioned properly, and the booster engine section separated satisfactorily.

This was the sixth Atlas (SLV-3) flight using new lightweight engine boots. In certain previous flights, with the Atlas using the old type boots, telemetry data indicated that engine compartment temperatures were considerably higher than normal. This condition was probably caused by insufficient shielding of the engine compartment and its components from engine radiation and hot gases. Since high temperatures could prove injurious to the engine compartment components and wiring, new engine boots were designed to provide improved shielding. The engine compartment temperatures on this mission were satisfactory and compared favorably with expected temperatures. The locations at which temperatures were measured are shown in figure V-1, and the measured values are given in table V-I.

TABLE V-I. - ATLAS ENGINE COMPARTMENT

TEMPERATURES, ATS-3

Measurement	Units	Flight values at -	
		Lift-off	Atlas booster engine cutoff
A743T	K	278	284
	^{o}F	40	52
A745T	K	297	307
	^{o}F	76	94
P15T	K	294	294
	^{o}F	70	70
P16T	K	289	293
	^{o}F	60	68
P671T	K	287	287
	^{o}F	58	58

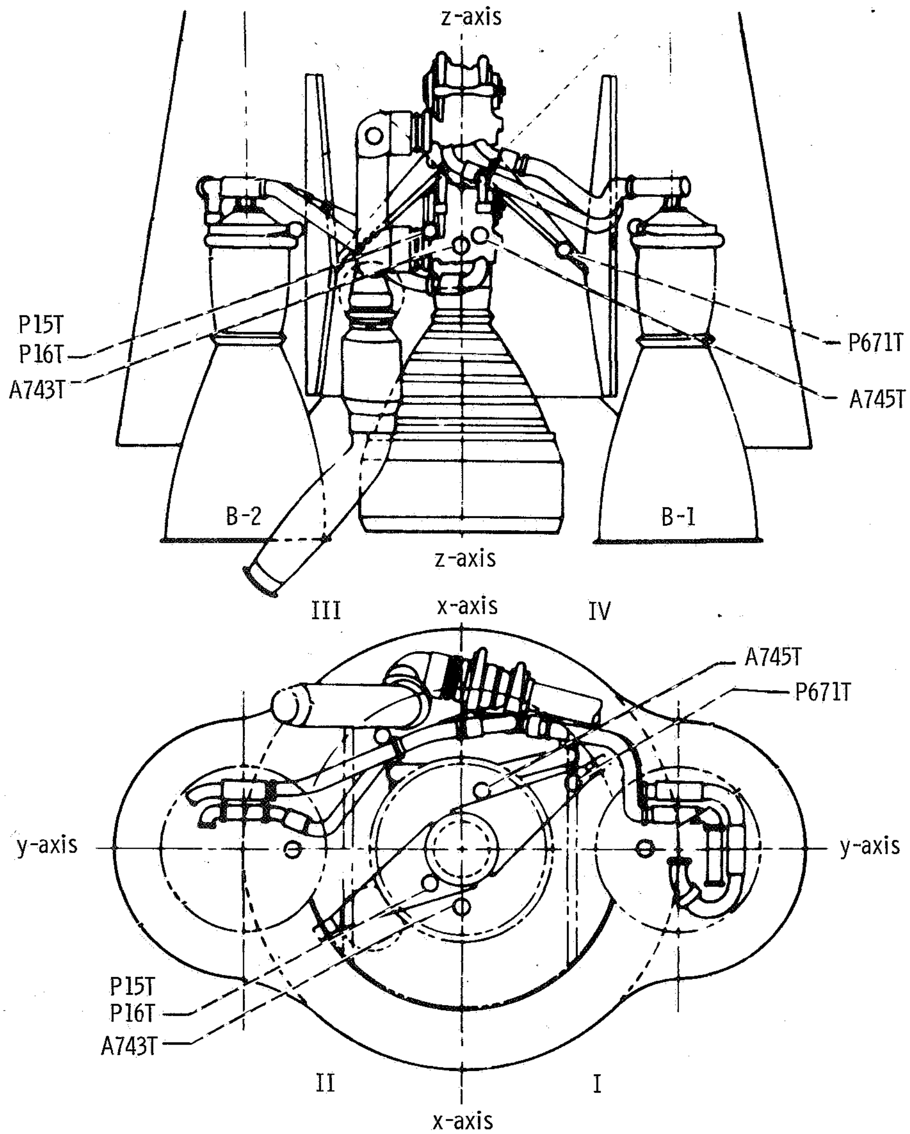


Figure V-1. - Atlas engine compartment temperature instrumentation, ATS-3.

PROPULSION SYSTEM

by Charles H. Kerrigan

Description

The Atlas engine system (fig. V-2) consists of a booster engine, a sustainer engine, two vernier engines, an engine tank system (pressurization and auxiliary propellant), and an electrical control system. The engines are of the single-burn type. During engine start, electrically fired pyrotechnic igniters are used to ignite the gas generator propellants for driving the turbopumps; hypergolic igniters are used to ignite the propellants in the thrust chambers of the booster, sustainer, and vernier engines. The propellants are liquid oxygen, oxidizer, and RP-1 (kerosene), fuel.

The booster engine, rated at 1467.9×10^3 newtons (330×10^3 lb) thrust at sea level, is made up of two gimbaled thrust chambers, propellant valves, two oxidizer and two fuel turbopumps driven by one gas generator, a lubricating oil system, and a heat exchanger. The sustainer engine, rated at 253.5×10^3 newtons (57×10^3 lb) thrust at sea level, consists of a thrust chamber, propellant valves, one oxidizer and one fuel turbopump driven by a gas generator, and a lubricating oil system. The entire sustainer engine system gimbals. Each vernier engine is rated at 2.98×10^3 newtons (669 lb) thrust at sea level when supplied with propellants from the sustainer turbopumps during sustainer engine operation. In the vernier phase of flight, each vernier engine is rated at 2.34×10^3 newtons (525 lb) thrust at sea level. For this phase, the vernier engines are supplied with propellants from the engine tank system because the sustainer turbopumps do not operate after sustainer engine cutoff. The vernier engines also gimbal. The engine tank system is composed of two small propellant tanks (each approximately 51 cm (20 in.) in diameter) and a pressurization system. This system supplies propellants for starting the engines and also for vernier engine operation after sustainer engine cutoff.

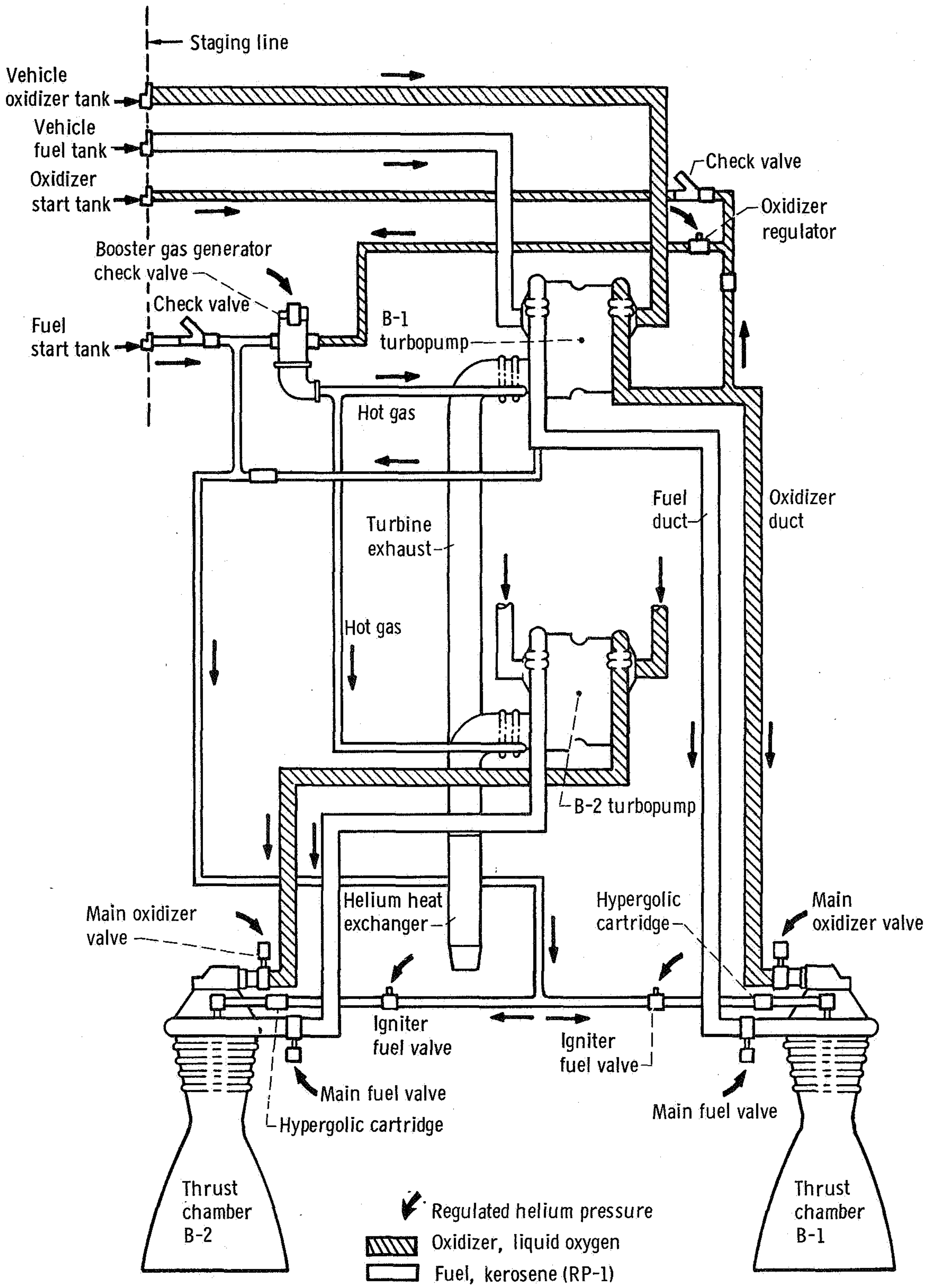
Performance

The performance of the Atlas propulsion system for the ATS-3 mission was satisfactory. During the engine start phase, valve opening times and starting sequence events were within tolerances. This was verified by landline recorders and by telemetry. The flight performance of the engines was evaluated by comparing measured engine parameters with the expected values. These are tabulated in table V-II. For portions of the flight, the measured value of sustainer engine thrust chamber pressure was slightly greater than expected, but the increased pressure was not detrimental to engine operation. The number 2 vernier engine chamber pressure data showed a gradual decrease of 17 newtons per square centimeter (24 psi) beginning at T + 262 seconds; it abruptly recovered at T + 270 seconds. This false indication of chamber pressure decay resulted from a buildup of carbon in the chamber pressure instrumentation sensing port. Engine performance did not reflect an actual pressure decay. All engine cutoff signals were issued by the Mod III Radio Guidance System commands and were properly executed. Transients at engine shutdown were normal.

TABLE V-II. - ATLAS PROPULSION SYSTEM PERFORMANCE, ATS-3

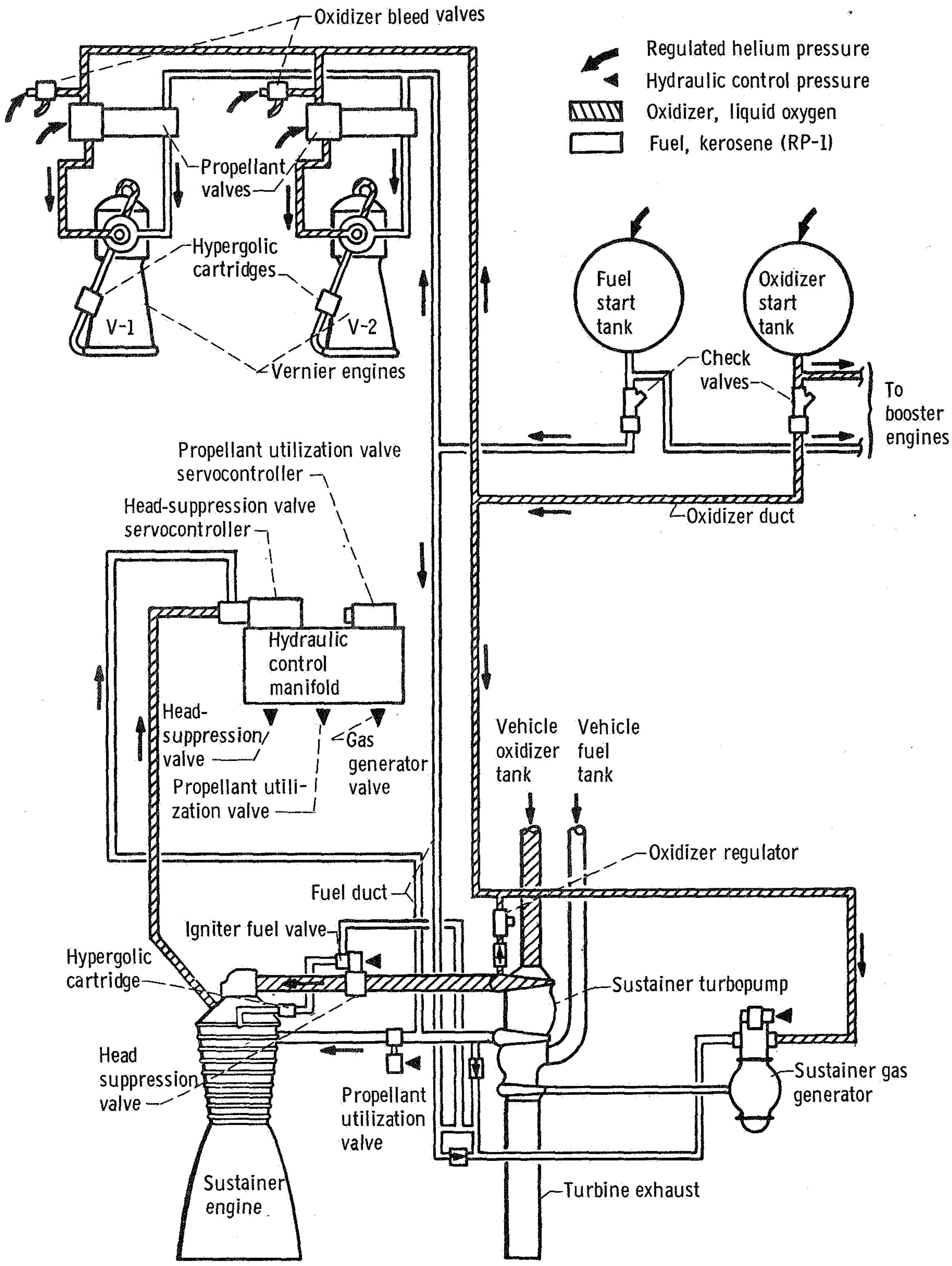
Perfromance parameters	Units	Expected operating range	Flight values at -			
			T + 10 sec	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Booster engine:						
Number 1 thrust chamber	N/cm[2]	386 to 410	392	398	(a)	(a)
pressure (absolute)	psi	560 to 595	568	576		
Number 2 thrust chamber	N/cm[2]	386 to 410	392	398	(a)	(a)
pressure (absolute)	psi	560 to 595	568	576		
Gas generator chamber	N/cm[2]	351 to 382	364	364	(a)	(a)
pressure (absolute)	psi	510 to 555	528	528		
Number 1 turbopump speed	rpm	6225 to 6405	6 306	6 306	(a)	(a)
Number 2 turbopump speed	rpm	6165 to 6345	6 217	6 217	(a)	
Sustainer engine:						
Thrust chamber pressure	N/cm[2]	469 to 493	496	496	490	(a)
(absolute)	psi	680 to 715	720	720	710	
Gas generator discharge	N/cm[2]	407 to 473	452	452	447	(a)
pressure (absolute)	psi	620 to 680	656	656	648	
Turbopump speed	rpm	10 025 to 10 445	10 414	10 384	10 450	(a)
Vernier engine:						
Number 1 thrust chamber pressure	N/cm[2]	172 to 183	179	179	179	(a)
when pump supplied (absolute)	psi	250 to 265	260	260	260	(a)
Number 1 thrust chamber pressure	N/cm[2]	145 to 155	(a)	(a)	(a)	152
when tank supplied (absolute)	psi	210 to 225				220
Number 2 thrust chamber pressure	N/cm[2]	172 to 183	179	179	179	(a)
when pump supplied (absolute)	psi	250 to 265	260	260	260	
Number 2 thrust chamber pressure	N/cm[2]	145 to 155	(a)	(a)	(a)	152
when tank supplied (absolute)	psi	210 to 225				220
Duration of burn:						
Booster engine	sec	129.5	------	128.4	--------	-----
Sustainer engine	sec	291.5	------	--------	289.1	-----
Vernier engine	sec	311.4	------	--------	--------	309.2

[a]Not applicable.



(a) Booster.

Figure V-2. - Atlas propulsion system, ATS-3.



(b) Sustainer and vernier.

Figure V-2. - Concluded.

HYDRAULIC SYSTEM

by Eugene J. Cieslewicz

Description

The Atlas hydraulic system consists of two independent systems: the booster system and the sustainer-vernier system (fig. V-3). The booster hydraulic system provides power for gimbaling the two thrust chambers of the booster engine system. System pressure is supplied by a single, pressure-compensated, variable-displacement pump driven by the engine turbopump accessory drive. Other components of the system include four servocylinders, a high-pressure relief valve, accumulators, and a reservoir. Engine gimbaling in response to flight control commands is accomplished by the servocylinders which provide separate pitch, yaw, and roll control during the booster phase of flight. The maximum booster engine gimbal angle capability is $\pm 5^{\circ}$ in both the pitch and yaw planes.

The sustainer-vernier hydraulic system is similar to the booster hydraulic system in providing hydraulic power for gimbaling the sustainer engine and for gimbaling of the two vernier engines. In addition the sustainer-vernier hydraulic system provides hydraulic power for controlling the sustainer engine propellant utilization and head-suppression valves.

The sustainer engine is held in the centered position until booster engine cutoff. Any disturbances created by booster engine differential cutoff impulses are damped by gimbaling the sustainer and vernier engines. The sustainer engine is again centered during booster engine section jettison. During the sustainer phase, roll control is maintained by differential gimbaling of the vernier engines, and pitch and yaw control is maintained by gimbaling the sustainer engine. During vernier solo operation, after sustainer engine cutoff, the vernier engine gimbal actuators provide pitch, yaw, and roll control and are provided with hydraulic pressure from two accumulators previously pressurized during sustainer operation. Actuator limit travel of the vernier engines is $\pm 70^{\circ}$ in pitch and roll and plus 20° minus 30° in yaw. The actuator limit travel of the sustainer engine is $\pm 3^{\circ}$.

Performance

Hydraulic system pressure data for both the booster and sustainer circuits are shown in table V-III. The transfer of fluid power from ground to airborne hydraulic systems prior to lift-off was normal. Starting transients associated with the transfer caused the normal 10 percent overshoot in hydraulic pump discharge pressures. Pressures after start remained stable throughout the boost flight phase. The steady-state absolute pressure values after transfer were 2172 newtons per square centimeter (3150 psi) for the booster system and 2169 newtons per square centimeter (3145 psi) for the sustainer-vernier system. During the vernier solo phase of flight, the pressure changed from the sustainer engine cutoff value of 2124 to 796 newtons per square centimeter (3080 to 1155 psi) as expected.

TABLE V-III. - ATLAS HYDRAULIC SYSTEM PERFORMANCE DATA, ATS-3

Parameter	Units	Flight values at -			
		Lift-off	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
B-1 engine accumulator	N/cm^2	2193	2172	(a)	(a)
pressure (absolute)	psi	3180	3150		
Booster pump discharge	N/cm^2	2172	2172	(a)	(a)
pressure (absolute)	psi	3150	3150		
Sustainer pump discharge	N/cm^2	2169	2158	2144	(b)
pressure (absolute)	psi	3145	3130	3110	
Sustainer-vernier pressure	N/cm^2	2144	2144	2124	796
(absolute)	psi	3110	3110	3080	1155

[a]Not applicable after booster engine cutoff.
[b]Not applicable after sustainer engine cutoff.

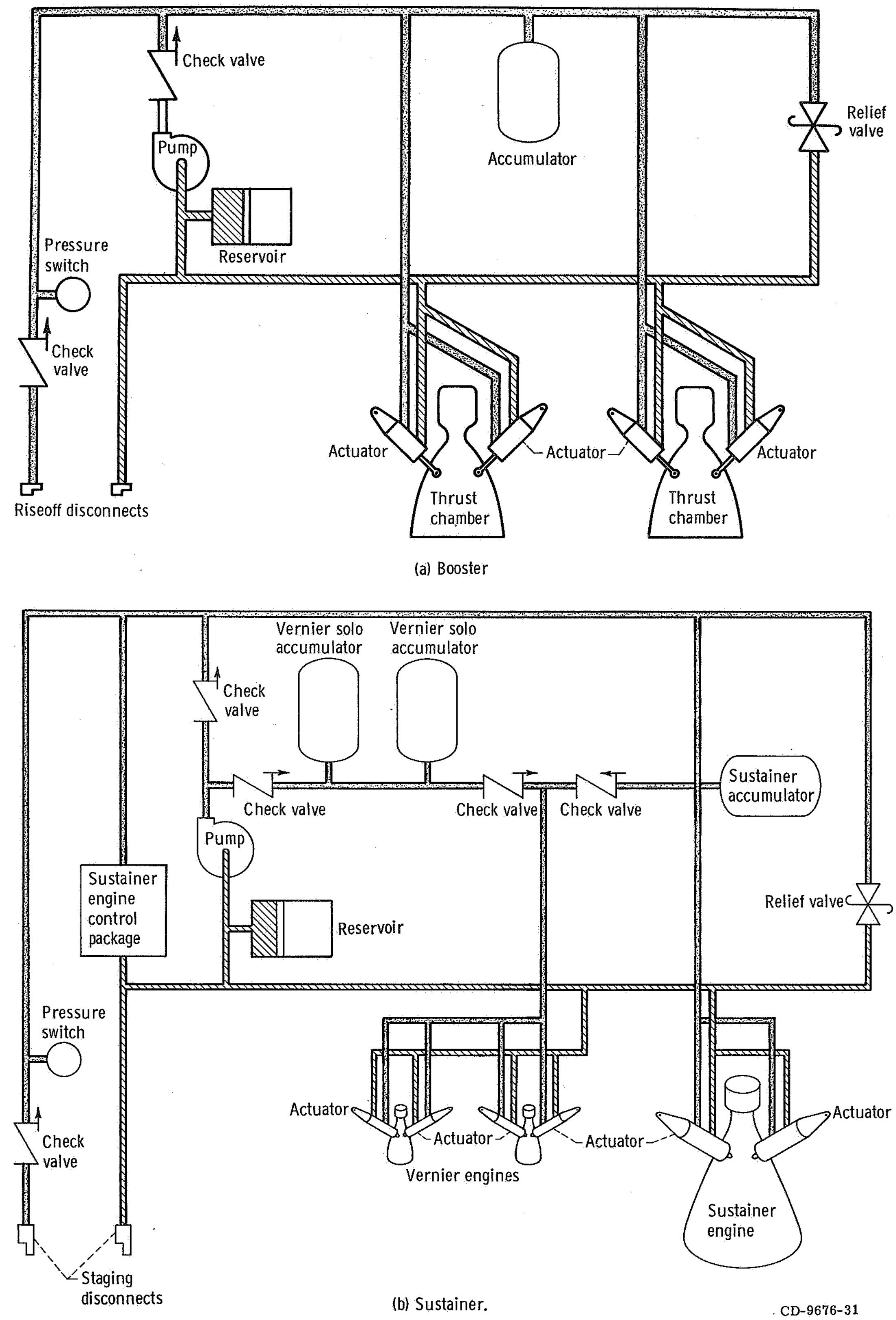


Figure V-3. - Atlas hydraulic system, ATS-3.

