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Standard Launch Vehicle I (SLV-I) System Description and Operation Summary

SEPTEMBER 1963

Prepared by R. H. McCULLOCH
Reliability Department

Prepared for COMMANDER SPACE SYSTEMS DIVISION
UNITED STATES AIR FORCE
Inglewood, California

ENGINEERING DIVISION • AEROSPACE CORPORATION
CONTRACT NO. AF 04(695)-269

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Report No.
TDR-269(4303)-2

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Prepared by R. H. McCulloch,
Surveillance and
Failure Analysis

This technical documentary report has been reviewed and is approved.

For Space Systems Division
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FOREWORD

This report presents a brief summary of SLV-I flight parameters and hardware performance as reported in the contractor flight evaluation report for each flight. The primary purpose of the report is to present a malfunction summary for ease of reference in conducting reliability evaluations. Since the SLV-I program is complex, a system description was deemed necessary in order to understand the system problems, capabilities, and configurations. With the inclusion of the system description the report becomes a ready reference to the SLV-I program for all interested parties.

This report should be used only for informational planning purposes. Detailed information relating to specific capabilities of the SLV-I vehicle should be requested from the Air Force Standard Launch Vehicle I (SLV-I) Program Office (SSVB).

ABSTRACT

An SLV-I system description and flight operation summary is presented for ready reference to the SLV-I program and to facilitate reliability evaluations. The system description is brief and primarily concerned with defining configuration differences. The flight operation summaries are extracts of system hardware performance and anomalies as reported in contractor flight evaluation reports.

The SLV-I is an inexpensive, diversified missile used for space exploration. Basic configurations of SLV-I used are: LV-1B, three-stage (unguided); SLV-IB, four-stage (unguided); BSI*, three-stage (guided); BSII*, four-stage (guided); and SLV-IA, four-stage (guided). Various payload requirements can be met by utilizing available motors. Flight histories, diagrams, charts, schematics, and tables are included.

*BSI is Blue Scout I, now obsolete.
BSII is Blue Scout II, now obsolete.

REVISION INSTRUCTIONS

The Standard Launch Vehicle I (SLV-I) System Description and Operation Summary has been compiled as a central reference source for all launch vehicles of the SLV-I family. Information contained in this report is considered valid as of July 1963.

This report will be updated periodically as additional information becomes available. Changes will be accomplished either by reissue of the entire document or by transmitting one or more revision pages. Reissues (complete republication) will be identified by the word Reissue and a sequential letter of the alphabet, beginning with capital A, placed below the report number on each page. If changes are made by revision pages, each page will be identified by the abbreviation Rev and a sequential Arabic number, beginning with one, placed immediately below the report number. The revision number reflects the total number of revisions to the particular report issue, not necessarily the number of revisions to any particular page.

A new title page and Revision Summary Sheet will accompany each revision. Revision title pages will carry the revision date below the revision designation. Title pages for reissues will carry a new publication date and supersession statement.

Revision summary sheets should be retained as a permanent part of the particular document issue to serve as a record of all change action. Revision summary sheets replace previously issued sheets if they include all prior change information; otherwise they are handled as additional pages and numbered accordingly.

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SECTION I

STANDARD LAUNCH VEHICLE I FAMILY

1.1 INTRODUCTION

The SLV-I program provides the Air Force and other governmental agencies with a family of economical and versatile standard space booster vehicles for supporting space systems, tests of space subsystems, and research programs. The SLV-I vehicles have the inherent simplicity and reliability of solid propellant vehicles and are more economical, for many applications, than the liquid boosters derived from ballistic missile programs.

1.1.1 Capability

The SLV-I vehicles will currently:

a. Place a 200-pound payload into a 400 nautical mile circular orbit

b. Boost a 200-pound payload to 4,000 nautical miles on a probe trajectory

c. Boost a 35-pound payload to 75,000 nautical miles on a probe trajectory

d. Place a 400-pound payload into a boost-glide trajectory at a velocity of 20,500 feet per second at 250,000 feet altitude.

Besides orbital flights, probes, and boost-glide trajectories, the vehicle provides downward booster high-speed reentry profiles. Data recovery, attitude-stabilized final stage, and payload capabilities, are also provided. Performance parameters are increasing due to continued booster motor development and modification. Performance comparisons of SLV-I vehicles are shown in Figure 1.

1.1.2 Program Phasing

The program is divided into two phases, development and application. The development phase is being used to refine and flight test the solid propellant vehicles; to train Air Force personnel in preparation and launch of the vehicles; and to accomplish the SLV-I Program objectives. The application phase will support user-program objectives. Vehicle receipt, assembly, payload mating, checkout, and launch will be accomplished by Air Force personnel during this phase.

From 26 April 1962, NASA and Air Force personnel, in a coordinated effort, have launched the new series of SLV-I missiles (111, 112 etc.). BSI and BSII (guided) have been phased out of the program; however, the SLV-IB and LV-IB (unguided) missiles remain an Air Force responsibility. The SLV-IA (guided) missile, with the newer or modified motors, will remain the basic SLV-I of NASA and the Air Force. Joint effort by the Air Force and NASA was maintained until a successful launch was accomplished on the new series of SLV-IA missiles; the Air Force military personnel then assumed prime responsibility on all subsequent Air Force flights from the Pacific Missile Range, with NASA retaining launch responsibilities for launches from Wallops Island.

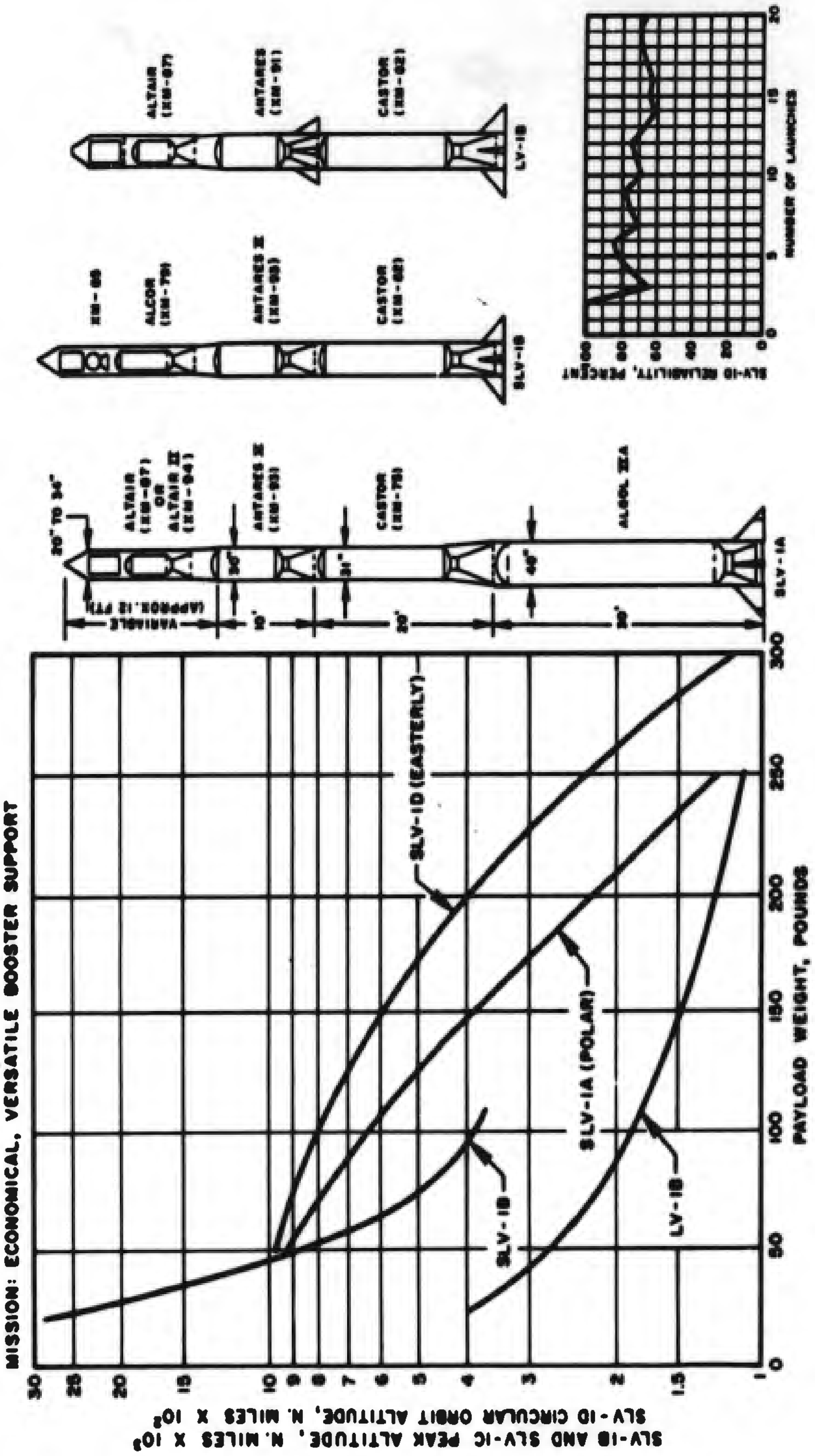


Figure 1. SLV-I General Configuration and Performance

1.2 VEHICLE CODING SYSTEM

A new coding number designation for the SLV-I family has been initiated, and for this publication the new numbers will be used. The new coding numbers for the SLV-I vehicles are shown below:

Old Number	New Number	Motor Configuration*
Scout I	BSI (Discontinued)	Guided, 1 - 2 - 3
Scout II	BSII (Discontinued)	Guided, 1 - 2 - 3 - 4
Scout Jr.	SLV-IB	Unguided, 2 - 3 - 5 - 6
Modified Scout (Scout Jr)	LV-IB	Unguided, 2 - 3 - 4
Scout (Guided) (AF and/or NASA)	SLV-IA	Guided, 1 - 2 - 3 - 4 or 1 - 2 - 3 - 7**

The current SLV-I vehicles will no longer be referred to as Blue Scout or Scout but as SLV-IA, SLV-IB and LV-IB, whichever is applicable.

1.3 ROCKET MOTOR DESIGNATIONS

Rocket motor designations and configurations are summarized in Table 1.

*See Table 1 for motor description.

**Non-standard 5th stage.

Table 1. ROCKET MOTOR DESIGNATIONS

A. The general rocket motor configurations for current and future SLV-I vehicles are listed below:

AF Desig.	NASA	Motor Configuration	Former Desig.
SLV-IB	--	2-3-5-6	Blue Scout Jr.
LV-IB	--	2-3-4	Modified B. S. Jr.
SLV-IA	Scout	1-2-3-4 or 1-2-3-7*	Air Force Scout

*nonstandard

B. Specific motor types are assigned to the numbered configurations above, to meet mission requirements, as follows:

Motor Config. No.	Manufacturer	DESIGNATION Manuf.	AF	NASA	Using Vehicle
1	Aerojet Gen. Corp.	AJ Sr.	XM-68	ALGOL ID	SLV-IA (obsolete)
1a	Aerojet Gen. Corp.	AJ Sr.		ALGOL IIA	SLV-IA
2	Thiokol Chem Corp.	XM33E5	XM-75	CASTOR	SLV-IA
2a	Thiokol Chem Corp.	XM33E7	XM-82	CASTOR	SLV-IB, LV-IB
3	Allegany Ball.Lab	X254A1	XM-91	ANTARES	LV-IB (obsolete)
3a	Allegany Ball.Lab	X259A2	XM-93	ANTARES II	SLV-IA
3b	Allegany Ball.Lab	X259A4	XM-93	ANTARES II	SLV-IB
4	Allegany Ball.Lab	X248A5	XM-87	ALTAIR	LV-IB, SLV-IA
4a	Allegany Ball.Lab	X258A1	XM-94	ALTAIR II	SLV-IA
5	Aerojet Gen. Corp.	AJ10-41	XM-79	ALCOR	SLV-IB
6	Naval Ord.Test Sta.	17 in. Spherical	XM-78	CETUS	SLV-IA* (obsolete)
6a	Naval Ord.Test Sta.	Impr. 17 in. Sph.	XM-85	CETUS	SLV-IB, SLV-IA
7	Lockheed Prop. Co.	MG-18	--	--	SLV-IA

*also used as nonstandard 5th stage

C. The following is the status of previously used TS609A/Blue Scout Program vehicle designations:

Popular Name	AF Desig.	Configuration	Remarks
Blue Scout I	XRM-89	1-2-3	Vehicle obsolete
Blue Scout II	XRM-90	1-2-3-4	Vehicle obsolete
Blue Scout Jr.	XRM-91	2-3-5-6	Redesignated SLV-IB
Modified B. S. Jr.	--	2-3-4	Redesignated LV-IB
A. F. Scout	XRM-92	1-2-3-4S	Redesignated SLV-IA

D. Four motor subtypes used during the development program are obsolete and are not included in the above motor list. These are two Aerojet Seniors (ALGOL IB and ALGOL IC), the ABL X254A2 (XM-70) and the ABL X248A6 (XM-69) motors.

E. Three motors are in the process of being phased out by improved motors. These are the ALGOL ID by the ALGOL IIA, the ABL X248 (XM-87) by the ABL X258 (XM-94), and the NOTS 17" spherical motor XM-78 by the improved NOTS 17" spherical motor XM-85.

F. Motor configurations used on SLV-I launches are governed by the primary mission objectives of each flight. (See page 52, Table 8, for motor configuration tabulation on past launches.)

SECTION II

SLV-IB VEHICLE SYSTEM

2.1 DESCRIPTION

The unguided SLV-IB vehicle consists of four solid propellant rocket motors with associated interstages and wiring. The vehicle is aerodynamically stabilized by fins on the first stage. The vehicle is initially spun by jettisonable spin rockets at launch and spin is maintained by canted aerodynamic fins. A second stage burnout spin rate of approximately 180 rpm maintains the attitude of the third and fourth stages.

The four stages are ignited in sequence. The first is ignited by ground command through the aft umbilical. The spin rockets are fired by a lanyard cable as the missile clears the launching boom. Second stage ignition is commanded by a first stage pressure switch, and separation is effected by fracturing a diaphragm connecting upper C section to lower C section (Figure 2). During flight through the atmosphere, the third stage, fourth stage, and payload are protected by a single clamshell heatshield. Heatshield separation is commanded by a timer located in the interstage between the third and fourth stage motors. The heatshield separates into halves and spin forces them away from the vehicle.

A third stage pyrotechnic delay causes the third stage to ignite approximately one second after heatshield separation (about six seconds after second stage burnout). Upon ignition, third stage separation is effected by diaphragm fracture. Within two seconds of third stage burnout, the fourth stage is ignited by command from the same timer used for heatshield separation and third stage ignition. Separation is accomplished by diaphragm fracturing at the forward end of the interstage. Payload timing functions, if required, are provided as an integral part of the payload. Payload timers are mechanical devices activated at launch through the forward umbilical.

2.1.1 System Elements

Subsystems on the vehicle include one destruct system, two ignition systems and one telemetry system.

2.1.1.1 Flight Termination

The command destruct components, located at the aft end of the second stage motor, provide for destruction of the first two stages and interruption of ignition of the last two stages during the first two periods of powered flight. Destruction is accomplished by 500-gram conical destruct charges located in transition section B.

2.1.1.2 Ignition

Two ignition systems are provided, one for the first two stages, and another for the last two stages. The former is located at the forward end of the first stage motor and the latter is located at the forward end of the third stage motor. Connection of the two ignition systems is through the command destruct relays. The second stage is ignited by a pressure switch on the first stage, whereas ignition of the third and fourth stages is from a timer started at launch.

2.1.1.3 Telemetry

The telemetry system is located in the payload compartment ahead of the fourth stage motor. The telemetry system is of special design utilizing small, lightweight components. Typically, four- to ten-channel FM/FM systems are used.

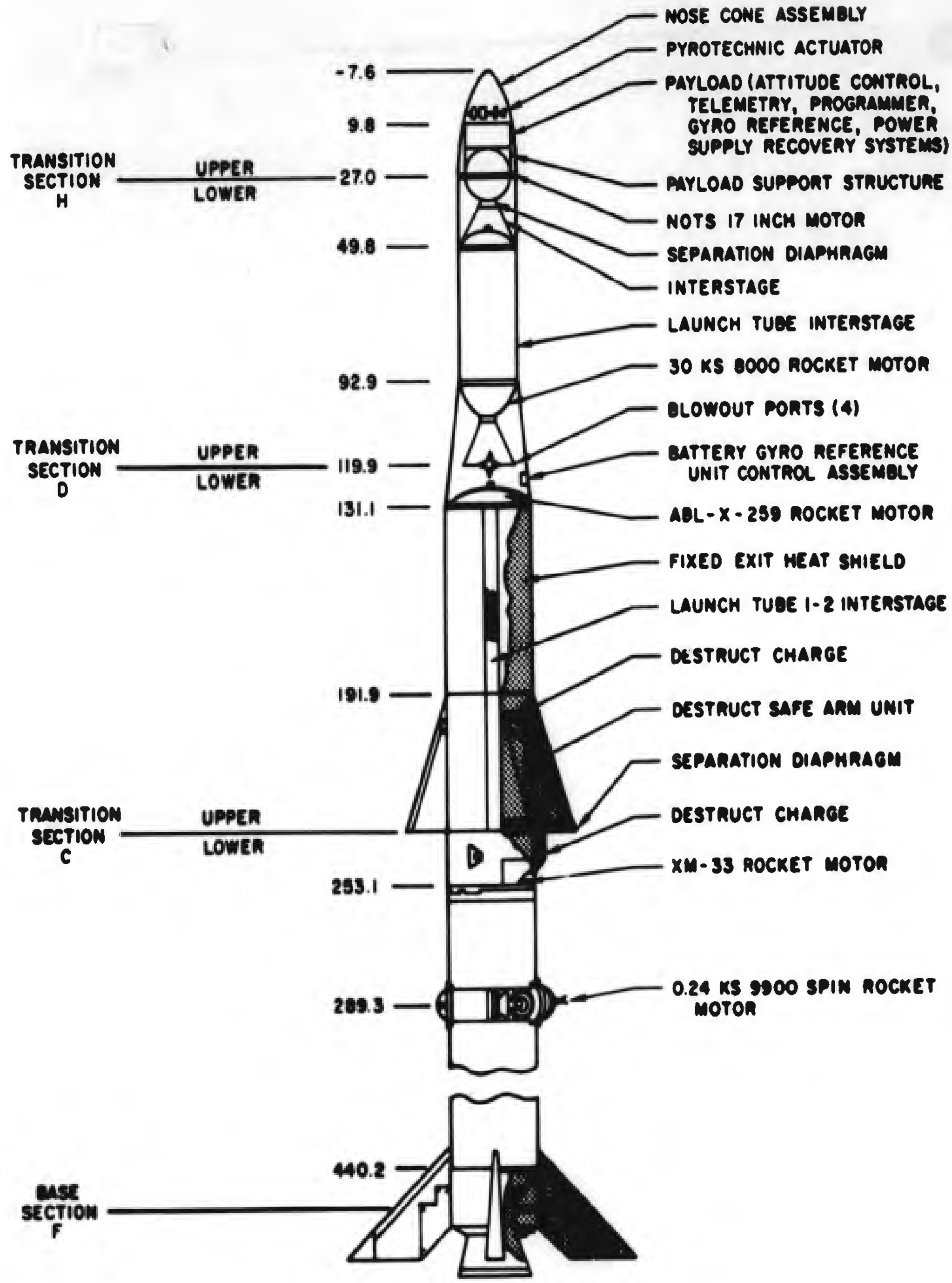
2.1.2 SLV-IB Vehicle Flight Sequence (Typical)

First stage ignition	T=0
Spin rocket ignition (lanyard)	T+0.65 sec
Spin rocket jettison (delay squib)	T+1.49 sec
Second stage ignition (timer, pressure switch)	T+37.1 sec
Nose cone separation (timer)	T+81.8 sec
Nose halves separation (delay squib)	T+82.3 sec
Third stage ignition (delay squib)	T+82.8 sec
Fourth stage ignition (timer)	T+115.8 sec
Apogee	T+4.75 hours
Nominal impact	9.35 hours

2.1.3 Payload

Major payload components are power supply, telemetry system, mechanical timer, checkout relays, and the payload structure.

An electrical interface between the payload and the forward umbilical connector consists of wires bonded to the case of the fourth stage motor. Wiring extends through pull-away plugs at the aft end of the motor, to the umbilical plug mounted externally on the interstage between the third and fourth stage motors. The wiring is required for arming and disarming the payload during prelaunch countdown and for firing squibs in payload timers at launch.



NOTE:
THE AERODYNAMIC STABILIZER FINS ON THE SECOND STAGE OF THE NEW SLV-1B HAVE BEEN ELIMINATED

Figure 2. SLV-1B, General View of Old Configuration

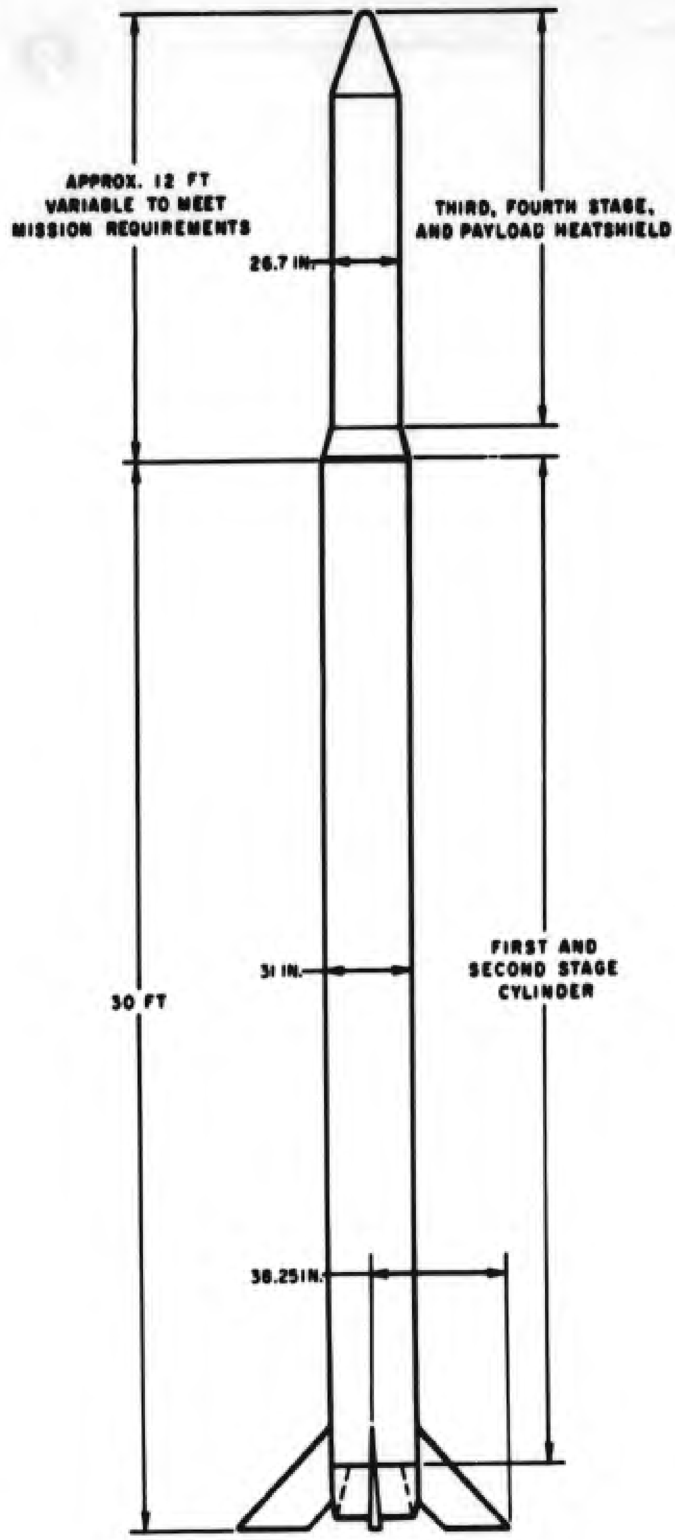


Figure 3. SLV-1B, External View of New Configuration

SECTION III

BSI AND BSII VEHICLE SYSTEM

3.1 DESCRIPTION

These configurations are obsolete developmental types and are no longer in use. This section is for historical reference only, since the only guided SLV-I vehicle now in production is the standardized SLV-IA (see section IV, page 23).

The BSI configuration consisted of three solid propellant stages. The spacecraft was attached to the forward end of the third stage motor.

The BSII configuration had the same external appearance; however, a solid propellant motor and a hot-gas control system were incorporated within the envelope of the payload section, providing a fourth stage motor and fourth stage attitude control. The payload volume was thereby reduced to gain increased vehicle performance and an orbital capability. The gyro reference unit for both BSI and BSII configurations was installed within the spacecraft envelope.

A standard set of interstage structures was utilized which provided structural continuity and housed the attitude control system, telemetry, and destruct components. The separation devices were also provided in the interstages. Base section A (Figure 4), attached to the aft end of the first stage, incorporated four stabilizing fins with aerodynamic control surfaces at the tips to provide proportional attitude control. These control surfaces were linked to jet vanes in the rocket exhaust to provide control at the initial, low vehicle velocities. Base A housed the hydraulic equipment and associated electrical power supply to operate the fins, and contained provisions for mounting the vehicle on the launch pad. The flare provided additional equipment space and increased aerodynamic stability.

The first-to-second stage interstage structure, section B, housed attitude control systems, separation diaphragm, and elements of the command destruct system. The BSI and BSII configurations utilized a standard interstage structure that connected the payload section of the spacecraft to the third stage motor. In the BSII this section incorporated an attitude control system for the fourth stage, and a shaped-charge separation device.

