DEVELOPMENT PLAN FOR ABLE 3-4

(Earth Satellite, Lunar Satellite, Deep Space Probe)

DOWNGRADE TO:
~~SECRET ON~~ ________
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1 June 1959

REVIEW DRAFT

CONTENTS

1. INTRODUCTION

This development plan describes an earth satellite, a lunar satellite, and a deep space probe. The earth satellite (Able-3), a Thor-boosted vehicle, will be launch about 7 August 1959. This satellite will be used to study the space environment about the earth and to prove out the payload and vehicle configuration. The lunar satellite (Able-4 Atlas), to be launched about 4 September 1959, will be an Atlas-boosted vehicle which will provide long-term data on the characteristics of space between the earth and the moon and contribute to our knowledge of the moon. The deep space probe (Able-4 Thor), to be launched about 4 November, will be a Thor-boosted vehicle that will be sent into the orbit of Venus. The experiments conducted on this probe will extend our knowledge of the inner solar system. All of these flights will establish basic parameters relating to space communication and guidance for future space missions.

II. BACKGROUND INFORMATION

Early in 1958, the Air Force Ballistic Missile Division (BMD) in its Project Able developed an Advanced Re-entry Test Vehicle (ARTV) for the purpose of testing ablating nose cones at the ICBM range of 5500 nautical miles. The two-stage ARTV utilized the Thor ballistic missile for the first stage and a second-stage liquid propulsion system which was an outgrowth of that developed for the Vanguard program. Three flight tests were conducted, two of which were highly successful.

On 27 March 1958, the Advanced Research Projects Agency (ARPA) directed AFBMD to proceed with a program of three lunar probe vehicles, using a third stage on the Able vehicle. This program was designated Project Able-1. Three lunar vehicles were launched, one each in August, October, and November 1958. Two of these flights provided much information concerning space flight and the space environment.

In late 1958, the National Aeronautic and Space Agency (NASA) directed AFBMD to proceed with two additional space programs, Able-3 (two earth satellites) and Able-4 (two interplanetary probes). The Able-3 program originally called for launchings in February and April of 1959. AFBMD/STL recommended that the February Able-3 launching be delayed until September 1959. NASA concurred with this decision. Both Able-4 launchings were scheduled for June 1959.

In April 1959, the scheduled Able-3 launching was postponed to 8 May. In early May, the Able-3 program was delayed further. Since the Able-4 program was predicated upon the Able-3 launch and since the success of the Able-4 Venus probes depended upon launching the vehicles on certain days, the flights were delayed and the missions changed.

At conferences held in May among AFBMD, NASA, and STL, the following missions and tentative schedules were assigned for the Able-3 and Able-4 program. The Able-3 (earth satellite) mission remained the same, but with launch date in early August. The Able-4 Atlas will be a lunar satellite, to be launched in September. The Able-4 Thor will be a deep space probe, to be launched about the middle of November. The second Able-3 missile, whose launch date was originally delayed until September, has not yet been rescheduled.

These launch dates are flexible and will be changed if the probability of success of the assigned missions is thereby increased.

III. DESIGN CHARACTERISTICS

A. Able-3 (Thor-Boosted Earth Satellite)

1. Objectives

The primary objectives of the Able-3 project are to place an instrumented payload into an elliptical orbit about the earth in order to make scientific measurements of the environment encountered and to demonstrate satisfactory operation of the vehicle, payload, and supporting ground stations to be used in the Able-4 space probes. These objectives include:

a. Receiving telemetered measurements of temperature, micrometeorite flux and momentum, magnetic field, radiation flux and species, whistler mode propagation, and position of the satellite relative to the earth by means of facsimile television.

b. Demonstrating satisfactory operation of the payload equipment: the primary transmitter, digital telemetry system, solar power supply system, doppler and command system, and the experiment sensors.

c. Demonstrating satisfactory open loop operation of the second stage guidance receiver in conjunction with the space guidance system ground station.

d. Demonstrating satisfactory operation of the special tracking stations to be used for the Able-4 launchings.

e. Determining the effect of vernier rocket firing on terminal vehicle dynamics.

The secondary objectives of the Able-3 program include determining the propagation characteristics of the ionosphere and troposphere, evaluating portions of the system which have either already operated reliably on previous programs or which will not be used in the Able-4 program, and evaluating first stage airframe, propulsion, and modified autopilot control systems.

2. Technical Summary

The first stage will consist of a Thor without its normal inertial guidance system and with a modified autopilot control system. This stage will boost the vehicle to an inertial velocity of 15,911 ft/sec at burnout (T + 159.4 seconds). First-second staging occurs at T + 161.4 seconds.

The second stage is essentially the same as that used on the Able-1 firings. It consists of an Aerojet 10-101A engine, autopilot control, instrumentation, and spin rockets, and will boost the vehicle to a velocity of 24,293 ft/sec at burnout (T + 274.8 seconds). Stage 2 burnout activates a 2-second timer which ignites the spin rockets at T + 276.8 seconds and a 2.7-second timer which causes second-third staging at T + 277.5 seconds.