PROPELLANT UTILIZATION SYSTEM

by Clifford H. Arth

Description

The Atlas propellant utilization system (fig. V-4) is designed to cause near-simultaneous depletion of both propellants. This system is a digital type system which adjusts the operating mixture ratio of the sustainer engine by sampling the volume ratio of the propellants 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 exceeds the limit error times for a sensor pair, the propellant utilization valve is commanded to the full open or closed position, depending on which sensor uncovers first. If the actual error time is less than the limit error time for a sensor pair, the valve is commanded to a partly open or closed position, which would theoretically result in a zero error time when the liquid level reaches the next sensor pair.

The difference in uncovering time for each sensor pair is measured and is transmitted to a hydraulic control unit. This hydraulic control unit directly controls the position of the propellant utilization (fuel) valve and indirectly controls the position of the liquid-oxygen head-suppression 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 fuel flow increases. The liquid-oxygen head-suppression servocontrol senses this decreasing pressure and causes the liquid-oxygen head-suppression valve to move to restrict the flow of the liquid oxygen to the thrust chamber. Thus, the liquid-oxygen injection pressure is decreased by approximately the same amount as the decrease in RP-1 (fuel) pump discharge pressure. The combined operation of the liquid-oxygen head-suppression valve and propellant utilization system results in the total flow weight of propellants to the sustainer thrust chamber being nearly constant.

Performance

The burnable propellant residuals in the propellant tanks at sustainer engine cutoff are shown in the table V-IV. The residual would have allowed the sustainer engine to burn an additional 3.8 seconds, if required. However, the proper velocity was attained, and the guidance system commanded the engine to shut down. Fuel and oxidizer pressure sensing ports provide the final propellant level data. The differential pressure between the ullage and the port pressure is measured. When the pressure differential indicates zero, the propellant level is below the port. As the sensing port uncovers, a time interval is calculated from the instant of port uncovering to sustainer engine cutoff. This time interval is used to determine the propellant residuals.

Table V-V shows the error times between the fuel and liquid-oxygen sensor uncovering for each of the six sensor pairs. The data indicate 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. - PROPELLANT RESIDUAL AND SENSING PORT UNCOVERING TIMES

	Units	Propellants	
		Liquid oxygen	Fuel (RP-1)
Burnable propellant residuals	kg	328	232
	lb	722	516
Fuel remaining at theoretical liquid-oxygen depletion	kg	0	88
	lb	0	194
Time from fuel and liquid-oxygen sensor port uncovering until sustainer engine cutoff	sec	4.7	2.7

TABLE V-V. - LEVEL SENSOR ERROR TIMES, ATS-3

Sensor number	Limit error time, sec	Actual error time, sec	First sensor uncovered
[a]1	1.018	0.9	Liquid oxygen
2	.968	.2	Liquid oxygen
3	.78	.3	RP-1 (kerosene)
4	1.89	.4	RP-1 (kerosene)
5	8.4	.6	RP-1 (kerosene)
6	4.2	2.7	Liquid oxygen

[a]Sensor pair number 1 uncovers first.

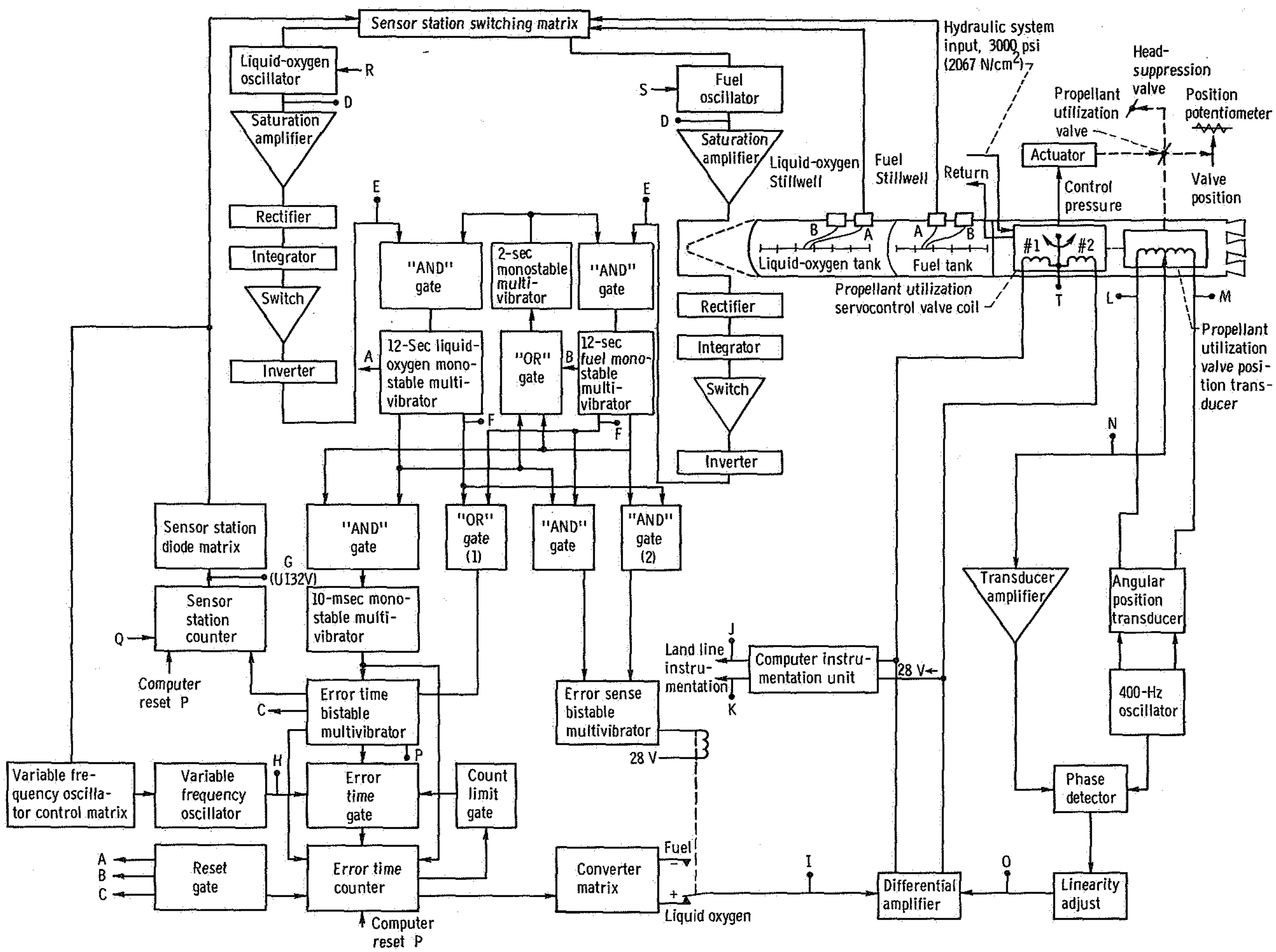


Figure V-4. - Atlas propellant utilization system, ATS-3.

PNEUMATIC SYSTEM

by Eugene J. Fourney

Description

The Atlas pneumatic system (fig. V-5) supplies helium gas for tank pressurization and for various vehicle control functions. The system consists of three independent subsystems: propellant tank pressurization, engine control, and booster engine section jettison.

Propellant tank pressurization subsystem. - This subsystem is used to maintain propellant tank pressures at required levels to support the pressure stabilized tank structure and to satisfy the inlet pressure requirements of the engine turboupmps. In addition, helium is supplied from the fuel tank pressurization line to pressurize the hydraulic reservoirs and turbopump lubricant storage tanks. The subsystem consists of six shrouded helium storage bottles, a heat exchanger, and fuel and oxidizer tank pressure regulators and relief valves. The six shrouded helium storage bottles with a total capacity of 724 144 cubic centimeters (44 190 $in.^3$) are mounted in the jettisonable booster engine section. The bottle shrouds are filled with liquid nitrogen during prelaunch operations to chill the helium in order to increase the storage capacity to about 2068 newtons per square centimeter (3000 psi). The liquid nitrogen drains from the shrouds at lift-off. During flight, the cold helium passes through a heat exchanger located in the booster engine turbine exhaust duct and is heated before being supplied to the tank pressure regulators. Control of the propellant tank pressurization subsystem is switched from the ground to the airborne regulators at about T - 60 seconds. The airborne regulators are set to control fuel tank gage pressure between 44.13 and 46.2 newtons per square centimeter (64 and 67 psi) and the oxidizer tank pressure between 19.65 and 21.37 newtons per square centimeter (28.5 and 31.0 psi). Pneumatic regulation of tank pressure is terminated at booster engine staging. Thereafter, the fuel tank pressure decays slowly. The oxidizer tank pressure is augmented by liquid-oxygen boiloff, and the decay in this tank pressure is much slower.

Engine controls subsystem. - This subsystem supplies helium pressure for actuation of engine control valves, for pressurization of the engine start tanks, for purging booster engine turbopump seals, and for the reference pressure to the regulators which control oxidizer flow to the gas generator Control pressure in this subsystem is maintain through Atlas-Agena separation. These pneumatic requirements are supplied from a 76 200-cubic-centimeter (4650-$in.^3$) storage bottle pressurized to an absolute pressure of about 2068 newtons per square centimeter (3000 psi) at lift-off.

Booster engine jettison subsystem. - This subsystem supplies pressure for release of the pneumatic staging latches to separate the booster engine section. A command from the Atlas flight control system opens two explosively actuated valves to supply helium pressure to the 10 piston-operated staging latches. Helium for this subsystem is supplied by a single 14 257-cubic-centimeter (870-$in.^3$) bottle pressurized to an absolute pressure of 2068 newtons per square centimeter (3000 psi).

Performance

The pneumatic system performance was satisfactory. Propellant tank pressures were maintained satisfactorily, and all control functions were performed properly. Pneumatic system performance data are presented in table V-VI. Liquid-oxygen-tank ullage-pressure oscillations were within the range experienced on other similar flights. Prior to lift-off, oscillation frequencies of 3.25 hertz were measured. The liquid-oxygen-tank ullage-pressure oscillations begain to damp at approximately T + 7 seconds and were completely damped by T + 29 seconds. These oscillations result from the configuration of the regulator and are considered normal.

The oscillation amplitudes of the differential pressure across the oxidizer - fuel-tank bulkhead varied with a maximum peak-to-peak amplitude of 2.41 newtons per square centimeter (3.5 psi). After lift-off, these oscillations increased in frequency to 5.25 hertz and caused the amplitude of the differential pressure across the oxidizer - fuel-tank bulkhead to vary with a maximum peak-to-peak amplitude of 6.894 newtons per square centimeter (10.0 psi).

TABLE V-VI. - ATLAS PNEUMATIC SYSTEM PERFORMANCE DATA, ATS-3

Parameter	Units	Range	Relief valve operates	Flight values at -				
				T - 10 sec	T - 0	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Oxidizer tank ullage	N/cm^2	[a]19.7 to 21.0	23.0 to 23.4	22.3	20.9	20.5	[b]19.9	19.9
pressure (gage)	psi	28.5 to 31.0	33.0 to 34.0	32.3	30.3	29.7	28.9	28.9
Fuel tank ullage	N/cm^2	[a]44.0 to 46.2	47.9 to 48.6	46.0	45.0	46.0	[b]33.0	33.0
pressure (gage)	psi	64.0 to 67.0	69.5 to 70.5	66.8	65.3	66.7	47.9	47.9
Sustainer controls helium	N/cm^2	[c]2344	(d)	2096	2027	1858	1737	1117
bottle pressure (absolute)	psi	3400		3040	2940	2695	2520	1620
Booster helium bottle	N/cm^2	[c]2344	(d)	2079	2051	458	------	------
pressure (absolute)	psi	3400		3015	2975	665		
Booster helium bottle	K	[e]86 to 77.5	(d)	80.5	78.5	45	------	------
temperature	^{o}F	-305 to -320		-315	-319	-379		

[a]Range after Atlas engine start.
[b]Helium supply bottles are jettisoned at booster engine cutoff + 3 sec. No additional helium gas is supplied to the propellant tanks.
[c]Required range for T - 0 (maximum).
[d]Not applicable.
[e]Range prior to T - 0.

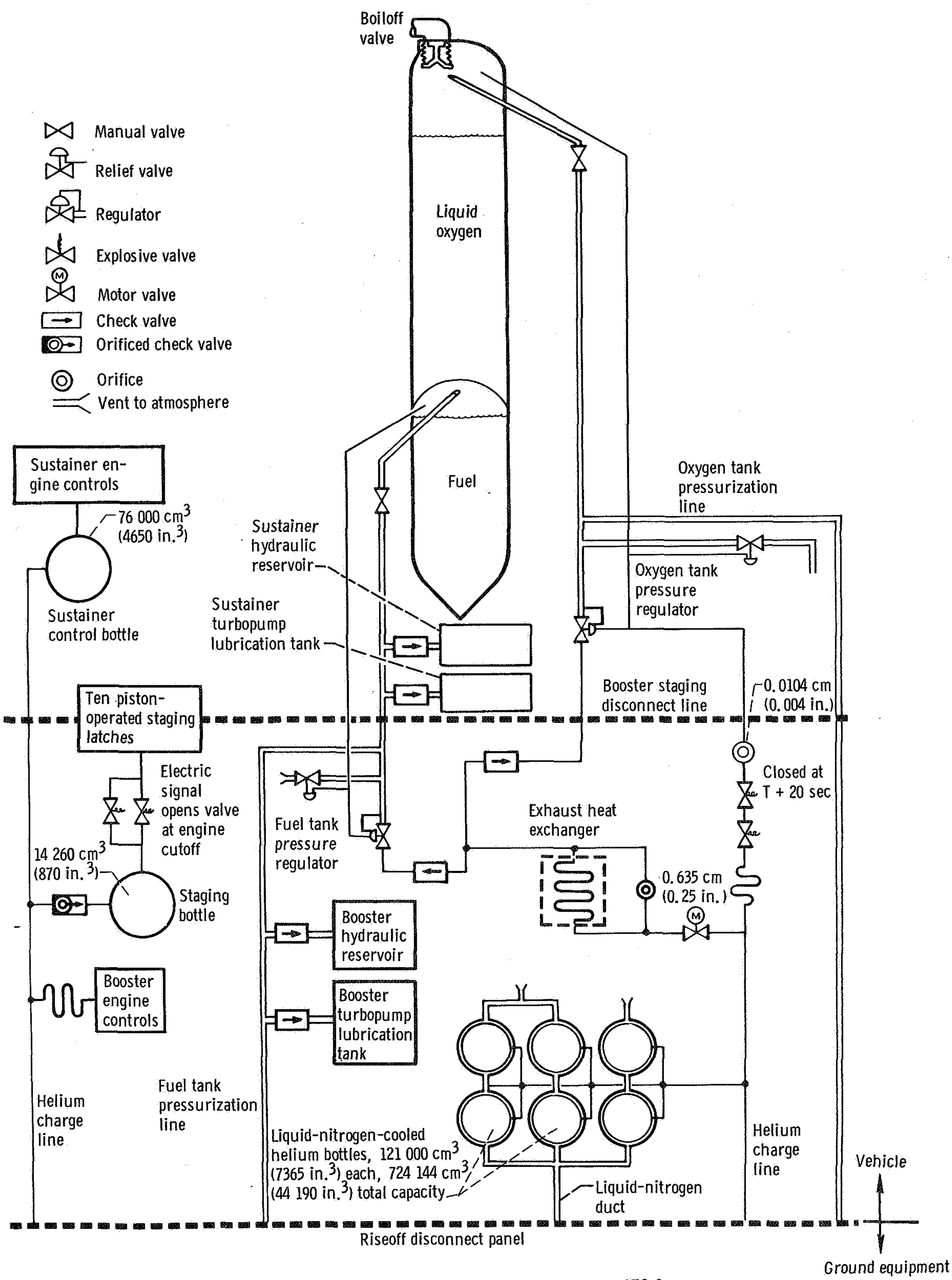


Figure V-5. - Atlas vehicle pneumatic system, ATS-3.

GUIDANCE AND FLIGHT CONTROL SYSTEM

by Dean W. Bitler and James L. Swavely

Description

The Atlas flight path is controlled by two interrelated systems: the flight control system and the Mod III Radio Guidance System. The flight control system directs the vehicle in a programmed open-loop mode from lift-off through vernier engine cutoff. During the period between T + 100 and T + 110 seconds, the Mod III Radio Guidance System may generate and transmit pitch steering signals to the vehicle. During the sustainer and vernier solo phases, the Mod III Radio Guidance System may generate and transmit pitch and yaw steering signals to the vehicle. The transmitted steering signals are received in the airborne guidance system and routed to the flight control system to provide corrections for vehicle deviations from the programmed trajectory.

The Mod III Radio Guidance System is the primary source for initiation of discrete commands for booster engine cutoff (BECO), sustainer engine cutoff (SECO), vernier engine cutoff (VECO), start Agena timer (SAT), and Atlas-Agena separation.

The Atlas flight control system (fig. V-6) consists of four major components:

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

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

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

(4) The programmer canister contains an electronic timer; arm-safe switch; high, low, and medium power electronic switches; the fixed pitch program; and circuitry to

set the roll program prior to launch. The programmer issues discrete commands to other subsystems.

The Mod III Radio Guidance System includes the Atlas airborne pulse beacon, rate beacon, and decoder; and a ground station comprised of a monopulse X-band (radar) track susbystem, a continuous wave L-band rate subsystem, and a digital guidance computer subsystem. The major functions (fig. V-7) are described in the following paragraphs.

The ground track subsystem measures range, azimuth, and elevation and transmits a composite message train containing an address code and the coded steering or discrete commands. When the address code of the received signal is correct, the airborne pulse beacon transmits a return pulse to the ground station, and the airborne decoder translates the message and then issues steering signals or discrete comands to the flight control system. The resulting steering outputs from the decoder are 400 hertz square waves of variable phase and amplitude which are transmitted to the autopilot to torque the appropriate gyro. The gyro torque rate is proportional to the decoder output. The maximum gyro torque rate is 2 degrees per second for 100 percent steering commands.

The ground rate subsystem transmits two continuous wave signals of different frequencies from a single ground antenna. The airborne rate beacon is interrogated by the signals from the ground subsystem. The rate beacon transmits a continuous wave signal at a frequency equal to the arithemetic average of the frequencies of the received signals. This signal is received by the central rate station and two outlying rate stations The two-way doppler shifts and phase relations of the signals, as received at these ground stations, are used to determine the vehicle range, azimuth, and elevation rates.

The position and rate information from the ground track and ground rate subsystems is sent to the ground computer. The ground computer solves the guidance equations every 1/2 second using position and rate information. The ground computer then generates steering and discrete commands which are transmitted from the computer to the ground track subsystem and then to the vehicle.

The ground track subsystem conical scan antenna acquires the vehicle during an early portion of the flight. Once the vehicle is acquired by the conical scan antenna, tracking is automatically switched to the main track antenna which is on the same mount as the conical scan antenna. The ground rate subsystem antennas are electronically slaved to the main track antenna.

The primary method used to acquire the vehicle is known as cube acquisition. In the cube acquisition method, the conical scan antenna is pointed to one of seven predetermined cubes along the programmed trajectory. If the vehicle is not acquired in the first cube selected, the conical scan antenna may be automatically switched to subsequent predetermined cubes. Alternate methods of acquisition are optical tracking or slaving to real-time vehicle coordinates supplied by the Eastern Test Range.

Performance

The performance of the Atlas flight control system was satisfactory. Lift-off transients in pitch, yaw, and roll were within acceptable limits. The maximum vehicle displacement angles during the lift-off transients were 0.5° counterclockwise in roll, 0.26° up in pitch, and 0.32° right in yaw. The roll program was initiated at T + 2 seconds and ended at T + 15 seconds as planned. The programmed vehicle roll required to achieve the desired launch azimuth was 15.18°. The actual vehicle roll from roll rate gyro data was 15.0°; this value was within acceptable vehicle roll requirements.

The pitch program was initiated at T + 15 seconds as planned. The programmed times, actual times, and pitch rates for each step of the pitch program are listed in table V-VII. The actual pitch maneuver dispersions were within acceptable limits.

Maximum dynamic pressure occurred at T + 63.7 seconds. Dynamic disturbances during the period of maximum dynamic pressure were small and resulted in the booster engines gimbaling a maximum of 1.15° in pitch and 0.72° in yaw. This represented 23 percent of the engine capability in pitch and 14.4 percent in yaw. The gimbal angles were within the maximum gimbal angle predictions based on data from atomospheric wind soundings taken at T - 0 (see IV. <u>TRAJECTORY AND PERFORMANCE</u>).

Vehicle displacements resulting from booster engine shutdown were 0.14° up in pitch and 0.65° left in yaw and were within acceptable limits. The booster engine jettison sequence was normal, and the resulting small yaw transient was quickly damped. The pitch transient resulted in an oscillation of 0.25 hertz with a peak-to-peak amplitude of 1.75°. This oscillation was reduced to within ±10 percent by T + 153.5 seconds. Sustainer steering was initiated at T + 137.4 seconds and was normal for this period of flight. Vernier phase of flight was normal and steering commands were within acceptable limits.

The command to separate Agena was initiated by radio guidance. At this 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 airborne guidance equipment performed satisfactorily.

The track subsystem conical scan antenna acquired the vehicle in the first cube at T + 59.1 seconds. The automatic switch to monopulse tracking with the main antenna occurred at T + 64.2 seconds, and good data were presented to the computer by T + 67.4 seconds.

Track lock was continuous from acquisition until T + 424.4 seconds, 111.3 seconds after the Atlas-Agena separation discrete command. Track lock was then intermittent until final loss of lock at the ground station, which occurred at T + 424.6 seconds. At this time, the Atlas was at an elevation angle of 1.97° above the horizon. The signal received (during the period of lock) by the ground track subsystem was within 5 decibels of the theoretically expected level.

