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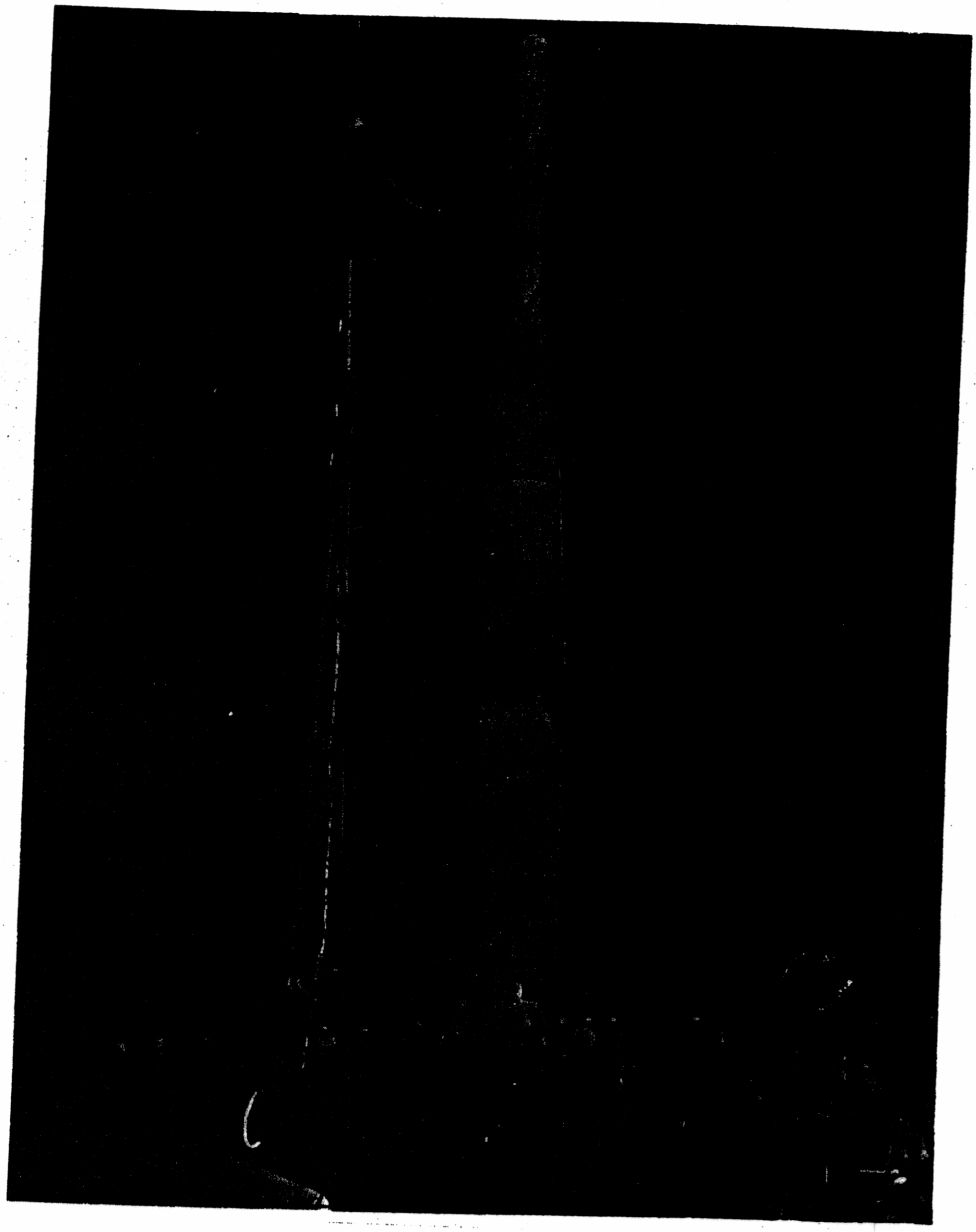

# WS-315A PRELIMINARY FLIGHT TEST REPORT NO.48

THOR MISSILE NO.134

AIR FORCE SERIAL NO. 57-2642

AIR FORCE DESIGNATION XSM-THOR-ABLE PHASE III

FIRED 7 AUGUST 1959

AIR FORCE MISSILE TEST CENTER

CAPE CANAVERAL, FLORIDA

BUREAU OF BUDGET APPROVAL NO. 21R138



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The Preliminary Flight Test Report for Missile 134 was prepared by the members of the WS-315A Flight Test Working Group. This test evaluation report contains the test objectives and results, missile history and configuration, conclusions and recommendations, countdown review, instrumentation summaries, and test data.

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TABLE OF CONTENTS

TABLE OF CONTENTS
(Continued)



TABLES

TABLE OF CONTENTS
(Continued)

Page

FIGURES

## 1.0 SUMMARY

The four stage Able-3 test vehicle was launched at 0923:22.239 EST, 7 August 1959. The flight, AFMTC Test No. 1005, began from Launch Stand 17A. The pitch plane azimuth was 48 degrees True.

The multi-stage vehicle consisted of:

1. A first stage Douglas Aircraft modified WS-315A Thor (Missile 134).
2. A second stage Aerojet AJ 10-101A propulsion system with Space Technology Laboratories, Inc. control, instrumentation, electric power, and guidance (open-loop) systems.
3. A third stage Allegheny Ballistic Laboratory 248 A4 engine.
4. A fourth stage STL payload incorporating an Atlantic Research Corporation 1 KS 420 injection rocket.

The payload was injected into the desired highly eccentric elliptical orbit. All primary and secondary test objectives appear to have been achieved. It has not been necessary to fire the injection rocket. Satellite lifetime is expected to be in excess of one year.

First stage performance was very good. Velocity at main engine cutoff was 154 feet per second above nominal.

