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FLIGHT PERFORMANCE OF ATLAS-AGENA LAUNCH VEHICLES IN SUPPORT OF THE LUNAR ORBITER MISSIONS III, IV, AND V

Lewis Research Center
Cleveland, Ohio

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ABSTRACT

Atlas-Agena launch vehicles successfully boosted a series of five Lunar Orbiters into proper lunar transfer trajectories. This report discusses the flight performance of the last three Atlas-Agena launch vehicles (for Lunar Orbiters III, IV, and V) from lift-off through the Agena retromaneuver. The objective of these flights was to perform selenographic investigations including photography of the lunar surface. Lunar Orbiter V, launched in August 1967, concluded the Lunar Orbiter missions.

CONTENTS

I. SUMMARY

This report evaluates the performance of the last three of five successful Atlas-Agena vehicles launched from the Eastern Test Range in support of the Lunar Orbiter series of photographic and selenographic missions. These flights were so similar that results are combined into a single report. Lunar Orbiter III was launched February 5, 1967 (referenced to GMT), followed by Lunar Orbiter IV on May 4, 1967, and Lunar Orbiter V on August 1, 1967. All three flights used a similar mode of ascent wherein the Atlas boosted the Agena-Lunar Orbiter into a proper suborbital coast ellipse. After separation of the Agena-spacecraft from the Atlas, the Agena engine was started and the Agena-spacecraft was injected into the desired approximate 185 kilometer circular parking orbit. After an orbital coast period, the Agena engine was restarted and the spacecraft was injected into the lunar transfer trajectory. Insertion of the spacecraft into the translunar trajectories was very accurate and out of the allocated 75 meter per second spacecraft midcourse correction capability, only 3.3 meters per second would have been required for Lunar Orbiter III; 10.8 meters per second for Lunar Orbiter IV; and 19.5 meters per second for Lunar Orbiter V to place the spacecraft onto the proper trajectory for the spacecraft deboost into its initial orbit around the moon. All three Lunar Orbiters successfully achieved the desired lunar orbits.

II. INTRODUCTION

The primary purpose of the Lunar Orbiter project was to obtain photography of the Earth's moon to aid in the selection of Apollo and Surveyor landing sites and to increase our scientific knowledge of the moon. Other objectives were to provide information on the micrometeoroid environment and the gravity field of the moon.

The Lunar Orbiters were a series of five lunar satellites launched by the Atlas-Agena. The project management of the Atlas-Agena launch vehicle, which supported the Lunar Orbiter missions, was under the direction of the Lewis Research Center. Lunar Orbiter I was launched in August of 1966 and Lunar Orbiter II was launched in November of 1966. These first two missions were site search missions which photographically examined potential landing areas for the Surveyor and Apollo. Lunar Orbiter III was launched in February 1967 and was a site confirmation mission. In May 1967 Lunar Orbiter IV was launched solely to gather scientific data of the moon. Lunar Orbiter V was launched in August 1967 as a further site examination and scientific mission.

The Atlas vehicle was used to boost the combined Agena - Lunar Orbiter onto a sub-orbital coast ellipse. After the Agena - Lunar Orbiter was separated from the Atlas, the Agena performed two separate engine firings to place the Lunar Orbiter spacecraft onto the proper translunar trajectory.

This report discusses the flight performance of the Atlas and Agena launch vehicles for Lunar Orbiters III, IV, and V from lift-off through Agena retromaneuver.

III. LAUNCH VEHICLE DESCRIPTION

by Joseph A. Ziemianski and Eugene E. Coffey

The Atlas-Agena is a two and one-half stage launch vehicle consisting of an Atlas first and one-half stage and an Agena second stage connected by a booster adapter. The Atlas is 3.05 meters (10 ft) in diameter except that the forward section of the tank is conical and tapers to a diameter of about 2 meters (6 ft). The Agena is 1.52 meters (5 ft) in diameter. The composite vehicle including the spacecraft shroud and booster adapter is 31.7 meters (104 ft) in length. The vehicle weight at lift-off approximates 126 500 kilograms (279 000 lb). Figure III-1 shows an Atlas-Agena launch vehicle lift-off with a Lunar Orbiter spacecraft.

The Atlas SLV-3 (fig. III-2) is 21.34 meters (70 ft) long. It is propelled by a standard Rocketdyne MA-5 propulsion system consisting of a booster engine having two thrust chambers with a total thrust at sea level of 1467.9×10^{3} newtons (330 000 lb); a sustainer engine with a thrust at sea level of 253.55×10^{3} newtons (57 000 lb); and two vernier engines, each with a thrust at sea level of 2.98×10^{3} newtons (669 lb). All engines use liquid oxygen and high grade kerosene propellants and are ignited prior to lift-off. The booster thrust chambers are gimbaled for pitch, yaw, and roll control during the booster phase of flight. This phase is completed when the vehicle acceleration equals about 6 g's. The booster engines are 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 only. The sustainer engine burns until the vehicle achieves the desired suborbital coast ellipse as determined by the radio guidance system. After sustainer engine shutdown, the vernier engines continue to burn for a short period of time prior to the Atlas-Agena separation. During this vernier phase the vernier engines are gimbaled to provide vehicle attitude control and fine trajectory corrections. After vernier engine shutdown, the Atlas is severed from the Agena by the firing of a Mild Detonating Fuse (Primacord) severance system located on the booster adapter. The firing of a retrorocket system, mounted on the booster adapter, then separates the Atlas and the booster adapter from the Agena.

The Agena stage and the shroud protecting the spacecraft are shown in figure III-3. This stage including the shroud is about 10.36 meters (34 ft) in length. The Agena is powered by a model 8096 Bell Aerosystems engine with a rated vacuum thrust of 71.17×10^{3} newtons (16 000 lb) and has a two burn capability. This engine uses unsymmetrical dimethylhydrazine and inhibited red fuming nitric acid as propellants. During powered flight, pitch and yaw control is 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. A retrorocket system on the Agena is fired, after spacecraft release, to ensure that the Agena will neither interfere with the spacecraft nor impact the moon. A metal shroud is used to provide an environmental protection for the Lunar Orbiter spacecraft during ascent. This shroud is jettisoned after vernier engine shutdown just prior to Atlas-Agena separation.

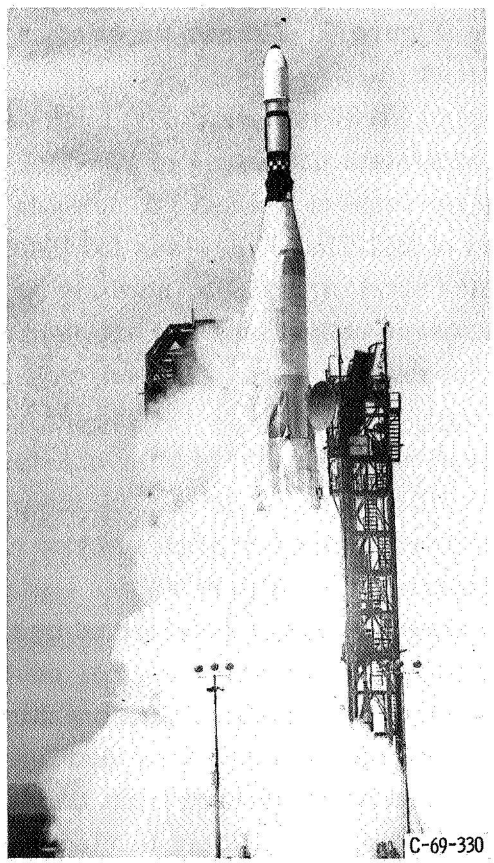


Figure III-1. - Atlas-Agena vehicle lifting off with Lunar Orbiter.

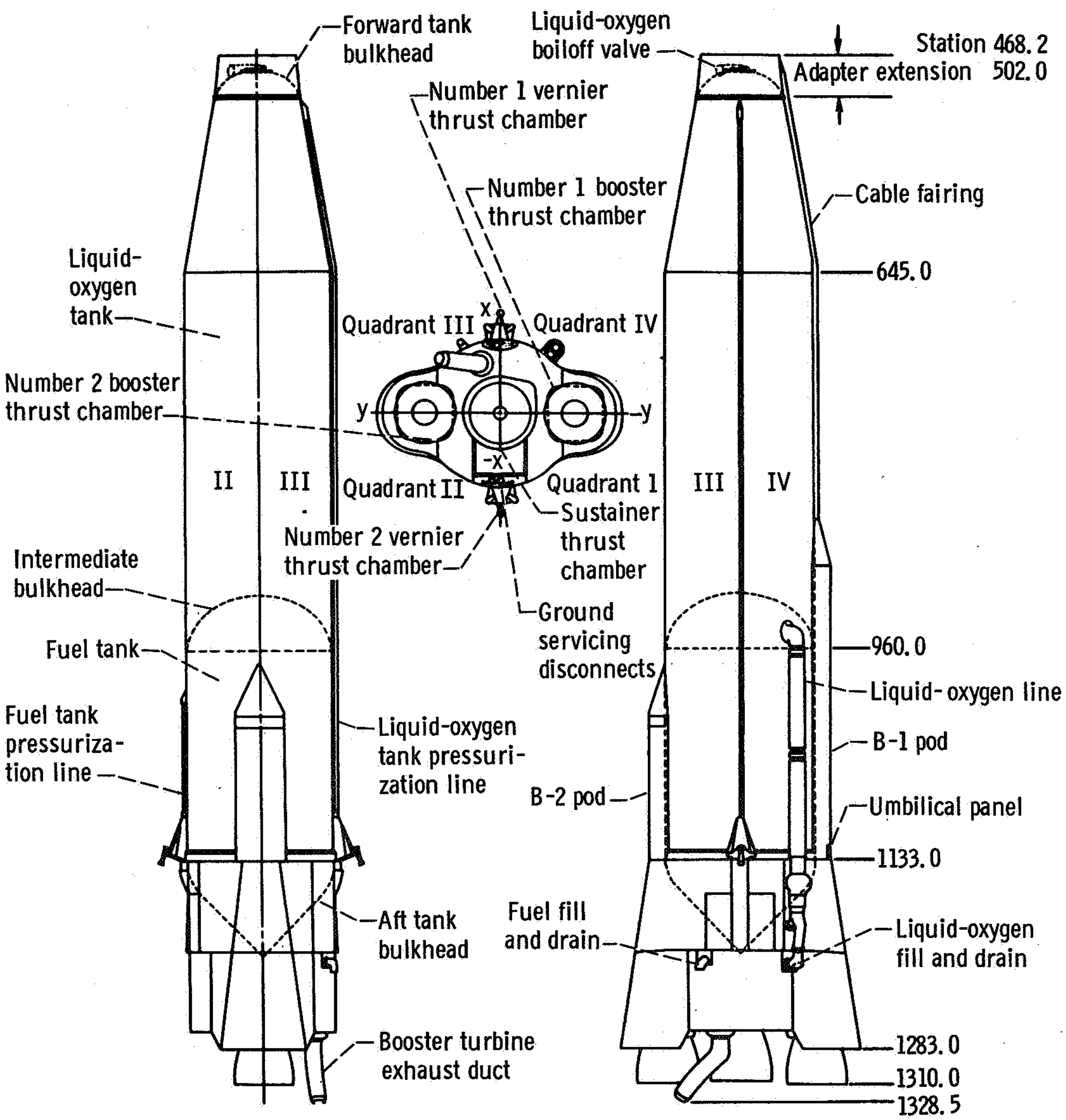


Figure III-2. - Atlas SLV-3 configuration, Lunar Orbiter.

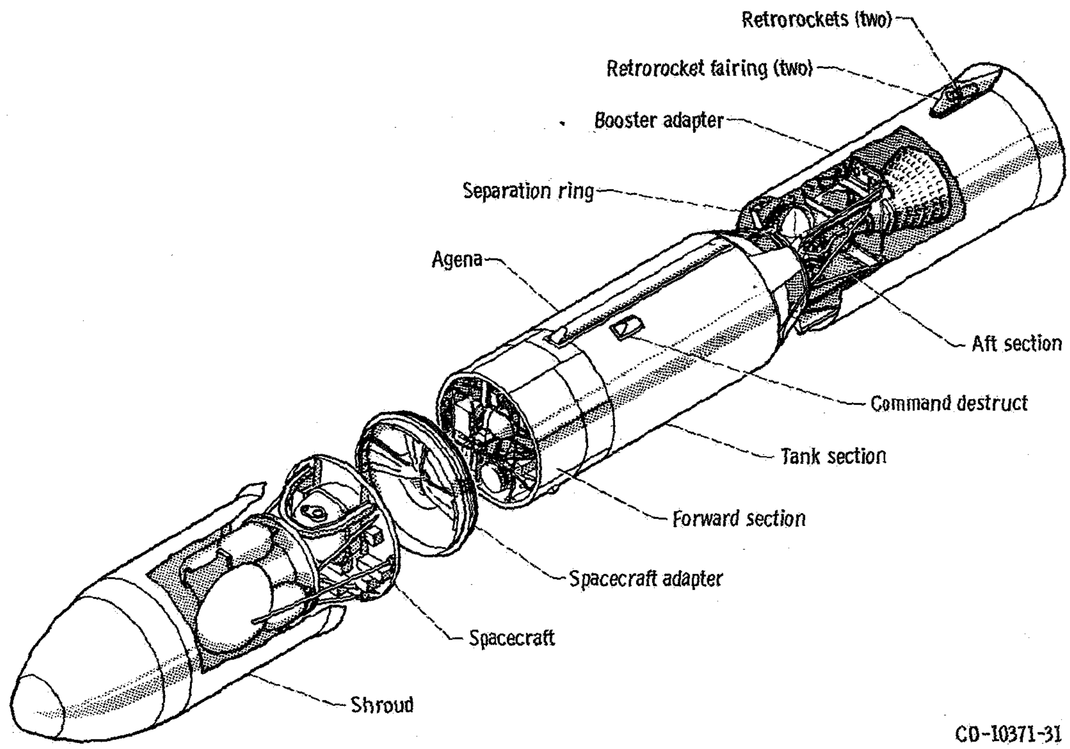


Figure III-3. - Agena-shroud-spacecraft-adapter, Lunar Orbiter.

IV. TRAJECTORY AND PERFORMANCE

by James C. Stoll and Kenneth A. Adams

TRAJECTORY PLAN

The Atlas boosts the Agena - Lunar Orbiter onto a prescribed suborbital coast ellipse. The Atlas flight consists of three powered phases: a booster phase, a sustainer phase, and a vernier phase. Shroud ejection, followed by Atlas-Agena separation, occurs after the vernier phase is completed. Following Atlas-Agena separation, the Agena engine first burn places the Agena - Lunar Orbiter onto a circular parking orbit at an altitude of 185 kilometers (100 n mi). The Agena - Lunar Orbiter coasts in the parking orbit to a predetermined point where the Agena engine is reignited. The Agena engine second burn places the Agena - Lunar Orbiter onto a lunar transfer orbit. After the spacecraft is separated, the Agena performs a retrosequence to ensure that it will neither interfere with the spacecraft Canopus seeker nor impact the moon.

Lunar and planetary missions require a flight azimuth which is a function of lift-off time. In practice, flight azimuths for the midpoints of successive 15-minute intervals throughout the daily launch window are precalculated. These intervals are identified as "Launch Plans." The precalculated flight azimuth for the designated launch plan is programmed in the Atlas airborne flight control programmer. The programmer controls roll gyro torque during the 13 seconds of flight starting at lift-off plus 2 seconds (T + 2). If launch occurs at a time other than the instant corresponding to the programmed flight azimuth, a discrepancy exists between the programmed flight azimuth and the desired flight azimuth. This error requires a guidance correction during ascent. The correction, generated by the ground radio guidance computer, is accomplished by yaw steering during the sustainer phase of the Atlas flight.

The coast time between Agena engine first burn and Agena engine second burn varies from mission to mission and also from Launch Plan to Launch Plan for the same mission. This variation is caused by the changing position of the predetermined point for Agena engine second ignition relative to the launch site. The changing position of this point is a result of the Earth's rotation on its axis and the motion of the target body relative to the Earth.

TRAJECTORY RESULTS - LUNAR ORBITER III

Lift-Off Through Atlas Booster Phase

Lunar Orbiter III (LO-III) was successfully launched from Complex 13, Eastern Test Range, on February 5, 1967 at 0117:01.120 GMT on Launch Plan 5C (see ref. 1). This launch plan required a programmed flight azimuth of 81.6°. The launch pad azimuth is 105.18°. The desired flight azimuth for the actual lift-off time was 80.8°. A comparison of actual and expected times for major flight events for this mission is presented in appendix A under LO-III.

Winds aloft at launch were predominately from the west with a maximum velocity of 58.5 meters per second (192 ft/sec) at an altitude of 12 588 meters (41 300 ft) as shown in figures IV-1 and IV-2. The winds had only a minor effect on the vehicle flight path. Abrupt changes in wind velocity at altitudes between 10 972 meters (36 000 ft) and 12 496 meters (41 000 ft) did produce strong wind shears.

As calculated from the T - 0 (lift-off) weather balloon data, the maximum vehicle bending response was 54.0 percent of the critical value and occurred at the Lunar Orbiter - spacecraft adapter interface (Agena Station 247) at 3003.194 meters (9853 ft) altitude. From the same data the maximum booster engine gimbal angle was calculated to be 44 percent of the available gimbal angle in the pitch plane and to occur at 3079 meters (10 102 ft) altitude.

Radar tracking data show that the vehicle flight path was slightly lower than the expected trajectory during the Atlas booster phase of the flight. Telemetry data indicate that the major cause of the depressed flight path was an approximate 0.5° excess total pitchover command by the autopilot during the booster phase. At booster engine cutoff (BECO) the actual trajectory was 1.2 kilometers (0.67 n mi) downrange of the expected trajectory.

The trajectory deviated in the horizontal plane only slightly from the expected trajectory, resulting in a position 0.4 kilometers (0.22 n mi) left at BECO. This effect was due primarily to a 0.2° excess vehicle roll during the programmed roll maneuver. Thrust misalinement and yaw gyro drift were small and had little effect on the trajectory.

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 station between T + 100 and T + 110 seconds. No booster pitch steering occurred during this interval since the velocity vector angle dispersion in the pitch plane was less than the predetermined threshold for booster pitch steering. Radio guidance yaw steering was not programmed to be used during the booster phase of flight. BECO occurred at T + 129.8 seconds by ground radio guidance at a vehicle longitudinal acceleration of 6.19 g's. The acceleration level at BECO was

0.05 g higher than expected but within the guidance tolerance of ±0.2 g. The booster engines were jettisoned at T + 132.99 seconds.

Atlas Sustainer Phase

The actual trajectory remained depressed and to the left during the sustainer phase. Sustainer steering was started at T + 138.2 seconds. The vehicle was pitched up approximately 5.5° and yawed left about 1.0° by the first steering commands. These maneuvers were made to compensate for the low trajectory and to steer the vehicle to the desired flight azimuth. No corrections were made for the crossrange displacement errors accumulated during the booster phase. Therefore, these errors were in evidence at sustainer engine cutoff (SECO). SECO was commanded by ground radio guidance at T + 288.0 seconds, 0.3 second earlier than expected. Tracking data indicate that the vehicle position was 2.7 kilometers (1.46 n mi) downrange, 2.3 kilometers (1.24 n mi) left, and 2.5 kilometers (1.35 n mi) low, as compared to the expected position, at SECO. The vehicle velocity (relative to a rotating earth) at SECO was 1.22 meters per second (4 ft/sec) greater than expected. The total Atlas performance including this velocity increment and the depressed trajectory of the vehicle resulted in the desired energy for the expected suborbital coast ellipse. The suborbital coast ellipse parameters are given in table IV-I.

Atlas Vernier Phase

Vernier engine thrust duration after SECO was approximately 21.2 seconds. During the vernier phase pitch-down and yaw-right steering commands were issued by radio guidance in order to place the vehicle in the proper attitude before Atlas-Agena separation These commands displaced the vehicle 0.45° down in pitch and 0.89° right in yaw. Vernier engine cutoff (VECO) occurred by ground radio guidance at T + 309.21 seconds, 0.5 second later than expected. Atlas insertion velocities at VECO are given in table IV-II.

Shroud separation was commanded by radio guidance at T + 311.43 seconds, 0.5 second later than expected. This delay was due to a guidance equation requirement that the transmission of this discrete must occur during the ground computer computation cycle following VECO plus 2.0 seconds.

Agena Engine First Burn Phase

Atlas-Agena separation squibs were fired at T + 313.6 seconds, 0.6 second later than expected but consistent with the guidance equation requirement that the discrete must occur during the ground computer computation cycle following VECO plus 4.0 seconds. After Atlas-Agena separation, an Agena pitchdown maneuver placed the vehicle in the proper attitude for initiating Agena engine ignition.

The start Agena primary timer (SAT) discrete had been transmitted during the Atlas phase by the radio guidance system at T + 297.36 seconds, 3.5 seconds later than expected. The ground 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 predicted. The guidance system adjusted SAT by 3.5 seconds so that Agena first burn would occur at the proper altitude. Since the SAT discrete was 3.5 seconds later than expected, the actual times for Agena first burn phase events listed in appendix A under LO-III are later than expected by this amount within the ±0.2 second timer tolerance.

Agena first ignition occurred at T + 370.4 seconds. Thrust duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 155.7 seconds, which was 1.0 second longer than expected. The longer thrust duration was attributed to a slightly lower than expected thrust level. Velocity meter shutdown indicated that the proper velocity was gained. First burn thrust decay velocity was 2.7 meters per second (8.9 ft/sec) compared to an expected 3.1 meters per second (10.1 ft/sec). After the first firing phase, the Agena coasted for 577.1 seconds on the nearly circular parking orbit to the proper spatial position for the Agena second burn phase. The actual parking orbit parameters are listed in table IV-III.

Agena Second Burn Phase

The start restart timer (SRT) discrete had been generated during the Atlas phase at T + 270.48 seconds, 0.3 second later than expected. The time of SRT was adjusted 0.3 second by the ground guidance system so that Agena second burn would occur at the proper spatial position. Since the SRT discrete was transmitted 0.3 second later than expected, the actual times for Agena second burn phase events listed in appendix A under LO-III are later by this amount within the ±0.2 second timer tolerance.

Agena second ignition occurred at T + 1104.5 seconds. Second burn thrust duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 88.7 seconds, 0.3 second longer than expected. As in the case of first burn phase the longer thrust duration was the result of lower than expected thrust level. Velocity meter shutdown indicated that the proper velocity was gained. Second burn phase thrust decay velocity was 9.876 meters per second (32.4 ft/sec) as compared to the expected value of 13.045 meters per second (42.8 ft/sec).

Lunar Orbiter Injection Phase

The spacecraft was separated from the Agena at T + 1358.6 seconds. The spacecraft trajectory parameters at final injection are given in table IV-IV. Uncorrected target point parameters for the spacecraft are given in table IV-V. The spacecraft midcourse correction which would have been required 15 hours after spacecraft injection to achieve target parameters was calculated to be 3.3 meters per second (10.827 ft/sec). This calculated correction indicated that the launch vehicle performance was within the 3 sigma allowances. The midcourse corrections actually performed were based on mission considerations not discussed in this report.

Postinjection and Agena Retrosequence

The Agena attitude control system began a 180° yaw maneuver at T + 1361.5 seconds and completed it at T + 1421.5 seconds. At T + 1958.5 seconds the Agena retrorocket was fired and the retrosequence was successfully completed. The Agena did not interfere with the spacecraft's Canopus seeker nor impact the moon. Integration of the resulting Agena position and velocity vectors to the moon resulted in the lunar miss distances tabulated in table IV-VI.

The Lunar Orbiter spacecraft achieved its desired orbit and performed satisfactorily.

TRAJECTORY RESULTS - LUNAR ORBITER IV

Lift-Off Through Atlas Booster Phase

Lunar Orbiter IV (LO-IV) was successfully launched from Complex 13, Eastern Test Range, on May 4, 1967 at 2225:00.571 GMT on Launch Plan 4H (see ref. 2). This launch plan required a programmed flight azimuth of 101.6° from the launch pad azimuth of 105.18°. The desired flight azimuth for the actual lift-off time was 106.1°. A comparison of actual and expected times for major flight events for this mission is presented in appendix A under LO-IV.

Winds aloft at launch were predominately from the west with a maximum velocity of 33.8 meters per second (111 ft/sec) at an altitude of 15 088 meters (49 500 ft) as shown in figures IV-3 and IV-4. The winds had only a minor effect on the vehicle flight path. Abrupt changes in wind velocity at altitudes between 9144 meters (30 000 ft) and 13 716 meters (45 000 ft) did produce strong wind shears.

As calculated from the T - 0 (lift-off) weather balloon data, the maximum vehicle bending response was 45.2 percent of the critical value and occurred at the Lunar Orbiter - spacecraft-adapter interface (Agena station 247) at 9395.5 meters (30 825 ft). From the same data the maximum booster engine gimbal angle was calculated to be 49.4 percent of the available gimbal angle in the pitch plane and to occur at 9395.5 meters (30 825 ft) altitude.

A comparison of radar position data with the expected trajectory shows that the vehicle flight profile was very near that expected during the Atlas booster phase of the flight. By booster engine cutoff (BECO) the actual trajectory was only 0.24 kilometer (0.13 n mi) lower than and 0.18 kilometer (0.1 n mi) downrange of the expected trajectory.

The trajectory deviated in the horizontal plane only slightly from the expected trajectory, resulting in a position 0.09 kilometer (0.05 n mi) right at BECO. This effect was primarily due to a programmed roll dispersion of 0.45° less than required. Thust misalinement and yaw gyro drift were small and had little effect on the trajectory.

During the booster phase of flight the capability of 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 station between T + 100 and T + 110 seconds. No booster pitch steering occurred during this interval since the velocity vector angle dispersion in the pitch plane was less than the predetermined threshold beyond which booster steering takes place. Radio guidance yaw steering was not programmed to be used during the booster phase of flight. BECO occurred at T + 128.3 seconds by ground radio guidance at a vehicle longitudinal acceleration level of 6.20 g's. The acceleration level at BECO was 0.04 g higher than expected but well within the guidance tolerance of ±0.2 g. The booster engines were jettisoned at T + 131.54 seconds.

Atlas Sustainer Phase

The actual trajectory remained slightly low and to the right during the sustainer phase. Sustainer steering was started at T + 136.9 seconds. The vehicle was pitched up approximately 2.9° and yawed left about 2.5° by the first steering commands. These maneuvers were made to compensate for the slightly low trajectory and to steer the vehicle to the desired flight azimuth. As in the case of LO-III no corrections were made for the crossrange displacement errors accumulated during the booster phase.

Therefore, these errors were in evidence at sustainer engine cutoff (SECO). SECO occurred on ground radio guidance command at T + 289.4 seconds, 0.93 second earlier than expected. Tracking data indicate that the vehicle position was 0.67 kilometer (0.36 n mi) downrange, 0.01 kilometer (0.05 n mi) right, and 0.30 kilometer (0.16 n mi) high, as compared to the expected position, at SECO. The vehicle velocity (relative to a rotating earth) at SECO was 25.6 meters per second (84 ft/sec) lower than expected. The total Atlas performance including this velocity decrement and the lofted trajectory resulted in the desired energy for the expected suborbital coast ellipse. The suborbital coast ellipse parameters are given in table IV-I.

Atlas Vernier Phase

The vernier engine thrust duration after SECO was approximately 20.72 seconds. During the vernier phase pitch-up and yaw right steering commands were issued by radio guidance in order to place the vehicle in the proper attitude before Atlas-Agena separation. These commands displaced the vehicle 0.3° up in pitch and 0.7° right in yaw. Vernier engine cutoff (VECO) occurred by ground radio guidance at T + 310.15 seconds, 0.45 second earlier than expected. Atlas insertion velocities at VECO are given in table IV-II.

Shroud separation was commanded by radio guidance at T + 312.53 seconds, 0.47 second earlier than expected. This was consistent with the guidance equation requirement that the transmission of this discrete must occur during the ground computer computation cycle following VECO plus 2.0 seconds.

Agena Engine First Burn Phase

Atlas-Agena separation squibs were fired at T + 314.65 seconds, 0.35 second earlier than expected but consistent with the guidance equation requirement that the discrete must occur during the ground computer computation cycle following VECO plus 4.0 seconds. After Atlas-Agena separation, an Agena pitch-down maneuver placed the vehicle in the proper attitude for initiating Agena first engine ignition.

The start Agena primary timer (SAT) discrete had been transmitted during the Atlas phase by the radio guidance system at T + 292.0 seconds, 1.15 seconds earlier than expected. The ground guidance system had determined that the Agena would be injected onto the suborbital coast ellipse at an altitude higher than expected and, therefore, the Agena would reach first burn altitude earlier than predicted. The guidance system adjusted SAT by 1.15 seconds so that Agena first burn would occur at the proper altitude.

Since the SAT discrete was 1.15 seconds earlier than expected, the actual times for Agena first burn phase events listed in appendix A under LO-IV are earlier than expected by this amount within the ±0.2 second timer tolerance.

Agena first ignition occurred at T + 364.85 seconds. Thrust duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 152.00 seconds, 0.57 second shorter than expected. The shorter thrust duration was attributed to a slightly high acceleration which was a result of slightly higher than expected thrust level. Velocity meter shutdown indicated that the proper velocity was gained. First burn phase thrust decay velocity was 2.7 meters per second (8.9 ft/sec) compared to an expected 3.08 meters per second (10.1 ft/sec). After the first firing phase, the Agena coasted for 1241.87 seconds on the nearly circular parking orbit to the proper spatial position for Agena second burn phase. The actual parking orbit parameters are listed in table IV-III.

Agena Second Burn Phase

The start restart timer (SRT) discrete had been generated during the Atlas phase at T + 269.07 seconds, 0.79 second earlier than expected. The time of SRT was adjusted 0.79 second by the ground guidance system so that Agena second burn would occur at the proper spatial position. Since the SRT discrete was transmitted 0.79 second earlier than expected, the actual times for Agena second burn phase events listed in appendix A under LO-IV are earlier by this amount within the ±0.2 second timer tolerance.

Agena second ignition occurred at T + 1760.0 seconds. Second burn thrust duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 87.32 seconds, 0.11 second longer than expected. Velocity meter cutoff indicated that the proper velocity was gained. Second phase decay velocity is not available because there was no radar coverage at that point.

Lunar Orbiter Injection Phase

The spacecraft was separated from the Agena at T + 2013.07 seconds. The spacecraft trajectory parameters at final injection are given in table IV-IV. Uncorrected target point parameters for the spacecraft are given in table IV-V. The spacecraft midcourse correction which would have been required 15 hours after spacecraft injection to achieve target parameters was calculated to be 10.8 meters per second (35.43 ft/sec). This calculated correction indicated that the launch vehicle performance was within the 3 sigma allowances. The actual midcourse corrections actually performed are based on mission considerations not discussed in this report.

Postinjection and Agena Retrosequence

The Agena attitude control system began a 180° yaw maneuver at T + 2016.9 seconds and completed it at T + 2076.0 seconds. At T + 2613.13 seconds the Agena retrorocket was fired and the retromaneuver was successfully completed. The Agena did not interfere with the spacecraft Canopus seeker nor impact the moon. Integration of the resulting Agena position and velocity vectors to the moon resulted in the lunar miss distances tabulated in table IV-VI.

The Lunar Orbiter spacecraft achieved its desired orbit and performed satisfactorily.

TRAJECTORY RESULTS - LUNAR ORBITER V

Lift-Off Through Atlas Booster Phase

Lunar Orbiter V (LO-V) was successfully launched from Complex 13, Eastern Test Range, on August 1, 1967 at 2233:00.352 GMT on Launch Plan 1K (see ref. 3). This launch plan required a programmed flight azimuth of 105.6° from the launch pad azimuth 105.18°. The desired flight azimuth for the actual lift-off time was 104.9°. A comparison of actual and expected times for major flight events for this mission is presented in appendix A under LO-V.

Winds aloft at launch were predominately from the west and northwest with a maximum velocity of 12.8 meters per second (42 ft/sec) at an altitude of 12 802 meters (42 000 ft) as shown in figures IV-5 and IV-6. The winds had only a minor effect on the vehicle flight path. Above 7620 meters (25 000 ft) the winds were tail winds and tended to depress the trajectory. Abrupt changes in wind velocity at altitudes between 11 278 meters (37 000 ft) and 15 240 meters (50 000 ft) did produce moderate wind shears.

As calculated from the T - 0 (lift-off weather balloon data, the maximum vehicle bending response was 41.3 percent of the critical value and occurred at the Lunar Orbiter - spacecraft-adapter interface (Agena station 247) at 8433.2 meters (27 668 ft) altitude. From the same data the maximum booster engine gimbal angle was calculated to be 46.4 percent of the available gimbal angle in the pitch plane and to occur at 8500.3 meters (27 888 ft) altitude.

Radar tracking data show that the vehicle flight path was lower than the expected trajectory during the Atlas booster phase of the flight. The depressed trajectory was due in part to tail winds. Telemetry data indicate that the major cause of the depressed flight path was an approximate 0.46° excess total pitchover command by the autopilot during the booster phase. At booster engine cutoff (BECO) the actual trajectory was 1.4 kilometers (0.32 n mi) uprange of the expected trajectory.

The trajectory deviated in the horizontal plane only slightly from the expected trajectory, resulting in a position 0.09 kilometer (0.05 n mi) right at BECO. This effect was due primarily to a programmed roll dispersion of 0.1° less than required. Thrust misalinement and yaw gyro drift were small and had little effect on the trajectory.

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 station between T + 100 and T + 110 seconds. No booster pitch steering occurred during this interval since the velocity vector angle dispersion in the pitch plane was less than the predetermined threshold beyond which booster steering takes place. Radio guidance yaw steering was not programmed to be used during the booster phase of flight. BECO occurred at T + 128.6 seconds by ground radio guidance at a vehicle longitudinal acceleration of 6.08 g's. The acceleration level at BECO was 0.08 g higher than expected but within the guidance tolerance of ±0.2 g. The booster engines were jettisoned at T + 131.7 seconds.

Atlas Sustainer Phase

The actual trajectory remained depressed and to the right during the sustainer phase. Sustainer steering was started at T + 137.6 seconds. The vehicle was pitched up approximately 4.7° and yawed left about 1.2° by the first steering commands. These maneuvers were made to compensate for the low trajectory and to steer the vehicle to the desired flight azimuth. As in the case of LO-III and LO-IV, no corrections were made for the crossrange displacement errors accumulated during the booster phase. Therefore, these errors were in evidence at sustainer engine cutoff (SECO). SECO was commanded by ground radio guidance at T + 288.6 seconds, 1.68 seconds earlier than expected. Tracking data indicate that the vehicle position was 7.01 kilometers (3.79 n mi) up range, 0.9 kilometers (0.49 n mi) right, and 2.3 kilometers (1.24 n mi) low, as compared to the expected position at SECO. The vehicle velocity (relative to a rotating earth) at SECO was 7.6 meters per second (25 ft/sec) greater than expected. The total Atlas performance including this velocity increment and the depressed trajectory of the vehicle resulted in the desired energy for the expected suborbital coast ellipse. The suborbital coast ellipse parameters are given in table IV-I.

Atlas Vernier Phase

Vernier engine thrust duration after SECO was approximately 19.3 seconds. During the vernier phase pitch-down and yaw-right commands were issued by radio guidance in order to place the vehicle in the proper attitude before Atlas-Agena separation. These commands displaced the vehicle 0.2° down in pitch and 0.4° right in yaw. Vernier engine cutoff (VECO) occurred by ground radio guidance command at T + 307.9 seconds, 2.5 seconds earlier than expected. Atlas insertion velocities at VECO are given in table IV-II.

Shroud separation was commanded by radio guidance at T + 310.3 seconds, 2.2 seconds earlier than expected. This time was consistent with the guidance equation requirement that the transmission of this discrete must occur during the ground computer computation cycle following VECO plus 2.0 seconds.

Agena Engine First Burn Phase

Atlas-Agena separation squibs were fired at T + 312.4 seconds, 2.1 seconds earlier than expected, but consistent with the guidance equation requirement that the discrete must occur during the ground computer computation cycle following VECO plus 4.0 seconcs. After Atlas-Agena separation, an Agena pitch-down maneuver placed the vehicle in the proper attitude for initiating Agena first engine ignition.

The start Agena primary timer (SAT) discrete had been transmitted during the Atlas phase by the radio guidance system at T + 296.3 seconds, 4.05 seconds later than expected. The ground guidance system had determined that the Agena would be injected onto the suborbital coast ellipse at an altitude lower than predicted. The guidance system adjusted SAT by 4.05 seconds so that Agena first burn would occur at the proper altitude. Since the SAT discrete was 4.05 seconds late, the actual times for Agena first burn phase events listed in appendix A under LO-V are later than expected by this amount within the ±0.2 second timer tolerance.

Agena first ignition occurred at T + 369.3 seconds. Thrust duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 153.13 seconds, 1.42 seconds longer than expected. The longer thrust duration was attributed to a slightly low acceleration as a result of slightly lower than expected thrust level. Velocity meter shutdown indicated that the proper velocity was gained. First burn phase thrust decay velocity was 3.048 meters per second (10.0 ft/sec) as compared to a predicted 3.091 meters per second (10.14 ft/sec). After the first firing phase, the Agena coasted for 1355.82 seconds on the nearly circular parking orbit to the proper spatial position for Agena second burn phase. The actual parking orbit parameters are listed in table IV-III.

Agena Second Burn Phase

The start restart timer (SRT) discrete had been generated during the Atlas phase at T + 272.4 seconds, 0.46 second later than expected. The time of SRT was adjusted 0.46 second by the ground guidance system so that Agena second burn would occur at the proper spatial position. Since the SRT discrete was transmitted 0.46 second later than expected, the actual times for Agena second burn phase events listed in appendix A under LO-V are later than expected by this amount within the ±0.2 second timer tolerance.

Agena second ignition occurred at T + 1879.4 seconds. Second burn thrust duration (90 percent chamber pressure to velocity meter cutoff) was 87.5 seconds, 0.91 second longer than expected. As in the case of first burn the longer thrust duration was a result of lower than expected thrust level. Velocity meter shutdown indicated that the proper velocity had been gained. Second burn phase decay velocity is not available.