3.1.1 Flight Sequence

Flight was controlled by the guidance and control system (BSII only). The guidance system generated signals to dynamically stabilize the vehicle, and generated a pitch program to obtain the desired trajectory. The programmer generated discrete signals required to accomplish engine ignition, stage separation, heat shield separation, and other functions of an electrical nature. Predetermined coast periods between stages were programmed to satisfy trajectory requirements. The programmer was capable of timing 28 separate events.

In normal flight, the first stage burned approximately 40 seconds. Burnout was followed by a 20-second coast period before second stage ignition. The second stage burned approximately 30 seconds. Exit heat shields were separated and third stage ignited immediately after burnout of the second stage. The third stage burned approximately 35 seconds, and, for the four-stage configuration, burnout was followed by a fourth stage burning period of approximately 40 seconds. Recovery vehicle separation was normally commanded from a downrange ground station.

3.1.1.1 Ignition

The first stage was ignited through umbilical cables. Electrical power and timer signals were provided to ignite subsequent stages. Standard pyrotechnic igniters were utilized on motors. Ignition wiring was redundant for increased reliability.

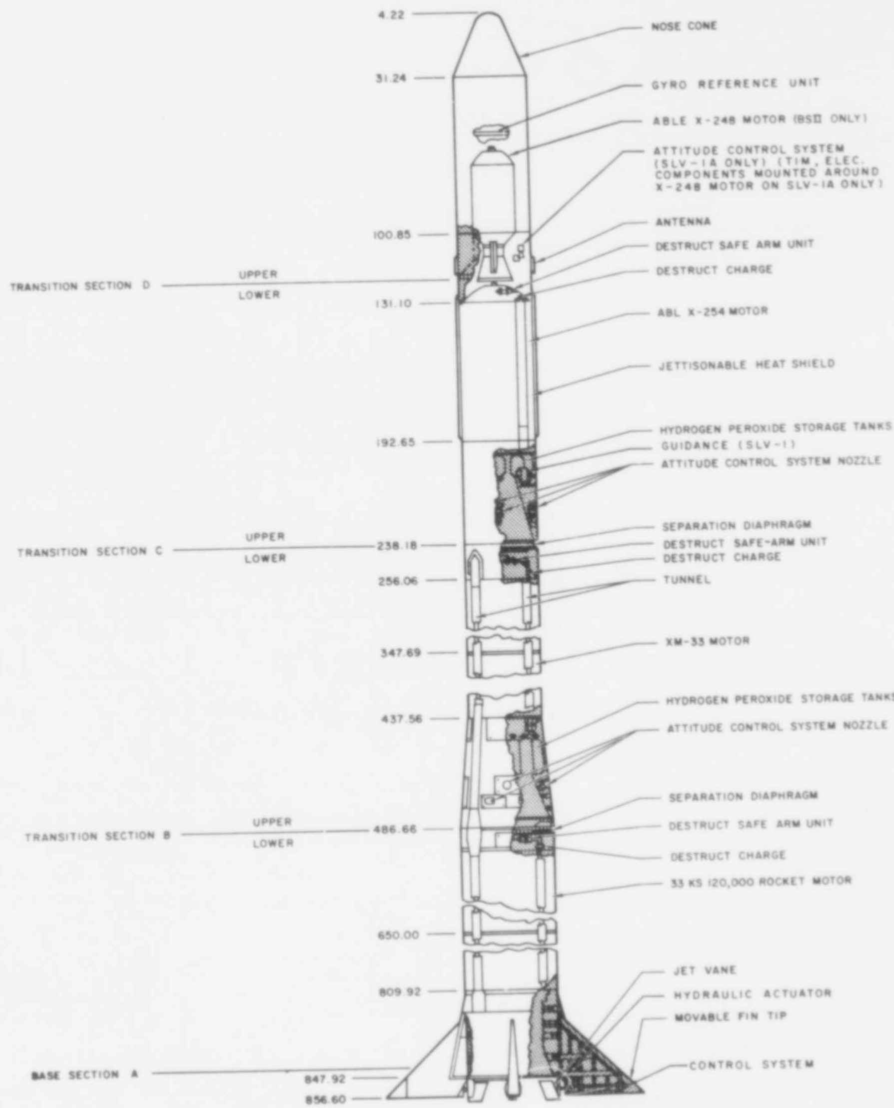


Figure 4. BSI and BSII, General View

3.1.1.2 Separation

First-to-second stage and second-to-third stage separation was achieved by fracture of an interstage diaphragm. The diaphragm was a structural member which secured the upper and lower interstage structure. It was mounted at the motor nozzle exhaust and slotted to cause breakup when the motor fired. In the BSII configuration, fourth stage separation was achieved with a linear-shaped charge installed around the internal periphery of the third-to-fourth stage interstage structure.

3.2 GUIDANCE (BSII)

The guidance system (BSII only) generated signals to stabilize the vehicle on three flight axes and, in addition, controlled the trajectory by open loop pitch programming. The system consisted of three single-axis displacement gyros, three single-axis rate gyros, a pitch programmer, timer, integrators, poppet valve electronics, and power supply (see Figure 5).

The guidance system provided attitude and stabilization control signals for the first stage hydraulic servo control system, the second and third stage hydrogen peroxide reaction control systems, and, on the BSII configuration, the fourth stage hot gas control system. It provided ignition sequence signals to second, third, and fourth stage rocket motors; initiated separation of the third and fourth stages; and initiated fourth stage spin rocket ignition.

The guidance system initiated yaw and roll stabilization and pitch program control signals to provide vehicle control during flight. These signals were generated in the inertial reference package. Displacement error signals from miniature integrating gyros were combined with rate error signals from the rate gyro unit and amplified. During first stage operation the amplified signals were applied to the hydraulic servo control system. During second and third stage operation the amplified signals were applied to the poppet valve electronics and converted to on - off signals for control of hydrogen peroxide reaction thrust systems.

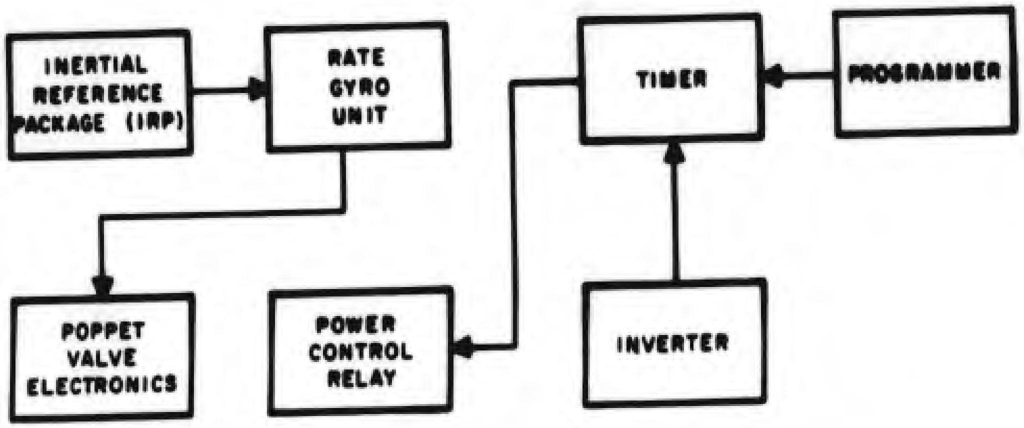


Figure 5. BSII Guidance System Block Diagram

Yaw and roll displacement error signals were initiated by deviations in vehicle yaw and roll attitude, using orientation at launch as a reference. Pitch program displacement signals were initiated by a DC signal from the programmer to the pitch displacement gyro torque generator. The pitching rate was determined by signal current, and the programmed angle was determined by the time interval the signal was applied. Ten different pitch signal levels (program steps) were available from the programmer. They were routed through the timer to control the sequence and interval of pitch signal application.

Rocket motor ignition, fourth stage separation, and spin motor ignition signals were sequenced during flight by timed switch closures in the system timer.

Pitch programming and timed function requirements varied with individual missile objectives. The required number of pitch program steps, the proper rate and total angle of each step, and the required sequencing of time functions for a specific mission, were preset in the guidance system by rewiring the timer and programmer.

3.2.1 Control

Vehicle attitude control during first stage operation was achieved by proportional control of jet vanes and aerodynamic tip fins connected on common torque bars. Hydraulic actuators were the prime movers. The second and third stage control consisted of hydrogen peroxide reaction jets. The primary or high thrust pitch and yaw nozzles were mounted normal to the vehicle axes.

The four roll nozzles were mounted circumferentially and worked in tandem, providing pure couples. The reaction jets used hydrogen peroxide fuel which was stored in the interstage areas and was pressure fed with compressed nitrogen.

Each jet or motor incorporated a silver screen catalyst bed, and a solenoid-actuated control valve. The command signal for all peroxide motors was provided by poppet valve electronics located in the spacecraft. The BSII four stage vehicle incorporated a solid propellant hot gas system for control during fourth stage burning. In addition, a low thrust nitrogen system was used during the vehicle coast phase. The high thrust unit consisted of a hot gas valve porting to two expansion nozzles. The system provided proportional-type control upon signals from the spacecraft guidance system.

3.2.1.1 <u>Fourth Stage Control</u>

Components of the BSII fourth stage control system were divided into two systems, hot gas and cold gas. The hot gas (high thrust) system was employed to provide reaction control forces for compensation of thrust misalignment of the fourth stage rocket motor. The cold gas (low thrust) system was employed to provide reaction control forces for programmed turns and for compensation of fourth stage residual motion at burnout.

The hot gas and cold gas systems were divided into four identical and completely separate subsystems for maximum reliability. Each subsystem, mounted on a removable panel for ease of installation, checkout, and maintenance, controlled the output of two hot gas and two cold gas thrust nozzles (see Figure 6).

In the hot gas system, each of the four subsystems or controller units consisted of a solid fuel gas generator and a diversion (constant area) valve for varying the flow direction, and consequently the reaction thrust, between two opposing nozzles. The valve had a single moving part, a flapper, which was flexibly mounted between two opposing electromagnets and valve seats. This valve was operated by pulse duration modulation in which the flapper was cycled constantly between the valve seats at a frequency of 20 cycles per second. Control of the differential flow or thrust was achieved by increasing

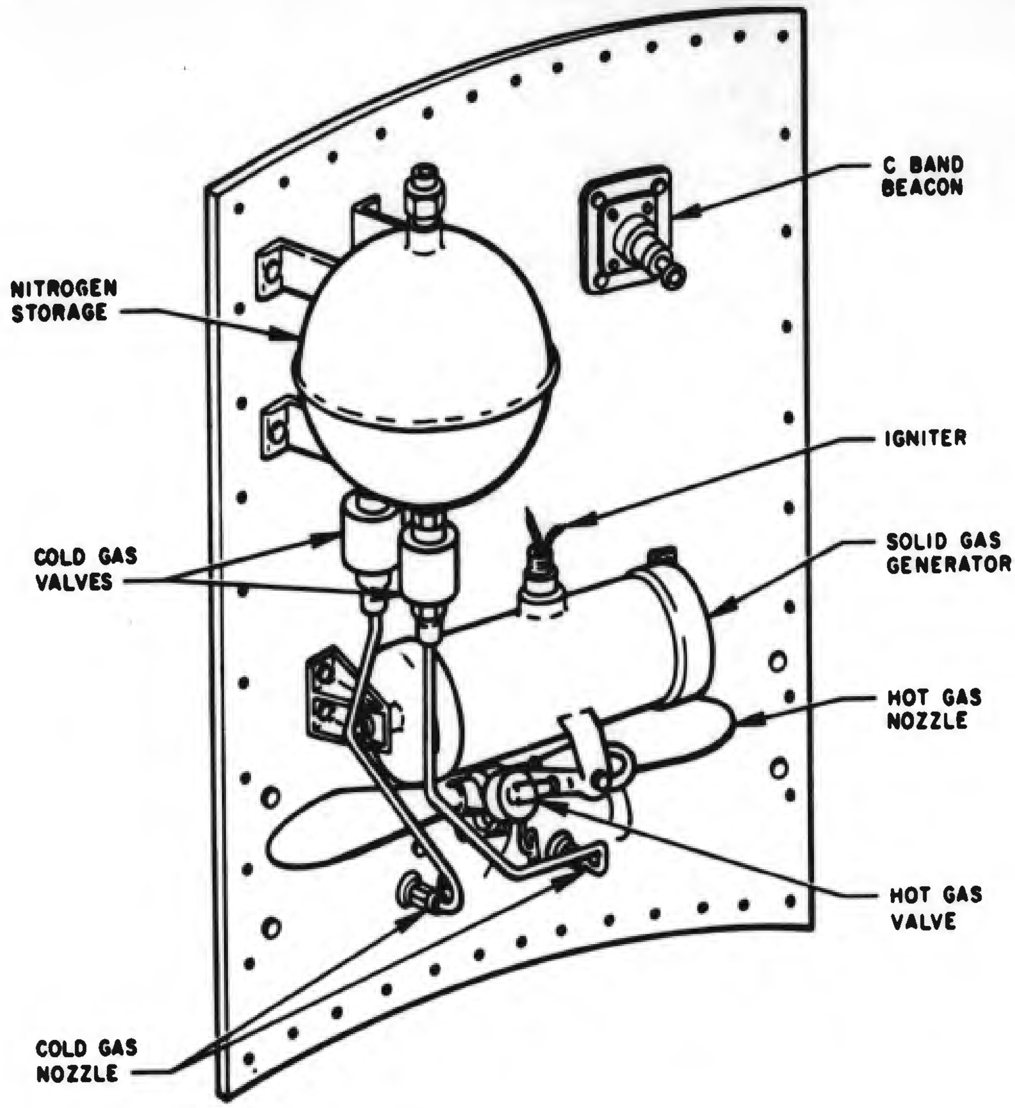


Figure 6. BSII Hot and Cold Gas Control Panel

the dwell time of the flapper on one valve seat and decreasing it for the other. This resulted in proportional-type control.

The system could be generated in a saturated mode of control. With a "hard over" signal the flapper shut off flow to one nozzle and full thrust was realized for the opposite nozzle. A 1.1-pound, dual-grain charge provided a minimum of 4.5 pounds thrust per nozzle for a total duration of greater than 42 seconds. Dual squib ignition, an integral part of the main motor ignition system, initiated full control within 100 milliseconds and a full second before motor ignition (delayed with a pyrotechnic delay train). Control was maintained for approximately 2 seconds after main motor burnout.

Roll-yaw and roll-pitch mixing of the control error signals was employed for uniform and redundant control capacity.

In the cold gas system, a stored nitrogen reaction system was used for coast because of the low control torques and total control impulse required. This system was a simple blowdown device with solenoid valves controlling on-off flow to expansion nozzles.

A spherical nitrogen supply bottle containing approximately 0.5 pounds of dry nitrogen at 3000 psi was mounted on each of the controller panels. Gas was ported through capillary tubes (for flow regulation) to two solenoid valves and expansion nozzles. The capillary flow restrictors were sized to provide 0.2 pounds thrust per nozzle at full tank pressure. Thrust levels decreased as tank pressure decreased due to expanding nitrogen. This resulted in low-rate limit cycle operation and allowed long periods of coast control with minimum gas storage.

3.2.2 Telemetry

The basic telemetering system consisted of a ten-channel FM-FM data link. One or two of the channels might be commutated, resulting in 54 commutated channels in addition to the remaining eight FM channels. Two such telemetery systems could be used. The telemetry system was powered by two 28-volt battery packs installed in the spacecraft. The telemetry antenna system consisted of a pair of slotted-blade antennas spaced 180 degrees apart to provide as nearly omnidirectional patterns as possible. A signal conditioning system mounted in the spacecraft was used to convert the various data inputs into a form suitable for transmission over the telemetry data link.

3.2.3 Flight Termination

The command destruct system consisted of two command receivers, two sets of decoding relays, one arming unit per stage, and a linear shaped charge mounted along the side of each motor case. Receipt of the command-destruct signal from a ground command transmitter would cause termination of the flight by igniting the charges which would, in turn, rupture the motor cases. In addition, an auto-destruct system was provided. If premature stage separation occurred, the lower stages would have been automatically destroyed.

3.2.4 Spacecraft

The spacecraft for the BSI vehicle was 30 inches in diameter, providing a total payload volume of 28.8 cubic feet. It was attitude-stabilized throughout flight.

The spacecraft for the BSII vehicle was identical to the BSI except for incorporation of the fourth stage motor, a fourth stage control system, and a stage separation system. Available payload volume was decreased to 16.9 cubic feet. The shaped-charge separation system, control valves, nozzles, and other fourth stage attitude control system components were located in the interstage structure.

SECTION IV

SLV-IA VEHICLE SYSTEM

4.1 DESCRIPTION

The standard SLV-IA vehicle is a four stage* solid propellant missile with spin-stabilized fourth stage, eliminating the need for a fourth stage control system. The gyro reference package is located in the third stage. Thus the payload weight and volume are increased while the orbital capability of the vehicle is retained.

A standard set of interstage structures is utilized which provides structural continuity and houses the attitude control system, telemetry, destruct components, and separation devices. Base A, attached to the aft end of the first stage (Figure 7), incorporates four stabilizing fins with aerodynamic control surfaces at the tips to provide proportional attitude control. These control surfaces are linked to jet vanes in the rocket exhaust to provide attitude control at initial low vehicle velocities. Base A houses the hydraulic equipment and associated electrical power supply to operate the fins, and contains provisions for mounting the vehicle on the launch pad.

The first-to-second stage interstage structure, section B, and the second-to-third stage interstage structure, section C, house the attitude control systems separation diaphragms, and elements of the command destruct system. This configuration utilizes a standard D section that connects the fourth stage motor and payload to the third stage motor. The upper D section provides for mounting of the fourth stage motor and contains fourth stage spinup rockets.

*A NOTS 17-inch spherical (XM-78 or XM-85) motor may be incorporated into the payload area as a fifth stage motor to meet specific specialized high speed reentry (probe) mission requirements.

4.1.1 Flight Sequence

Flight is governed by the guidance and control system (see G & C description, page 27). The guidance system generates signals to dynamically stabilize the vehicle, and generates a pitch program to obtain the desired trajectory. The timer (see guidance timer description, page 27) also generates discrete signals to accomplish engine ignition, stage separation, heatshield separation, and other functions of an electrical nature. Predetermined coast periods between stagings are programmed to satisfy trajectory requirements. The timer is capable of timing 28 separate events.

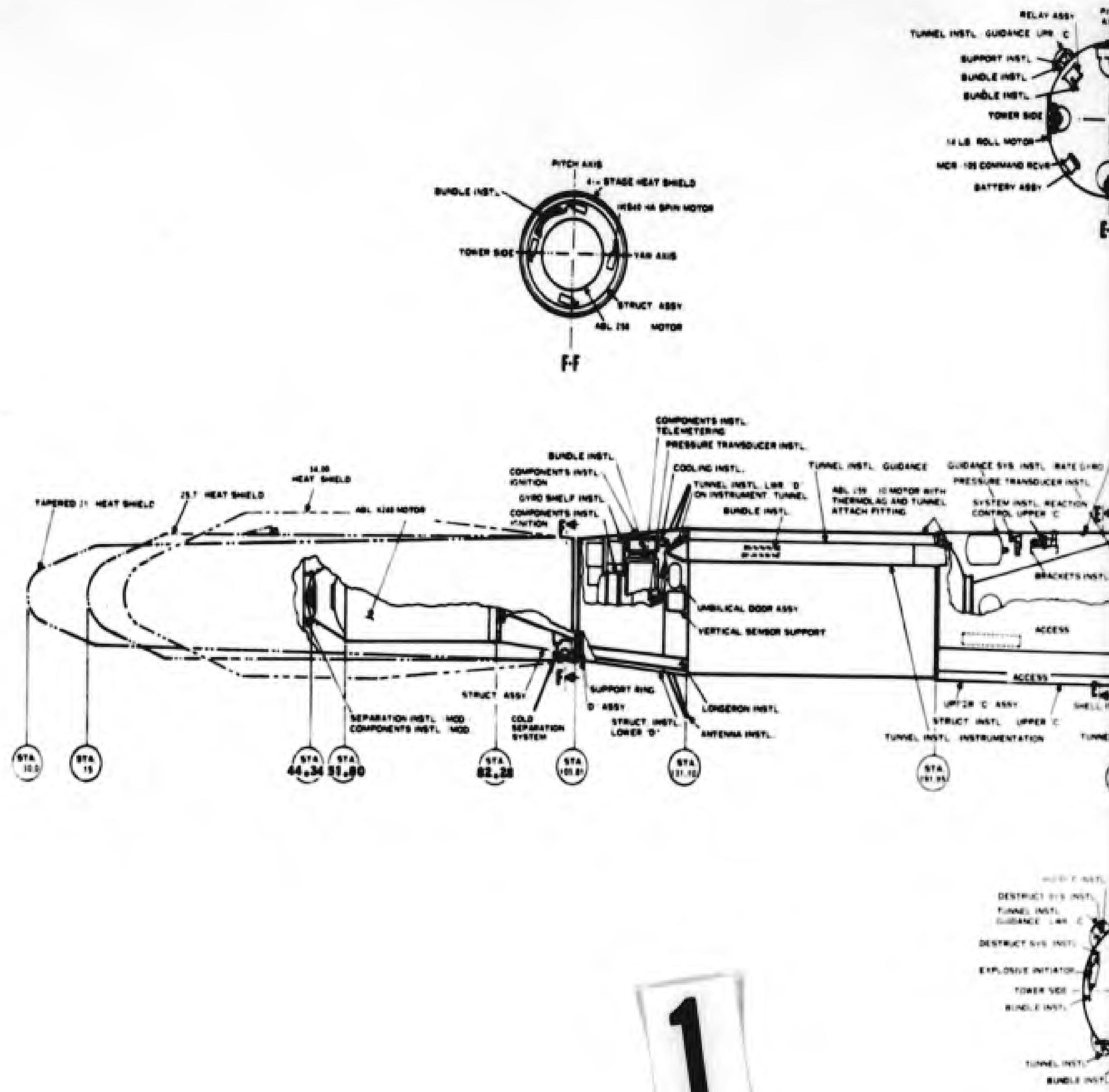
In normal flight the first stage burns approximately 68 seconds and is followed by a 20-second coast period before second stage ignition. The second stage burns approximately 40 seconds. Exit heatshields are ejected and third stage ignited immediately after burnout of the second stage. The third stage burns approximately 30 seconds. As required, to meet mission objectives, the fourth stage and burned-out third stage will coast from 0 to 10 minutes in a stabilized attitude. At the end of coast the fourth stage is spun to 150 rpm and then ignited. The fourth stage then burns approximately 23 seconds.

4.1.2 Command Destruct System

Command destruct components are located in the interstage at the forward end of the third stage motor to provide destruction of the first three stages, and ignition interruption of the fourth stage, during the first periods of powered flight. Destruct is accomplished by a linear shaped charge mounted on each of the first three stages.