Second stage engine performance was nominal. The second stage autopilot functioned properly and the programmed pitch command ntinued)

d correctly. Second stage telemetry functioned
d signals of good quality were received on all
Open-loop operation of the AGS transponder was
y.

engine performance was nominal.

transmitters are functioning and returning good
als on all channels. The solar cell power supply
nctioning. The worldwide network of tracking
rated properly.

## 2.0 TEST OBJECTIVES

### 2.1 Primary Test Objectives

2.1.1 Place an instrumented payload into an elliptical orbit about the earth to make scientific measurements of the environment encountered at altitudes from approximately 500 to 20,000 nautical miles.

2.1.2 Demonstrate satisfactory operation of the test vehicle, payload, and supporting ground stations to be utilized for Able-4 launchings. This includes the following:

a. Demonstrate satisfactory operation of the payload equipment to be used on Able-4 launching.

b. Demonstrate satisfactory open-loop operation of the second stage guidance transponder in conjunction with the Advanced Guidance Studies ground station.

c. Demonstrate satisfactory performance of the third stage propulsion system.

d. Demonstrate satisfactory operation of the special tracking stations and equipment which will be used to support the Able-4 launching.

3. Demonstrate satisfactory operation of the second/third and third/fourth stage separation mechanisms.

2.1 Primary Test Objectives (Continued)

f. Demonstrate satisfactory operation of the second stage autopilot control system and the spin control system.

2.2 Secondary Test Objectives

2.2.1 Conduct electromagnetic propagation experiments from an earth satellite to determine the propagation characteristics of the ionosphere and the troposphere.

2.2.2 Evaluate operation of portions of the Able-3 test vehicle which will not be carried on the Able-4 vehicle. These secondary objectives include the following:

a. Evaluate performance of the second stage propulsion system.

b. Demonstrate satisfactory performance of the fourth stage injection rocket (if commanded to fire).

c. Evaluate operation of the first/second stage separation mechanism.

2.2.3 Evaluate first stage airframe, propulsion, and modified autopilot control systems.

# 3.0 FLIGHT TEST RESULTS

## 3.1 First Stage

### 3.1.1 Trajectory

Liftoff from Launch Stand 17A was normal. Table II contains liftoff weight, thrust, and acceleration data.

AZUSA data indicate that the missile response to the roll and pitch rates commanded by the flight controller programmer was normal.

Maximum observed angular deviations of the velocity vector from nominal occurred at main engine cutoff (MECO) and were 2.5 degrees high in pitch and 1.4 degrees south in azimuth.

The velocity time history was nominal except that the 0.4 seconds longer than nominal burning time enabled the missile velocity to be 154 feet per second greater than nominal at main engine cutoff. These deviations were within the missile specifications.

Table II contains the missile velocity, flight path angle, altitude, range, and acceleration at main engine cutoff as obtained from AZUSA data. Also listed in Table II are the thrust, weight, and propellant utilization as obtained from kinematic data.

3.1.1 Trajectory (Continued)

The first stage impact point, assuming a vacuum re-entry, was 1488 nautical miles from the launch stand.

According to all available trajectory data, the performance of the first stage was more than needed to achieve the desired 144 nautical mile perigee and 19,585 nautical mile apogee orbit assuming nominal second and third stage performance.

3.1.2 Propulsion System

First stage propulsion system performance was excellent. Significant test data are presented in Tables I and II.

Preliminary data indicated a normal start sequence, with liftoff occurring 3.07 seconds after initiation of the start command.

A review of the ring camera films revealed that the main engine transition from gas generator blade valve opening to main stage was rough. This was evidenced by distortion of the main chamber bell during this period. This condition began with pick-up of the main engine chamber pressure switch and continued for a duration of 420 milliseconds.

3.1.2 Propulsion System (Continued)

A review of the propulsion system parameters during this period of time indicated normal engine operation. It should be noted, however, that the pressure oscillation frequency noted on the films is on the order of 18 cps which is beyond the capability of PDM/FM telemetry.

The main chamber pressure trace indicated a pressure of 576 psia at liftoff and 566 psia at cutoff, as compared to the expected value of 525 psia. This is not substantiated by other propulsion system performance data. The similarity of the main chamber pressure trace to the main fuel injector pressure trace, and the fact that the main chamber pressure instrumentation installation on Missile 136 was attached to a fuel injector pressure tap indicates that main fuel injector pressure had been measured instead of main chamber pressure.

The indicated turbopump speed during flight was 8500 rpm, as compared to the past MB-1 nominal of 6095 rpm. Since other performance parameters indicate approximately nominal thrust, the high measurement of turbopump speed is not believed to be indicative of a propulsion system problem. This and the high main chamber pressure measurement are discussed further in the instrumentation section of this report.

3.1.2 Propulsion System (Continued)

The malfunction of the low level liquid oxygen float switch prevented determination of propellant utilization by float switch data. The liquid oxygen float switch closed at 151.81 seconds starting the 5.2 second timer and indicating 1700 ± 88 pounds of liquid oxygen remaining. The float switch reopened at 155.46 seconds and remained opened for the remainder of the flight. It is worthy to note that if the float switch had not reopened, main engine cutoff would have been signalled after the expiration of the 5.2 second timer at 157.01 seconds.

The data indicated that cutoff resulted from the main chamber pressure switch dropout at fuel depletion; however, there was very little liquid oxygen or fuel aboard at cutoff (based on kinematic data). Propellant utilization was essentially 100 per cent.

Vernier engine solo operation subsequent to first stage cutoff was 7.7 seconds.

3.1.3 Control System

The control system performed satisfactorily in maintaining missile stability and executing the desired roll and pitch commands.