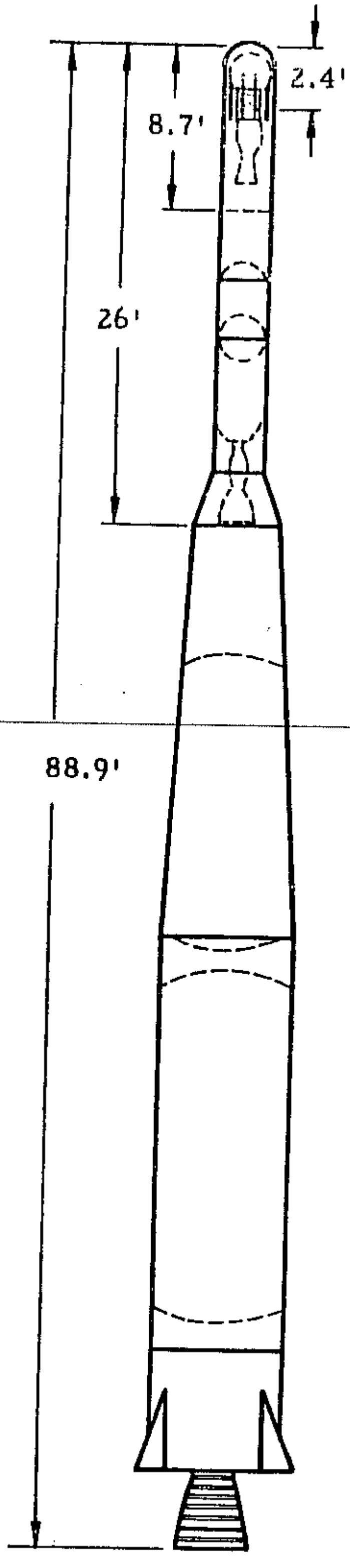
The third stage also will be the same as used for Able-1, an Allegheny Ballistic Laboratory 248 A3 solid propellant engine, which will boost the vehicle to 33,683 ft/sec at burnout (T + 314.7 seconds). Third-fourth staging occurs at T + 485 seconds, initiated by a 5-minute timer activated at T + 185 seconds when the third-fourth nose fairing is jettisoned.

The fourth stage will contain payload electronics, two vernier rockets, and an injection rocket which will be fired to insure adequate lifetime if necessary. The injection rocket, an ARC 1KS 420 solid propellant rocket, will provide approximately 500 pound-seconds of impulse. Power for the payload electronics will be provided by a solar cell system with the cells located on four paddles whose arms will fold out from the payload at second stage burnout. The fourth stage will weigh about 145 pounds.

The vehicle is shown in Figure 1.

It is proposed to fly the basic instrumentation in an elliptical orbit with an apogee of over 19,000 nautical miles and a perigee sufficiently high to insure a lifetime of over a year. Present range safety limitations lead to orbits having a maximum north latitude excursion of about 48 degrees.

The 19,000 nautical mile elliptical orbit permits a payload weight of 145 pounds. Table 1 shows the present allocation of weight in the payload.



FOURTH STAGE

ARC 1KS420 Injection Rocket	
Thrust at Altitude	434 lb
Vacuum, I_{sp}	248 lb-sec/lb
Total Impulse	498 lb-sec
Burnout Acceleration	3.3 g

THIRD STAGE

ABL-248 A3 Engine	
Thrust at Altitude	3150 lb
Vacuum, I_{sp}	250.5 lb-sec/lb
Total Impulse	116,400 lb-sec
Burnout Acceleration	16 g

SECOND STAGE

AGC 10-101A Engine	
Thrust at Altitude	7664 lb
Thrust (mass burnout)	7327 lb
Vacuum, I_{sp} (nominal)	271 lb-sec/lb
Total Impulse (nominal)	8.69×10^5 lb/sec
Burnout Acceleration	4.54 g

FIRST STAGE - THOR

Rocketdyne MB-1 Engine	
Sea Level Thrust	153,248 lb
Sea Level, I_{sp} (nominal)	248.7 lb-sec/lb
Vacuum, I_{sp} (nominal)	289.7 lb-sec/lb
Total Impulse (nominal)	2.67×10^7 lb-sec
Burnout Acceleration	12.5 g

Figure 1. Outline and Performance Parameters of Able-3 Test Vehicle.

Table 1. Able-3 Payload Weight.

Item	Weight (lb)
Structure	28.4
Injection Rocket	5.3
Vernier Rockets	2.0
Temperature Control Paint	1.0
Solar Cells	13.6
Receiver and Control	9.7
Transmitter System	10.8
Telemetry System	15.1
Converters (including heat sink)	6.5
Batteries	19.6
Accelerometer	2.0
Wire, Fasteners, Miscellaneous	8.0
Experiments	19.6
TOTAL	141.6

The instrumentation proposed for inclusion in the payload is designed to provide as much knowledge as possible of the space environment about the earth. The experiments are outlined in Table 2. They include a micrometeorite detector which will measure the momentum spectrum of the particles and, with the position indicator, enable a rough determination of their direction. The magnetometer experiment is designed, together with the position indicator, to give the magnitude and direction of the local magnetic field. A number of experiments designed to measure the quantity, the kind, and the energy of the radiation striking the vehicle are included. The local dielectric current is determined from the change in doppler shift between the two telemetry frequencies. In addition, propagation through the troposphere and ionosphere as a function of angle above the horizon will be measured at the basic frequencies which have been considered for communication satellite purposes.