All rate antennas were locked on the vehicle by T + 57.1 seconds, and good data were presented to the computer by T + 58.7 seconds. Rate lock was continuous unitl T + 409.7 seconds. It then became intermittent and was finally lost at T + 414.5 seconds, when the received rate signal had decreased to the "noise" level. 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 3 decibels of those received at the central rate antenna.

Performance of the computer subsystem was satisfactory throughout the countdown and vehicle flight. 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 (AGC) monitor indicated a received signal strength of -53 dBm (decibels referenced to 1 mW) at the time of acquisition (T + 59.1 sec) with a signal increase to approximately -32 dBm at T + 64 seconds. The received signal strength reached a maximum of -7 dBm at T + 74.5 seconds, and gradually decreased to -34 dBm at Agena separation. The signal strength continued to decrease until T + 417 seconds, when the received signal strength was less than -78 dBm. Momentary loss of pulses occurred as expected during the booster engine staging sequence. This loss occurred because of signal attenuation when the booster engine section separated from the Atlas.

The magnetron current monitor data obtained on Atlas telemetry also indicates satisfactory performance of the airborne pulse beacon. Both AGC and magnetron current data show normal transients due to loss of telemetry signal during booster staging.

The rate beacon automatic gain control monitors, numbers 1 and 2, indicated that the received signal strength of the two carrier frequencies was intermittent during acquisition until T + 55.5 seconds, they reached a signal level greater than -75 dBm, and remained so until approximately T + 397 seconds. Both signal strengths gradually decreased to the threshold sensitivity of the receiver, -85.5 dBm, at approximately T + 409.5 seconds. The rate beacon phase detector and power output monitors indicated that the received signals were processed and that the return signal was properly transmitted to the ground station during the period from T + 54 to T + 409.5 seconds.

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

Spurious pitch and yaw commands were observed, as in prior flights, during the periods of intermittent pulse beacon lock between T + 51.0 and T + 56.5 seconds during the first cube acquisition period. These spurious commands were less than ±20 percent of maximum steering commands during the interval from T + 51.0 to T + 56.5 seconds, and were properly inhibited by the airborne flight control system.

During the booster phase, the capability of the airborne flight control system to accept Mod III Radio Guidance commands was enabled at T + 80 seconds.[1] No steering commands were transmitted during this interval since the pitch attitude errors were within the ±1-sigma error band.

Sustainer steering was initiated at T + 137.4 seconds. The maximum steering command outputs from the decoder were a pitch up of 92 percent (100 percent equals 2 deg/sec gyro torque rate)[2] at T + 139.5 seconds and a yaw right of 12 percent at T + 138 seconds. Pitch and yaw steering commands from the decoder were reduced to within ±10 percent by T + 153.5 seconds and remained within these limits until sustainer engine cutoff. The amplitude and duration of these commands indicated normal steering by radio guidance.

The largest steering command output from the decoder during vernier solo phase was a 45-percent pitch-down command, which was reduced to within ±10 percent during this phase. These commands were within the acceptable limits.

Table V-VIII lists the times at which the actual booster engine cutoff (BECO), sustainer engine cutoff (SECO), start Agena timer (SAT), vernier engine cutoff (VECO), and Atlas-Agena separation discretes were generated at the guidance computer.

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[1]However, pitch steering commands were inhibited by the groud guidance system except for the period between T + 100 to T + 110 sec.

[2]See IV. TRAJECTORY AND PERFORMANCE for correlation of percentages with absolute values of steering commands.

TABLE V-VII. - ATLAS PITCH PROGRAM, ATS-3

Time interval		Step level	
Programmed, sec	Actual, sec	Programmed, deg/sec	Actual, deg/sec
0 to 15	0 to 15	0	0
15 to 35	15 to 35	1.018	1.000
35 to 45	35 to 45	.848	.809
45 to 58	45 to 58	.509	.485
58 to 70	58 to 70	.678	.646
70 to 82	70 to 82	.806	.806
82 to 91	82 to 91	.678	.647
91 to 105	91 to 105	.551	.485
105 to 120	105 to 120.1	.382	.322
120 to 138.2	120.1 to sometime during sustainer pitch steering[a]	.254	.242
138.2 to sustainer engine cutoff	Sometime during sustainer pitch steering[a] to sustainer engine cutoff	.042	(b)

[a]Specific time determination cannot be made because of telemetry channel resolution.
[b]Less than telemetry channel resolution.

TABLE V-VIII. - MOD III RADIO GUIDANCE COMPUTER DISCRETE TIMES, ATS-3

Flight event	Guidance computer generated discrete times,[a] sec
Booster engine cutoff	T + 128.155
Sustainer engine cutoff	T + 288.948
Start Agena timer	T + 300.038
Vernier engine cutoff	T + 309.138
Atlas-Agena separation	T + 313.155

[a]Time discrete was initiated by ground guidance system.

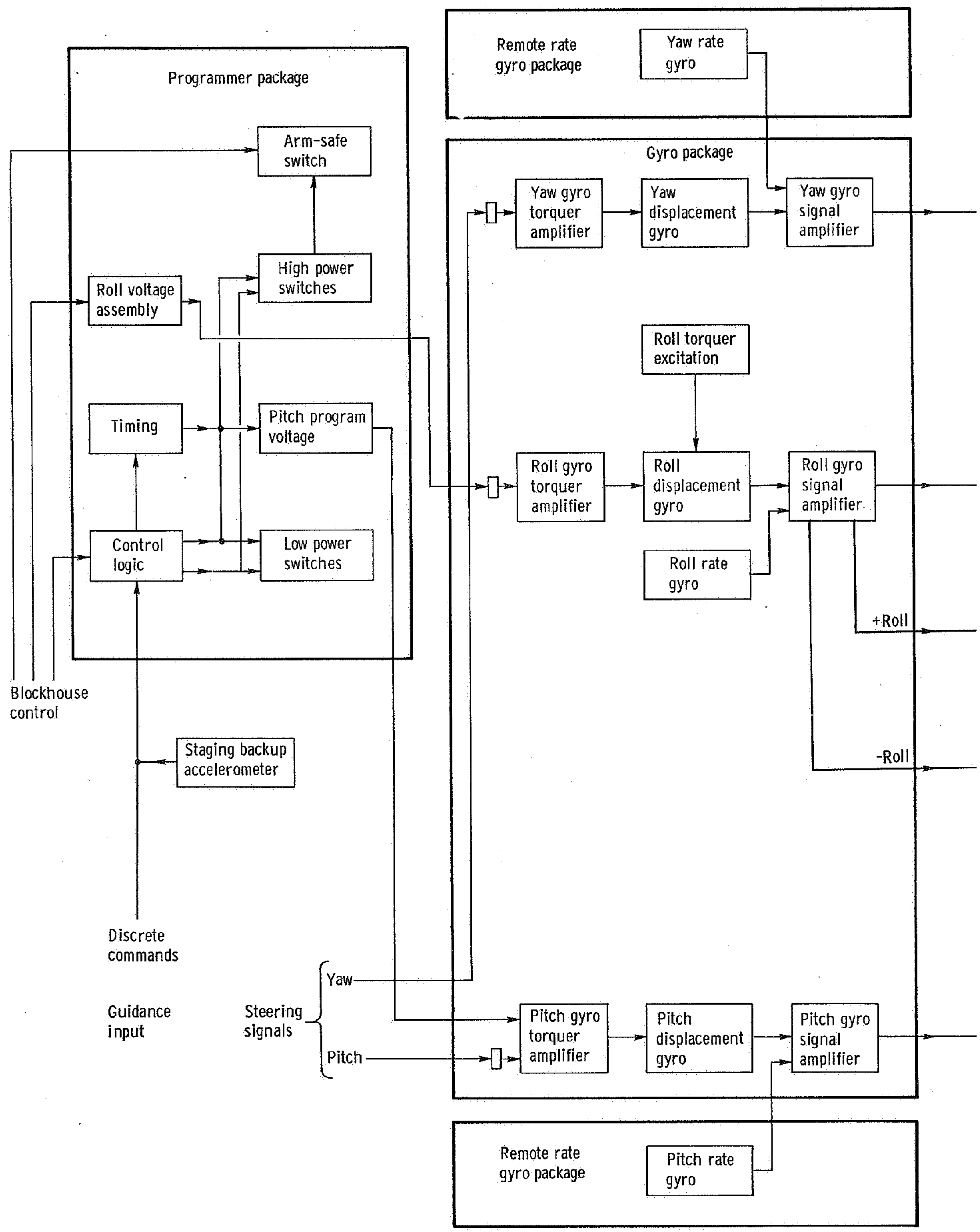
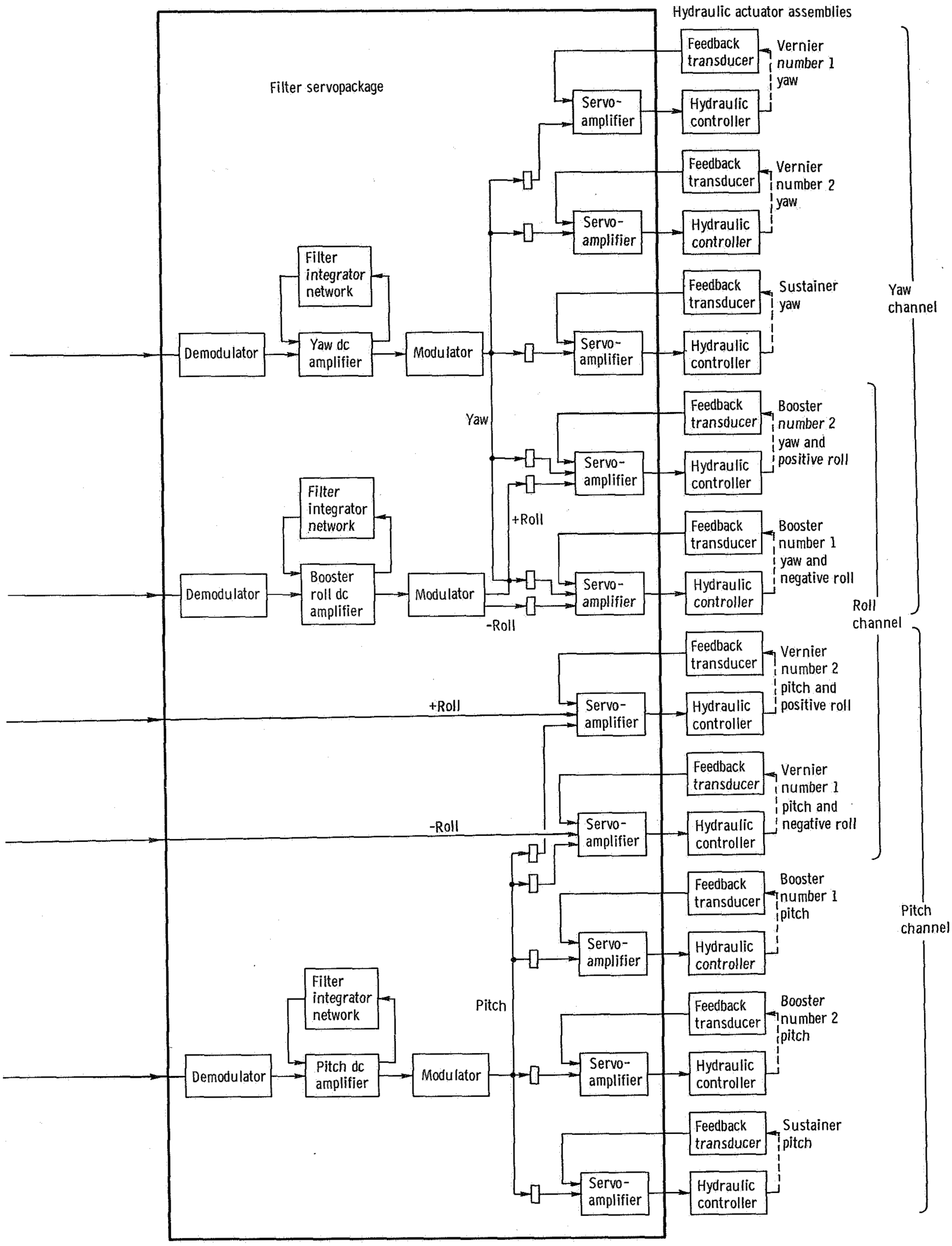


Figure V-6. - Atlas flight cc



system block diagram, ATS-3.

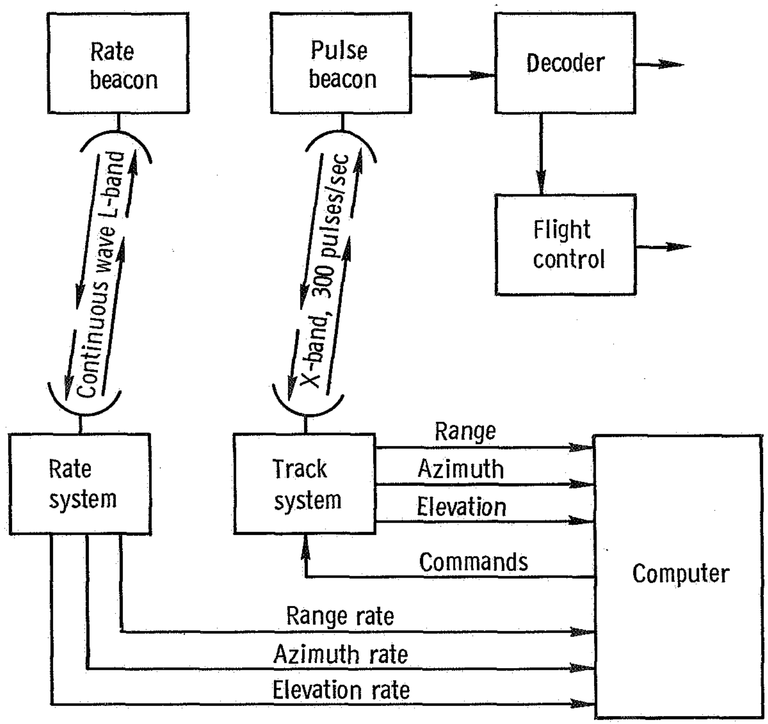


Figure V-7. - MOD III Guidance System block diagram, ATS-3.

ELECTRICAL SYSTEM

by Clifford H. Arth

Description

The Atlas 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, three-phase, 400-hertz inverter; a power changeover switch; a distribution box; two junction boxes; and related electrical harnesses. The main 28-volt dc battery supplies power to the flight control system, the airborne radio guidance system, the propellant utilization system, the propulsion system, and the inverter. Another 28-volt dc battery supplies power to the telemetry system, and the remaining two 28-volt dc batteries supply power to the flight termination system. The inverter supplies power to the flight control system, the propellant utilization system, and the airborne radio guidance system. Phase A of the inverter is used as a phase reference in the flight control and the radio guidance system.

The vehicle flight control, propulsion, airborne radio guidance, and propellant utilization systems operate from ground regulated dc and ac power sources until 2 minutes prior to lift-off. At this time, the power changeover switch is used to transfer from ground power sources to vehicle electrical power supply.

Performance

All measured electrical system parameters were within specifications at all times. Table V-IX shows the maximum inverter frequency deviation was 0.8 hertz at lift-off, the maximum deviation from a nominal of 115 volts ac was 0.3 volt ac for phase A, and that the battery voltage deviated 0.3 volt dc from the nominal of 28 volts dc.

TABLE V-IX. - ATLAS ELECTRICAL SYSTEM PERFORMANCE, ATS-3

Characteristic	Units	Tolerance	Lift-off	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff	Atlas-Agena separation
Main battery voltage	V dc	$28^{+2.0}_{-1.5}$	27.9	27.7	27.9	27.9	27.9
Inverter frequency	Hz	400±6	399.2	400.0	400.0	400.3	400.3
Phase A voltage	V ac	115±0.5	115.3	115.1	114.9	114.9	114.9
Phase B voltage	V ac	115±1.7	115.9	115.9	115.8	115.7	115.7
Phase C voltage	V ac	115±1.7	116.0	115.9	115.9	115.8	115.8

TELEMETRY SYSTEM

by Edwin S. Jeris

Description

The Atlas telemetry system consists of a telemetry package, a manually activated 28-volt dc battery, associated transducers, wiring harnesses, and two antennas. Appendix B summarizes the launch vehicle instrumentation.

The 18-channel Pulse Amplitude Modulation/Frequency Modulation/Frequency Modulation (PAM/FM/FM) telemetry package consists of a transmitter, commutator assemblies, signal conditioning components, and the subcarrier oscillators. The telemetry transmitter has an output power level of 3.5 to 6 watts and requires 28 volts dc for operation. The transmitter is designed to use standard Interrange Instrumentation Group (IRIG) subcarrier channels 1 to 18 (see table V-X). The outputs of all subcarrier channels are multiplexed to allow continuous frequency modulation of the 249.9 megahertz carrier wave.

Performance

Atlas telemetry performance was satisfactory on ATS-3. All 114 measurements (table V-XI) flown yielded usable data. No telemetry problems occurred during the countdown or during the flight. Signal strength was adequate during flight except for the expected 1-second loss of signal at booster engine section jettison. This loss of signal occurs as a result of signal attenuation by the Atlas sustainer engine exhaust plume. Carrier frequency and commutator speeds were stable and no data playback difficulties were encountered. The location of the telemetry stations used for support and the telemetry coverage provided are shown in appendix C.

TABLE V-X. - ATLAS TRANSMITTER

SUBCARRIER CHANNELS, ATS-3

Channel	Type
1	Not used for this program
2	Not used for this program
[a]3	Continuous direct (no subcarrier oscillator)
[a]4	Continuous direct (no subcarrier oscillator)
5	Continuous (subcarrier oscillator)
6	
7	
8	
9	
10	
11	Commutated at 2.5 revolutions/sec
[a]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

[a]Transducer provides the modulating frequency.

TABLE V-XI. - ATLAS

[Blocked-in measurements are either inactive or will indicate less than 5 percent of

Analysis category		Interrange Instrumentation Group									
		1	2	3	4	5	6	7	8	9	10
		Continuous									
Airframe	Adapter										
	Structural							M79A			
	Engine area										
Propulsion system	Engine controls								P77X P347X		
	Propellant feed			P349B	P84B						
	Engine performance										
Control systems	Flight control					S54B M32X S359X	S209V	S254D M30X	S53R	S52R S234X S241X S245X S248X	
	MOD III guidance					Y41X					
Support systems	Propellant utilization										
	Pneumatic										
	Hydraulic										
	Electrical										E151V
	Range safety					D1V					
Number of measurements		0	0	1	1	5	1	3	3	5	1
Total number of											

INSTRUMENTATION SUMMARY, ATS-3

information bandwidth prior to launch. They become active either at engine ignition or later in flight.]

(IRIG) channel number and commutation rate								Alternate measurements indicative of performance
11	12	13	14	15	16	17	18	
2.5 rps	Continuous	5 rps	Open	10 rps		Open	30 rps	
Y45T		Y44P						
	U101A							F1P P116P F3P
A743T P15T A745T P16T P671T								
		P26P P344P			P616X			F291P S241X H140P S245X S236X
P117T P530T		P2P P47P P27P P49P P30P P55P		P83B P100P P330P			P1P P56P	
							P6P P59P P28P P60P P29P P339P	P330P U101A
				S61D S62D S63D S190V S384X	S252D S259D S253D S260D S255D S261D S256D S290X S257D S291X S258D			M79A P77X P347X P616X U101A
		G296V G298V G354V		G4C G280V G82E G282V G3V G287V G279V G288V	G590V G592V G591V G593V			S61D S234X S62D S241X S63D S243X S52R S248X S53R U101A S54R
		P529D U113V P830D U132V		U112V	U80P U81P		U134V U605V U135V	
P247T		F125P F288P F246P F291P		F1P F3P			F116P	
		H3P		H33P H140P H130P H224P			H601P	
		E95V E96V					E28V E52V E31V E53V	
				D1V D7V	D3X			
9	1	23	0	25	19	0	17	

measurements - 114

FLIGHT TERMINATION SYSTEM

by Edwin S. Jeris

Description

The Atlas contains an airborne, flight termination system (fig. V-8) which is designed to function on receipt of command signals from the ground stations. This system includes two receivers and two batteries for redundancy, two antennas (shared with the telemetry system), a power control unit, an electrical arming unit, a ring coupler, and a destructor. The two batteries operate independently of the vehicle main 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 conical shaped charge, and the liquid propellants are dispersed. The operation of the flight termination system is under command of the Range Safety Officer only.

Performance

Performance of the flight termination system was satisfactory. Prelaunch checks were completed without incident. The only measurements telemetered were receiver number 1 automatic gain control and engine cutoff and destruct commands. The automatic gain control measurement on receiver number 1 indicated that the capability to terminate the flight was maintained throughout powered flight. Minimum signal strength measured at the receiver (airborne) was 121 microvolts except for the expected loss of signal at booster engine section jettison. The minimum required signal strength for receiver operation is 5 microvolts. Receiver number 2 was not instrumented. No flight termination commands were required, nor were any commands inadvertently generated by any vehicle system.