3.1.3 Control System (Continued)

The transients which occurred at MECO are considered normal and result from changes in body attitude due to thrust misalignment. The rather violent first stage motion indicated by telemetry is believed to be the result of loss of some telemetry functions due to second stage ignition. Real motion may have resulted from blast effects from second stage ignition into the first stage guidance compartment since there was no blast shield over the guidance compartment on this flight. There was evidently no interference problem in staging.

3.1.4 Electrical and Instrumentation Systems

All AC and DC voltages were normal.

Program steps, separation, and all other electrical systems performed satisfactorily.

Telemetry performance was satisfactory with the exception of the following telemetered functions:

A. Main engine chamber pressure indicated in excess of 570 psia. Investigation of Missile 136 (the last MB-1 type missile at CCMTA) revealed that the main

3.1.4 Electrical and Instrumentation Systems(Continued)

engine chamber pressure transducer was actually plumbed to a main fuel injector pressure port. This would account for the unusually high main engine chamber pressure indication on Missile 134 and for the similarity between main engine chamber pressure and main fuel injector pressure on Missiles 135 and 137. Recent MB-1 type missiles have experienced high main engine chamber pressure. The error may have occurred when aluminum tubing was replaced by stainless,steel since installation drawings are correct.

B. Turbopump speed indicated a level of 8570 rpm as compared to a nominal of 6080. It is not presently known where the malfunction originated. It is known that a procedural calibration of the converter at the launch stand was satisfactory.

AZUSA performance was satisfactory.

3.0 FLIGHT TEST RESULTS (Continued)

3.2 Second, Third, and Fourth Stages

3.2.1 Flight Mechanics and Propulsion Systems

Pressurization of the second stage began 14 minutes before liftoff with the oxidizer burst diaphragm bursting approximately 13.5 minutes before liftoff and the fuel diaphragm bursting approximately 12.8 minutes before liftoff. Pressurization continued until 2 minutes prior to launch. The fuel and oxidizer tank and helium bottle pressures at liftoff were 314, 310, and 1660 psia, respectively. The MECO signal occurred at 160.2 seconds after liftoff with second stage ignition occurring 162.2 seconds after liftoff. Eighty per cent chamber pressure was reached in 0.4 seconds and full chamber pressure of 208-210 psia was reached 0.9 seconds after second stage firing.

Steady-state operation was normal with a chamber pressure of 209 psia and regulator pressure of 366-378 psia. Steady-state thrust was 7,950 pounds, and total impulse was 930,150 pound-seconds, based on an assumed $C_F$ at altitude of 1.75. Burning time was 117.0 seconds.

3.2.1 Flight Mechanics and Propulsion Systems (Continued)

Preliminary data from the integrating accelerometer indicates a velocity increment during second stage burning of within 100 feet/second of the nominal value of 9420 feet/second. No explanation is presently available for the 0.32 seconds extended burning time. Trajectory data is not currently available for second stage powered flight.

The third stage ignition signal was received at 282.2 seconds after liftoff, and stages two and three separated less than 0.1 second later. Based on angular accelerometer data, third stage tipoff is estimated to be about 2 degrees. It is believed that a portion of the tipoff was due to the asymmetrical configuration resulting from one solar paddle failing to lock in the erect position. Third stage burnout occurred at 320.8 seconds after liftoff, with a total burning time of 38.6 seconds, as compared to nominal burning time of 36.8 seconds. The doppler range rate data indicate a velocity increment during third stage burning about 75 feet/second greater than expected. Preliminary trajectory analysis indicates the inertial velocity at third stage burnout was about 130 feet/second above that expected. The velocity vector at burnout appears to have been lofted about 3 degrees and was about

3.2.1 Flight Mechanics and Propulsion Systems (Continued)

0.8 degrees to the right in azimuth. The data indicate satisfactory operation of the third stage.

Third/fourth stage separation occurred 433.7 seconds after liftoff. The tipoff angle is estimated about 1 degree. A precession present at payload separation damped out within 20 seconds after separation. Spin rate after separation was 2.7 to 2.8 cps.

The payload went into orbit. Approximate orbital parameters are as follows:

| | | |
|---|---|---|
| Period | 768 | minutes |
| Perigee | 136 | nautical miles |
| Apogee | 22,911 | nautical miles |
| Lifetime | Greater than one year | |

3.2.2 Control and Guidance Systems

The second stage autopilot functioned properly in stabilizing the vehicle attitude. The pitch program operated normally for a period of 100.6 seconds. The roll control system also operated normally, with a 5 degree peak to peak limit cycle. Jet "ON" time varied from 0.08 to 0.13 seconds for each actuation. Total jet "ON" time was about two seconds.

Missile-borne and ground equipment of the guidance system functioned correctly. A manual second stage

3.2.2 Control and Guidance Systems (Continued)

cutoff command was transmitted after telemetry burnout indication to provide a back-up for initiating the spin-up and staging sequence. The second stage was tracked until burnout and the payload transponder was acquired and tracked until turned off by a manual command.

A manual payload accelerometer "ON" command was transmitted immediately after the loss of the payload transponder signal indicated that the manual transmitter "OFF" command was received. Pitch stop, yaw right, and yaw left commands were transmitted, open-loop, to the missile during second stage burning. These commands were properly executed and simulated torquing signals were applied to a dummy load. The pitch commands were not executed since this would interfere with the desired open loop pitch program.

3.2.3 Instrumentation

Second stage and payload telemetry operated satisfactorily, with signals of good quality on all channels. All experiments seem to be functioning properly. At present, no comparison is available between the digital and analog data.

3.2.4 Other Subsystems

Spin rocket firing was accomplished satisfactorily.