4.1.3 Ignition System

Ignition of the first stage is commanded through the umbilical cable. Ignition of the subsequent three stages is accomplished by command from the guidance


PITCH AXIS
BUNDLE INSTL.
TOWER SIDE
YAW AXIS
STRUCT ASSY
F-F
RELAY ASSY
SUPPORT INSTL.
BATTERY ASSY
COMPONENTS INSTL. TELEMETERING
PRESSURE TRANSDUCER INSTL.
COOLING INSTL.
TUNNEL INSTL. GUIDANCE
GYRO SHELF INSTL.
UMBILICAL DOOR ASSY
VERTICAL SENSOR SUPPORT
SUPPORT RING
LONGERON INSTL.
ANTENNA INSTL.
STRUCT ASSY
COLD SEPARATION SYSTEM
ACCESS
BRACKETS INSTL.
TUNNEL INSTL. INSTRUMENTATION
DESTRUCT SYS INSTL.
EXPLOSIVE INITIATOR
TOWER SIDE
TUNNEL INSTL.
STA 44.34
STA 51.80
STA 82.28

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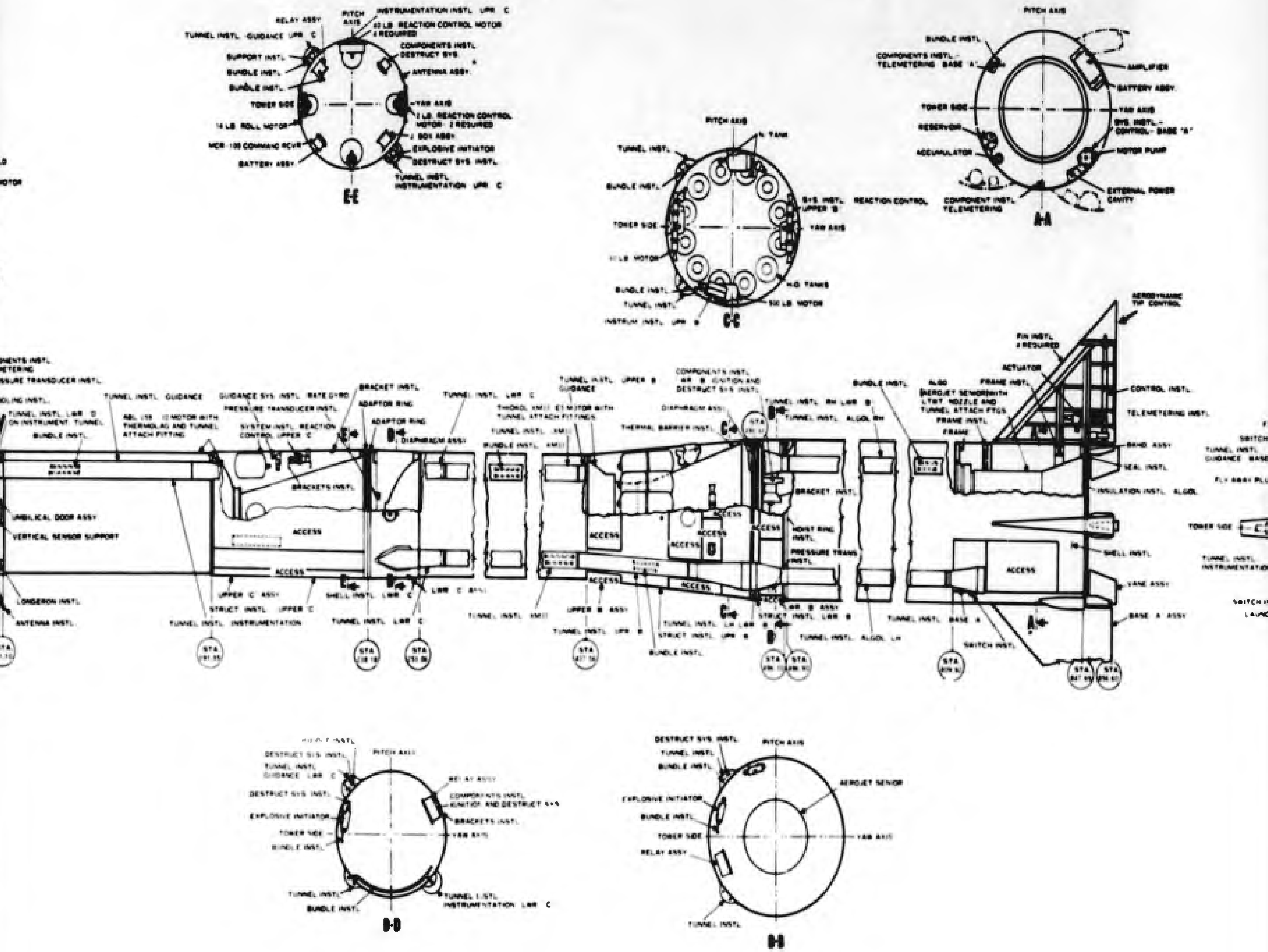
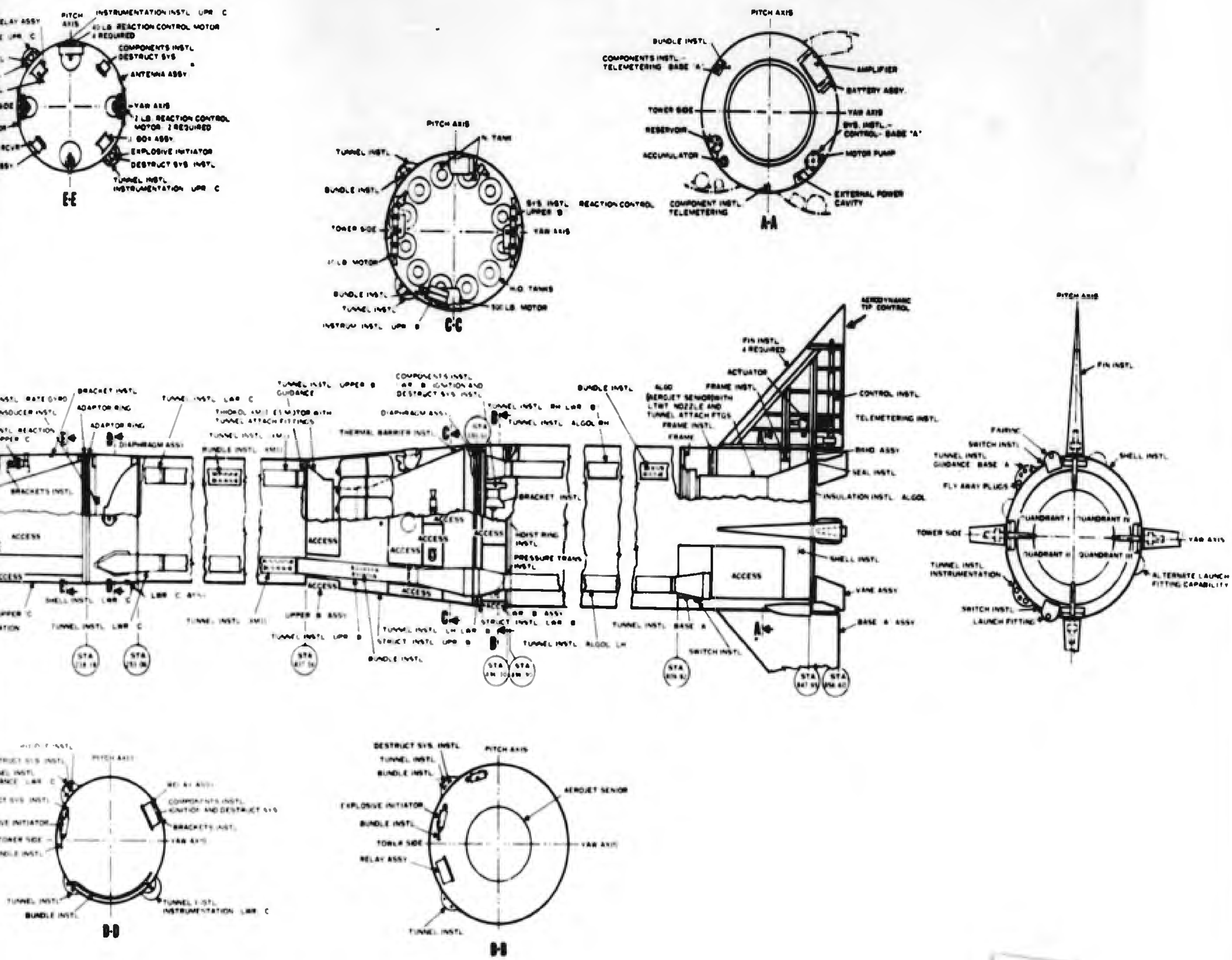


Figure 7. SLV-1A Basic Vehicle

2



3

Figure 7. SLV-1A Basic Vehicle

system timer. Interlock is provided with the command destruct system and pressure switches mounted on each stage.

4.1.4 Guidance System (Minneapolis - Honeywell)

All guidance system components, with the exception of the rate gyro package, are located at the forward end of the third stage motor. The rate gyro package is located at the aft end of the third stage motor. The guidance system pitch rate program is predetermined by digital and analog computer calculations, to provide the desired trajectory. Guidance of the fourth stage is accomplished by aiming and spinning the fourth stage prior to its ignition. The guidance timer initiates switch closures at precisely predetermined times, allowing proper selection of the attitude program rate in the pitch gyro programmer; selection of vehicle command functions such as motor ignition, control gain changes, and heatshield ejection as well as initiation of experiment command functions in the spacecraft.

The timer is capable of providing up to 28 functions (relay closures) over a time interval of 10,000 seconds. It uses the basic 400 cps reference power as a time base. Accuracy is limited only by the 400 cps input frequency.

4.1.5 Guidance and Control Description

The guided versions of the SLV-I vehicle are guided throughout boost by accurate aiming of the motor thrust vector, accomplished by precision control of the vehicle body attitude in space.

Three single degree of freedom gyros are rigidly mounted to the airframe and aligned along the three mutually orthogonal principal axes of the vehicle. They establish a fixed inertial reference attitude in space. Angular deviations of the vehicle from this reference are measured about the output axes of the gyroscopes. These angular deviations, or error signals, command control system torque to rotate the vehicle into the reference attitude. See Figure 8.

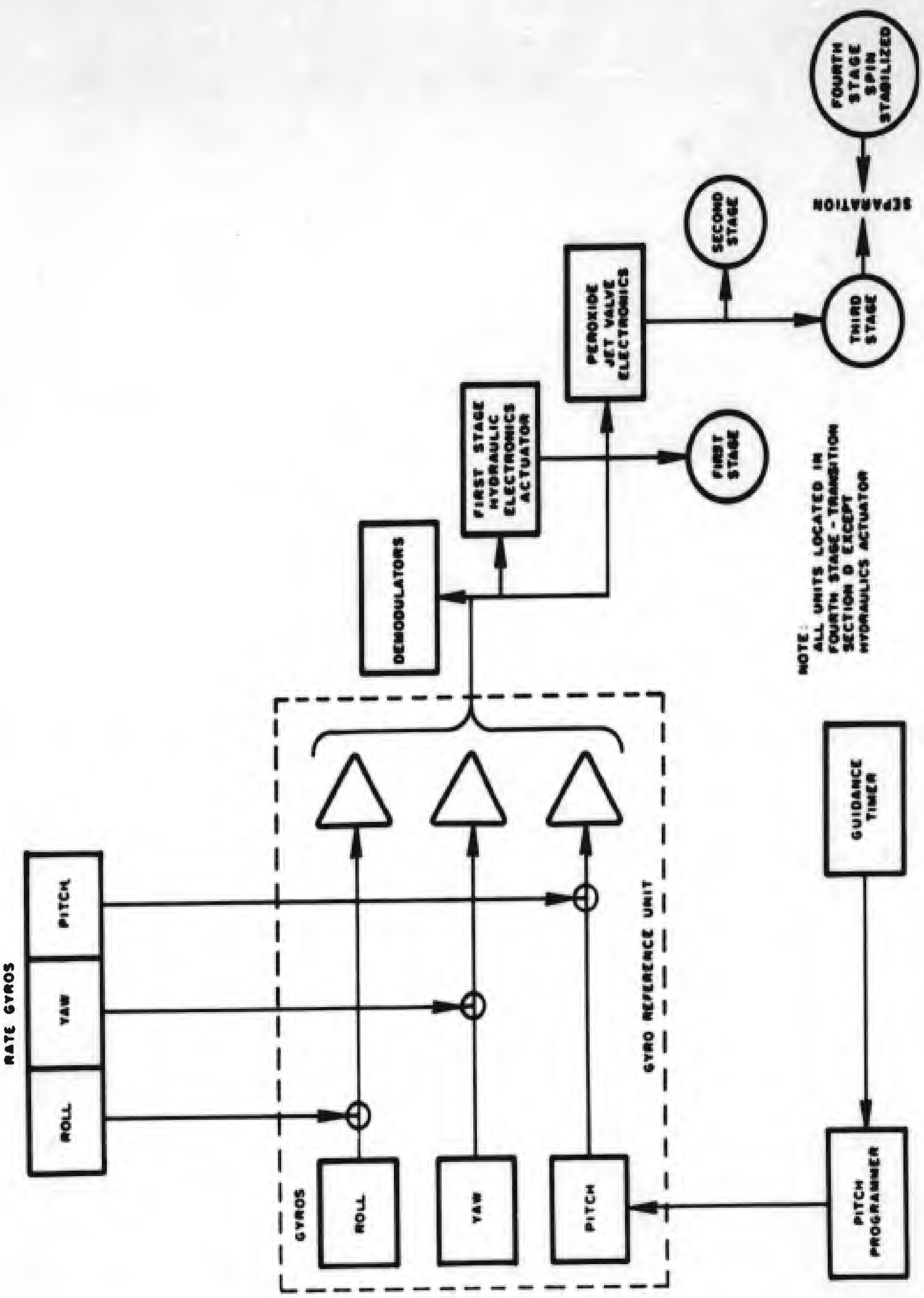


Figure 8. SLV-1A Guidance and Control System Schematic

Change of reference attitude as a function of time is achieved by electromagnetic torquing of the gyro wheel about its input axes. This constitutes open loop time programming of the desired vehicle attitude in space, hence, continuous aiming of the vehicle thrust vector.

Normally gyro torquing is employed about the vehicle pitch axis only. The yaw or azimuth gyro and roll gyro are initially aligned in the guidance plane, thus constraining the vehicle to fly a fixed azimuth heading with a program change in pitch attitude and a resulting change in the in-plane flight path angle. A modified guidance system is available, however, which provides yaw axis programming if it is required.

The control systems of each stage have been chosen to optimize vehicle performance over a broad spectrum of missions. As a result, three modes of vehicle attitude control are sequentially employed through boost.

During the first stage of flight, through the more dense atmosphere, control torque is generated by vanes in the first stage motor exhaust and by movable tip surfaces on the delta planform stabilizing fins. Proportional vane control is provided by a hydraulic actuator servo system located in the motor base section.

Upper stage control utilizes a reaction jet control system. The second and third motor stages are controlled by on-off type jet systems exhausting the products of decomposed hydrogen peroxide.

4.1.6 Control System

Control of the first stage is accomplished by hydraulically actuated jet vanes located at the exit of the first stage nozzle. The jet vanes are linked to movable tips on the first stage fins, providing aerodynamic control to augment the action of the jet vanes and provide control during coast. Second and third stage control is accomplished by operation of hydrogen peroxide jets at the aft ends of the second and third stage motors.

4.1.6.1 Control Components

Each control subsystem for the individual stage is packaged in the interstage structure aft of its corresponding motor. Primary electronics associated with control (signal processing, generation of control error signals) are located in the third stage. This minimizes the number of components and assures control signal continuity through each of the stage separations.

Control system loop damping is achieved by use of three attitude rate gyros orthogonally mounted in the upper C section. This location was chosen after consideration of aeroelastic problems.

a. First Stage Jet Vane Control

The guided vehicles employ a four vane system using individual hydraulic actuators to position each jet vane. A tip fin is connected to each jet vane by a torque bar.

Upper and lower vanes provide yaw torque by in-phase movement and roll torque through differential positioning. The port and starboard vanes provide pitch moment control.

The hydraulic servo system consists of a 28-volt battery supply, pump, accumulator, two servo amplifiers, control signal modulators, and associated mixing electronics. All are located within the interstage structure.

Vane displacements are stop-limited to 20 degrees from null. Servo loop frequencies are maintained at approximately 15 radians per second, with linear operation to servo rate limits of approximately 30 degrees per second. The jet vane/fin torquing system provides adequate torque to pitch the vehicle at maximum design rates of 10 degrees per second at liftoff, and to maintain prescribed coast characteristics to an altitude of 120,000 feet.

b. Second and Third Stage Control

The hydrogen peroxide reaction control systems used on the second and third stages are monopropellant systems utilizing the gaseous products of catalytic decomposition of 90 percent hydrogen peroxide. See Figure 9.

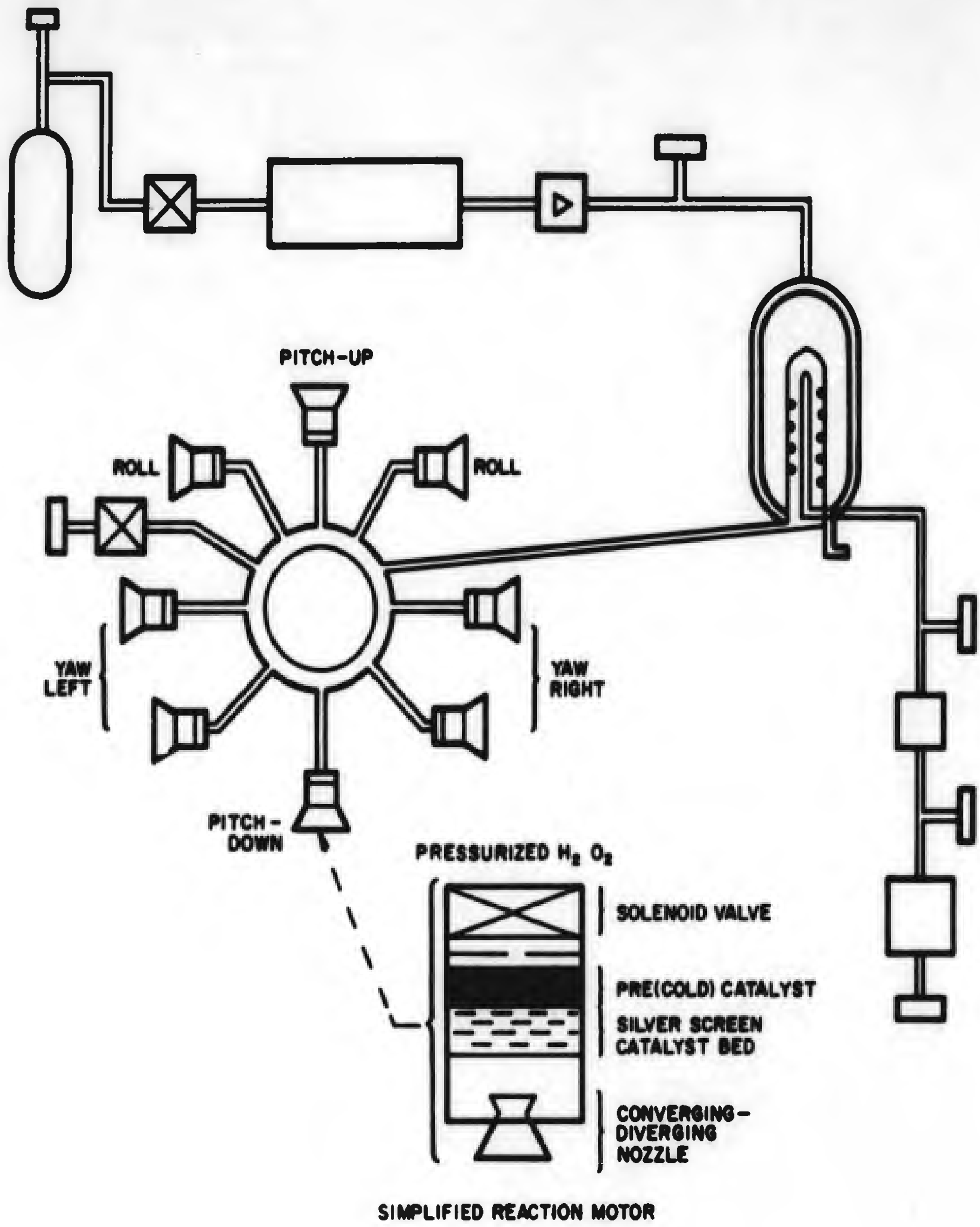


Figure 9. H_2O_2 Reaction Control System

Each reaction motor assembly consists of a normally closed solenoid valve, a catalyst bed to decompose the hydrogen peroxide, and a De Laval nozzle to expand the superheated steam. The catalyst bed consists of a pre- or cold catalyst pack and a normal silver screen catalyst bed. The pre-catalyst bed initiates decomposition at relatively low fuel temperatures, i.e., 40 degrees F. Because of the short active life of the pre-catalyst, the silver screen is necessary to sustain decomposition after the first few pulses on each motor.

The propellant, 90 percent hydrogen peroxide, is stored in aluminum tanks lined with a flexible bladder. Nitrogen pressure on the outside of the bladder is supplied by the pressurization system. The bladder serves as a positive expulsion service to prevent peroxide fumes from migrating into the nitrogen system. All reaction motors are connected to a common fuel manifold within each stage.

The pressurization system consists of commonly-manifolded gaseous nitrogen storage bottles, pressure regulator, normally closed squib valve, relief and charging valves, and vents. The nitrogen is expanded through a pressure regulator which insures the correct thrust output from the reaction jets. A relief valve set at approximately 575 psig protects the fuel system from over-pressurization in the event of regulator failure. A normally closed squib-actuated valve separates the pressure supply from the fuel system. Since the squib is not actuated until the last few minutes of the countdown, tower personnel are not subjected to the inherent dangers of a pressurized fuel system.

4.1.7 Telemetry System

The telemetry system consists of two RF links and two sets of subcarrier oscillators. The first is a standard 15-channel unit with two channels commutated, located at the forward end of the third stage motor. The second system is located in the payload area and is designed to meet payload requirements. Typically, a 15-channel unit, consisting of light weight components, is utilized.

4.1.8 Gyro Reference Unit

The gyro reference unit is a single-domed cylindrical package approximately 13 inches in diameter and weighing 19 pounds. The unit contains three single-degree-of-freedom, floated, rate integrating attitude gyros. The gyros are orthogonally mounted on a heater block and temperature is maintained within plus or minus five degrees F. A unique feature of the instrument is the wide input axis freedom; nominally plus or minus ten degrees. This freedom is required to prevent gyro bottoming or loss of attitude reference during transient conditions near the maximum control torque capability of the vehicle. A permanent magnet torque generator is employed for torque rates to 400 milliradians per second. A compensating winding in the dualsyn is provided to null out gravity-insensitive torques. Included in the package are the associated electronics to provide for gyro caging, trim balance, signal generator excitation, signal processing, and relay gain changing.

4.1.9 Pitch Attitude Programmer

The pitch attitude program is implemented by torquing the pitch gyro wheel. Constant torque results in a constant rate of the gyro wheel, hence, an attitude program consists of selected constant rates or straight line variations in reference attitude as a function of time.

The programmer is a precision voltage supply capable of providing eleven preselected individual levels corresponding to vehicle pitch attitude rates of from 0 to 10 degrees per second.

4.2 THE SLV-IA PAYLOAD

Payload is defined as all instrumentation, mounting structures, separation systems, and other hardware attached to the fourth stage motors. Exceptions are spin motors, rocket motors, nozzle skirt and (when used in the payload stage) guidance and control. There are three standard heatshield configurations available; 21-inch tapered, 25.7-inch, and 34-inch.

The payload compartment volume is dictated by the inside dimensions of the heatshield, with nominal clearances between the heatshield, motor, and payload. The heatshield length and width may be varied to meet specific mission requirements. Integration of payload requirements, including essential mechanical and electrical interface arrangements, should be coordinated through the Air Force Standard Launch Vehicle-I Office (SSVB). See Reference 3 for further information on the SLV-IA Vehicle and payload configurations.

SECTION V

SLV-I DEVELOPMENTAL PROPULSION

5.1 BACKGROUND AND DESCRIPTION

Seven solid propellant rocket motors are utilized in the SLV-I Program. The vehicle configurations are designated according to motor sequence employed, such as 1-2-3 or 2-3-5-6. The various rocket motors are as follows:

5.1.1 Motor No. 1 (Algol)

The Aerojet 40KS-120,000 is a conventional solid-propellant motor with steel case and nozzle, containing an aluminized polyurethane propellant. This motor is utilized as a first stage motor on the 1-2-3, 1-2-3-4, and 1-2-3-4-5 vehicle configurations.

The Algol was originally developed by Aerojet General Corporation as the Jupiter Senior rocket motor, in connection with the Jupiter program under Navy Contract NOrd 17012. The motor underwent a limited static test program which, though completely successful, was discontinued when the Jupiter program was supplanted by the Polaris program. Reliability of hardware was fully established; optimization of hardware weight, particularly the nozzle weight, was not initially attempted. An improved, optimized Algol (Algol IIA) was adopted as standard in late 1962 – it has a Jato designation of 68-KS-80,000.

5.1.1.1 Hardware

The Algol is 40 inches in diameter and approximately 358 inches long. The 0.109-inch, 4130 steel chamber wall was established as standard, while the 1020 steel nozzle was used without modification. The present Algol IIA uses a fiberglas nozzle.

5.1.1.2 Propellant and Core Configuration

An improved polyurethane propellant was selected for the Algol, since it has better physical properties, is easier to process, and produces a more neutral performance curve. The eight-point-star propellant grain configuration incorporated in the Algol is essentially the same as that originally used in the Jupiter Senior. The Algol IIA uses a cloverleaf grain configuration and is the current first-stage motor.

5.1.1.3 Static Tests and Nominal Performance

Four successful qualification static tests of limited duration were conducted at 90, 70, 50, and 30 degrees F. One result of this qualification test program was a slight design modification of the Algol C motors, which required changing the firing temperature limits to a range of from 50 to 90 degrees F. Some of the Algols were IB motors (see page 52 for motor configurations) which have a temperature range from 70 to 90 degrees F. Table 2 lists the nominal performance values.

Table 2

Algol-ID Nominal Sea Level Performance Values

Data Item	Nominal (a)	1-Sigma Deviation From Nominal* (a)	Percent of 1-Sigma Deviation From Nominal (b)	Manufacturers Prediction For Motor C-23 (a)
T	70	--	--	70
W_m	22,648	53	0.23	22,598 (c)
W_p	18,998	47	0.25	18,942 (c)
W_f	3,443	59	1.71	3,424 (c)(d)
t_b	36.06 (c)	0.65	1.80	NA
t_f	41.29 (c)	1.76	4.26	NA
$\int_{t_o}^{t_f} pdt$	16,980	450	2.65	NA
I_{sl}	4,077,800	17,700	0.43	4,059,588 (c)
I_v	4,274,395 (b)	18,500 (b)	0.43	4,256,473 (b)

(a) Supplied by Aerojet

(b) Determined by NASA

(c) Used for preflight trajectory calculations

(d) Used for postflight trajectory calculations

NA Not Available

* 1-sigma deviation is computed by the equation:

$1\sigma = \left(\frac{\Sigma D^2}{N}\right)^{1/2}$ where D^2 is the sum of the squares of the deviations from the nominal and N is the number of samples.

5.1.2 Motor No. 2 (Castor)

The Thiokol XM-33-E7 and XM-33-E5 are also of conventional design and utilize a polybutadiene-acrylic acid propellant (PBAA). The E5 and E7 motors differ only in nozzle configuration, depending on the use of the motoı as a first or second stage. The E5 motor is used as a second stage on the 1-2-3 and 1-2-3-4 vehicle configurations. The E7 motor is used as a first stage on the 2-3-5-6 vehicle configuration.

The SLV-I Castor XM33-E5, with a high performance polybutadiene acrylic acid propellant, was developed for the Scout and Mercury Little Joe programs. It is, in turn, an elongated XM-12 Sergeant booster.

5.1.2.1 Hardware

The Castor is 31 inches in diameter, approximately 244 inches long, and has a 0.1 inch wall thickness. The Sergeant 4130 steel case was used with no change in chamber shape or wall thickness. New developments are a 4130 steel nozzle and a new plastic cased pyrogen igniter utilizing a PBAA propellant.

5.1.2.2 Propellant and Core Configuration

The core, a five point star yielding a saddle-shaped pressure curve, was used with no modification. A PBAA propellant was developed to suit the Scout and Little Joe Castor requirements.

5.1.2.3 Static Tests and Nominal Performance

Qualification testing consisted of eleven static firings at temperatures of 20 to 100 degrees F. Nominal performance values are presented in Table 3.