Lunar Orbiter Injection Phase

The spacecraft was separated from the Agena at T + 2133.42 seconds. The spacecraft trajectory parameters at final injection are given in table IV-IV. Uncorrected target point parameters for the spacecraft are given in table IV-V. The spacecraft midcourse correction which would have been required at 15 hours after spacecraft injection to achieve target parameters was calculated to be 19.5 meters per second (64 ft/sec). This calculated correction indicated that the launch vehicle performance was within the 3 sigma allowance. The midcourse corrections actually performed were based on mission considerations not discussed in this report.

Postinjection and Agena Retrosequence

The Agena attitude control system began a 180° yaw maneuver at T + 2136.39 seconds and completed it at T + 2196.27 seconds. At T + 2733.5 seconds the Agena retrorocket was fired and the retromaneuver was completed successfully. The Agena did not interfere with the spacecraft Canopus seeker nor impact the moon. Integration of the resulting Agena position and velocity vectors to the moon resulted in the lunar miss distance tabulated in table IV-VI.

The Lunar Orbiter spacecraft achieved its desired orbit and performed satisfactorily.

TABLE IV-I. - ATLAS SUBORBITAL COAST ELLIPSE PARAMETERS

Parameter	Units	Lunar Orbiter flight					
		III		IV		V	
		Expected	Actual	Expected	Actual	Expected	Actual
Semimajor axis	km	4423.380	4423.273	4423.380	4423.277	4423.380	4423.270
	n mi	2388.434	2388.376	2388.434	2388.379	2388.434	2388.375
Semiminor axis	km	3873.581	3873.422	3873.581	3873.658	3873.581	3873.618
	n mi	2091.566	2091.481	2091.566	2091.608	2091.566	2091.586
Radius vector magnitude at apogee	km	6559.171	6559.138	6559.159	6559.222	6559.144	6558.781
	n mi	3541.669	3541.651	3541.663	3541.697	3541.654	3541.458
Inertial velocity at apogee	km/sec	5.60597	5.60591	5.60603	5.60512	5.60646	5.60652
	ft/sec	18 392.3	18 392.1	18 392.5	18 389.5	18 393.9	18 394.1
Inclination	deg	29.51	(a)	29.74	(a)	31.07	(a)
Period	min	48.50	(a)	48.61	(a)	48.62	(a)

[a]Not applicable.

TABLE IV-II. - ATLAS INSERTION VELOCITIES AT VERNIER ENGINE CUTOFF

Parameter	Units	Lunar Orbiter flight					
		III		IV		V	
		Expected	Actual	Expected	Actual	Expected	Actual
Velocity magnitude	m/sec	564.94	564.93	564.44	564.43	564.96	564.94
	ft/sec	18 534.9	18 534.4	18 518.5	18 518.0	18 535.3	18 534.8
Altitude rate	m/sec	485.58	485.06	456.07	456.80	486.40	482.74
	ft/sec	1593.1	1591.4	1496.3	1498.7	1595.8	1583.8
Yaw velocity	m/sec	0	0.8534	0	0.2438	0	0.6096
	ft/sec	0	2.8	0	0.8	0	2.0

TABLE IV-III. - AGENA PARKING ORBIT

PARAMETERS (ACTUAL)

Parameter[a]	Units	Lunar Orbiter flight		
		III	IV	V
Apogee	km	201.868	192.608	196.312
	n mi	109.0	104.0	106.0
Perigee	km	179.644	177.792	176.644
	n mi	97.0	96.0	97.0
Inclination	deg	29.5	29.8	31.1
Period	min	88.2	88.2	88.2

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

TABLE IV-IV. - LUNAR ORBITER SPACECRAFT TRAJECTORY

PARAMETERS AT FINAL INJECTION (ACTUAL)

Parameter	Units	Lunar Orbiter flight		
		III	IV	V
Vis viva energy, [a] C_3	(km/sec^2)	-1.869	-1.832	-2.077
	[(n mi)/sec]^2	-0.5449	-0.5341	-0.6056
Radius	km	6729.907	6739.878	6645.815
	n mi	3633.859	3639.243	3588.453
Velocity	km/sec	10.380	10.372	10.923
	ft/sec	34 055.118	34 028.871	35 836.614
Flight path angle	deg	9.457	9.462	6.568
Inclination	deg	29.766	29.858	31.302

[a]Defined using the geocentric radius R and the inertial velocity V of the spacecraft as follows: $C_3 = V^2 - (2GM_E/R)$, where G is the universal gravitational constant and M_E is the mass of the Earth.

TABLE IV-V - UNCORRECTED LUNAR TARGET POINT

PARAMETERS, SPACECRAFT (ACTUAL)

Parameter[a]	Units	Lunar Orbiter flight		
		III	IV	V
$\overline{B} \cdot \overline{T}$	km	5132	9012	6933
	n mi	2771.058	4866.091	3743.521
$\overline{B} \cdot \overline{R}$	km	-1812	-2759	3477
	n mi	-978.402	-1489.741	1877.430
T_f	hr	92.5167	89.8870	91.3820

[a]The target parameter $\overline{B}$ is a vector directed from the center of the moon to the incoming asymptote of the approach hyperbola at arrival time and perpendicular to it. It is resolved into the components $\overline{B} \cdot \overline{T}$ and $\overline{B} \cdot \overline{R}$ in the selenocentric RST coordinate system, where $\overline{S}$ is a unit vector having the same direction as the incoming asymptote ($\overline{B}$ and $\overline{S}$ are perpendicular) and $\overline{T}$ is a unit vector perpendicular to $\overline{S}$, parallel to B, in the true lunar equatorial plane of date and directed such that $\overline{S} \times \overline{T}$ has the approximate direction of the moon's south pole. $\overline{R}$ is a unit vector completing the right orthogonal system with $\overline{S}$ and $\overline{T}$; $\overline{R} = \overline{S} \times \overline{T}$. The parameter T_f is the time of flight from the injection point to encounter.

TABLE IV-VI. - LUNAR MISS DISTANCE PARAMETERS, AGENA

Parameter[a]	Units	Lunar Orbiter flight					
		III		IV		V	
		Deep Space Instrumentation Facility[b]	Eastern Test Range[c]	Deep Space Instrumentation Facility	Eastern Test Range	Deep Space Instrumentation Facility	Eastern Test Range
$\overline{B} \cdot \overline{T}$	km	27 132	30 672	26 666	30 406	31 551	31 045
	n mi	14 650.108	16 561.555	14 398.488	16 417.927	17 036.177	16 762.959
$\overline{B} \cdot \overline{R}$	km	-4802	-5162	-2256	-2279	570	505
	n mi	-2592.873	-2787.257	-1218.143	-1230.562	307.775	272.678
T_f	hr	98.4089	99.1269	95.7157	96.7515	95.5261	95.4414

[a]The target parameter $\overline{B}$ is a vector directed from the center of the moon to the incoming asymptote of the approach hyperbola at arrival time and perpendicular to it. It is resolved into the components $\overline{B} \cdot \overline{T}$ and $\overline{B} \cdot \overline{R}$ in the selenocentric RST coordinate system, where $\overline{S}$ is a unit vector having the same direction as the incoming asymptote ($\overline{B}$ and $\overline{S}$ are perpendicular), and $\overline{T}$ is a unit vector perpendicular to $\overline{S}$, parallel to B, in the true lunar equatorial plane of date and directed such that $\overline{S} \times \overline{T}$ has the approximate direction of the moon's south pole. $\overline{R}$ is a unit vector completing the right orthogonal system with $\overline{S}$ and $\overline{T}$; $\overline{R} = \overline{S} \times \overline{T}$. The parameter T_f is the time of flight from the injection point to encounter.

[b]Includes an expected retrosimulation.

[c]Does not include a retrosimulation.

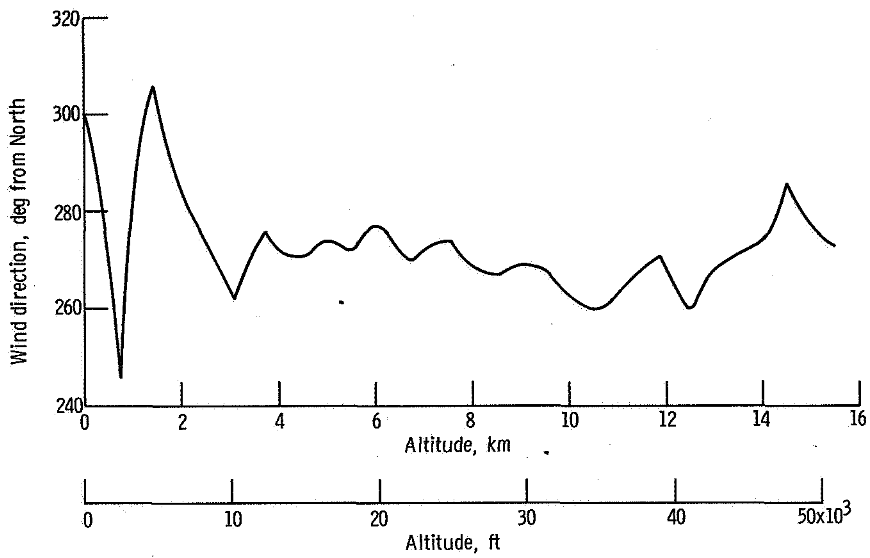


Figure IV-1. - Wind direction at lift-off, Lunar Orbiter III.

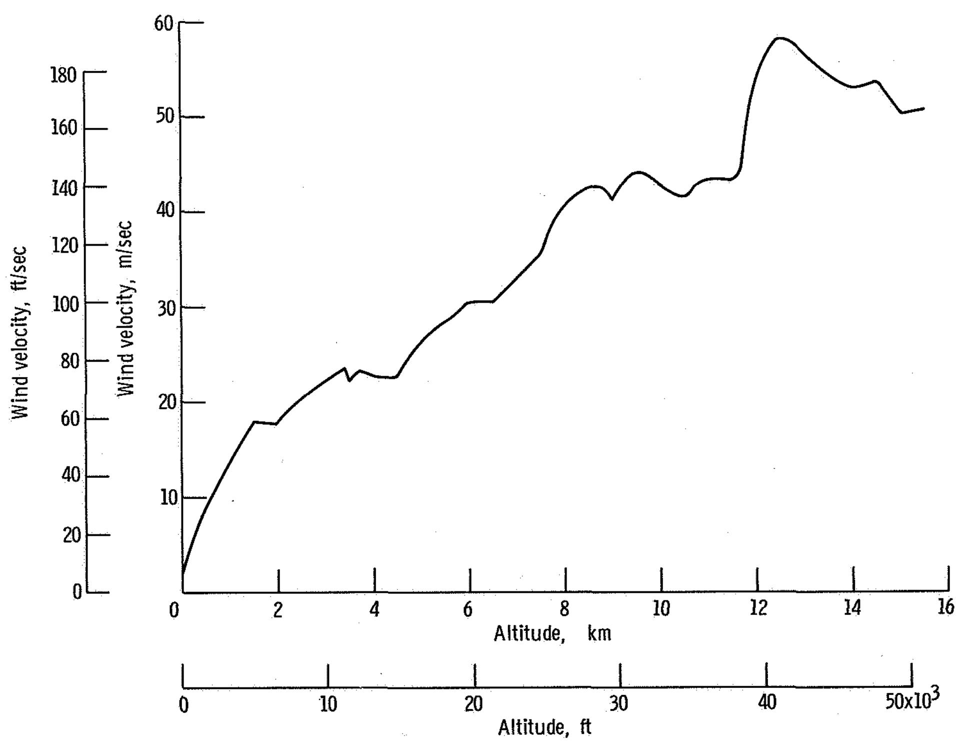


Figure IV-2. - Wind velocity at lift-off, Lunar Orbiter III.

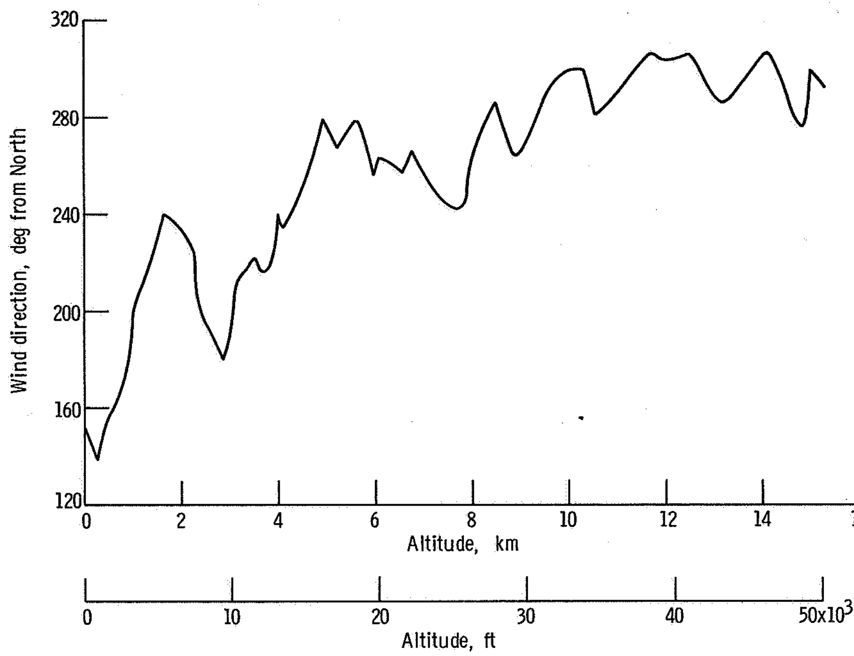


Figure IV-3. - Wind direction at lift-off, Lunar Orbiter IV.

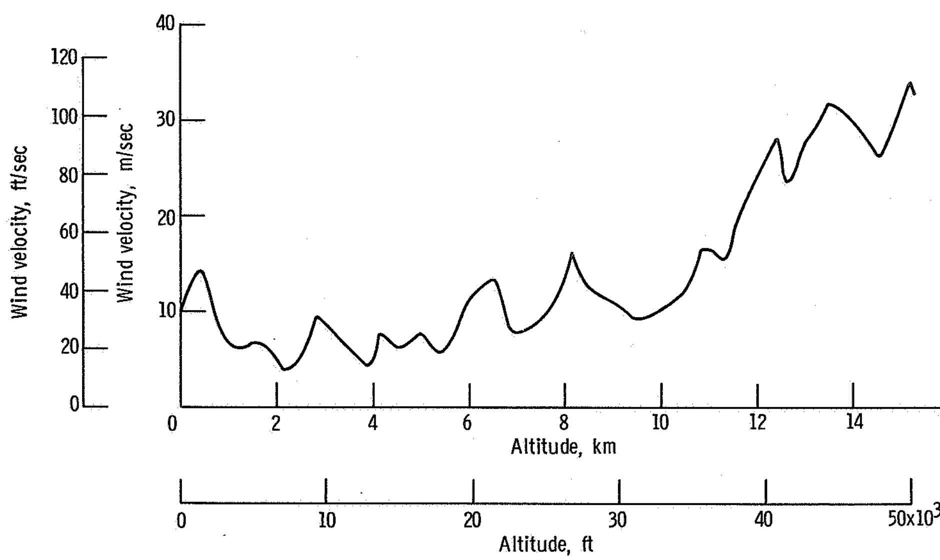


Figure IV-4. - Wind velocity at lift-off, Lunar Orbiter IV.

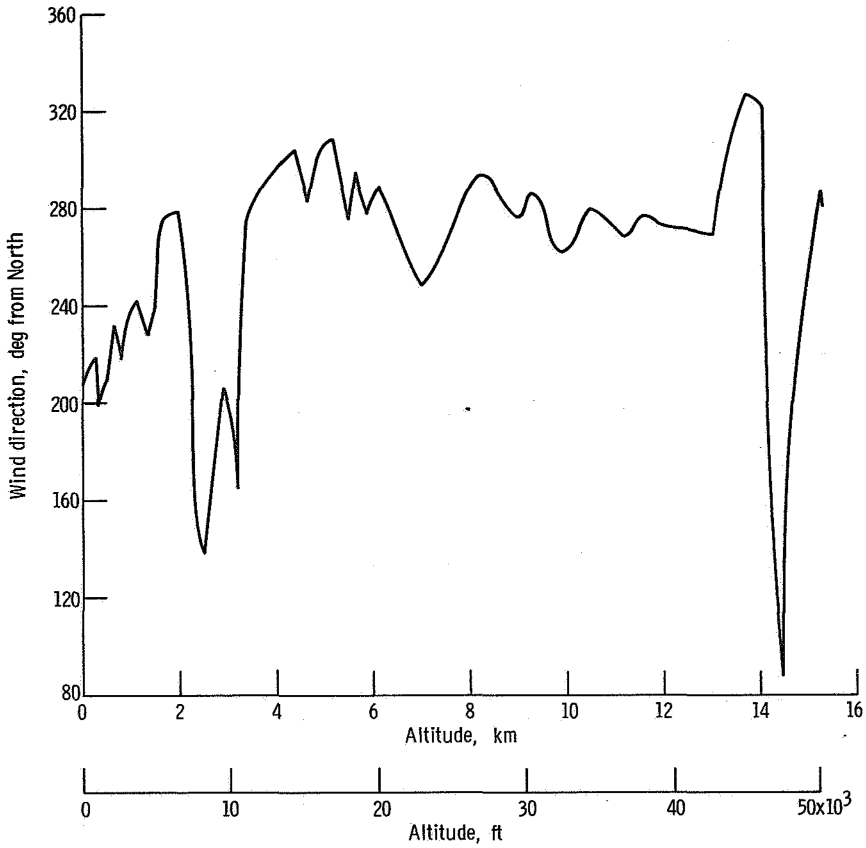


Figure IV-5. - Wind direction at lift-off, Lunar Orbiter V.

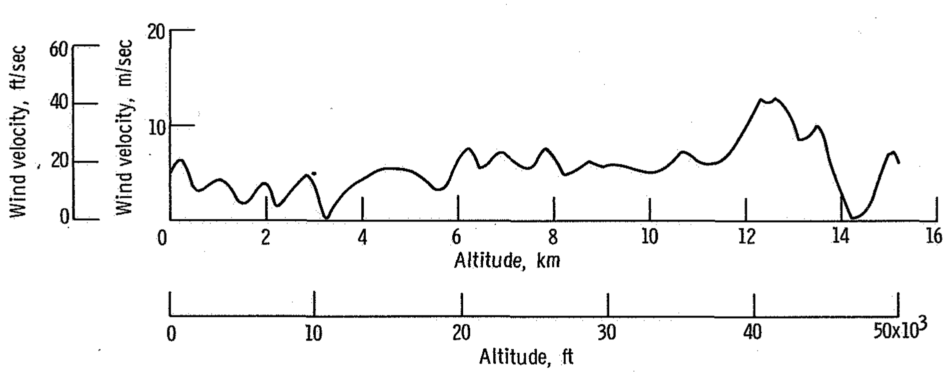


Figure IV-6. - Wind velocity at lift-off, Lunar Orbiter V.

V. ATLAS VEHICLE SYSTEM PERFORMANCE

VEHICLE STRUCTURE SYSTEM

by Richard T. Barrett

Description

The Atlas structure system consists of two major sections; the propellant tank section and the booster engine section (fig. III-2). The propellant tank section consists of thin-walled, pressure-stabilized, stainless-steel monocoque sections of welded construction. This section is divided by a bulkhead into a fuel (kerosene) tank and an oxidizer (liquid-oxygen) tank. The maximum allowable differential pressure between the oxidizer and fuel tanks is limited by the strength of the intermediate bulkhead. The fuel tank pressure must always be greater than the oxidizer tank pressure to prevent reversal of the intermediate bulkhead. The tank section is 3.048 meters (10 ft) in diameter and 18.5 meters (60.9 ft) in length. The forward bulkhead is ellipsoidal and the aft bulkhead is conical. The sustainer engine is mounted on the thrust cone. Two equipment pods are attached to the sides of the tank.

The booster engine section consists of protective fairings, a thrust structure, and a booster engine with two thrust chambers. The booster engine section is attached to a thrust ring at the aft end of the fuel tank by a latch mechanism which allows the booster engine section to be jettisoned.

Performance

The vehicle structure performance was satisfactory. All measured loads were within the expected limits. The peak longitudinal load factor during flight occurred at booster engine cutoff and was 6.19 g's for Lunar Orbiter III, 6.20 g's for Lunar Orbiter IV, and 6.08 g's for Lunar Orbiter V. The command to actuate the booster release latching mechanism was given at T + 132.9, T + 131.5, and T + 131.7 seconds for Lunar Orbiters III, IV, and V, respectively. In each case the mechanism functioned properly and the booster engine section jettisoned satisfactorily.

PROPULSION SYSTEM

by Charles H. Kerrigan

Description

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

The booster engine, rated at 1468×10^3 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 engine tank system is composed of two small propellants tanks (each approximately 51 cm (20 in.) in diameter) and a pressurization system. This system supplies propellants for starting the engines and slso for vernier engine operation after sustainer engine cutoff.

Performance

The performance of the propulsion systems on the Atlas boosters used for Lunar Orbiter missions III, IV, and V was satisfactory. During the engine start phase, valve opening times and starting sequence events were within tolerances. The flight performance of the engines was evaluated by comparing measured engine parameters with the expected values. These are tabulated in tables V-I, V-II, and V-III. All engine cutoff signals were issued by guidance system commands and were properly executed. Transients at engine shutdown appeared normal.

TABLE V-I. - ATLAS PROPULSION SYSTEM PERFORMANCE, LUNAR ORBITER III

Performance parameters	Units[a]	Expected operating range	Flight values at -			
			T + 10 sec	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Booster engine:						
Number 1 thrust chamber pressure	N/cm^2	386 to 410	392	386	-----	-----
	psi	560 to 595	568	560	-----	-----
Number 2 thrust chamber pressure	N/cm^2	386 to 410	392	386	-----	-----
	psi	560 to 595	568	560	-----	-----
Gas generator chamber pressure	N/cm^2	351 to 382	358	364	-----	-----
	psi	510 to 555	519	528	-----	-----
Number 1 turbopump speed	rpm	6225 to 6405	6267	6277	-----	-----
Number 2 turbopump speed	rpm	6165 to 6345	6166	6096	-----	-----
Sustainer:						
Engine thrust chamber pressure	N/cm^2	469 to 493	487	485	483	-----
	psi	680 to 715	706	704	700	-----
Engine gas generator discharge	N/cm^2	407 to 473	436	441	441	-----
pressure	psi	590 to 686	632	640	640	-----
Engine turbopump speed	rpm	10 025 to 10 445	10 323	10 216	10 414	-----
Vernier:						
Engine number 1 thrust chamber	N/cm^2	172 to 183	174	171	177	-----
pressure when pump supplied	psi	250 to 265	252	248	256	-----
Engine number 1 thrust chamber	N/cm^2	145 to 155	-----	-----	-----	155
pressure when tank supplied	psi	210 to 225	-----	-----	-----	224
Engine number 2 thrust chamber	N/cm^2	172 to 183	174	171	182	-----
pressure when pump supplied	psi	250 to 265	252	248	264	-----
Engine number 2 thrust chamber	N/cm^2	145 to 155	-----	-----	-----	160
pressure when tank supplied	psi	210 to 225	-----	-----	-----	232
Duration of engine burn:						
Booster	sec	129.9	-----	129.8	-----	-----
Sustainer	sec	288.3	-----	-----	288.0	-----
Vernier	sec	308.7	-----	-----	-----	309.2

[a]All pressures are absolute.

TABLE V-II. - ATLAS PROPULSION SYSTEM PERFORMANCE, LUNAR ORBITER IV

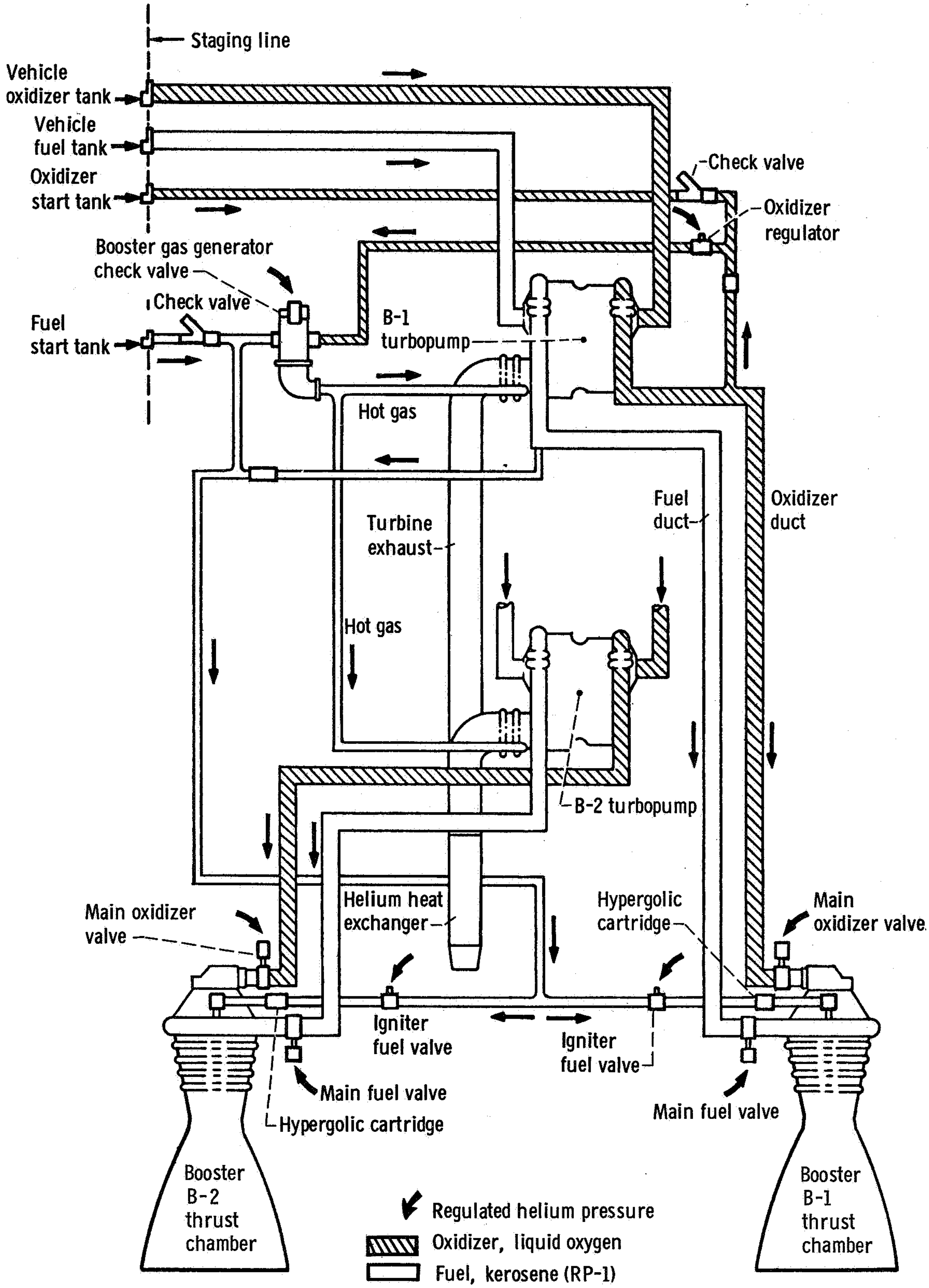
Performance parameters	Units[a]	Expected operating range	Flight values at -			
			T + 10 sec	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Booster engine:						
Number 1 thrust chamber pressure	N/cm^2	386 to 410	395	398	------	------
	psi	560 to 595	573	576	------	------
Number 2 thrust chamber pressure	N/cm^2	386 to 410	395	398	------	------
	psi	560 to 595	572	578	------	------
Gas generator chamber pressure	N/cm^2	351 to 382	360	358	------	------
	psi	510 to 555	527	519	------	------
Number 1 turbopump speed	rpm	6225 to 6405	6277	6287	------	------
Number 2 turbopump speed	rpm	6165 to 6345	6274	6336	------	------
Sustainer:						
Engine thrust chamber pressure	N/cm^2	469 to 493	485	486	(b)	(b)
	psi	680 to 715	703	705	------	------
Engine gas generator discharge pressure	N/cm^2	407 to 473	436	433	439	------
	psi	590 to 686	632	628	636	------
Engine turbopump speed	rpm	10 025 to 10 445	9943	10 011	9913	------
Vernier:						
Engine number 1 thrust chamber pressure when pump supplied	N/cm^2	172 to 183	179	179	182	------
	psi	250 to 265	260	260	264	------
Engine number 1 thrust chamber pressure when tank supplied	N/cm^2	145 to 155	----	------	------	155
	psi	210 to 225	----	------	------	224
Engine number 2 thrust chamber pressure when pump supplied	N/cm^2	172 to 183	171	171	177	------
	psi	250 to 265	248	248	256	------
Engine number 2 thrust chamber pressure when tank supplied	N/cm^2	145 to 155	----	------	------	149
	psi	210 to 225	----	------	------	216
Duration of engine burn:						
Booster	sec	128.44	----	128.3	------	------
Sustainer	sec	290.36	----	------	289.37	------
Vernier	sec	310.6	----	------	------	310.1

[a]All pressures are absolute.
[b]Data dropped out after T + 273 sec.

TABLE V-III. - ATLAS PROPULSION SYSTEM PERFORMANCE, LUNAR ORBITER V

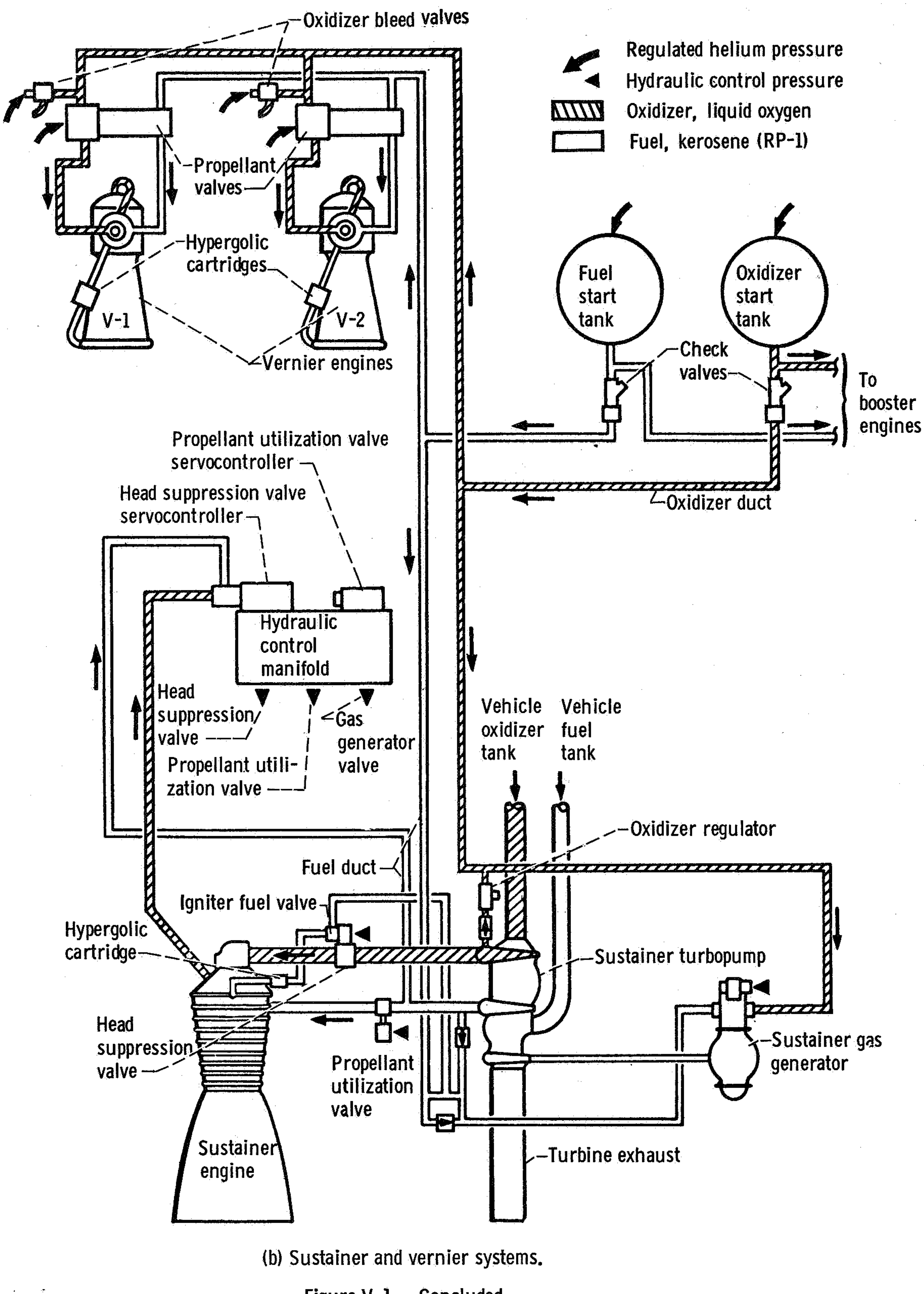
Performance parameters	Units[a]	Expected operating range	Flight values at -			
			T + 10 sec	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Booster engine:						
Number 1 thrust chamber pressure	N/cm^2	386 to 410	392	398	------	------
	psi	560 to 595	568	576	------	------
Number 2 thrust chamber pressure	N/cm^2	386 to 410	398	398	------	------
	psi	560 to 595	576	576	------	------
Gas generator chamber pressure	N/cm^2	351 to 382	358	354	------	------
	psi	510 to 555	519	513	------	------
Number 1 turbopump speed	rpm	6225 to 6405	6295	6285	------	------
Number 2 turbopump speed	rpm	6165 to 6345	6247	6278	------	------
Sustainer:						
Engine thrust chamber pressure	N/cm^2	469 to 493	496	496	489	------
	psi	680 to 715	720	720	710	------
Engine gas generator discharge	N/cm^2	407 to 473	452	452	452	------
pressure	psi	590 to 686	656	656	656	------
Engine turbopump speed	rpm	10 025 to 10 445	10 415	10 335	10 475	------
Vernier:						
Engine number 1 thrust chamber	N/cm^2	172 to 183	182	182	182	------
pressure when pump supplied	psi	250 to 265	264	264	264	------
Engine number 1 thrust chamber	N/cm^2	145 to 155	------	------	------	157
pressure when tank supplied	psi	210 to 225	------	------	------	228
Engine number 2 thrust chamber	N/cm^2	172 to 183	182	182	182	------
pressure when pump supplied	psi	250 to 265	264	264	264	------
Engine number 2 thrust chamber	N/cm^2	145 to 155	------	------	------	157
pressure when tank supplied	psi	210 to 225	------	------	------	228
Duration of engine burn:						
Booster	sec	128.44	------	128.6	------	------
Sustainer	sec	287.94	------	------	288.6	------
Vernier	sec	310.43	------	------	------	307.9

[a]All pressures are absolute.



(a) Booster system.

Figure V-1. - Atlas propulsion system, Lunar Orbiter.



(b) Sustainer and vernier systems.

Figure V-1. - Concluded.

PROPELLANT UTILIZATION SYSTEM

by Clifford H. Arth

Description

The Atlas propellant utilization system is designed to cause near simultaneous depletion of both propellants (see fig. V-2). This system is a digital type, which adjusts the operating mixture ratio of the sustainer engine by sampling the propellant volume ratio at six discrete points during flight. Six fuel and six oxidizer level sensors are positioned in the propellant tanks so that both sensors will uncover simultaneously if the propellants are being consumed at the proper ratio. If the propellant usage ratio is incorrect, one sensor of a pair will uncover before the other sensor. The time difference in the uncovering of the sensors comprising a pair is directly proportional to the propellant usage ratio error. If this time difference is greater than the limit error times for each sensor pair the propellant utilization valve is commanded to the full open or closed position, depending on which sensor uncovers first. If the actual error time is less than the limit error time, the valve would be commanded to something less than full open or closed position; this adjustment would theoretically result in a zero error time when the liquid level reaches the next sensor pair.

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

Performance

Shown in table V-IV are the burnable propellant residuals in the propellant tanks at sustainer engine cutoff for each of the three flights. These residuals would have allowed the sustainer engine to burn the additional times indicated in table V-IV. However, the proper velocity had been attained and the guidance system commanded the engine to shut down. If these flights would have continued to theoretical liquid oxygen depletion, the total fuel remaining would have been as shown in table V-IV.

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 sense port uncovers a time interval is calculated from the instant of port uncovering to sustainer engine cutoff. This time interval is used in determining the propellant residuals.