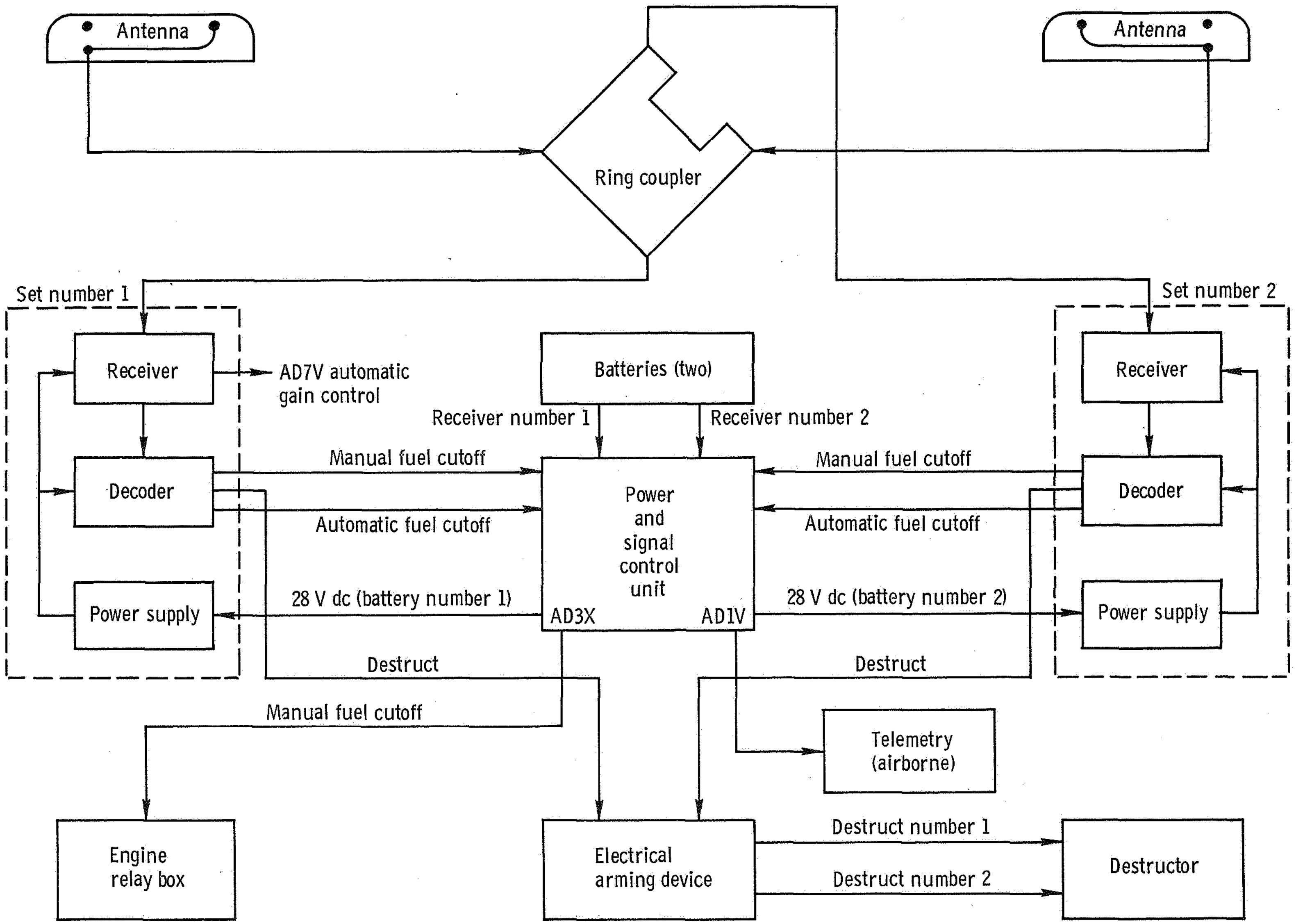


Figure V-8. - Atlas flight termination system, ATS-3.

VI. AGENA VEHICLE SYSTEM PERFORMANCE

VEHICLE STRUCTURE SYSTEM

by Robert N. Reinberger

Description

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

Performance

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

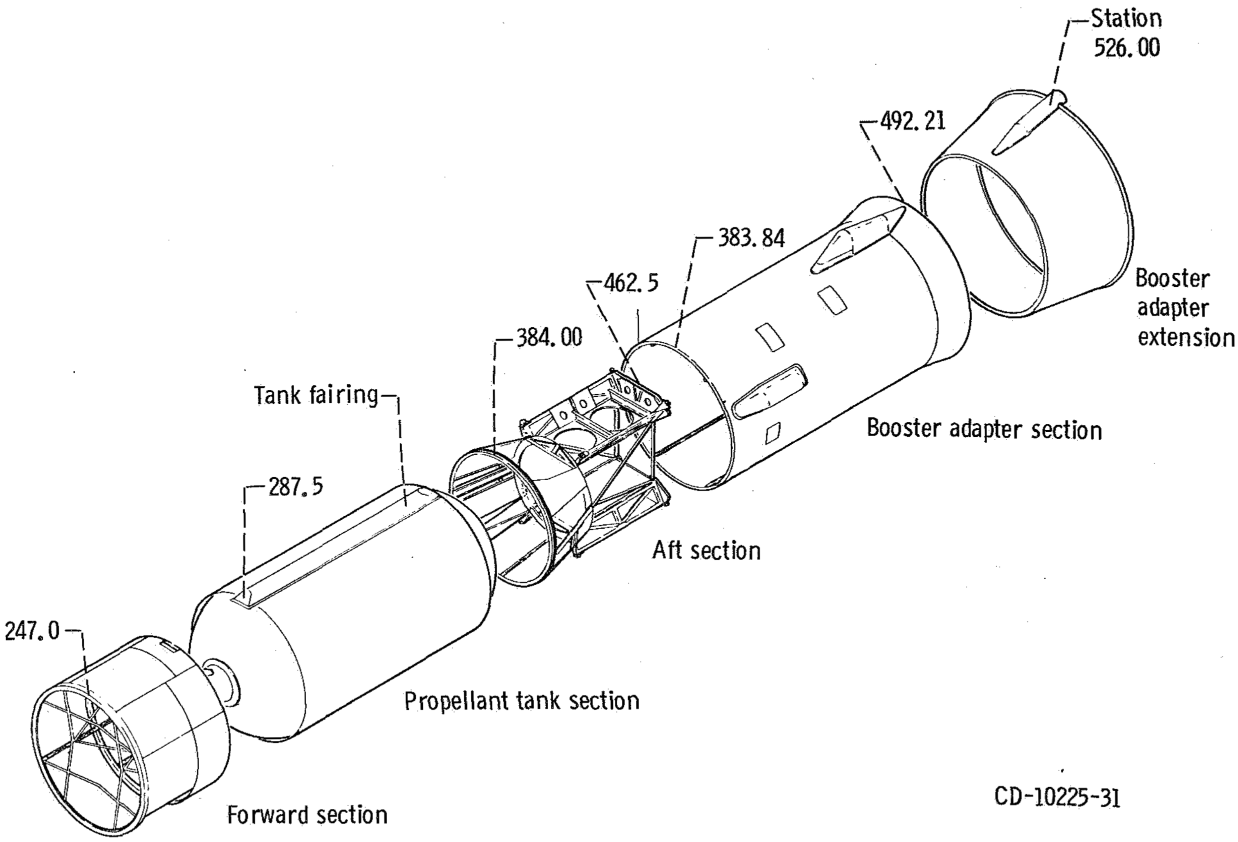


Figure VI-1. - Agena vehicle structure system, ATS-3.

SHROUD SYSTEM

by C. Robert Finkelstein and W. Dyer Kenney

Description

The shroud system is the Standard Agena Clamshell (SAC) shroud with minor mission modifications incorporated. The shroud provides environmental protection for the spacecraft prior to and during launch.

The Standard Agena Clamshell shroud (fig. III-4) is 5.72 meters (18.78 ft) long and weighs 328.24 kilograms (723 lb). It consists of an aluminum transition ring and two shroud halves. These two shroud halves form a fairing with a 1.65-meter- (5.42-ft-) diameter cylindrical section, a 15^{o} half-angle conical section, and a 0.61-meter- (2-ft-) diameter hemispherical nose cap. The shroud halves are constructed of laminated fiberglass strengthened by internal, aluminum longerons and ribs. Microquartz thermal insulation blankets in the cylindrical section of each shroud half and an aluminum foil covering in the conical section of each half provide thermal protection for the spacecraft. The shroud halves are held together by a nose latch, two flat bands around the cylindrical section, and a V-band around the base of the cylindrical section. The top, middle, and bottom bands are tensioned to 22 250, 11 570, and 35 600 newtons (5000, 2600, and 8000 lb).

The V-band clamps the shroud to the transition ring, which is approximately 0.051 meter (0.17 ft) high and is bolted to the forward end of the Agena. The transition ring attaches both the shroud and the spacecraft adapter to the Agena. Figure VI-2 shows a simulated ATS spacecraft and spacecraft adapter mounted on the forward end of the transition ring. A metal diaphragm attached to the transition ring isolates the shroud cavity from the Agena. During ascent, the shroud cavity is vented through four ports in the cylindrical section of the shroud. These ports permit venting in an outward direction only.

Shroud jettison is initiated by a Mod III Radio Guidance discrete approximately 10 seconds after the initiation of Agena first burn. At this time, Agena electrical power is used to fire two pyrotechnic bolt cutters in the nose latch assembly and two explosive bolts in each of the three bands. The firing of one bolt cutter in the nose latch and one bolt in each of the bands will effect shroud separation. When the bolt cutter and the bands are released, two pairs of springs in each shroud half thrust against the transition ring and provide the energy to rotate each shroud half about hinges mounted on the transition ring. At the time of shroud separation, the Agena has an acceleration of approximately 1 g. At this acceleration level, each shroud half rotates through an angle of about 75^{o} before it leaves the hinges and falls free. The shroud separation springs have enough energy to jettison the shroud halves successfully during a vehicle longitudinal acceleration up to 3.5 g's.

The shroud system is instrumented with two temperature transducers and one differential pressure transducer. The temperature transducers are located on the inner surface of the shroud fiberglass skin at Agena station 176. The pressure transducer measures differential pressure between the shroud cavity and the Agena.

The ATS-3 spacecraft - adapter and the shroud are mated to the transition ring in an environmentally clean room, and the complete assembly is then transported to the launch pad and mated to the Agena.

Performance

The shroud internal wall temperature history is shown in figure VI-3. The maximum temperature measured was 367.8 K (202° F) at approximately T + 180 seconds. This measured temperature was well within the maximum predicted temperature of 466.7 K (380° F) for a 3-sigma depressed trajectory.

The history of the differential pressure across the diaphragm separating the shroud cavity and the Agena is shown in figure VI-4. The differential pressure was essentially zero during the early portion of the flight. During the transonic portion of the flight, the differential pressure was -0.57 newton per square centimeter (0.83 psi). When the pressure in the shroud cavity is less than the pressure in the Agena forward equipment section, the resulting differential pressure is defined as negative. This negative differential pressure was caused by shock waves on the vehicle during transonic flight. After the transonic portion of the flight, the differential pressure became slightly positive for a short period of time; it then returned essentially to zero for the remainder of the flight.

Shroud pyrotechnics were fired at T + 381.1 seconds. At this time, the vehicle roll and yaw rates were very nearly zero, and the pitch rate was at the programmed geocentric value. No measurable Agena roll, pitch, or yaw rates developed as a result of shroud jettison.

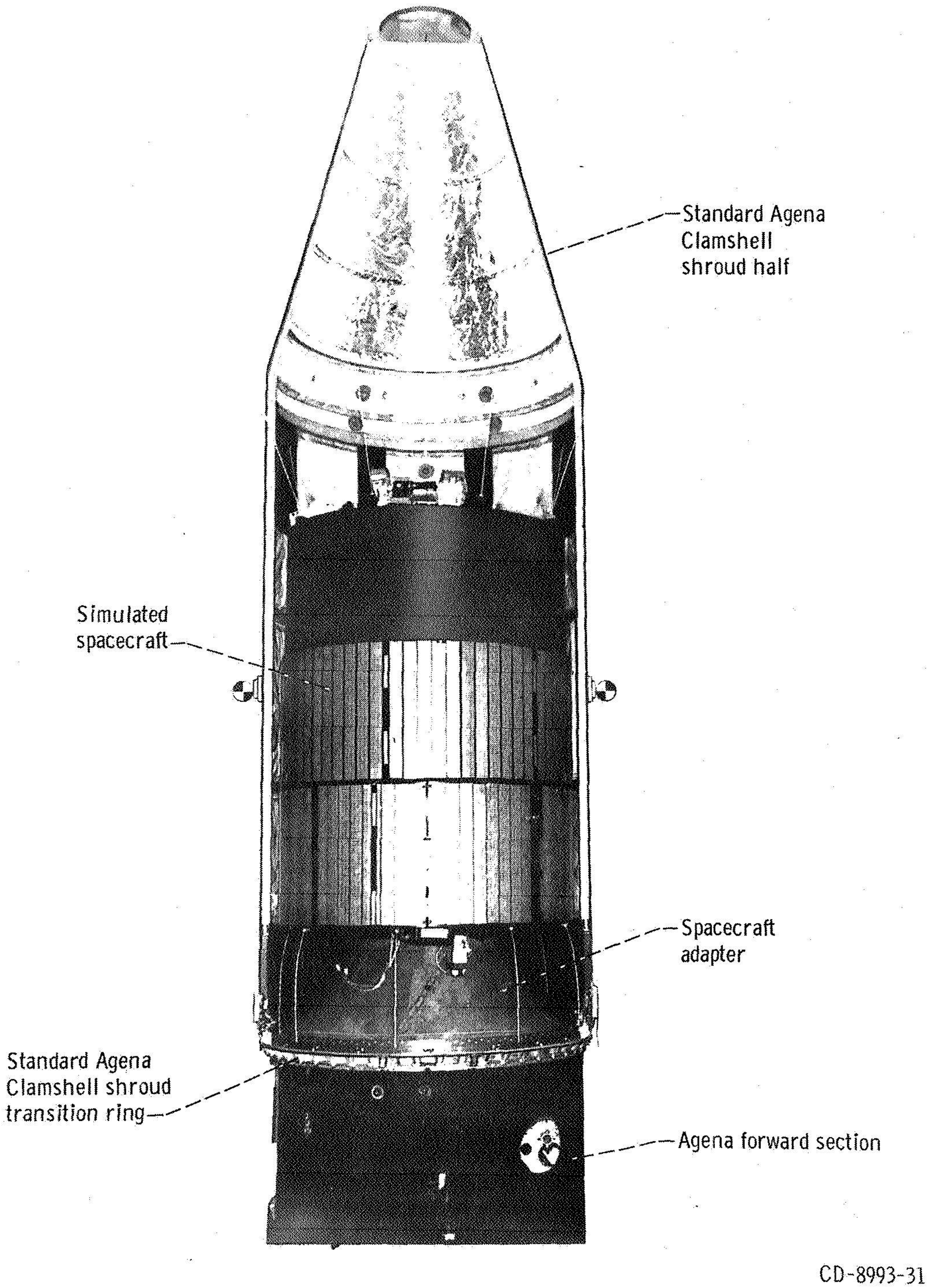


Figure VI-2. - Shroud - simulated spacecraft, ATS-3.

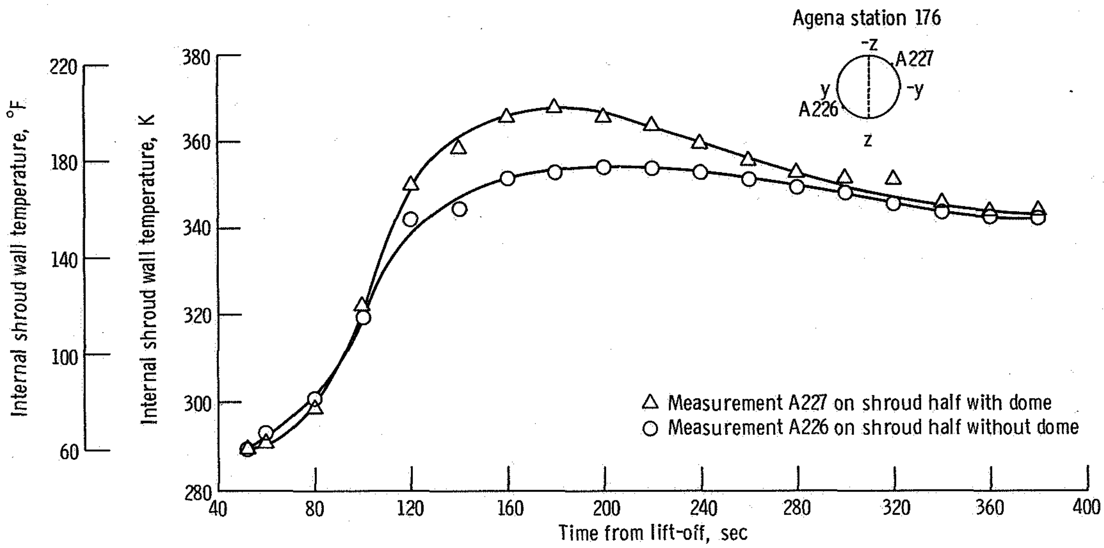


Figure VI-3. - Shroud internal wall temperature history, ATS-3.

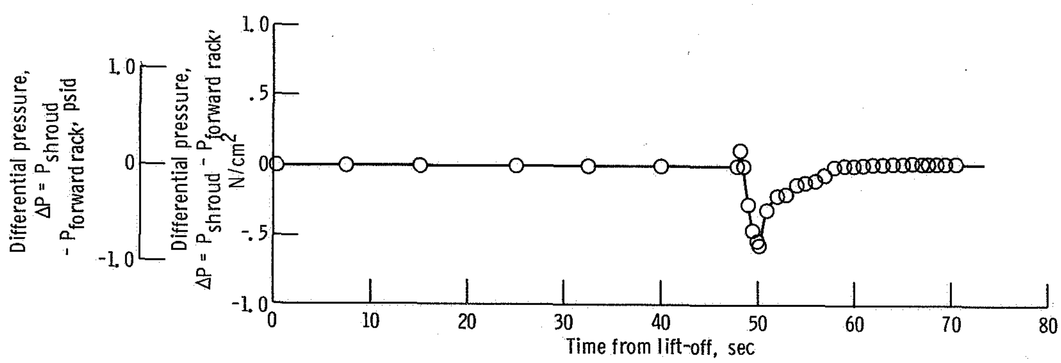


Figure VI-4. - Diaphragm differential pressure history, ATS-3. Measurement number, A519.

PROPULSION SYSTEM

by Robert J. Schroeder

Description

The Agena propulsion system, as shown in figure VI-5, consists of a propellant tank pressurization system, a propellant management system, and an engine system. Also considered a part of the propulsion system are the Atlas-Agena separation system and Agena vehicle pyrotechnic devices.

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

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

The Agena engine is a Bell Aerosystems Company Model 8096 liquid bipropellant engine, which uses unsymmetrical dimethylhydrazine (UDMH) fuel and inhibited red fuming nitric acid (IRFNA) oxidizer. Rated thrust in a vacuum is 71 171 newtons (16 000 lb) with a nozzle expansion area ratio of 45. The engine has a regeneratively cooled thrust chamber and a turbopump-fed propellant flow system. Turbine rotation is initiated for each engine firing by a solid propellant start charge. The turbine is driven during steady-state operation by hot gas produced in a gas generator. Propellants to the gas generator are supplied by the turbopump. An oxidizer fast-shutdown system consisting of a pyrotechnically operated valve and a high pressure nitrogen storage cylinder is used to close the main oxidizer valve rapidly at first-burn cutoff. Engine thrust vector control is provided by the gimbal-mounted thrust chamber. A pair of hydraulic actuators provides the force for pitch and yaw thrust chamber movement in response to signals produced by the Agena guidance system.

Atlas-Agena separation is accomplished by firing a Mild Detonating Fuse which structurally severs the booster adapter circumferentially near the forward end. The Atlas and booster adapter are then separated from the Agena by firing two solid propellant retrorockets mounted on the booster adapter. Rated average sea level thrust of each retrorocket is 2180 newtons (490 lb) with an action time of 0.93 second.

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

Performance

The Agena engine first burn was initiated by the primary sequence timer at T + 371.1 seconds. Telemetered data of the engine switch group monitor indicated a normal start sequence of the engine control valves. At T + 372.3 seconds, 90 percent of combustion chamber pressure was reached. The average steady-state thrust generated by the Agena engine was 71 728 newtons (16 125 lb) compared with an expected value of 72 203 newtons (16 232 lb). The Agena engine burn was terminated by the velocity meter cutoff command at T + 533.6 seconds. The engine burn duration, measured from 90 percent chamber pressure to velocity meter cutoff command was 161.3 seconds. This was 1.1 seconds longer than the expected value of 160.2 seconds. The actual burn time and thrust level indicate that engine performance was within the allowable 3-sigma limit.

The propellant tank pressurization system supplied the required tank pressures. This was evidenced by the fuel and oxidizer pump inlet pressure values which were within 1.4 newtons per square centimeter (2 psi) of the expected values during the first burn.

The propellant isolation valves were commanded to close at T + 541.1 seconds (approximately 8 sec after Agena engine cutoff). The fuel and oxidizer propellant isolation values functioned normally, as evidenced by the valve position measurement and the decrease in pump inlet pressures.

Near the end of the Agena coast period, the propellant isolation valves were commanded to open at T + 1502.1 seconds and both valves functioned normally. The Agena engine second burn was initiated 2 seconds later at T + 1504.1 seconds. Telemetered data of the engine switch group monitor indicated a normal start sequence of the engine control valves. At T + 1505.2 seconds, 90 percent combustion chamber pressure was reached. The average steady-state thrust generated by the Agena engine was 71 599 newtons (16 096 lb) compared with an expected value of 71 865 newtons (16 156 lb). The engine second burn was terminated by the velocity meter cutoff command at T + 1583.5 seconds. The engine burn duration, measured from 90 percent combustion chamber pressure to velocity meter cutoff command, was 78.3 seconds. This was 0.1 second longer than the expected value of 78.2 seconds. The actual burn time and thrust level indicate that engine performance was within the allowable 3-sigma limit.

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

The Atlas-Agena separation system performance was normal. Separation was commanded by the ground Mod III Radio Guidance System and occurred at T + 313.5 seconds when the Mild Detonating Fuse and the two retrorockets were ignited. Complete separation of the Atlas and the Agena was accomplished in 2.5 seconds.

All the Agena pyrotechnic devices performed their intended functions satisfactorily. The oxidizer fast-shutdown-valve squib produced a momentary shorting of the bridge wires after ignition, but this had no adverse effect on vehicle performance.

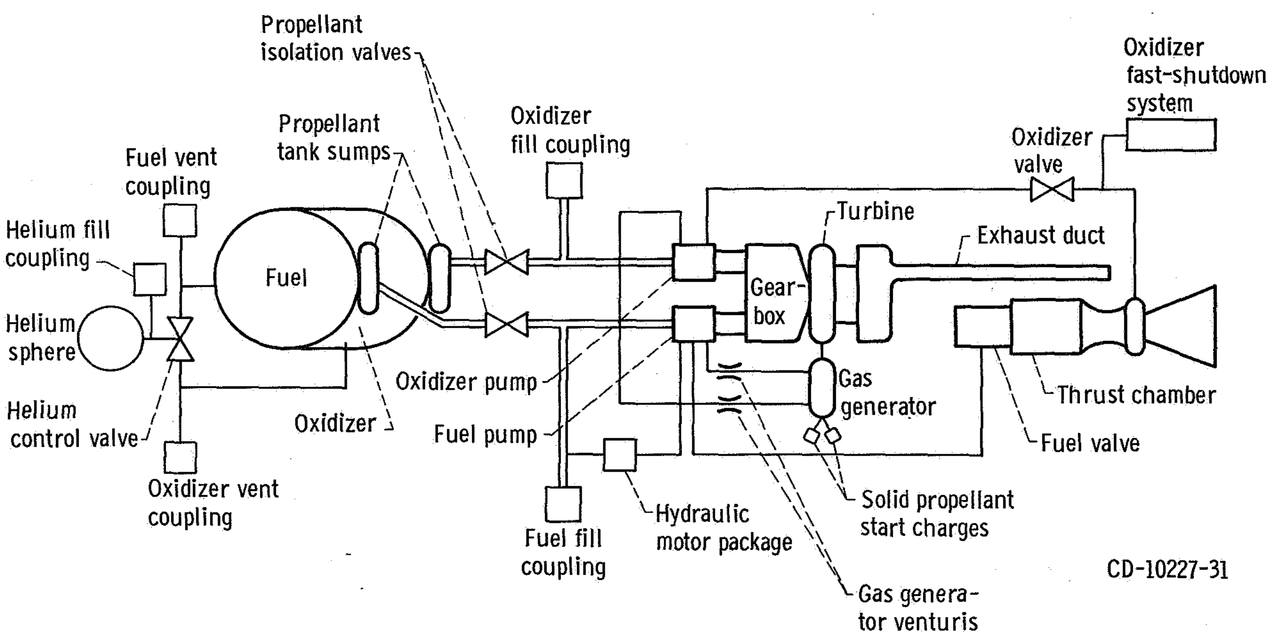


Figure VI-5. - Agena propellant system schematic, ATS-3.