3.2.4 Other Subsystems (Continued)

Spin rate after second/third stage separation was 2.8 rps. Microswitch indication of solar paddle erection occurred 0.8 seconds after spin initiation. There was indication of only three paddles being latched in position. Since paddle erection starts with firing of the cable cutters at second stage TPS shutdown and normally takes about one second, erection should be completed before spin initiation. The excessive erection time is not understood at present. The indication that one paddle came up part way and did not latch is strengthened by data from the payload angular accelerometer which exhibited a steady-state zero shift after payload separation. The most probable reason for this shift is that the payload is spinning about a principal axis which does not coincide with the axis of symmetry, due to dynamic unbalance caused by one partly erected paddle. It is estimated that the present spin axis is about 5 degrees from the axis of symmetry. In addition, there are indications of a low charging rate for the payload batteries. This has necessitated a lowered duty cycle for the 5-watt transmitter.

All other subsystems appear to have operated satisfactorily.

# 4.0 CONCLUSIONS AND RECOMMENDATIONS

## 4.1 Conclusions

The first stage performance was very good.

Powered flight performance of the second and third stages was approximately nominal.

The desired elongated elliptical orbit was achieved.

Satellite lifetime is estimated to be in excess of a year.

All payload equipment functioned properly except that one solar cell paddle apparently failed to lock in the erect position. The lowered battery charging rate has necessitated a lower duty cycle for the 5-watt (378.21 mc) transmitter.

The special tracking stations performed satisfactorily and are recording scientific data for all experiments.

Satisfactory open-loop operation of the second stage guidance transponder was observed.

All stage separation mechanisms operated properly.

The second stage autopilot control system and the spin control system operated correctly.

4.1 Conclusions (Continued)

It has not been necessary to fire the fourth stage injection rocket.

4.2 Recommendations

The possible failure of one solar paddle to lock in the fully erected position should be investigated.

Flight testing of the Able-4 vehicles should be conducted as planned.

5.0 HISTORY

| | |
|---|---|
| 12-17-58 | DAC missile arrived at Cape Canaveral Missile Test Annex (CCMTA). |
| 4-9-59 | STL second stage arrived at CCMTA. |
| 4-13-59 | Second stage electro-mechanical check completed. |
| 4-15-59 | Missile first moved to Launch Stand 17A. |
| 4-16-59 | Second stage covers-off test completed. |
| 4-22-59 | Second stage covers-on test completed. |
| 4-24-59 | Second stage mounted on Missile 134. |
| 4-30-59 | Notification received of program delay. |
| 4-30-59 | Second stage removed and stored in Hangar E. |
| 5-1-59 | Missile returned to Hangar M. |
| 5-7-59 to 7-7-59 | Missile stored in Hangar L. |
| 6-17-59 | Second stage removed from storage. |
| 7-13-59 | Second stage electro-mechanical check performed. |
| 7-13-59 | Missile moved to Launch Stand 17A. |
| 7-20-59 | Second stage covers-off test completed. |
| 7-22-59 | Second stage covers-on test completed. |
| 7-23-59 | Second stage mounted on Missile 134. |
| 7-30-59 | T-6 day checks and acceptance checks. |
| 7-30-59 | Second stage integrated acceptance test completed. |
| 8-3-59 | Mock countdowns accomplished. |
| 8-4-59 | STL third stage installation. |
| 8-4-59 | T-3 day checks. |
| 8-5-59 | Second stage ordnance and payload installation. |
| 8-6-59 | Launch countdown started. |
| 8-7-59 | Launch. |

## 6.0 CONFIGURATION

The physical configuration of the missile is shown in Table V. Sign Convention and Station Locations are shown in Figure 2. Weight information is included in Tables II and III.

### 6.1 First Stage

#### 6.1.1 Airframe

The first stage is a standard Douglas Aircraft Company Thor airframe modified to accommodate the additional stages. These modifications include:

A. Strengthening of the guidance section to withstand the additional load of the additional stages.

B. Removal of the following items:

1. Re-entry vehicle latches.
2. Missile center section beams.
3. One PDM/FM set and associated wiring and end instruments.
4. Two FM/FM sets and associated wiring and end instruments.
5. The instrumentation inverter and associated batteries.
6. Retro rockets and associated wiring. The fairings will be retained.
7. The complete DOVAP system and antennas.
8. Power factor correction box.
9. The quad-plexer and PDM power amplifier.

6.1.1 Airframe (Continued)

10. ACSP air conditioning ducts from the guidance section.

C. Retention of one light weight nickel cadmium battery in the forward section to power the telemetry instrumentation.

D. Powering of the AZUSA transponder by the control inverter through fuzes.

E. Providing proper loading balances of the control inverter by the use of dummy loads.

6.1.2 Propulsion System

The propulsion for the first stage is supplied by a Rocketdyne MB-1 system providing a nominal sea level thrust of 152,000 pounds using liquid oxygen and RP-1 as propellants.

The propulsion system includes one main engine and two vernier engines, a gas generator and turbine power system, a turbopump propellant feed system, a helium pressurization and pneumatic system, and an integrated start propellant system with interconnecting plumbing. The regeneratively cooled main engine thrust chamber is gimbal mounted and capable of developing 150,000 pounds of thrust under sea level

6.1.2 Propulsion System (Continued)

conditions. An additional 2000 pounds of thrust is developed by the two gimballed vernier engines which are provided for roll control and impulse adjustment.

The following modifications are incorporated:

1. An MB-3 turbine exhaust system with liners removed from the bellows is installed.
2. The liners are removed from the fuel and liquid oxygen inlet bellows.
3. An improved quill shaft with new spiral lock retainers is installed.
4. G-9 bearings are installed in the turbopump.
5. The gear case is pressurized.
6. The turbopump bearing lube oil supply lines are insulated.
7. A 100-mesh screen is installed over the liquid oxygen duct inlet at the bottom of the main liquid oxygen tank.
8. The liquid oxygen pump is reworked by Rocketdyne by the staking of threads to prevent the backing off of the liquid oxygen pump bearing retainer unit. A check is made to insure proper clearance between the liquid

6.1.2 Propulsion System (Continued)

oxygen pump impeller and adapter and between the retainer ring bolts and impeller. The pump is also checked for excess lubricant.