Table 3

Castor XM33-E5 Nominal Vacuum Performance Values

Data Item	Nominal (a)	1-Sigma Deviation From Nominal (a)	Percent of 1-Sigma Deviation From Nominal (a)	Manufacturers Prediction for Motor 92 (b)
T	77	NA	--	77
W_m	8, 845	36	0. 41	8, 858 (c)
W_p	7, 320	26	0. 36	7, 294 (c)
W_f	1, 390	47	3. 38	1, 424 (c)(d)
t_b	27. 20	0. 51	1. 88	27. 69 (c)
t_f	39. 9	1. 1	2. 76	NA
$\int_{t_o}^{t_f} P\,dt$	15, 960	130	0. 81	NA
I_v	1, 957. 000	12, 940	0. 66	1, 941, 500 (c)*

(a) Determined by NASA

(b) Supplied by Thiokol

(c) Used for preflight trajectory calculations

(d) Used for postflight trajectory calculations

NA Not Available

* Adjusted by NASA

5.1.3 Motor No. 3 (Antares)

The ABL (Allegany Ballistics Laboratory) X-259 motor case is constructed of filament wound Fiberglas impregnated with an epoxy resin and utilizes a cast, aluminized, double-base propellant. The nozzle is compression molded phenolic-Fiberglas. The X-259 motor replaced the X-254 rocket motor used in earlier TS609A development flights. The motors are physically interchangeable but a higher energy propellant formulation is used to improve performance.

The Antares X254-A1, based on the X-248 design, was developed for use as the third stage propulsion system.

5.1.3.1 Hardware

The chamber is a filament-wound, glass fiber-reinforced epoxy resin structure incorporating integrally wound forward and aft adapters of high strength aluminum. The forward adapter serves as a resonance suppressor-igniter support and the aft adapter as a nozzle attachment fitting. The chamber is 30.050 inches in diameter and 76.1 inches long. The overall motor length, including nozzle, is 114.7 inches. The nominal wall thickness is 0.1 inch. The ends of the case are wound as ellipsoidal domes for maximum strength-to-weight ratio.

5.1.3.2 Propellant and Core Configuration

The Antares uses the cast double-base propellant previously developed for the fourth stage Altair motor. The propellant charge is a single-perforated, five-slot design having ellipsoid-shaped head and aft ends to conform to the chamber contours.

5.1.3.3 Static Tests and Nominal Performance

The Antares Qualification Test Program consisted of 20 static test firings with temperature operating limits of 50 to 100 degrees F. Nominal performance values for the X-254 are presented in Table 4. The X-259 has approximately 35 percent more total impulse.

Table 4

Antares X254-A1 Nominal Performance Values

Data Item	Nominal (a)	1-Sigma Deviation From Nominal (a)	Percent of 1-Sigma Deviation From Nominal (a)	Manufacturers Prediction for Motor No. F-1[illegible] (b)
T	70	NA	--	70
W_m	2,285	16	0.70	2,286 (c)
W_p	2,084	6	0.29	2,080 (c)
W_f	178	7	3.93	182 (c)(d)
t_b	36.80 (c)	0.96	2.61	NA
t_f	39.7 (c)	0.8	2.02	NA
$\int_{t_o}^{t_f} P\,dt$	11,686	51	0.44	NA
I_v	534,080	490	0.09	532,688 (c)

(a) Determined by NASA

(b) Supplied by ABL

(c) Used for preflight trajectory calculations

(d) Used for postflight trajectory calculations

NA Not Available

5.4.1 Motor No. 4 (Altair)

The ABL X-258 is of the same construction and utilizes the same fuel as the X-259. This motor replaces the X-248 motor which was utilized in earlier TS609A development flights.

The Altair X248 was developed for the Vanguard program and is used as the third stage of the LV-IB vehicle. The SLV-IA vehicle currently uses the X-258 as its standard fourth stage.

5.1.4.1 Hardware

The case is a filament-wound, glass fiber-reinforced epoxy resin structure 18.0 inches in diameter. The overall motor length is 59.1 inches. The case wall thickness is 0.055 inch. The ends are wound as hemispherical domes and Fiberglas shoulders (doublers) are wound in the forward and aft ends of the chamber.

5.1.4.2 Propellant and Core Configuration

The Altair has a cast double-base propellant developed for this motor in the Vanguard program. The charge is a single-perforated, eight-slot design having hemispheric-shaped forward and aft ends to conform to the chamber contours.

5.1.4.3 Static Tests and Nominal Performance

Four static quality control tests were performed and the motor was qualified from 50 to 100 degrees F. Nominal performance values are presented in Table 5.

Table 5

Altair X248-A5-S Nominal Performance Values

Data Item	Nominal (a)	1-Sigma Deviation From Nominal (a)	Percent of 1-Sigma Deviation From Nominal (a)	Manufacturers Prediction for Motor No. SV-118 (b)
T	70	--	--	70
W_m	515	1	0.19	515.5 (c)
W_p	456	1	0.22	456.25 (c)
W_f	50	1	2.0	51.05 (c)(d)
t_b	38.5 (c)	1.82	4.73	NA
t_f	41.4 (c)	1.82	4.40	NA
$\int_{t_o}^{t_f} Pdt$	8,975	135	1.50	NA
I_v	116,840	630	0.54	116,500 (c)

(a) Determined by NASA

(b) Supplied by ABL

(c) Used for preflight trajectory calculations

(d) Used for postflight trajectory calculations

NA Not Available

5.1.5 Motor No. 5 (Alcor Aerojet AJ10-41)

The Aerojet 30KS-8000 motor case is a steel-type wound structure bonded with epoxy resin. The motor has a phenolic-Fiberglas nozzle exit cone and contains an aluminized polyurethane propellant. This motor is utilized as the third stage motor in the 2-3-5-6 configuration. See Table 6.

5.1.6 Motor No. 6 (NOTS 17" Sphere)*

The NOTS (U.S. Naval Ordnance Test Station) Model 100A is a spherical rocket motor. The motor case is fabricated of stainless steel and employs a phenolic-lined internal nozzle. An aluminized polyurethane propellant is used. This motor is used as the fifth stage on the SLV-IA vehicles, as well as the fourth stage on SLV-IB.

5.1.7 Motor No. 7 (MG-18)*

This motor is manufactured by Lockheed Propulsion Company with one motor designation MG-18. No further information is available.

5.2 ROCKET MOTOR CONFIGURATIONS

Rocket motor designations and configurations are summarized in Table 1, page 5.

5.3 ROCKET MOTOR PERFORMANCE

Typical performance data for Algol, Castor, Antares, Altair, and Alcor motors are presented in Table 6.

*For motor parameters contact SSVB Scout Program Office.

TABLE 6

SLV-I MOTOR DATA (TYPICAL)

ALGOL

	ALGOL ID	ALGOL IIA
Total impulse, lb/sec, sea level	4,077,800	4,740,000
Specific impulse, lb/sec/lb, sea level	214.4	223
Total burning time, sec	41.3	68
Avg web thrust, lb, sea level	102,290	111,580
Total weight, lb	22,178	23,600
Fuel weight, lb	18,998	21,200
Mass ratio, W_p/W_t	.840	0.895
Nozzle expansion ratio	4.64	7.35
Weight consumed, lb	19,205	21,500

CASTOR

	CASTOR XM-33-E7	CASTOR XM-33-E5
Total impulse. lb/sec, vacuum	1,635,000	2,004,000
Specific impulse, lb/sec/lb, vacuum		267.4
Total burning time, sec	27.39	40.27
Avg web thrust, lb. vacuum	53,800	64,340
Total weight. lb		8,867
Fuel weight, lb		7.313
Mass ratio, W_p/W_t		.825
Nozzle expansion ratio		15.8
Weight consumed, lb		7,427

Reference: "The Scout Solid Propellant Launch Vehicles", Chance Vought Corp. Scout Manual, dated October 1962 (U).

Table 6 (Continued)

ANTARES

	Antares X-254-A1	Antares X-259-A2
Total impulse, lb/sec, vacuum	720,000	719.900
Specific impulse, lb/sec/lb, vacuum	281	281.2
Total burning time, sec		33.2
Avg web thrust, lb, vacuum		23,760
Total weight, lb	2,785	2,785
Fuel weight, lb	2,562	2,562
Mass ratio, W_p/W_t	0.920	0.912
Nozzle expansion ratio	17.5	17.93
Weight consumed, lb	2,585	2,587

ALTAIR

	Altair X-258	Altair X-248-A5
Total impulse, lb/sec, vacuum	143,000	116,500
Specific impulse, lb/sec/lb, vacuum	277	256.0
Total burning time, sec	23.5	41.4
Avg web thrust, lb, vacuum		3000
Total weight, lb	578.1	513
Fuel weight, lb	516,0	456
Mass ratio, W_p/W_t	0.906	0.89
Nozzle expansion ratio	18.0	25.8
Weight consumed, lb	516.1	464

Table 6 (Continued)

ALCOR

	Alcor AJ10-41
Total impulse, lb/sec, vacuum	242,540
Specific impulse, lb/sec/lb, vacuum	275.3
Total burning time, sec	30.9
Avg web thrust, lb, vacuum	7,849
Total weight, lb	970.0
Fuel weight, lb	875.0
Mass ratio, W_p/W_t	0.902
Nozzle expansion ratio	21.9
Weight consumed, lb	895

PART 2

SLV-I OPERATION SUMMARY

1.1 INTRODUCTION

The purpose of this section is to present factual summaries of pertinent operation and failure information contained in the Flight Test Evaluation Reports published by the responsible agencies; Aeronutronic for USAF (during the TS609A program), and Vought Astronautics for NASA and the USAF operational program. A brief summary of SLV-I operation and flight history is included for reference (Table 7, page 50).

Anomaly summaries are presented to provide a data source for evaluation and correlation purposes. It should be noted, however, that the SLV-I is a "family" of different vehicles, with primary mission objectives dictating which configuration of the SLV-I family was used (See Table 8, page 52, for motor configuration). Therefore, discretion should be used when making evaluations. However, an overall success to total trial ratio graph, with the different vehicle configurations launched, is presented in Figure 10, page 54. Failure data on the motors used is included in Table 7.

Table 7. SLV-I Operation and Flight History to 20 July 1963

LAUNCH NO.	LAUNCH DATE	LAUNCH LOC.	MISSILE S/N	OLD DESIGNATION	NEW DESIGNATION	MOTOR CONFIGURATION	SCOUT MISSION RESULT S OR F	CUMULATIVE RATIO	20-MISSILE MOVING RATIO	REMARKS	MOTORS XM68	XM75	XM70	XM87	XM79	XM78	MO18
0	4-18-60	WI	SX-1*	NASA	SLV-IA	1-2-3	F										
1	7-1-60	WI	ST-1	NASA	SLV-IA	1-2-3-4-S	S	1.00		4th Stage disabled by Safety Officer due to false radar tracking interpretation.	○	○	○	△			
2	9-21-60	AMR	D-1	JR.	SLV-IB	2-3-5-6	S	1.00		Vehicle flight satisfactory, but payload telemetry failed at approximately 100 n mi altitude.		○	○		○	○	
3	10-4-60	WI	ST-2	NASA	SLV-IA	1-2-3-4-S	S	1.00		Flight successful.	○	○	○	○			
4	11-8-60	AMR	D-2	JR.	SLV-IB	2-3-5-6	F	0.75		2nd Stage failure caused by improperly installed motor nose plug.		○	□		△	△	
5	12-4-60	WI	ST-3	NASA	SLV-IA	1-2-3-4-S	F	0.60		2nd Stage ignition failure.	○	□	△	△			
6	1-7-61	AMR	D-3	BSI		1-2-3	S	0.67		Flight successful.	○	○	○				
7	2-16-61	WI	ST-4	NASA	SLV-IA	1-2-3-4-S	S	0.72		Flight successful. Explorer IX put in orbit	○	○	○	○			
8	3-3-61	AMR	D-4	BSII		1-2-3-4	S	0.75		Flight successful.	○	○	○	○			
9	4-12-61	AMR	D-5	BSII		1-2-3-4	S	0.78		Flight successful.	○	○	○	○			
10	5-9-61	AMR	D-6	BSI		1-2-3	F	0.70		Loss of 2nd stage yaw control attributed to faulty wiring in electrical harness.	○	□	△				
11	6-30-61	WI	ST-5	NASA	SLV-IA	1-2-3-4-S	F	0.64		Nozzle plug forced out due to improper venting, breaking 3rd stage ignition circuit leads.	○	○	□	△			
12	8-17-61	AMR	0-1	JR.	SLV-IB	2-3-5-6	S	0.67		Vehicle flight successful, but payload telemetry lost as on D-1. Telemetry system subsequently redesigned by AFSWC.		○	○		○	○	
13	8-25-61	WI	ST-6	NASA	SLV-IA	1-2-3-4-S	S	0.69		Explorer XIII put in orbit for 2 1/2 days. Unsatisfactory orbit achieved because of 4th stage tipoff (approx. 11 deg.).	○	○	○	○			
14	10-19-61	WI	ST-7	NASA	SLV-IA	1-2-3-4-S	S	0.72		4th Stage tipoff again observed (at least 7 deg.).	○	○	○	○			
15	11-1-61	AMR	D-8	AES*	SLV-IA	1-2-3-4-S	F	0.67		Control instability caused by improperly connected gyro leads. Vehicle auto-destructed after approximately 30 sec of flight.	□	△	△	△			
16	12-4-61	PMR	0-2	JR.	SLV-IB	2-3-5-6	S	0.69		4th Stage tipoff depressed trajectory; telemetry lost.		○	○		○	□	
17	3-1-62	WI	ST-8**	NASA	SLV-IA	1-2-3-4-5-S	S	0.71		Flight successful. Guided vehicle tipoff modifications proven successful.	○	○	○	○		○	
18	3-29-62	WI	ST-9	NASA	SLV-IA	1-2-3-4-S	S	0.72		Flight successful. Tipoff modifications again effective.	○	○	○	○			
19	4-12-62	AMR	D-7	BSI		1-2-3	F	0.68		Pressure decay switch circuit malfunction prevented 2nd stage ignition.	○	□	△				
20	4-26-62	PMR	111	AF/NASA	SLV-IA	1-2-3-4-S	F	0.65	0.65	Control lost during 3rd stage burn (peroxide fueling malfunction).	○	○	□	△			

Table 7. SLV-I Operation and Flight History to 20 Julv 1963 (continued)

LAUNCH NO.	LAUNCH DATE	LAUNCH LOC.	MISSILE S/N	OLD DESIGNATION	NEW DESIGNATION	MOTOR CONFIGURATION	SCOUT MISSION RESULT S OR F	CUMULATIVE RATIO	20-MISSILE MOVING RATIO	REMARKS	MOTORS XM68	XM75	XM70	XM87	XM79	XM78	MG18
21	5-23-62	PMR	112	AF/NASA	SLV-IA	1-2-3-4-S	F	0.619	0.60	Premature separation of 3rd stage resulted in destruction of vehicle.	○	□	△				△
22	5-31-62	PMR	102	JR.	LV-IB	2-3-4	S	0.636	0.60	Flight successful.		○	○	○			
23	7-24-62	PMR	101	JR.	LV-IB	2-3-4	S	0.652	0.60	Flight successtul.		○	○	○			
24	8-22-62	PMR	117	AF	SLV-IA	1-2-3-4-S	S	0.662	0.65	Flight successful, close to nominal orbit achieved.	○	○	○				○
25	8-31-62	WI	114**	NASA	SLV-IA	1-2-3-4-5-S	F	0.640	0.65	3rd stage ignited 51 seconds late. 5th stage thrusted for 27 instead of 43 seconds.	○	○	□	○		□	
26	11-21-62	PMR	201	AF	LV-IB	2-3-4	S	0.653	0.65	Flight successful.		○	○	○			
27	12-16-62	WI	115	AF	SLV-IA	1-2-3-4-S	S	0.666	0.65	Flight successful. Explorer XVI put in orbit.	○	○	○	○			
28	12-18-62	PMR	211	AF	SLV-IB/C	2-3-5	S	0.678	0.65	Flight successful. Close to nominal orbit achieved.	○	○	○	○			
29	12-18-62	PMR	118	AF	SLV-IA	1-2-3-4-S	S	0.689	0.65	Flight successful.		○	○		○		
30	2-1-63	PMR	202	AF	LV-IB	2-3-4	S	0.700	0.70	Flight successful.		○	○	○			
31	2-19-63	PMR	126	AF	SLV-IA	1-2-3-4-S	S	0.709	0.75	Flight successful.	○	○	○				○
32	3-13-63	PMR	203	AF	LV-IB	2-3-4	S	0.718	0.75	Flight successful.		○	○	○			
33	4-5-63	PMR	119	AF	SLV-IA	1-2-3-4-S	F	0.696	0.70	3rd stage H_2O_2 malf.	○	○	□	△			
34	4-25-63	PMR	121	AF	SLV-IA	1-2-3-4-S	F	0.676	0.65	3rd stage failure - unknown.	○	○	□				△
35	5-17-63	PMR	204	AF	LV-IB	2-3-4	S	0.685	0.70	Flight successful.		○	○	○			
36	5-22-63	WI	116***	NASA	SLV-IA	1-2-3-4-S	S	0.694	0.70	Flight successful.	○	○	○	○			
37	6-15-63	WI	120	NASA	SLV-IA	1-2-3-4-S	S	0.702	0.70	Flight successful.	○	○	○	○			
38	6-28-63	WI	113	NASA	SLV-IA	1-2-3-4-S	S	0.710	0.70	Flight successful.	○	○	○	○			
39	7-20-63	WI	110**	NASA	SLV-IA	1-2-3-4-S	F	0.695	0.70	First stage motor nozzle failure at approx. T + 4 sec due to a possible defect between the plies of the fiberglass forward insert or the gap between the insert and the graphite throat which eroded and allowed the flame to cut through the steel closure and undercut adjacent structures.	□	△	△	△			

*The SX-1 Vehicle was used for a research and development launch for experimental feasibility purposes only. This launch is not included in the success-to-total-trial ratio graph.

**Fifth Stage Vehicle - 5th Stage Note 17-inch spherical motor XM-85 (old XM-78).

***Used new XM93(ABL X259) 3rd stage motor

****XM85 motor replaced XM78 motor

○ Stage satisfactory

□ Malfunction

△ Not tested

Table 8. SLV-I Flight Motor Configurations

Note: 1. All "ST" Launches - Wallops Island
2. All "O" Launches - PMR
3. 111 and 112 - PMR D-1 thru D-8 AMR
4. 5 Stage Vehicle - 5th Stage Nots 17 in. Spherical Motor (XM-78)

Launch No.	Launch Date	Missile No.	AF And/Or NASA Vehicle	XRM No.	First Stage Motor Designations				Second Stage Motor Designations				Third Stage Motor Designations				Fourth Stage Motor Designations			
					NASA	AF	Manuf.	Jato	NASA	AF	Manf.	Jato	NASA	AF	Manf.	Jato	NASA	AF	Manf.	Jato
1	7-1-60	ST-1	NASA SLV-1A	92	Algol 1-B	XM-68	Aerojet Senior 1	33-KS-120,000	Castor E5	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares IA1	XM-91	Abl X-254 A1	38-DS-14000	Altair IA5S	XM-69	Abl X-248 A5	38-DS-3100
2	9-21-60	D-1	Jr. SLV-1B	91	Castor	XM-82	Thiokol XM-33-E7	27-KS-55,000	3 Antares	XM-70	Abl X-254A2	38-DS-14000	Alcor	XM-79	Aerojet AJ10-41	30-KS-8000	Sphere	XM-78	NOTS 100A	40-KS-900
3	10-4-60	ST-2	NASA SLV-1A	92	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS 14000	Altair	XM-69	Abl X-248 A6	38-DS-3100
4	11-8-60	D-2	Jr. SLV-1B	91	Castor IE-7	XM-82	Thiokol XM-33-E7	27-KS-55,000	Antares	XM-70	Abl X-254 A2	38-DS-14000	Alcor	XM-79	Aerojet AJ10-41	30-KS-8000	Sphere	XM-78	NOTS 100A	40-KS-900
5	12-4-60	ST-3	NASA SLV-1A	92	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000	Altair	XM-69	Abl X-248 A5	38-DS-3100
6	1-7-61	D-3	BSI	89	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000				
7	2-16-61	ST-4	NASA SLV-1A	92	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000	Altair	XM-69	Abl X-248 A5	38-DS-3100
8	3-3-61	D-4	BSII	90	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000	Altair	XM-69	Abl X-248 A5	38-DS-3100
9	4-12-61	D-5	II	90	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000	Altair	XM-69	Abl X-248 A5	38-DS-3100
10	5-9-61	D-6	I	89	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-93	Abl X-259 A1	38-KS 18550				
11	6-30-61	ST-5	NASA SLV-1A	92	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000	Altair	XM-69	Abl X-248 A5	38-DS-3100
12	8-17-61	O-1	Jr. SLV-1B	91	Castor IE-7	XM-82	Thiokol XM-33-E7	27-KS-55,000	Antares	XM-91	Abl X-254 A1	38-KS-14000	Alcor	XM-79	Aerojet AJ10-41	30-KS-8000	Sphere	XM-85	NOTS	40-KS-900
13	8-25-61	ST-6	NASA SLV-1A	92	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000	Altair	XM-69	Abl X-248 A5	38-DS-3100
14	10-19-61	ST-7	NASA SLV-1A	92	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000	Altair	XM-69	Abl X-248 A5	38-DS-3100
15	11-1-61	D-8	AES SLV-SIA	92	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS 14000	Altair	XM-69	Abl X-248 A5S	38-DS-3100
16	12-4-61	O-2	Jr. SLV-1B	91	Castor IE-7	XM-82	Thiokol XM-33-E7	27-KS-55,000	Antares	XM-76	Abl X-254 A1	38-DS-14000	Alcor	XM-79	Aerojet AJ10-41	30-KS-8000	Sphere	XM-85	NOTS	40-KS-900
17	3-1-62	ST-8*	NASA SLV-1A	92	Algol 1 C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000	Altair	XM-69	Abl X-248 A5	38-DS-3100
18	3-29-62	ST-9	NASA SLV-1A	92	Algol 1-D	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-93	Abl X-259 A1	38-KS-18550	Altair	XM-69	Abl X-248 A5	38-DS-3100
19	4-12-62	D-7	BSI	9	Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-91	Abl X-254 A1	38-DS-14000				
20	4-26-62	111	AFNASA SLV-1A	92	Algol 1-D	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-93	Abl X-259 A1	38-KS-18550	Altair	XM-69	Abl X-248 A5	38-DS-3100

Table 8. SLV-I Flight Motor Configurations (continued)

Launch No.	Launch Date	Missile No.	AF And/Or NASA Vehicle	XRM No.	First Stage Motor Designations: NASA	First Stage: AF	First Stage: Manuf.	First Stage: Jato	Second Stage Motor Designations: NASA	Second Stage: AF	Second Stage: Manf.	Second Stage: Jato	Third Stage Motor Designations: NASA	Third Stage: AF	Third Stage: Manf.	Third Stage: Jato	Fourth Stage Motor Designations: NASA	Fourth Stage: AF	Fourth Stage: Manf.	Fourth Stage: Jato
21	5-23-62	112	AFNASA SLV-1A	92	Algol 1-D	XM-66	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55000	Antares	XM-93	Abl X-259 A1	38-KS-18550	MG-18		Lockheed	
22	5-31-62	102	Jr. LV-1B	91	Castor IE-7	XM-82	Thiokol XM-33-E7	27-KS-55,000	Antares	XM-91	Abl X-254 A1	38-DS 14000	Altair	XM-87	Abl X-248 A5	38-DS-3100				
23	7-24-62	101	Jr. LV-1B	91	Castor IE-7	XM-82	Thiokol XM-33-E7	27-KS-55,000	Antares	XM-91	Abl X-254 A1	38-DS-14000	Altair	XM-87	Abl X-248 A5	38-DS-3100				
24	8-22-62	117	AF SLV-1A	92	Algol 1-D	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259 A2	38-KS-18,550	MG-18	None	None	
25	8-31-62	114[4]	NASA SLV-1A	92	Algol II-A	None	Aerojet Senior 1	33-KS-111,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259 A2	38-KS-18,550	Altair	XM-69	Abl X-248 A5	38-DS-3100
26	11-21-62	201	LV-1B		Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares	XM-70	Abl X-254-A1	38-DS-14,000	Altair	XM-69	Abl X-248-A5	38-DS-3100				
27	12-16-62	115	SLV-1A		Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	Altair	XM-69	Abl X-248-A5	38-DS-3100
28	12-18-62	211	SLV-1B/C		Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	Alcor	XM-79	Aerojet AJ10-41	30-KS-8000				
29	12-18-62	118	SLV-1A		Algol 1-C	XM-68	Aerojet Senior 1	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	Altair	XM-69	Abl X-248-A5	38-DS-3100
30	2-1-63	202	LV-1B		Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares	XM-70	Abl X-254-A1	38-DS-14,000	Altair	XM-87	Abl X-248-A5	38-DS-3100				
31	2-19-63	126	SLV-1A		Algol II-A	None	Aerojet Senior	33-KS-111,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	MG-18			
32	3-13-63	203	LV-1B		Castor	XM-82	Thiokol XM-33-E7	27-KS-55,000	Antares	XM-91	Abl X-254-A1	38-DS-14,000	Altair	XM-87	Abl X-248-A5	39-DS-3100				
33	4-5-63	119	SLV-1A		Algol II-A	None	Aerojet Senior	33-KS-111,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	Altair	XM-69	Abl X-248-A5	38-DS-3100
34	4-25-63	121	SLV-1A		Algol 1-D	XM-68	Aerojet Senior	33-KS-120,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	MG-18			
35	5-17-63	204	LV-1B		Castor	XM-82	Thiokol XM-33-E7	27-KS-55,000	Antares	XM-91	Abl X-254-A1	38-DS-14,000	Altair	XM-87	Abl X-248-A5	38-DS-3100				
36	5-22-63	116	SLV-1A		Algol II-A	None	Aerojet Senior	33-KS-111,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	Altair	XM-69	Abl X-248-A5	38-DS-3100
37	6-15-63	120	SLV-1A		Algol II-A	None	Aerojet Senior	33-KS-111,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	Altair	XM-69	Abl X-248-A5	38-DS-3100
38	6-28-63	113	SLV-1A		Algol II-A	None	Aerojet Senior	33-KS-111,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	Altair	XM-69	Abl X-248-A5	38-DS-3100
39	7-20-63	110[4]	SLV-1A	A	Algol II-A	None	Aerojet Senior	33-KS-111,000	Castor	XM-75	Thiokol XM-33-E5	27-KS-55,000	Antares II-A2	XM-93	Abl X-259-A2	38-KS-18,550	Altair	XM-69	Abl X-248-A5	38-DS-3100