ELECTRICAL SYSTEM

by Edwin R. Procasky

Description

The Agena electrical system (fig. VI-6) supplies all power for the guidance, telemetry, flight termination, propulsion, and pyrotechnic systems. The electrical system consists of two silver-zinc primary batteries, two nickel-cadmium secondary batteries, a dc-dc converter, one solid-state three-phase inverter, and associated electrical harnesses. Another dc-dc converter is located in the telemetry system.

The two primary batteries, with a minimum 3-sigma rating of 1320 watt-hours each, provide power to all systems except the flight termination system, which is powered by the two secondary batteries. The three-phase inverter supplies regulated 115 volts ac (rms) at 400 hertz (±0.02 percent) to the guidance system. One dc-dc converter supplies regulated ±28 volts dc to the guidance system, while the other dc-dc converter supplies 28 volts dc to the telemetry system.

Performance

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

The battery current load profile was as expected for this mission. The inverter and converter voltages were within specification at lift-off and remained essentially constant throughout flight. Table VI-I summarizes the electrical system performance.

The inverter frequency is not monitored on Agena vehicles; however, performance of the guidance system indicated that the inverter frequency was normal and stable.

TABLE VI-I. - AGENA ELECTRICAL SYSTEM FLIGHT PERFORMANCE SUMMARY, ATS-3

Measurement	Tolerance[a]	Measurement number	Flight values at -					
			Lift-off	First ignition	First shutdown	Second ignition	Second shutdown	Spacecraft separation
Pyrotechnic battery voltage, V dc	22.5 to 29.5	C141	26.2	26.2	26.2	26.2	26.2	26.2
Main battery voltage, V dc	22.5 to 29.5	C1	25.1	24.9	25.0	24.9	25.4	25.4
Battery current, A	-------------	C4	14	17	14	17	14	17
Guidance converter, 28.3 V dc regulated	27.7 to 28.9	C3	27.7	27.7	27.7	27.7	27.7	27.7
Guidance converter, -28.3 V dc regulated	-27.7 to -28.9	C5	-28.7	-28.7	-28.7	-28.7	-28.7	-28.7
Guidance inverter, phase AB, V ac (rms)	112.7 to 117.3	C31	114.7	114.7	114.7	114.7	114.7	114.7
Guidance inverter, phase BC, V ac (rms)	112.7 to 117.3	C32	114.7	114.7	114.7	114.7	114.7	114.7
Telemetry converter number 1, 28.3 V dc regulated	27.7 to 28.9	H204	28.4	28.4	28.4	28.4	28.4	28.4

[a]For telemetry data.

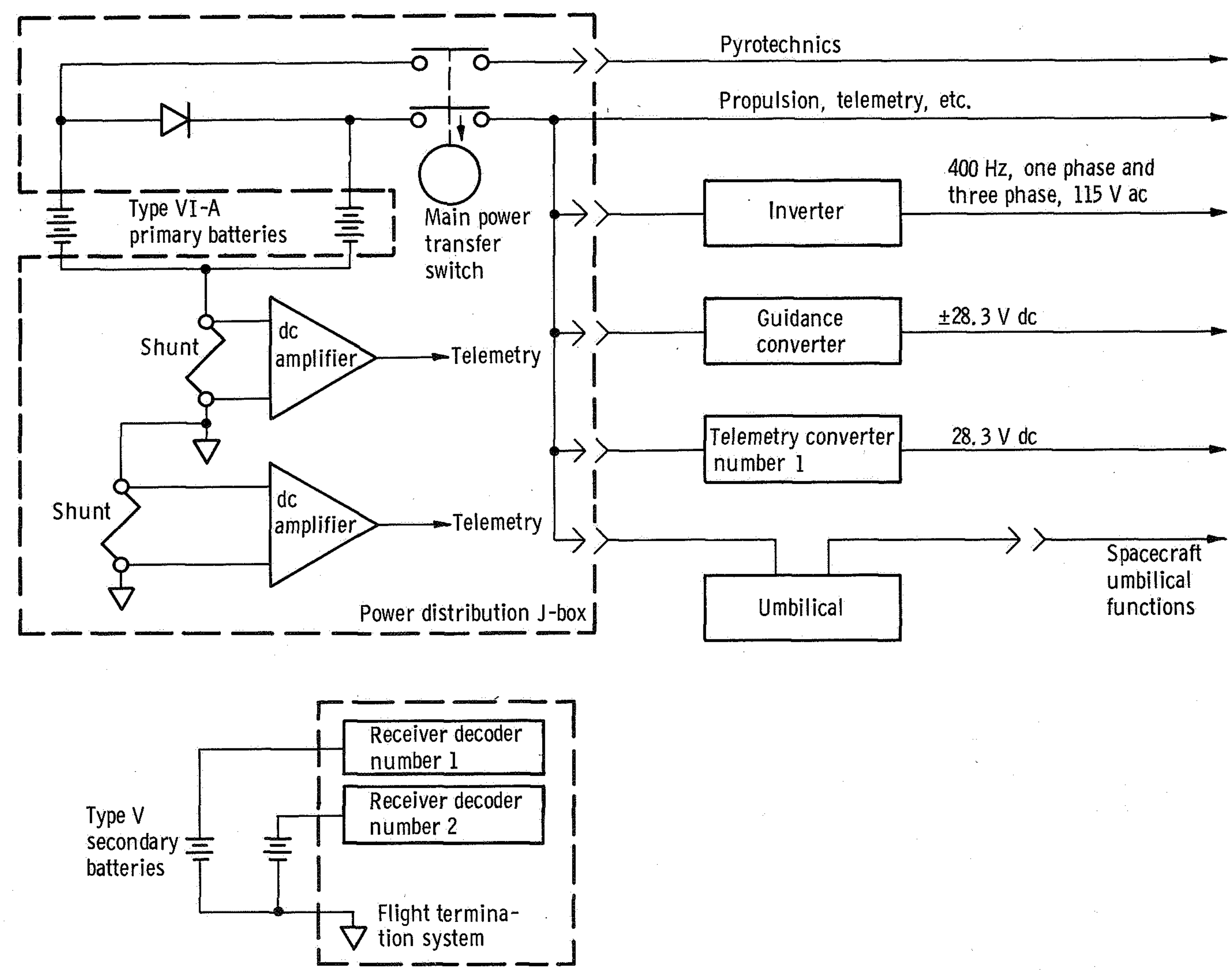


Figure VI-6. - Agena electrical system, ATS-3.

GUIDANCE AND FLIGHT CONTROL SYSTEM

by Howard D. Jackson

Description

The Agena guidance and flight control system (fig. VI-7) 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: guidance, control, and flight programming.

The Agena guidance subsystem consists of an inertial reference package (IRP), horizon sensors, velocity meter, and a guidance junction box. Primary attitude reference is provided by three orthogonal rate-integrating gyroscopes in the inertial reference package. The infrared horizon sensors provide corrections in pitch and roll to the inertial reference package as required. Yaw attitude is referenced to the attitude of the Atlas at the time of Atlas-Agena separation and is then corrected by gyro-compassing techniques during long (greater than 10 min) coast periods. During Agena engine burns, longitudinal acceleration is sensed by the velocity meter accelerometer. The accelerometer produces output pulses that count down a predetermined "velocity-to-be-gained" binary number in the velocity meter counter. A signal is generated to terminate engine thrust when the Agena velocity has increased by a predetermined number. A first-burn velocity-to-be-gained number is loaded into the counting register before lift-off. The second-burn "velocity-to-be-gained" number is loaded into a storage register before lift-off and transferred to the counting register after Agena engine first cutoff.

The Agena flight control subsystem, which controls vehicle attitude, consists of a flight control electronics unit, a cold-gas attitude 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 cold-gas and hydraulic systems. During Agena coast periods, the cold-gas system consisting of six thrusters provides roll, pitch, and yaw control. These thrusters operate on a mixture of nitrogen and tetrafluoromethane gas. During powered flight, the hydraulic system provides pitch and yaw control by means of two hydraulic actuators which gimbal the Agena engine thrust chamber, and the cold-gas system provides roll control. A patch panel in the flight control junction box provides the means for 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 with multiple switch closure capability provides 22 usable, discrete event times and has a maximum running time of 6000 seconds. Two timers (a primary and a restart timer) are required if the Agena mission duration exceeds 6000 seconds or if more than 22 discrete events are required. Since the ATS-3 mission requires more than 22 discrete events, two sequence timers were used. The primary sequence timer is started by a ground initiated Mod III Radio Guidance command. The start time is determined by the ground-based computer after compensating for the trajectory dispersions of the booster. These commands are received by the Atlas, where they are decoded, and then a signal is sent to the Agena. The Agena restart timer is started by the primary timer.

Performance

The guidance and flight control system performance on ATS-3 was satisfactory throughout flight. All flight events were initiated within tolerance by the sequence timers. A comparison of the expected and actual times of programmed events is given in appendix A. The rates imparted to the Agena at Atlas-Agena separation and the attitude errors that existed following separation were within previous flight experience and are shown in the following table:

Rates imparted to Agena at separation, deg/sec			Attitude errors at cold gas activation, deg		
Yaw	Roll	Pitch	Yaw	Roll	Pitch
0.35 left	0.1 CW[a]	0.1 up	1.1 left	0.9 CW[a]	0.1 up

[a]Clockwise (CW) roll reference applies when looking forward along Agena longitudinal axis (see fig. VI-8).

The cold-gas attitude-control system reduced these errors to within the dead band limits of ±0.2° pitch, ±0.18° yaw, and ±0.6° in roll within 14 seconds. The vehicle then completed a programmed pitch down of 9.9° and the programmed geocentric rate of 3.25 degrees per minute pitch down was applied. The vehicle had stabilized in all axes by the time of first ignition. For the Agena first burn, the pitch horizon sensors were set at a pitch bias angle of ±5.34° (nose up).

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

Roll rate, deg/sec	Maximum roll displacement error, deg	Time to reverse initial rate, sec
2.5 CW[a]	1.41 CCW[a] first overshoot	1.7

[a]See fig. VI-8.

The hydraulic and cold-gas attitude-control activity was normal throughout the Agena engine first burn. Shroud jettison (initiated 10 sec after first-burn ignition) occurred during the portion of the 1.41° counterclockwise roll displacement when the vehicle roll rate had reached a minimum (approximately zero). The horizon sensors indicated a slight disturbance approximately 2.0 seconds after the firing of the shroud pyrotechnics; however, gyro data indicated 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.

Engine shutdown was commanded by the velocity meter after the vehicle had attained the required velocity increment. The roll transients, caused by engine shutdown (i.e., turbine spin and turbine exhaust decay), were as experienced on similar previous flights. The maximum time required to reduce the roll excursions to within the attitude control dead bands was 20 seconds. Approximately 23 seconds after Agena engine shutdown, the programmed geocentric pitch rate was increased to 4.47 degrees per minute pitch down, and the horizon sensor bias angle was decreased to zero. Horizon sensor and cold-gas attitude-control data indicated that the vehicle maintained the proper attitude in the coast phase.

Gas generator turbine spin-up at Agena engine second igintion resulted in a roll rate and maximum roll displacement error as follows:

Roll rate, deg/sec	Maximum roll displacement error, deg	Time to reverse initial rate, sec
2.1 CW[a]	4.44 CW[a]	2.2

[a]See fig. VI-8.

The hydraulic and cold-gas attitude-control activity was normal throughout the Agena engine second burn. Engine shutdown was commanded by the velocity meter after the vehicle had attained the required velocity increment. A normal second-burn shutdown transient resulted in a displacement of 1.74° roll counterclockwise. Cold-gas thruster activity removed the roll displacement, and the vehicle assumed a roll limit cycle with a rate of 0.025 hertz and a peak displacement of 1.16°.

The horizon sensors were disconnected from the gyros, and the vehicle pitched up 10.1° and yawed left 52.9° as programmed. The Agena was stable within the dead band limits of the cold-gas attitude-control system at the time of spacecraft separation.

After spacecraft separation, the vehicle performed a yaw-right maneuver of 211.4° to orient the vehicle in the desired nose-aft position. At the completion of this maneuver, the horizon sensors were reconnected to the gyros, and a geocentric pitch rate of 3.62 degrees per second was applied to maintain this attitude.

The Agena attitude-control gas usage was less than anticipated. A comparison of the predicted and actual gas usage is given in table VI-II. These data indicate that no unusual attitude perturbing forces were encountered in flight.

TABLE VI-II. - ATTITUDE CONTROL GAS LOADING REQUIREMENTS AND USAGE, ATS-3

Event	Units	Predicted	Actual
Lift-off	kg	13.5 load	13.3 load
	lb	29.7	29.3
Agena first coast	kg	2.6 usage	.9 used
	lb	5.7	2.0
Agena first burn	kg	1.8 usage	.9 used
	lb	4.0	1.9
Parking orbit coast	kg	.3 usage	.2 used
	lb	.6	.4
Agena second burn	kg	1.1 usage	.5 used
	lb	2.4	1.1
Prespacecraft separation maneuver	kg	.5 usage	.4 used
	lb	1.1	.8
Postspacecraft separation maneuver	kg	.1 usage	(a)
	lb	.3	
Loss of signal	kg	7.1 surplus	[b]10.3 surplus
	lb	15.6	22.8

[a]Not available.

[b]Value based on the assumption that 0.1 kg (0.3 lb) of gas was actually used for the postspacecraft separation maneuver.

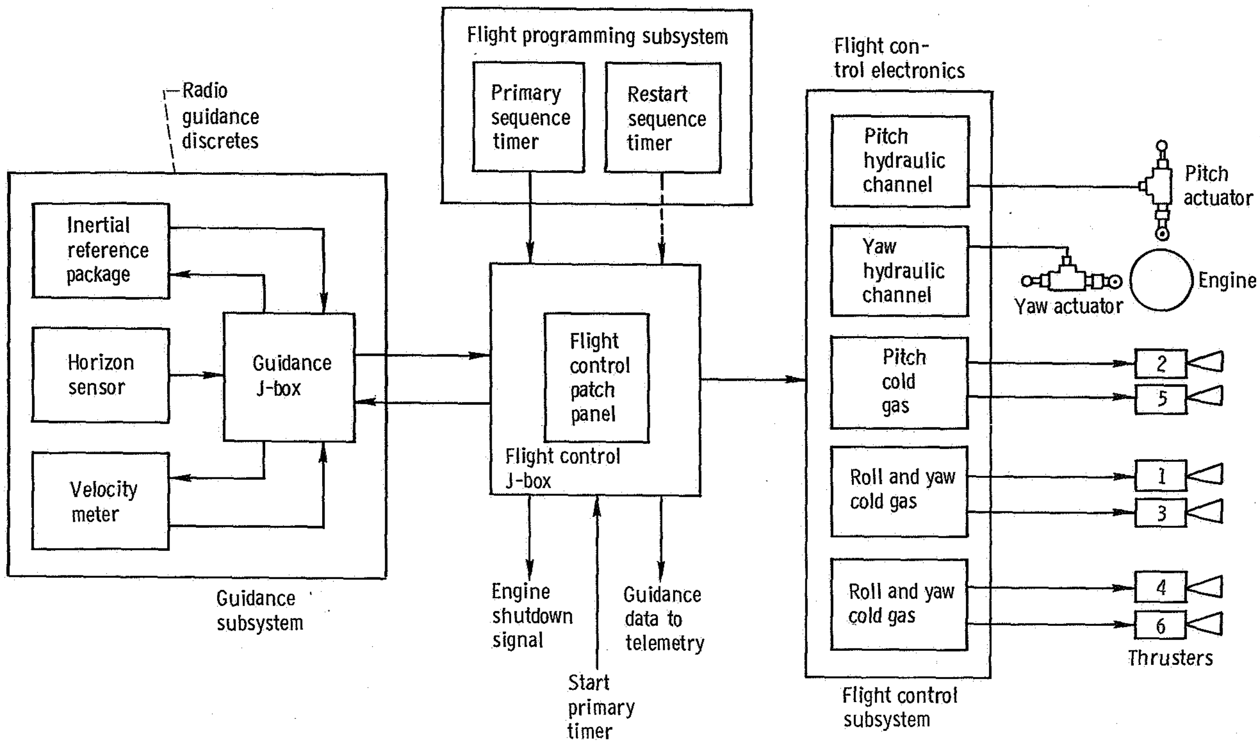


Figure VI-7. - Agena guidance and flight control system block diagram, ATS-3.

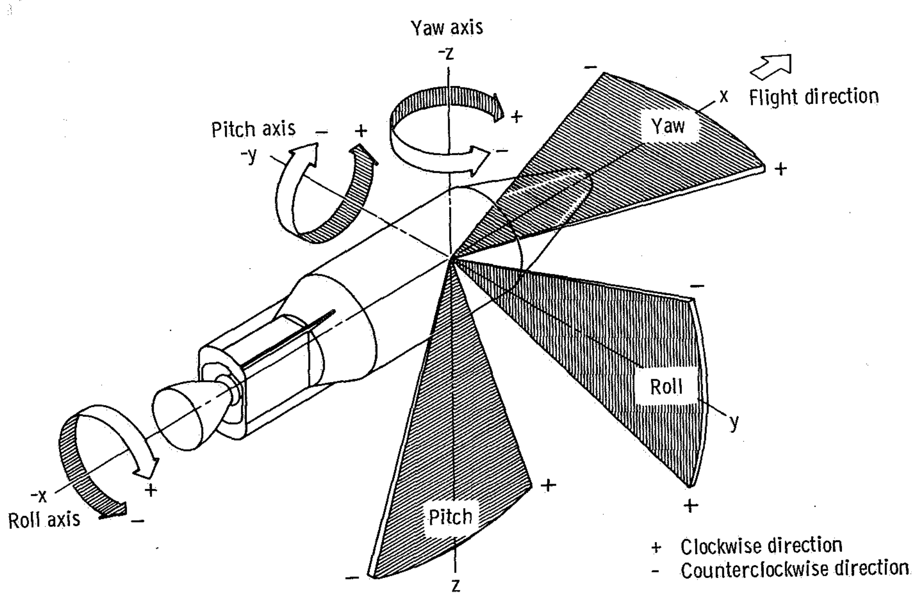


Figure VI-8. - Agena vehicle axes and vehicle movement designations, ATS-3.

COMMUNICATION AND CONTROL SYSTEM

by Richard L. Greene

Description

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

The telemetry subsystem is mounted in the forward section. It monitors and transmits the Agena functional and environmental measurements during ascent. The Frequency Modulation/Frequency Modulation (FM/FM) telemetry unit contains a very high frequency (VHF) transmitter, voltage controlled oscillators, a commutator, a switch and calibrate unit, 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 subsystem consists of seven continuous subcarrier channels and two subcarrier channels commutated at 5 revolutions per second with 60 segments on each channel. The subsystem also includes a radiofrequency switch and an antenna. The radiofrequency switch connects the telemetry output to the umbilical for ground test or to the antenna for flight.

A total of 59 measurements are telemetered from the Agena vehicle. Appendix B summarizes the launch vehicle instrumentation by measurement description. Five continuous channels were used for acceleration and vibration data, one continuous channel monitored the gas valve current signals, and one continuous channel was time shared by the velocity meter accelerometer and velocity meter counter. The turbine speed signal did not use a subcarrier oscillator but directly modulated the transmitter during engine operation. The remaining 50 measurements were monitored on the two commutated subcarrier channels.

The airborne tracking subsystem includes a C-band beacon transponder, a radiofrequency switch, and an 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 upon the rates transmitted from the ground interrogating stations and the number of stations interrogating the vehicle at any one time.

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 batteries, one destruct initiator with two detonators and a shaped charge, and two receiver-decoders which are coupled to two antennas by a multicoupler. With the exception of the multicoupler and destruct change, these units are connected to provide redundant flight termination capability. Flight termination, if necessary, is inititiated by the Range Safety Officer through commands from the range safety transmitter. The destruct charge, near the fuel-oxidizer bulkhead, ruptures both propellant tanks and effects dispersion of the propellants. The flight termination subsystem is disarmed shortly after Agena engine first cutoff.

Telemetry Subsystem Performance

The telemetry subsystem performance was satisfactory throughout the flight. Stations at Cape Kennedy and Antigua provided complete telemetry coverage from lift-off through Agena engine first cutoff. The Agena engine second burn was monitored by the range instrumentation ship, Twin Falls. The preseparation attitude maneuvers, spacecraft separation, and the final Agena yaw maneuver were monitored by a range instrumentation aircraft. Signal strength data from all stations showed an adequate and continuous signal level from the vehicle telemetry transmitter. The telemetry data indicated that the performance of the telemetry system was satisfactory. A description of the tracking and data acquisition network used in support of the ATS-3 flight is shown in appendix C.

Commutated measurement D60, hydraulic oil pressure, failed at Agena first ignition. The transducer operated erratically for the remainder of first burn. The output of the transducer was more stable during second burn but was 50 percent lower than expected. Since steering control was maintained even when the transducer indicated zero hydraulic pressure, it is concluded that the pressure transducer failed.

Continuous measurement A5, a tangential low-frequency accelerometer, exhibited intermittent erratic behavior during ascent. The first obvious anomaly appeared in the transonic region where A5 indicated g-levels several times higher than A4, a second tangential accelerometer. The subsequent events during which the output of A5 appeared erratic include horizon sensor fairing jettison, Atlas-Agena separation, shroud separation, and Agena second-burn ignition. Because of its intermittent erratic behavior, the validity of all dynamic data from A5 are questionable.

Flight data of channels 17 and 18 indicate that the wire harnesses for the longitudinal vibrometer (measurement A520) and the radial vibrometer (measurement A524) were interchanged. Therefore, the flight dynamics data for A520 were transmitted over channel 17 instead of channel 18, and the data for A524 were transmitted over channel 18 instead of channel 17. The flight dynamic data shown in appendix D reflect the "as-flown" configuration.

Tracking Subsystem Performance

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. The radar coverage provided by each supporting station is shown in appendix C.

Flight Termination Subsystem Performance

The flight termination subsystem maintained the capability to destruct the vehicle during the flight. The received signal strength remained essentially constant and well above the minimum required for operation of the flight termination subsystem throughout the period during which destruct capability was required. However, no flight termination signals were required.