Main engine cutoff is to be effected by a chamber pressure switch upon depletion of propellants (as indicated by a drop in chamber pressure to 90 per cent of normal). A back-up cutoff from a five second timer, initiated at the closure of the first propellant float switch, is also incorporated.

Main engine cutoff is designed to occur at approximately 159 seconds after launch for no remaining RP. The vernier engines are to continue to furnish thrust through first/second stage separation about 3.8 seconds later.

6.1.3 Control System

The control system is altered from standard Thor configuration due to variations in aerodynamic parameters, airframe physical characteristics, and trajectory requirements for this flight.

A. The over-all system gains are reduced initially with a single gain change occurring during powered flight. This change is similar to the rate and attitude loop gain

6.1.3 Control System (Continued)

reduction on standard Thors except that it occurrs at 110.0 seconds instead of 108.5 seconds. Consequently vernier propellant tank pressurization occurrs at 110.0 seconds. Alterations are made in the compensation networks of the Control Electronics Assembly.

B. All three HIG-4 attitude gyros are modified to provide compensation for positive restraint due to reaction torque.

C. The pitch and yaw rate gyros are relocated at Station 117.0 in the guidance section.

D. Inverter voltage is adjusted to off-set possible voltage waveform distortion which affects desired pitch command.

E. Major events programmed on the film strip timer are as follows:

Roll Program

| Req'd Azimuth | Deg. Roll Angle | Deg/Sec Roll Rate |
|---|---|---|
| 48.0 | 42.0 | 6.0 |

6.1.3 Control System (Continued)

Pitch Program:

| TIME (Sec) | (Deg/Sec) |
|---|---|
| 0-10 | 0 |
| 10-28 | -0.50518 |
| 28-70 | -0.76810 |
| 70-98 | -0.54511 |
| 98-140 | -0.31517 |
| 140-Burnout | 0 |

6.1.4 Electrical and Instrumentation Systems

The first stage primary cutoff circuit consists of two main chamber pressure switches in series which are armed by a program step at 149 seconds. A secondary, or back-up, system consists of the liquid oxygen float switch and the fuel float switch in parallel which are armed by the programmer at 149 seconds. The float switches in turn arm a 5.25 second timer which provides MECO.

Second stage start is provided at MECO through the K-59 relay.

First stage instrumentation consists of a thirty channel PDM/FM telemetry system with a standard Thor-type antenna, a coherent AZUSA transponder, and a standard Thor dual-destruct system. The RF carrier will operate on a frequency of 225.5 megacycles per second.

6.0 CONFIGURATION (Continued)

6.2 First/Second Interstage

The Aerojet first/second interstage assembly consists of a conical skirt structure in two parts. One part of the interstage forms an extension of the second stage skin at the aft end; the other part is an extension of the forward part of the first stage. The skirt consists of a stainless steel outer skin and a Mag-Thorium inner skin, supported by stringers. At the separation plane are rings on each half, held together by four explosive bolts. The bolts are fired by means of a chamber pressure switch set to operate at 60 per cent of second stage engine thrust.

6.3 Second Stage

The second stage consisted of an Aerojet General Corporation AJ 10-101A propulsion system, Serial Number P SA-3-1, a Space Technology Laboratories control compartment, instrumentation, and spin rockets. Purpose of the second stage is to boost the vehicle to an inertial velocity of 24,030 feet/second. Propulsion termination is at incipient propellant depletion (TPS).

6.3.1 Airframe

The second stage is primarily composed of three Type 410 stainless steel tanks welded into an integral

6.3.1 Airframe (Continued)

assembly providing compartments for fuel, helium, and oxidizer. A spherical helium tank is located in the middle of the tank assembly, one hemisphere forming the aft bulkhead of the forward (fuel) tank, the other hemisphere forming the forward bulkhead of the rear (oxidizer) tank.

The engine is housed in an aluminum skin and stringer structure aft of the oxidizer tank. Piping and wiring running along the outer surface of the tanks is covered by fairings.

The STL control compartment is at the forward end of the stage. Attached to the control compartment are the second/third interstage support struts and the solar paddle fishpoles. The latter are two long metal tubes which extend forward along the third stage, terminating near the ends of the paddles. The fishpoles contain cable-cutters at their ends which cut the solar paddle tiedown cords. The fish-poles are bent inward toward the longitudinal axis of the third stage, supplying tension to the tiedown cords. Firing of the cable-cutters occurs at second stage TPS shutdown. Paddle erection follows within one second.

6.3 Second Stage (Continued)

6.3.2 Propulsion System

The second stage is powered by an AJ 10-101A propulsion system manufactured and prepared by Aerojet General Corporation. The system provides a nominal vacuum thrust of 7,800 pounds, using WIFNA and UDMH as propellants.

The system consists of a regeneratively cooled rocket thrust chamber mounted on a gimbal, propellant feed system, helium pressurization system, interconnecting plumbing, gimbal actuation system, and roll control system.

The propellants utilized are fed to the thrust chamber and injected into the chamber at a nominal instantaneous mixture ratio of 2.8:1 by weight of oxidizer to fuel. The propellants are pressure-fed to the thrust chamber by regulated helium pressure from a high pressure helium supply, supplemented by a solid propellant heat generator.

6.3.3 Control System

An STL programmed autopilot is used for guidance. The autopilot program consists only of a pitch rate of 0.00706356 rad/sec. during the time interval (from liftoff) of 262.0 - 275.8 seconds. There are

6.3.3 Control System (Continued)

no programmed yaw or roll maneuvers. Roll control is provided by discharging helium from the pressurization tank through two nozzles for clockwise and two nozzles for counterclockwise control. Pitch and yaw control is achieved by gimballing the thrust chamber.