Table V-V shows the error times between the fuel and liquid oxygen sensor uncovering for all 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-V. - LEVEL SENSOR ERROR TIMES

Sensor number	Limit error time, sec	Lunar Orbiter flight					
		III		IV		V	
		Actual error time, sec	First sensor uncovered	Actual error time, sec	First sensor uncovered	Actual error time, sec	First sensor uncovered
1	1.018	0.9	Liquid oxygen	0.8	Liquid oxygen	0.55	Liquid oxygen
2	.968	.8		.6	Liquid oxygen	.20	Liquid oxygen
3	.78	.2		.05	Liquid oxygen	.50	Liquid oxygen
4	1.89	.3		.10	Fuel	.50	Fuel
5	8.4	2.6		2.5	Fuel	2.80	Fuel
6	4.2	3.5		2.0	Fuel	.50	Fuel

TABLE V-IV. - PROPELLANT UTILIZATION

SYSTEM, PERFORMANCE

	Units	Lunar Orbiter flight		
		III	IV	V
Burnable propellant residuals:				
Fuel	lb	642	645	746
	kg	291	293	338
Liquid oxygen	lb	1127	991	962
	kg	511	449	436
Residual sustainer engine burn time	sec	6.15	5.25	4.9
Remaining fuel at theoretical liquid oxygen depletion	lb	85	205	336
	kg	39	93	152
Fuel and liquid oxygen sensor port uncovering times:				
Fuel	sec	1.15	1.2	0
Liquid oxygen	sec	2.4	3.3	3.4

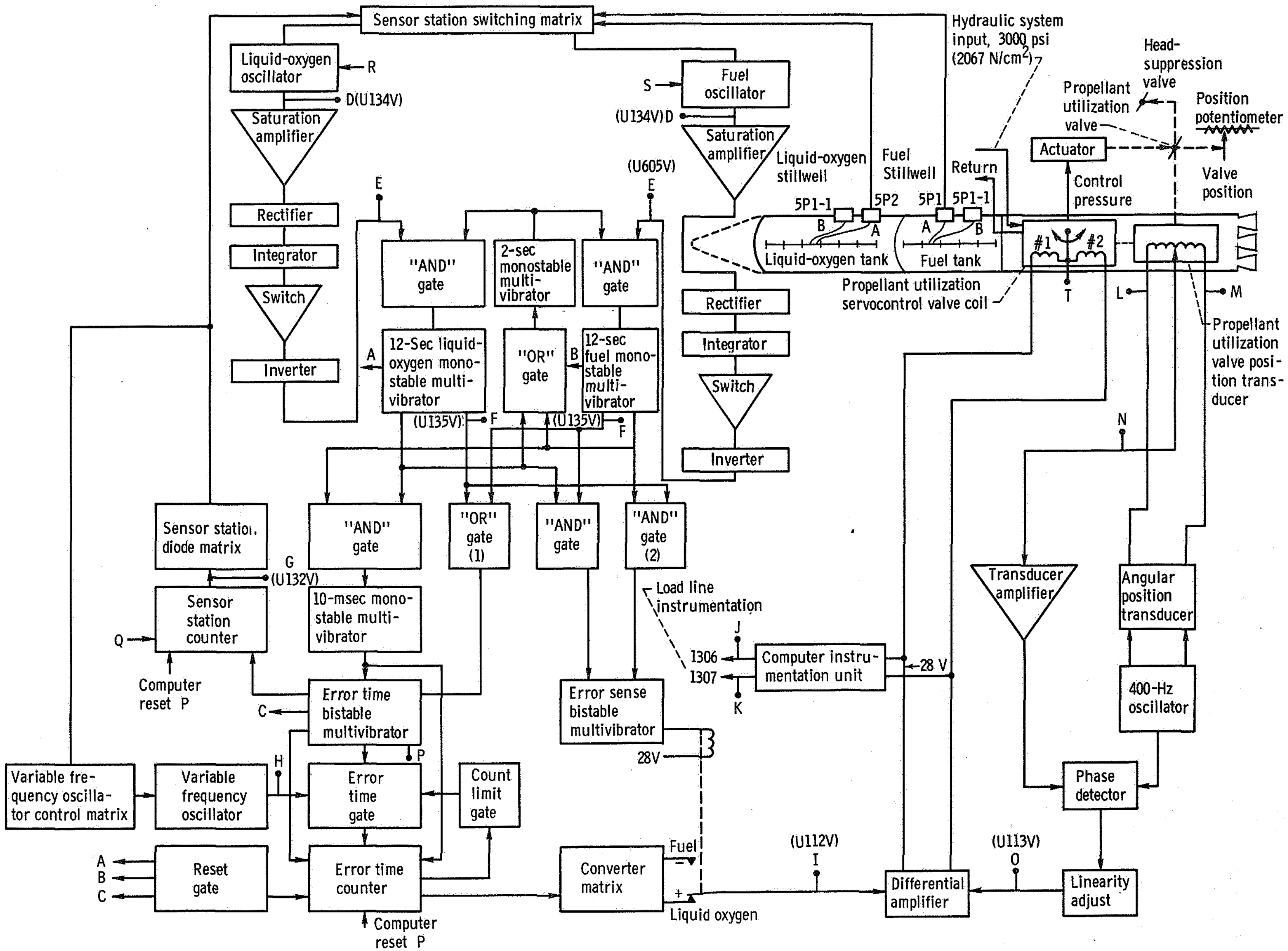


Figure V-2. - Atlas propellant utilization system, Lunar Orbiter.

HYDRAULIC SYSTEM

by Eugene J. Cieslewicz

Description

The Atlas hydraulic system consists of two independent systems: the booster system and the sustainer-vernier system (see 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 stage uses a system similar to the booster but, in addition, provides hydraulic power for sustainer engine propellant utilization and head suppression valves and for gimbaling of the two vernier engines.

The sustainer engine is held in the centered position until booster engine cutoff. Any disturbances created by the engine differential cutoff impulses are damped by gimbaling the sustainer and vernier engines. The sustainer engine is again centered during booster engine package jettison. Vehicle roll control is maintained throughout the sustainer phase by differential gimbaling of the vernier engines. After sustainer engine cutoff, during vernier solo operation, the vernier engine gimbal actuators are provided with hydraulic pressure from two accumulators previously pressurized during sustainer operation. Actuator limit travel of the vernier engines is $\pm 70^{\circ}$, and the sustainer engine is $\pm 3^{\circ}$.

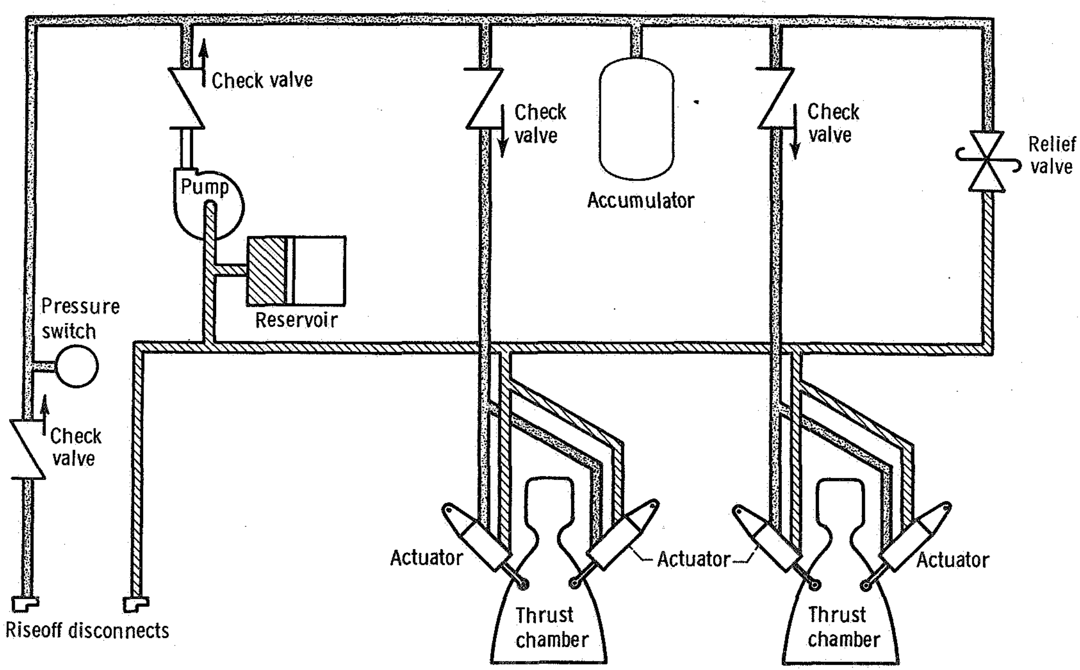
Performance

The hydraulic systems performed properly throughout the three flights. Hydraulic system pressure data for both the booster and sustainer-vernier circuits are shown in table V-VI for Lunar Orbiters III, IV, and V. Transfer of fluid power from ground to airborne systems before lift-off was normal. Starting transients produced a normal overshoot of about 10 percent in the hydraulic pump discharge pressures. Pressures, except for the expected transients at lift-off, booster engine cutoff, and sustainer engine cutoff, were stable throughout the boost flight phase. Gimbaling of the engines was well within the gimbal capabilities and in accordance with the flight control and guidance commands.

TABLE V-VI. - ATLAS HYDRAULIC SYSTEM PERFORMANCE DATA,

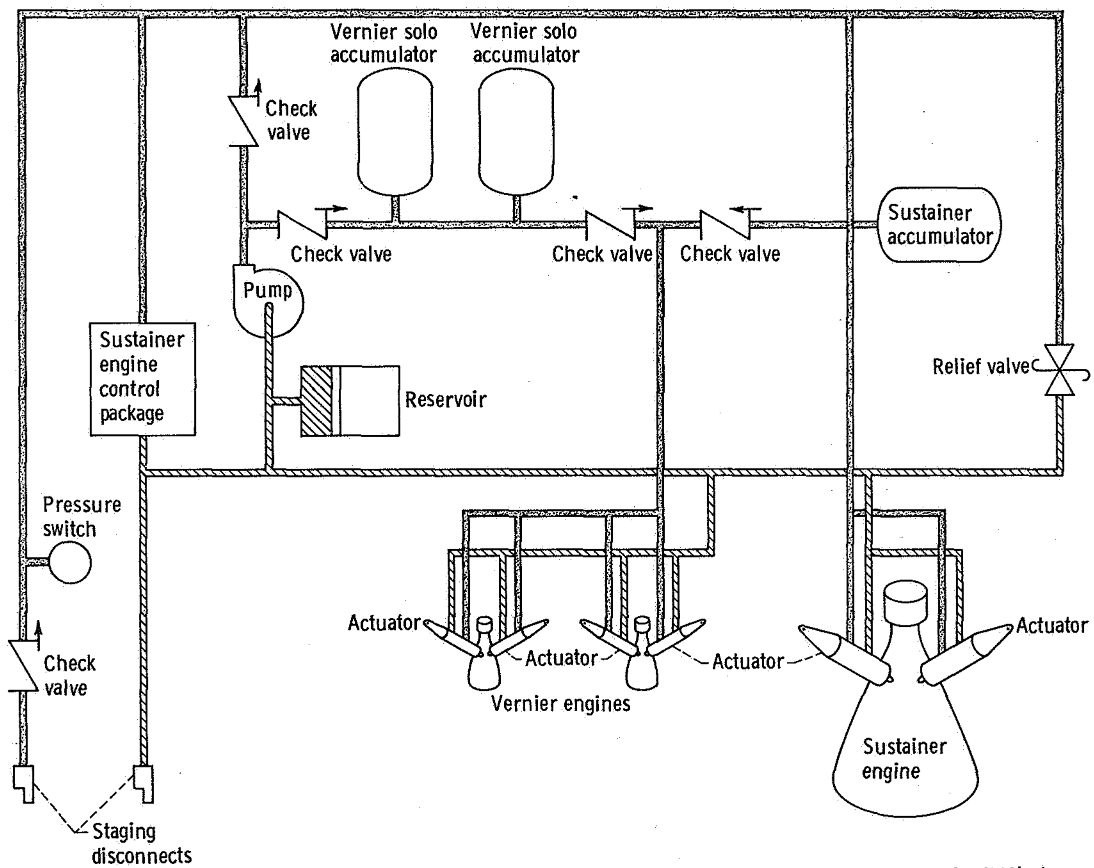
LUNAR ORBITERS III, IV, AND V

Flight	Measurement (absolute)	Units	Lift-off	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
III	Booster pressure	N/cm^2	2172	2148	----	----
		psi	3150	3115	----	----
	Sustainer-vernier pressure	N/cm^2	2124	2124	2124	917
		psi	3080	3080	3080	3080
IV	Booster pressure	N/cm^2	2148	2100	----	----
		psi	3115	3045	----	----
	Sustainer-vernier pressure	N/cm^2	2148	2132	2124	827
		psi	3115	3095	3080	1200
V	Booster pressure	N/cm^2	2148	2075	----	----
		psi	3115	3010	----	----
	Sustainer-vernier pressure	N/cm^2	2196	2144	2124	843
		psi	3185	3110	3080	1223



(a) Atlas booster.

Figure V-3. - Hydraulic system, Lunar Orbiter.



(b) Atlas sustainer.

Figure V-3. - Concluded.

PNEUMATIC SYSTEM

by Eugene J. Fourney

Description

The Atlas pneumatic system 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. This system schematic is shown in figure V-4.

Propellant tank pressurization subsystem. - This subsystem is used to maintain propellant tank pressures at required levels to (1) support the pressure stabilized tank structure, and (2) satisfy the inlet pressure requirements of the engine turbopumps. In addition, helium is supplied from the fuel tank pressurization line to pressurize the hydraulic reservoirs and turbopump lubricant storage tanks. The subsystem consists of six shrouded 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 provide a maximum storage capacity at 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. The propellant tank pressurization subsystem control 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 staging. Thereafter the fuel tank pressure decays slowly. The oxidizer tank pressure is augmented by liquid oxygen boil-off and the decay in this tank pressure is small.

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 controlled oxidizer flow to the gas generator. Control pressure in this subsystem is maintained through Atlas-Agena separation. These pneumatic requirements are supplied from a 76 200-cubic-centimeter (4650 in.^{3}) storage bottle pressurized to a gage 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 charged to a gage pressure of 2068 newtons per square centimeter (3000 psi).

Performance

The pneumatic system performance was satisfactory during the three flights. All tank pressures were satisfactory and all control functions were performed properly. Pneumatic system parameters are compared with measured values during the three flights in tables V-VII, V-VIII, and V-IX. Liquid oxygen tank ullage pressure oscillations were as experienced on similar Atlas vehicles. Prior to lift-off, oscillation frequencies of approximately 3.25 hertz were measured. The oscillation amplitudes as measured by the differential pressure across the bulkhead varied with a maximum peak-to-peak amplitude of approximately 2.07 newtons per square centimeter (3.0 psi). After lift-off, these oscillations increased in frequency to approximately 5.25 hertz and increased in amplitude slightly. These oscillations damped out within 26 seconds as has been experienced on other, similar Atlas vehicles.

During dual propellant loading (DPL) of Atlas 5803 (Lunar Orbiter III) unusual vibrations were noted on various spacecraft accelerometers; however, these did not exceed allowable spacecraft vibration levels. These vibrations occurred twice during liquid oxygen loading and existed for approximately 1 minute each. These vibrations were related to a "fluttering" condition of the boiloff valve. In a normal pressurization sequence, the boiloff valve relieves at 4.0 newtons per square centimeter (5.3 psi) and reseats at 3.24 newtons per square centimeter (4.7 psi) and the boiloff valve cycles opened and closed at the approximate rate of 1 hertz every 3 seconds. During the two above noted vibration periods, the liquid oxygen tank ullage pressure remained at approximately 3.59 newtons per square centimeter (5.2 psi) and the valve fluttered at the rate of approximately 25 to 26 hertz allowing the ullage tank pressure to remain fairly constant.

The acoustical resonance of the Atlas liquid oxygen tank changes as the tank is filled with liquid oxygen. Analysis indicated that the second harmonic of the liquid oxygen tank acoustic resonance at 30 percent liquid oxygen load is approximately 25 hertz, the first harmonic of the tank acoustic resonance at 70 percent load is approximately 25 hertz, and at 95 percent load the fundamental frequency of the tank acoustic resonance is also approximately 25 hertz. The vibration on Atlas 5803 (Lunar Orbiter III) occurred twice during dual propellant loading at approximately 30 percent and at 70 percent liquid oxygen load.

It was determined that the boiloff valve should be cycled to the locked position at the onset of "flutter." This immediately stops the "flutter." The valve is then unlocked after 2 seconds. Should "flutter" reappear the procedure is repeated as often as required. This procedure was incorporated into all Atlas tanking and launch countdown procedures to prevent this boiloff valve vibration.

TABLE V-VII. - ATLAS PNEUMATIC SYSTEM PERFORMANCE, LUNAR ORBITER III

Parameter	Measurement number	Units	Design range	Flight values at -					Remarks
				T - 10 sec	T - 0	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff	
Oxidizer tank ullage pressure (gage)	AF1P	N/cm^2 psi	19.65 to 21.37 28.5 to 31.0	21.72 31.5	20.89 30.3	19.86 28.8	[a]18.20 26.4	18.20 26.4	
Fuel tank ullage pressure (gage)	AF3P	N/cm^2 psi	44.13 to 46.20 64.0 to 67.0	45.02 65.3	45.02 65.3	44.68 64.8	[a]33.03 47.9	33.03 47.9	
Intermediate bulkhead differential pressure	AF116P	N/cm^2 psi	0.35 (min.) .5 (min.)	----- -----	11.03 16.0	----- -----	----- -----	----- -----	The lowest ΔP was 6.89 N/cm^2 (10 psi) at approximately T + 1 sec
Sustainer controls bottle pressure (absolute)	AF291P	N/cm^2 psi	2344 (max.) 3400 (max.)	2117 3070	2065 2995	1793 2600	1734 2515	1110 1610	
Booster helium bottle pressure (absolute)	AF246P	N/cm^2 psi	2344 (max.) 3400 (max.)	2124 3080	2079 3015	434 630	----- -----	----- -----	
Booster helium bottle temperature, deg	AF247T	K ^{o}F	------------- -------------	77.59 -320	74.59 -323	42.67 -385	----- -----	----- -----	

[a]Helium supply bottles are jettisoned with the booster package at booster engine cutoff + 3 sec; therefore, regulator design range is no longer a criterion.

TABLE V-VIII. - ATLAS PNEUMATIC SYSTEM PERFORMANCE, LUNAR ORBITER IV

Parameter	Measurement number	Units	Design range	Flight values at -					Remarks
				T - 10 sec	T - 0	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff	
Oxidizer tank ullage pressure (gage)	AF1P	N/cm^2 psi	19.65 to 21.37 28.5 to 31.0	21.58 31.3	20.2 29.3	20.62 29.9	[a]18.82 27.3	18.82 27.3	
Fuel tank ullage pressure (gage)	AF3P	N/cm^2 psi	44.13 to 46.20 64.0 to 67.0	45.02 65.3	45.02 65.3	45.36 65.8	[a]32.96 47.8	32.96 47.8	
Intermediate bulkhead differential pressure	AF116P	N/cm^2 psi	0.35 (min.) .5 (min.)	----- -----	11.38 16.5	----- -----	----- -----	----- -----	The lowest value was 7.93 N/cm^2 (11.5 psi) at T + 1 sec
Sustainer controls bottle pressure (absolute)	AF291P	N/cm^2 psi	2344 (max.) 3400 (max.)	2124 3080	2075 3010	1731 2510	1689 2450	1062 1540	
Booster helium bottle pressure (absolute)	AF246P	N/cm^2 psi	2344 (max.) 3400 (max.)	2124 3080	2055 2980	434 630	(a) -----	----- -----	
Booster helium bottle temperature, deg	AF247T	K $^{\circ}F$	------------- -------------	77.59 -320	75.59 -322	43.15 -382	(a) -----	----- -----	

[a]Helium supply bottles are jettisoned with the booster package at booster engine cutoff + 3 sec; therefore, regulator design range is no longer a criterion.

TABLE V-IX. - ATLAS PNEUMATIC SYSTEM PERFORMANCE, LUNAR ORBITER V

Parameter	Measurement number	Units	Design range	Flight values at -					Remarks
				T - 10 sec	T - 0	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff	
Oxidizer tank ullage pressure (gage)	AF1P	N/cm^2 psi	19.65 to 21.37 28.5 to 31.0	21.58 31.3	21.12 30.8	20.13 29.2	[a]19.79 28.7	19.79 28.7	
Fuel tank ullage pressure (gage)	AF3P	N/cm^2 psi	44.13 to 46.20 64.0 to 67.0	46.20 67.0	46.20 67.0	45.98 66.7	[a]33.58 48.7	33.58 48.7	
Intermediate bulkhead differential pressure	AF116P	N/cm^2 psi	0.35 (min.) .5 (min.)	11.93 17.3	10.68 15.5	12.89 18.7	13.78 20.0	13.78 20.0	The lowest value was 5.17 N/cm^2 (7.5 psi) at T + 0.7 sec
Sustainer controls bottle pressure (absolute)	AF291P	N/cm^2 psi	2344 (max.) 3400 (max.)	2124 3080	2068 3000	1917 2780	1737 2520	1131 1640	
Booster helium bottle pressure (absolute)	AF246P	N/cm^2 psi	2344 (max.) 3400 (max.)	2124 3080	2068 3000	459 665	(a) -----	----- -----	
Booster helium bottle temperature, deg	AF247T	K °F	------------- -------------	79.08 -317.5	77.8 -319	43.15 -382	(a) -----	----- -----	

[a]Helium supply bottles are jettisoned with the booster package at booster engine cutoff + 3 sec; therefore, regulator design range is no longer a criterion.

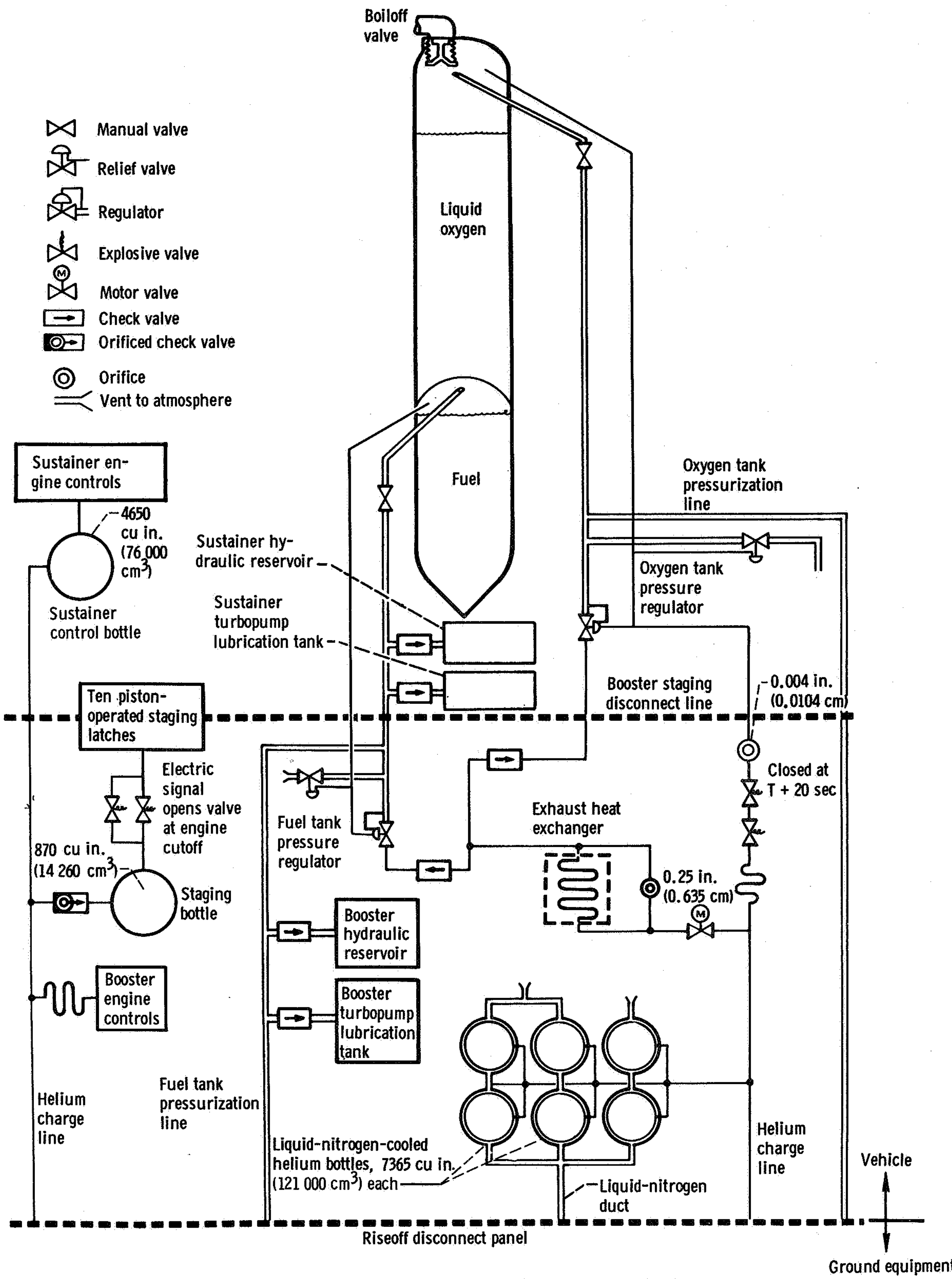


Figure V-4. - Atlas vehicle pneumatic system, Lunar Orbiter.

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 seconds 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 phase of flight 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 vehicle flight control system to provide corrections for vehicle deviations from the programmed trajectory.

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

The Atlas flight control system figure V-5 consists of the following major components mounted to the vehicle airframe:

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

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

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

The Mod III Radio Guidance System includes the Atlas vehicleborne pulse beacon, rate beacon, and decoder; and a ground station comprised of a monopulse X-band (radar) track subsystem, a continuous wave L-band rate subsystem, and a digital guidance computer subsystem. The major functions (fig. V-6) are more fully 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 discrete commands. When the address code of the received signal is correct, three events occur. The vehicleborne pulse beacon transmits a return pulse to the ground station and the vehicleborne decoder receives and decodes the message, and issues the pitch and yaw steering signals and discrete commands to the flight control system.

The ground rate subsystem transmits two continuous wave signals of different frequencies from a single ground antenna. The vehicleborne rate beacon is interrogated by the signals from the ground subsystem. The rate beacon transmits a continuous wave signal at a frequency equal to the arithmetic average of the frequencies of the received signals. This signal is received by ground based stations. The two-way doppler shift and phase relation of the signals as received at three separate 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 subsystem is sent to the ground computer. The guidance ground computer, solves the guidance equations every 1/2 second using the position and rate information. This results in transmission of steering and discrete commands from the computer to the ground tract station and then to the vehicle and its control systems.

Acquisition of the vehicle by the ground antenna is accomplished by one of three methods; cube procedure, or optical tracking acquisition aid, or by slaving to range azimuth and elevation data supplied by the Eastern Test Range. In the acquisition cube procedure, which is the primary method of acquisition, the ground conical scan antenna is directed to one of seven predetermined positions along the programmed trajectory. These positions represent cubes defined by range, azimuth, and elevation. The conical scan antenna is on the same mount as the X-band track subsystem antenna. Once the vehicle is acquired by the conical scan antenna, tracking is automatically switched to the main track antenna. The rate subsystem antennas are slaved to the track subsystem antennas.

Performance

The performance of the Atlas flight control was satisfactory for all three missions. Lift-off transients in pitch, yaw, and roll were within acceptable limits. The maximum vehicle displacement during the lift-off transients are given in the following table:

Vehicle	Maximum peak-to-peak amplitude, deg		
	Roll	Pitch	Yaw
Lunar Orbiter III	0.37 CCW[a]	0	0.17 right
Lunar Orbiter IV	.72 CCW[a]	.15 up	.38 right
Lunar Orbiter V	.47 CCW[a]	.26 up	.25 right

[a]Counterclockwise (CCW) roll reference applies when looking downrange along the Atlas longitudinal axis.

The roll program was initiated at T + 2 seconds and discontinued at T + 15 seconds as planned. The following table shows the programmed vehicle roll required to achieve the desired launch azimuth, and also the actual vehicle roll:

Vehicle	Programmed vehicle roll, deg	Actual vehicle roll[a], deg
Lunar Orbiter III	23.57	23.8
Lunar Orbiter IV	3.58	3.13
Lunar Orbiter V	.42	.32

[a]Roll rate gyro data.

The required pitch program was the same for all three vehicles. 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-X. The actual pitch maneuver dispersions were within acceptable limits.

Dynamic disturbances during the period of maximum dynamic pressure were small as indicated in the following table. The gimbal angles were within the maximum gimbal angle predictions based on atmospheric data (wind soundings) taken at T - 0. (See IV. TRAJECTORY AND PERFORMANCE section.)

Vehicle	Time of maximum engine gimbal angle, sec	Maximum booster engine gimbal angle, sec		Booster engine gimbal capacity, percent	
		Pitch	Yaw	Pitch	Yaw
Lunar Orbiter III	T + 68.3	1.93	0.97	38.6	19.4
Lunar Orbiter IV	T + 73.3	.95	.7	19	14
Lunar Orbiter V	T + 79.8	1.06	.45	21.2	9

Vehicle displacements resulting from booster engine shutdown were within acceptable limits and are shown as follows:

Vehicle	Vehicle displacement at booster engine shutdown, deg	
	Pitch	Yaw
Lunar Orbiter III	0.13 up	0.34 left
Lunar Orbiter IV	.08 up	.83 left
Lunar Orbiter V	.14 up	.42 left

The booster engine jettison sequence was normal through each of the lunar flights and the resulting small pitch and yaw transients were quickly damped. The sustainer engine in each case then returned the vehicle to the correct attitude.

After completion of sustainer burn and vernier solo the command to separate Agena was initiated on each of the three flights by radio guidance. At the time, each of the vehicles was stable in attitude, and separation was successfully completed. One anomaly occurred on Lunar Orbiter III; the "initial separation sequence" backup programmer function could not be observed on telemetry due to the generation of a guidance discrete at the same time. This condition was corrected for the Lunar Orbiter IV and V flights.

The performance of both the ground station and vehicleborne Mod III radio guidance equipment for all three missions was satisfactory. The performance of the track subsystem is summarized in the following table:

Event	Lunar Orbiter flight		
	III	IV	V
Acquisition mode Time	Acquisition Cube T + 62.9 sec in first cube	Acquisition Cube T + 60.7 sec in first cube	Acquisition Cube T + 60.1 sec in first cube
Automatic switch to monopulse tracking, sec	T + 68.0	T + 65.9	T + 65.5
Computer acquisition, sec	T + 71.1	T + 69.2	T + 68.8
Initial loss of lock, sec	T + 374.5	T + 376.3	T + 442.7
Final loss of lock, at elevation:			
sec	T + 375.7	T + 395.2	T + 442.1
deg	4.59	3.60	1.3

For Lunar Orbiters III and V, the signal received by the track subsystem was within 5 decibels of predicted; for Lunar Orbiter IV the signal received was within 3 decibels of predicted.

Significant performance data of the rate subsystems are shown in the following table:

Event	Lunar Orbiter flight		
	III	IV	V
Initial lock at all antennas, sec	T + 53.9	T + 58.2	T + 57.5
Computer acquisition, sec	T + 55.1	T + 58.5	T + 61.8
Initial loss of lock[a], sec	T + 374.8	T + 374.1	T + 419.2
Final loss of lock at elevation:			
sec	T + 375.7	T + 395.2	T + 434.6
deg	4.59	3.60	----
Maximum deviation of actual from theoretical signal strength, dB:			
Central rate antenna	4	5	5
Rate leg antennas	2	3	3

[a]Booster staging resulted in normal momentary loss of lock.

'he ground computer subsystem performance was satisfactory for each vehicle . A postflight simulation of these flights successfully verified the guidance program ıdicated that no transient error occurred during these flights.

[The performance of the airborne pulse beacon was satisfactory for each flight. Significant performance data from the pulse beacon automatic gain control (AGC) monitors as observed from Atlas telemetry are shown in the following table:

Event	Lunar Orbiter flight					
	III		IV		V	
	Time, sec	Signal strength, dB	Time, sec	Signal strength, dB	Time, sec	Signal strength, dB
Acquisition	T + 63.2	-43	T + 58.8	-51	T + 58.9	-52
Switch to automatic track	T + 68	---	T + 66	-23	T + 65.7	-33
Maximum signal strength	T + 77	-8	T + 75.4	-8	T + 77.3	-10
Atlas-Agena separation	T + 313.6	-33	T + 314.6	-28	T + 312.4	-34
Final loss of signal	T + 374.5	-70	T + 373.5	-77	T + 442.5	-79
Significant periods of missing pulses due to antenna look angles	None		T + 86.3 to T + 87.3		T + 123 to T + 125 T + 128.3 to T + 128.8	

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 on each flight.

Peformance of the airborne rate beacon was satisfactory for each flight. Significant performance data from the rate beacon automatic gain control (AGC) monitors as observed from Atlas telemetry are shown in the following table:

Event	Lunar Orbiter flight		
	III	IV	V
	Time, sec		
Initial acquisition	-------	T + 40.5	T + 38.5
Signal strength reaches -75 dBm	T + 56.5	T + 57	T + 61.7
Signal strength drops below -75 dBm	T + 360	T + 369.5	T + 419
Signal strength drops to threshold (-85 dBm)	T + 376	T + 374.5	T + 442.5
Overall period of good reception and transmission of data[a]	T + 56.5 to T + 376	T + 57 to T + 374.5	T + 49.5 to T + 419

[a]Based on interpretation of rate beacon phase detector and power output monitors.

The steering and discrete commands transmitted from the ground station were properly received and processed by the airborne decoder. Spurious pitch and yaw commands were observed on each flight during the period of intermittent pulse beacon lock in the first cube acquisition period. The time of occurrence and peak spurious command (percent of maximum) for each flight are shown in the following table:

	Lunar Orbiter flight		
	III	IV	V
Time period, sec	T + 52 to T + 63	T + 50.8 to T + 58.7	T + 51.4 to T + 59.6
Peak spurious command, percent	±24	±15	±17

These spurious commands, which are normally encountered during acquisition, were properly inhibited by the airborne flight control system.

Mod III radio guidance steering was enabled within the airborne flight control system at T + 80 seconds on each flight. As noted in the Description section only pitch steering commands may be generated by the Mod III radio guidance system during the booster phase. However, the pitch steering commands are constrained by the ground to the time period between T + 100 seconds and T + 110 seconds. No steering commands were transmitted in these flights during the booster phase since the flight path deviations in the pitch plane were within the ground computer threshold limits of implementation.

During the sustainer and vernier phases of flight, pitch and yaw steering signals may be provided. A summary of these signals for each flight is shown in the following paragraphs.

For Lunar Orbiter III, sustainer steering began at T + 138.2 second. The maximum steering command outputs from the decoder were a yaw-left command of 45 percent of maximum[1] at T + 139 seconds and a pitch-up command of 100 percent at T + 140 seconds. The pitch-up command at T + 140 seconds was followed by a pitch-down command of 12 percent at T + 145 seconds. Pitch and yaw steering commands from the decoder were within ±10 percent by T + 159 seconds, and remained within these limits until sustainer engine cutoff.

During the vernier phase, the largest steering command outputs from the decoder were a 28-percent pitch-down command, which was reduced to within ±10 percent within 1.0 second, and a 60-percent yaw-right command that was reduced to within ±10 percent within 1.5 seconds.

For Lunar Orbiter IV, sustainer steering began at T + 136.9 seconds. The maximum steering command outputs from the decoder were a pitch-up command of 71 percent of maximum and a yaw-left command of 60 percent at T + 138 seconds, followed by a pitch-down command of 9 percent at T + 143 seconds, and a yaw-right command of 17 percent at T + 147.5 seconds. Pitch and yaw steering commands from the decoder were reduced to within ±10 percent of maximum steering by T + 157 seconds, and remained within these limits until sustainer engine cutoff, except for a pitch-down command of 10 percent at T + 260 seconds.

During the vernier phase, the largest steering command outputs from a 13-percent pitch-up command from the decoder which was reduced to within ±10 percent within 1.0 second, and a 42-percent yaw-right command that was reduced to within ±10 percent within 1.5 seconds.

For Lunar Orbiter V, sustainer steering began at T + 137.6 seconds. The maximum steering command outputs from the decoder were a pitch-up command of 92 percent of maximum at T + 139.6 seconds followed by a pitch-down command of 9 percent at

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[1]See IV. <u>TRAJECTORY AND PERFORMANCE</u> for correlation of percentages with absolute values of steering commands.

T + 144.5 seconds, and a yaw-left command of 57 percent at T + 138.8 seconds followed by a yaw-right command of 19 percent at T + 144.5 seconds. Both pitch and yaw commands from the decoder were reduced to within ±10 percent by T + 148 seconds and remained within these limits until sustainer engine cutoff.

During the vernier phase, the largest steering command outputs from the decoder were a 9-percent pitch-down command and an 18-percent yaw-right command from the decoder at T + 289 seconds, both of which were reduced to within ±10 percent of maximum steering command in 0.6 second.

Tables V-XI, V-XII, and V-XIII give the times at which the actual Booster Engine Cutoff (BECO). Start Restart Timer (SRT), Sustainer Engine Cutoff (SECO), Start Agena Timer (SAT), Vernier Engine Cutoff (VECO), Shroud Separation, and Atlas-Agena Separation discretes were generated by the ground guidance station computer for Lunar Orbiters III, IV, and V.

TABLE V-X. - ATLAS PITCH PROGRAM, LUNAR ORBITERS III, IV, AND V

Time interval, sec				Step level, deg/sec			
Programmed	Actual			Programmed	Actual		
	Lunar Orbiter flight				Lunar Orbiter flight		
	III	IV	V		III	IV	V
0 to 15	0 to 15	0 to 15.5	0 to 15	0	0	0	0
15 to 35	15 to 35	15.5 to 36	15 to 35	1.018	1.000	1.004	.998
35 to 45	35 to 45	36 to 45.3	35 to 45	.848	.835	.837	.805
45 to 58	45 to 58	45.3 to 58	45 to 58	.509	.510	.585	.483
58 to 70	58 to 70	58 to 70	58 to 70	.678	.665	.669	.676
70 to 82	70 to 82	70 to 83	70 to 82	.806	.805	(a)	.805
82 to 91	82 to 91	83 to 92.2	82 to 91	.678	.670	.669	.644
91 to 105	91 to 105	92.2 to 105.4	91 to 105	.551	.570	.502	.515
105 to 120	105 to 120	105.4 to 121.2	105 to 120	.382	.370	.334	.354
120 to 138.2	120 to (b)	121.2 to (b)	120 to (b)	.254	.270	(c)	.241
138.2 to sustainer engine cutoff	(b) to sustainer engine cutoff	(b) to sustainer engine cutoff	(b) to sustainer engine cutoff	.042	(c)	(c)	(c)

[a]Data during period of maximum dynamic pressure could not be read.

[b]Time cannot be determined.

[c]Data not valid since the values were less than the telemetry accuracy tolerance.