VII. LAUNCH OPERATIONS

by Frank E. Gue

PRELAUNCH ACTIVITIES

The Atlas, Agena, and spacecraft arrived at the Eastern Test Range August 31, September 7, and October 3, 1967, respectively. A calendar of the major prelaunch activities is shown in table VII-I.

All launch vehicle prelaunch tests were completed satisfactorily. Only minor operational and ground equipment problems were encountered while conducting these tests. During the Joint Flight Acceptance Tests, the telemetry ground stations noted wavetrain fluctuations on the Agena commutated telemetry channels. These fluctuations resulted from erratic speed changes within the Agena telemetry switch motor module. These speed changes were within specification. During the simulated launch demonstration, a leak was detected in the ground side of the Agena nitrogen quick disconnect coupling. Pressure was vented and a seal was tightened correcting the problem.

COUNTDOWN AND LAUNCH

The launch attempt on November 3 was terminated because the amount of oxidizer loaded on the Agena could not be determined. The problems encountered during the countdown for this launch attempt are as follows:

(1) Shortly after the launcher holddown cylinders were pressurized to a gage pressure of 4140 newtons per square centimeter (6000 psi) for launch, leakage was observed in the quadrant II-III valve main poppet seat. The seat assembly was replaced but the replacement seat assembly also failed during checkout. The pneumatic section of the valve was then replaced; however, the replacement section failed to maintain pressure after being pressurized to 4140 netwons per square centimeter (6000 psi). The pneumatic section was again replaced and, after a successful test, was ready to support the launch.

(2) The Agena telemetry switch motor module did not start when telemetry was turned on and the module was replaced.

(3) During the replacement of the telemetry switch motor module while power was on the Agena, an unexpected velocity meter counter warning was received at the block-

house. It was determined that removal of the telemetry switch motor module connectors resulted in isolating the signal ground from the unregulated ground on the vehicle. This allowed the monitoring voltages from the gas valves to activate the counter warning monitors. Subsequent testing confirmed that there had been no actual velocity meter malfunction.

(4) The Agena oxidizer transfer unit failed to shut off at 100 percent load, resulting in an oxidizer overload in the Agena. Oxidizer was drained back to correct for the overload. However, since it was possible for oxidizer to have entered the tank vent line, the amount of oxidizer in the Agena could not be determined with certainty. Because of this uncertainty in oxidizer load, the launch attempt was terminated.

During the investigation that followed, the twisted ends of an identification tag attachment wire were noted to protrude so as to interefere randomly with the horizontal drive shaft for the over-under indicator of the oxidizer transfer unit. This interference prevented the scale from reaching balance until sufficient force was applied to push the tag attachment wire from the path of the scales. It was concluded that this mechanical interference caused the scales to read incorrectly, resulting in an Agena oxidizer overload condition.

The launch attempt on November 4 was terminated because the Atlas liquid-oxygen tanking could not be completed. The problems encountered during countdown for this launch attempt are as follows:

(1) The fuel pump discharge pressure transducer on the Agena fuel transfer unit failed and was replaced during Agena tanking preparations.

(2) A filter in the Agena oxidizer transfer unit exhibited a high-pressure drop, and the filter was replaced.

(3) At the beginning of Atlas liquid-oxygen tanking, a short circuit occurred in a 440-volt ac cable at the circuit breaker panel in the liquid-oxygen storage area. The circuit breaker, the breaker housing, and associated wiring were damaged by the short. Since Atlas liquid-oxygen tanking could not be completed, the launch attempt was terminated. The circuit breaker was replaced and the control cables were repaired prior to the next launch attempt.

ATS-3 was successfully launched on November 5, 1967. During the November 5 launch countdown, Aerospace Ground Equipment performance was satisfactory and no problems occurred.

TABLE VII-I. - PRELAUNCH ACTIVITIES

COMPLEX 12, ATS-3

Date	Major events
9/8/67	Atlas erection
10/2-5/67	Atlas propellant tanking test number 1
10/11/67	Booster flight acceptance test
10/17/67	Atlas propellant tanking test number 2
10/24/67	Agena mate
10/27/67	Joint flight acceptance test
10/28/67	Spacecraft mate
10/31/67	Simulated launch demonstration
11/3/67	First launch attempt
11/4/67	Second launch attempt
11/5/67	Launch

VIII. CONCLUDING REMARKS

The Atlas-Agena vehicle successfully boosted the 714-kilogram (1574-lb) ATS-3 onto the required transfer orbit. The ATS-3 apogee motor performed as planned placing the spacecraft onto a near-synchronous Earth orbit.

The Standard Agena Clamshell shroud was used on this flight and performed satisfactorily by providing adequate aerodynamic shielding for the spacecraft during ascent through the atmosphere. The Atlas-Agena was launched on the third attempt (previous attempts were made on Nov. 3 and 4). All aerospace ground equipment performed satisfactorily during the launch on November 5, 1967.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, April 14, 1969,
491-05-00-02-22.

APPENDIX A

SEQUENCE OF MAJOR FLIGHT EVENTS

by Richard L. Greene

Nominal time, sec	Actual time,[a] sec	Event description	Source	Event monitor[b]
0	0	Lift-off (1837:00.257 EST)		5.08-cm (2-in.) motion switch
129.5	128.4	Atlas booster engine cutoff	Mod III Radio Guidance	Longitudinal accelerometer (A9)
291.5	289.1	Atlas sustainer engine cutoff		Longitudinal accelerometer (A9)
295.7	300.1	Start Agena primary timer		Guidance and control monitor (D14)
311.4	309.2	Atlas vernier engine cutoff	Agena primary timer	
		Jettison horizon sensor fairings		
		Uncage Agena gyros		
315.5	313.5	Atlas-Agena separation	Mod III Radio Guidance	
318.0	315.6	Activate cold-gas attitude control	Agena separation switch	
341.7	346.1	Initiate -119.1 deg/min pitch rate	Agena primary timer	Pitch torque rate (D73)
346.7	351.1	Transfer to -3.25 deg/min pitch rate		Pitch torque rate (D73)
366.7	371.1	Agena engine first-ignition squibs		Switch group Z (B13)
		Deactivate pitch and yaw cold-gas attitude control		Guidance and control monitor (D14)
367.9	372.3	Agena engine thrust at 90 percent chamber pressure		Chamber pressure (B91)

[a] Event times monitored on D14, D51, and D73 are based on data sampled every 0.2 sec. Event times monitored on B91 are based on data sampled every 0.1 sec. Event times monitored on B13 are based on data sampled every 0.05 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. Designation in parentheses is monitor measurement designation. See LAUNCH VEHICLE INSTRUMENTATION SUMMARY (appendix B) for measurement range and channel assignment.

Nominal time, sec	Actual time,[a] sec	Event description	Source	Event monitor[b]
368.2	372.5	Fire helium pressure squibs	Agena primary timer	(c)
376.7	381.1	Fire shroud separation squibs		Shroud separation monitor (A52)
483.7	488.1	Arm Agena engine shutdown circuit		(c)
528.0	533.6	Agena engine first shutdown	Velocity meter	Chamber pressure (B91)
		Activate pitch and yaw cold-gas attitude control	Velocity meter	Guidance and control monitor (D14)
536.7	541.1	Connect velocity meter counter output to telemetry	Agena primary timer	Velocity meter accelerometer-counter (D83/D88)
		Close propellant isolation valve		Propellant isolation valve monitor (B130)
551.7	556.1	Disarm Agena command destruct		Pitch torque rate (D73)
		Transfer to -4.47 deg/min pitch rate		Pitch torque rate (D73)
556.7	561.1	Transfer to second-burn velocity increment		Velocity meter accelerometer-counter (D83/D88)
758.7	763.1	Fire helium valve squib (close)		(c)
		Start restart timer		(c)
1497.7	1502.1	Enable velocity meter		Velocity meter accelerometer-counter (D83/D88)
		Open propellant isolation valves		Propellant isolation valve monitor (B130)
1499.7	1504.1	Deactivate pitch and yaw cold-gas system		Guidance and control monitor (D14)
		Fire Agena second-ignition squibs		Switch group Z (B13)

[a]Event times monitored on D14, D51, and D73 are based on data sampled every 0.2 sec. Event times monitored on B91 are based on data sampled every 0.1 sec. Event times monitored on B13 are based on data sampled every 0.05 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. Designation in parentheses is monitor measurement designation. See LAUNCH VEHICLE INSTRUMENTATION SUMMARY (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]
1500.9	1505.2	Agena engine thrust at 90 percent chamber pressure	Agena primary timer	Chamber pressure (B91)
1574.7	1579.0	Arm engine shutdown	Agena primary timer	(c)
1579.0	1583.5	Agena engine second shutdown	Velocity meter	Chamber pressure (B91)
		Pitch and yaw cold-gas system on	Velocity meter	Guidance and control monitor (D14)
1582.7	1587.0	Connect velocity meter counter output to telemetry	Agena primary timer	Velocity meter accelerometer-counter (D83/D88)
1666.7	1671.1	Transfer to 27.54 deg/min pitch rate	Agena auxiliary timer	Pitch torque rate (D73)
1688.7	1693.1	Stop 27.54 deg/min pitch rate		Pitch torque rate (D73)
1705.7	1710.0	Initiate -176.18 deg/min yaw rate		Yaw torque rate (D51)
1723.7	1728.0	Stop -176.18 deg/min yaw rate		Yaw torque rate (D51)
1743.7	1748.1	Fire spacecraft separation squibs	Agena primary timer	Longitudinal vibration (A520)
1746.7	1751.0	Initiate 176.18 deg/min yaw rate		Yaw torque rate (D51)
1818.7	1823.0	Stop 176.18 deg/min yaw rate		Yaw torque rate (D51)
1818.7	1823.0	Initiate 3.62 deg/min pitch rate	Agena auxiliary timer	Pitch torque rate (D73)
2118.7	2123.1	Remove all power from vehicle except C-band	Agena auxiliary timer	Telemetry signal strength

[a]Event times monitored on D14, D51, and D73 are based on data sampled every 0.2 sec. Event times monitored on B91 are based on data sampled every 0.1 sec. Event times monitored on B13 are based on data sampled every 0.05 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. Designation in parentheses is monitor measurement designation. See LAUNCH VEHICLE INSTRUMENTATION SUMMARY (appendix B) for measurement range and channel assignment.

[c]No direct measurement for these events.

APPENDIX B

LAUNCH VEHICLE INSTRUMENTATION SUMMARY

by Edwin S. Jeris and Richard L. Greene

TABLE B-I. - ATLAS TELEMETRY, ATS-3

Measurement number	Description	Channel[a] assignment	Measurement range (low to high)	
			SI Units	U.S. Customary Units
A743T	Ambient temperature at sustainer instrumentation panel	11-41	227.5 to 561 K	-50° to 550° F
A745T	Ambient temperature at sustainer fuel pump	11-45	227.5 to 561 K	-50° to 550° F
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	(b)	
E28V	Main dc voltage	18-1/31	20 to 35 V dc	
E51V	400 Hz, ac, phase A	18-11	105 to 125 V dc	
E52V	400 Hz, ac, phase B	18-29	105 to 125 V ac	
E53V	400 Hz, ac, phase C	18-41	105 to 125 V ac	
E95V	28 V dc guidance power input	13-15	20 to 35 V dc	
E96V	115 V ac, 400 Hz, phase A to guidance	13-37	105 to 125 V ac	
E151V	400 Hz, phase A waveform	10	0 to 150 V ac	
F1P	Liquid-oxygen-tank helium pressure (absolute)	15-9	0 to 34.5 N/cm^2	0 to 50 psi
F3P	Fuel-tank helium pressure (absolute)	15-11	0 to 68.9 N/cm^2	0 to 100 psi
F116P	Differential pressure across bulkhead	18-13/43	0 to 17.2 N/cm^2	0 to 25 psi
F125P	Booster control pneumatic regulator output pressure (absolute)	13-21	0 to 689 N/cm^2	0 to 1000 psi

[a]First number indicates the Inter-Range Instrumentation Group (IRIG) subcarrier channel used; second number indicates the commutated position for the measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

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

TABLE B-I. - Continued. ATLAS TELEMETRY, ATS-3

Measurement number	Description	Channel[a] assignment	Measurement range (low to high)	
			SI Units	U.S. Customary Units
F246P	Booster tank helium bottle pressure (absolute)	13-55	0 to 2413 N/cm^2	0 to 3500 psi
F288P	Start pneumatic regulator output	13-1	0 to 551.5 N/cm^2	0 to 800 psi
F291P	Sustainer control helium bottle	13-3	0 to 2413 N/cm^2	0 to 3500 psi
F247T	Booster tank helium bottle temperature	11-31	33.5 to 116.5 K	-400° to -250° F
G4C	Pulse beacon magnetron average current	15-15	0 to 5 V dc	
G82E	Rate beacon radiofrequency output	15-17	0 to 5 V dc	
G3V	Pulse beacon automatic gain control	15-19	0 to 5 V dc	
G279V	Rate beacon automatic gain control number 1	15-21	0 to 5 V dc	
G280V	Rate beacon automatic gain 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 2413 N/cm^2	0 to 3500 psi
H33P	B1 hydraulic accumulator pressure (absolute)	15-31	0 to 2413 N/cm^2	0 to 3500 psi
H130P	Sustainer hydraulic pump discharge pressure (absolute)	15-33	0 to 2413 N/cm^2	0 to 3500 psi
H140P	Sustainer-venier hydraulic pressure (absolute)	15-35	0 to 2413 N/cm^2	0 to 3500 psi
H224P	Booster hydraulic system low pressure (absolute)	15-7	0 to 413.6 N/cm^2	0 to 600 psi
H601P	Sustainer hydraulic return line pressure (absolute)	18-7/37	0 to 413.6 N/cm^2	0 to 600 psi

[a]First number indicates the Inter-Range Instrumentation Group (IRIG) subcarrier channel used; second number indicates the commutated position for the measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

TABLE B-I. - Continued. ATLAS TELEMETRY, ATS-3

Measurement number	Description	Channel[a] assignment	Measurement range (low to high)	
			SI Units	U. S. Customary Units
M79A	Vehicle axial accelerometer fine	7	-0.5 to 0.5 g's	
M30X	Vehicle 5.08-cm (2-in.) motion	7-S	(b)	
M32X	Jettison system command	5-S	(b)	
P83B	Booster 2 pump speed	15-41	3974 to 7005 rpm	
P84B	Booster 1 pump speed	4	5976 to 6946 rpm	
P349B	Sustainer pump speed	3	9350 to 10 870 rpm	
P529D	Sustainer main liquid-oxygen valve	13-43	0° to 90°	
P330D	Sustainer fuel valve position	13-35	21.3° to 52.6°	
P1P	Booster 1 liquid-oxygen-pump inlet pressure (absolute)	18-9	0 to 103.4 N/cm^2	0 to 150 psi
P2P	Booster 1 fuel-pump-inlet pressure (absolute)	13-31	0 to 68.9 N/cm^2	0 to 100 psi
P6P	Sustainer thrust chamber pressure (absolute)	18-3/33	0 to 689 N/cm^2	0 to 1000 psi
P26P	Booster liquid-oxygen regulator reference pressure (absolute)	13-17	344.7 to 689 N/cm^2	500 to 1000 psi
P27P	Vernier fuel tank pressure (absolute)	13-39	0 to 689 N/cm^2	0 to 1000 psi
P28P	Vernier 1 thrust chamber pressure (absolute)	18-15	0 to 275.8 N/cm^2	0 to 400 psi
P29P	Vernier 2 thrust chamber pressure (absolute)	18-17	0 to 275.8 N/cm^2	0 to 400 psi
P30P	Vernier liquid-oxygen-tank pressure (absolute)	13-53	0 to 689 N/cm^2	0 to 1000 psi
P47P	Vernier 1 liquid-oxygen-inlet pressure (absolute)	13-45	0 to 413.7 N/cm^2	0 to 600 psi
P49P	Vernier 1 fuel-inlet pressure (absolute)	13-49	0 to 413.7 N/cm^2	0 to 600 psi
P55P	Sustainer fuel-pump-inlet pressure (absolute)	13-5	0 to 68.9 N/cm^2	0 to 100 psi
P56P	Sustainer liquid-oxygen pump-inlet pressure (absolute)	18-5	0 to 103.4 N/cm^2	0 to 150 psi
P59P	Booster 2 thrust chamber pressure (absolute)	18-19	0 to 511.6 N/cm^2	0 to 800 psi
P60P	Booster 1 thrust chamber pressure (absolute)	18-21	0 to 551.6 N/cm^2	0 to 800 psi
P100P	Gas generator combustion chamber pressure (absolute)	15-51	0 to 413.7 N/cm^2	0 to 600 psi
P330P	Sustainer fuel pump discharge pressure (absolute)	15-55	0 to 1034.2 N/cm^2	0 to 1500 psi
P339P	Sustainer gas generator discharge pressure (absolute)	18-55	0 to 551.6 N/cm^2	0 to 800 psi

[a]First number indicates the Inter-Range Instrumentation Group (IRIG) subcarrier channel used; second number indicates the commutated position for the measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

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

TABLE B-I. - Continued. ATLAS TELEMETRY, ATS-3

Measurement number	Description	Channel[a] assignment	Measurement range (low to high)	
			SI Units	U.S. Customary Units
P334P	Sustainer liquid-oxygen regulator reference pressure (absolute)	13-19	344.7 to 689 N/cm^2	500 to 1000 psi
P15T	Engine compartment air temperature	11-35	227.5 to 561 K	-50° to 550° F
P16T	Engine compartment component temperature	11-55	255.5 to 477.5 K	0° to 400° F
P117T	Booster 2 fuel pump inlet	11-53	255.5 to 311 K	0° to 100° F
P530T	Sustainer liquid-oxygen pump-inlet temperature	11-1	89 to 105.5 K	-300° to -270° F
P671T	Thrust section ambient temperature quadrant IV	11-15	227.5 to 561 K	-50° to 550° F
P77X	Vernier cutoff relay	8-S	(b)	
P347X	System cutoff relay	8-S	(b)	
P616X	Booster flight lock-in relay	16-19	(b)	
S61D	Roll displacement gyro signal	15-29	-3° to 3°	
S62D	Pitch displacement gyro signal	15-37	-3.5° to 3.5°	
S63D	Yaw displacement gyro signal	15-39	-4° to 4°	
S252D	Booster 1 yaw roll	16-15	-6° to 6°	
S253D	Booster 2 yaw roll	16-55	-6° to 6°	
S254D	Booster 1 pitch	7	-6° to 6°	
S255D	Booster 2 pitch	16-1	-6° to 6°	
S256D	Sustainer yaw	16-41	-4° to 4°	
S257D	Sustainer pitch	16-45	-4° to 4°	
S258D	Vernier 1 pitch roll	16-3	-70° to 70°	
S259D	Vernier 2 pitch roll	16-5	-70° to 70°	
S260D	Vernier 1 yaw	16-7	-55° to 5°	
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)	

[a]First number indicates the Inter-Range Instrumentation Group (IRIG) subcarrier channel used; second number indicates the commutated position for the measurement. If no second number is indicated, the channel is used continuously for the designated transducer.
[b]Items are determined from a step change in voltage

TABLE B-I. - Concluded. ATLAS TELEMETRY, ATS-3

Measurement number	Description	Channel[a] assignment	Measurement range (low to high)	
			SI Units	U. S. Customary Units
S290X	Programmer output	16-29	0 to 28 V dc	
	Spare			
	Booster jettison			
	Enable discretes			
S291X	Programmer output	16-31	0 to 28 V dc	
	Booster engine cutoff			
	Sustainer engine cutoff			
	Vernier engine cutoff			
S359X	Booster staging backup	5-S	(b)	
S384X	Spin motor test output	15-5	0 to 5 V dc	
U101A	Axial acceleration	12	0 to 10 g's	
U80P	Liquid-oxygen-tank head pressure (differential)	16-11	0 to 3.4 N/cm^2	0 to 5 psi
U81P	Fuel-tank head pressure (differential)	16-13	0 to 1.7 N/cm^2	0 to 2.5 psi
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	
U44P	Interstage adapter pressure (absolute)	13-23	0 to 10.3 N/cm^2	0 to 15 psi
Y45T	Interstage adapter temperature	11-5	144 to 366.5 K	-200° to 200° F
Y41X	Start Agena primary timer	5-S	(b)	

[a]First number indicates the Inter-Range Instrumentation Group (IRIG) subcarrier channel used; second number indicates the commutated position for the measurement. If no second number is indicated, the channel is used continuously for the designated transducer.
[b]Items are determined from a step change in voltage.