6.3.4 Electrical and Instrumentation Systems

Second stage instrumentation consisted of an FM/FM telemetry set and associated transducers, including an accelerometer. Tracking data is obtained from the guidance receiver. The telemetry system consists of five FM subcarrier channels, one 18-segment PAM/FM subcarrier channel, and one 14-segment PAM/FM subcarrier channel. The phase-modulated r-f carrier operates at 238.5 mc. The antenna system consists of a double-yogi configuration with the antennas spaced 180 degrees apart. The system is self-contained and uses its own batteries for primary power. The transmitted data consists of second stage airframe, propulsion, and control measurements, plus events timing, as shown in Table IX. The second stage also contains a guidance transponder which is to be operated open-loop in conjunction with the Advanced Studies ground station at Cape Canaveral. Simulated guidance system steering commands will be transmitted from the ground at 401.848 mc and the transponder will transmit the guidance action at 378.21 mc.

6.3.4 Electrical and Instrumentation Systems (Continued)

The second stage also carries an acceleration switch to lock out second stage ignition until 0.8 seconds after an acceleration of about 6.7 g's is achieved during first stage burning. Detonating cord is carried on the stage for range safety purposes so that the second stage can be destroyed simultaneously with the first stage during first stage powered flight.

6.3.5 Spin Rockets

Eight Atlantic Research Corporation 0.5 KS 130 spin rockets are mounted on the second stage. These are ignited from a signal sent by a two-second timer started at second stage TPS shutdown. and impart a 2.8 rps spin to the vehicle to spin-stabilize the third and fourth stages.

6.4 Second/Third Interstage

The second/third interstage structure consists of four support struts, a retention ring, a separation band, two bushings, and two explosive bolts. The support fans are hinged at their lower ends. The hinge brackets are mounted on the second stage control compartment. On the upper ends of the fans are bulbs which fit into the third stage retention ring groove. The aluminum separation band is in two parts, held together by explosive bolts

6.4 Second/Third Interstage (Continued)

which fit into bushings inserted into the band trunnions. The band wraps around the upper ends of the fans, forcing the fan bulbs into the retention ring groove. Separation is controlled by a 3.1 second timer started at second stage TPS shutdown. At this time the second, third, and fourth stages are spinning. A signal is sent to fire the separation bolts and to ignite the third stage engine, simultaneously. After the bolts fire and the band separates, centrifugal force causes the support struts to rotate about their hinges, pulling the bulbs out of the retention ring and breaking physical connection between the stages.

6.5 Nose Fairing

An aerodynamic nose fairing fits over the third and fourth stages. It is in two symmetrical halves, and when assembled has the shape of a cylinder with a hemispheric nose. A groove in the lower end of the fairing fits around a bulb on the second stage control compartment. The upper portions of the fairing are held together by two explosive bolts. Two gas-driven actuators in the plane of the fairing center of gravity hold the central portion of the fairing together and separate the halves at fairing jettison time. The explosive bolts are shielded to provide blast and fragmentation protection to the payload.

6.5 Nose Fairing (Continued)

Jettison of the fairing is accomplished by a signal from a 46-second timer in the second stage. Jettison takes place about 25 seconds after start of the second stage engine. The explosive bolts and the actuators are simultaneously energized, and the actuator pistons drive the halves of the fairing apart. A lanyard attached to the fairing starts the third stage four-minute timer.

6.6 Third Stage

The third stage is a solid-propellant rocket engine, Type ABL 248 A4, manufactured by Allegheny Ballistic Laboratory (engine Serial No. 52). Modifications to the ABL 248 include the following.

A retention ring is bonded to the motor case at the aft end of the cylindrical portion. The ring contains a machined groove to fit the bulbs on the second/third interstage support fans.

A four-minute timer and associated battery are mounted 180 degrees apart on the retention ring. The purpose of the timer is to initiate third/fourth stage separation. The timer is started at the time of nose fairing jettison by means of a lanyard attached to the nose fairing.

Four solar paddle support brackets are mounted 90 degrees apart around the third stage motor case.

6.6 Third Stage (Continued)

Two tube (fishpole) support pads are mounted near two opposite paddle support brackets.

The third stage portion of the third/fourth interstage structure is bonded to the forward end of the motor case.

The third stage carries no control system or instrumentation. Various third stage functions are monitored by second and fourth stage telemetry. The stage attitude is spin-stabilized by the second stage spin rockets.

The purpose of the third stage is to boost the payload velocity from 24,030 ft/sec to 33,368 ft/sec, which is sufficient to enable the payload to attain an apogee of 19,585 nautical miles above the earth.

6.7 Third/Fourth Interstage

The third/fourth interstage structure consists of a plastic cylinder attached to the payload, an aluminum cylinder attached to the head end of the third stage, a coil spring, a separation band, two explosive bolts, and anti-fragmentation shielding. The plastic and aluminum portions are mated by means of tapered shear pins in the aluminum portion which fit into holes in the plastic portion. The central coil spring between the stages is put into compression when the stages are mated. In the vicinity of the separation plane, the plastic and aluminum portions of the interstage structure

6.7 Third/Fourth Interstage (Continued)

extend radially outward to form a lip with trapezoidal cross-section. The steel separation band is in two parts, held together by the explosive bolts. On each half of the band are mounted four aluminum Marmon clamps which fit over the interstage lip. When the band is installed and the bolts torqued down, these clamps hold the stages together against the spring force. Bolt firing impulse comes from the four-minute timer on the third stage. After the bolts are fired and the band separates, the stages are forced apart by the spring. Shields around the bolts protect the payload from blast and fragmentation damage.