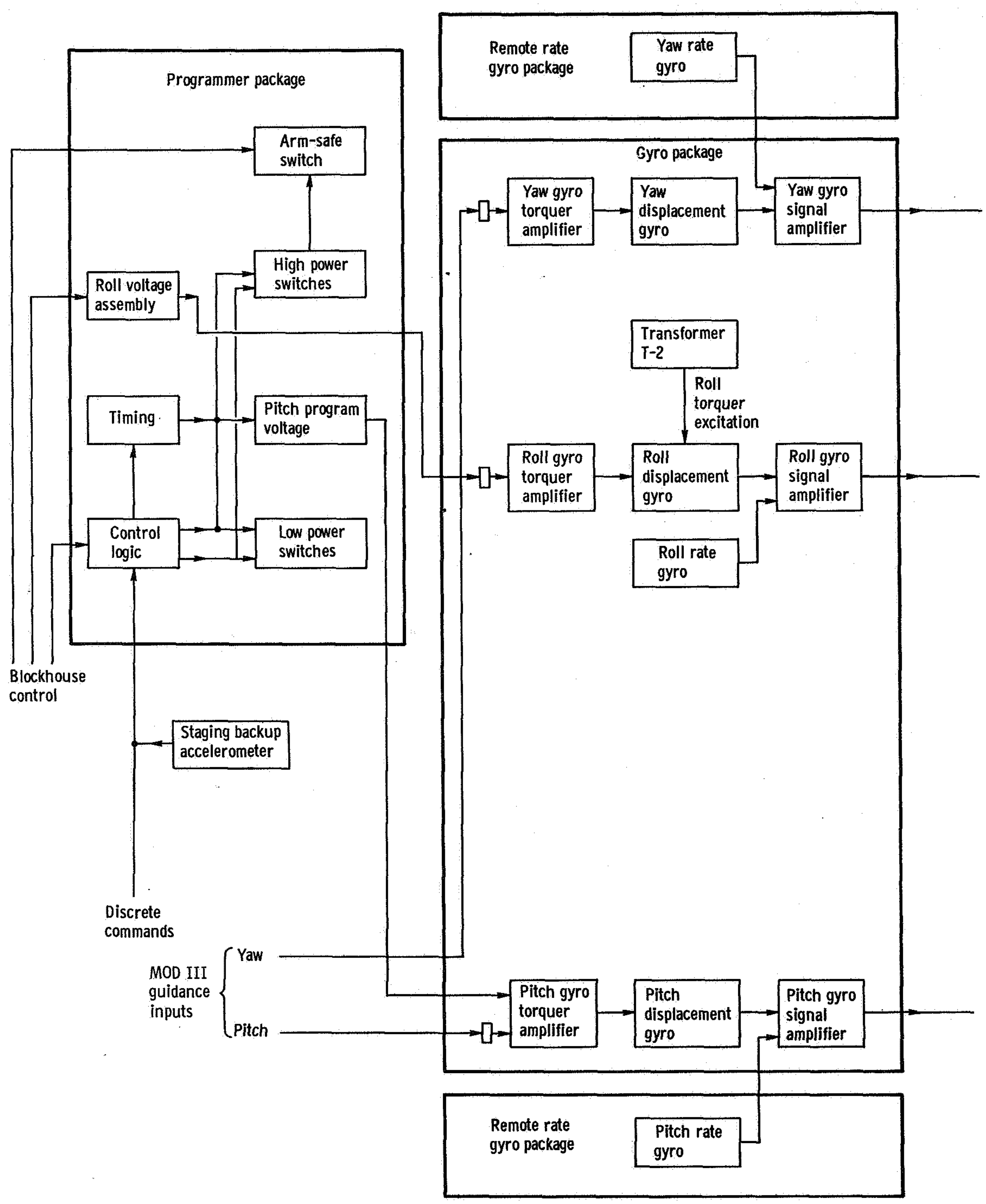
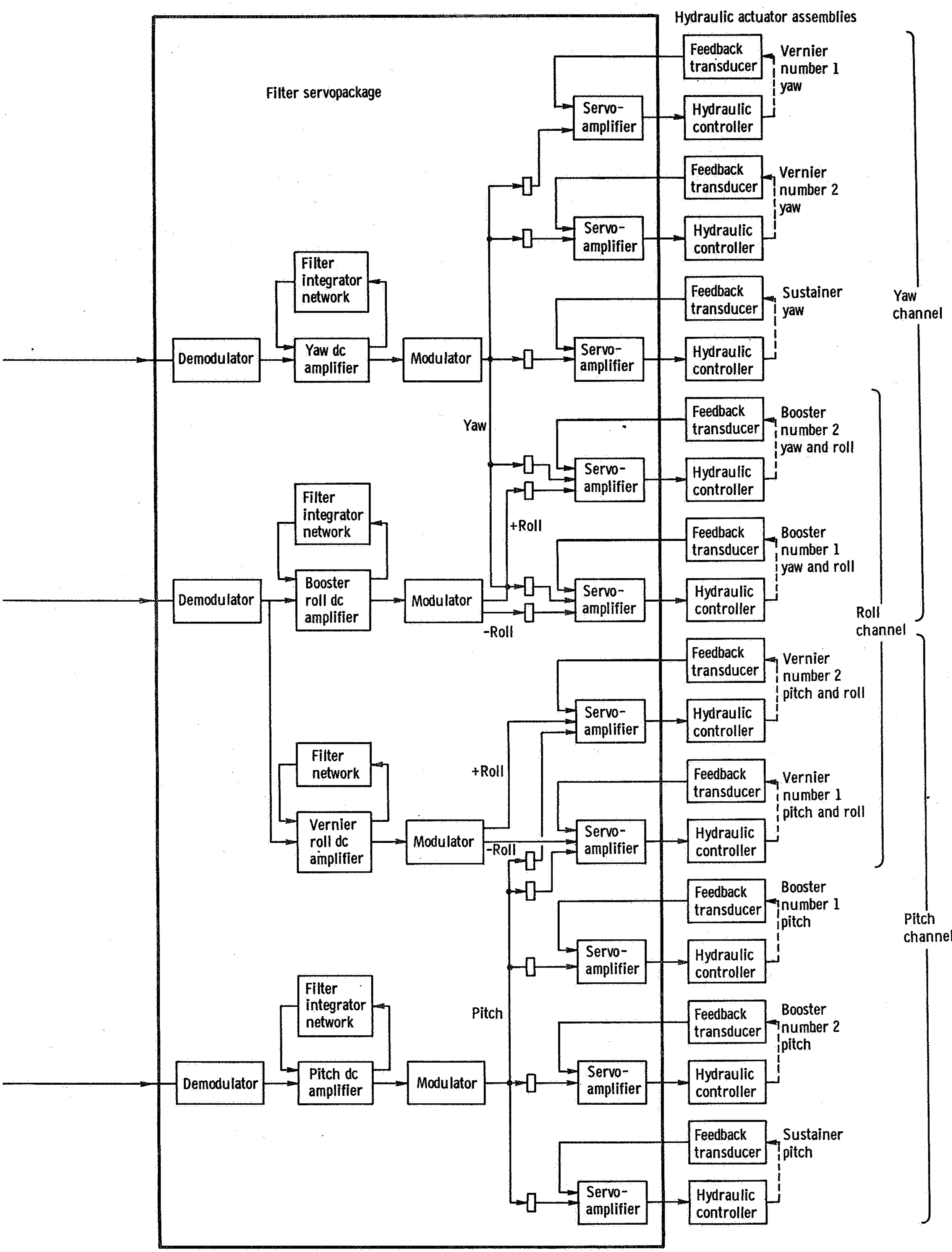


Figure V-5. - Atlas flight



ystem block diagram, Lunar Orbiter.

TABLE V-XI. - MOD III RADIO GUIDANCE DISCRETE TIMES AND COORDINATES,

LUNAR ORBITER III

Flight event and trajectory function	Units	Actual discrete generation time after lift-off, sec	Vehicle coordinates at time of discrete	Vehicle velocity at time of discrete	Discrete duration, sec
Booster engine cutoff:		129.378			0.497
Range	m		100 807		
	ft		330 731		
Azimuth	deg		78.865		
Elevation	deg		30.231		
Range rate	m/sec			2 519	
	ft/sec			8 265	
Start restart timer		270.478			0.897
Sustainer engine cutoff:		287.881			0.993
Range	m		672 628		
	ft		2 206 785		
Azimuth	deg		80.744		
Elevation	deg		9.901		
Range rate	m/sec			5 127	
	ft/sec			16 820	
Start Agena timer		297.325			0.550
Vernier engine cutoff:		309.151			0.723
Range	m		782 757		
	ft		2 568 099		
Azimuth	deg		80.778		
Elevation	deg		8.485		
Range rate	m/sec			5 145	
				16 820	
Shroud separation		311.378			0.497
Atlas-Agena Separation		313.378			To end

TABLE V-XII. - MOD III RADIO GUIDANCE DISCRETE TIMES AND COORDINATES,

LUNAR ORBITER IV

Flight event and trajectory function	Units	Actual discrete generation time after lift-off, sec	Vehicle coordinates at time of discrete	Vehicle velocity at time of discrete	Discrete duration, sec
Booster engine cutoff:		127.957			0.497
Range	m		97 116		
	ft		318 623		
Azimuth	deg		98.317		
Elevation	deg		31.990		
Range rate	m/sec			2 479	
	ft/sec			8 133	
Start restart timer		269.010			0.943
Sustainer engine cutoff:		289.277			0.667
Range	m		673 278		
	ft		2 208 917		
Azimuth	deg		101.006		
Elevation	deg		10.189		
Range rate	m/sec			5 154	
	ft/sec			16 911	
Start Agena timer		291.897			0.557
Vernier engine cutoff:		310.090			0.863
Range	m		781 072		
	ft		2 562 572		
Azimuth	deg		101.006		
Elevation	deg		8.691		
Range rate	m/sec			5 148	
	ft/sec			16 891	
Shroud separation		312.457			0.497
Atlas-Agena separation		314.457			To end

TABLE V-XIII. - MOD III RADIO GUIDANCE DISCRETE TIMES AND COORDINATES,

LUNAR ORBITER V

Flight event and trajectory function	Units	Actual discrete generation time after lift-off, sec	Vehicle coordinates at time of discrete	Vehicle velocity at time of discrete	Discrete duration, sec
Booster engine cutoff:		128.253			-----
Range	m		96 287		
	ft		315 901		
Azimuth	deg		102.204		
Elevation	deg		31.748		
Range rate	m/sec			2 458	
	ft/sec			8 063	
Start restart timer		272.340			-----
Sustainer engine cutoff:		288.500			-----
Range	m		670 236		
	ft		2 198 938		
Azimuth	deg		105.052		
Elevation	deg		9.988		
Range rate	m/sec			5 176	
	ft/sec			16 981	
Start Agena timer		296.253			-----
Vernier engine cutoff:		307.880			-----
Range	m		768 558		
	ft		2 521 515		
Azimuth	deg		105.056		
Elevation	deg		8.663		
Range rate	m/sec			5 166	
	ft/sec			16 950	
Shroud separation		310.253			-----
Atlas-Agena separation		312.253			-----

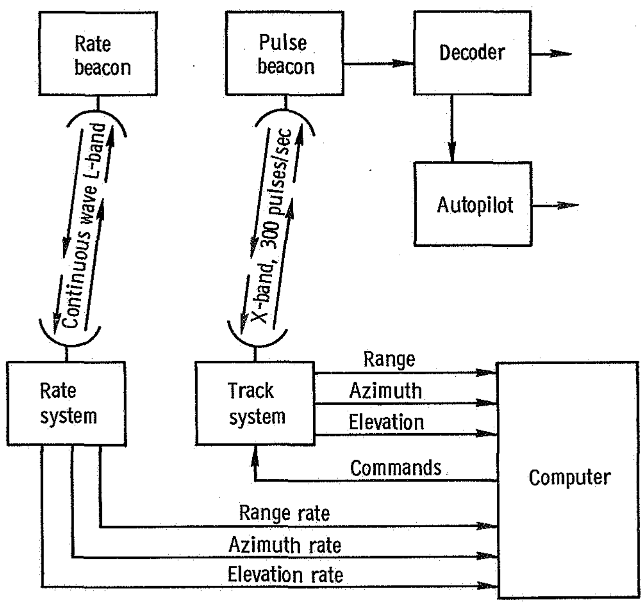


Figure V-6. - MOD III Guidance System block diagram.

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 V dc manually activated batteries; a 115 V ac, 3 phase, 400 hertz inverter; a power changeover switch; a distribution box; two junction boxes; and related electrical harnesses. The main missile 28 V dc battery supplies power to the flight control system, the airborne radio guidance system, the propellant utilization system, the propulsion system, and the inverter. One other 28 V dc battery supplies power to the telemetry system and the remaining two 28 V dc batteries supply power to the flight termination system. The inverter supplies power to the flight control, the propellant utilization, and the airborne radio guidance systems. Phase A of the inverter is used as 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. The maximum inverter frequency deviation was 2.0 hertz at booster engine cutoff for Lunar Orbiter III as shown in table V-XIV. Table V-XIV also shows a maximum deviation of 0.5 V ac (phase A) from the nominal 115 V ac occurring with Lunar Orbiter IV and a maximum battery voltage deviation of 0.3 V dc from the nominal of 28 V dc with Lunar Orbiter III.

A low-amplitude voltage oscillation was observed on the Lunar Orbiter flights. These oscillations were approximately 0.6 V peak-to-peak at a frequency of approximately 20 hertz on the main vehicle dc bus. The time during which they occurred was about T + 310 seconds until T + 445 seconds. The oscillations have been attributed to an instability inherent in the design of the inverter frequency controller and the voltage regulator.

TABLE V-XIV. - ELECTRICAL SYSTEM PERFORMANCE, LUNAR ORBITERS III, IV, AND V

	Tolerance	Lift-off	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff	Atlas-Agena separation
Lunar Orbiter III						
Main battery voltage, V dc	$28^{+2.0}_{-1.5}$	27.8	27.7	27.8	28.0	28.0
Inverter frequency, Hz	400±6	398.0	398.0	398.5	398.5	398.5
Phase A voltage, V ac	115±0.5	115.4	115.3	115.0	115.0	115.0
Phase B voltage, V ac	115±1.7	115.9	115.6	115.4	115.4	115.4
Phase C voltage, V ac	115±1.7	115.8	115.6	115.5	115.5	115.5
Lunar Orbiter IV						
Main battery voltage, V dc	$28^{+2.0}_{-1.5}$	28.1	28.0	28.0	28.1	28.1
Inverter frequency, Hz	400±6	399.5	400.1	400.7	400.7	400.7
Phase A voltage, V ac	115±0.5	115.5	114.9	114.7	114.7	114.8
Phase B voltage, V ac	115±1.7	115.8	115.7	115.5	115.5	115.5
Phase C voltage, V ac	115±1.7	115.8	115.8	115.4	115.4	115.4
Lunar Orbiter V						
Main battery voltage, V dc	$28^{+2.0}_{-1.5}$	28.1	28.0	28.1	28.1	28.2
Inverter frequency, Hz	400±6	398.8	398.8	399.4	399.7	399.7
Phase A voltage, V ac	115±0.5	115.2	115.2	114.8	114.6	114.9
Phase B voltage, V ac	115±1.7	115.8	115.7	115.5	115.5	115.4
Phase C voltage, V ac	115±1.7	115.9	115.6	115.6	115.5	115.5

TELEMETRY SYSTEM

by Edwin S. Jeris

Description

The Atlas telemetry system consists of a telemetry package, a manually activated 28 V dc battery, associated transducers, wiring harness, and two antennas. Figure V-7 shows the telemetry system. Appendix B summarizes the launch vehicle instrumentation.

The 18 channel Pulse Amplitude Modulation/Frequency Modulation/Frequency Modulation (PAM/FM/FM) telemetry system 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 V dc for operation. The transmitter is designed to use standard Interrange Instrumentation Group (IRIG) subcarrier channels of 1 to 18. Channels 3, 4, and 12 are continuous direct. On these channels the transducer provides the modulating frequency. Channels 2 and 5 to 10 are continuous, using subcarrier oscillators. Channel 11 is commutated at 2.5 rps; channel 13 is commutated at 5 rps; channels 15 and 16 are commutated at 10 rps and channel 18 is commutated at 30 rps. Channels 1, 14, and 17 were not used for these flights. 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 Lunar Orbiters III, IV, and V. One-hundred sixteen measurements were made on each vehicle. Table V-XV shows Atlas telemetry parameters. No telemetry problems occurred during prelaunch activities or during flight. Signal strength was adequate during flight except for a 1-second loss of signal at booster engine staging. This loss of signal occurs as a result of signal attenuation by the Atlas sustainer exhaust plume. Carrier frequency and commutator speeds were stable and no data playback difficulties were encountered.

One minor instrumentation anomaly occurred on Lunar Orbiter III. Helium bottle pressure measurement (F246P) exhibited a 10 percent noise level from T + 15 to T + 25 seconds. This measurement uses a potentiometer transducer and the probable cause of the anomaly was an intermittent wiper contact under vibration.

Two instrumentation failures occurred on Lunar Orbiter IV. The measurements affected were sustainer fuel pump discharge pressure (P330P) which decreased to zero after T + 65 seconds and sustainer thrust chamber pressure (P6P) gradually decreased from the normal absolute pressure of 476 newtons per square centimeter (690 psi) indication to 345 newtons per square centimeter (500 psi) at T + 272 seconds. Sustainer thrust chamber absolute pressure measurement continued to indicate 345 newtons per square centimeter (500 psi) after sustainer engine cutoff. Both anomalies are believed to be caused by freezing transducer or transducer sense lines due to propellant leakage.

Two minor transducer anomalies occurred on Lunar Orbiter V. Noisy data were received from vernier fuel tank pressure measurement (P27P) from lift-off to T + 18 seconds. The noise obscured the data between T + 80 and T + 129 seconds. Noise and transients were present on the booster engine number 1 thrust chamber measurement (P60P) from T + 60 seconds to T + 122 seconds. Noise of the type encountered on measurements P60P and P27P is generally caused by intermittent loss of contact between the wipers and resistance elements of potentiometer transducers.

The telemetry stations used for support and the telemetry coverage provided are shown in appendix C (figs. C-4 to C-6).

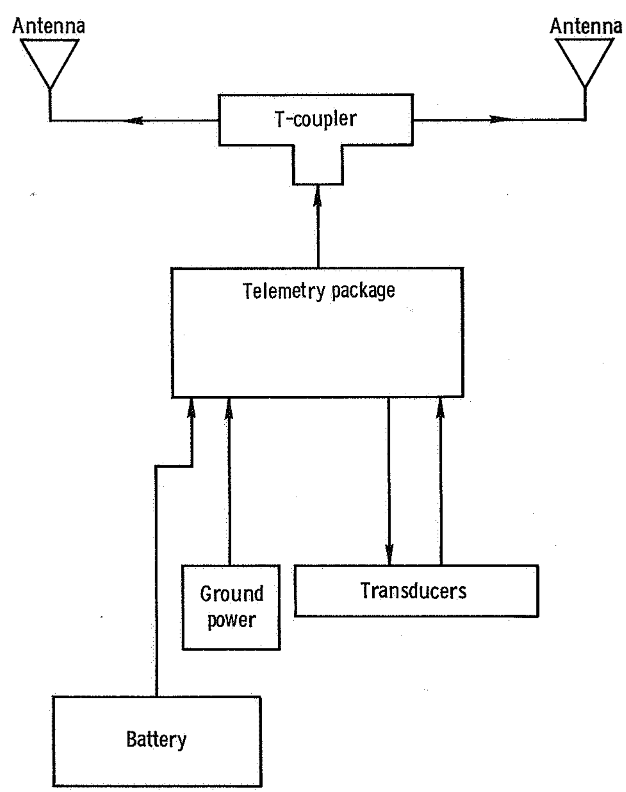


Figure V-7. - Atlas airborne telemetry system, Lunar Orbiter.

TABLE V-XV. - ATLAS TELEMETRY

[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 S236X S241X S245X S248X	
	MOD III guidance		G364X			Y41X	G363X				
Support systems	Propellant utilization										
	Pneumatic										
	Hydraulic										
	Electrical										E151V
	Range safety					D1V					
Number of measurements		0	1	1	1	5	2	3	3	5	1
		Total number of									

PARAMETERS, LUNAR ORBITERS III, IV, AND V

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	Con- tinuous	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 - 115

FLIGHT TERMINATION SYSTEM

by Edwin S. Jeris

Description

The Atlas contains a vehicleborne flight termination system which is designed to function on receipt of command signals from the ground stations. This system includes redundant receivers, a power control unit, an electrical arming unit, a destructor, and two batteries which operate entirely independent of the main vehicle power system (see fig. V-8).

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

Performance

Performance of the flight termination system was satisfactory. Prelaunch checks were completed without incident. The only measurement telemetered on receiver number 1 was the automatic gain control. The automatic gain control measurement on receiver number 1 indicated that the capability to terminate flight was maintained throughout powered flight. Minimum signal strength measured at the receiver on Lunar Orbiters III, IV, and V was 12.4, 66, and 62 microvolts, respectively. Five microvolts is the minimum required signal strength for receiver operation. Receiver number 2 was not instrumented. No flight termination commands were required, nor were any commands inadvertently generated by any vehicle system.

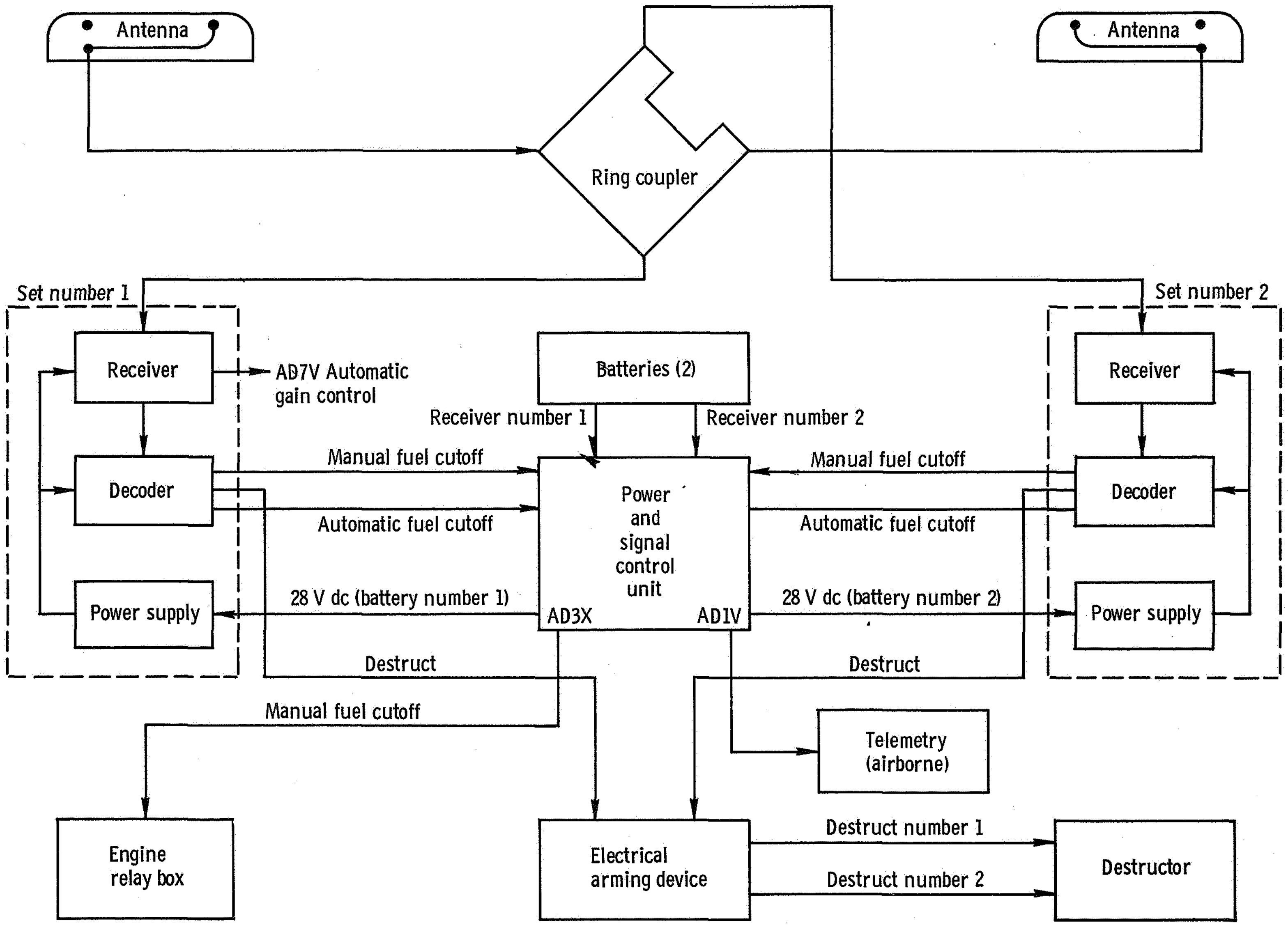


Figure V-8. - Atlas flight termination system, Lunar Orbiter.

VI. AGENA VEHICLE SYSTEM PERFORMANCE

VEHICLE STRUCTURE SYSTEM

by C. Robert Finkelstein

Description

The Agena vehicle structure system (fig. VI-1) consists of four major sections: the forward section, tank section, aft section, and the booster adapter assembly. Together they provide the aerodynamic shape, structural support, and environmental protection for the vehicle. The forward section is basically an aluminum structure with beryllium and magnesium panels. This section encloses most of the electrical, guidance, and communication equipment and provides the mechanical and electrical interface for the spacecraft adapter and shroud. The tank section consists of two integral aluminum propellant 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 supports the Agena and remains with the Atlas after Atlas-Agena separation. The forward section of the Lunar Orbiter - Agena was modified by the installation of a spacecraft ground air-conditioning duct. The duct assembly extends from an access panel on the forward section to an air distribution manifold on the spacecraft adapter sealing diaphragm.

Performance

The measured dynamic environment of the structure system was within design limitations. The measured data are presented in appendix D. The peak steady state longitudinal load factors were 6.08 g's for Lunar Orbiter V and 6.2 g's for both Lunar Orbiters III and IV.

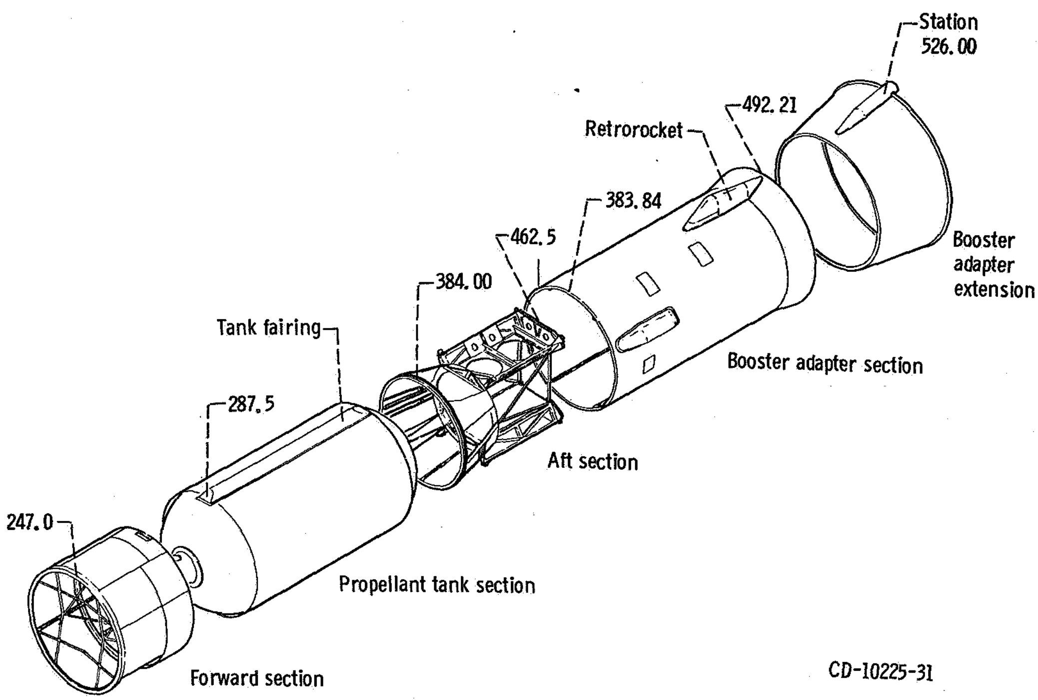


Figure VI-1. - Agena vehicle structure system, Lunar Orbiter.

SHROUD-ADAPTER SYSTEM

by C. Robert Finkelstein

Description

The shroud-adapter system for each Lunar Orbiter consists of an all metal "over-the-nose" shroud and a spacecraft adapter. The system is shown in figures VI-2 and VI-3. The shroud provides environmental protection for the spacecraft prior to and during launch. The adapter is the transition section between the Agena-shroud and Agena Agena-spacecraft.

The shroud is 3.98 meters (13.0 ft) long and weighs 163.4 kilograms (360 lb). The shroud consists of a 1.66-meter-(5.4-ft-)diameter cylindrical section, a 15° half-angle conical section, an ogive section, and a 0.72-meter- (2.4-ft-) diameter hemispherical nose cap. The nose cap is made of beryllium; all other sections are fabricated of magnesium alloy. The shroud is strengthened by magnesium alloy internal rings to which polished aluminum liners are attached to protect the spacecraft from thermal radiation.

The shroud is clamped to the spacecraft adapter by a V-band which is tensioned to approximately 30 000 newtons (6750 lb). Jettison of the shroud is initiated by a radio guidance discrete approximately 2 seconds after Atlas vernier engine cutoff. At this time, Agena electrical power is used to fire two pyrotechnic-actuated release devices in the shroud separation V-band. Either release device will effect shroud separation. When the V-band is released, four pairs of spring loaded pushrods in the shroud bearing against the adapter provide the energy to eject the shroud "over the nose" of the spacecraft at a relative velocity of 2 meters per second (7 ft/sec).

The adapter, 1.67 meters (5.5 ft) in diameter and 0.26 meter (0.84 ft) high, is fabricated of aluminum and magnesium and is bolted to the Agena forward section at station 247. A magnesium diaphragm is mounted across the bottom of the adapter at station 247 to isolate the shroud compartment from the Agena. The spacecraft is clamped to the adapter by a V-band tensioned to approximately 17 792 newtons (4000 lb). Spacecraft separation is initiated by an Agena sequence timer discrete following Agena injection into a translunar trajectory. Agena electrical power is used to fire pyrotechnic release devices which sever the spacecraft V-band in two places 180° apart. Four spring mechanisms, located in the adapter and reacting on the spacecraft interface plane, then impart the energy to attain a differential velocity of approximately 1/2 meter per second (1.6 ft/sec) between the Agena and the spacecraft. The V-band is restrained from possible impact on the spacecraft by a series of retention springs.

The shroud-adapter system is instrumented with six accelerometers and vibrometers mounted in the adapter, two shroud separation switches, two strain gages to measure separation V-band tension, a temperature transducer and a pressure transducer mounted in the adapter, and three linear potentiometers arranged to measure spacecraft separation.

The adapter incorporates a ground cooling air distribution system (fig. VI-4) required for the Lunar Orbiter spacecraft. Ground cooling air for the spacecraft is conveyed by duct through the Agena forward section to a fiberglass manifold riveted to the top of the diaphragm beneath the spacecraft equipment mounting deck. Various items of electronic equipment on this mounting deck dissipate heat loads into the deck varying from 100 to 250 watts. The distribution of ground cooling air provided by the manifold maintains the average temperature of the equipment mounting deck at 381±3.5 K ($46^{\circ}\pm6^{\circ}$ F). During ground cooling, the shroud cavity is vented through an exhaust coupling mounted on the side of the shroud. At lift-off this exhaust coupling is closed, and the shroud cavity is vented into the Agena forward section through a pressure relief valve in the diaphragm. The Agena forward section is vented through the cable tunnels on the side of the Agena to the aft section, where external vent holes are provided. A check valve in the air-conditioning duct prevents leakage from the shroud cavity during flight, and the door assembly on an access panel on the forward section prevents leakage from the exterior environment into the shroud cavity.

The Lunar Orbiter spacecraft and the shroud are mated to the adapter in an environmental clean room, and the complete assembly is then transported to the launch pad and mated to the Agena. Since the spacecraft carries fuel, the clean room is also required to be an explosive-safe area.

Performance

During flight for Lunar Orbiters III, IV, and V, the shroud cavity pressure decayed as expected, indicating proper operation of the pressure relief valve in the diaphragm and of the air-conditioning duct assembly. The shroud cavity temperature remained within the expected range of 277.5 to 286 K (40° to 55° F) for all three flights.

Shroud pyrotechnics were fired at approximately T + 311 seconds. The variation in pyrotechnic ignition times for the three flights was 2 seconds. Spacecraft accelerometer data indicated no disturbance of the spacecraft during shroud jettison.

For Lunar Orbiter III, both separation switches operated simultaneously, indicating that shroud jettison was initiated properly.

For Lunar Orbiter IV, only one shroud separation switch operated at the time of shroud separation. The switch that did not operate during flight was known to be inoperative at the time of shroud installation. The pyrotechnic shock accelerometer signature at shroud separation was similar to the signatures of Lunar Orbiters I, II, and III and, therefore, indicated normal shroud separation.

For Lunar Orbiter V, both shroud separation switches operated prematurely. The probable cause of this anomaly was marginal switch plunger operation (overtravel) coupled with the flexibility of the thin metal cover that held the switch plunger depressed. The pyrotechnic shock acceleration signature at shroud separation of Lunar Orbiter V was compared with the signature of Lunar Orbiter I and was found to be similar. This similarity of signatures indicates normal shroud separation. Lunar Orbiter I was chosen for comparison because the trajectories for both Lunar Orbiters I and V were about 0.2 sigma depressed.

Spacecraft separation V-band pyrotechnics were fired at T + 1358.6 seconds for Lunar Orbiter III, T + 2013.07 seconds for Lunar Orbiter IV, and T + 2133.2 seconds for Lunar Orbiter V. The variation in spacecraft separation times results from a variation in Agena coast times. For each flight the V-band strain gage data indicated normal V-band release. For these three flights, the adapter and spacecraft accelerometers indicated that the spacecraft separated cleanly and did not contact the adapter.

For Lunar Orbiter III, the separation potentiometers showed that the spacecraft-Agena differential velocity was 0.48 meter per second (1.56 ft/sec). Figure VI-5 shows the data obtained from the three spacecraft separation potentiometer for Lunar Orbiter III and is representative of the action of the separation potentiometers for the other Lunar Orbiter flights. For Lunar Orbiter IV, the spacecraft-Agena differential velocity was 0.40 meter per second (1.30ft/sec). For Lunar Orbiter V, the spacecraft Agena differential velocity was 0.37 meter per second (1.20 ft/sec).

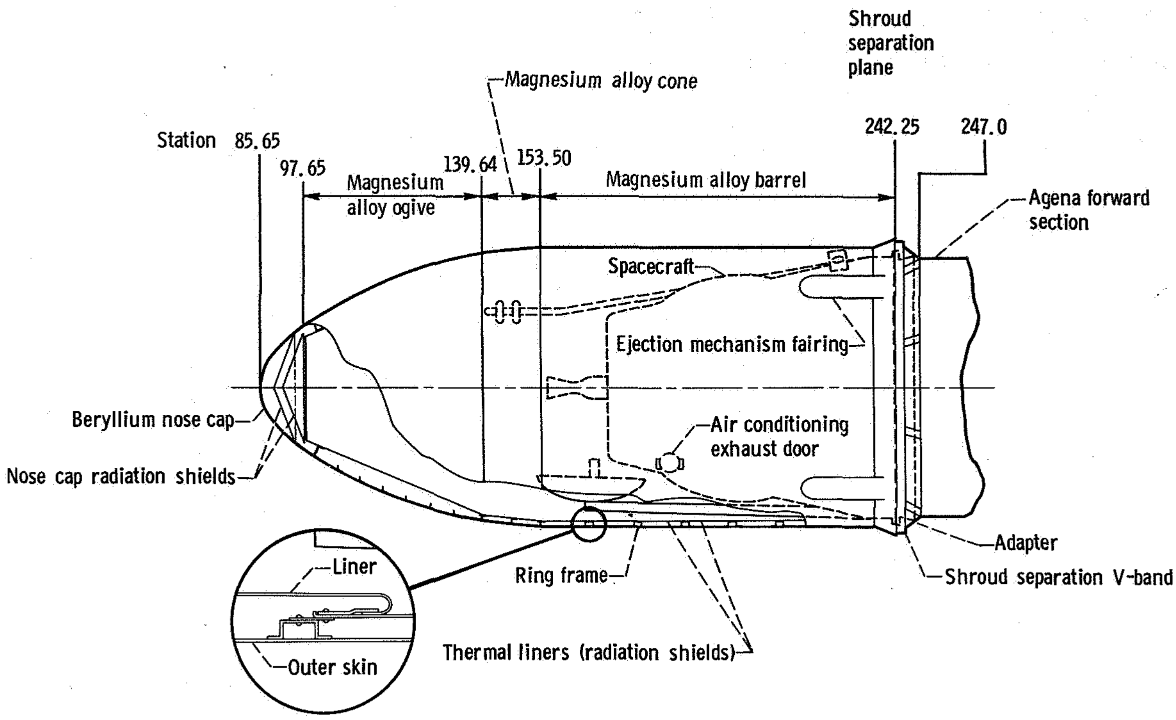


Figure VI-2. - Shroud system, Lunar orbiter.