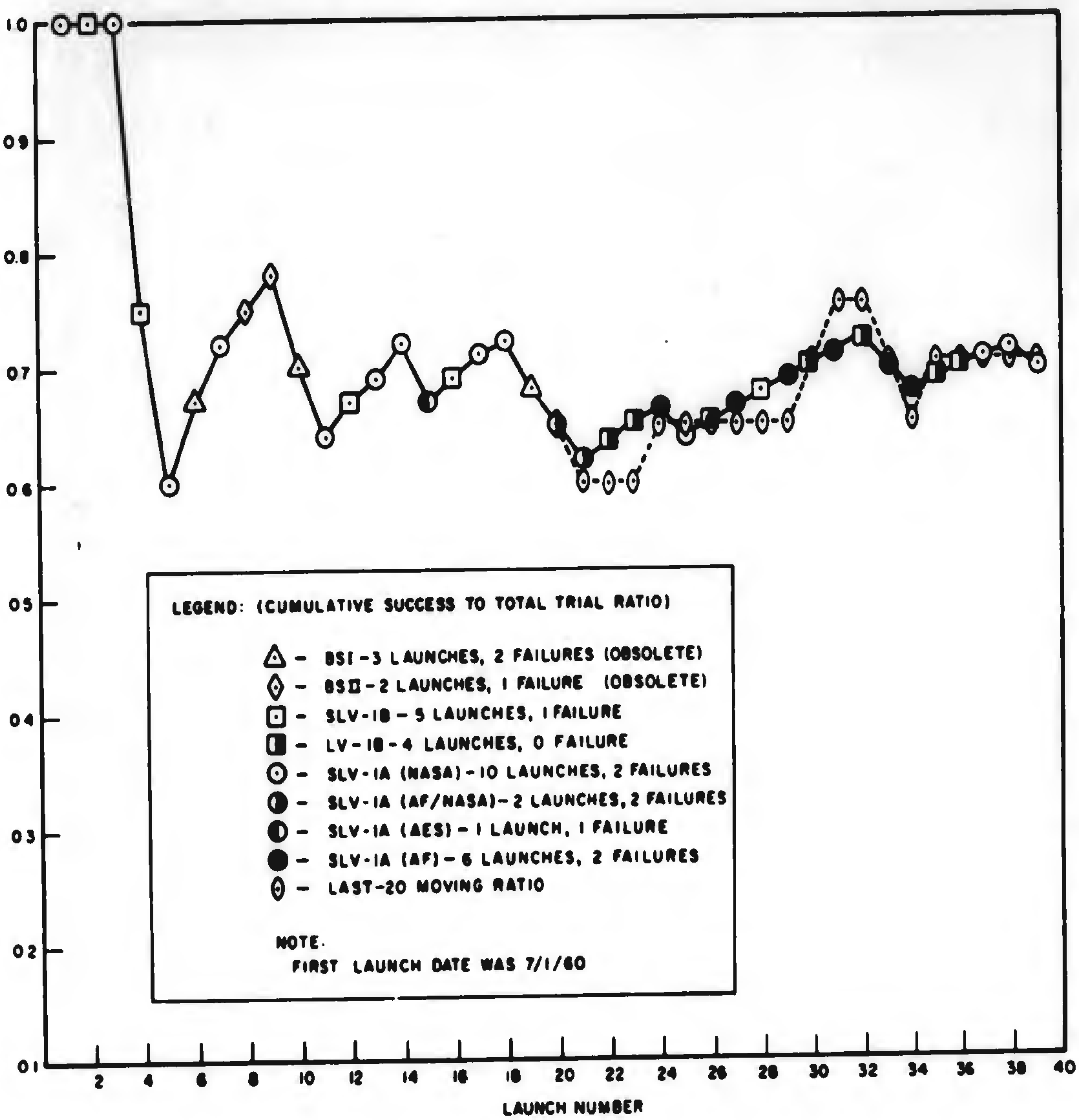


Figure 10. SLV-I Success Ratio vs Launch Number

Project: R and D

Launch Date: 18 April 1960

Launch Area: Wallops Island

Countdown Holds: 0

Launch No.: 0

Missile Type: SLV- 1A

Serial No.: SX-1

Result: Failure

PRIMARY OBJECTIVE

The primary objective of this first firing was to obtain flight experience with critical hardware items at the earliest possible date. This test system differed from the standard SLV-1A system in that it was unguided and spinning, the second stage Castor motor contained ballast instead of propellant, the third stage Altair motor was replaced by a steel nose cap of weight equal to this stage, and the fin tip controls and jet vanes were fixed at an angle of 8 degrees.

MISSION SUMMARY

The objective was not achieved due to two structural failures. At approximately T+3 seconds, an object fell from the base region of the first stage. This object was approximately the size of a fin tip control or jet vane, but positive identification was not possible because of obscurity from the exhaust of the first stage motor. At T+16 or 17 seconds the heat shield around the third stage motor separated and disassembled. This heat shield was supposed to remain on the vehicle for the duration of flight. At approximately T+38 seconds, a structural failure occurred which allowed a forward portion of the vehicle to separate (possibly transition section C).

PERFORMANCE

Heat Shield

Breakup of the third stage heat shield was due to excessive pressure differential. The heat shield was designed to withstand a pressure differential of 2 psi, but tests made after the flight indicated that approximately 4 psi differential existed at the time of heat shield failure. This failure appears to be a transonic problem as it occurred in a speed range between Mach 0.9 to 1.0. When the heat shield was torn off, the ignition circuitry for the third stage was destroyed by the supersonic free-stream flow.

Structure

At approximately T+38 seconds the vehicle was spinning at a rate of 220 rpm (about 28 percent higher than nominal). At the same time, the amplitude of the induced vibrations increased and oscillated between +20 g and -30 g during and after the vehicle structural failure. Calculations of vehicle spin velocity showed that the first-body bending mode occurred at this speed. It appears that the rolling frequency, coupled with this first-body bending mode, caused a section of the vehicle to be overstressed to the point of failure. The weakest structural link of the system was at transition section C, or about 122 inches from missile Station 0.

Film data indicated that a 10-foot object (about the size of the section of the vehicle forward of transition section C) came off the vehicle at the time of failure. This also indicates that the hardware component which failed could have been the transition section C.

Trajectory

At T+37 seconds the test vehicle flight trajectory had a large dispersion in the azimuth plane. The wind-compensated flight path differed from the nominal by 1.5 degrees and from the flight test path by 12.5 degrees. The large dispersion between the flight test and wind-compensated flight path could be due to unaccounted-for thrust misalignment of the first stage motor, surface gusts, angular velocity input from launcher base support pins, or a combination of these influences. Sufficient data are not available to determine a probable explanation of this dispersion. The fact that the vehicle was launched at a large elevation angle (74 degrees elevation setting) would tend to magnify a small lateral tipoff in the azimuth plane (104 degrees azimuth setting). For example, at an elevation angle of 81 degrees a 1 degree lateral tipoff would be represented by approximately 9 degrees dispersion in the azimuth plane.

REFERENCE

"Scout SX-1 Flight Test Results," NASA, 15 September 1960, Unclassified.

Project: Space Probe	Launch No.: 1
Launch Date: 1 July 1960	Missile Type: SLV-1A
Launch Area: Wallops Island	Serial No.: ST-1
Countdown Holds: 0	Result: Success

PRIMARY OBJECTIVES

Establish a space probe mission that would permit radar tracking and telemetry acquisition through fourth stage burnout.

MISSION SUMMARY

Most test objectives were achieved, although erroneous radar tracking resulted in prevention of fourth stage motor ignition by the range safety officer.

Guidance accuracy for the flight was determined from a comparison of measured and predicted trajectories. The comparison indicated the actual flight path angle was about 1.5 degrees higher than predicted, and the angular difference in azimuth track was 0.8 degree. These differences are within control system design specifications. Flight simulation studies have shown that part of the difference can be attributed to variations in motor performance, thrust misalignment and winds, especially during first stage burning.

PERFORMANCE

Propulsion

All motors operated within in-flight tolerances.

Guidance and Control

First and second stage controls functioned normally. The third stage functioned normally except for overpowering of the roll jets by an unexpected rolling moment disturbance near third stage burnout. Although the control system regained command of the vehicle at a new roll reference, the hold-fire signal, which could not be countermanded, had been given at T+151.3 seconds and the fourth stage motor did not ignite.

Large vibration amplitudes coincided with the large roll disturbance near third stage motor burnout. These vibrations caused an acceleration switch to chatter and resulted in a constant switching in and out of the high and low reaction-jet controls during third stage burning.

Airframe

Except for premature loss of the third stage heat shield as the vehicle entered the transonic speed range, structural integrity of the vehicle was demonstrated. At approximately T+16 seconds, the third stage heat shield detached prematurely, due to high pressure loads overcoming the yield load of the heat shield latching mechanism during transonic speed. Wind tunnel data indicated high negative pressure over the forward end of the heat shield at subsonic speeds. The pressure reached maximum at Mach 0.9. There was, in effect, no venting of the inside of the heat shield, and the heat shield latching mechanism was subjected to loads arising from the low-pressure region over the forward end of the heat shield. Corrective action consisted of drilling six equally spaced 0.250-inch diameter holes in the heat shield. The vent area of these holes was sufficient to maintain the pressure inside the heat shield at less than 0.1 psi above the outside pressure.

Telemetry

Data acquisition was satisfactory except for sidelobe tracking from launch to near third stage burnout. Due to the fourth stage not spinning up (RSO hold-fire signal at T+151.3 seconds) data were not obtained on the operation of the solar aspect system (channel 14).

REFERENCE

"NASA Scout ST-1 Flight Test Results," Langley Research Center, NASA TN D-1240, Unclassified.

Project: Space Probe

Launch Date: 21 Sept 1960

Launch Area: AMR-18

Countdown Holds: 0

Launch No.: 2

Missile Type: SLV-1B

Serial No.: D-1

Result: Success

PRIMARY OBJECTIVES

1. Prove the capability of the 2-3-5-6 booster configuration to place small instrument payloads into high altitude probe trajectories.
2. Obtain telemetered experimental data from the instrument payload during flight.

MISSION SUMMARY

Flight test proving of the booster configuration was successful. In-flight experimental data were not received, due to telemetry system failure at T+151 seconds.

PERFORMANCE

Propulsion

The vehicle launcher was set to provide an effective launch azimuth of 105 degrees and an effective launch elevation of 70 degrees. The vehicle was aerodynamically and spin stabilized along a gravity-turn trajectory during boost of the first two stages. It was also spin stabilized along a constant-attitude trajectory during third and fourth stage boost, and during coast to apogee.

All motors fired and the durations of thrust for the first three motors were within tolerances. Burning time of the fourth stage motor cannot be defined, due to loss of telemetry.

The following changes were made on subsequent SLV-1B vehicles:

1. Zero-wind launcher settings at AMR will be 69.2 degrees elevation and 107 degrees azimuth, to provide an effective ground track of 105 degrees.
2. First stage fin tip settings were increased to 6.0 degrees and second stage fins were reset to 0.35 degree to increase roll rate.

Airframe

There was an apparent malfunction of the nose cone separation mechanism. Normally, an ignition signal is given simultaneously to the nose cone retaining strap cutter, the nose cone pin puller, and the third stage motor igniter. The strap cutter releases a compressed spring that sends the nose cone forward, and 0.5 second later the pin puller releases another spring, sending the nose cone halves away from the centerline. One second later, the third stage motor ignites and accelerates forward.

An unexpected impulse, indicated by the lateral accelerometers at approximately the time for the payload to overtake the separated nose cone, led to the conclusion that one antenna on the payload struck the nose cone. Apparently the nose cone halves did not separate. Possible causes of pin puller failure, if it occurred, are lack of electrical signal due to broken wires, or inadequate forces generated by the device.

A new nose cone segment-separation mechanism was employed on subsequent SLV-1B vehicles.

Telemetry

Thirty-two of the 38 programed channels provided useful data. The malfunctions were: three breakwire channels erroneously indicated events, command destruct current was erroneously indicated, the X-254 heat shield temperature signal was lost at T+35 seconds, and one lateral accelerometer appeared to be coupled with the longitudinal accelerometer.

Abrupt loss of the telemetered signal occurred at T+151 seconds, 8 seconds prior to normal fourth stage burnout. Signal loss was probably due to soft solder melting in several pull-away plugs located near the nozzle of the motor, causing a short circuit in the telemetry power supply. On subsequent SLV-1B vehicles, telemetry power leads are opened by a relay in the payload area at third stage ignition.

REFERENCE

"Blue Scout Jr. Flight Test Report D-1," Aeronutronic, 21 November 1960.

Project: Space Probe

Launch Date: 4 Oct 1960

Launch Area: Wallops Island

Countdown Holds: 0

Launch No.: 3

Missile Type: SLV-1A

Serial No.: ST-2

Result: Success

PRIMARY OBJECTIVES

1. Prove the overall capability and reliability of the SLV-1 system on a probe-type mission.
2. Measure environmental conditions and performance characteristics.
3. Gain additional experience in preflight preparation and launching of the vehicle.
4. Evaluate modifications made following the ST-1 launch. These were:
   a) The yoke and lanyard method of latching and jettisoning the third stage Antares heat shield was replaced by piano hinges and ballistic actuators.
   b) Larger roll motors (2.2 pounds to 14 pounds) were used.

MISSION SUMMARY

All flight objectives were accomplished.

A delay in the start of the timer, similar to the ST-1 flight (launch No. 1) but of somewhat longer duration (0.43 sec), was observed at lift-off. Again this was attributed to the lanyard which pulls the flyaway umbilical and actuates the timer. Approximately 33 seconds after lift-off, a false signal indication was received that the third stage heat shield was jettisoned. Prior to launch, a spacer was installed between the heat shield and jettison-indication switch in order to depress the switch plunger. It is possible that heat deformation caused sufficient movement of the spacer to allow the switch to open and give a false signal indication of heat shield removal. The jettison indicator switch was replaced by a breakwire on subsequent vehicles. A rolling moment again occurred, as on the ST-1 flight, but the larger roll motors, installed for this flight, were sufficient to overcome the disturbance.

PERFORMANCE

Propulsion

Motor thrust could not be determined from acceleration data because the longitudinal accelerometers were improperly mounted. Pressure and velocity were, therefore, the criteria used for motor performance analysis.

Overall performance was less than 0.2 percent higher than preflight predicted velocity. Third stage experienced a sharp pitch-up motion at time of ignition. At 12 seconds before tail-off, a roll left (ccw) disturbance of 1-ft/lb was experienced, followed in rapid succession by roll right (cw), roll left, and roll right disturbances. Magnitudes of 70 and 40 ft/lb (0.05 sec pulses) were measured in the roll right direction and 30 ft/lb (0.05 sec pulse) in the roll left direction. This disturbance was slightly earlier than on the ST-1 flight and was characterized by reversals, whereas the ST-1 flight disturbance remained continually in the roll right direction.

Resonant burning, or unstable combustion of the third stage motor (creating high-frequency high-amplitude pressure oscillation about the mean chamber pressure), was still present on this flight.

Guidance and Control

One roll jet operated out of sequence due to an electrical unbalance in the poppet valve electronics (relay amplifier), resulting in excessive compensation for pitch and yaw jet flow angularity in combination with radial center of gravity offset.

Telemetry

One channel in the base A section, which measured compartment temperature, was broken during erection of the vehicle and was not replaced prior to launch. All other channels operated properly and tracked for 400 seconds. The payload FM/FM telemetry had one unusable longitudinal accelerometer channel (incorrect installation). The 584 S-band radar became erratic at third stage firing and was lost at 121 seconds. The Mod II radar experienced trouble in "locking on" the beacon and was lost at 123 seconds.

Temperature

The thermocouple located on the skin at transition section C did not function during flight.

REFERENCE

"Preliminary Flight Test Results of Scout ST-2," Langley Research Center, 16 December 1960, Unclassified.

Project: Probe

Launch Date: 8 Nov 1960

Launch Area: AMR Pad 18A

Countdown Holds: 0

Launch No.: 4

Missile Type: SLV-1B

Serial No.: D-2

Result: Failure

PRIMARY OBJECTIVES

1. Prove the capability of the 2-3-5-6 booster configuration to place a small instrument payload into a high-altitude probe trajectory.
2. Obtain telemetered data from the instrument payload during flight.

MISSION SUMMARY

Objectives were not accomplished. Second stage motor thrust was lower than nominal from ignition at T+36.46 seconds until it exploded at T+62.9 seconds. One spin motor clamp did not eject as planned. Telemetry signals from five channels were completely lost at approximately T+52 seconds. Inner skin temperatures in the section near the front of the second stage motor increased sharply after second stage motor ignition.

Following thrust deterioration, the second stage was tracked to near impact, which was approximately 256 nautical miles downrange from the launcher at T+609 seconds. The postflight predicted impact target was $27^{\circ}32.42'$ north latitude and $75^{\circ}54.91'$ west longitude.

PERFORMANCE

Propulsion

Prior to second stage ignition the temperatures recorded were approximately $100^{\circ}F$ at Station 100, $60^{\circ}F$ at Station 120, and $80^{\circ}F$ at Station 140. From second stage ignition (T+36.46 seconds) until loss of usable data at T+48 seconds, a $15^{\circ}F/sec$ rate of temperature increase was indicated at the missile stations noted above.

Second stage velocity at the time of destruction (T+62.9 seconds) was 500 fps below nominal (7000 fps). This is much more than can be explained by abnormal gravitational and drag losses. It was concluded that the second stage motor (X-254) performed below its nominal propulsive capability, which supports the possibility of exhaust gases leaking at the motor headcaps and producing a temperature rise in the transition section forward of the second stage motor immediately after second stage motor ignition.

Hot gases escaping around a loose motor plug (at the head end of the X-254 motor) caused a temperature increase forward of the motor. Erosion of the plug weakened the suppressor paddle support. The suppressor paddle support failed and became wedged in the X-254 motor, causing overpressure and rupturing of the motor sidewall. Radar and camera coverage showed the X-254 motor flamed out almost completely at 26.44 seconds after ignition, indicating a large motor wall rupture.

Telemetry

The quality of data from certain channels was poor during the entire flight, possibly due to RF feedback and high noise levels. The C-band transponder shifted frequency at T+200 seconds, which made tracking marginal and the signals were lost at T+348 seconds. Excessive heating subsequent to second stage failure possibly contributed to the frequency shift.

Severe heating of the instrument tray located at missile Station 120 was recorded immediately after X-254 motor ignition at T+36.46 seconds. This heat may have contributed to the loss of the pedestal amplifier voltage which supplies voltage to the ADF subcarrier oscillators, causing a frequency shift of the ADF oscillators at T+48 seconds and resulting in a final dropout of these five oscillators at T+52.8 seconds.

Other telemetry dropouts and failures were: a 4-second dropout at T+58.3 seconds, a 25-second dropout at T+100.5 seconds, three breakwire channels erroneously indicated events, two temperature probes broke during second stage burning, and the command-destruct channel falsely indicated destruct at T+38 seconds. Although a command destruct signal was erroneously indicated, the flight termination subsystem functioned properly.

Ground Support

This was the first missile procured by the Air Force from NASA. The manufacturer shipped the missile under the procedures outlined by NASA, which required the plugs to be loose for instrumentation installation. These shipping instructions were not furnished to the Air Force. A review of Air Force prelaunch procedures for this vehicle revealed that inspection of motor plugs was not specified, or made, prior to launch.

Prelaunch checkout procedures were amended to include mandatory physical torque checks of all motor plugs prior to launch. Motor production and shipping procedures were also revised, requiring the manufacturer to properly install and tighten all motor plugs prior to shipping.

REFERENCE

"Final Test Report, Blue Scout Junior 2356, Vehicle Flight Test D-2," Aeronutronic, 8 January 1961.

Project: Explorer

Launch Date: 4 Dec 1960

Launch Area: Wallops Island

Countdown Holds: 0

Launch No.: 5

Missile Type: SLV-1A

Serial No.: ST-3

Result: Failure

PRIMARY OBJECTIVE

Place an inflatable satellite payload into orbit on a planned west-to-east trajectory.

MISSION SUMMARY

The objective was not achieved due to failure of second stage ignition. At the programmed time (T+70.7 seconds) second stage ignition failed and the first stage did not separate, resulting in destruction of the vehicle on impact at T+256 seconds.

PERFORMANCE

Propulsion

The first stage Algol motor flight velocity was greater than the theoretical velocity at all times during burning. The velocity varied from 9.2 percent greater at 15 seconds to about 11.8 percent greater at 35 seconds, and then to 1.6 percent greater at burnout. The headcap chamber pressure measured 17, 340 psia/sec (nominal total pressure integral 16, 980 psia/sec) which was 2.1 percent higher than nominal. The flight web burning time was 33.94 seconds (nominal 36.06 seconds at 70°F) which was about 5.9 percent shorter than nominal, and the average pressure was about 7 percent higher than average nominal.

First stage pressure records indicated vibration of approximately 60 cps, 6.6 psi (peak-to-peak), from 21 to 34 seconds. This vibration was also observed in the base on the normal accelerometer, and on the payload longitudinal accelerometer. Similar vibrations were noted (62.5 cps, 6.1 psig, 21 seconds to burnout) by Aerojet on the last motor (C-13) in the ground test program.

Ignition

There were several areas of possible failure:

1. An ignition plug located in section D (fourth stage) was not safety-wired, and a common wire in the motion switch circuit could have been broken during checkout. However, some of the wires through this plug performed properly during flight.

2. Ignition arming bars, located in the transition sections of all four stages, may not have been in the correct position prior to launch. These arming bars could be in three positions: open, shorted, or flight.

3. Short circuits in several parts of the ignition wiring system could cause the failure noted. Final checks and safety wiring of the electrical system took place before ignition sequence tests on the launcher. Few plugs are broken after this test, but continuity tests are not made on the power side of the circuit after this test. Ignition circuit resistance tests are run only on the squib side of the arming bars.

4. Short circuit in the battery. Although extremely unlikely (without a subsequent explosion) there is a possibility of a short occurring and causing ignition failure.

5. Failure of both first-motion switches in Base A section. Firing two squibs in section D connects the timer to the ignition circuit. These switches open in such a manner that debris could cause jamming. The switches cannot be checked once the vehicle is on the launch pad.

In view of the above failure possibilities, the following precautionary measures were taken:

a) Double safety wiring of the ignition plug. Inspection and work log certification at each level during final check and countdown.

b) Photographic record of each transition and arming area before hatch closure.

c) Preflight checkout procedures will require a complete electrical inspection of the entire launch harness.

d) Future vehicles will use separate destruct and ignition batteries as single-dual units; each battery supplying one ignition and one destruct circuit.

e) Mode of operation of first-motion switches will be changed and provisions for checkout added on subsequent vehicles.

REFERENCE

"Preliminary Flight Tests Results of Scout ST-3, " Langley Research Center, 15 December 1960, Unclassified.

Project: Probe	Launch No.: 6
Launch Date: 7 Jan 1961	Missile Type: Blue Scout I
Launch Area: AMR-18B	Serial No.: D-3
Countdown Holds: 0	Result: Success

PRIMARY OBJECTIVES

1. Demonstrate the capability of the booster vehicle to place a 392-pound payload into a predetermined high-altitude probe trajectory.
2. Obtain data related to the vehicle performance.
3. Obtain data from the experiments in the payload.
4. Evaluate and recover the re-entry vehicle.

MISSION SUMMARY

The overall objectives of this first SLV-1 vehicle launch (XRM-89, obsolete) were achieved with the exception of the payload and third stage motor heat shields not ejecting due to a wiring malfunction, preventing exposure of the experiments and reducing the third stage range. Third stage apogee of 750 n mi was 150 n mi short of the preflight estimate. Recovery vehicle impact point was 120 n mi short of predicted. The re-entry vehicle was not recovered.

PERFORMANCE

Propulsion

First Stage. Performed at higher thrust level, but for shorter period than predicted. Total impulse was 0.25 percent low.

Second Stage. Lower nominal thrust for longer period than predicted. Total impulse was 3 percent low.

Third Stage. Lower nominal thrust for longer period than predicted. Total impulse was 3 percent low. Burnout velocity was approximately 1260 fps low due to ejection failure of heat shields at third stage ignition and low third stage performance.

The highest vibrations recorded in the base section at Station 105 of the payload carrier were 7 g at 1475 and 2000 cps, and 2.5 g at 500 cps. Vibration levels recorded throughout the payload carrier were of the same magnitude, or less, apparently caused by resonant burning of the third stage motor for 5 seconds near burnout (approximately 133 to 138 seconds).

Guidance and Control

First Stage. Dispersions were greater than predicted and controls bottomed near second stage ignition. Most of the dispersions were credited to thrust misalignment, winds, and performance changes. Also contributing were errors in computing flight test data, guidance characteristics, programmed pitch rate, and aerodynamic stability and control parameters. Control surface bottoming was attributed to wind gradients near 90,000 feet which caused oscillations in pitch.