6.8 Fourth Stage

The fourth stage consists of an STL payload structure, payload instrumentation and a solar cell power supply system, and an Atlantic Research Corporation solid-propellant injection rocket. Payload serial number is 003.

6.8.1 Structure

The structure consists of a 29-inch diameter approximate sphere, flanked by four paddles equally spaced around the sphere's equator. The paddles are light aluminum spars into which are fastened modular pallets of solar cells on both sides. The sphere is composed of a central platform, a support structure, a lower cover and an upper cover. The central platform is

6.8.1 Structure (Continued)

made of fiberglass honeycomb. The support structure consists of a welded aluminum tube truss with two circumferential rings to transfer load from the platform and paddle hinge brackets to the truss. A ring at the equator of the sphere provides for attachment of the covers, which consist of thin formed sheet metal. All electronic equipment and storage batteries are attached to the central platform. The paddles are hinged so as to remain in either of two stable positions. At launch they are folded down symmetrically inside the aerodynamic nose fairing along the third stage engine. At stage two shutdown the solar cell paddle cable is cut and springs on the paddle hinge brackets cause the paddles to move to an extended position and latch. In the extended position, two of the paddle arms lie 22.5 degrees above the sphere equatorial plane and two lie 22.5 degrees below the equatorial plane. The paddles are canted with respect to the brackets to maximize the average number of cells normal to the sun's radiation. Extension of the solar paddles closes a microswitch for activation of the payload transmitter.

6.8.2 Electronics and Instrumentation

The payload carries the following electronic equipment:

(a) Sensors to measure aspects of the space environment.

6.8.2 Electronics and Instrumentation (Continued)

(b) Three telemetry transmitters (108.09 mc, 108.06 mc, 378.21 mc)

(c) A doppler and command system, including a receiver (401.848 mc)

(d) Analog and digital instrumentation systems, including programmer.

Able-3 payload experiments consist of measurements of the space environment and propagation experiments as follows:

Environment Experiments

(a) Micrometeorite Flux and Momentum
An apparatus similar to that used on Able-1, flights 1 and 2, is used to count micrometeorite impacts above about $10^{-4}$ g cm/sec momentum. Two momentum levels are measured.

(b) Search-Coil Magnetometer
An STL search-coil magnetometer is used in conjunction with a flux-gate magnetometer to map the vector magnetic field.

(c) Flux-Gate Magnetometer
This equipment is used in conjunction with the search-coil magnetometer to measure the spin axis component of the magnetic field.

6.8.2 Electronics and Instrumentation (Continued)

(d) Vehicle Position Determination

A facsimile system consisting of both optical and electronic equipment is contained in the payload to determine the position of the vehicle relative to the earth. Transmitted pictures of the earth have a resolution of about five miles and provide meteorological information such as cloud cover.

(e) Ion Chamber and Geiger Tube

An ionization chamber to measure total radiation flux, and a Geiger-Muller tube for count rate, both supplied by the University of Minnesota, are carried. The combination of these instruments measure mean specific ionization per particle.

(f) Scintillation Counter

An STL scintillation counter is used to determine total radiation flux. This equipment uses different shielding than for the University of Minnesota experiment.

(g) Cosmic Ray

A University of Chicago triple-coincidence proportional counter telescope obtains a total count of charged particles above

6.8.2 Electronics and Instrumentation (Continued)

two energy threshholds. The seven counters are arranged concentrically to provide single-incidence and triple-coincidence measurements.

(h) Aspect Indicator

A phase comparator which measures the phase relationship between the output of a photoelectric "sun-scanner" and the search-coil magnetometer provides the "H" direction of the magnetic field.

(i) VLF Propagation

A VLF receiver supplied by Stanford University monitors the propagation of 15.5 KC signals from NSS, Annapolis, to study the dispersive properties of the atmosphere at very low frequencies.

(j) Other Measurements

Other data that is taken includes temperature readings of the payload compartment and solar cell paddles, performance measurements on the solar cell and battery system, and performance measurements on a temperature control unit (to be used on Atlas Able-4).

Propagation Experiments

(a) Electron Density

Two coherent transmitters operating at 108.06

6.8.2 Electronics and Instrumentation (Continued)

mc and 378.21 mc are used for electron density measurements by comparing the doppler shift of the received signals at the Hawaii station.

(b) Faraday Rotation

The faraday rotation caused by a change in the total ions along the propagation path from payload to ground is measured in Hawaii by observing rotation of the plane of polarization of the received 108.06 mc signal.

(c) Amplitude and Phase Fluctuation

Amplitude and phase fluctuations induced by the ionosphere are determined by measurement of the 108.06 mc signal with two receivers at the National Bureau of Standard Laboratory, Boulder, Colorado.

In addition to measurements of space environment, information on fourth-stage operation is telemetered as indicated in Table X. The 378.21 mc transmitter is the primary transmitter and the only one that will be used on Able-4 flights. It telemeters all experiments. The other two transmitters also telemeter this information and also telemeter accelerometer and blip strip information. Operation of all transmitters

6.8.2 Electronics and Instrumentation (Continued)
is controlled by the doppler and command system.

The doppler and command system is used for tracking the payload and for controlling various functions of the payload equipment. Position and velocity is determined by use of a transponder in the payload. Transmission from ground stations is at 401.848 mc; response is at 378.21 mc. A command is available for turning off all radiation from the payload to avoid interference with other spectrum users.

6.8.3 Solar Cell Power Supply System
Payload electronics are powered by storage batteries which are kept charged by impingement of solar radiation on banks of solar cells in the paddles. A total of 8,000 cells are carried with an active cell area of 12.2 square feet. Output, except during an eclipse is 30 watts. Storage batteries with a capacity of 50 watt-hours are charged by part of the output of the solar cells to provide for intermittent power needs in excess of the 30 watts from the solar cells.