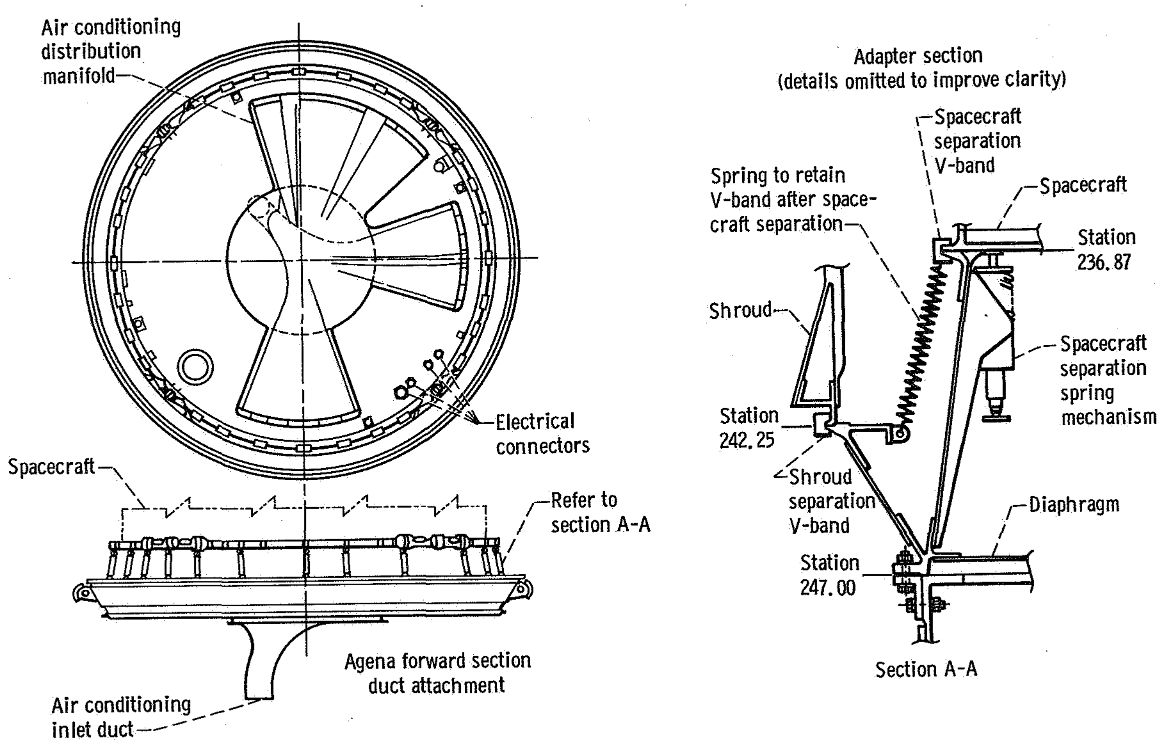


Figure VI-3. - Spacecraft adapter, Lunar orbiter.

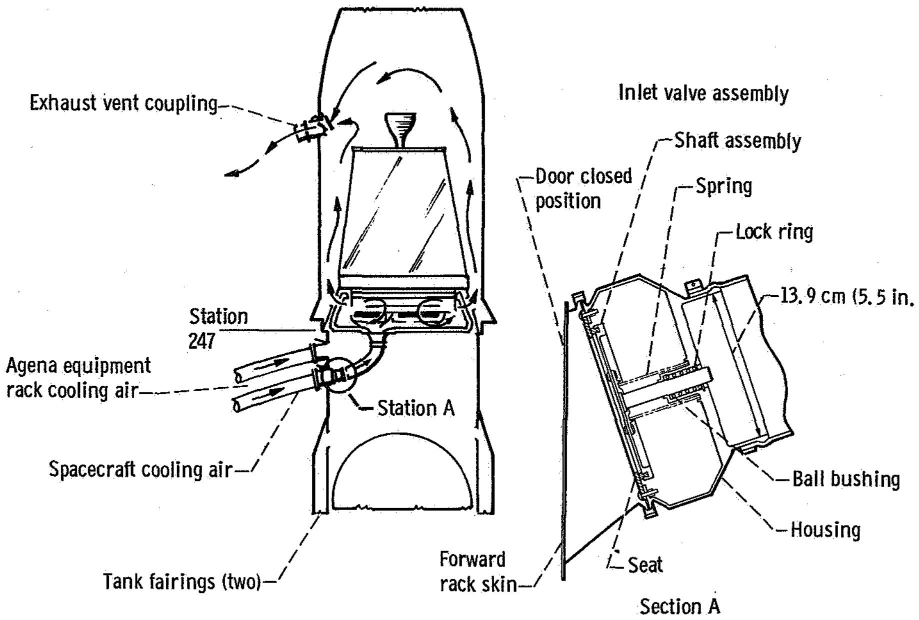


Figure VI-4. - Ground cooling air distribution system, Lunar orbiter.

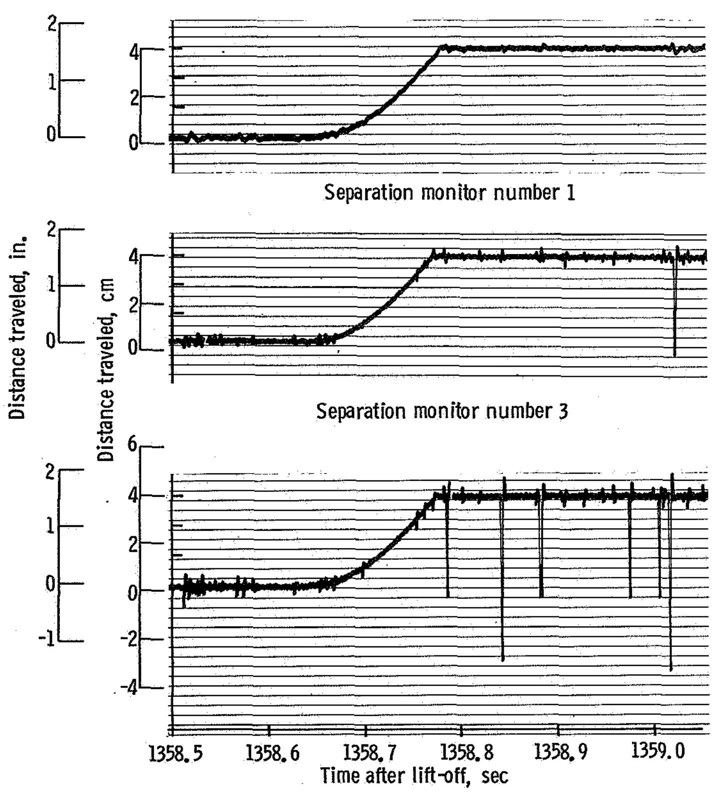


Figure VI-5. - Potentiometer travel, Lunar orbiter III separation.

PROPULSION SYSTEM

by Robert J. Schroeder

Description

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

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

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

The Agena engine is a Bell Aerosystems Company Model 8096 liquid bipropellant engine which uses unsymmetrical dimethylhydrazine (UDMH) fuel and inhibited red fuming nitric acid (IRFNA) oxidizer. Rated thrust in a vacuum is 71 168 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 is used to provide a rapid closure of the main oxidizer valve at first burn cutoff. This oxidizer fast-shutdown system consists of a pyrotechnically operated valve and a high-pressure nitrogen storage cylinder. Engine thrust vector control is provided by the gimbal mounted thrust chamber. A pair of hydraulic actuators provides the force for thrust chamber movement in response to signals produced by the Agena guidance and control system.

Atlas-Agena separation, after Atlas vernier engine cutoff, is accomplished by firing a Mild Detonating Fuse which cuts 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.

The propulsion system also includes a solid propellant retrorocket motor mounted on the aft rack of the Agena. Rated average vacuum thrust is 609 newtons (137 lb) with an action time of 16.6 seconds. The retrorocket imparts a velocity change to the Agena vehicle to reduce the probability of impact of the Agena vehicle on the moon and to ensure that the Agena will not interfere with the spacecraft Canopus seeker. This retrorocket is fired during the Agena retromanuever sequence.

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 reposition the horizon sensor heads, and to activate the oxidizer fast-shutdown system. Ignitors are used for the solid propellant engine start charges and solid propellant 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 V-band release devices.

Performance

The Agena engine first burn was initiated by the primary sequence timer at T + 370.4 seconds on Lunar Orbiter III, T + 364.85 seconds on Lunar Orbiter IV, and T + 369.3 seconds on Lunar Orbiter V. Start sequencing of the engine control valves was normal. Engine start transients were normal and 90 percent combustion chamber pressure reached within approximately 1.2 seconds on each flight. The average engine steady-state thrust was slightly lower than expected on Lunar Orbiters III and V. Lunar Orbiter IV had a slightly higher thrust level than expected. Engine burn was terminated by velocity meter cutoff command at T + 527.4 on Lunar Orbiter III, T + 518.16 seconds on Lunar Orbiter IV, and T + 523.58 seconds on Lunar Orbiter V. The actual thrust duration measured from 90 percent chamber pressure time to velocity meter cutoff command time was longer than expected on Lunar Orbiters III and V. Lunar Orbiter IV experienced a shorter-than-expected thrust duration. Average steady-state flight values of thrust level and thrust duration are summarized and compared with corresponding preflight predicted values in table VI-I. The flight values were within the allowable 3 sigma limits.

The propellant tank pressurization system performed as expected. The fuel pump inlet pressure measurement B-1 failed approximately 11 seconds before engine first cutoff on Lunar Orbiter IV.

The propellant isolation valves were commanded to close after the Agena engine first burn by the primary sequence timer at T + 533.4 seconds on Lunar Orbiter III, T + 527.9 seconds on Lunar Orbiter IV, and T + 523.3 seconds on Lunar Orbiter V. The propellant isolation valves functioned as expected and vented the propellants in the engine pumps and feed lines.

At the end of the Agena coast phase, the propellant isolation valves were commanded to open 2 seconds prior to initiating the Agena engine second burn. Engine ignition sequence for second burn occurred at T + 1104.5 seconds on Lunar Orbiter III, T + 1760.0 seconds on Lunar Orbiter IV, and T + 1879.4 seconds on Lunar Orbiter V. Start sequencing of the engine control valves was normal. Engine start transients were as expected and 90 percent combustion chamber pressure reached within approximately 1.2 seconds on each flight. The engine burn was terminated by velocity meter cutoff command at T + 1194.37 on Lunar Orbiter III, T + 1848.49 seconds on Lunar Orbiter IV, and T + 1967.6 seconds on Lunar Orbiter V. Second burn deviations in thrust level and thrust duration were similar to those experienced on the first burn. Average steady-state flight values of thrust level and thrust duration are summarized and compared with corresponding preflight predicted values in table VI-II. All flight values were within the allowable 3 sigma limits.

Flight data indicated normal pump inlet pressures except for the previously noted failure of the fuel pump inlet pressure on Lunar Orbiter IV.

The Agena retrorocket was ignited at T + 1958.5 seconds on Lunar Orbiter III, T + 2613.1 seconds on Lunar Orbiter IV, and T + 2733.5 seconds on Lunar Orbiter V. Action time appeared normal on all three flights as evidenced by the Agena velocity meter accelerometer.

TABLE VI-I. - AGENA ENGINE PERFORMANCE FIRST BURN

Parameter	Unit	Lunar Orbiter					
		III		IV		V	
		Expected value	Actual value	Expected value	Actual value	Expected value	Actual value
Average steady-state engine thrust	N	71 155	70 786	71 969	72 591	72 369	72 035
	lb	15 997	15 924	16 180	16 320	16 270	16 195
Thrust duration (90 percent chamber pressure to velocity meter cutoff)	sec	154.77	155.74	152.56	152.00	151.71	153.13

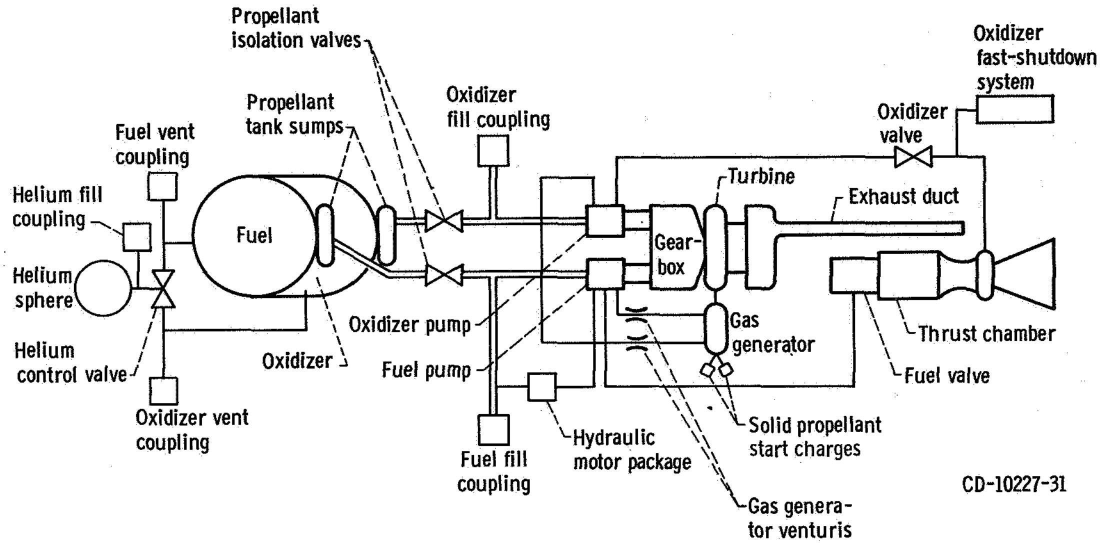


Figure VI-6. - Agena propulsion system schematic, Lunar Orbiter.

TABLE VI-II. - AGENA ENGINE PERFORMANCE SECOND BURN

Parameter	Units	Lunar Orbiter					
		III		IV		V	
		Expected value	Actual value	Expected value	Actual value	Expected value	Actual value
Average steady-state engine thrust	N	70 884	70 425	71 711	71 880	72 146	71 969
	lb	15 936	15 833	16 122	16 160	16 220	16 180
Thrust duration (90 percent chamber pressure to velocity meter cutoff)	sec	88.37	88.70	87.20	87.32	86.58	87.50

ELECTRICAL SYSTEM

by Edwin R. Procasky

Description

The Agena electrical system (see fig. VI-7) supplies all power, frequency, and voltage requirements 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, three dc-dc converters, and one solid-state three-phase inverter.

The two primary batteries, with a minimum rating of 966 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 V ac (rms) at 400 hertz (± 0.02 percent) to the guidance system. One dc-dc converter supplies regulated ± 28 V dc to the guidance system, while the other two dc-dc converters supply 28 V 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 each of the three flights.

The battery current load profile was as expected for all three missions and compared closely with the two previous Lunar Orbiter missions. The inverter and converter voltages were within specification at lift-off and remained essentially constant throughout each flight. Table VI-III summarizes the electrical system performance during the three missions.

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

TABLE VI-III. - AGEAN ELECTRICAL SYSTEM FLIGHT PERFORMANCE SUMMARY, LUNAR ORBITER

[Tolerances listed are for telemetry data. The tolerances listed in the System Description are equipment specifications.]

Measurement	Tolerance	Measurement number	Flight	Flight event at -					
				Lift-off	First ignition	First shutdown	Second ignition	Second shutdown	Spacecraft separation
Pyrotechnic battery voltage, V dc	22.5 to 29.5	C-141	III	26.3	26.3	26.3	26.3	26.3	26.3
			IV	25.9	25.9	25.9	25.9	25.9	25.9
			V	25.9	25.9	25.9	25.9	25.9	25.9
Main battery voltage, V dc	22.5 to 29.5	C-1	III	25.6	25.3	25.6	25.3	25.6	25.6
			IV	25.3	24.7	25.2	24.7	25.2	25.2
			V	25.3	24.7	25.3	24.7	25.3	25.3
Battery current, A dc		C-4	III	15	18	15	18	15	15
			IV	15	18	15	18	15	15
			V	15	18	15	18	15	15
Guidance converter 28.3 V dc regulated	27.7 to 28.9	C-3	III	28.5	28.5	28.5	28.5	28.5	28.5
			IV	28.0	28.0	28.0	28.0	28.0	28.0
			V	28.3	28.3	28.3	28.3	28.3	28.3
Guidance converter -28.3 V dc regulated	-27.7 to -28.9	C-5	III	-28.2	-28.2	-28.2	-28.2	-28.2	-28.2
			IV	-28.7	-28.7	-28.7	-28.7	-28.7	-28.7
			V	-28.7	-28.7	-28.7	-28.7	-28.7	-28.7
Guidance inverter phase AB, V ac rms	112.7 to 117.3	C-31	III	116.1	116.1	116.1	116.1	116.1	116.1
			IV	113.4	113.4	113.4	113.4	113.4	113.4
			V	113.4	113.4	113.4	113.4	113.4	113.4
Guidance inverter phase BC, V ac rms	112.7 to 117.3	C-32	III	116.1	116.1	116.1	116.1	116.1	116.1
			IV	113.4	113.4	113.4	113.4	113.4	113.4
			V	113.4	113.4	113.4	113.4	113.4	113.4
Telemetry converter number 1 28.3 V dc regulated	27.7 to 28.9	H204	III	28.0	28.0	28.0	28.0	28.0	28.0
			IV	28.0	28.0	28.0	28.0	28.0	28.0
			V	28.0	28.0	28.0	28.0	28.0	28.0
Telemetry converter number 2 28.3 V dc regulated	27.7 to 28.9	H400	III	28.0	28.0	28.0	28.0	28.0	28.0
			IV	28.0	28.0	28.0	28.0	28.0	28.0
			V	28.0	28.0	28.0	28.0	28.0	28.0

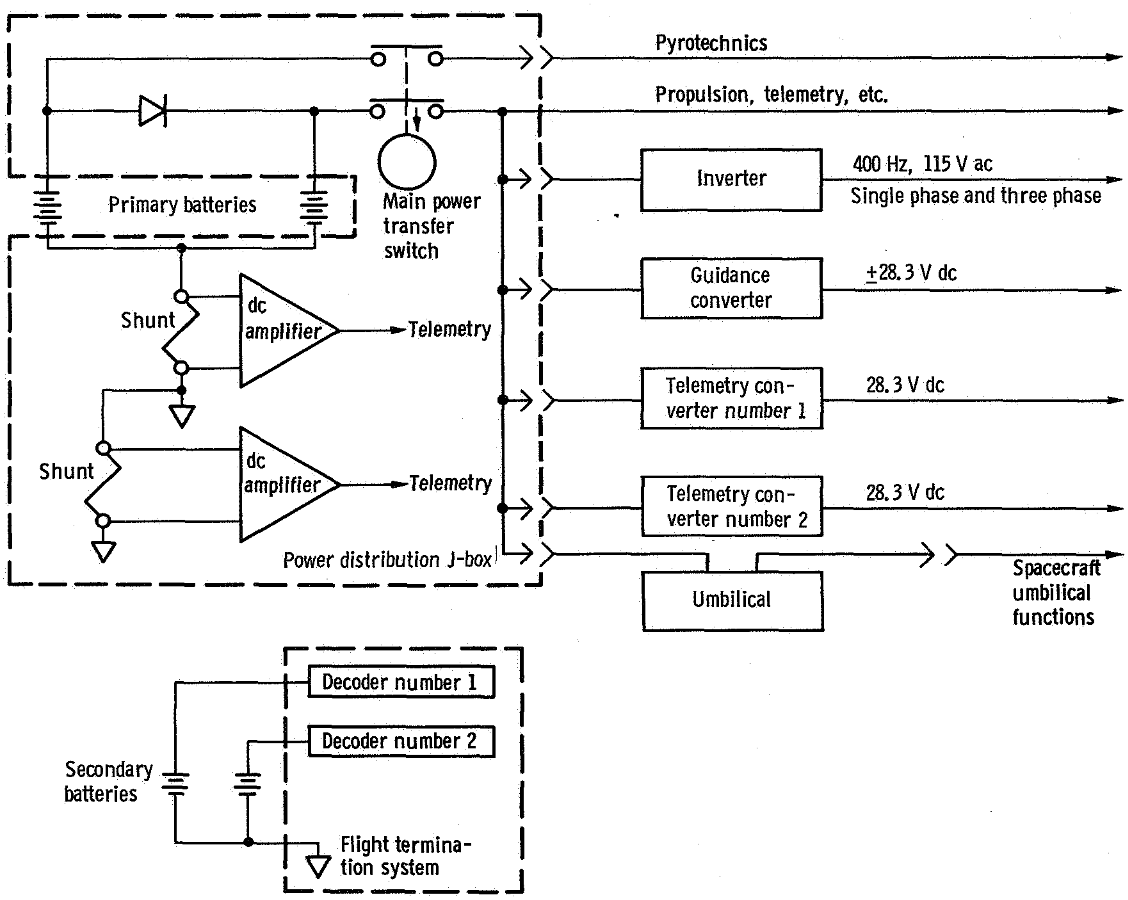


Figure VI-7. - Agena electrical system, Lunar Orbiters III, IV, and V.

GUIDANCE AND FLIGHT CONTROL SYSTEM

by Howard D. Jackson

Description

The Agena guidance and flight control system performs the vehicle guidance, control, and flight programming functions necessary to accomplish the vehicle mission after Atlas-Agena separation. The system consists of three subsystems; guidance, control, and flight programming functions. A block diagram of the system is shown in figure VI-8.

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 IRP. The infrared horizon sensors provide corrections in pitch and roll to the IRP 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. Longitudinal acceleration is sensed by the velocity meter accelerometer and the velocity meter counts down the first burn "velocity-to-be-gained" binary number. The velocity meter counter then generates a signal to terminate engine thrust when the vehicle velocity increases by a predetermined increment. The second burn "velocity-to-be-gained" number is transferred from a storage register 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 IRP are conditioned and amplified by the flight control electronics to operate the cold gas and hydraulic systems. During Agena coast periods, the cold gas system consisting of six thrusters provides roll, pitch, and yaw control. These thrusters operate on a mixture of nitrogen and tetrafluoromethane compound. During powered flight the hydraulic system provides pitch and yaw control by means of two hydraulic actuators which gimbal the Agena engine thrust chamber; roll control is provided by the cold gas system. A patch panel in the flight control junction box provides the means for varying the interconnections of the guidance and flight control system to suit mission requirements.

The flight programming subsystem uses sequence timers to program Agena flight events. A sequence timer provides 22 usable, discrete event times with multiple switch closure capability and has a maximum running time of 6000 seconds. Two timers (a primary and restart timer) are used for lunar or interplanetary probe missions. The primary sequence timer is started by a ground initiated ratio guidance command at a time determined by the ground-based computer after compensating for trajectory dispersions of the booster. These commands are routed through the Atlas to the Agena. The primary sequence timer is programmed to stop during the Agena coast period. Prior to Agena second ignition, the primary timer is restarted by the restart sequence timer. The restart time, which is variable and which is determined prior to Atlas-Agena separation, controls the duration of the coast period so that the spacecraft will be at the proper injection point after second burn. Three methods of varying the restart time are provided:

(1) Coarse adjustments can be made on the restart sequence timer for various launch days.

(2) Time may be run off the restart sequence timer in 45 second increments during the countdown.

(3) A radio guidance command, which starts the restart sequence timer in flight, provides fine adjustments to compensate for lift-off time and trajectory.

Performance

The guidance and flight control system performance on Lunar Orbiters III, IV, and V was satisfactory throughout flight with the exception of the Lunar Orbiter III velocity meter accelerometer which exhibited a slight null shift at the end of Agena first burn. This null shift was evident in the data prior to second burn and after second burn through loss of signal; however, this shift did not have any adverse effect on the mission. 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 were within previous flight experience and are shown in the following table:

Lunar Orbiter flight	Vehicle rates, deg/sec			Attitude errors at cold gas activation, deg		
	Yaw	Roll	Pitch	Yaw	Roll	Pitch
III	0.26 left	0.03 CW[a]	0.24 down	0.8 left	0.5 CW	0.6 down
IV	.3 left	(b)	.15 down	.6 left	.2 CCW[c]	.3 down
V	(b)	.05 CW	(b)	(b)	.4 CW	.2 down

[a]Clockwise (CW) roll reference applies when looking forward along the Agena longitudinal axis (see fig. VI-9).
[b]Negligible.
[c]Counterclockwise (CCW).

The times required by the cold gas attitude control system to reduce these errors to within the dead band limits of $\pm0.2^{\circ}$ pitch, $\pm0.18^{\circ}$ yaw, and $\pm0.6^{\circ}$ in roll were 4.2 seconds on Lunar Orbiter III and 1.5 seconds on Lunar Orbiters IV and V.

Subsequent to the cold gas attitude control activation, each vehicle completed a programmed pitch down of 10° and the programmed geocentric rate of 3.21 degrees per minute pitch down was applied. For the Agena first burn, the pitch horizon sensors were set at a pitch bias angle of 5.22° (nose up) for Lunar Orbiter III and a pitch bias angle of 5.05° (nose up) for Lunar Orbiters IV and V. Each of the vehicles had stabilized in all axes by the time of first ignition.

Gas generator spin-up at Agena engine first ignition resulted in roll rates and induced maximum displacement errors as follows:

Lunar Orbiter flight	Roll rate, deg/sec	Maximum displacement error, deg	Time to reverse initial rate, sec
III	2 CW[a]	2.6 CW	2.2
IV	2.46 CW	2.8 CW	1.3
V	3.08 CW	2.55 CW	1.4

[a]Clockwise (CW) roll reference applies when looking forward along Agena longitudinal axis (see fig. VI-9).

The hydraulic and cold gas attitude control activity was normal throughout the Agena engine first burn for each of the three vehicles. In each flight, engine shutdown was commanded by the velocity meter after the vehicle had attained the required velocity increment.

The roll transients, caused by engine shutdowns (i.e., turbine spin and turbine exhaust decay), were as experienced on previous flights. The maximum time required to reduce the roll excursions to within the cold gas attitude control dead bands was 23 seconds for Lunar Orbiter V.

During the engine thrust decay period for Lunar Orbiter III the velocity meter accelerometer pulse rate decreased as expected for the first 18 seconds. There were no pulses for the next 39 seconds whereupon the accelerometer established a low frequency pulse rate of approximately 0.27 pulses per second output (rather than the expected zero pulse/sec) which continued until loss of signal. This essentially constant pulse rate represented approximately 1.08×10^{-3} g's and is attributed to a null shift of the accelerometer. Similar velocity meter accelerometer null shifts have been experienced on some previous flights; this shift did not adversely affect the mission.

Approximately 20 seconds after Agena engine shutdown, the programmed geocentric pitch rate was increased to 4.20 degrees per minute pitch down and the horizon sensor bias angle was decreased to -0.12° (nose down) for Lunar Orbiter III and 0.21° for Lunar Orbiters IV and V. Horizon sensor and cold gas attitude control data show that the vehicles maintained the proper attitude in the coast phase.

Gas generator turbine spin-up at Agena engine second ignition resulted in roll rates and induced maximum displacement errors as follows:

Lunar Orbiter flight	Roll rate, deg/sec	Maximum displacement error, deg	Time to reverse initial rate, sec
III	2.3 CW[a]	3.2 CW	1.9
IV	2.1 CW	2.2 CW	1.1
V	2.7 CW	2.4 CW	1.2

[a]Clockwise (CW) roll reference applies when looking forward along the Agena longitudinal axis (see fig. VI-9).

The hydraulic and cold gas attitude control activity was normal throughout the Agena engine second burn for each of the three vehicles. In each flight, engine shutdown was commanded by the velocity meter after the vehicles had attained the required velocity increment.

Subsequent to spacecraft separation the vehicles performed an approximate 180° yaw maneuver. This yaw maneuver and subsequent retrorocket firing were successful and the retrosequence was properly accomplished on each flight.

The Agena attitude control gas usage was less than anticipated on all three flights. A comparison of predicted and actual gas usage is given in table VI-IV. These data indicate that the Agena attitude was stable throughout flight.

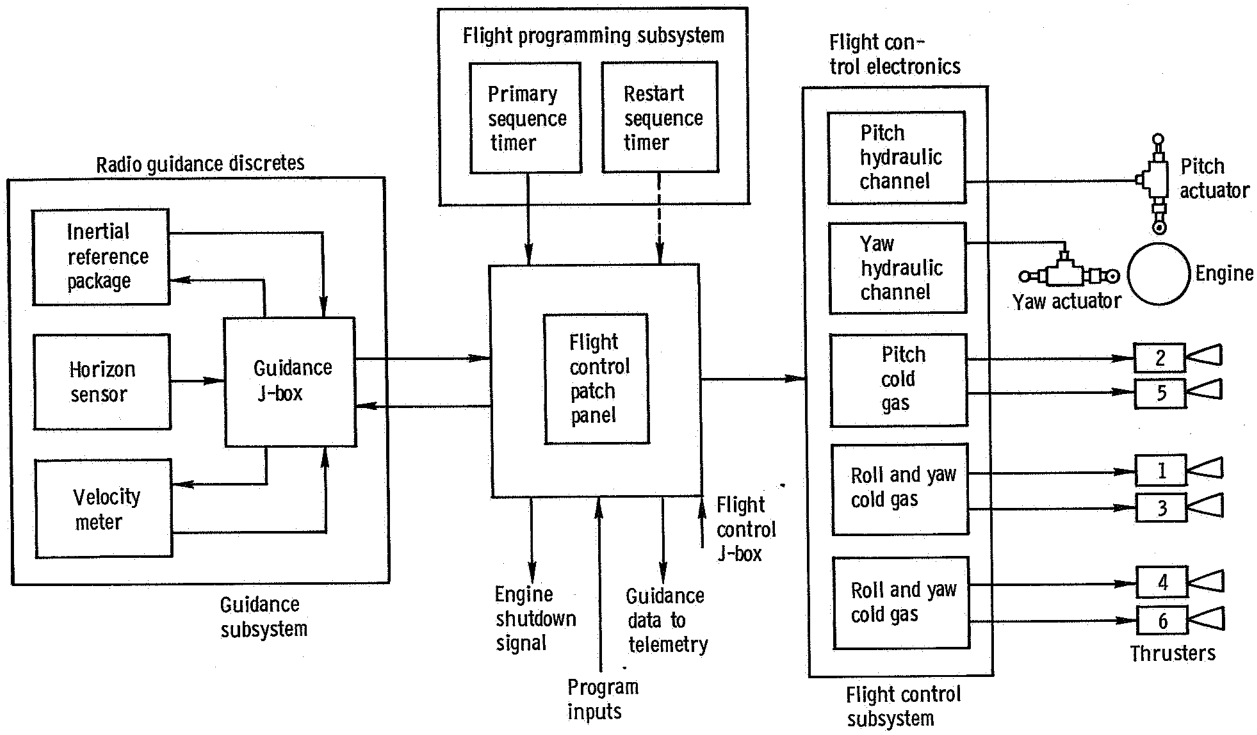


Figure VI-8. - Agena guidance and flight control system block diagram, Lunar Orbiter.

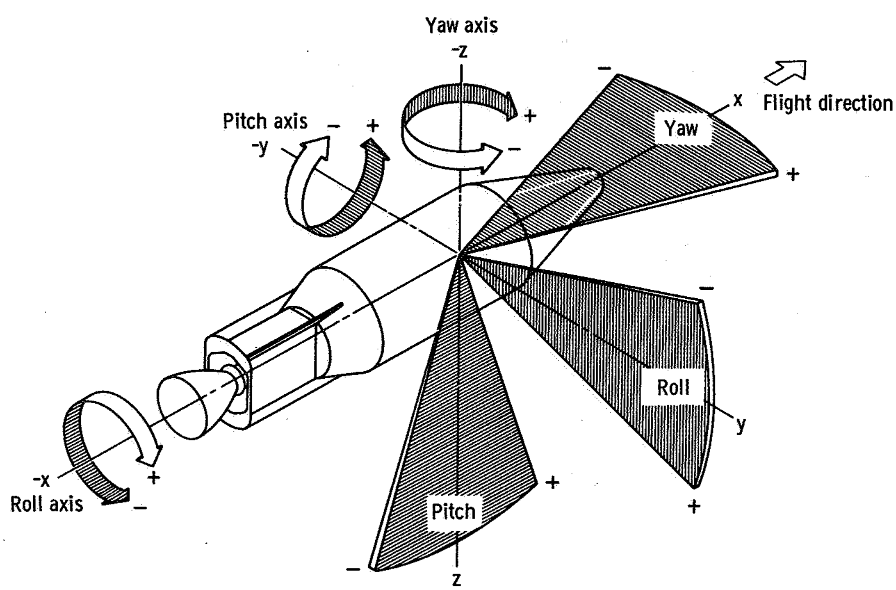


Figure VI-9. - Agena vehicle axes and vehicle movement designations, Lunar Orbiter.

TABLE VI-IV. - ATTITUDE CONTROL GAS LOADING REQUIREMENTS AND USAGE, LUNAR ORBITER

Flight sequence	Units	Predicted nominal	Lunar Orbiter flight		
			III	IV	V
			Actual		
Lift-off	kg lb	13.5 load 29.7	13.3 loaded 29.3	14.2 loaded 31.2	13.9 loaded 30.5
Agena first coast	kg lb	2.7 usage 6.0	0.3 used 0.7	0.6 used 1.4	0.3 used 0.7
Agena engine first burn	kg lb	1.4 usage 3.0	0.1 used 0.3	1.5 used 3.4	1.8 used 3.9
Parking orbit coast	kg lb	0.6 usage 1.3	0.7 used 1.6	0.3 used 0.6	0.5 used 1.2
Agena engine second burn	kg lb	0.8 usage 1.7	0.3 used 0.6	1.3 used 2.7	1.4 used 3.0
Prespacecraft separation maneuver	kg lb	0.9 usage 2.0	0.6 used 1.3	0.2 used 0.5	0.05 used 0.1
Postspacecraft separation maneuver	kg lb	2.3 usage 5.0	1.8 used 4	2.1 used 4.6	0.3 used 0.7
Loss of signal	kg lb	4.85 surplus 10.7	9.5 surplus 20.8	8.2 surplus 18.0	9.5 surplus 20.9

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. This system is essentially identical for all the Lunar Orbiter - Agena vehicles. Appendix B summarizes the launch vehicle instrumentation.

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 VHF transmitter, voltage controlled oscillators, a commutator, a switch and calibrate unit, and two dc-dc converters. The transmitter operates on an assigned frequency of 244.3 megahertz at a power output of 10 watts. The telemetry subsystem consists of thirteen continuous channels and two commutated channels. Eight continuous subcarrier channels are used for accelerometer data, three of which are switched to the spacecraft separation monitors after Agena engine second burn; two continuous channels are for spacecraft V-band tension measurements; one continuous channel transmits spacecraft engineering data; one continuous channel monitors the Agena cold gas attitude control thrusters and another continuous channel is time shared by the Agena velocity meter accelerometer and velocity meter counter.

Two channels are commutated at 5 rps with 60 segments on each channel. These two channels monitor 50 (49 for Lunar Orbiter III) of 67 measurements. Lunar Orbiter IV - Agena and Lunar Orbiter V - Agena vehicles incorporated a new measurement (B130) which was added to monitor the actuation of the propellant isolation valves.

The turbine speed measurement does not utilize a subcarrier channel but directly modulates the transmitter during engine operation. The Lunar Orbiter III - Agena vehicle was the first NASA-Agena vehicle to use a filter between the turbine speed pickup and the associate signal conditioning unit. The purpose of the filter is to improve the waveform and reduce the noise from the pickup.

The airborne tracking subsystem includes a C-band beacon transponder, RF switch, and an antenna. The transponder receives coded signals from the ground 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) of 0 to 1600 pulses per second. The pulse rate varies inversely with range. The pulse rate varies inversely with range. The RF switch connects the output of the transponder to either the umbilical for ground checkout or the antenna for flight.

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

Telemetry Subsystem Performance

The Agena telemetry subsystem performance on Lunar Orbiter flights III, IV, and V was satisfactory. Stations at Cape Kennedy and Antigua provide complete telemetry coverage from lift-off through Agena engine first burn. The Agena engine second burn, Agena - Lunar Orbiter separation, and Agena retromaneuver were monitored by Eastern Test Range downrange stations and Manned Space Flight Network stations. Signal strength data from all the stations showed an adequate and continuous signal level from the vehicle telemetry transmitter for these periods. The telemetry data indicated that the performance of the voltage controlled oscillators, switch and calibrate unit, dc-dc converters, and commutator was satisfactory. The telemetry stations used for support and the telemetry coverage provided are shown in figures C-4 to C-6 (see appendix C).

On the Lunar Orbiter III flight, all measurements were satisfactory except the four environmental low frequency accelerometers; A4 tangential, A5 tangential, A9 longitudinal and A523 radial. These accelerometers exhibited 2 to 5 percent step changes in acceleration level during the flight which are believed to be caused by a bias change in the reference level of the accelerometer-amplifier combination. These accelerometers are of the unbonded strain gage type. Similar accelerometers have exhibited this phenomenon on previous flights and there is no correlation between the occurrence of this pheomenon and flight events. The effects of the bias shifts have been compensated for in the presentation of data in appendix D.

It was noted prior to lift-off that the spacecraft composite signal on Channel F was being superimposed on the two spacecraft Channels 12 and 13. However, data recorded from these channels were not significantly degraded. It was also noted, during prelaunch test, that the signal from accelerometer A9 was inverted because of a vehicle wiring error. However, this condition was determined to be acceptable for flight.

On Lunar Orbiter IV all measurements were satisfactory except the fuel pump inlet pressure, measurement B1, which falsely indicated an abrupt pressure drop from 22.75 newtons per square centimeter (33 psi) to zero at 11 seconds prior to Agena engine first cutoff. This malfunction was probably due to an electrical failure of the transducer. Two instruments were known to have been faulty prior to launch; one of the three spacecraft separation potentiometers (A007) was open, and one of the two shroud separation switches (both switches are monitored by measurements A52) indicated a shroud off condition. Since these instruments were inaccessible, no attempt was made to replace them. Valid shroud and spacecraft separation data were obtained from the remaining instruments.

All measurements on the Lunar Orbiter V flight were satisfactory except the shroud separation monitor (measurement A52). This measurement indicated that the two switches for this measurement operated prematurely near the end of the Atlas booster phase. Analysis of the flight data shows conclusively that normal shroud separation occurred. For details of this instrumentation anomaly see the discussion of shroud performance for the Lunar Orbiter V in section VI. AGENA VEHICLE SYSTEM PERFORMANCE, SHROUD-ADAPTER SYSTEM.

The quality of turbine speed data from Lunar Orbiters III, IV, and V was improved by the addition of a filter from the turbine speed pickup as discussed in the Telemetry System description.