Second Stage. Telemetry data indicated that the lower right-roll control motor was not operating. Coil voltage was confirmed, but there was no indication of pressure switch action from the pressure switch telemetry. This condition persisted until late in second stage operation, at which time intermittent, then apparently proper, operation was obtained. Poor resolution of the telemetered rate signal did not allow positive verification of no-thrust by measurement of the acceleration levels. The erratic duty cycle could indicate either this unbalanced and intermittent thrust action or merely that non-steady roll torques were generated by the Castor motor. Failure or sticking of the pressure switch and/or a cold-start problem with the hydrogen peroxide roll motor are also possible.

Third Stage. The roll motors were apparently overpowered by main motor torque approximately 10 seconds from the end of burn, resulting in a large roll transient and loss of roll reference. For future flights, 14.5 pound thrust motors will be used to provide adequate control force.

A pitch transient occurred just prior to re-entry (T+1246 seconds). Circumstantial evidence indicates a control system malfunction.

Airframe

The payload and third stage heat shields did not eject due to a wiring malfunction. Breakwires and temperature transducers were still intact past the normal time of separation and for as long as data were obtained. Flight test data indicated a direct short for approximately 0.2 second in battery No. 2, which supplies voltage to the explosive bolts of the payload and the squib actuators of the motor heat shields. The time period was coincident with initiation of heatshield ejection.

At T+101.1 seconds, similar signal discontinuities were noted in all systems to which battery No. 2 supplied power (telemetry system No. 2, signal condition box No. 2, destruct system No. 2, radar beacon, WADD-7, and destruct system No. 2 monitors). After the 0.2-second interval all functions resumed their normal operating levels, indicating that either the 28-volt conductor burned through or the guidance and control timer relay contact (2 amp rating) burned out. Corrective action was incorporation of complete redundancy in the payload heatshield bolt ignition circuitry, and modification of preflight checkout procedures by including additional heatshield assembly detail and checkout procedures.

Electrical

All electrical equipment and wiring operated as designed except for the power circuits associated with heat shield ejection. Improved ejection circuitry and more complete checkout procedures are to be used on future vehicles.

Recovery Vehicle

No usable signal was received from telemetry due to heat shield malfunction. All recovery aids functioned except the flashing light, but the vehicle was not recovered.

Facility and Ground Support Equipment

The launcher azimuth ring was locked in position after launch when the azimuth seal melted and the molten metal ran into the bearing, welding it in position. Some erosion and distortion of the rings occurred. A heat shield or blast deflector will be added to eliminate this problem.

REFERENCES

"Trajectory and Aerodynamic Information for TS609A (D-3) Vehicle," Aeronutronic publication No. C-997, 19 September 1960.

"Additional Supplemental Trajectory Information for TS609A 123A (D-3) Vehicle," HETS memo 1838, 5 December 1960.

Aerojet-General Letter SRP: 5720: 0989, dated 30 August 1960.

"Information and Preliminary Data on the Scout ST-1 Launching," NASA Letter Report to Chance Vought Aircraft, dated 3 February 1961.

Project: Explorer IX

Launch Date: 16 Feb 1961

Launch Area: Wallops Island

Countdown Holds: 0

Launch No.: 7

Missile Type: SLV-1A

Serial No.: ST-4

Result: Success

PRIMARY OBJECTIVE

Place the Explorer IX satellite into orbit on a planned west-to-east trajectory.

SUMMARY

The objective was achieved. Injection of the payload occurred at 363.8 n mi, with an apogee of 1400.7 n mi and a perigee of 342.42 n mi.

PERFORMANCE

Propulsion

All motors except the fourth stage Castor motor exhibited a shorter web time than nominal, but within predicted deviations.

Guidance and Control

After liftoff, pitch-up (0.5 degree) of the vehicle occurred at approximately T+1.5 seconds, with control surface deflections (pitch-down) correcting by T+3 seconds. This motion had not been as evident on previous flights. The cause is not definitely determined but investigations will be made on future flights.

Telemetry

The S-56A tracking beacon on the payload did not operate. When the payload came out of the earth's shadow, the higher solar cell voltage and the high internal battery resistance allowed an excessive voltage (30 to 35v) to be applied to its transistors, causing breakdown.

REFERENCE

"Preliminary Flight Test Results of Scout ST-4," Langley Research Center, 8 March 1961, Unclassified.

Project: Probe

Launch Date: 3 March 1961

Launch Area: AMR Pad 18B

Countdown Holds: 1

Launch No.: 8

Missile Type: Blue Scout II

Serial No.: D-4

Result: Success

PRIMARY OBJECTIVES

1. Place a 170-pound payload into a predetermined high-altitude probe trajectory.
2. Obtain data from the experiments in the payload.

MISSION SUMMARY

The mission was considered successful, although one anomaly was noted: fourth stage impact prediction was 102 n mi short and 74 n mi to the right due to a 2-degree-high flight path angle at fourth stage burnout.

The countdown hold (2 minutes) was for a procedural change to permit a final functional demonstration and/or preheating of the hydrogen peroxide systems. Flight countdown time was 420 minutes.

PERFORMANCE

Guidance and Control

First Stage. A slightly early initial pitch command caused the actual flight path to be below the nominal flight path for the first 35 to 40 seconds. The motor performance was higher than nominal, which, coupled with downrange winds, caused dispersions in pitch.

Second Stage. Roll reference was lost for approximately 8 seconds at second stage ignition. The cause of this roll transient has not been determined although first tendencies are to attribute large roll moments to the motor. Evidence indicates misalignment of the high-thrust pitch and yaw jets which immediately fired at second stage ignition. The maximum available roll control torque decreases appreciably when the large pitch and yaw jets are operating, since they share the same hydrogen peroxide reservoir. While the roll jets indicated ON there was a period of 4 seconds with apparent zero roll acceleration which cannot be explained. Thorough examination of data is required. There is disagreement between roll-right deadzone and roll-left deadzone (1.8 degrees and 3.5 degrees, respectively) which requires further telemetry data analysis.

Third Stage. Roll control deadzones indicated the same out-of-tolerance condition noted on second stage operation. These deadzones were designed to be identical for second and third stages. The data did not indicate whether the error was in telemetry or in the control system.

The roll control jet thrust, which had been increased to 14.5 pounds due to the anomaly which occurred on the D-3 flight (Launch No. 6), operated successfully.

Fourth Stage. Premature switchoff of the peroxide system and change-over to the hot gas control system occurred approximately 1 second prior to third stage burnout. The single timer signal to fire the separation charge, transfer control from the H_2O_2 to the hot gas system, ignite the hot-gas generators, and ignite the pyrotechnic delay squib in the motor (1.8 seconds delay) occurred approximately 1 second before closure of the motor pressure switch (set at 50 psi). This resulted from long burning of the third stage motor. Normal operation requires closure of the pressure switch as an arming device prior to timer signal initiation of the above functions. The flight timer for missile D-5 (launch No. 9) was modified to add an additional 3-second delay to this time function, at a slight penalty in vehicle performance.

The error of 1 second in time sequencing caused a corresponding period of "no control" while the third stage motor was in its "tail-off" condition. Hot-gas electronics and valves were functioning properly, but the generators were not ignited until pressure switch closure.

Approximately 4 seconds after fourth stage ignition the vehicle rolled clock-wise, bottomed the roll gyros, and developed a maximum roll rate of approximately 1 rps by the end of fourth stage burn. This roll moment was induced by the X-248 motor. Because yaw and roll share the same hot gas valves, the amount of control force available for roll is limited by the amount required to counter-balance the existing thrust misalignment in yaw. Of the total 9 lb available for roll-yaw control, approximately 6.5 lb were required for yaw thrust alignment, leaving 2.5 lb, or approximately 3 ft-lb of torque, available for roll control. This net torque of 9 ft-lb can account for the initial 25 deg/sec^2 of roll acceleration. A roll rate gyro is being provided for the D-5 vehicle to aid in defining the roll moment characteristics of the X-248 motor.

Coast control was lost in both pitch and yaw due to a second timer sequencing error. This error resulted from early ignition of the hot gas system, and a subsequent long burn of the fourth stage motor. These compounding events led to switching off the hot gas control approximately 2 seconds prior to fourth stage motor burnout. Although the proper valves turned on and stayed on in an attempt to recapture, the control fuel was depleted before the low-thrust-level system could appreciably reduce the rate errors induced during burn of the motor.

The D-5 timer is being changed to add a 5-second delay to the "hot gas to cold gas" switch-over signal. This 5-second delay includes the 3-second delay noted for hot gas initiation and an extension of 2 seconds in the operating time of the hot gas system.

Telemetry

Both ADF and AFSWC telemetry system signal strengths were normal except for an 11.5-second period of erratic operation which started at T+82 seconds.

Ground Equipment

Hot gas control console land-line instrumentation appeared to be faulty. The blockhouse control circuit for H_2O_2 control system preheat functional check-out did not function properly. Both problems were resolved before the next launch (vehicle D-5).

REFERENCE

"Preliminary Test Report, Blue Scout II, 1234A Vehicle Flight Test D-4," Aeronutronic, 17 March 1961.

Project: Space Probe

Launch Date: 12 April 1961

Launch Area: AMR Pad 18B

Countdown Holds: 1

Launch No.: 9

Missile Type: Blue Scout II

Serial No.: D-5

Result: Success

PRIMARY OBJECTIVES

1. Place a 365-pound payload into an attitude-programmed, high altitude probe trajectory.
2. Prove the basic design concepts of the vehicle (XRM-90).

MISSION SUMMARY

The objectives were successfully accomplished. At T-125 seconds one unscheduled hold was incurred. A cold solder joint failed in a plug connection between the command/destruct box and command receiver. Approximately 80 percent of the hold time was required to remove and replace the heat shield to effect repairs. Countdown time was 420 minutes; total hold time 217 minutes.

PERFORMANCE

Propulsion

First Stage. Motor performance was at a higher thrust level but for a shorter burning time than predicted due to downrange winds which caused the vehicle to pitch over during the first 20 to 25 seconds of flight. After 25 seconds, the high down-range wind component carried the vehicle to a pitch-up condition and, with the increased velocity, caused the flight path to be above the nominal. These winds, coupled with the variations encountered in motor performance, are major causes of pitch dispersion.

Second Stage. The second stage motor performed at lower nominal thrust for a longer period of time than predicted. Analysis, after recovery of the first stage rocket motor, indicated that the second stage diaphragm did not function as designed. The diaphragm did not break, nor did all the blow-out doors on the interstage release as designed. Separation was accomplished by deformation of the diaphragm, which permitted the nozzle and diaphragm threads to disengage. No immediate corrective action is necessary as the mechanism of separation is apparently adequate. A severe roll transient at second stage motor ignition, possibly augmented by the XM-33 motor during initial burning, resulted in a 2-degree loss of roll reference. Improper functioning of the separation diaphragm did not appear to have affected these transients. Abnormal roll rate and loss of roll reference show some correlation with the pitch-down and yaw-right jets.

Third Stage. The third stage motor performed at a lower nominal thrust for a slightly longer period of time. A roll disturbance was noted during resonant burn of the ABL X-254 motor; control, however, was adequate and the transient action does not represent a problem.

During X-254 motor burning, the longitudinal vibration levels were generally under 4 g for the range of 0 to 2550 cps. Exceptions existed at three distinct peaks noted during a period of 30 to 35 seconds after ignition. Two peaks, one 16 g at 1480 cps and the other 20 g at 1980 cps, compare in frequency with the D-4 flight data (launch No. 8) but are approximately twice the magnitude. The third peak of 10 g was noted at 510 cps.

Fourth Stage. The fourth stage motor burned approximately 5 seconds longer than expected, at an appreciably lower chamber pressure (440 fps low). This longer operation could be attributed to the lower-than-normal conditioning temperature which was provided in the payload compartment to meet experimental requirements.

High roll torques, generated by the X-248 motor, were experienced during resonant burning of the fourth stage motor. A counterclockwise moment in roll overpowered the control system, and caused the vehicle to roll, bottoming the roll gyro. The roll rate continued to increase until a maximum of 2 rev/sec was reached at the end of boost. The roll moment was attributed to the X-248 motor transient. At T+167 seconds, control was lost simultaneously in pitch and yaw. Pitch-up and yaw-right motion of the vehicle were severe enough to cause gyro bottoming and permanent loss of control. The control system operation appeared to be normal in that it functioned to oppose gyro errors (roll gyros reacted in same manner).

Changes in thrust misalignment components in pitch and yaw induced by resonant burning of the fourth stage motor appear as a possible reason for the loss of control in both planes.

The data recovery vehicle did not separate from the vehicle and consequently was not recovered. Only limited data were obtained on the recovery vehicle extension mechanism. A premature tone 7 signal at T+1178 seconds resulted in an early signal for retraction of the re-entry vehicle heat shield. Because of loss of fourth stage roll stability, the mechanism did not fully retract and latch, thereby preventing firing of the recovery vehicle explosive bolt mechanism. Elimination of the fourth stage roll problem would assure proper operation of the extension mechanism.

During X-248 motor burning, the longitudinal vibration levels, for the period 13 to 23 seconds after ignition, were approximately 15 g at 2500 to 2600 cps and 17.5 g at 2200 to 2500 cps, with a maximum vibration peak of 21 g at 2250 cps.

During the period of 27 to 31 seconds a sharp vibration peak was experienced at 570 cps, with a magnitude that exceeded the telemetry band width of 45 g. Additional peaks at 1140 cps of 17.5 g and at 1710 cps of 4 g were measured which appear to be harmonics of the 570 cps vibration. No similar harmonics were noted on the D-4 flight. In general, harmonic levels measured in the longitudinal direction on this flight were higher than those previously obtained on flight D-4 during the X-248 motor burning.

REFERENCE

"Flight Test Report - TS 609A, Blue Scout II, 1234B Vehicle Flight Test D-5," Aeronutronic, 22 May 1961, Unclassified.

Project: Space Probe

Launch Date: 9 May 1961

Launch Area: AMR Pad 18B

Countdown Holds: 0

Launch No.: 10

Missile Type: Blue Scout I

Serial No.: D-6

Result: Failure

PRIMARY OBJECTIVES

1. Demonstrate the capability of the booster vehicle to place the 445-pound payload into a predetermined high-altitude probe trajectory.
2. Obtain vehicle performance figures.
3. Obtain data from experiments in the payload.
4. Recover and evaluate the re-entry vehicle.

MISSION SUMMARY

The vehicle achieved a normal liftoff and satisfactory first stage operation. It veered off course shortly after second stage ignition (T+64.0 seconds), making destruction necessary at T+82.0 seconds.

No test objectives were achieved beyond the demonstration of satisfactory first stage performance. No experimental or recovery vehicle data were obtained.

PERFORMANCE

Propulsion

Preliminary estimates of vehicle velocity taken from radar data indicate that the first stage velocity was approximately 100 fps low at second stage ignition and 750 fps low at the time of destruct initiation (80 seconds). At least 200 fps of the velocity loss during second stage occurred during the first 11 seconds of burning, indicating that the second stage was performing below specifications from time of ignition.

The first stage motor experienced a slightly higher chamber pressure and shorter burning time than predicted. The second stage chamber pressure data are of questionable value because a much lower pressure level is shown during the first 18 seconds of burning than is compatible with the radar velocity or telemetered accelerometer data. Following destruct action, the telemetered indication immediately jumped to a higher level, which was nearer the expected value. It was determined that the problem existed in the monitoring system rather than in the second stage motor.

First stage separation occurred with normal pitch, yaw, and roll disturbances. Longitudinal vibration data indicated a higher than normal transient at the time of second stage ignition. In addition, the telemetry drop-out, which was attributed to antenna blanking by the ignition plume, was more severe than on any previous flight.

It is not obvious why second stage thrust was not terminated coincident with third stage destruction. It has been concluded that the most probable cause was that the power leads to the second stage destruct system were destroyed by detonation of the third stage shaped charge. The leads were in the same tunnel with the charge, and the squib firing time tolerance is ±50 percent (with roughly 95 percent of the firing time needed for heating the squib bridge wire). The third stage system could therefore fire in advance of the second stage system. In addition, the second stage system received less current because of the longer leads to the battery, tending to slow the firing action. Future vehicles will be modified as follows:

a) All destruct-system wiring will be routed down the instrumentation tunnel away from the shaped charge installation.

b) Limiting resistors will be installed in series with each destruct system safe/arm unit to balance the current supplied to each stage.

Control System

Approximately 18 seconds after second stage ignition, the yaw-right control motor failed to produce thrust due to a break in signal continuity. This resulted in steady state deviations from the programmed flight path. Contributing circumstances were possible thrust misalignment in yaw of about 0.025 degree left, or 0.2 deg/second^2, and winds from the south in a cross-plane direction. The vehicle was destroyed by the RSO at approximately T+82 seconds.

All observations concerning loss of yaw-right control pointed to a break in signal continuity in the control system between the poppet valve electronics and the hydrogen peroxide valve coils. Also suspect were wiring defects in telemetry leads on the three peroxide motors (yaw-right, pitchdown and lower right-roll motor).

The exact cause or type of wiring harness failure could not be determined. A detailed inspection of control system wiring harnesses delivered for subsequent vehicles disclosed excessive and improper splices. It was necessary to completely rewire the electrical harnesses for vehicles D-7, D-8, and D-9 in the field. The defective harness, produced prior to the establishment of Vought Astronautics Division as prime system contractor, had been manufactured by another contractor. All subsequent harnesses procured have been Vought-manufactured, with special emphasis on quality control. This procedure produced fault-free electrical harnesses as determined by inspection. No further or similar control system harness failures were encountered on subsequent vehicles.

REFERENCE

"Flight Test Report, Blue Scout I, 123B Vehicle, Flight Test D-6," Aeronutronic, 26 June 1961.

Project: S-55 Satellite

Launch Date: 30 June 1961

Launch Area: Wallops Island, Pad 3

Countdown Holds: 1

Launch No.: 11

Missile Type: SLV-1A

Serial No.: ST-5

Result: Failure

PRIMARY OBJECTIVES

1. Perform a vehicle development flight.
2. Inject the S-55 micrometeoroid satellite into a near-earth orbit.

MISSION SUMMARY

Objectives of this flight were not achieved. The third stage did not ignite (programmed for T+140.2 seconds) and the second stage failed to separate from the third stage due to improper venting of the nozzle closure plug located in the third stage ignition circuit. The vehicle was destroyed by the Range Safety Officer shortly after this failure occurred. Telemetry signals were lost at T+246.67 seconds.

The countdown hold lasted 7 minutes, due to the blockhouse console battery activation switch which remained in the "on" position after the battery activation signal was commanded. The switch was turned off and proper voltages were monitored.

PERFORMANCE

Propulsion

The first and second stage motor burning time was, respectively, 2.44 and 2.90 seconds longer than predicted.

Ignition

Improper venting of the nozzle closure plug, located at Station 238.3, allowed a differential between the third stage headcap pressure and ambient pressure to increase to a point where the flight closure plug and ignition leads were forced out of the nozzle.

A decrease of third stage headcap pressure of only 3.5 psia and a decrease in ambient pressure of 12.5 psia were recorded prior to T+38 seconds. Immediately after T+38 seconds, the third stage headcap pressure dropped abruptly below the ambient pressure and leveled off at 1 psia above the ambient pressure. This erratic headcap pressure fluctuation indicates improper venting of the flight closure plug. The flight closure plug for the third stage, as unpackaged, had neither silica gel nor masking tape installed in the two counterbored cavities in the aft end of the plug. Silica gel bags were obtained from a spare plug and were installed. The cavities were then covered with masking tape and the tape was to have been perforated prior to launching. No positive verification can be made that this was done.

Telemetry

Telemetry data from the third stage were unreadable due to the failure of the second stage to separate. The third stage instrumentation shares common channels with the second stage instrumentation and is programmed to commence transmitting upon second stage separation. Following separation failure, both second and third stages continued to transmit. As a result, an unreadable mixture of signals was transmitted. Interference was also noted between the radar beacon and certain telemetered channels due to signals being fed from the radar beacon through the power source and into the telemetry system. In the future, separate power supplies will be used for the radar beacon and telemetry system.

As a result of the poor telemetry data, neither the attitude nor pitch rate could be determined and consequently the nominal planned pitch program was used for postflight trajectory calculations. There was no indication, from these calculations, of deviations from the intended pitch program.

REFERENCE

"Scout ST-5 Final Flight Report," Vought Astronautics, 21 August 1961, AST/EIR-13409, Unclassified.

Project: Space Probe

Launch Date: 17 Aug 1961

Launch Area: AMR

Countdown Holds: 3

Launch No.: 12

Missile Type: SLV-1B

Serial No.: 0-1

Result: Success

PRIMARY OBJECTIVES

1. Put a payload into space.
2. Demonstrate the operational capability of the vehicle.

MISSION SUMMARY

Operation of the vehicle was generally satisfactory until 16 seconds after fourth stage (XM-85) rocket motor ignition, when telemetry was lost. Loss of telemetry occurred at the time squibs were fired to activate an experiment in the payload. It could not be determined if the expected apogee of 126,000 nautical miles was attained.

The countdown holds were: (1) rain and threat of rain prevented removal of panels from vehicle for a period of 141 minutes, (2) lack of experienced ordnance personnel held countdown for 52 minutes, and (3) a 5-minute extra hold for beacon warm-up time.

PERFORMANCE

Guidance and Control

The combination of spin rockets and aerodynamic forces on the first stage fin tips and second stage fins was not adequate to attain or maintain the desired spin rate of 3 - 4 rps. Modifications to subsequent missiles were accomplished to maintain a spin rate of 3 rps during flight.

Telemetry

All battery power to the payload passes through a miniaturized magnetic latching relay which is closed before launch. A switch closes at 15 g during third stage rocket motor operation and starts a timer which, after 30 seconds, closes the circuit to fire four guillotine squibs. These squibs, when fired, cut strings holding protective covers over electrostatic analyzers in the experiment package. Data analysis showed that loss of telemetry occurred at the time these guillotine squibs were fired. Subsequent bench tests showed that these squibs can short out upon firing and draw 20-25 amperes through the power control relay. As this power control relay had a rating of only one ampere, the high current flow caused the circuit to open, thus cutting the battery power to the payload. It appears that loss of flight telemetry was the result of a similar power loss brought on by shorting of the guillotine squibs. In future designs of pyrotechnic circuits, provisions were made to eliminate the possibility of high current drains as a result of shorted squibs.

No structural environment, engine pressures, or other pertinent data were programmed which might have been useful in immediately determining the malfunction which occurred during fourth stage operation. Adequate instrumentation will be provided on subsequent vehicles to determine operational capability.

Trajectory

The vehicle heading departed left (2.75 degrees) and upward (2.16 degrees) from the planned trajectory. This heading, however, did not exceed the three-sigma preflight prediction. A study has been made of the method for calculating azimuth and elevation prelaunch windage conditions.

REFERENCE

"Blue Scout Jr. (XRM-91) 0-1 Vehicle Flight Test Report," BSD, 19 September 1961, DWZS-TM-61-1.

Project: Explorer XIII

Launch Date: 25 Aug 1961

Launch Area: Wallops Island Pad 3

Countdown Holds: 1

Launch No.: 13

Missile Type: SLV-1A

Serial No.: ST-6

Result: Success

PRIMARY OBJECTIVE

Inject a second model of the micrometeroid satellite (S-55a), Explorer XIII, into a near-earth orbit.

MISSION SUMMARY

Not all objectives were achieved, due to failure of the satellite to remain in orbit the planned predicted life (2-1/2 years).

Telemetry and tracking radar data analyses indicated that all systems performed normally up to the point of fourth stage ignition. After fourth stage ignition (484.04 seconds), data are not available on vehicle behavior. The decay of the Explorer XIII orbit was so rapid that Minitrack stations had difficulty in tracking; consequently, orbital conditions have not been defined to date.

The countdown hold was due to plug J209 which had several wires shorted to ground. Rewiring and retaping the plug caused excessive pressure due to bulkiness when wires were reinserted in the plug. The shielded wire penetrated the wire insulation and shorted. Each wire was rewrapped with teflon insulation, the plug was changed, and the adaptor clamp removed from the plug to correct the condition.

PERFORMANCE

Trajectory

The flight path error angle was 4 degrees, as compared to a nominal of 0.013 degree. There are two probable causes of this error: (1) a pitch rate signal error of so small a magnitude as to be undetected by the instrumentation system, or (2) a tipoff error at the time of fourth stage ignition.

Pitch program calibration seems suspect, but due to limitations in instrumentation accuracy a positive conclusion cannot be reached. The pitch program obtained from the flight records was adjusted by adding a constant rate (-0.104 deg/sec) in order to obtain the proper trajectory at third stage burnout. Close agreement with the nominal pitch program was obtained and the integrated values of the adjusted pitch rate showed that the actual pitch rate did not return to zero at 222 seconds, but continued at a rate of approximately 0.04 deg/sec negative. This amount of drift resulted in an attitude error of -11.1 degrees at fourth stage ignition. The trajectory computed with this pitch program resulted in an injection altitude of 246.15 nautical miles and a flight path angle of -4.396 degrees, values which are close to the values of the Minitrack data.

Corrective action for this problem (made on vehicle ST-8, launch No. 17) was use of a "cold" fourth stage separation system.

REFERENCE

"Scout ST-6 Final Flight Report," Vought Astronautics, 12 October 1961, AST/EIR-23.3, Unclassified.

Project: Probe

Launch No.: 14

Launch Date: 19 Oct 1961

Missile Type: SLV-1A

Launch Area: Wallops Island, Pad 3

Serial No.: ST-7

Countdown Holds: 1

Result: Success

PRIMARY OBJECTIVES

1. Perform a vehicle development flight.
2. Place the P-21 Electron Density Profile payload in orbit.

MISSION SUMMARY

All objectives of this flight were achieved. Launch azimuth was 0.98 degrees and the vehicle launch angle was 84.9 degrees.