6.8.4 Injection Rocket
The payload carries an Atlantic Research Corporation 1 KS 420 solid-propellant rocket. This rocket is

6.8.4 Injection Rocket (Continued)

ignited by command from the doppler and command system only under certain specified conditions. Purpose of the injection rocket is to impart additional velocity to the payload if a satisfactory satellite lifetime cannot be assured without injection rocket firing. The rocket serial number is S6.

## 7.0 COUNTDOWN REVIEW

### 7.1 Mock Countdowns (3 August 1959)

A mock countdown for the Able personnel began at 1730 EST, 3 August 1959. The mock countdown was conducted to familiarize Able personnel with the Able ordnance and preparation task.

A mock countdown for all personnel began at 1300 EST, 3 August 1959. This mock countdown was conducted to familiarize all personnel with the tower removal, final preparations, and terminal count tasks.

### 7.2 Flight Test Countdown (6 August 1959)

The flight test countdown, AFMTC Test No. 1005, began at 1325 EST, 6 August 1959.

During the payload experiment task, certain payload telemetry subcarriers were found to be slightly out of band. This was due to high temperature of the payload and was accepted.

The main fuel tank pressure transducer was replaced twice during instrumentation calibrations.

While conducting electrical systems tests, an irregularity was noted in the output of the AZUSA transponder and it was planned to replace the transponder. A special test was made at 2051 EST and a slight improvement was noted.

7.2 Flight Test Countdown (Continued)

It was decided to recheck the transponder after tower removal.

Because of the uncertainties of the operation, the payload solar cell paddle installation and checkout were conducted during the built-in two hour hold. All modules operated correctly.

Fuel flow to the first stage began at 0315 EST, but was stopped at 0317 EST because of a leak in the vernier fuel vent umbilical. This leak was corrected and fuel flow was resumed at 0332 EST. Fuel flow was stopped at a sight gage level of 30.95 inches and a Brodie flowmeter reading of 4901 gallons.

Liquid oxygen flow to the first stage began at 0625 EST and was stopped at 0745 EST when the 100 per cent float switch was activated. Topping was initiated at 0824 EST.

At 0824 EST, T-18 minutes, a truck was reported stuck in the ELSSE site,making the site inoperative. A hold was called at 0852 EST, T-8 minutes,to allow removal of the truck. Also, weather conditions were unsatisfactory for a brief period during this hold.

After holding for 15 minutes, the AGS calibrations to determine angular constants were no longer valid and had to be rerun.

7.2 Flight Test Countdown (Continued)

The count was resumed at 0915 EST, T-8 minutes, with the ELSSE site still inoperative. The AGS calibrations were completed at approximately T-5 minutes. Liftoff occurred at 0923:22.239 EST, 7 August 1959.

The start and completion times for major countdown tasks are given below:

| | Actual Start | | Actual Completion | |
|---|---|---|---|---|
| | T-Time | Real Time | T-Time | Real Time |
| Countdown Initiation | 995 | 1325 | 990 | 1330 |
| Payload Experiment | 990 | 1330 | 870 | 1530 |
| Able Guidance Link Test | 840 | 1600 | 800 | 1640 |
| Instrumentation Calibrations | 800 | 1650 | 707 | 1813 |
| Engine Checks | 735 | 1745 | 615 | 1945 |
| Electrical Systems Checks | 645 | 1915 | 525 | 2115 |
| Second Stage Fueling | 510 | 2130 | 315 | 0045 |
| Built-In Hold | 300 | 0100 | 300 | 0300 |
| Payload Electrical Checks | 300* | 0200 | 300* | 0245 |
| First Stage Fueling | 300* | 0255 | 245 | 0355 |
| Second Stage Ordnance | 300* | 0255 | 120 | 0600 |
| Regulator Settings | 300 | 0300 | 233 | 0407 |
| First Stage Ordnance | 233 | 0407 | 170 | 0510 |
| Liquid Oxygen Filling | 120 | 0600 | 35* | 0800 |
| Second Stage Helium Checks | 120 | 0600 | 95 | 0625 |
| Tower Removal | 120 | 0600 | 35 | 0725 |
| Final Preparations | 75 | 0645 | 35 | 0825 |
| Built-In Hold | 35 | 0725 | 35 | 0825 |
| Terminal Count | 35 | 0825 | 0 | 0923:22.239 |
| Liftoff | | | 0 | 0923:22.239 |

*During Hold

## 8.0 EXTERNAL INSTRUMENTATION

External optical instrumentation consisted of 11 metric, 21 engineering sequential and 21 documentary cameras. A listing of these items and their disposition appears in Table VI.

Radar coverage was obtained from Mod II 1.4, 1.5, Mod IV, and the FPS-16 located on the Cape. Coverage was also obtained from the XN-1 at Patrick Air Force Base and the XN-2 at Grand Bahama Island.

An AZUSA transponder was carried aboard the missile for trajectory information and for range safety purposes. Good coverage was obtained from 13 to 250 seconds with a data dropout occurring between 151 to 156 seconds.

Telemetry ELSSE obtained coverage for the first 300 seconds of flight.

Telemetry data was received at the Cape Telemetry Station and from Spruce Creek, Vero Beach, and Grand Bahama Island.

A listing of electronic items is contained in Table VII.

Figures 3, 4, and 5 show the instrumentation sites.

Payload telemetry was received and tracked by all stations in the worldwide network used for this operation, including the STL stations at Manchester, Hawaii, and Singapore; NASA Minitrack stations at Antafagasta, Woomera, Antigua, Grand Turk, Cape Canaveral, NEL-San Diego, and Johannesburg; and the Millstone station.

Powered flight tracking data was also received from Millstone skin-tracking, Minitrack, the ABMD Microlock trailer, the STL telemetry van, and the NASA Cape doppler station.