Tracking Subsystem Performance

The tracking subsystem performance was satisfactory and the C-band transmitted a continuous response to received interrogations for the required tracking periods for each flight. On Lunar Orbiter III, the C-band beacon was not interrogated for a period of approximately 40 seconds just prior to Agena engine second burn. This loss of interrogation was caused by an obstruction in the line of sight between the vehicle and the Canary Island ground radar antenna. The location of the radar stations used for support and the radar coverage provided is shown in figures C-7 to C-9 (see appendix C).

Flight Termination Subsystem Performance

The flight termination receivers (two per vehicle) functioned satisfactorily during prelaunch tests and flights of Lunar Orbiters III, IV, and V. Each receiver signal strength remained well above the threshold of 2 microvolts from lift-off to disabling of the flight termination subsystem shortly after Agena engine first cutoff.

VII. LAUNCH OPERATIONS

by George M. Michalson, Joseph A. Ziemianski, and Alvin C. Hahn

LUNAR ORBITER III

Prelaunch Activities

Major activities at the Eastern Test Range for Lunar Orbiter III are shown in table VII-I. All prelaunch tests were completed satisfactorily. The significant schedule delays and problems which occurred during the prelaunch period are as follows:

(1) Atlas Propellant Tanking Test Number 1; the test was postponed for 2 days because

(a) The airborne fuel regulator did not function properly and required replacement twice.

(b) A leak was detected at the fuel probe boss. Since the leak was small and was not considered detrimental to the flight, a Teflon bag was attached around the boss to contain the leakage.

(2) Booster Flight Acceptance Test (B-FACT) Number 2; an Atlas battery voltage redline was exceeded at about T - 38 seconds. The voltage drop in the battery simulator cable was sufficient to cause low voltage at the vehicle. An adjustment of the power supply output corrected the problem.

(3) Joint Flight Acceptance Composite Test (J-FACT); the thermocouple attached to the Agena helium sphere became detached from the sphere. The thermocouple was reattached.

(4) Simulated Launch Test; a spacecraft external power supply failure caused a 73-minute hold at T - 40 minutes. In addition, the spacecraft up-link signal strength varied as much as 15 decibels below and 8 decibels above specified values. The signal strength was improved by electrically boresighting the umbilical antenna with the Deep Space Instrumentation Facility (DSIF-71). A subsequent RF system verification test was satisfactorily completed.

(5) Subsequent spacecraft power supply problems caused a 1-day postponement of the launch.

Countdown and Launch

Lunar Orbiter III was successfully launched from ETR Launch Complex 13 on February 5, 1967 (referenced to GMT). Lift-off, 5.08-centimeter (2-in.) motion, occurred at 0117:01.120 GMT. The countdown proceeded as planned. The performance of the launch vehicle ground equipment during the countdown was satisfactory. Spacecraft air-conditioning became inoperative at about T - 50 minutes and was transferred to the backup system without incident.

LUNAR ORBITER IV

Prelaunch Activities

Major activities at the Eastern Test Range for Lunar Orbiter IV are shown in table VII-I. All prelaunch tests were completed satisfactorily. The significant schedule delays and problems which occurred during the prelaunch period are as follows:

(1) Atlas-Agena Mate; the Atlas-Agena mating occurred 8 days late because of the late replacement of the Agena propellant isolation valves. Consequently, the Agena-Spacecraft Mate was 2 days late and the Joint Flight Acceptance Composite Test (J-FACT) was 4 days late. These delays resulted in performing the Simulated Launch Test without launch vehicle participation. However, the late Agena mating did not impact the launch date.

(2) Booster Flight Acceptance Test (B-FACT) Number 1:

(a) The Atlas pod cooling system circuit breaker opened at about T - 7 minutes and caused a hold of about 4 minutes. The repair of a short circuit corrected the problem.

(b) A zero voltage reading at the digital voltmeter on the blockhouse vehicle power panel was detected at T - 55 seconds and the countdown was recycled to T - 7 minutes. An adjustment to the sensitivity potentiometer of the voltmeter corrected the problem.

(3) Atlas Propellant Tanking Test; the panel lights in the blockhouse indicated that the fuel ground fill and drain valve was open, though the valve was determined to be closed. An adjustment of the valve microswitch corrected the problem.

(4) Joint Flight Acceptance Composite Test (J-FACT); the Atlas autopilot pitch and yaw steering commands during the Guidance Command Test were not recorded. The replacement of a fuse on the recorder preamplifier corrected the problem. A broken wire in the patch panel of a recorder in the blockhouse necessitated the rechanneling of the Atlas Acoustica computer reset measurement (V1125X) during the test.

(5) Velocity Meter and Counter; analysis of velocity meter data on the day before launch showed a possible very small scale factor shift. Additional velocity meter check-outs could not verify a scale factor shift. The velocity meter and counter were replaced. Completion of confidence testing occurred early in the countdown.

Countdown and Launch

Lunar Orbiter IV was successfully launched from the Eastern Test Range Launch Complex 13 on May 4, 1967. Lift-off, 5.08-centimeter (2-in.) motion, occurred at 2225:00.571 GMT. The countdown proceeded as planned. The performance of the launch vehicle ground equipment during the countdown was satisfactory.

LUNAR ORBITER V

Prelaunch Activities

Major activities at the Eastern Test Range for Lunar Orbiter V are shown in table VII-I. All prelaunch tests were completed satisfactorily. The significant schedule delays and problems which occurred during the prelaunch period are as follows:

(1) Atlas Propellant Tanking Test:

(a) The panel lights in the blockhouse indicated that the fuel ground fill and drain valve was open, though the valve was determined to be closed. An adjustment of the valve microswitch corrected the problem.

(b) The premature and intermittent indication by panel lights in the blockhouse of 94 percent liquid oxygen fill caused a hold at T - 7 minutes during the liquid oxygen tanking test. The problem was evaluated and it was determined that the cause was splashing of liquid oxygen.

(c) The usual liquid oxygen tank oscillations were evident after switching to internal pressurization. Liquid oxygen tank pressures exceeded the redline value of 22.4 newtons per square centimeter (32.5 psi) by 0.76 newton per square centimeter (1.1 psi) during two separate changes to internal pressurization. These tank oscillations damped below redline values by T - 18 seconds and were essentially zero at T - 0.

(2) Booster Flight Acceptance Test (B-FACT) Number 2; the sustainer yaw telemetry transducer output remained at approximately 80 percent of full scale during the Guidance Command Tests. An investigation showed that the U-clamp holding the transducer to the engine was broken and it was replaced.

(3) Joint Flight Acceptance Composite Test (J-FACT):

(a) An Agena range safety command battery failed during the plus count. A subsequent test showed that the battery was faulty.

(b) Several voltage level changes occurred on the Atlas 400-hertz, three-phase output at about T + 100 seconds. This phenomenon has occurred before. Its cause was attributed to an interaction between the gyro heater magnetic amplifiers and the inverter.

(c) The signal to one of the Agena propellant pump drain line sealing mechanisms did not appear at the Pyrotechnic Monitor Console at T - 0. The cause was a broken wire at a connector in the console panel.

(4) Simulated Launch Test:

(a) The reading for the staging bottle helium pressure (measurement F1304P) became intermittent during the test. Replacement of the landline transducer for the measurement corrected the problem.

(b) The remote pressure gage for the Agena umbilical boom hydraulic pressure indicated low pressure. The replacement of the connector and the transducer in the readout system corrected the problem.

Countdown and Launch

Lunar Orbiter V was successfully launched from the Eastern Test Range Launch Complex 13 on August 1, 1967. Lift-off, 5.08-centimeter (2-in.) motion, occurred at 2233:00.352 GMT. A velocity meter replacement and severe weather conditions caused a delay of 2 hours and 24 minutes in the launch time. The countdown progressed smoothly until a velocity meter scale factor shift occurred at T - 295 minutes during the Velocity Meter Test. The velocity meter replacement resulted in a 55-minute hold at T - 155 minutes and also resulted in delaying gantry removal until completion of the new velocity meter checkout. The planned hold time compensated for this 55-minute unscheduled hold.

High winds and an approaching thunderstorm caused a 2 hour and 24 minute hold and required the return of the gantry around the vehicle at T - 90 minutes. The countdown proceeded smoothly after the hold.

The performance of the launch vehicle ground equipment during the countdown was satisfactory.

TABLE VII-I. - MAJOR PRELAUNCH ACTIVITIES AT EASTERN TEST RANGE,

LUNAR ORBITERS III, IV, AND V

Date (actual)			Event
Lunar Orbiter			
III	IV	V	
11/22/66	3/01/67	5/25/67	Agena arrived at ETR
12/05/66	3/07/67	5/27/67	Atlas arrived at ETR
12/08/66	3/13/67	6/08/67	Erection of Atlas
12/21/66	3/30/67	7/07/67	Booster Flight Acceptance Test (B-FACT) Number 1
1/13/67	4/06/67	6/28/67	Atlas Propellant Tanking Test Number 1
1/16/67			Atlas Propellant Tanking Test Number 2
1/21/67	4/28/67	7/19/67	Atlas-Agena mate
1/25/67	4/25/67	7/20/67	Booster Flight Acceptance Test (B-FACT) Number 2
1/27/67	5/01/67	7/24/67	Joint Flight Acceptance Composite Test (J-FACT)
1/31/67	[a]5/02/67	7/28/67	Simulated Launch Test
2/05/67 (GMT)	5/04/67 (GMT)	8/01/67 (GMT)	Launch

[a]Partial test.

VIII. CONCLUDING REMARKS

The Atlas-Agena vehicles performed satisfactorily for Lunar Orbiters III, IV, and V, as evidenced by the small midcourse correction which would have been required by the spacecraft of only 3.3 meters per second for Lunar Orbiter III, 10.8 meters per second for Lunar Orbiter IV, and 19.5 meters per second for Lunar Orbiter V. On these flights, the "over-the-nose" shroud jettison and Lunar Orbiter separation occurred without imparting any disturbance to either the Lunar Orbiter spacecraft at shroud jettison or to the spacecraft adapter at spacecraft separation.

The launch of Lunar Orbiter V in August 1967 concluded the Lunar Orbiter program. The Lunar Orbiter series of missions successfully supported the Apollo program and provided high quality scientific data to confirm Apollo landing sites and for investigation of the physical features of the moon in the vicinity of these sites.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, January 8, 1969,
441-05-00-02-22.

APPENDIX A

SEQUENCE OF MAJOR FLIGHT EVENTS, LUNAR ORBITERS III, IV, AND V

Event description	Lunar Orbiter					
	III		IV		V	
	Expected time, sec	Actual time, sec	Expected time, sec	Actual time, sec	Expected time, sec	Actual time, sec
Lift-off	0	0	0	0	0	0
Booster engine cutoff (BECO)	129.9	129.78	128.44	128.3	128.44	128.6
Booster engine jettison	132.9	132.99	131.44	131.54	131.44	131.7
Start Agena restart timer (SRT)	270.21	270.48	269.86	269.07	271.94	272.4
Sustainer engine cutoff (SECO)	288.3	288.02	290.36	289.43	290.28	288.6
Start Agena primary timer (SAT)	293.88	297.36	293.17	292.02	292.25	296.3
Vernier engine cutoff (VECO)	308.7	309.21	310.6	310.15	310.43	307.9
Shroud separation	311.0	311.43	313.00	312.53	312.5	310.3
Fire Atlas-Agena separation squibs	313.0	313.57	315.00	314.65	314.5	312.4
Fire first-burn ignition squibs	366.88	370.42	366.17	364.85	365.25	369.3
First-burn 90 percent chamber pressure	368.03	371.64	367.32	366.16	366.4	370.45
Agena first-burn cutoff	522.81	527.38	519.88	518.16	518.11	523.58
Fire second-burn ignition squibs	1104.21	1104.51	1760.86	1760.03	1878.94	1879.4
Second-burn 90 percent chamber pressure	1105.36	1105.65	1762.01	1761.17	1880.09	1880.1
Agena second-burn cutoff	1193.73	1194.37	1849.22	1848.49	1966.68	1967.6
Agena-spacecraft separation	1358.21	1358.55	2013.86	2013.07	2132.94	2133.42
Start yaw maneuver	1361.21	1361.46	2016.86	2016.9	2135.94	2136.39
Stop yaw maneuver	1421.21	1421.46	2076.86	2076.0	2195.94	2196.27
Initiate retrorocket fire	1958.21	1958.55	2613.86	2613.13	2732.94	2733.5
Retrorocket burnout	1974.21	1974.99	2629.86	2629.12	2748.88	2750.5

APPENDIX B

LAUNCH VEHICLE INSTRUMENTATION SUMMARY, LUNAR ORBITERS III, IV, AND V

by Edwin S. Jeris and Richard L. Greene

TABLE B-I. - ATLAS TELEMETRY

Measurement number	Description	Channel assignment (a)	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 V	
D3X	Range safety command destruct output	16-S	0 to 6 V dc	
E28V	Main dc voltage	18-1/31	20 to 35 V dc	
E51V	400 Hz, ac, phase A	18-11	105 to 125 V dc	
E52V	400 Hz, ac, phase B	18-29	105 to 125 V ac	
E53V	400 Hz, ac, phase C	18-41	105 to 125 V ac	
E95V	28 V dc guidance power input	13-15	20 to 35 V dc	
E96V	115 V ac, 400 Hz, phase A to guidance	13-37	105 to 125 V ac	
E151V	400-Hz, phase A waveform	10	0 to 150 V ac	
F1P	Liquid-oxygen tank helium pressure (absolute)	15-9	0 to 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
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

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

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

TABLE B-I. - Continued. ATLAS TELEMETRY

Measurement number	Description	Channel assignment (a)	Measurement range (low to high) SI Units	Measurement range (low to high) U.S. Customary Units
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 control number 2	15-13	0 to 5 V dc	
G282V	Rate beacon phase detector number 1	15-45	0 to 5 V dc	
G287V	Decoder pitch output	15-47	0 to 5 V dc	
G288V	Decoder yaw output	15-49	0 to 5 V dc	
G296V	Pulse beacon 15 V dc power supply	13-9	0 to 5 V dc	
G298V	Decoder 10 V dc power supply	13-13	0 to 5 V dc	
G354V	Rate beacon 25 to 30 V dc power supply	13-11	0 to 5 V dc	
G590V	Discrete binary 1	16-33	0 to 5 V dc	
G591V	Discrete binary 2	16-35	0 to 5 V dc	
G592V	Discrete binary 4	16-37	0 to 5 V dc	
G593V	Discrete binary 8	16-39	0 to 5 V dc	
G363X	Jettison shroud	6-S	(b)	
G364X	Start Agena restart timer	2	(b)	
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-vernier 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	18-7/37	0 to 413.6 N/cm^2	0 to 600 psi
M79A	Missile axial accelerometer fine	7	-0.5 to 0.5 g's	
M30X	Missile 2-in. (5.08-cm) motion	7-S	(b)	

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

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

TABLE B-I. - Continued. ATLAS TELEMETRY

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			SI Units	U.S. Customary Units
M32X	Jettison system valve command	5-S	(b)	
P83B	Booster 2 pump speed	15-41	4000 to 7000 rpm	
P84B	Booster 1 pump speed	4	6000 to 6950 rpm	
P349B	Sustainer pump speed	3	9900 to 11 200 rpm	
P529D	Sustainer main liquid-oxygen valve	13-43	0^{o} to 90^{o}	
P330D	Sustainer fuel valve position	13-35	23^{o} to 54^{o}	
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 551.6 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

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

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

TABLE B-I. - Continued. ATLAS TELEMETRY

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			SI Units	U.S. Customary Units
P339P	Sustainer gas generator discharge pressure (absolute)	18-55	0 to 551.6 N/cm^2	0 to 800 psi
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	277.5 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° to 3°	
S63D	Yaw displacement gyro signal	15-39	-3° to 3°	
S252D	Booster 1 yaw roll	16-15	-6° to 6°	
S253D	Booster 2 yaw roll	16-55	-6° to 6°	
S254D	Booster 1 pitch	7	-6° to 6°	
S255D	Booster 2 pitch	16-1	-6° to 6°	
S256D	Sustainer yaw	16-41	-4° to 4°	
S257D	Sustainer pitch	16-45	-4° to 4°	
S258D	Vernier 1 pitch roll	16-3	-70° to 70°	
S259D	Vernier 2 pitch roll	16-5	-70° to 70°	
S260D	Vernier 1 yaw	16-7	-5° to 55°	
S261D	Vernier 2 yaw	16-9	-5° to 55°	
S52R	Roll rate gyro signal	9	-8 to 8 deg/sec	
S53R	Pitch rate gyro signal	8	-6 to 6 deg/sec	
S54R	Yaw rate gyro signal	5	-6 to 6 deg/sec	
S190V	Pitch gyro torque amplifier	15-43	-1 to 1 V ac	
S209V	Programmer 28 V dc test	6	20 to 35 V dc	
S236X	Booster cutoff discrete	9-S	(b)	
S241X	Sustainer cutoff discrete	9-S	(b)	
S245X	Vernier cutoff discrete	9-S	(b)	
S248X	Release payload discrete	9-S	(b)	

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

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

TABLE B-I. - Concluded. ATLAS TELEMETRY

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			SI Units	U.S. Customary Units
S290X	Programmer output Spare Booster jettison Enable discretes	16-29	0 to 28 V dc	
S291X	Programmer output Booster engine cutoff Sustainer engine cutoff Vernier engine cutoff	16-31	0 to 28 V dc	
S359X	Booster staging backup	5-S	(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 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	Acoutica valve position feedback	13-33	0 to 5 V dc	
U132V	Acoustica station counter output	13-7	0 to 5 V dc	
U134V	Acoustica time shared oscillator output	18-23/53	0 to 5 V dc	
U135V	Acoustica sensor signal	18-39	0 to 5 V dc	
U605V	Acoustica time shared integrator switch	18-35	0 to 5 V dc	
Y44P	Interstage adapter pressure (absolute)	13-23	0 to 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 Interrange Instrumentation Group (IRIG) subcarrier channel used; second number indicates commutated position for measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

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

TABLE B-II. - AGENA TELEMETRY

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			SI Units	U.S. Customary Units
A4	Tangential accelerometer	10	-5 to 5 g's	
A5	Tangential accelerometer	11	-5 to 5 g's	
A9	Longitudinal accelerometer	[b]9	-4 to 12 g's	
A52	Shroud separation	16-19/49	(c)	
A007	Spacecraft separation monitor 1	9	0 to 4.45 cm	0 to 1.75 in.
A008	Spacecraft separation monitor 2	13	0 to 4.45 cm	0 to 1.75 in.
A009	Spacecraft separation monitor 3	12	0 to 4.45 cm	0 to 1.75 in.
A105	Spacecraft V-band tension 1	5	0 to 18.7×10^3 N	0 to 4.2×10^3 lb
A106	Spacecraft V-band tension 2	6	0 to 18.7×10^3 N	0 to 4.2×10^3 lb
A519	Shroud pressure internal (absolute)	16-17	0 to 10.32 N/cm^2	0 to 15 psi
A520	Spacecraft adapter longitudinal vibration	17	-20 to 20 g's	
A523	Radial accelerometer	8	-5 to 5 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)	16-23, 53	0 to 1034 N/cm^2	0 to 1500 psi
B12	Fuel venturi inlet pressure (absolute)	16-24, 54	0 to 1034 N/cm^2	0 to 1500 psi
B13	Switch group Z	15-7/22/37/52	(c)	
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		(d)	

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

[b]Subcarrier channels 9, 12, and 13 are switched from accelerometers to spacecraft separation monitors after Agena second burn.

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

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

TABLE B-II. - Continued. AGENA TELEMETRY

Measurement number	Description	Channel assignment (a)	Measurement range (low to high)	
			SI Units	U.S. Customary Units
B91	Combustion chamber pressure number 3 (gage)	15-4/34	328 to 379 N/cm^2	475 to 550 psi
B130	Propellant isolation valve monitor	15-11/13/23/27/31/33/41/45/48/51/53/56	(c)	
C1	28-V dc unregulated supply	16-40	22 to 30 V dc	
C3	28-V dc regulator (guidance and control)	15-12	22 to 30 V dc	
C4	28-V dc unregulated current	16-13/44	0 to 100 A	
C5	-28-V dc regulator (guidance and control)	15-30	-30 to -22 V dc	
C21	400-Hz, 3-phase, inverter temperature	15-14	255 to 367 K	0° to 200° F
C31	400-Hz, 3-phase, bus phase AB	15-18	90 to 130 V ac	
C32	400-Hz, 3-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	(c)	
D41	Horizon sensor pitch	16-45	-5° to 5°	
D42	Horizon sensor roll	16-46	-5° to 5°	
D46	Gas valve cluster temperature 1	15-39	228 to 339 K	-50° to 150° F
D47	Gas valve cluster temperature 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 heat temperature (right)	15-47	228 to 367 K	-50° to 200° F
D55	Horizon sensor head temperature (left)	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	16-41	-50 to 50 deg/min -4 to 4 deg/min	

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

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

TABLE B-II. - Concluded. AGENA TELEMETRY

Measurement number	Description	Channel assignment (a)	Measurement number (low to high)	
			SI Units	U.S. Customary Units
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	16-36	-10° to 10° -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	
D74	Yaw gyro output	16-39	-10° to 10° -5° to 5°	
D75	Roll gyro output	16-42	-10° to 10° -5° to 5°	
D83	Velocity meter acceleration	14	0 to 2000 pulses/sec	
D86	Velocity meter cutoff switch	16-28	(c)	
D88	Velocity meter counter	14	Binary code (50 pulses/sec)	
D129	Inertial reference package internal case temperature	15-54	255 to 342 K	0° to 155° F
D149	Gas valves 1 to 6 current	7	(f)	
H47	Beacon receiver pulse repetition frequency	16-15	0 to 1600 pulses/sec	
H48	Beacon transmitter pulse repetition frequency	16-16	0 to 1600 pulses/sec	
H101	Safe-Arm-Fire destruct number 1	16-20	(c)	
H103	Safe-Arm-Fire destruct number 2	16-21	(c)	
H204	Number 1 dc-dc converter	15-50	22 to 30 V dc	
H218	Telemetry transmitter temperature	16-30	283 to 350 K	50° to 170° F
H354	Destruct receiver number 1 signal level	16-25	0 to 40 V	
H364	Destruct receiver number 2 signal level	16-26	0 to 40 V	
H400	Number 2 dc-dc converter	16-12	22 to 30 V dc	
LA018	Spacecraft adapter skin temperature	16-31	255 to 339 K	0° to 150° F
PL30	Spacecraft longitudinal accelerometer	[b]13	-10 to 10 g's	
PL37	Spacecraft radial accelerometer	[b]12	-10 to 10 g's	

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

[b]Subcarrier channels 9, 12, and 13 are switched from accelerometers to spacecraft separation monitors after Agena second burn.

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

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

[e]This measurement was not used on Lunar Orbiter III - Agena vehicle.

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

APPENDIX C

TRACKING AND DATA ACQUISITION

by Richard L. Greene

The launch vehicle trajectories as projected on world maps for each Lunar Orbiter flight are presented in figures C-1 to C-3. Tracking and telemetry stations provided radar data (vehicle position, velocity, and acceleration) and telemetry data (vehicle subsystem performance) for the vehicles during the near-Earth phase of the Lunar Orbiter missions. Coverage was provided by the Eastern Test Range (ETR) uprange stations at Cape Kennedy, Grand Bahama Island, Grand Turk Island, and Antigua, and the ETR downrange stations at Ascension, Pretoria (South Africa), several Range Instrumentation Ships, and one Range Instrumentation Aircraft. Supplementing the ETR downrange station coverage were the NASA Manned Space Flight Network (MSFN) stations at Bermuda, Canary Islands, Tananarive (Malagasy Republic), and Carnarvon (Australia).

Telemetry Data

Telemetry data from the Atlas-Agena launch vehicle were recorded on magnetic tape by telemetry stations during all Atlas and Agena engine operations, Agena-spacecraft separation, Agena 180° yaw maneuver, and the Agena retromaneuver. These data were used for postflight analyses of launch vehicle performance. Real-time monitoring of specific Atlas and Agena functions was provided for verification of the occurrence of significant flight events. A submarine cable between the ETR uprange stations and Cape Kennedy permitted real-time monitoring of vehicle telemetered signals through Agena engine first cutoff. The subsequent flight events were monitored by the ETR downrange stations and by the MSFN stations. These events were reported back to Cape Kennedy in "near" real-time by ETR single side band radio links and the NASA Communication Network (NASCOM) voice and data circuits. Figures C-4 to C-6 show the specific telemetry coverage provided by each telemetry station for each Lunar Orbiter flight.

Radar Data

C-band radar data (time, elevation, azimuth, range) were provided for real-time operations and postflight analyses. Real-time radar data were provided for monitoring the launch vehicle flight performance for range safety purposes and for assisting the downrange stations in acquiring track of the vehicle. These data were also used for computation of parking orbit elements, injection conditions at Agena engine first cutoff, injection conditions at Agena engine second cutoff, and the Agena final orbit elements. The specific radar coverage by each tracking station is presented in figures C-7 to C-9 for each Lunar Orbiter flight.

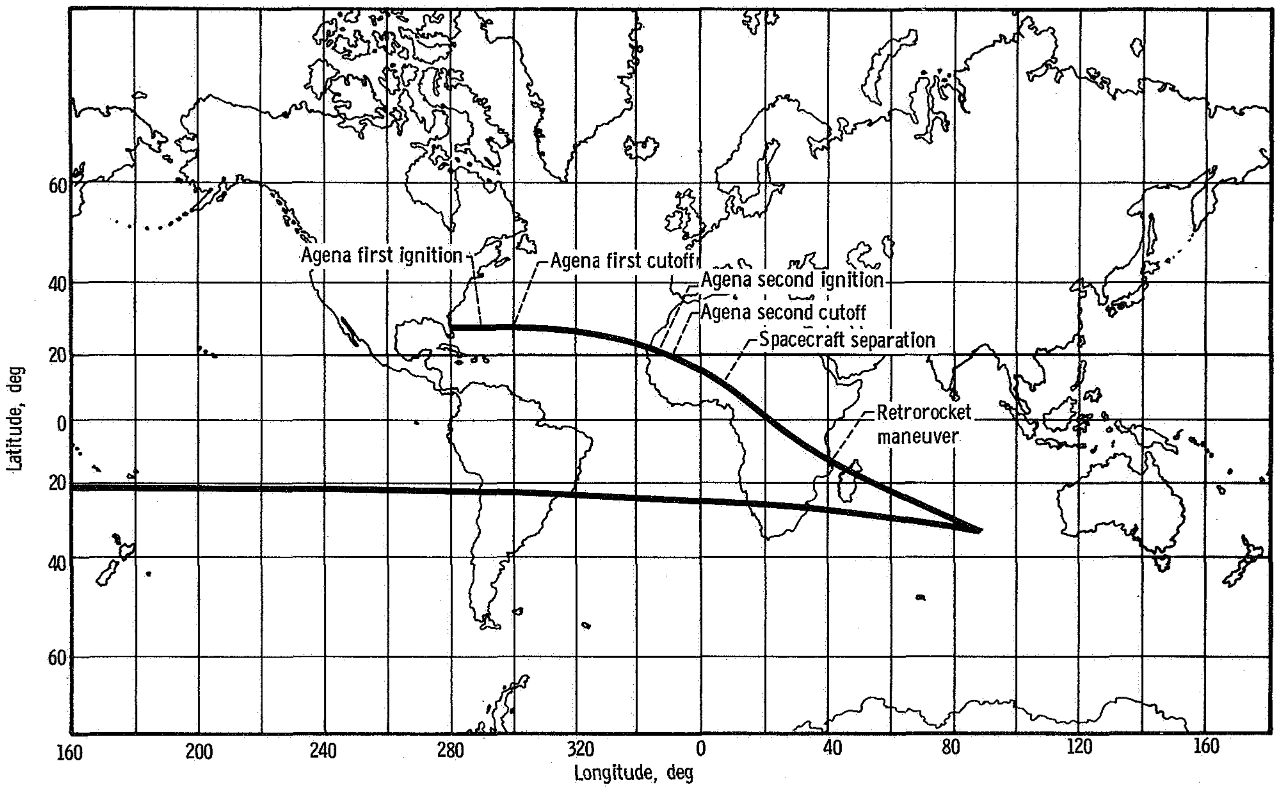


Figure C-1. - Ground trace for Lunar Orbiter III.

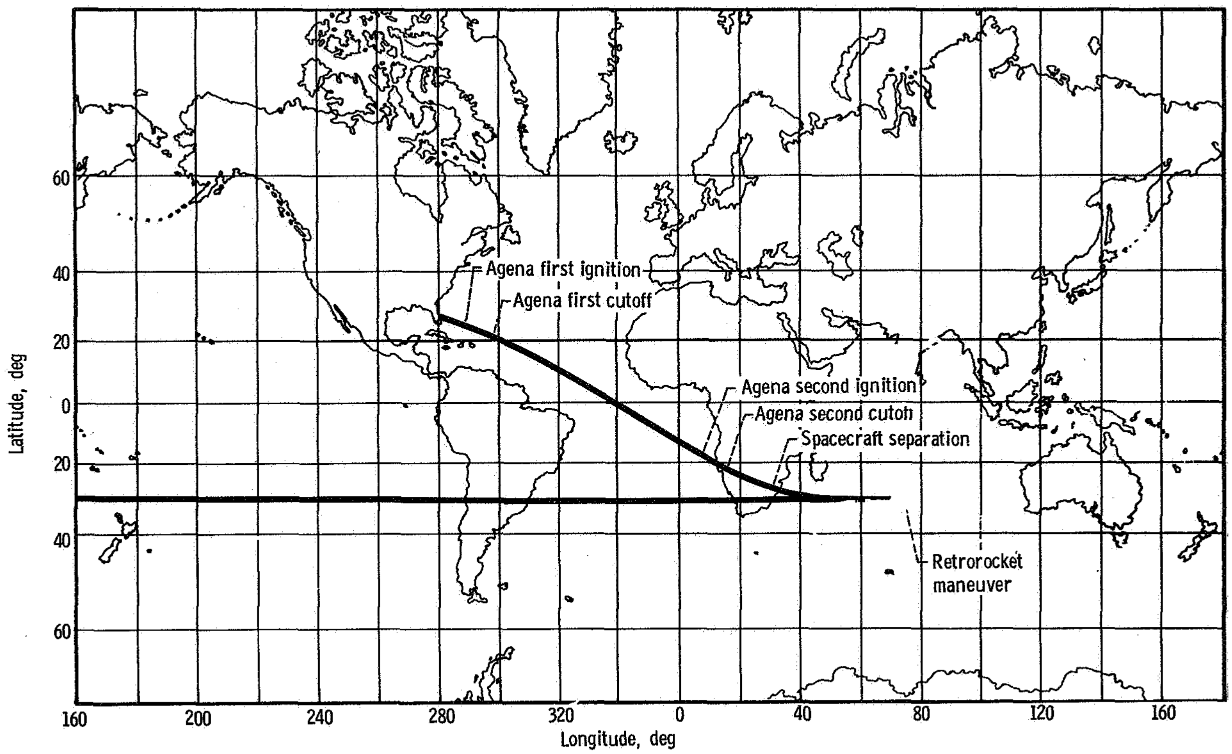


Figure C-2. - Ground trace for Lunar Orbiter IV.

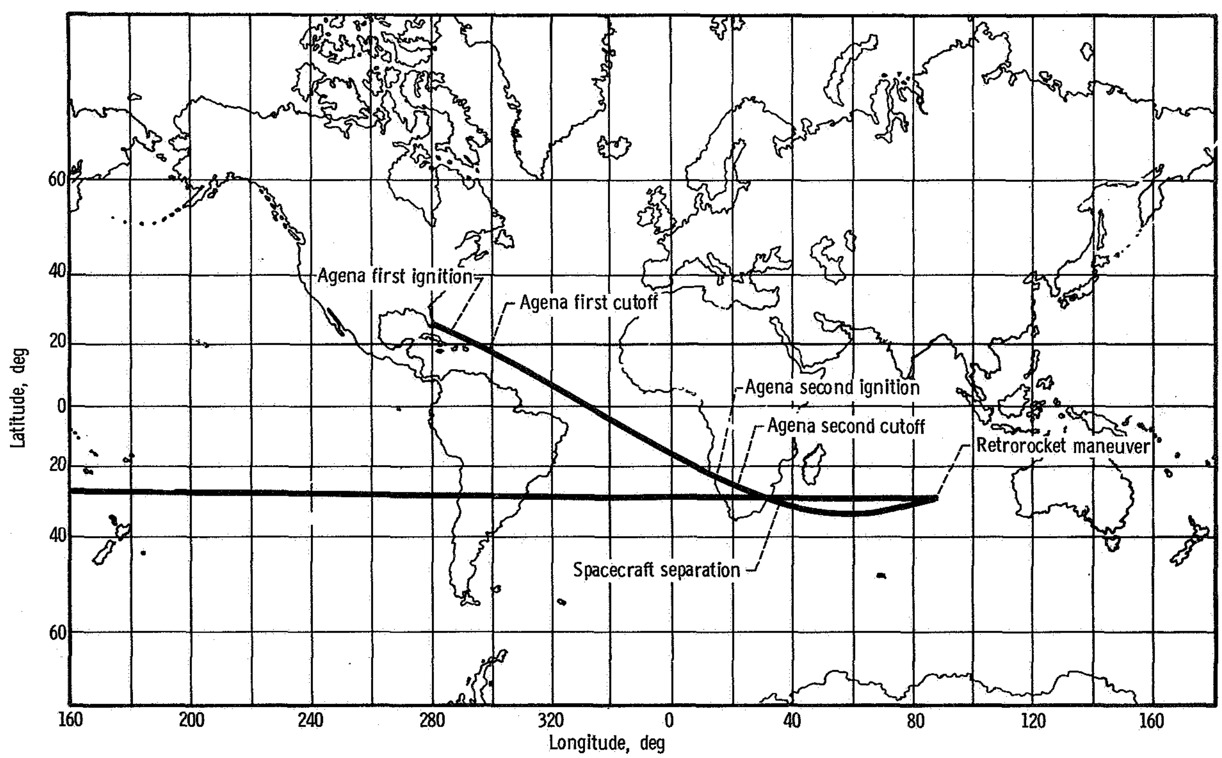


Figure C-3. - Ground trace for Lunar Orbiter V.

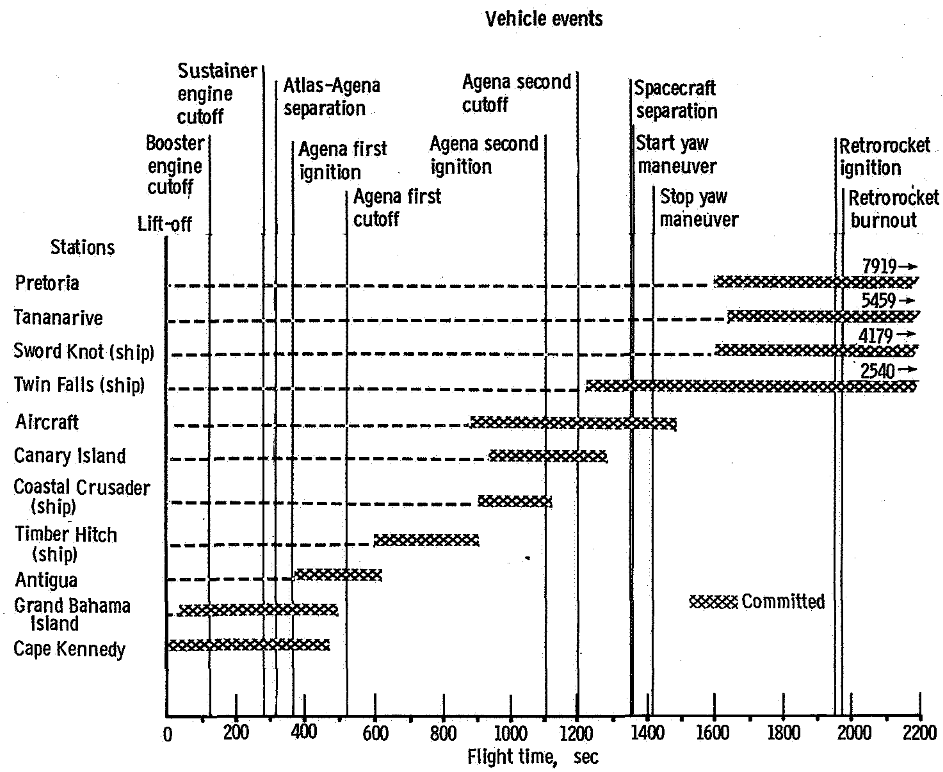


Figure C-4. - Launch vehicle telemetry coverage, Lunar Orbiter III.