The countdown hold was due to tube failure in the autocollimator electronics, Spandar radar failure, FRW-2 command destruct transmitter failure, and a broken umbilical bungee cable.

PERFORMANCE

Propulsion

A tipoff at fourth stage ignition resulted in a nose-down attitude error of 8 degrees. Preflight, postflight, and radar data presentations are made with the 8-degree error introduced for correlation. The 8-degree nose-down pitch attitude change was approximated by averaging the conditions shown by the NASA fourth stage payload data at fourth stage ignition. Predicted motor performance was used and analyses of conditions at fourth stage ignition were based upon the postflight trajectory which included winds, atmosphere, and thrust, using accelerometer data for first, second, and third stages, which were in close agreement with the radar tracking data.

Vibration

Oscillations (14 cps) were noted in the pitch and yaw rate at third stage burning which appeared to be greater than those previously obtained. These higher oscillations could be due to increased instrument sensitivity as they compared well with flight ST-6 (launch No. 13) except at ignition of the third stage. Calculations for the third stage with heat shield on show a first-bending mode frequency of 20 cps. The first-bending mode is expected to drop considerably due to removal of the forward heat shield, which stiffens the payload.

The decision was made to use a "cold" fourth stage separation system.

REFERENCE

"Scout ST-7 Final Flight Report," Vought Astronautics, AST/EIR-23.7, 7 December 1961, Unclassified.

Project: Mercury

Launch Date: 1 Nov 1961

Launch Area: AMR-18B

Countdown Holds: 3

Launch No.: 15

Missile Type: SLV-1A

Serial No.: D-8

Result: Failure

PRIMARY OBJECTIVES

1. Inject the payload into orbit over the Mercury worldwide tracking range.

2. Track the payload for approximately three days.

3. Obtain booster vehicle data during powered flight.

MISSION SUMMARY

The test objectives were not achieved due to a control malfunction which started at 0.5 second after liftoff and lasted until complete destruction of the vehicle at T+44.2 seconds.

The vehicle commenced to roll, yaw, and pitch in an erratic manner 0.5 seconds after liftoff, although flight was maintained. The nose and fourth stage heat shields separated prematurely from the vehicle at T+14.9 seconds. At T+26.7 seconds, transition section C of the third stage disassembled, causing autodestruction of the second and first stage motors. The Range Safety Officer commanded destruction of the third stage motor at T+44.2 seconds. Loss of third stage telemetry signals occurred at T+100 seconds and loss of payload telemetry signals at T+117 seconds.

The countdown holds were (1) checkout of S-band beacon interrogation scheme, (2) nonejection of payload umbilical, and (3) dissipated telemetry batteries in the lower D section. Flight countdown time was 420 minutes.

PERFORMANCE

Control System

All flight test data point to the interchanging of three connectors between the flight control rate gyros and the flight control system. Telemetered rates and composite error signals revealed an intermixing of control system channels. The yaw rate gyro signals and pitch composite signals had similarities in general waveform. These similarities were mechanization errors which resulted from attitude signals being added to cross-axis rate signals.

Determination of the intermixed transducers was made by integrating and summing various combinations of integrated rate signals (corresponding to a displacement) and rate signals with an analog computer, then comparing with the actual telemetered composite error signals.

First stage control is achieved by proportional control of a jet vane fin system located in the base A section. Control composite error signals for each channel are formed by adding the position error signal (from the MIG gyros) and the corresponding rate error signals (from the GNAT rate gyro package).

Prior to the next flight, a single-key connector was used to eliminate the possibility of incorrect coupling of connectors.

REFERENCE

"Scout 1234S D-8, Flight Test Report," Aeronutronic, 22 December 1961.

Project: Space Probe	Launch No.: 16
Launch Date: 4 Dec 1961	Missile Type: SLV-1B
Launch Area: PMR	Serial No.: 0-2
Countdown Holds: 0	Result: Success

PRIMARY OBJECTIVE

Measure low energy proton flux (solar wind) in regions beyond the outer radiation belt.

MISSION SUMMARY

Due to failure of all TLM-18 stations, except Vandenberg AFB, to acquire the telemetry signal, the ballistic trajectory of this flight is unknown. Telemetry data indicated an unusual occurrence at or near fourth stage ignition (T+115.22 seconds) in that the payload was wobbling through an angle of at least 55 degrees from its principal spin axis.

Impact of the first stage was approximately 3000 yards short and to the left of the nominal assigned range impact area - or within the cumulative failure impact area. This variation is attributed to the high winds encountered at apogee which induced tumbling of the first stage at approximately 100 seconds. The tumbling body, with its increased drag, assumed a new trajectory of shorter range.

Second stage impact was within the three-sigma dispersion area.

At T+90 to T+100 seconds, the radar skin track data became erratic. This condition is attributed to the inability of this type of radar to skin-track so small an object at so great a distance. Since these data did not become erratic until well into third stage burning, it is assumed that vehicle performance was normal up to that point.

PERFORMANCE

Trajectory

The preflight computer apogee was 24,000 nautical miles at T+372 minutes at latitude 70 degrees south and longitude 152 degrees east. The launcher conditions at liftoff were azimuth angle 188.050 degrees, elevation angle 69.732 degrees. These corrected angles were to negate wind effects so as to realize a flight path corresponding to nominal launcher settings of 185 degrees azimuth angle and 72 degrees elevation angle. The actual payload weighed 28.5 pounds instead of the 30 pounds weight used for preflight trajectory calculations. As a result, the true path of the vehicle varied from the nominal by a small amount. The actual powered portion of the flight path was high and to the left of that predicted. The launcher vector elevation setting of 69.732 degrees was based on winds at T+35 minutes, as was the vector azimuth setting of 188.050 degrees. Due to increased winds during the 35 minutes before launch, a closer setting in the elevation angle (69.005 degrees) and azimuth angle (189.432 degrees) would have produced angles closer to nominal.

Propulsion

Possible causes of the upsetting moment encountered at approximately T+115.22 seconds can be determined only by conjecture. These possibilities are:

a) A purge hose connected to the payload and to the third stage motor was designed to disconnect upon third stage separation with a 2-pound pull. Theoretical calculations reveal that this torque resulted in a 3-degree precession angle within 0.036 seconds; although this would not cause the entire observed precession angle, it would be a contributing factor.

b) If a fourth stage motor malfunction occurred immediately after fourth stage ignition, i.e., nozzle or case burn-through, nozzle blowout, or separation of the propellant from the case, the precession angle could be duplicated.

c) If the separation of the third and fourth stages was asymmetric, an upsetting moment could be induced.

Any combination of several possible malfunctions could have caused the wobble, however, no final conclusion can be made other than that a malfunction did occur at fourth stage ignition, inducing a precession angle of at least 55 degrees.

Environmental storage facilities are recommended to maintain vehicle motor temperatures within manufacturer's recommended limits.

Telemetry

Telemetry dropouts at the TLM-18 stations are attributed to the precession moment or wobble at fourth stage ignition.

It is recommended that an additional telemetry tracking station, with an assumed tracking resolution of 3 degrees, be located 15 degrees out of the orbit plane to obtain orbit solutions if malfunctions occur during powered flight. Also, more instrumentation should be provided for vehicle performance data.

REFERENCE

"Final Report for Blue Scout Junior Flight 0-2," AFSWC, TDR-62-24, March 1962.

Project: Probe	Launch No.: 17
Launch Date: 1 March 1962	Missile Type: SLV-1A
Launch Area: Wallops Island, Pad 3	Serial No.: ST-8
Countdown Holds: 5	Result: Success

PRIMARY OBJECTIVE

Perform a vehicle development flight. A re-entry type trajectory was planned which would satisfy the mission of five-stage Scout Re-entry Program Payload No. 1 (total heat transfer payload).

MISSION SUMMARY

All objectives were achieved. Launch azimuth was 129 degrees and the vehicle launch angle was 81 degrees. Weather conditions delayed launching five times.

PERFORMANCE

Propulsion

Maximum deviation in burntime of the first three rocket motors was 2.96 seconds and total impulse deviation was 2.64 percent. The fourth stage rocket motor thrusted 26.0 seconds instead of 41 seconds as predicted. Early thrust termination was probably due to a ruptured motor case (although structure analysis shows that the loads developed in flight should not have caused a failure). Due to failure of the fourth stage motor to burn the predicted time. the payload re-entry velocity was approximately 3,500 fps less than predicted.

REFERENCE

"Scout ST-8 Final Flight Report," Vought Astronautics, 3-13000/2R-157, 27 April 1962, Unclassified.

Project: Space Probe

Launch No.: 18

Launch Date: 29 March 1962

Missile Type: SLV-1A

Launch Area: Wallops Island, Pad 3

Serial No.: ST-9

Countdown Holds: 2

Result: Success

PRIMARY OBJECTIVES

1. Accomplish a probe-type trajectory designed to satisfy the mission of the P-21a Electron Density Profile Probe payload.

2. Continue development of the Scout vehicle.

MISSION SUMMARY

All objectives were achieved.

One countdown hold was due to radio frequency interference, a tripped circuit breaker in the guidance 37-volt power supply (reset and subsequently monitored due to nonavailability of a spare), and difficulties experienced in the radar data link between the launch site and the range control center subsequent to removal of vehicle heat. The second hold was due to shipping in the impact area.

SUBSYSTEM PERFORMANCE

Propulsion

The Algol, Castor, and Antares rocket motors closely approximated their predicted performance. This was the first SLV-1 vehicle to utilize the new Antares X-259 A1 motor. Based on programmed time, the deviation at the first pitch rate command signal was +0.18 second (programmed for 3.00 seconds).

Separation

After actuation of fourth step spring separation at 158.92 seconds of flight, a collision of the third and fourth stage occurred (at 160.06 seconds). Several analyses were performed to determine the cause of this collision, with the following conclusions:

1. The separation spring functioned properly and imparted the designed relative separation velocity between the third and fourth stages.

2. Based on the fact that the measured flight collision occurred at approximately 1.14 seconds after the spring separation event, the separation trajectory of the third stage (relative to the fourth stage) was such that the third stage collided with the motor nozzle of the fourth stage.

To preclude collision between the third and fourth stages, either the coast time may be increased by 2 seconds or the retrorockets may be fired 0.5 second after the spring separation event. If the coast time is increased by 2 seconds, then the peak separation distance is optimum before firing retrorockets. Firing of the retrorockets at 0.5 second would correct the separation trajectory for payloads up to 400 pounds.

REFERENCE

"Scout ST-9 Final Flight Report," Vought Astronautics, 24 May 1962, 3-13000/2R-173, Unclassified.

Project: Space Probe

Launch Date: 12 April 1962

Launch Area: AMR-18B

Countdown Holds: 0

Launch No.: 19

Missile Type: Blue Scout - I

Serial No.: D-7

Result: Failure

PRIMARY OBJECTIVES

1. Demonstrate the capability of the booster vehicle to place the re-entry vehicle in a predetermined hypersonic re-entry trajectory.

2. Obtain data related to booster vehicle performance.

MISSION SUMMARY

All test objectives were not achieved. The second stage motor did not ignite at the programmed time and the first stage failed to separate, resulting in loss of attitude control and destruction of the vehicle.

PERFORMANCE

Ignition

It is suspected that second stage ignition failure was due to a headcap pressure switch malfunction in the ignition circuit. At the time for second stage ignition (T+80.12 seconds), the 28-volt guidance and control battery is connected (by the timer and the first stage headcap pressure switch) to the ignition/destruct relays. The relays then close the circuit from the ignition/destruct batteries to the motor ignitors. Two connectors in the ignition circuit carry the second stage motor ignition signals: one is the headcap pressure switch connector (P103), the other is the ignition monitor connector (P112). All other functions which utilize these connectors and cables were operable. There was, however, no observed signal from the ignition current monitor (P112), although the guidance and control second stage ignition command was initiated.

At approximately T+30 to T+42 seconds an increase in vibration levels (3.75 g to 5.5 g) was recorded, possibly due to a resonant burning condition of the first stage motor. This increase in vibration levels could have caused the headcap pressure switch failure.

Corrective action to prevent ignition failure on subsequent vehicles was elimination of the headcap pressure switch from the ignition circuit.

Control System

Telemetered data indicated that the second stage hydrogen peroxide control system turned on in a normal manner and performed properly at the signal for second stage ignition. Since the first stage booster did not separate, the center of gravity of the vehicle was considerably aft, and the vehicle inertia was extremely high. This resulted in lowering the available control acceleration in pitch and yaw. The pitch and yaw jets were active for approximately 25 seconds, attempting to contain the initial capture rates, but the pitch and yaw gyros bottomed due to low acceleration levels and the hydrogen peroxide fuel supply became exhausted. Result was loss of attitude control and premature destruction of the vehicle.

REFERENCE

"Blue Scout I, 123C (XRM-89) D-7 Flight Test Report," Aeronutronic, 2 May 1962.

Project: Solrad IVB

Launch Date: 26 April 1962

Launch Area: PMR

Countdown Holds: 2

Launch No.: 20

Missile Type: SLV-1A

Serial No.: S-111

Result: Failure

PRIMARY OBJECTIVES

1. Inject the Solrad IVB satellite into a nominal 500 nautical mile circular orbit having an inclination of approximately 75.4 degrees.

2. Demonstrate the integrity of the various subsystems and techniques used for the flight.

MISSION SUMMARY

This was a series of firsts for this SLV-1 launch which used an improved Antares X-259 A1 motor.

1. First SLV-1 to be launched from PMR.

2. A 34-inch diameter heat shield installed on fourth stage.

The primary objectives were not achieved, due to lack of corrective moments about the third stage reaction control system.

One countdown hold was for a guidance timer that could not be reset. It was removed and a timer from S-113 was rewired and installed in S-111. The second hold was for a weak battery, which was replaced. A 28-volt battery had shorted leads and was replaced. A power control relay box for the third stage coast gain burned K6 contacts when 37 volts was shorted to ground during preflight checks.

PERFORMANCE

Control System

At third stage ignition (T+114.78 seconds) loss of attitude control occurred (simultaneously for all three vehicle axes) due to lack of third stage control response to guidance commands. Approximately 8 seconds later a decrease in relative velocity, in conjunction with a decay in altitude, was recorded. The vehicle impacted 270 nautical miles down range approximately 310 seconds after launch.

During third stage operation, reaction control systems provide vehicle stabilization and attitude control by means of reaction thrust motors that use 90 percent hydrogen peroxide as the propellant. These ON-OFF motors are arranged to produce moments about each of the three vehicle axes. They are actuated by guidance control signals. Nitrogen is used for pressurizing the system. Components of the third stage system are located in transition section C (third stage).

It is suspected that loss of third stage control during third stage burning resulted from the hydrogen peroxide passing through the supply solenoid valve and returning to the supply tank (ground support supply) when the system was pressurized for "burping". Under system pressurization, the hydrogen peroxide supply (fill) solenoid valves in the operational support building could open with a 300-psi differential across the valve ports, allowing the peroxide to return to the supply tank. Most of these recommended corrective actions have been implemented:

a) Reverse the solenoid-operated valves in the H_2O_2 lines.

b) Separate the N_2 servicing of the B and C sections so that B cannot act as an accumulator of C section and mask excessive pressure drops.

c) A thorough review with all launch personnel to assure accuracy and understanding.

d) A complete mock firing launch prior to each flight.

e) Three indications of GO on the peroxide system prior to launch, as follows:

1) Green light on the regulated N_2 pressure.

2) Satisfactory pressure reading on the peroxide transducer.

3) Correct pressure drop of unregulated N_2 at the time of system pressurization.

REFERENCE

"Scouts S-111 Final Flight Report," Vought Astronautics, 3-13000/2R-185. 18 June 1962, Unclassified.

Project: Classified

Launch Date: 23 May 1962

Launch Area: PMR

Countdown Holds: 0

Launch No.: 21

Missile Type: SLV-1A

Serial No.: S-112

Result: Failure

PRIMARY OBJECTIVE

Inject the payload into a 400 nautical mile circular orbit.

MISSION SUMMARY

The objective was not achieved due to premature separation of the second/third stage sections.

PERFORMANCE

First stage operation was entirely satisfactory, and integrity of the vehicle and its systems was partially demonstrated. A catastrophic failure, after approximately 72 seconds of normal flight, completely aborted the mission. This failure was attributed to the diaphragm in the C transition section backing out of the upper C adapter ring. This resulted in separation of the second stage/third stage plane.

Although it could not be determined how a diaphragm back-out could occur, it is known that the only prelaunch configuration discrepancy was in the C section diaphragm. The diaphragm was not properly pinned to the upper and lower C section adapter rings, thus leaving the diaphragm free to rotate. A design change was incorporated into vehicle ST-9 and subsequent vehicles to make the C section compatible with the improved Antares motor, but was not incorporated in the build-up procedures of the launch Wing. The old pin, which is 0.37 inch shorter than the new pin, was installed according to the then existing procedures. Assuming that the diaphragm could be rotated to a point of less than one thread engagement, then the diaphragm could not restrain any bending moment from possible thrust misalignment during the second stage burning phase.

This failure cause is the only one which is entirely consistent with the known telemetry information, and corrective action has been instituted by using the redesigned attachment pins for the C section diaphragm assembly and by updating assembly procedures.

REFERENCE

"Post Launch Report - Scout S-112," Chance Vought Corp., 16 July 1962, No. 3-13839/2R-16, Secret.

Project: Program 279

Launch Date: 31 May 1962

Launch Area: PMR Sunflare Pad

Countdown Holds: 2

Launch No.: 22

Missile Type: LV-1B

Serial No.: 102

Result: Success

PRIMARY OBJECTIVES

Classified

MISSION SUMMARY

This was the first launching of the Program 279 rocket (modified Blue Scout), and all primary and secondary mission objectives were achieved. The flight was highly successful. The vehicle was a three stage, unguided, spin-stabilized, rail-launched rocket. The payload weighed approximately 182 pounds.

Flight countdown time was 282 minutes. Total hold time of 88 minutes was for range clearance (train and ships in hazard area).

PERFORMANCE

Propulsion

The trajectory of this flight was slightly left (south) of the predicted track and below predicted trajectory. The pitchover rate was greater than predicted from 4 to 6 seconds after liftoff, but velocity and acceleration for the first 23 seconds of flight were slightly below predicted values, yielding a pitch attitude lower than predicted.

Recovery Vehicle

The spinup and decay times were very close to predicted, although the spin rate was 0.4 to 1.2 rps lower than anticipated.

Vibration

Lateral and longitudinal vibration levels of 40 to 50 g peak amplitude were noted approximately 1 second after heatshield jettisoning sequence. These vibration levels are being evaluated.

Telemetry

Telemetry system operated successfully until approximately 817 seconds. Just prior to termination of telemetry, the following indications were obtained:

a) Pitch and yaw gyro signals indicated changes in nutation rate.

b) Outer strain gauge signal became erratic, suggesting a pulling away from attach point and/or heating effects.

c) Antenna incident and reflected power signals indicated physical disabling of antenna. (Telemetry power supplies were steady until failure time, indicating this to be an antenna failure.)

d) Both RF links ceased abruptly and simultaneously.

The PMR radars for realtime plot of trajectory were not acquired at Pt. Mugu or San Nicholas tracking stations due to failure of liftoff tone communications.

The trajectory digital output from Point Arguello radar was not obtained due to timing system failure. Theodolite data (trajectory, velocity, acceleration, roll, pitch, and yaw) was lost after T+23 seconds because of timing system failure.

REFERENCE

"Program 279 Launch Report 2/102," VAFB, -0150, 31 May 1962, Secret.

Project: Program 279	Launch No.: 23
Launch Date: 24 July 1962	Missile Type: LV-1B
Launch Area: PMR-A	Serial No.: 101
Countdown Holds: 1	Result: Success

PRIMARY OBJECTIVES

Classified

MISSION SUMMARY

This was the second and final launching of Phase I vehicles (Program 279 rocket). The launch was highly successful in meeting all primary and secondary objectives. The LV-1B vehicle is a three stage, unguided, spin-stabilized, rail-launched rocket vehicle. The payload weighed approximately 188 pounds. All trajectory data indicate a trajectory which was low and to the left of nominal. Radar tabular data showed the vehicle velocity was approximately 1200 feet per second low and the altitude approximately 16 nautical miles low at third stage burnout. All impact points were within the allowable dispersion area.

A countdown hold of 28 minutes was called for range clearance. Flight countdown time was 179 minutes.

PERFORMANCE

Vehicle

At the approximate time of third stage ignition (T+84.56 seconds), both flight path angle and azimuth began to diverge more rapidly from the nominal; the flight path angle pitching down and the azimuth swinging left. Telemetry traces indicated a cyclic, oscillatory motion of both pitch and yaw rate, commencing at third stage ignition and damping out prior to third stage burnout.

No exact impact point can be determined due to absence of tracking data all the way to splash. Data were obtained, however, which yield an approximation of splash point.

Position data from TLM-18 telemetry tracking were obtained through T+1476 seconds, indicating an apogee altitude of 946 nautical miles.

The average apogee and impact point computed on a pure ballistic trajectory from data points obtained immediately after third stage burnout were 866.64 nautical miles, and 148.162 degrees west and 31.634 degrees north, respectively.

The vehicle spin rockets ignited as programmed at approximately 1 second after launch, accelerating the spin rate to a maximum of 2.76 rps, then decayed to 0.96 rps and aerodynamically increased to 3.47 rps. The spin rate was 3.00 rps at payload separation. The fin tip cant was set at 10 degrees, compared to 5.5 degrees on vehicle 2/102, to maintain the desired 3 rps spin rate.

Overall vehicle performance was satisfactory, although the first stage motor burned for 39.15 seconds compared to a nominal 37.3 seconds; the second stage motor burned 41.7 seconds compared to a nominal of 38.2 seconds; and the third stage motor burned 36.45 seconds compared to a nominal 40.4 seconds. There were changes in the pitch and yaw rates during third stage burning which are being analyzed for correlation with vehicle performance.

<u>Telemetry</u>

A slight difficulty was experienced in the telemetry system in playing back commutated telemetry data. The Range telemetry ground station could not maintain synchronization between the decommutation units and the signal. There were three factors contributing to this problem:

a) The commutator in the vehicle appeared to be changing speed.

b) The commutator duty cycle was higher than the Range was normally accustomed to.

c) During portions of the flight beginning sometime after third stage burnout there was a cyclic loss of signal, occurring at approximately the vehicle spin frequency (3 cps).

During telemetry coverage, a disturbance of the signal occurred for approximately 10 minutes, having maximum distortion of signal at T+9 minutes. This disturbance was cyclic in nature, occurring at a rate of 3.3 cps. The effect was first noticeable in the higher frequency channels and gradually progressed into the lower frequency channels. Data during the period of T+6 and 12 minutes were of little value, and data between T+5 and 6 and T+12 and 13 minutes were of doubtful value. Further analyses are being made.

REFERENCE

"Flight Test System 101 Final Test Report," Bendix System Division, 17 September 1962, BSC 35236, Secret.

Project: Classified

Launch Date: 23 August 1962

Launch Area: PMR Scout Pad

Countdown Holds: 6

Launch No.: 24

Missile Type: SLV-1A

Serial No.: S-117

Result: Success

MISSION

Mission objectives are classified.

SUMMARY

This was the third four-stage guided SLV-I vehicle launched from the Pacific Missile Range. The vehicle followed the predicted trajectory with such success that the observed variations were practically the same as the estimated accuracy of the acquired tracking data. Only small differences between the predicted and actual operation of the vehicle systems were noted.

The countdown holds were:

- 8-9-62 ---Range safety (railroad trains in launch area)
- 8-12-62 --Excessive drift in guidance system yaw displacement gyro
- 8-15-62 --Replace blockhouse console pressure switch which malfunctioned, indicating that "C" section yaw-right motor was inoperative
- 8-23-62 --Bad lendex switch connection
- 8-23-62 --Payload battery voltage dropped to 7 volts
- 8-23-62 --Power supply voltage in the blockhouse was low, resulting in non-operating "C" section pitch-up and yaw-right reaction control motors.

PERFORMANCE

Propulsion

All motors performed as predicted. Deviations in total impulse were:

Algol IIA	+1.79%
Castor XM-33E5	-0.08%
Antares ABL-X259	+0.77%
MG-18	+0.74%

Guidance System

Operated satisfactorily.

Control System

All control systems responded to guidance signals and altered vehicle attitude so as to nullify errors.

Airframe

No discrepancies noted.

Separation

Normal

Thermal Environment

All temperatures were normal except for Base A nozzle which was apparently caused by a thermistor location that was appreciably dissimilar from that required by existing procedures. The contractor will control the location of this nozzle thermistor on future flights.

Ignition and Destruct

Not required.

Telemetry

All data recovered.

REFERENCE

"Chance Vought Post Launch Report, Scout S-117," Report No. 3-13839/2R-20, dated 27 September 1962, secret.

Project: Space Probe

Launch Date: 31 Aug 1962

Launch Area: Wallops Island, Pad 3

Countdown Holds: 0

Launch No.: 25

Missile Type: SLV-1A

Serial No.: S-114

Result: Failure

PRIMARY OBJECTIVES

1. Transport the radiative heat-transfer payload along a precise trajectory.
2. Provide the boost necessary to obtain a reentry-type trajectory which would satisfy this program.

MISSION SUMMARY

Delayed third stage ignition initiated a sequence of events that precluded accomplishment of the primary mission. Limited payload data are available, but this anomaly as well as the information items listed below should be noted. Information is based on preliminary data.

This was the second five-stage SLV-1 vehicle utilized in radiative heat-transfer payloads and the first using the improved Aerojet Senior Algol IIA first-stage motor. The launch azimuth was 130.5 degrees true and the actual vehicle launch angle was 84.5 degrees.