TABLE B-II. - AGENA TELEMETRY, ATS-3

Measurement number	Description	Channel[a] assignment	Measurement range (low to high)	
			SI Units	U. S. Customary 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	273 to 533 K	32° to 500° F
A227	Shroud inside temperature	16-33	273 to 533 K	32° to 500° F
A519	Shroud cavity pressure (differential)	16-12/23/34/53	-3.4 to 3.4 N/cm^2	-5 to 5 psi
A520	Spacecraft adapter longitudinal vibration	17	-20 to 20 g's	
A524	Spacecraft adapter radial vibration	18	-20 to 20 g's	
B1	Fuel pump inlet pressure (gage)	15-15	0 to 69 N/cm^2	0 to 100 psi
B2	Oxidizer pump inlet pressure (gage)	15-17	0 to 69 N/cm^2	0 to 100 psi
B11	Oxidizer venturi inlet pressure (absolute)	15-19/49	0 to 1034 N/cm^2	0 to 1500 psi
B12	Fuel venturi inlet pressure (absolute)	15-23/53	0 to 1034 N/cm^2	0 to 1500 psi
B13	Switch group Z	15-7/22/37/52	(b)	
B31	Fuel pump inlet temperature	15-6	255 to 311 K	0 to 100° F
B32	Oxidizer pump inlet temperature	15-8	255 to 311 K	0 to 100° F
B35	Turbine speed	(c)		
B91	Combustion chamber pressure (gage)	15-4/34	328 to 379 N/cm^2	475 to 550 psi
B130	Propellant isolation valve monitor	15-41	(b)	
C1	28 V dc unregulated supply	16-40	22 to 30 V dc	
C3	28 V dc regulated (guidance and control) number 1	15-12	22 to 30 V dc	
C4	28 V dc unregulated current	16-13/44	0 to 100 A	
C5	-28 V dc regulated (guidance and control) number 1	15-30	-30 to -22 V dc	

[a]First number indicates the Inter-Range Instrumentation Group (IRIG) subcarrier channel used; second number indicates the commutated position for the measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

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

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

TABLE B-II. - Continued. AGENA TELEMETRY, ATS-3

Measurement number	Description	Channel[a] assignment	Measurement range (low to high)	
			SI Units	U.S. Customary Units
C21	400-Hz, three-phase inverter temperature	15-14	255 to 367 K	0° to 200° F
C31	400-Hz, three-phase bus phase AB	15-18	90 to 130 V ac	
C32	400-Hz, three-phase bus phase BC	15-20	90 to 130 V ac	
C38	Structure current monitor	15-10/25/40/55	0 to 50 A	
C141	Pyrotechnic bus voltage	15-5/35	22 to 30 V dc	
D14	Guidance and control monitor	16-27	(b)	
D41	Horizon sensor pitch	16-45	-5° to 5°	
D42	Horizon sensor roll	16-46	-5° to 5°	
D46	Gas valve temperature cluster 1	15-39	228 to 339 K	-50° to 150° F
D47	Gas valve temperature cluster 2	15-36	228 to 339 K	-50° to 150° F
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	228 to 367 K	-50° to 200° F
D55	Horizon sensor head temperature (left head)	15-46	228 to 367 K	-50° to 200° F
D59	Control gas supply high pressure (absolute)	16-47	0 to 2758 N/cm^2	0 to 4000 psi
D60	Hydraulic oil pressure (absolute)	15-21	0 to 2758 N/cm^2	0 to 4000 psi
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°	
D69	Yaw actuator position	15-24	-2.5° to 2.5°	
D70	Control gas supply temperature	15-42	228 to 367 K	-50° to 200° F
D72	Pitch gyro output (ascent mode)	16-36	-10° to 10°	
D72	Pitch gyro output (orbital mode)	16-36	-5° to 5°	
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	

[a]First number indicates the Inter-Range Instrumentation Group (IRIG) subcarrier channel used; second number indicates the commutated position for the measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

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

TABLE B-II. - Concluded. AGENA TELEMETRY, ATS-3

Measurement number	Description	Channel[a] assignment	Measurement range (low to high)	
			SI Units	U. S. Customary Units
D74	Yaw gyro output (ascent mode)	16-39	-10° to 10°	
D74	Yaw gyro output (orbital mode)	16-39	-5° to 5°	
D75	Roll gyro output (ascent mode)	16-42	-10° to 10°	
D75	Roll gyro output (orbital mode)	16-42	-5° to 5°	
D83	Velocity meter acceleration	14	0 to 2000 pulses/sec	
D86	Velocity meter cutoff switch	16-28	(b)	
D88	Velocity meter counter	14	Binary code (50 bits/sec)	
D129	Inertial reference package internal case temperature	15-54	255 to 342 K	0° to 170° F
D149	Gas valve 1 to 6, current moniter	7	(d)	
H47	Beacon receiver pulse repetition rate	15-27	0 to 1600 pulses/sec	
H48	Beacon transmitter pulse repetition rate	15-28	0 to 1600 pulses/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 2	15-50	22 to 30 V dc	
H218	Telemetry transmitter temperature	16-49	283 to 350 K	50° to 170° F
H354	Destruct receiver number 1 signal level	16-6	0 to 40 μV	
H364	Destruct receiver number 2 signal level	16-8	0 to 40 μV	

[a]First number indicates the Inter-Range Instrumentation Group (IRIG) subcarrier channel used; second number indicates the commutated position for the measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

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

[d]A unique voltage level is associated with any one or a combination of several gas thrusters firing.

APPENDIX C

TRACKING AND DATA ACQUISITION

by Richard L. Greene

The launch vehicle trajectory, as projected on a world map, is shown in figure C-1. Eastern Test Range (ETR) and Manned Space Flight Network (MSFN) 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, and Antigua, and the MSFN station at Bermuda. For Agena engine second burn and the prespacecraft separation maneuver, data coverage was provided by the instrumentation ship Twin Falls. (These events occurred beyond the view of any land based stations.) Data coverage during Agena-spacecraft separation was provided by an instrumentation aircraft over Angola, Africa. The instrumentation aircraft was used because the Twin Falls' location could not be optimized since the ship had to reposition to support another launch within a few days.

The ETR station at Pretoria, South Africa, and the MSFN station at Tananarive, Malagasy Republic, provided data coverage after Agena-spacecraft separation. These data were used primarily for the determination of the final Agena orbit.

Telemetry Data

Telemetry data from the Atlas-Agena launch vehicle were recorded on magnetic tape by telemetry stations during all engine operations, Agena pitch and yaw maneuvers, and spacecraft separation. These recorded data were used for postflight analysis of the launch vehicle performance.

Real-time monitoring of specific Atlas and Agena parameters was provided for verification of significant flight events. A submarine cable linking several ETR stations permitted real-time monitoring at Cape Kennedy of vehicle telemetered signals through Agena engine first cutoff. The subsequent flight events were monitored by the remote ETR and MSFN stations and the time of occurrence reported back to Cape Kennedy in "near" real time by single side band radio links.

Figure C-2 shows the specific telemetry coverage provided by each ETR and MSFN tracking station.

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 used 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, parking orbit parameters, injection conditions at Agena engine second cutoff, and transfer orbit parameters. The radar coverage provided by the ETR and MSFN tracking stations is presented in figure C-3.

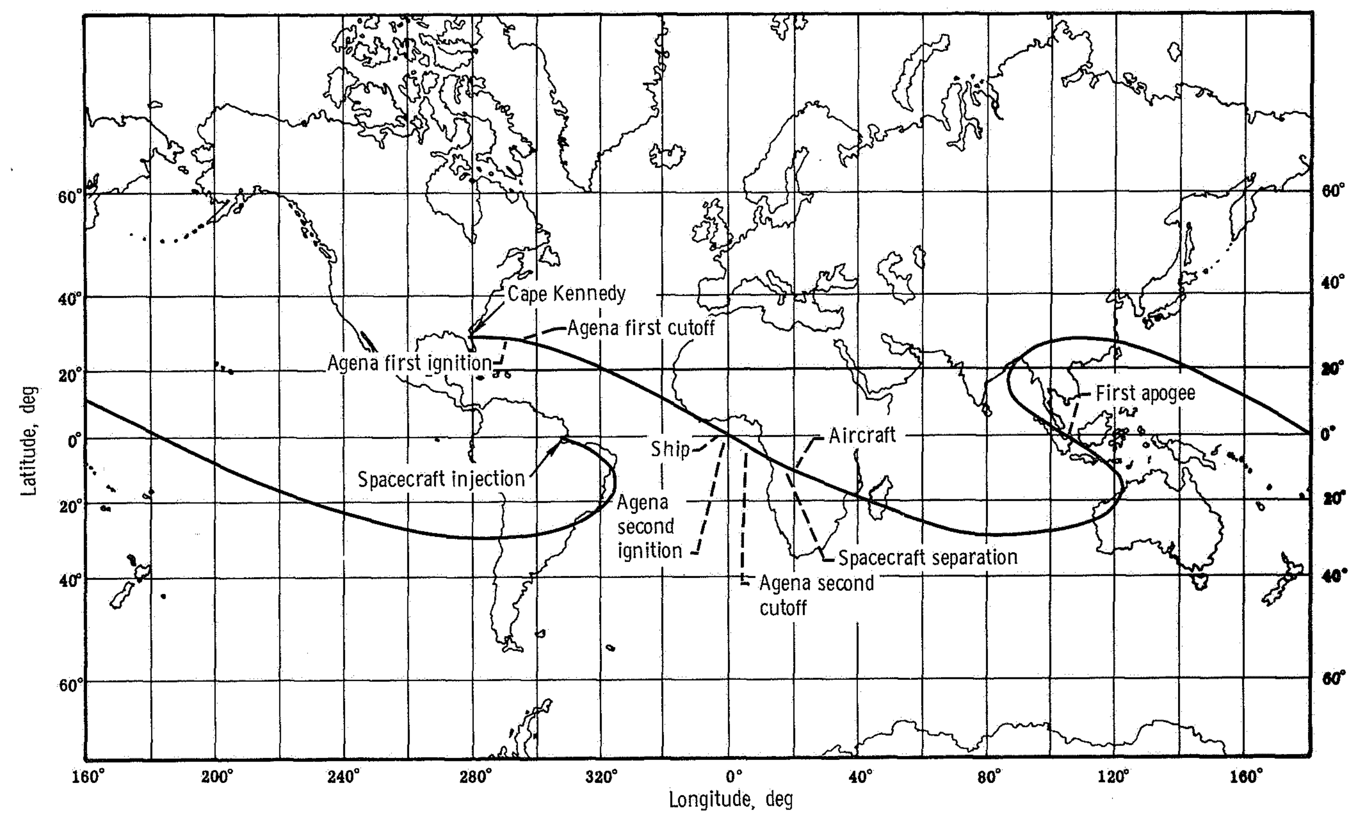


Figure C-1. - Ground trace, ATS-3.

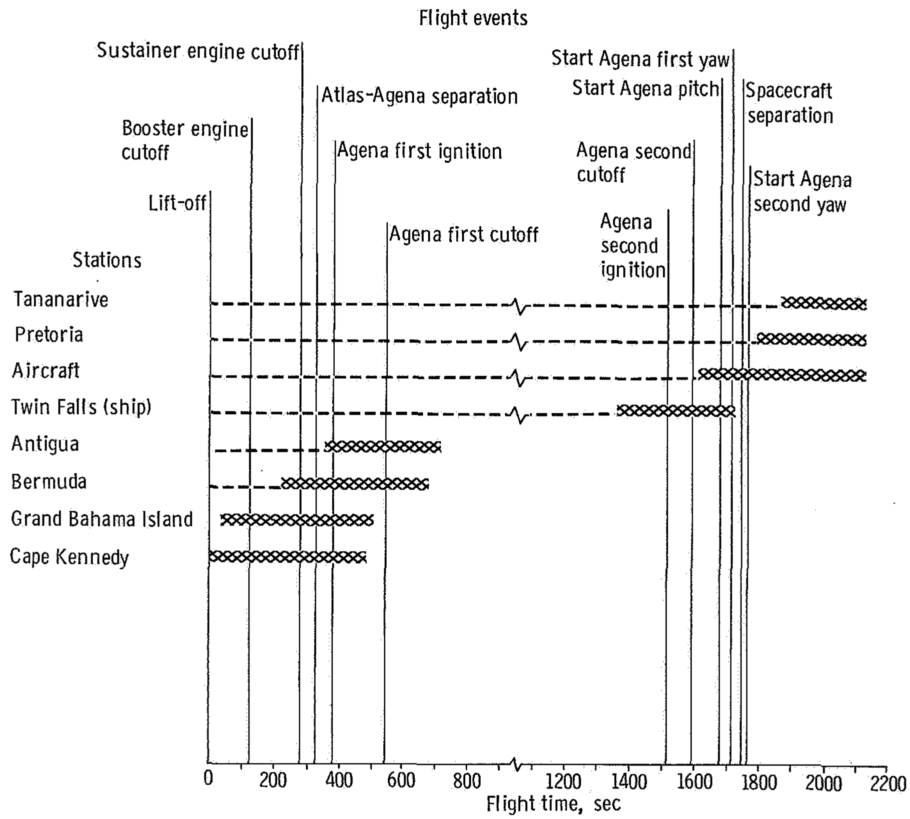


Figure C-2. - Launch vehicle telemetry coverage, ATS-3.

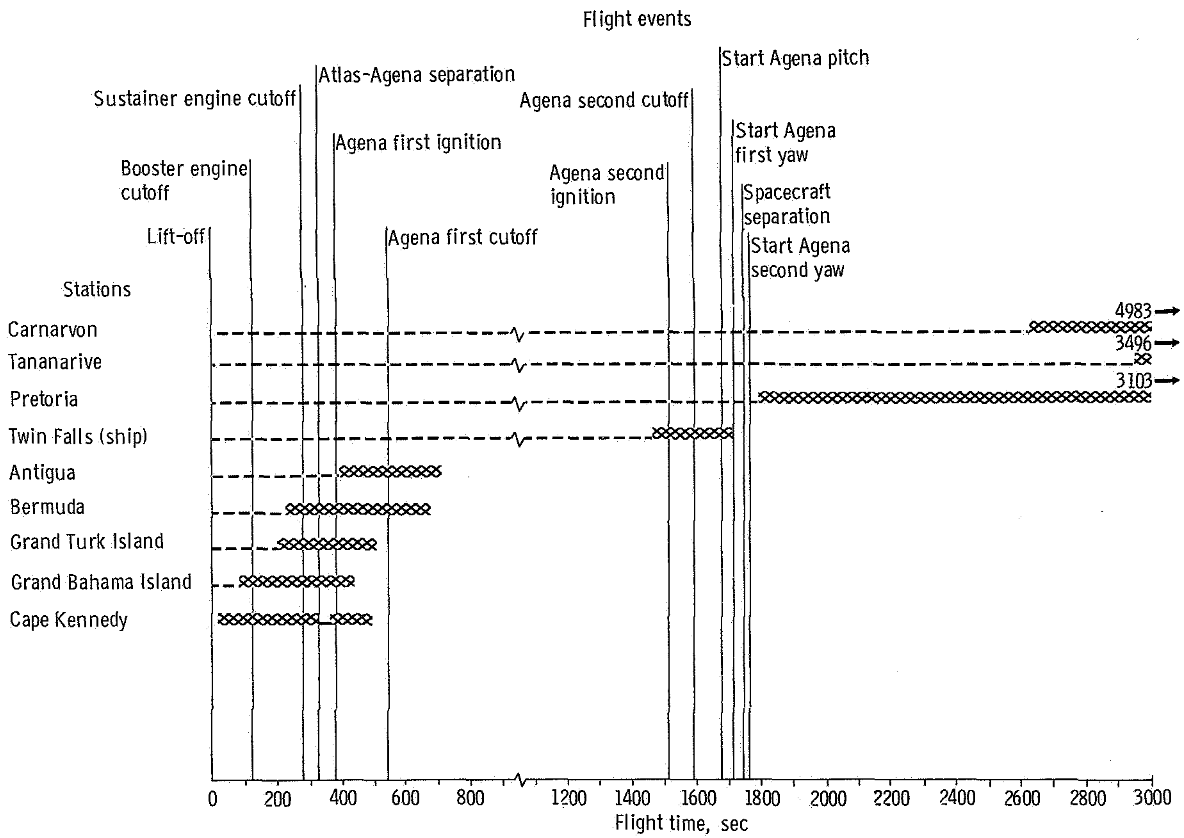


Figure C-3. - Launch vehicle radar coverage, ATS-3.

APPENDIX D

VEHICLE FLIGHT DYNAMICS

by Robert W. York

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

All instruments performed satisfactorily with the exception of accelerometer A5 on channel 11. The data received from this accelerometer during significant dynamic events were unreliable.

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

TABLE D-I. - SUMMARY OF DYNAMIC DISTURBANCES, ATS-3

Event causing disturbance	Time of dynamic disturbance, sec
Lift-off	0
Transonic region	T + 54 to T + 74
Booster engine cutoff	T + 128. 67
Sustainer engine cutoff	T + 289. 19
Horizon sensor fairing jettison	T + 309. 18
Atlas-Agena separation	T + 313. 35
Agena engine first ignition	T + 372. 27
Shroud separation	T + 381. 10
Agena engine first cutoff	T + 533. 59
Agena engine second ignition	T + 1505. 21
Agena engine second cutoff	T + 1583. 51

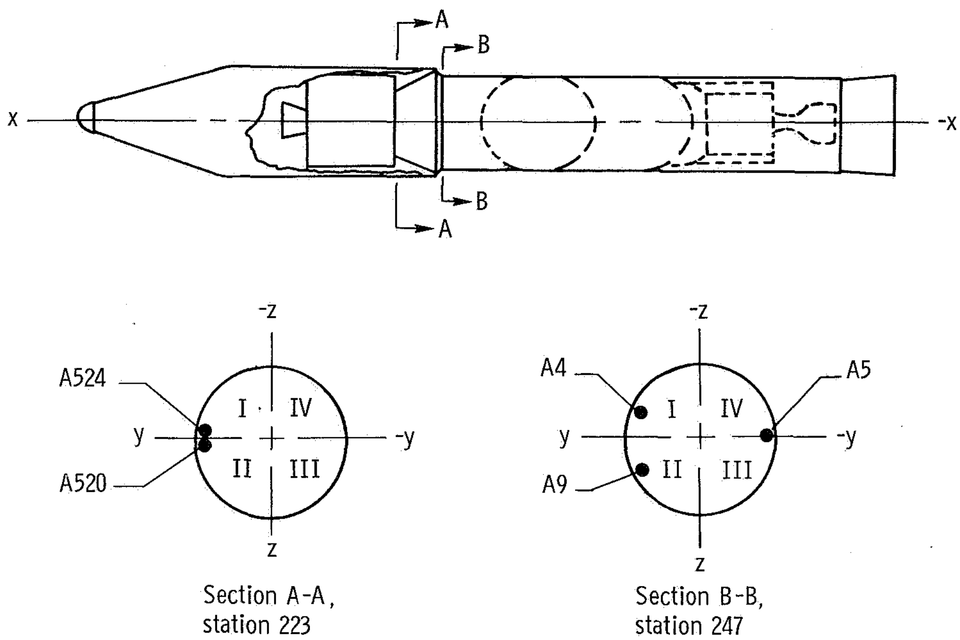
Table D-II shows the maximum acceleration levels and corresponding frequencies recorded at times of significant dynamic disturbances during flight. All acceleration levels are shown in g's zero-to-peak (0-P).

Dynamic environment data recorded for the preceding flight times are presented in figures D-2 to D-12.

TABLE D-II. - SUMMARY OF DYNAMIC ENVIRONMENT, ATS-3

Event causing disturbance	Time of dynamic disturbance, sec	Accelerometer						Vibrometer[a]			
		Channel 8		Channel 9		Channel 11		Channel 17		Channel 18	
		Measurement									
		A9, x-axis		A4, z-axis		A5, z-axis		A520, x-axis		A524, y-axis	
		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	0	Pulse[b] 5	[b]7.2 .4	Pulse	2.6	220	0.3	5	1.0	900	11.5
Transonic	T + 54 to T + 74	60	.3	180	.1	(c)	(c)	800	7.5	800	12.0
Booster engine cutoff	T + 128.67	15	.7	66	1.2	66	1.4	52	1.0	70	.5
Sustainer engine cutoff	T + 289.19	90	.2	100	.1	130	.2	60	.7	120	.5
Horizon sensor fairing jettison	T + 309.18	Pulse	7.8	95	.7	(c)	(c)	800	20	800	Over 20
Atlas-Agena separation	T + 313.35	Pulse	1.8	Pulse	.6	(c)	(c)	850	17	900	Over 20
Agena engine first ignition	T + 372.27	43	.3	56	.2	90	0.4	50	1.5	52	.4
Shroud separation	T + 381.10	Pulse	9.5	Pulse	3.4	(c)	(c)	950	Over 20	850	Over 20
Agena engine first cutoff	T + 533.59	80	.2	60	.4	40	0.4	50	2.0	100	.5
Agena engine second ignition	T + 1505.21	120	.3	80	.5	(c)	(c)	52	2.5	50	.5
Agena engine second cutoff	T + 1583.51	68	.6	50	.4	64	0.5	56	3.0	78	.5

[a]These spacecraft mounted transducers are the responsibility of Goddard Space Flight Center.
[b]Double entries indicate the frequencies and acceleration levels of two superimposed vibrations.
[c]Accelerometer output erratic during these events (see figs. D-3, D-6, D-7, D-9, and D-11).



Measurement number	Channel number	Measurement	Station	Frequency response, Hz	Range, g's
A9	8	x-axis, acceleration	247	0 to 35	-4 to 12
A4	9	z-axis, acceleration	247	0 to 110	±10
A5	11	z-axis, acceleration	247	0 to 160	±10
A520	17	x-axis, vibration	223	20 to 2000	±20
A524	18	y-axis, vibration	223	20 to 1500	±20

Figure D-1. - Flight instrumentation, ATS-3.

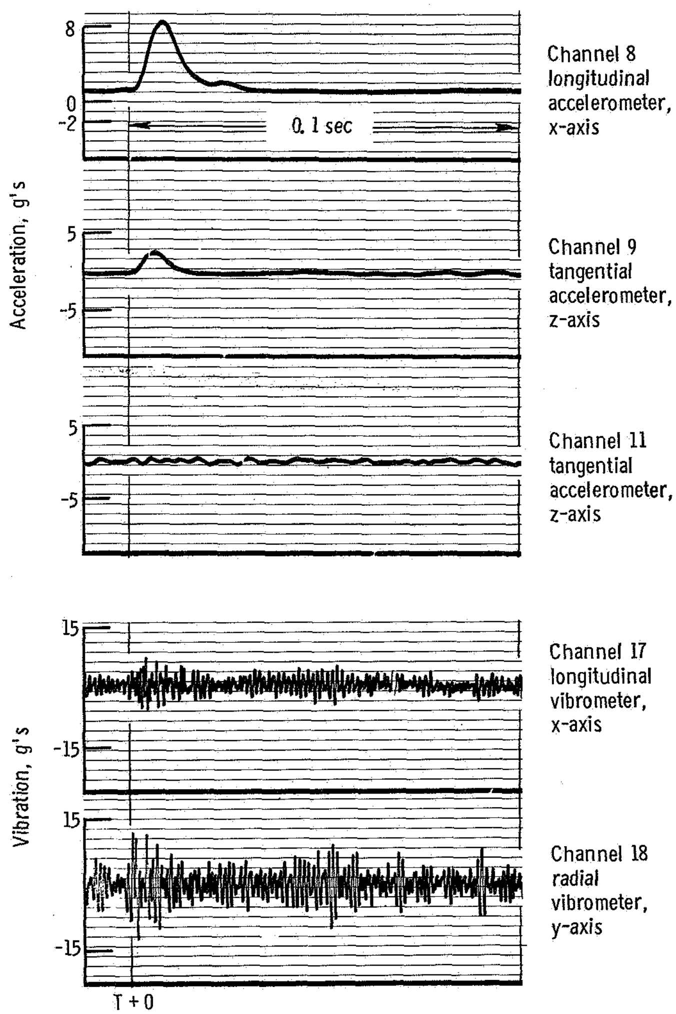


Figure D-2. - Dynamic data at lift-off, ATS-3.