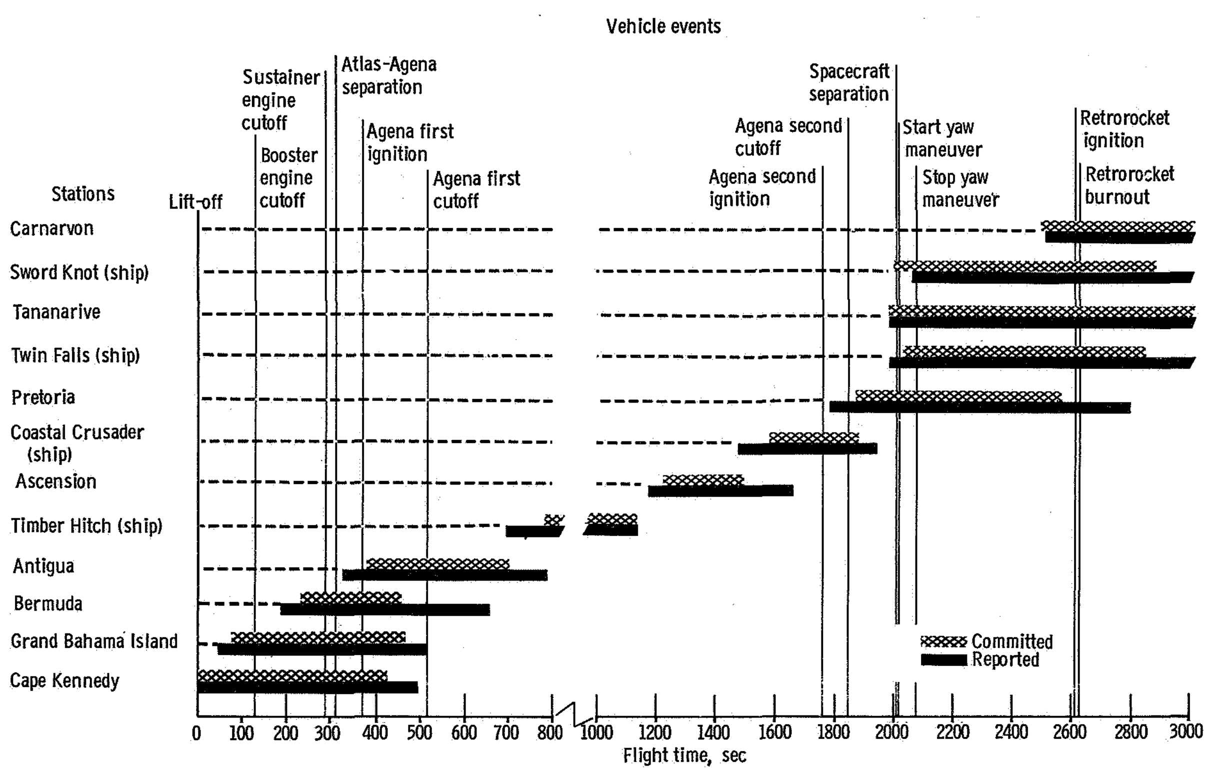


Figure C-5. - Launch vehicle telemetry coverage, Lunar Orbiter IV.

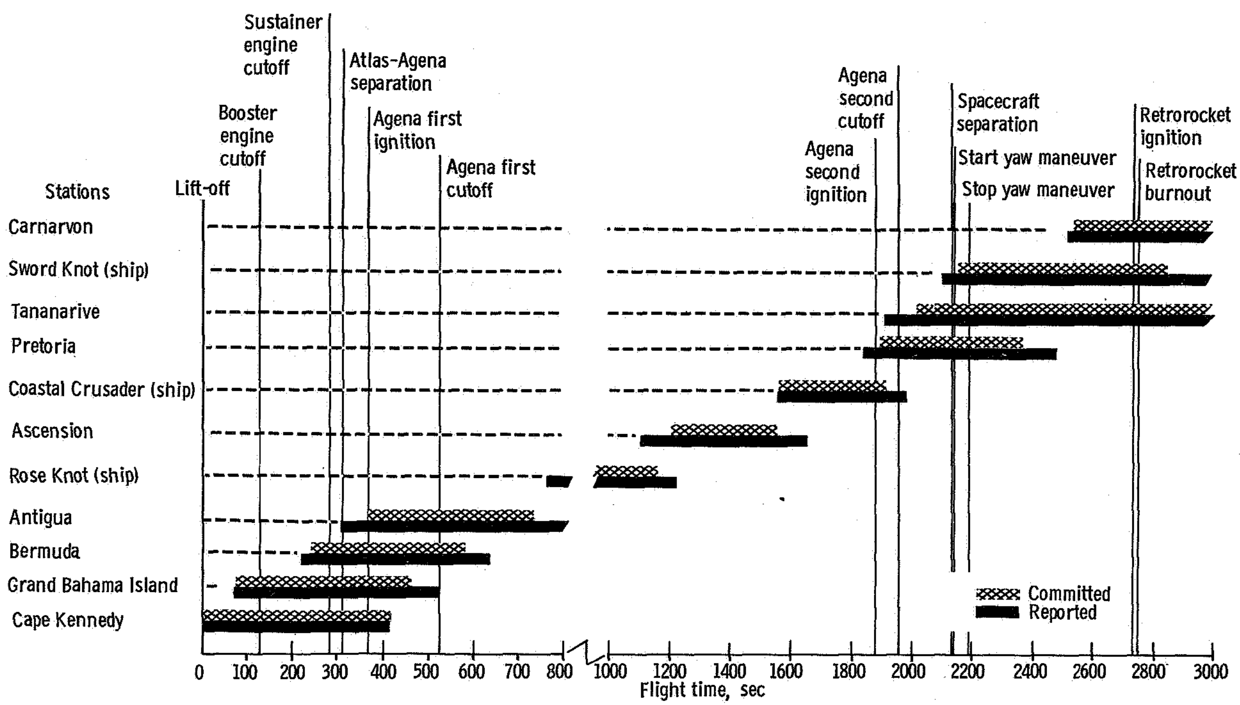


Figure C-6. - Launch vehicle telemetry coverage, Lunar Orbiter V.

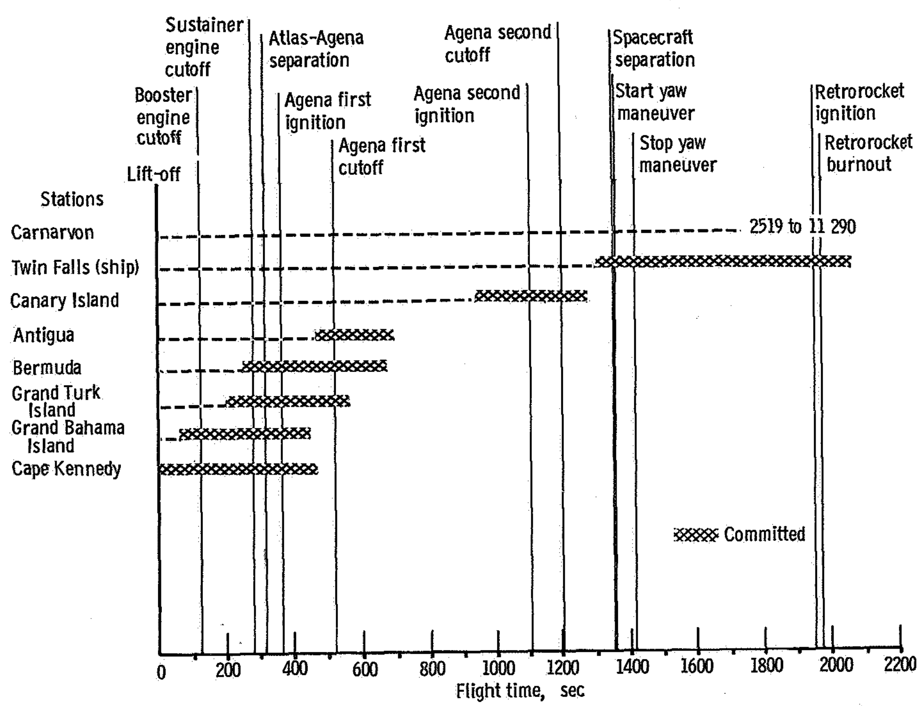


Figure C-7. - Launch vehicle radar coverage. Lunar Orbiter III.

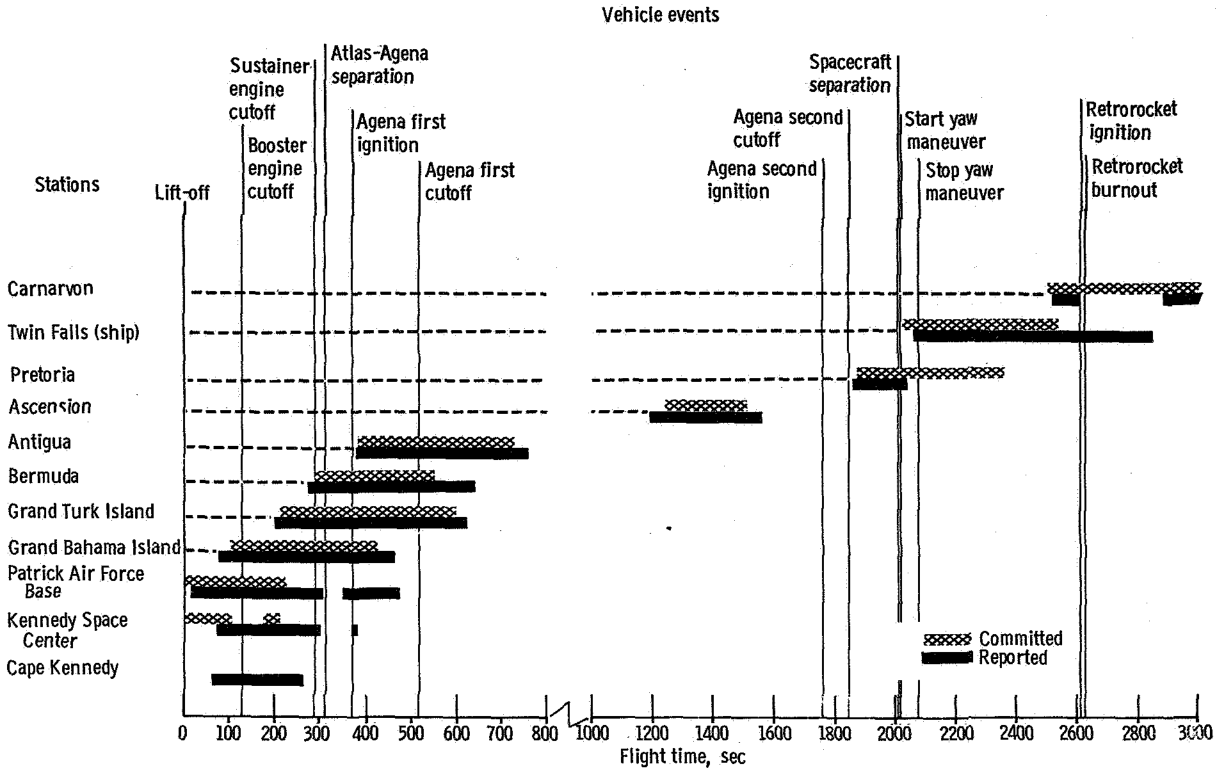


Figure C-8. - Launch vehicle radar coverage, Lunar Orbiter IV.

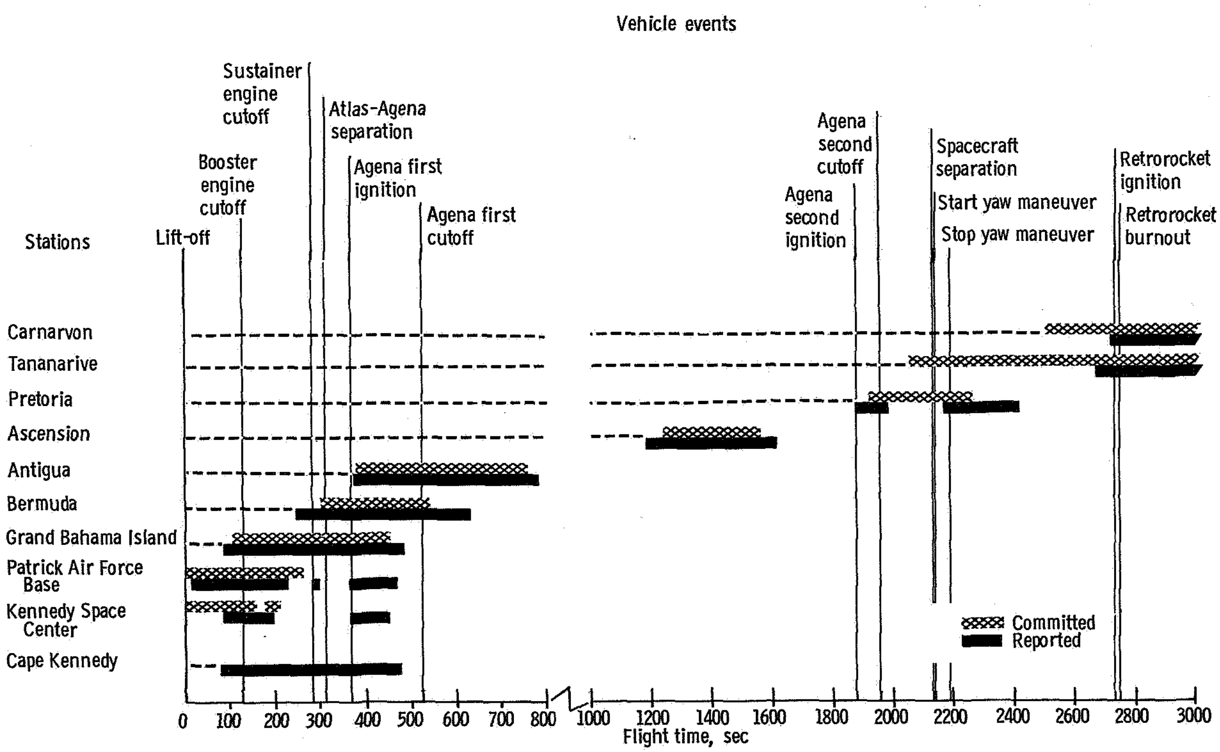


Figure C-9. - Launch vehicle radar coverage, Lunar Orbiter V.

APPENDIX D

VEHICLE FLIGHT DYNAMICS

by Robert W. York

Flight dynamic data for Lunar Orbiters III, IV, and V were obtained from four accelerometers installed in the Agena forward section, from two accelerometers on the spacecraft adapter, and from two accelerometers mounted on the spacecraft. A summary of instrument locations and characteristics is presented in figure D-1. The following table presents the actual flight times during which significant dynamic disturbances were recorded:

Event causing disturbance	Lunar Orbiter flight		
	III	IV	V
	Time of dynamic disturbance, sec after lift-off		
Lift-off	0	0	0
Transonic region	50 to 70	50 to 70	50 to 70
Booster engine cutoff (BECO)	129.7	128.43	128.75
Sustainer engine cutoff (SECO)	288.0	289.53	288.58
Horizon sensor fairing jettison	309.2	310.15	307.92
Shroud separation	311.4	312.50	310.29
Atlas-Agena separation	313.6	314.64	312.43
Agena engine first ignition	371.7	366.17	370.44
Agena engine first cutoff	527.4	518.17	523.56
Agena engine second ignition	1105.6	1761.17	1880.51
Agena engine second cutoff	1194.3	1848.47	1967.60

For the Lunar Orbiter III, IV, and V flights the dynamic environmental data indicates that the vibration and shock levels were comparable to those experienced on other Lunar Orbiter flights. Environmental data for Lunar Orbiter IV recorded at the time (near the flight event) when the dynamic disturbance occurred are presented in figures D-2 to D-23. These data are typical of the data on the three flights and consequently similar data for Lunar Orbiters III and V have not been included in this report. However, data for Lunar Orbiter III have been included (figs. D-24 to D-30) to show four measurement anomalies that occurred on this flight.

Accelerometer A9 (Channel 9, fig. D-24) was inadvertently reversed when it was mounted in the Agena forward section. Instead of having a range of -4 to 12 g's, this accelerometer had a range of -12 to 4 g's. Operation of the accelerometer was satisfactory until the vehicle acceleration exceeded 4 g's at approximately 8 seconds prior to booster engine cutoff and the instrument reading went off scale. Also, accelerometer data from channels 8 and 11 for Agena engine first cutoff, second ignition, and second cutoff (figs. D-25, D-27, and D-29) indicated abnormally high transient acceleration levels. The validity of these high transients is questionable since, on previous Agena flights, accelerometers of the same type have often been suspected of having been damaged during the Atlas booster phase of flight. Damage (or overstress) is evidenced by asymmetrically shaped output traces similar to those experienced on this flight. Since the accelerometer outputs do not conform to an expected waveform, it can be assumed either that the event environment was abnormal or that the accelerometers did not measure the actual environment. The latter assumption is the more likely because other Agena instrumentation (channels 9, 10, 12, 13, 17, and 18, shown in figs. D-25 to D-30) indicate normal conditions existed during Agena engine first cutoff, second ignition, and second cutoff. Additional facts which support the questionable reliability of the accelerometers are that (1) these accelerometers exhibited a simultaneous zero or dc bias shift several times during the ascent phase of flight, (2) historically, these accelerometers have had a high failure rate, and (3) the accelerometers are known to be sensitive to accelerations perpendicular to their sensitive axis.

Tables D-I to D-III summarize the maximum acceleration levels and corresponding frequencies experienced during significant flight events for each flight. All acceleration levels are shown in g's zero-to-peak.

TABLE D-I. - SUMMARY OF DYNAMIC ENVIRONMENT, LUNAR ORBITER III

Event causing disturbance	Time of dynamic disturbance, sec after lift-off	Accelerometer							
		Channel 8		Channel 9		Channel 10		Channel 11	
		Measurement							
		A523 Radial		A9 Longitudinal		A4 Tangential		A5 Tangential	
		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	(a)	0.7	(a)	0.6	(b)	0.5	(a)	0.6
Transonic region	50 to 70	(a)	1.3	(a)	.6	(b)	.7	(a)	.5
Booster engine cutoff (BECO)	129.7	8	.3	(a)	.2	65	.7	65	.8
Sustainer engine cutoff (SECO)	288.0	90	.4	85	.4	85	.3	85	.4
Horizon sensor fairing jettison	309.2	80	2.4	100	3.6	(b)	.9	(a)	2.3
Shroud separation	311.4	80	1.1	90	4.0	(b)	1.1	(a)	1.4
Atlas-Agena separation	313.6	100	1.2	(a)	2.3	(b)	1.2	(a)	1.1
Agena engine first ignition	371.7	88	.2	95	.3	(a)	.2	(a)	.2
Agena engine first cutoff	527.4	(a)	1.7	80	.8	104	.3	(a)	1.8
Agena engine second ignition	1105.6	(a)	2.1	(a)	.7	(a)	.4	(a)	1.9
Agena engine second cutoff	1194.3	(a)	2.5	(a)	1.2	(a)	.3	(a)	1.7

[a]No measurable frequency or acceleration.
[b]Frequency greater than 800 Hz.

TABLE D-I. - Concluded. SUMMARY OF DYNAMIC ENVIRONMENT, LUNAR ORBITER III

Event causing disturbance	Time of dynamic disturbance, sec after lift-off	Accelerometer							
		Channel 12		Channel 13		Channel 17		Channel 18	
		Measurement							
		PL 37 Radial		PL 30 Longitudinal		A520 Longitudinal		A524 Radial	
		Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)
Lift-off	0	(a)	0.3	(a)	0.6	(b)	8.0	(b)	9.2
Transonic region	50 to 70	(a)	(a)	400	.6	(b)	8.0	(b)	7.8
Booster engine cutoff (BECO)	129.7	9.6	.5	(a)	.5	(a)	.3	9.6	.4
Sustainer engine cutoff (SECO)	288.0	83	.2	(a)	.7	100	.4	80	.4
Horizon sensor fairing jettison	309.2	(a)	.2	450	1.2	(b)	10.4	(b)	18.0
Shroud separation	311.4	(a)	.4	440	3.0	(b)	11.2	(b)	Over 20
Atlas-Agena separation	313.6	640	.4	600	2.0	(b)	11.0	(b)	Over 20
Agena engine first ignition	371.7	(a)	.5	52	.5	72	4.4	72	1.6
Agena engine first cutoff	527.4	45	1.0	(a)	1.5	(b)	1.0	(b)	.6
Agena engine second ignition	1105.6	40	.8	30	1.4	72	1.2	(a)	.6
Agena engine second cutoff	1194.3	86	1.3	76	4.2	(b)	3.2	(b)	1.2

[a]No measurable frequency or acceleration.
[b]Frequency greater than 800 Hz.

TABLE D-II. - SUMMARY OF DYNAMIC ENVIRONMENT, LUNAR ORBITER IV

Event causing disturbance	Time of dynamic disturbance, sec after lift-off	Accelerometer							
		Channel 8		Channel 9		Channel 10		Channel 11	
		Measurement							
		A523 Radial		A9 Longitudinal		A4 Tangential		A5 Tangential	
		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	(a)	0.5	5.0	1.0	90	0.5	(a)	1.0
Transonic region	50 to 70	(a)	1.5	(a)	1.2	(a)	.9	(a)	.9
Booster engine cutoff (BECO)	128.43	68	.2	15	.7	64	.8	64	1.1
Sustainer engine cutoff (SECO)	289.53	90	.2	90	.2	82	.2	80	.3
Horizon sensor fairing jettison	310.15	48	2.1	(a), (b) 100	5.0, (b) 1.0	100	1.1	100	4.4
Shroud separation	312.50	54	2.0	90	3.8	(c), (b) 100	.3, (b) 1.4	120	2.0
Atlas-Agena separation	314.64	85	1.6	120	1.9	(c), (b) 104	.4, (b) 1.8	104	1.1
Agena engine first ignition	366.17	56	.2	54	.4	52	.2	(a)	.2
Agena engine first cutoff	518.17	64	.5	74	.9	110	.3	110	.5
Agena engine second ignition	1761.17	48	.4	80	.4	80	.3	80	.4
Agena engine second cutoff	1848.47	56	.3	72	.8	28, (b) 200	.5, (b) .7	44, (b) 200	.5, (b) .7

[a]No measurable frequency or acceleration.
[b]Double entries indicate the frequency and acceleration levels of two superimposed vibrations.
[c]Frequency greater than 800 Hz.

TABLE D-II. - Concluded. SUMMARY OF DYNAMIC ENVIRONMENT, LUNAR ORBITER IV

Event causing disturbance	Time to dynamic disturbance, sec after lift-off	Accelerometer							
		Channel 12		Channel 13		Channel 17		Channel 18	
		Measurement							
		PL 37 Radial		PL 30 Longitudinal		A520 Longitudinal		A524 Radial	
		Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)
Lift-off	0	5.0	0.6	400	0.8	(b)	12.0	(b)	6.0
Transonic region	50 to 70	(a)	.3	800	.5	(b)	15.2	(b)	6.8
Booster engine cutoff (BECO)	128.43	32	.4	64	1.2	14.5	.8	70	.2
Sustainer engine cutoff (SECO)	289.53	38	.6	24	.7	100	.4	120	.4
Horizon sensor fairing jettison	310.15	120	.4	180	2.2	800	Over 20	750	Over 20
Shroud separation	312.50	400	.4	350	2.9	750	Over 20	750	Over 20
Atlas-Agena separation	314.64	(b)	.4	800	1.8	(b)	Over 20	(b)	Over 20
Agena engine first ignition	366.17	50	.7	56	.6	80	.2	60	.2
Agena engine first cutoff	518.17	40	1.2	42	1.2	88	1.4	110	.8
Agena engine second ignition	1761.17	42	1.2	48	1.5	80	.6	90	.6
Agena engine second cutoff	1848.47	42	1.5	40	3.0	68	1.2	80	.8

[a]No measurable frequency or acceleration.

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

[c]Frequency greater than 800 Hz.

TABLE D-III. - SUMMARY OF DYNAMIC ENVIRONMENT, LUNAR ORBITER V

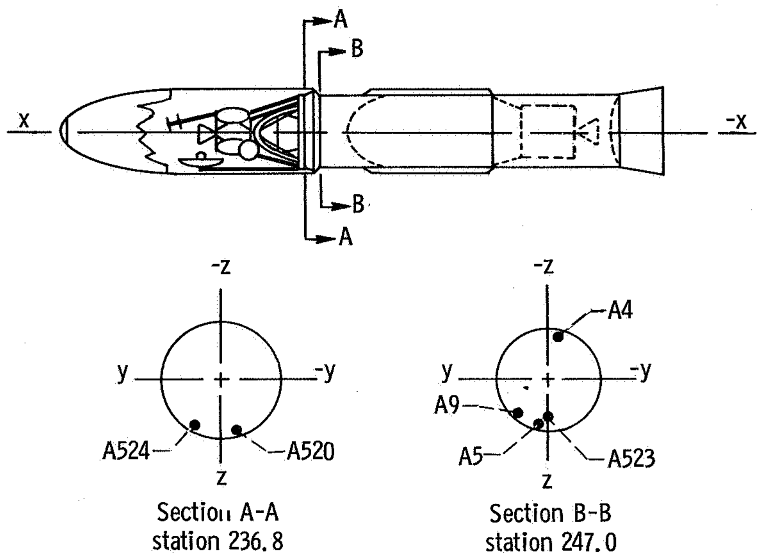
Event causing disturbance	Time of dynamic disturbance, sec after lift-off	Accelerometer							
		Channel 8		Channel 9		Channel 10		Channel 11	
		Measurement							
		A523 Radial		A9 Longitudinal		A4 Tangential		A5 Tangential	
		Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)
Lift-off	0	100	0.2	5.0	0.5	30	0.6	30	0.5
Transonic region	50 to 70	60	1.0	110	1.1	70	.5	70	.8
Booster engine cutoff (BECO)	128.75	4	.3	14	.6	65	1.2	65	1.1
Sustainer engine cutoff (SECO)	288.58	8	.1	24	.2	84	.1	80	.1
Horizon sensor fairing jettison	307.92	55	2.5	70	3.8	100	1.3	100	2.0
Shroud separation	310.29	77	4.4	100	9.2	700	2.0	0.02 sec pulse	3.2
Atlas-Agena separation	312.43	77	2.0	70	4.6	700	1.7	80	1.0
Agena engine first ignition	370.44	90	.1	60	.2	54	.2	86	.2
Agena engine first cutoff	523.56	60	.6	80	.8	83	.6	40	.4
Agena engine second ignition	1880.51	75	.5	80	.3	70	.2	93	.2
Agena engine second cutoff	1967.60	75	.5	120	.8	70	.4	58	.6

[a]Frequency greater than 800 Hz.
[b]No measurable frequency or acceleration.

TABLE D-III. - Concluded. SUMMARY OF DYNAMIC ENVIRONMENT, LUNAR ORBITER V

Event causing disturbance	Time of dynamic disturbance, sec after lift-off	Accelerometer							
		Channel 12		Channel 13		Channel 17		Channel 18	
		Measurement							
		PL 37 Radial		PL 30 Longitudinal		A520 Longitudinal		A524 Radial	
		Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)	Frequency, Hz	g's (zero to peak)
Lift-off	0	80	0.5	5	0.7	5	1.0	(a)	7.0
Transonic region	50 to 70	(b)	.2	33	.2	(a)	5	(a)	6.0
Booster engine cutoff (BECO)	128.75	4	.5	15	1.0	15	1.0	25	.5
Sustainer engine cutoff (SECO)	288.58	40	.5	23	.6	23	.5	.9	.5
Horizon sensor fairing jettison	307.92	100	.4	400	2.0	(a)	Over 20	(a)	Over 20
Shroud separation	310.29	18	.4	400	3.0	(a)	Over 20	(a)	Over 20
Atlas-Agena separation	312.43	130	.2	200	1.8	(a)	Over 20	(a)	19
Agena engine first ignition	370.44	85	.3	40	.6	750	3.5	80	.5
Agena engine first cutoff	523.56	40	1.1	40	1.6	80	1.6	80	.5
Agena engine second ignition	1880.51	45	.8	32	1.6	80	.5	80	.5
Agena engine second cutoff	1967.60	40	1.5	30	3.5	120	1.0	50	1.0

[a]Frequency greater than 800 Hz.
[b]No measurable frequency or acceleration.



Measurement number	Channel	Measurement description	Vehicle station	Frequency response, Hz	Range, g's
A4	10	Tangential acceleration, forward section	247.0	0 to 80	±5
A5	11	Tangential acceleration, forward section	247.0	0 to 110	±5
[a]A9	9	Longitudinal acceleration, forward section	247.0	0 to 60	+2 to -4
A520	17	Longitudinal acceleration, spacecraft adapter	236.8	20 to 2000	±20
A523	8	Radial acceleration, spacecraft adapater	247.0	0 to 45	±5
A524	18	Radial vibration, spacecraft adapter	236.8	20 to 2000	±20
PL30	13	Longitudinal acceleration, spacecraft	Not shown	5 to 1100	±10
PL37	12	Radial acceleration, spacecraft	Not shown	5 to 800	±10

[a]Installed reversed on Lunar Orbiter III. Data for this accelerometer reflects the reversed installation.

Figure D-1. - Dynamic flight instrumentation for Lunar Orbiters III, IV, and V.

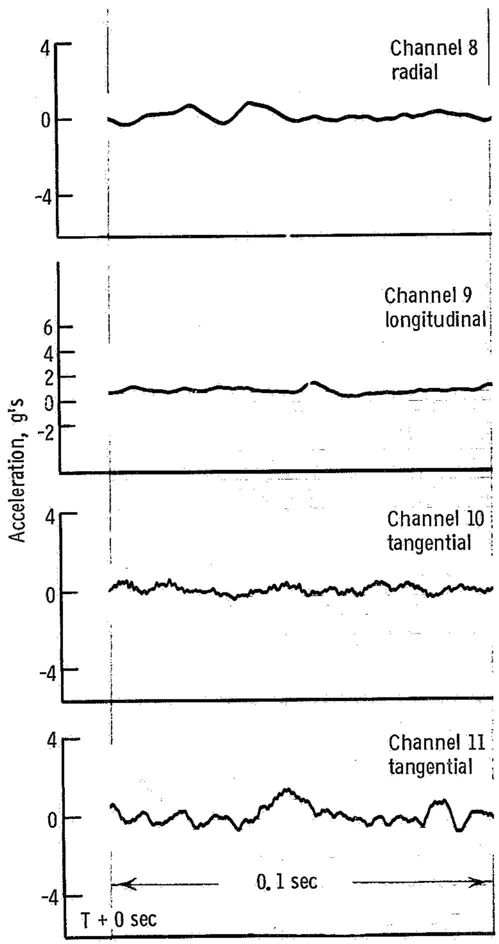


Figure D-2. - Dynamic data at lift-off, Lunar Orbiter IV.

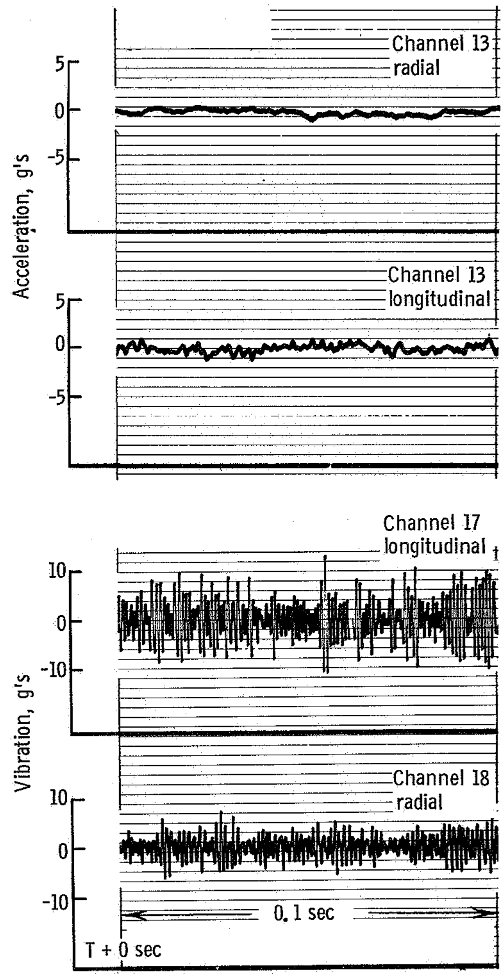


Figure D-3. - Dynamic data at lift-off, Lunar Orbiter IV.

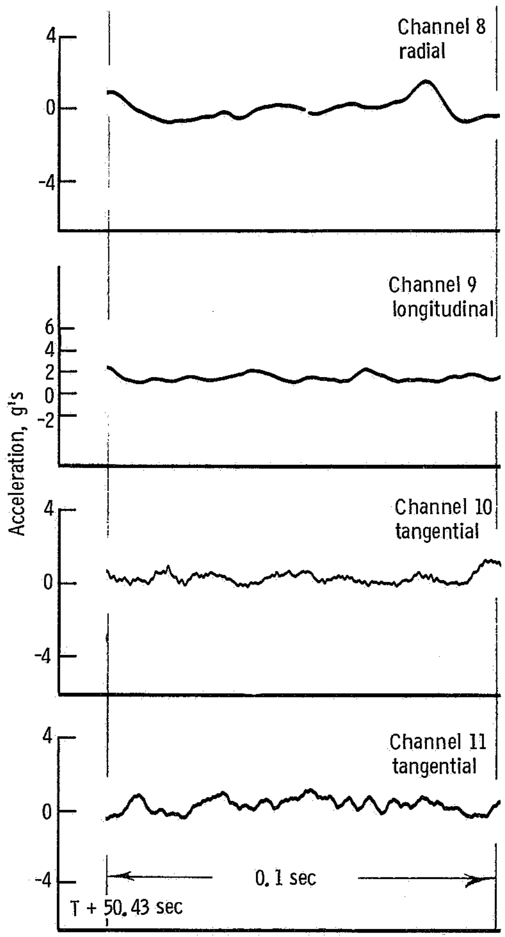


Figure D-4. - Dynamic data during transonic period, Lunar Orbiter IV.

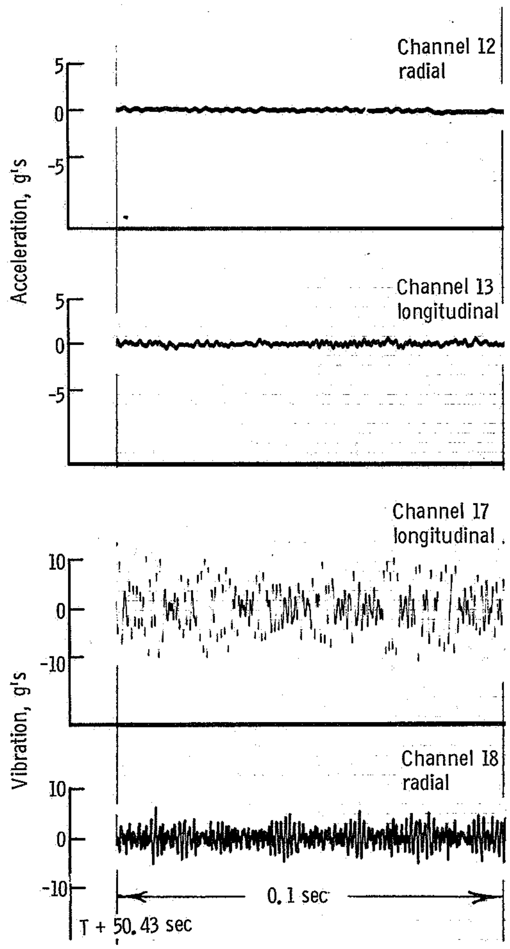


Figure D-5. - Dynamic data during transonic period, Lunar Orbiter IV.

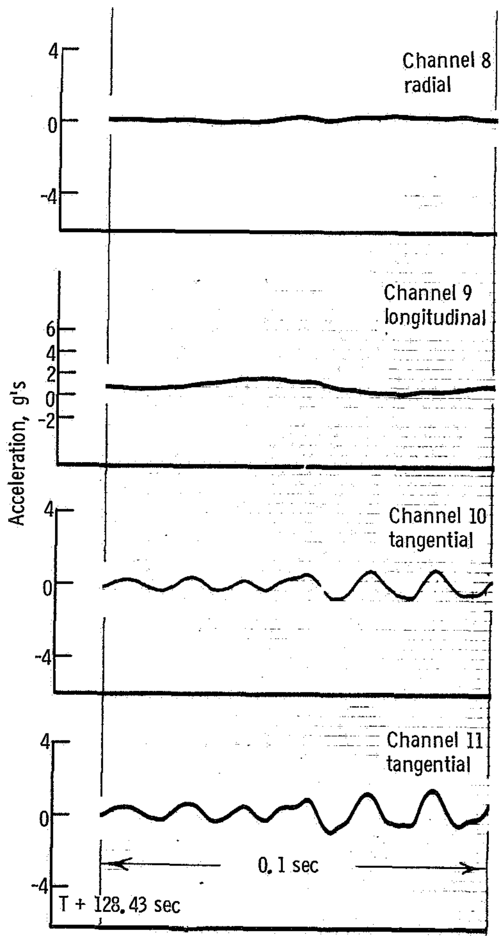


Figure D-6. - Dynamic data near time of booster engine cutoff, Lunar Orbiter IV.

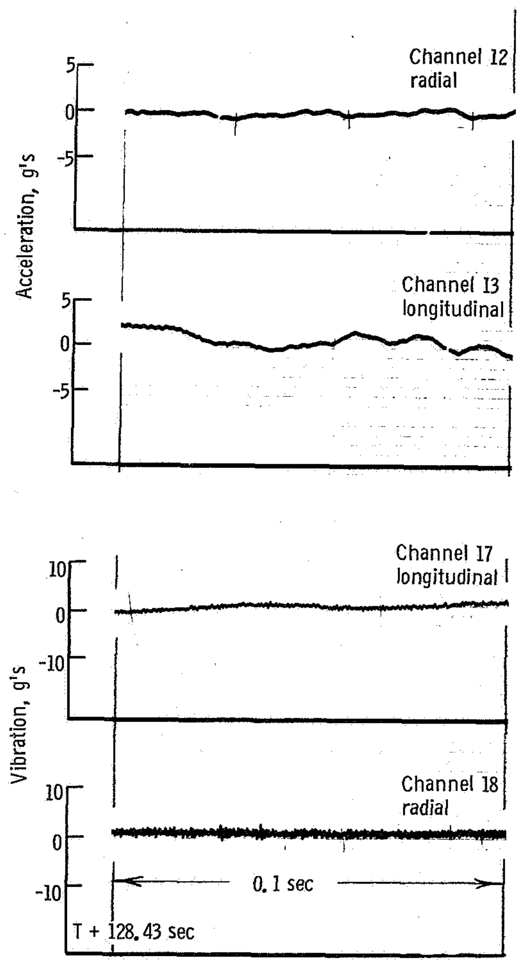


Figure D-7. - Dynamic data near time of booster engine cutoff, Lunar Orbiter IV.