The vehicle performance up to third stage ignition was satisfactory, although the trajectory was slightly high. Third stage ignition, programmed for ignition at 279 seconds, occurred at 329.96 seconds. This was approximately 1.2 to 2.0 seconds after spin motor ignition. Separation of third and fourth stages occurred when scheduled, as indicated by the transition section D accelerometer's peak acceleration of 21.5 g subsequent to separation (nominal was 11 g). Fifth stage ignition was approximately 2 seconds premature and thrusted for approximately 27 seconds, 16 seconds less than nominal.

PERFORMANCE

Propulsion

First Stage. The first stage motor web burn time was approximately 5 seconds shorter than predicted, with the relative velocity dropping approximately 100 fps below predicted at burnout. The burn time was 0.12 seconds shorter than predicted.

This was the first launch to utilize the Algol IIA motor (which replaces the Algol ID, Aerojet Senior, installed on SLV-1 vehicles S-110 and S-111). The Algol IIA rocket engine has the same type chamber as the Algol ID, but incorporates a higher performance propellant, higher propellant volumetric loading, a lightweight plastic nozzle, controlled pressure, and a computed grain-type igniter with insertable squibs.

The lightweight Fiberglas nozzle used on the Algol IIA rocket motor utilizes phenolic resin-impregnated Refrasil as an ablative heat sink. This heat sink virtually eliminates heating of the external surface of the first stage rocket motor during the burn and coast phases. As a result of this improvement, an insulation blanket around the nozzle is not required to protect equipment in Base A section.

Second Stage. The Castor motor total burn time was 42.78 seconds, as compared with a predicted time of 39.9 seconds. This motor has consistently shown a longer burn time than predicted.

Third Stage. Accelerometer data indicated that the third stage reached a peak acceleration of approximately 21.5 g, due to delayed third stage ignition and subsequent firing of the fourth stage motor.

The third stage motor (X-259-A-2) differs from the A-1 slightly in that less insulation is used. This permits a 27-pound increase in propellant.

Possible causes of late third stage ignition are:

1) Third stage ignition circuit failure.
2) Third stage ignition squib failure.
3) Short in the heat shield explosive bolt circuit which drained sufficient current from the third stage motor squib to preclude ignition until the short was removed at fourth stage spinup.
4) Improper attachment, or failure, of the heat shield disconnect plug lanyard.

Further analyses are being made and a subsequent report will be issued upon conclusion of the investigations.

Guidance and Control

The unregulated nitrogen pressure in transition B section appeared normal until activation of C section coast controls at approximately 318 seconds. At this time deadband limits are changed from 0.025 ± 10 percent to 0.004 ± 10 percent, increasing the duty cycle, and causing rapid decay of second stage nitrogen pressure. The third stage nitrogen pressure begins to decay somewhat prior to the time of C section burn control activation. The pressure continues to decrease at a nearly constant rate through 330 seconds of flight, at which time a momentary increase occurs, followed by a rapid decay in pressure due to retrorocket activation. Further analyses are being made of this phenomena.

Airframe

Analysis of the inside skin temperature of transition B section indicates a maximum of 350 degrees F at T+120 seconds. The previous maximum condition reached in this area was approximately 250°F, recorded on vehicles ST-6 and S-111.

Telemetry

A slight frequency shift on the pitch program channel during ground checkout was due to RF radiation from transition D section telemetry when the umbilicals were pulled. Further investigations are being performed on this situation. Other areas under analytical investigation are:

a) Fourth stage did not attain the desired spin rate of 170 ± 10 rpm. The fourth stage apparently had a spin rate of less than 25 rpm.

b) Data at fourth stage separation imply that the third stage pushed the fourth stage for 0.25 - 0.5 second and then tumbled the fourth stage.

c) Ejection of the heat shield was unconfirmed.

REFERENCE

"Scout S-114 Preliminary Flight Report," Vought Astronautics, 3-13000/2R-255, 18 September 1962, Unclassified.

Project: Classified

Launch Date: 21 Nov 1962

Launch Area: PMR-Complex A

Countdown Holds: 0

Launch No.: 26

Missile Type: LV-1B

Serial No.: 201

Result: Success

PRIMARY OBJECTIVES

All primary flight test objectives are directly related to payload performance, which is sensitive, and are therefore not included in this summary.

MISSION SUMMARY

The LV-1B was a three stage, unguided, spin-stabilized vehicle, which met all planned requirements and was within acceptable limits with the exception of spin motor separation. This anomaly has been corrected; however, no report as to the corrective action was issued.

PERFORMANCE

Propulsion

All stages operated within acceptable requirements.

Control System

All stages reacted as required.

Airframe

No structural failures occurred.

Ignition and Destruct

All stages ignited at proper times and the destruct system was not required.

Telemetry

All measurement sources were operative.

Launcher

No damage at launch.

REFERENCE

"6595th Aerospace Test Wing Report," 21 November 1962, ESSDV-63-0004, Secret.

Project: S-55B Satellite | Launch No.: 27

Launch Date: 16 December 1962 | Missile Type: SLV-1A

Launch Area: Wallops Island, Area 3 | Serial No.: S-115

Countdown Holds: 2 | Result: Success

PRIMARY OBJECTIVE

Inject the S-55B micrometeroid satellite in an elliptical orbit about the earth.

MISSION SUMMARY

The primary mission objective was accomplished. The orbital inclination was 52 degrees, the apogee was 637.9 n mi and the perigee was 404.5 n mi. Injection occurred at approximately T+6[illegible]2.5 seconds (0944:36.6 EST).

Countdown holds were due to weather (low ceiling) and for removal of insulating Strux blanket from the X-259 motor.

PERFORMANCE

Propulsion

All motors approximated predicted performance.

Guidance System

All guidance timer functions occurred as predicted.

Control System

The first, second, and third stage control systems reacted as required to guidance system commands. During initial second stage thrusting, both yaw motors functioned continuously for approximately 21 seconds. An unusual phenomenon occurred in that excitation of the second bending mode (20 cps) of the vehicle was excited by the response of individual jets to intermittent command signals sensed by the rate gyro responding to the bending mode. The bending mode was initiated by a sharp, short-duration ignition impulse of the second stage booster. This initial excitation of the bending mode caused both jets to fire within 0. 3 seconds of ignition. The right jet then fired twice, each time exciting the bending mode a little more. The command signal apparently became large enough to fire the left jet, which then increased the oscillation amplitude and was then sufficient to cause both jets to fire alternately.

This unusual phenomenon requires an unlikely combination of circumstances: very low thrust misalignment on one axis, proper phasing of ignition transients and motor response, position within the deadband, and phasing of bending oscillations relative to the rigid body motion. Studies are being performed to determine corrective action required to prevent a recurrence.

Airframe

No structural failures occurred.

Telemetry

Very high percentage of vehicle data recovered.

Ignition and Destruct

All stages ignited at proper time and destruct system was not required.

Facility and Launcher

No unusual damage occurred.

REFERENCE

"S-115 Final Flight Report," Chance Vought Corp., 26 February 1963, 3-13000/3R-49, Unclassified.

Project: Classified

Launch Date: 18 December 1962

Launch Area: PMR Complex A

Countdown Holds: 0

Launch No.: 28

Missile Type: LV-1B

Serial No.: 211

Result: Success

PRIMARY OBJECTIVES

All primary flight test objectives are directly related to payload performance, which is sensitive, and therefore are not included in this summary.

MISSION SUMMARY

The vehicle trajectory was within acceptable limits. Discrepancies are noted below.

PERFORMANCE

Propulsion

a) First stage motor performance was nominal.

b) Second stage motor burned 2.36 seconds longer than predicted.

c) Third stage ignition occurred 4.22 seconds late and burned 2.82 seconds longer than predicted. The additional burning time overlapped the 2-second delay for ignition of de-spin rockets and permitted de-spinning to occur while the third stage was still thrusting. Telemetry records indicate that at T+102.54 seconds, 24.64 seconds after third stage ignition, a decay in motor pressure to 100 psi occurred. This pressure stabilized at 250 psi, approximately one-half of its normal level.

Control System

First, second, and third stage systems reacted as required.

Airframe

No structural failures occurred.

Ignition and Destruct

All stages ignited at proper time and destruct system was not required.

Telemetry

Very high percentage of data recovered

Facility and Launcher

No damage to launch

REFERENCE

"Program 661-A Evaluation Report," 6595th Aerospace Test Wing, 28 February 1963, Confidential.

Project: Transit 5A

Launch Date: 18 December 1962

Launch Area: PMR, Scout Pad

Countdown Holds: 1

Launch No.: 29

Missile Type: SLV-1A

Serial No.: S-118

Result: Success

PRIMARY OBJECTIVE

Inject the payload into a near-circular orbit about the earth.

MISSION SUMMARY

The primary test objective was accomplished. Injection occurred at approximately T+670.39 seconds (0136:56 GMT) at an altitude of 397.2 n mi, an azimuth of 184.63 degrees, and a velocity of 24,598 fps. Orbital inclination was 90.62 degrees.

Range support problems accounted for the 5-hr hold time. Most of this time was made up and launch was approximately 10 min later than the scheduled T-0 time.

PERFORMANCE

Propulsion

All motors approximated predicted performance.

Guidance System

All guidance timer functions occurred as predicted.

Control System

First, second, and third stage control systems reacted as required to guidance system commands.

Airframe

No structural failures occurred.

Telemetry

Very high percentage of data recovered.

Ignition and Destruct

All stages ignited at proper time and destruct system was not required.

Facility and Launcher

No damage at launch.

REFERENCE

"S-118 Final Flight Report," Chance Vought Corp., 12 February 1963, 3-13000/3R-39, Unclassified

Project:

Launch No.: 30

Launch Date: 1 Feb 1963

Missile Type: LV-1B

Launch Area: PMR

Serial No.: 202

Countdown Holds:

Result: Success

PERFORMANCE

All missile subsystems operated satisfactorily.

Project: Classified

Launch Date: 19 Feb 1963

Launch Area: PMR Scout Pad

Countdown Holds: 0

Launch No.: 31

Missile Type: SLV-1A

Serial No.: S-126

Result: Success

MISSION

Mission objectives are classified.

SUMMARY

The primary objective of this four-stage vehicle was accomplished with only minor deviations from the predicted. For further details, see referenced document acquired through the Scout Office.

PERFORMANCE

Propulsion

The Algol IIA motor total impulse (-0.275%) was near nominal, but the motor exhibited an over-performance during the majority of the operation, which resulted in an altitude and velocity error. Total impulse for the second stage Castor (XM-33E5) motor was -0.89%. The third stage motor was outside its 3-sigma limit for both burn time and total impulse. Burning time for the first, second, and third stage motors was -1.26 seconds, +3.40 seconds, and +4.64 seconds from predicted. The fourth stage MG-18 motor was approximately 1.20% low in total impulse.

Guidance System

All guidance timer functions (except the fifth pitch program step) occurred as predicted, with a maximum deviation of -0.01 to +0.10 seconds. Proper attitude was maintained throughout flight.

Control System

The recurring problem of the control system exciting the basic body-bending frequency was observed during second and third stage operations. This caused increased control system operation, but did not interfere with the system's control of the vehicle.

Airframe

No structural failures occurred.

Thermal Environment

Transition ambient temperatures did not exceed specified limits.

Ignition and Destruct System

All stages ignited at the proper time and the destruct system was not required.

Telemetry

A good percentage of vehicle data were recovered.

Launcher

No damage at launch site.

REFERENCE

"Chance Vought Post Launch Report, Scout S-126," Report No. 3-13839/3R-8, dated 11 April 1963, secret.

Project: Classified

Launch Date: 13 March 1963

Launch Area: PMR Complex A

Countdown Holds: 0

Launch No.: 32

Missile Type: LV-1B

Serial No.: 203

Result: Success

PRIMARY OBJECTIVES

All primary flight test objectives are directly related to payload performance, which is sensitive, and therefore are not included in this summary.

MISSION SUMMARY

The LV-1B was a three stage, unguided, spin-stabilized vehicle which met all planned requirements and was within acceptable limits.

PERFORMANCE

Propulsion

All stages operated within nominal requirements.

Control System

All stages of the control system reacted as required.

Airframe

No structural failures occured.

Ignition and Destruct

All stages ignited at proper times and the destruct system was not required.

Telemetry

All measurement sources were operative.

Launcher

No damage at launch.

REFERENCE

"6595th Aerospace Test Wing Evaluation Report," 13 March 1963, ESN-3-63-0046, Secret.

Project: Program 435

Launch Date: 5 April 1963

Launch Area: PMR, Scout Pad

Countdown Holds: 2

Launch No: 33

Missile Type: SLV-1A

Serial No.: 119

Result: Failure

PRIMARY OBJECTIVE

Inject the payload into orbit about the earth.

MISSION SUMMARY

The vehicle closely followed the predicted trajectory and all systems appeared normal through third-stage burnout. At approximately 445 seconds flight time (during third stage coast) a malfunction in the reaction control system occurred which caused loss of the third stage hydrogen peroxide supply and resulted in vehicle instability which precluded the accomplishment of the primary mission. A vehicle telemetry malfunction occurred at approximately 345 seconds, with almost total telemetry loss.

The countdown was held for 30 minutes for non-support by Range during Task 2 first-motion checks, and for 60 minutes due to malfunction of FPS radar. This hold was continued longer due to trains in the area.

A series of tests will be conducted on a vehicle C section to determine the effects of various conditions on systems qualification and assurance from environment, life, over-pressure, vibration, etc.

PERFORMANCE

Control System

The telemetry malfunction precludes a thorough analysis. However, available data does provide a basis to conclude that the failure of the third stage reaction control system was specifically due to (1) a 2-pound thrust pitch motor that failed to respond to a pitch command at 444.25 seconds, and (2) loss of hydrogen peroxide between 445 and 511 seconds, probably due to a rupture in the system (possibly the peroxide line to the 2-pound motor).

Telemetry

The vehicle telemetry transmitter malfunction is not associated with the vehicle failure. This malfunction may have been the result of improper pre-flight operation of telemetry (coaxial cable from the antenna was disconnected at the transmitter). There is a history of three previous failures of the Teledynamics TDD-1009B transmitter operating without a normal load. In two cases the coaxial transmission line was disconnected at the antenna, and in one case the coaxial cable was disconnected at the transmitter. In each case a complete failure was noted within 2 minutes or less.

REFERENCE

"Post Flight Evaluation Report," SSD, 9 May 1963, Unclassified.

Project: Program 435	Launch No.: 34
Launch Date: 25 April 1963	Missile Type: SLV-1A
Launch Area: PMR Complex D	Serial No.: 121
Countdown Holds: 2	Result: Failure

PRIMARY OBJECTIVE

Inject the payload into orbit about the earth.

MISSION SUMMARY

The primary test objective was not accomplished due to an abrupt termination of thrust, loss of third stage attitude control, and loss of C transition section telemetry at T+145.46 seconds, approximately 8 seconds prior to third stage burnout.

PERFORMANCE

Propulsion

Stage 1 and 2 performance was nominal. During third stage burn, 8.45 seconds after ignition, a shock of 1-2 g was sensed by the longitudinal accelerometer. Telemetry data indicated an isolated and practically instantaneous event which terminated motor thrust, tumbled the vehicle, and cut off all information from the C and D sections.

It is suspected that premature activation of the destruct system, due to an electrical short circuit in the automatic destruct system, was the cause of third stage failure. Although a short circuit could occur in several locations, a highly suspect point is connector J318. This connector contains three combinations of adjacent pins, any of which will initiate destruct if shorted together. Corrective action will consist of (1) analyses and tests to determine the vulnerability of the SLV-1A vehicle destruct system to short circuits, spurious radiation, and mechanical failure of relays, (2) modification of the electrical system to assure that the destruct activation circuits and powered circuits are not located on adjacent connector pins, and (3) weather proofing of connectors and other sensitive components that are exposed to rain and mist during vehicle erection, countdown and launch.

Guidance and Control

All guidance timer functions occurred as programmed until the failure of the third stage.

Control System

An unknown malfunction occurred between T+40 to T+50 seconds in the first stage control system but was not associated with the failure which occurred with the third stage. The pitch fins went to full-down position for 6 seconds, then resumed normal operation. The trajectory was unaffected.

Airframe

No structural failures of stages 1 and 2. Stage 3 destructed at T+145 seconds.

Telemetry

Small discrepancies in transmitted data and the reported "on" time of the destruct transmitter.

Ignition and Destruct

Ignition of the first three stages was normal. Premature activation of the destruct system, possibly due to a short circuit in the destruct system of the third stage, destroyed vehicle.

Facility and Launcher

Nominal damage.

REFERENCE

"Post Flight Evaluation Report," SSD, 9 May 1963, Unclassified.

Project:	Launch No.: 35
Launch Date: 17 May 1963	Missile Type: LV-1B
Launch Area: PMR	Serial No.: 204
Countdown Holds:	Result: Success

PERFORMANCE

All missile subsystems operated satisfactorily.

Project: SNAP-10A Probe

Launch No.: 36

Launch Date: 22 May 1963

Missile Type: SLV-1A

Launch Area: Wallops Island

Serial No.: S-116

Countdown Holds: Many weather holds

Result: Success

MISSION

The primary test objective of this four-stage vehicle was to provide the required boost for the SNAP-10A safety flight test payload by flying a re-entry trajectory.

SUMMARY

The vehicle approximated the predicted trajectory and all vehicle systems operated normally, resulting in the payload being closely subjected to the planned re-entry conditions. Although no transmission was reported from the payload after onset of the re-entry blackout phenomena and recovery of the payload was not accomplished, optical coverage was considered satisfactory and the overall mission was successfully completed.

PERFORMANCE

Propulsion

All motors approximated predicted performance. Deviations from predicted burning time were:

First stage Algol IIA	-1.34 sec
Second stage Castor XM-33E5	+4.48 sec
Third stage Antares X259-A3	+1.17 sec

Guidance System

All guidance timer functions through initiation of retro occurred as predicted, with a maximum deviation of +0.08 to -0.06 seconds.

Control System

All control systems reacted to guidance system commands as required.

Airframe

No structural failures occurred.

Thermal Environment

Measured temperatures did not exceed specified limits.

Telemetry

A very high percentage of vehicle data were recovered. Usable data were obtained to 261 seconds of flight time.

Ignition and Destruct System

All stages ignited at the programmed times and the destruct system was not required.

Launcher

Minor damage was sustained by the launcher.

REFERENCE

"Chance Vought Final Flight Report, Scout S-116," Report No. 3-30000/3R-170, dated 18 July 1963, unclassified.

Project: Classified	Launch No.: 37
Launch Date: 16 June 1963	Missile Type: SLV-1A
Launch Area: PMR Scout Pad	Serial No.: S-120
Countdown Holds: 1	Result: Success

MISSION

The primary test objective of this four-stage vehicle was to provide the required boost and trajectory to orbit a special payload.

SUMMARY

The vehicle closely followed the predicted trajectory and all vehicle systems operated normally, resulting in the payload being placed in a precise orbit about the earth.

The one countdown hold (1 minute) was for confirmation of H_2O_2 pressure.

PERFORMANCE

Propulsion

All motors approximated predicted performance. Deviations from predicted burning time were:

First stage Algol IIA	-2.12 sec
Second stage Castor XM-33E5	+0.295 sec
Third stage Antares X259-A3	+0.083 sec

Guidance System

All guidance timer functions through initiation of retro occurred as predicted, with maximum deviations ranging from -0.04 to -0.11 seconds.

Telemetry

A very high percentage of vehicle data were recovered. Usable data were obtained to 648 seconds of flight time.

Ignition and Destruct Systems

All stages ignited at the programmed times and the destruct system was not required.

Launcher

No damage.

REFERENCE

"Chance Vought Preliminary Flight Report, Scout S-120," Report No. 3-30000/3R-165, dated 2 July 1963, Unclassified.

Project: Research Satellite

Launch No.: 38

Launch Date: 28 June 1963

Missile Type: SLV-1A

Launch Area: Wallops Island

Serial No.: S-113

Countdown Holds: 4

Result: Success

MISSION

The primary test objective of this four-stage SLV-1A vehicle was to provide the required boost and trajectory to place the Research Satellite for Geophysics in the specified orbit around the earth.

SUMMARY

The vehicle closely followed the predicted trajectory and all vehicle systems operated normally, resulting in the payload being placed in a precise orbit about the earth.

The countdown holds were:

1) Open circuit on the arming console during the negative battery to negative monitor check, due to lack of contact between pin and socket in CVC plug 356 which was installed by launch site personnel.
2) Reset "C" regulator when pressure would not exceed 450 psi (nominal 470 psi).
3) Three-minute hold to complete ignition battery heater cycle.
4) Connection of bungee cord to umbilical cable.

PERFORMANCE

Propulsion

This was the first SLV-1A vehicle to utilize the Altair X-258 rocket motor - prior vehicles have used the X-248 motor. All motors approximated predicted performance, with the following burn time deviations:

First stage Algol IIA	-0.07 sec
Second stage Castor XM-33E5	1.61 sec
Third stage Antares X-259-A3	0.55 sec
Fourth stage Altair X-258	Not available

Guidance

All guidance timer functions were accomplished.

Control Systems

All control systems reacted as required to guidance system commands.

Airframe

No structural failures occurred.

Thermal Environment

Not available at this time.

Telemetry

Usable data obtained beyond 517 seconds of flight time.

Command Destruct

Not required.

Launcher

Minor damage -- some replacement of cables.

REFERENCE

"Chance Vought Preliminary Flight Report, Scout S-113," Report No. 3-30000/3R-181, dated 23 July 1963, Unclassified.

Project: Thermal Protective Materials Experiment

Launch No.: 39

Launch Date: 20 July 1963

Missile Type: SLV-1A

Launch Area: Wallops Island

Serial No.: S-110

Countdown Holds: 3

Result: Failure

MISSION OBJECTIVE

The primary objective of this five-stage guided vehicle was to provide the required boost and trajectory for the Thermal Protective Materials Experiment.

SUMMARY

The three countdown holds were for (1) Mod-II radar problems and a ship in the first-stage impact area, (2) confirmation of down-range optics systems, and (3) payload ignition battery voltage went to zero as the arming console short switch was activated. This was a normal condition.

Catastrophic failure at approximately T+4 seconds after first stage ignition prevented the accomplishment of the primary test objective.

Failure Isolation

At approximately T+2.62 seconds after first stage ignition, smoke began to billow out of base section A just above the tower-side fin on the first stage. At approximately T+4 seconds the destruct charge on the third stage X-259 motor ignited, followed immediately by ignition of the second stage Castor destruct charge. The first stage Algol IIA motor destruct charge did not ignite. While the first stage motor continued to burn, the upper three stages broke off at transition section C and fell back to the pad near the launcher. The first stage, with the second stage Castor motor still attached, started to tumble and roll and finally impacted in a marsh approximately 0.9 miles north-northwest of the launcher. Telemetry transmission ceased at T+21 seconds, which is assumed to be the time at which the upper stages impacted on the pad. This flight abortion of vehicle S-110 appears to have been caused by failure of the nozzle on the Algol IIA motor.

From the pieces of nozzle found on the pad, the motor vendor personnel indicated that the most probable cause of the failure was a possible defect of the nozzle between the plies of the fiberglass forward insert or the gap between the insert and the graphite throat, which eroded and allowed the flame to cut through the steel closure and undercut adjacent structures.

PERFORMANCE

Propulsion

First-stage chamber pressure and acceleration were normal for approximately the first 4 seconds.

Guidance

The first commanded pitch rate change occurred within 0.05 seconds of programmed time.

Control Systems

The control fins did not appear to respond to the guidance commands after 0.6 seconds. Three possible failures could cause this malfunction: (1) loss of pitch and yaw-roll servo amplifier input, (2) telemetry not recording the actual fin deflections, and (3) mechanical restraint of fin tip-jet vane assemblies. The primary cause appears to have been caused by the fire and intense heat resulting from the failure of the first stage Algol IIA nozzle.

Airframe

Except for the first-stage nozzle, no structural failures occurred until after the destruct system ignited.

Telemetry

The telemetry system operated until impact (T+21 seconds).

Thermal Environment

Base A nozzle temperature started to increase at approximately T+3 seconds, then the circuit "opened".

Command-Destruct

Command-destruct was initiated at T+10 seconds: however, second and third stage destruct systems had been ignited (probably by an electrical short-circuit in Base A at approximately T+4 seconds. First stage did not destruct.

Launcher

The launcher area sustained abnormal damage from burning propellant falling on the electrical cables running from the tower to the terminal building.

REFERENCE

"Chance Vought Preliminary Flight Report, Scout S-110," Report No. 3-30000/3R-203, dated 7 August 1963, Unclassified.

GLOSSARY

ABL	Allegany Ballistics Laboratory
ADF	Aeronutronics Division, Ford
AFB	Air Force Base
AFSWC	Air Force Special Weapons Command
AMR	Atlantic Missile Range
cps	cycle per second
ccw	counter clockwise
cw	clockwise
FM	frequency modulation
fps	feet per second
g	earth gravitational constant
GMT	Greenwich mean time
I_{sl}	total impulse, sea level, lb/sec
I_v	total impulse, vacuum, lb/sec
NASA	National Aeronautics and Space Administration
n mi	nautical mile
NOTS	Naval Ordnance Test Station
MIG	miniature integrating gyro
Mod	modification
psia	pound per square inch, absolute
psig	pound per square inch, gage
PBAA	polybutadiene-acrylic acid

PMR	Pacific Missile Range
R&D	research and development
rev/sec	revolution per second
RF	radio frequency
rpm	revolution per minute
RSO	Range Safety Officer
sec	second (time)
SLV	Standard Launch Vehicle
T	temperature, ^{o}F
T-time	T = 0 is time of initiating firing circuit
t_b	web burning time, sec
t_f	total burning time, sec
W_f	expended motor weight, lb
W_m	loaded motor weight, lb
W_p	propellant weight, lb
W_t	total weight, lb
$\int_{to}^{t_f} pdt$	total pressure time integral, psia/sec

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