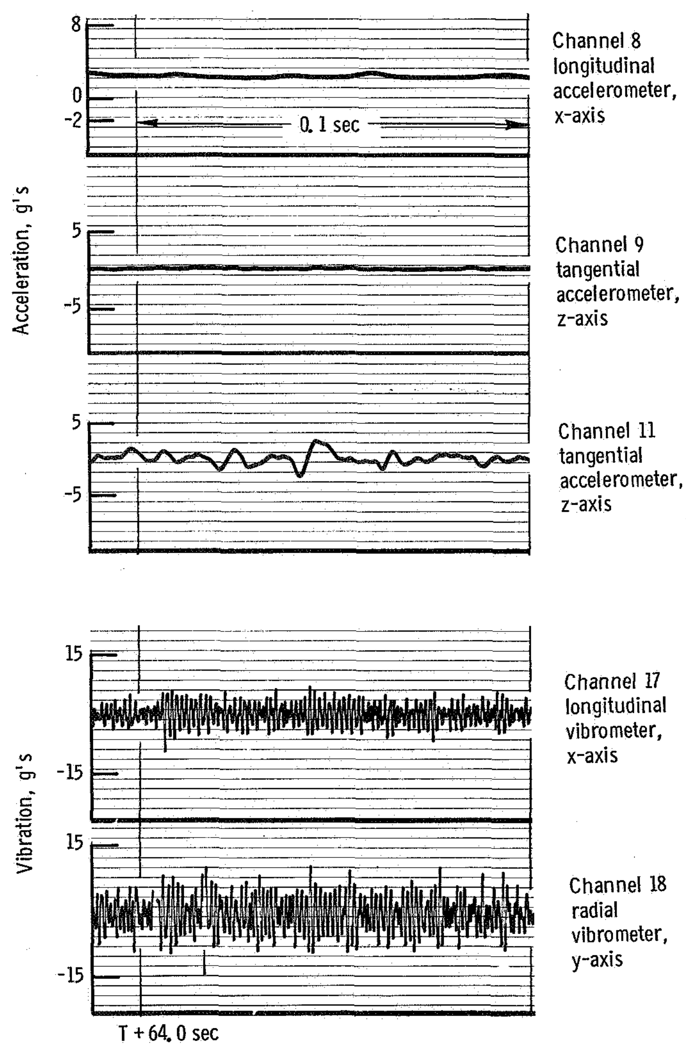


Figure D-3. - Dynamic data during transonic period, ATS-3.

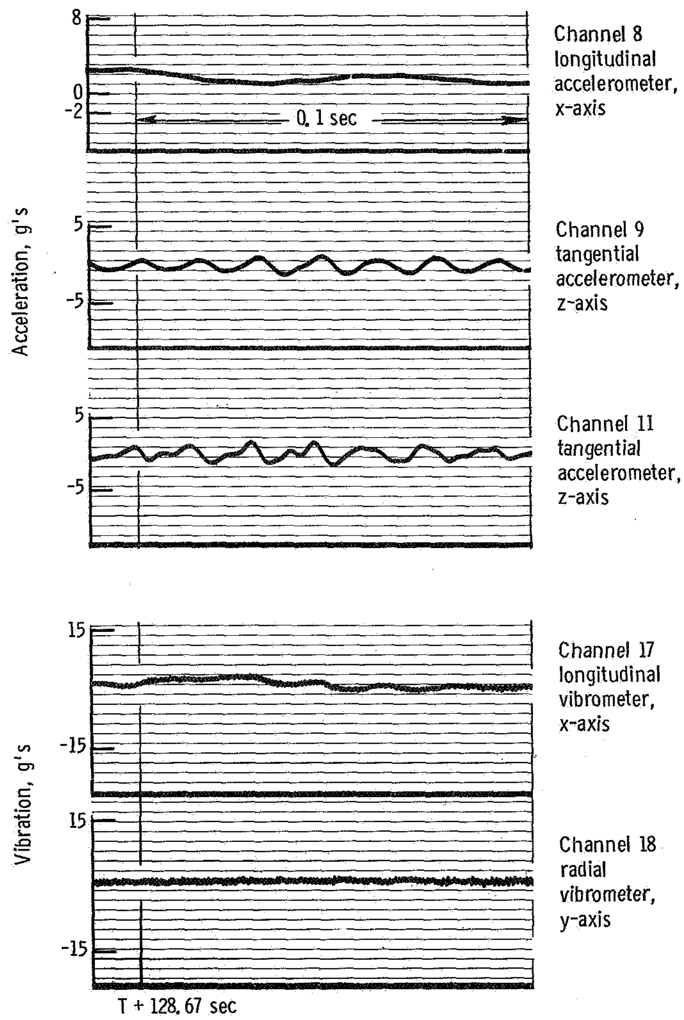


Figure D-4. - Dynamic data near time of booster engine cutoff, ATS-3.

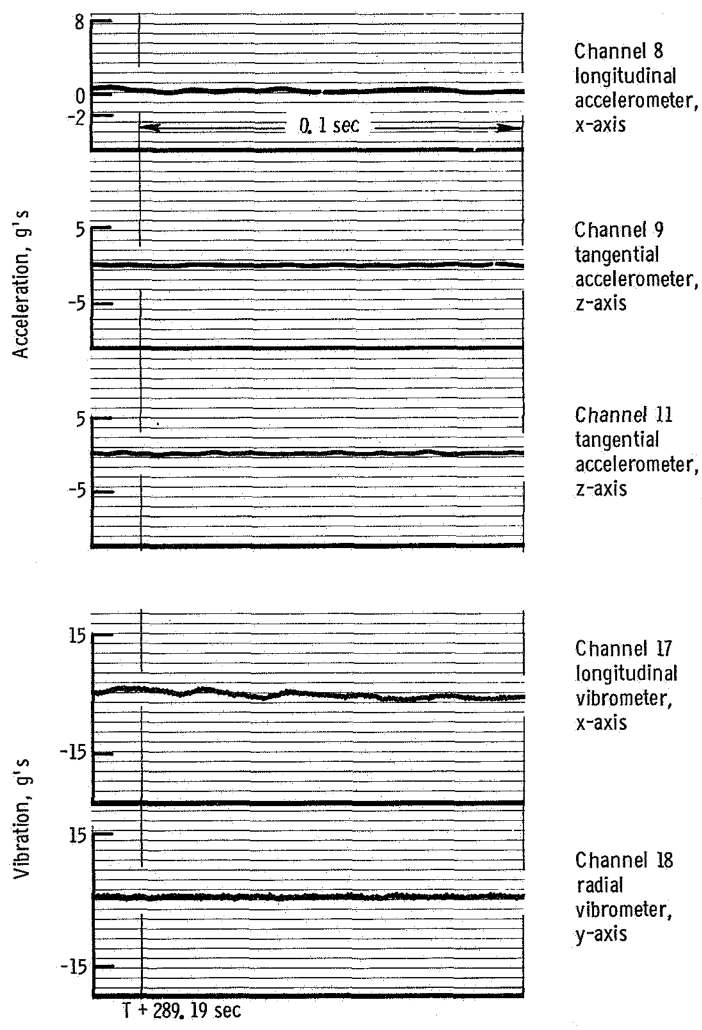


Figure D-5. - Dynamic data near time of sustainer engine cutoff, ATS-3.

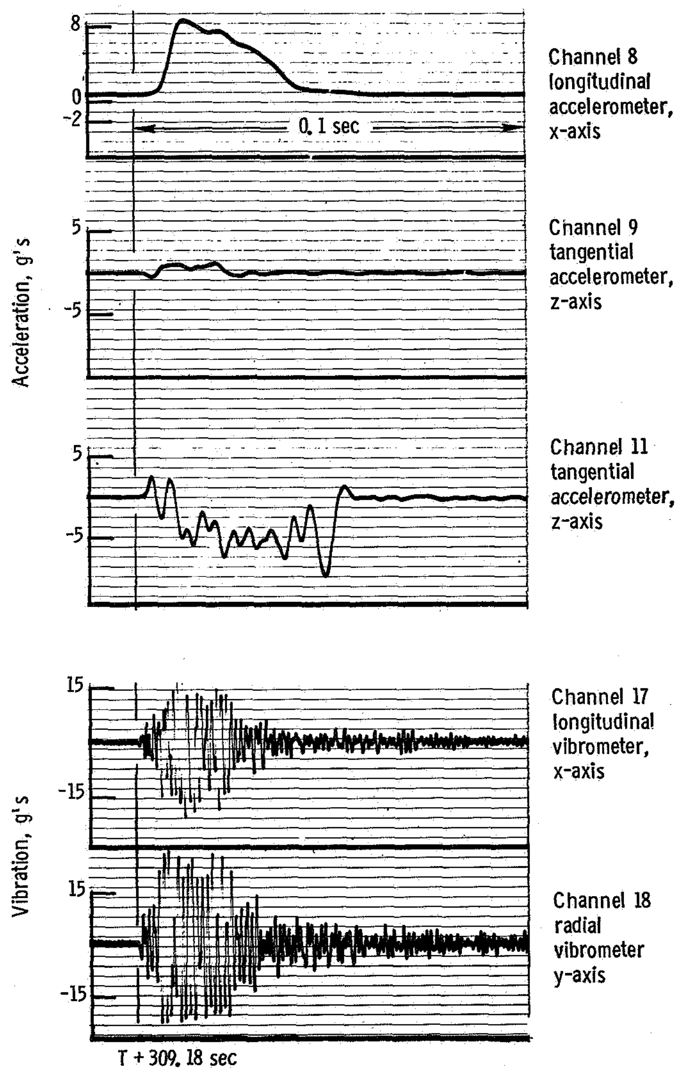


Figure D-6. - Dynamic data near time of horizon sensor fairing jettison, ATS-3.

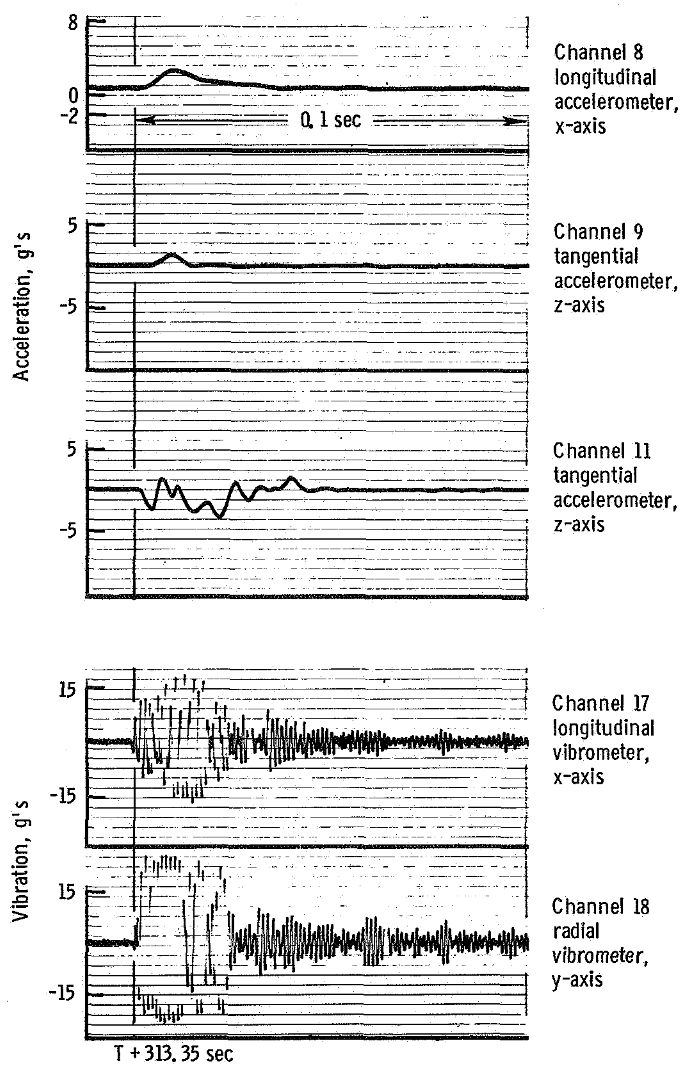


Figure D-7. - Dynamic data near time of Atlas-Agena separation, ATS-3.

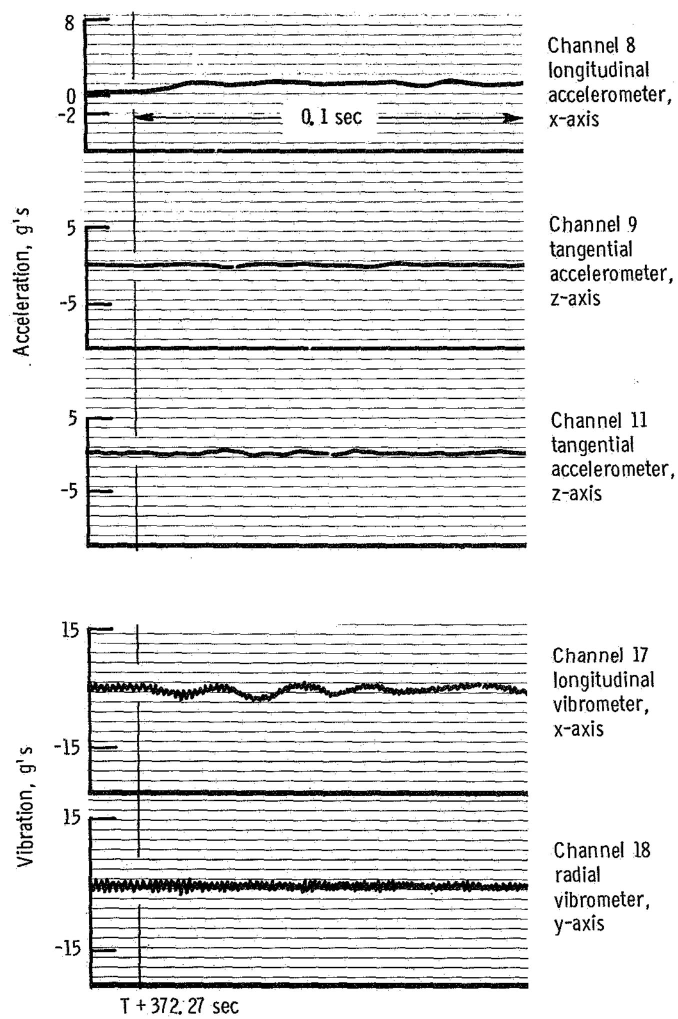


Figure D-8. - Dynamic data near time of Agena engine first ignition, ATS-3.

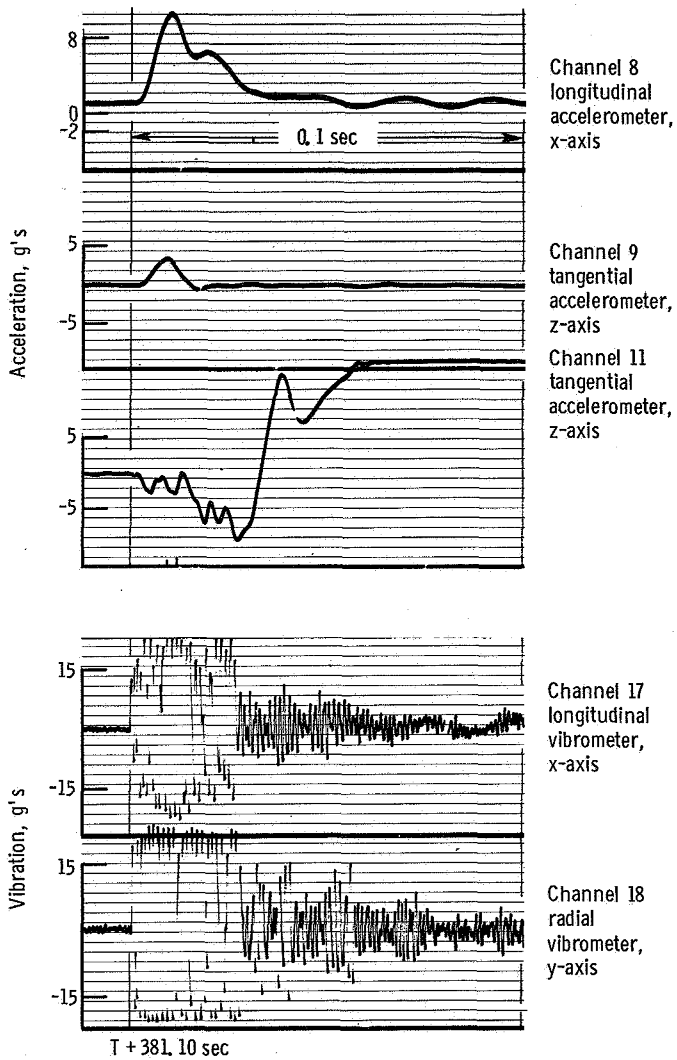


Figure D-9. - Dynamic data at time of shroud separation, ATS-3.

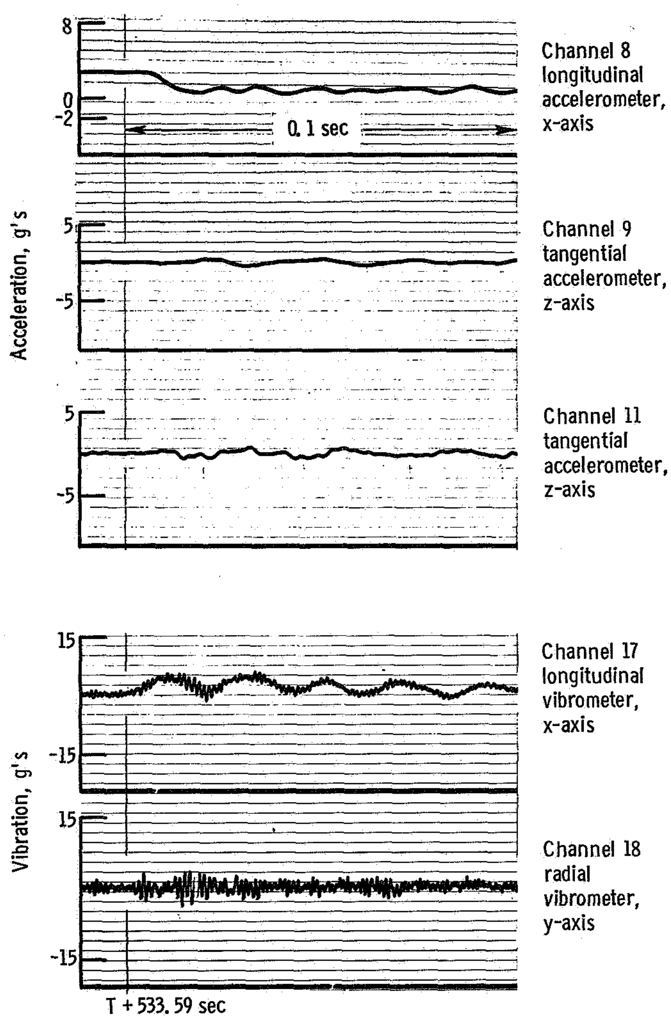


Figure D-10. - Dynamic data near time of Agena engine first cutoff, ATS-3.

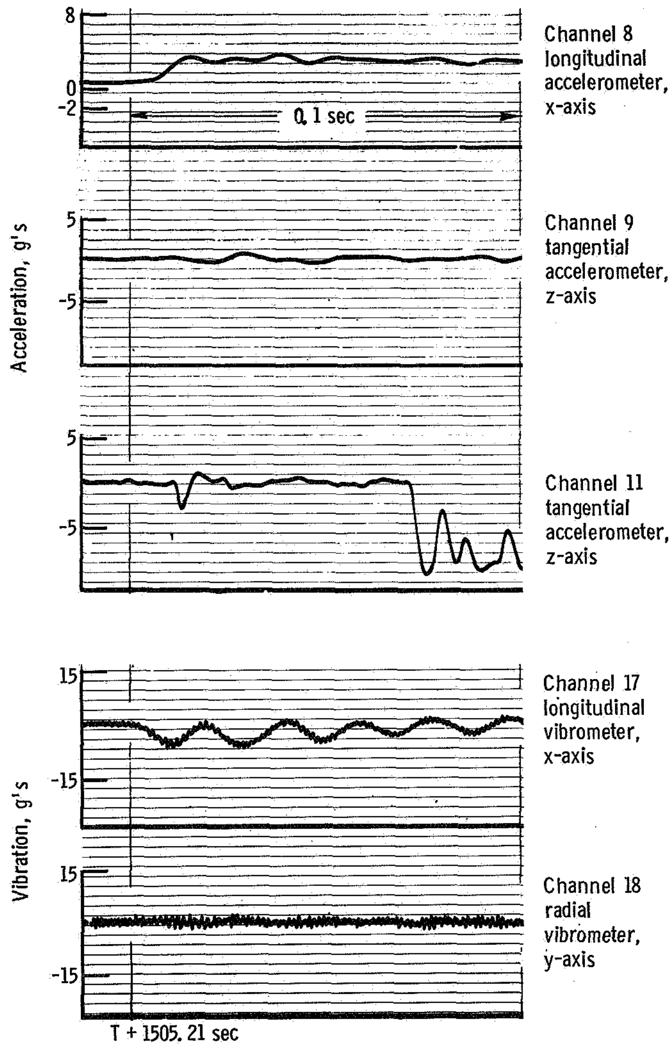


Figure D-11. - Dynamic data near time of Agena engine second ignition, ATS-3.

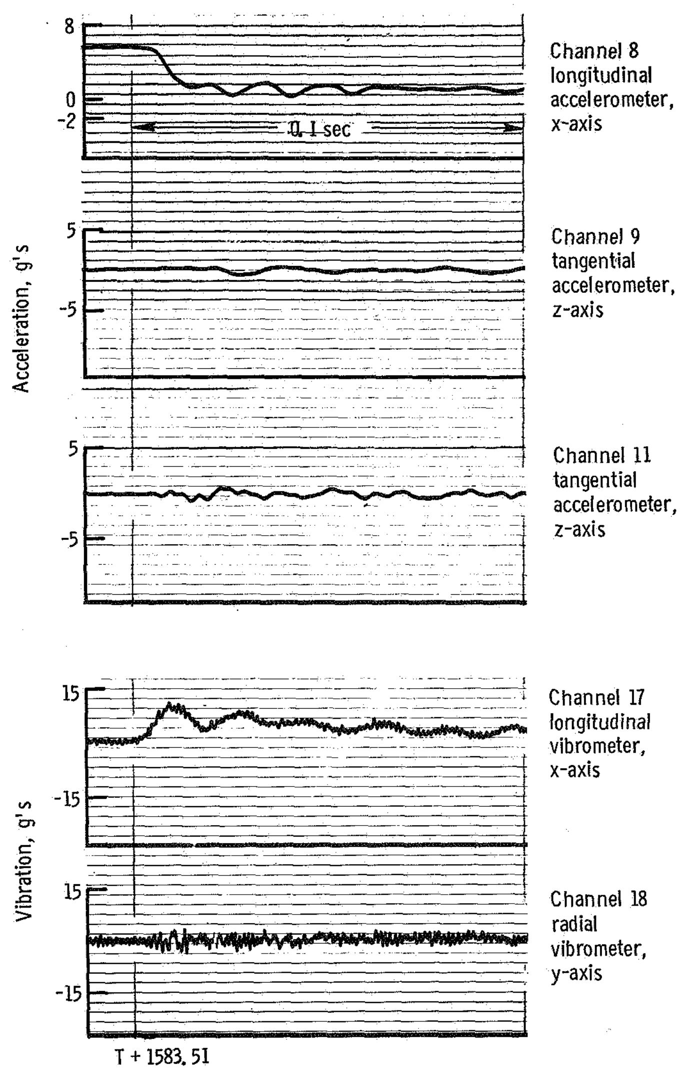


Figure D-12. - Dynamic data wear time of Agena engine second cut-off, ATS-3.

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