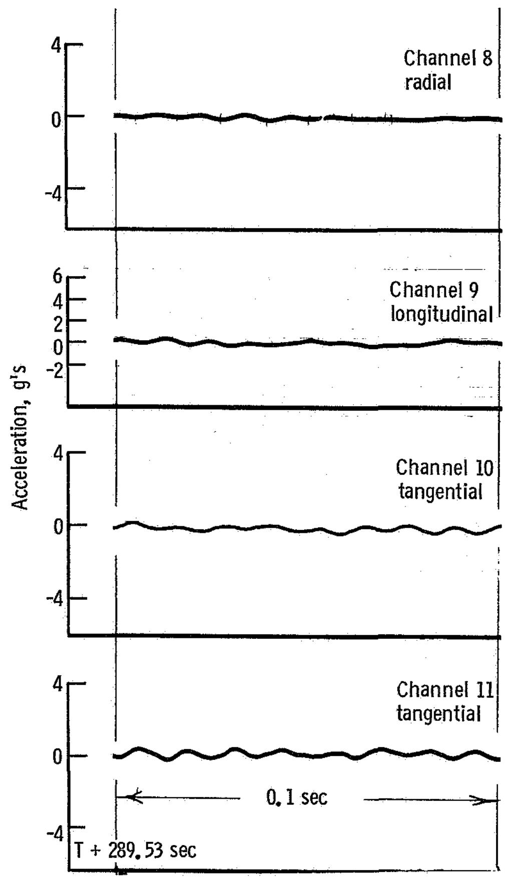


Figure D-8. - Dynamic data near time of sustainer engine cutoff, Lunar Orbiter IV.

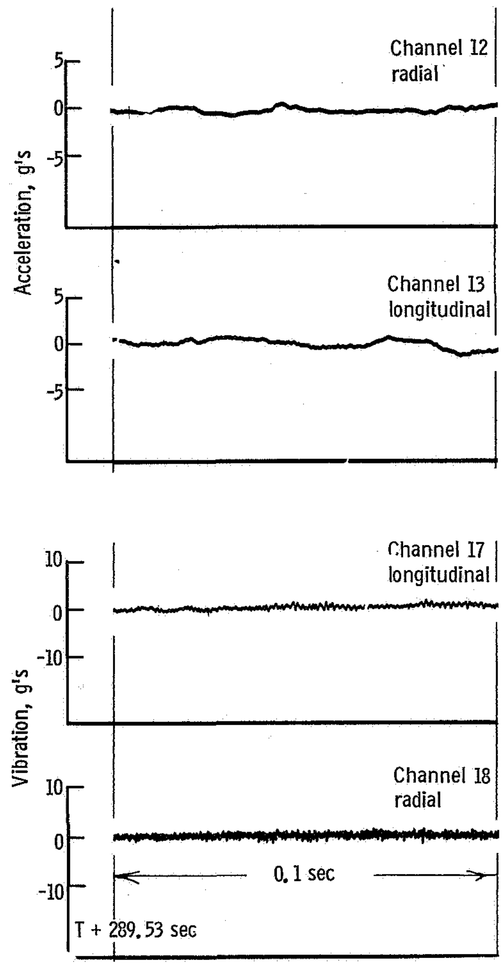


Figure D-9. - Dynamic data near time of sustainer engine cutoff, Lunar Orbiter IV.

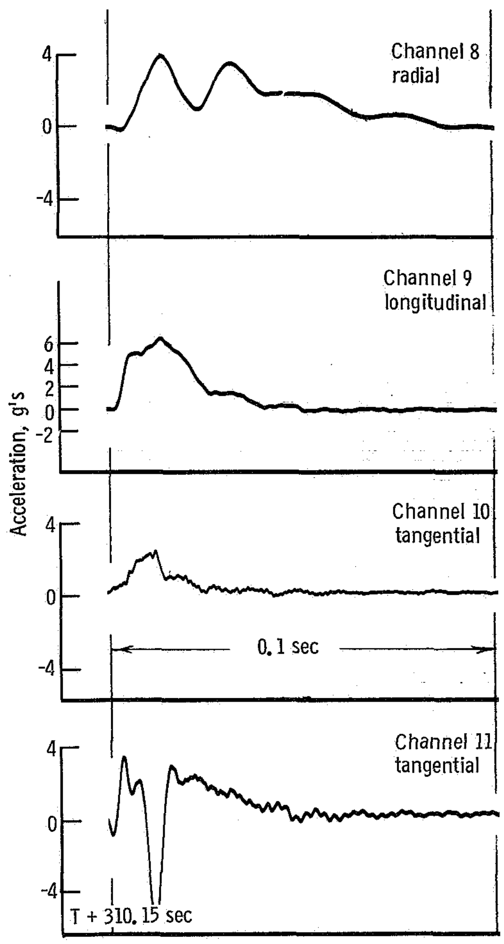


Figure D-10. - Dynamic data near time of horizon sensor fairing jettison, Lunar Orbiter IV.

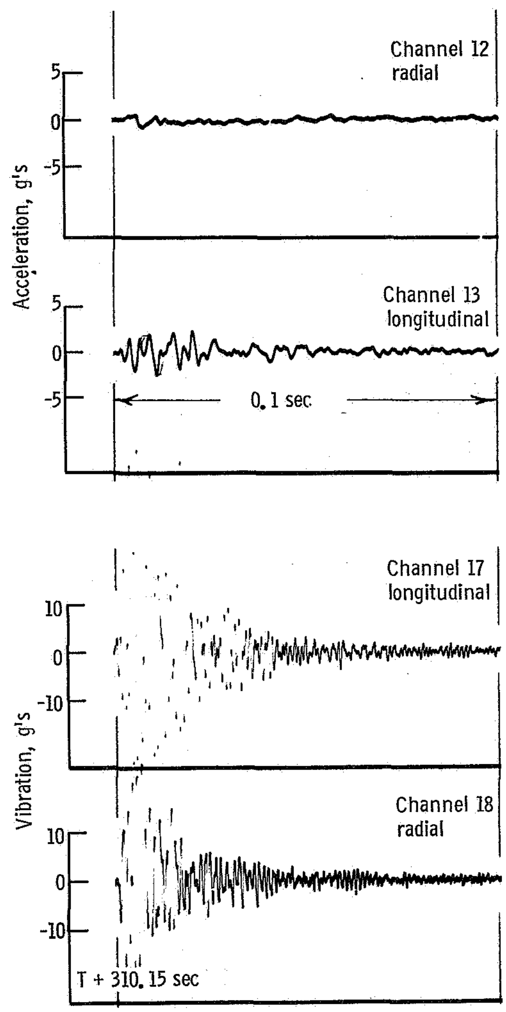


Figure D-11. - Dynamic data near time of horizon sensor fairing jettison, Lunar Orbiter IV.

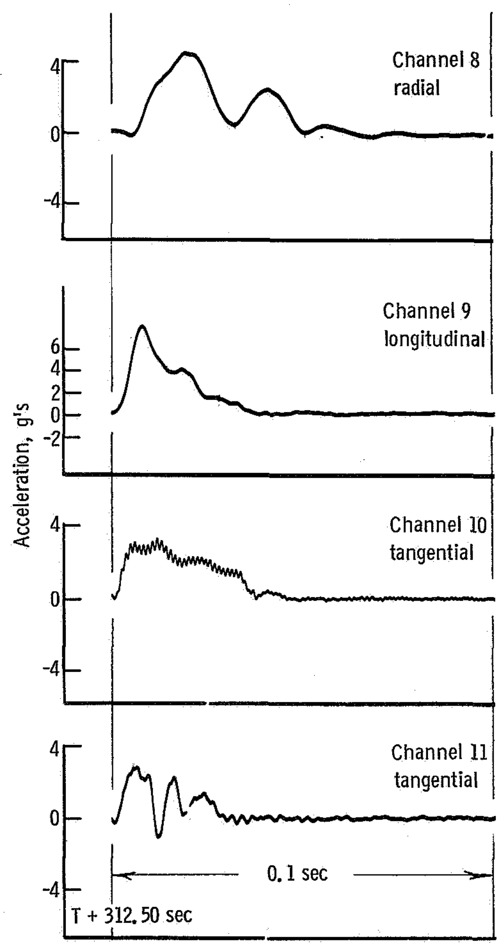


Figure D-12. - Dynamic data near time of shroud separation, Lunar Orbiter IV.

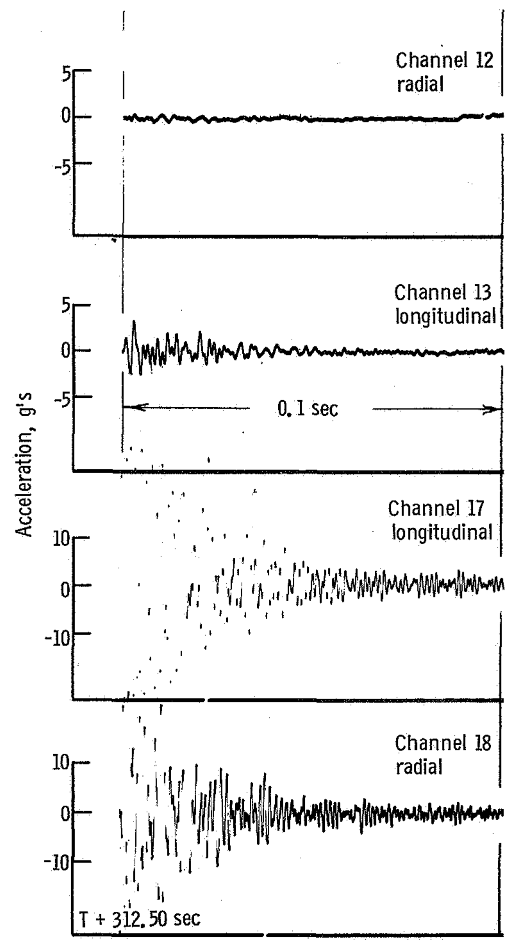


Figure D-13. - Dynamic data near time of shroud separation, Lunar Orbiter IV.

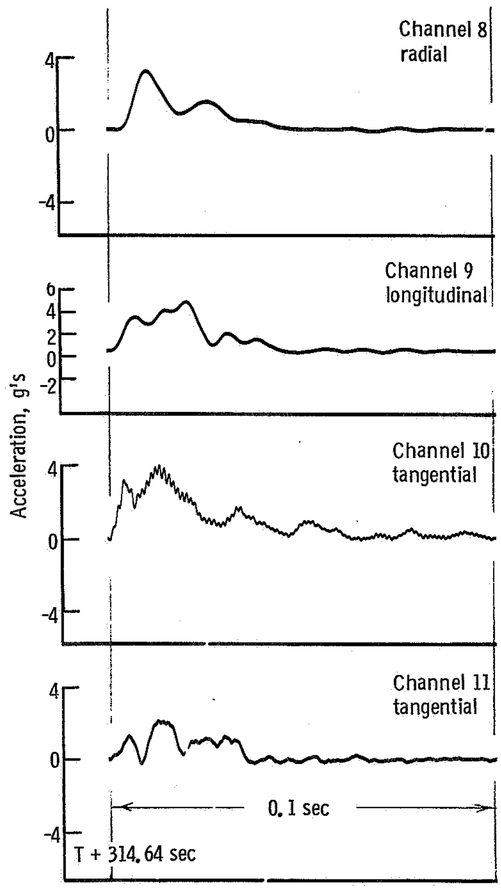


Figure D-14. - Dynamic data at Atlas-Agena separation, Lunar Oribter IV.

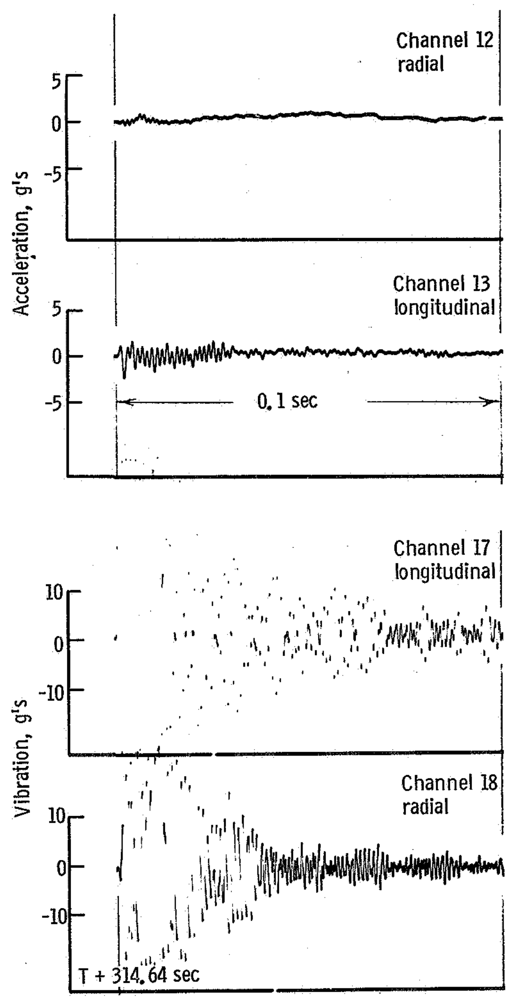


Figure D-15. - Dynamic data at Atlas-Agena separation, Lunar Orbiter IV.

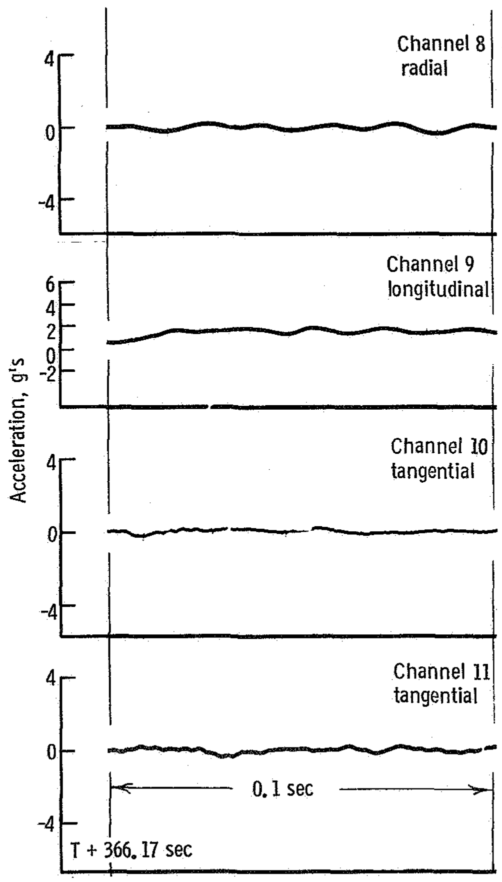


Figure D-16. - Dynamic data near time of Agena engine first ignition, Lunar Orbiter IV.

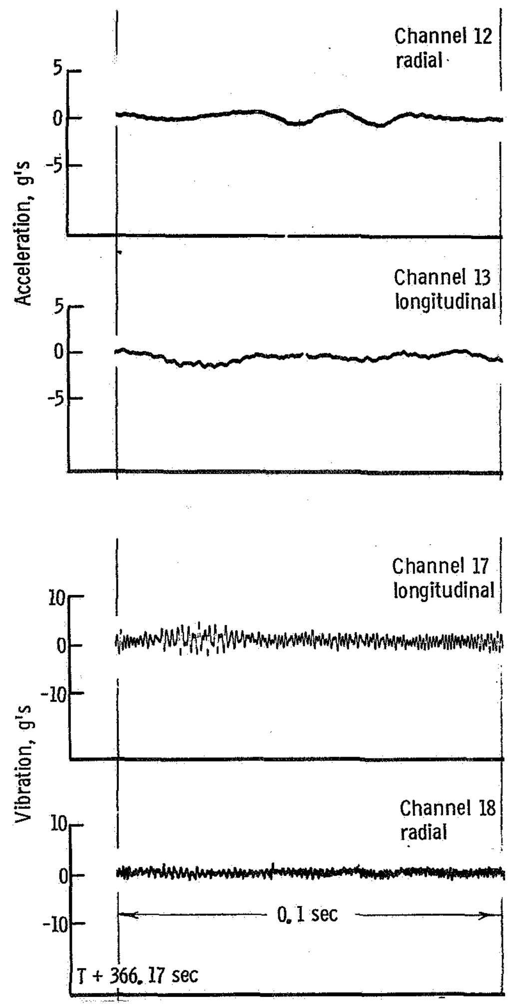


Figure D-17. - Dynamic data near time of Agena engine first ignition, Lunar Orbiter IV.

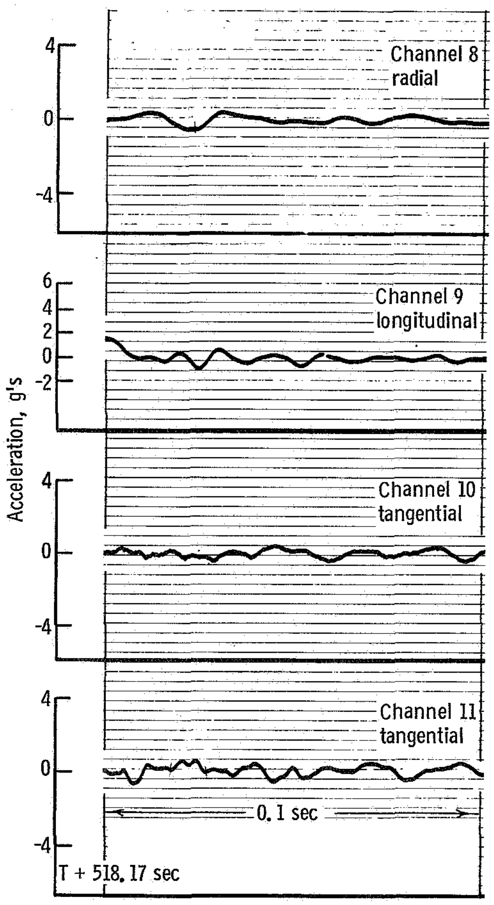


Figure D-18. - Dynamic data near time of Agena engine first cutoff, Lunar Orbiter IV.

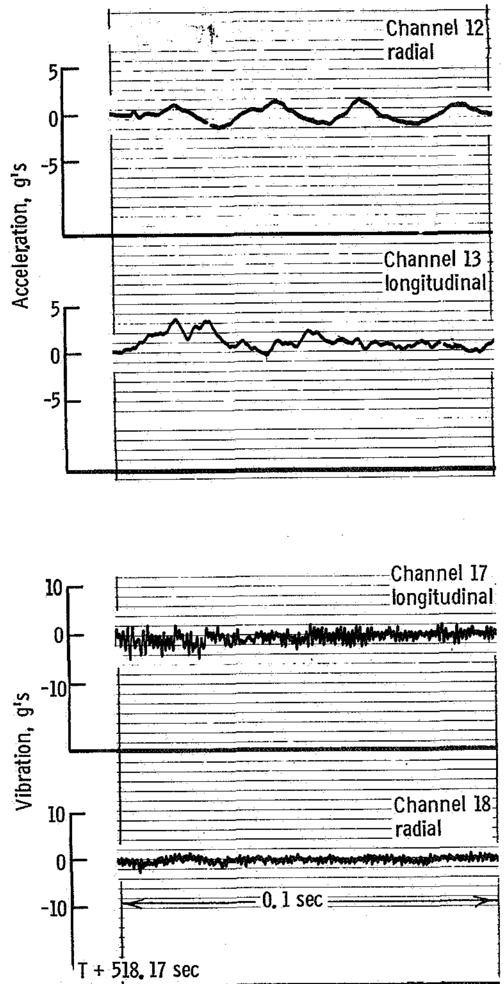


Figure D-19. - Dynamic data near time of Agena engine first cutoff, Lunar Orbiter IV.

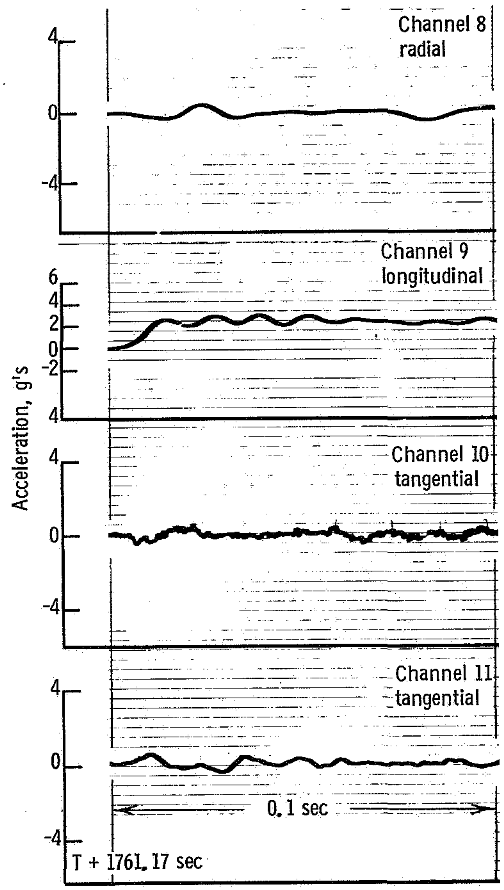


Figure D-20. - Dynamic data near time of Agena engine second ignition, Lunar Orbiter IV.

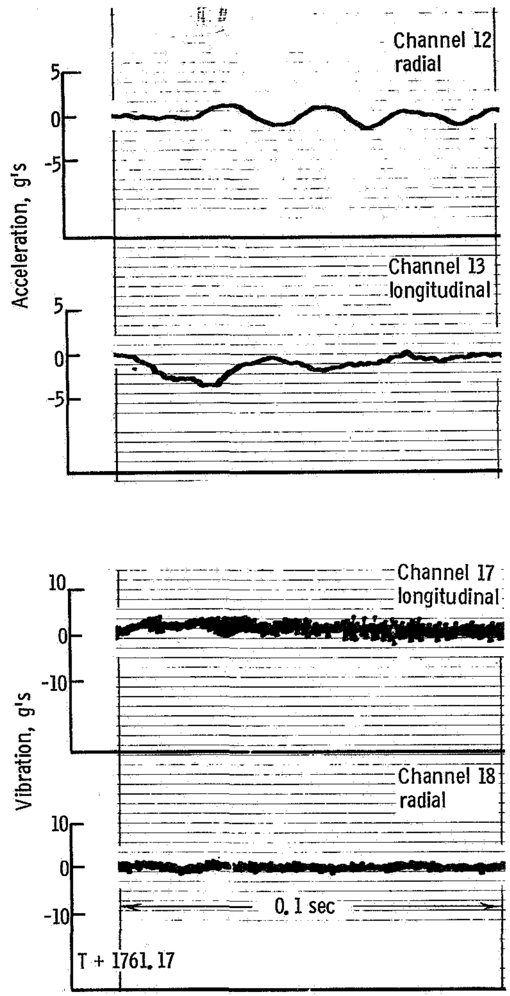


Figure D-21. - Dynamic data near time of Agena engine second ignition, Lunar Orbiter IV.

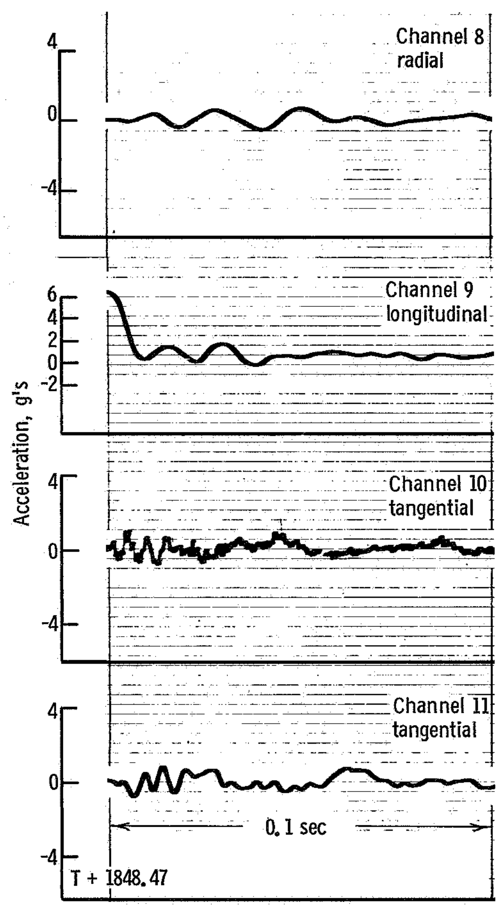


Figure D-22. - Dynamic data near time of Agena engine second cutoff, Lunar Orbiter IV.

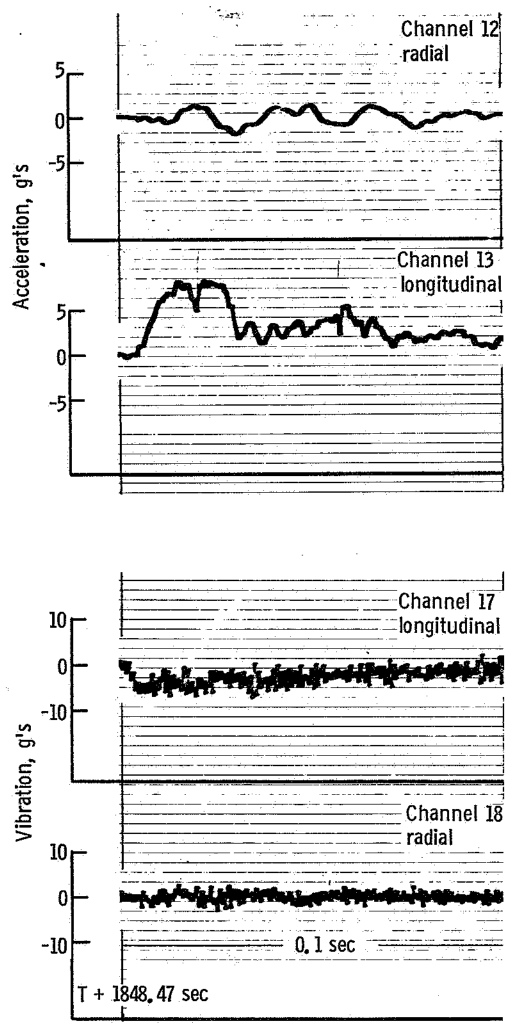


Figure D-23. - Dynamic data near time of Agena engine second cutoff, Lunar Orbiter IV.

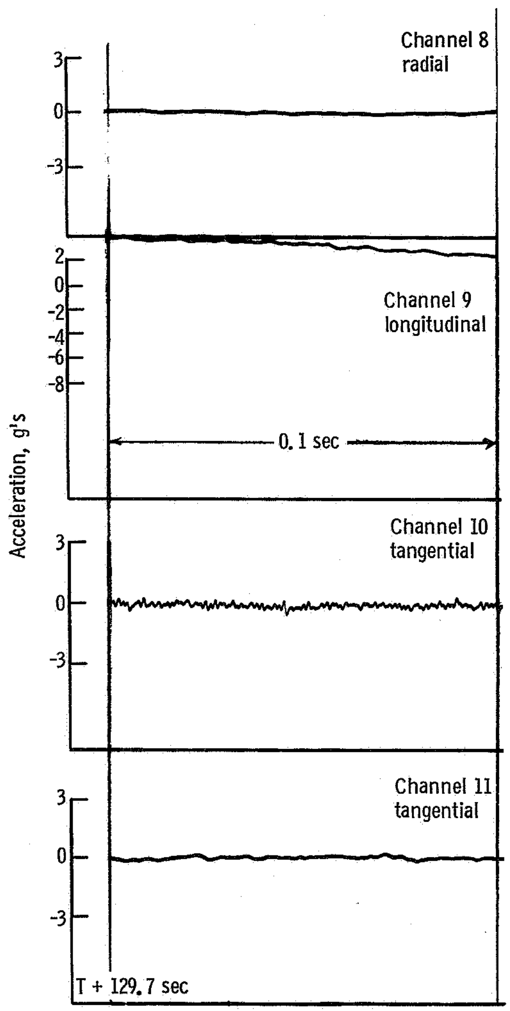


Figure D-24. - Dynamic data near time of booster engine cutoff, Lunar Orbiter III.

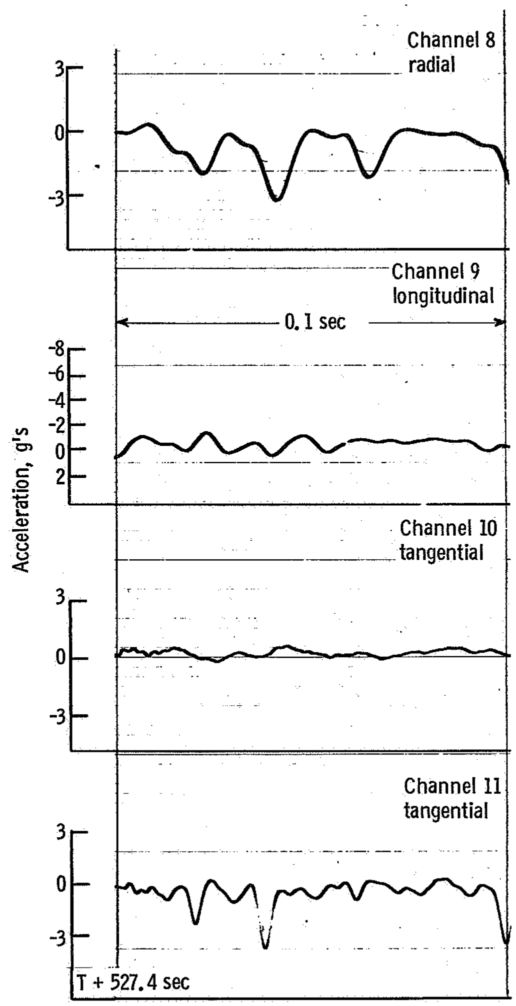


Figure D-25. - Dynamic data near time of Agena engine first cutoff, Lunar Orbiter III.

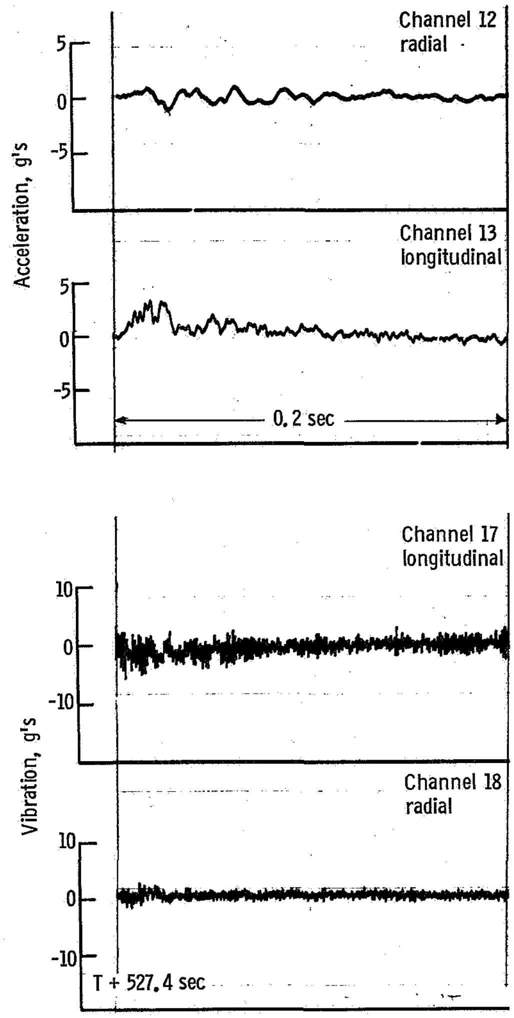


Figure D-26. - Dynamic data near time of Agena engine first cutoff, Lunar Orbiter III.

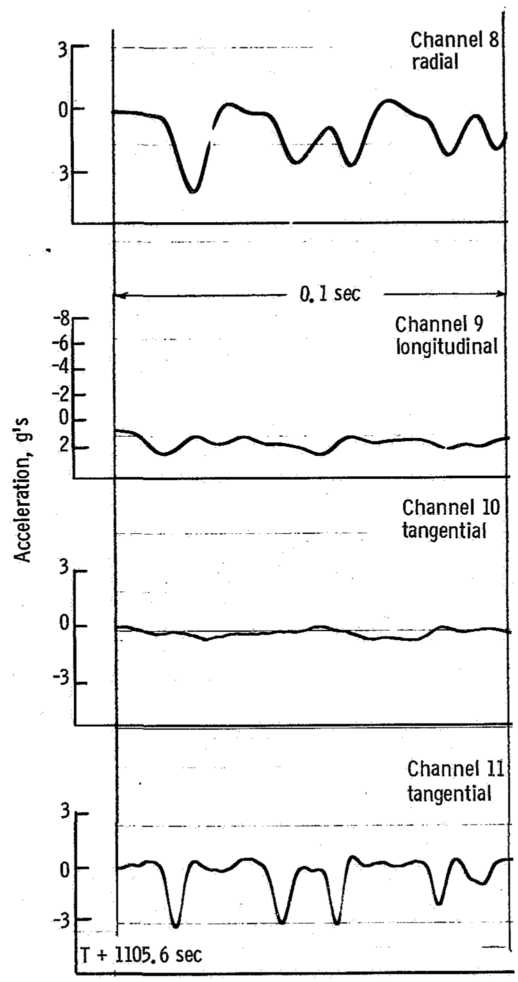


Figure D-27. - Dynamic data near time of Agena engine second ignition, Lunar Orbiter III.

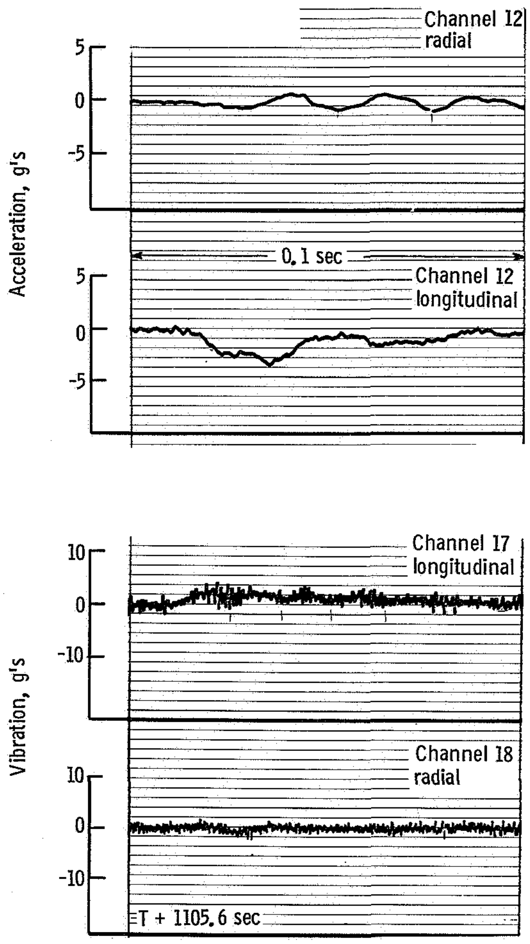


Figure D-28. - Dynamic data near time of Agena engine second ignition, Lunar Orbiter III.

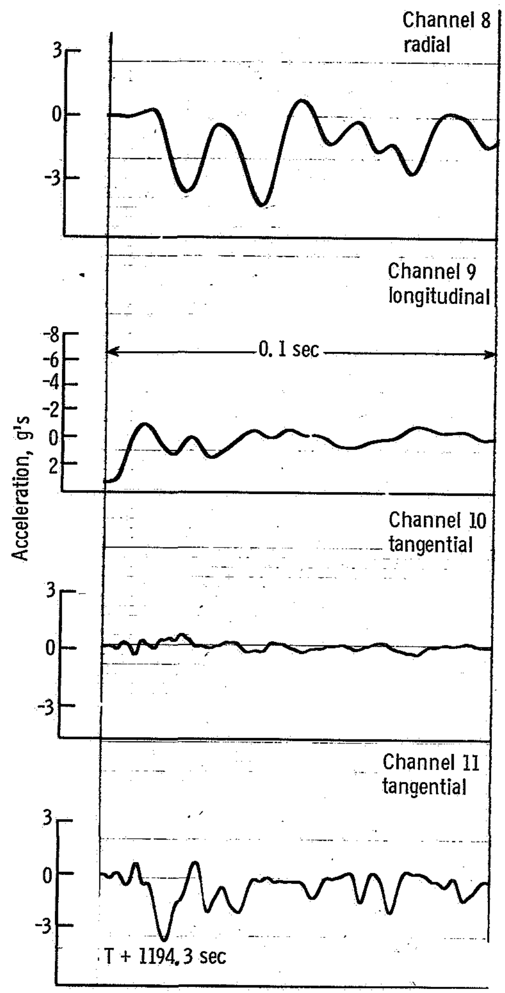


Figure D-29. - Dynamic data near time of Agena engine second cutoff, Lunar Orbiter III.

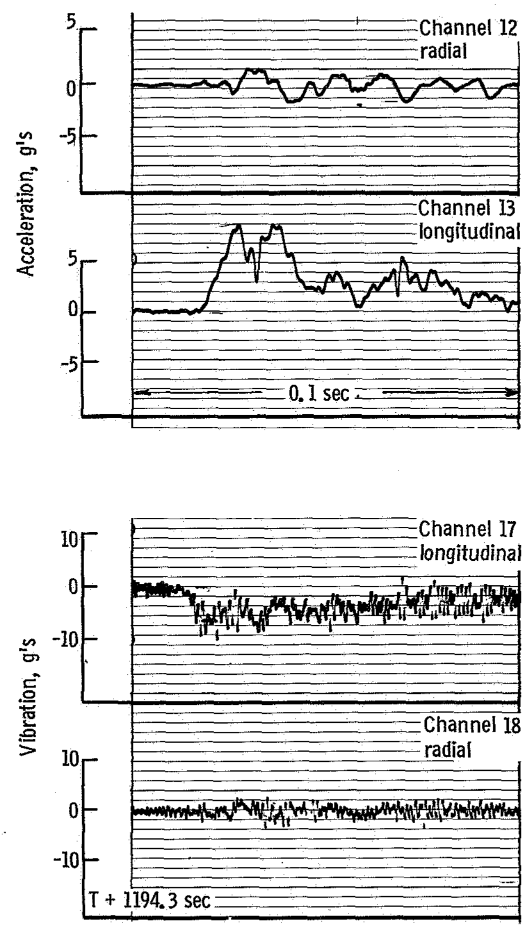


Figure D-30. - Dynamic data near time of Agena engine second cutoff, Lunar orbiter III.

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