Table 2. Able-3 Basic Instrumentation Package.

Experiment	Quantity Measured	Importance	Weight (lb)	Power (mw)	To Be Supplied By
1. Ion Chamber and Geiger-Muller	Radiation level in earth-moon space and mean specific ionization per particle to 1 per cent	Radio propagation, solar physics, cosmology theory, magnetic modulation of cosmic ray intensity	2.1	140	U. of Minnesota
2. Proportional Radiation Counter	Intensity distribution		5.3	190	U. of Chicago
3. Scintillation Counter	Radiation flux in the space about the earth		4.8	320	STL
4. Magnetic Field and Vehicle Aspect using Search Coil Magnetometer	Field strength; component perpendicular to vehicle spin axis	Radio propagation, magnetic fields of earth, electric currents in space	1.6	50	STL
5. Flux Gate Magnetometer	Field strength; component along spin axis		2.2	180	STL
6. Meteorites and Interplanetary MM Spectrograph	Intensity and momentum distribution	Nighttime E layer, effect on re-entry problem, cosmology theory	0.9	70	AFCRC
7. Very Low Frequency Monitor	Magnetically guided radiation at very low frequencies	Solar activity; electron density	0.6	86	Stanford
8. Temperatures	Internal payload temperatures and solar cell paddle temperatures	Effect of space environment on solid propellants and instrumentation	0.2	1	STL
9. TV	Images of the earth from space	Cosmology theory and terrestrial weather prediction	1.9	230	NRL/STL

The power supply consists of solar cells which charge storage batteries to provide for continuous operation of the telemetry and intermittent command operation of the repeaters. The storage batteries have sufficient capacity to provide a duty cycle of more than 50 per cent with the 5-watt transmitter and of about 1 per cent with the 150-watt transmitter, and also to carry on operations through the eclipse of the satellite by the earth.

Guidance and tracking is accomplished by the doppler command transponder developed for the Able-1 probe. It provides a shutoff command to the second stage and commands to the vernier rockets and the injection rocket. Tracking with suitable ground equipment, it measures the range and radial velocity of the fourth stage.

3. Trajectory Considerations

a. Selection of Trajectory

The Able-3 orbit has been selected to optimize the information gathered from the experimental equipment. It is highly elliptical and inclined by about 45 degrees with respect to the equator. This will result in the orbit precessing by about 90 degrees during its nominal lifetime of a year. Near the end of the year, apogee will occur almost at the equator. This motion will substantially increase the amount and kind of information gathered.

b. Free Flight Trajectory

This section provides free flight trajectory information for an Able-3 vehicle launch.

The nominal orbit has an apogee of 19,000 nautical miles above the earth's surface, a perigee of 145 nautical miles, a period of 10 hours and 20 minutes, and an orbital inclination of 47.7 degrees. The vehicle is to be launched from Stand 17A (latitude 28.44687°N, longitude 80.56517°W) at an azimuth of 48 degrees true. This azimuth is compatible with range safety requirements and results in a satisfactory orbital inclination angle.

At Stage 3 burnout, the velocity vector makes an angle of 85.8 degrees from the local vertical. This angle represents the best compromise between apogee and perigee altitudes for a given total payload weight. A smaller angle will result in a greater apogee, but will reduce the perigee while at the same time increasing the sensitivity of perigee altitude to propulsion and control system dispersions. The trajectory selected will result in a nominal perigee of 145 nautical miles without firing the fourth stage injection rocket. This perigee will result in an orbital lifetime of over one year.

The fourth stage injection rocket, a 1KS420 rocket weighing five pounds and providing an impulse of 498 pound-seconds, can be fired to insure an adequate lifetime. This results in a vectorial velocity increment of 119.7 ft/sec. The purpose of this rocket is to increase the perigee (nominally by 198 nautical miles). For the nominal flight, the perigee without rocket firing is satisfactory. However, in the case of a malfunction, the rocket can mean the difference between no lifetime and a satisfactory lifetime.

Figure 2 shows the projection of the nominal vehicle on the earth for each of the first four days of orbit. Positions of apogee and perigee are indicated.

4. Vehicle Characteristics

a. Description of Vehicle

The Able-3 vehicle consists of three propulsion stages plus a terminal stage and a nose fairing. The first stage is the standard Thor ballistic missile less the guidance system and nose cone. This vehicle has been proven in the Thor flight test program. The second stage is the Aerojet General Corporation AJ 10-101A liquid propellant rocket used in the Able-1 program plus a control compartment designed by STL. This propulsion system is a modified version of the AJ 10-40 system which was proven in the Project Able ARTV flights. The third stage is an Allegheny Ballistic Laboratory X-248-A3 solid propellant rocket engine used in the Able-1 program. This unit was originally developed for the advanced Vanguard program. The

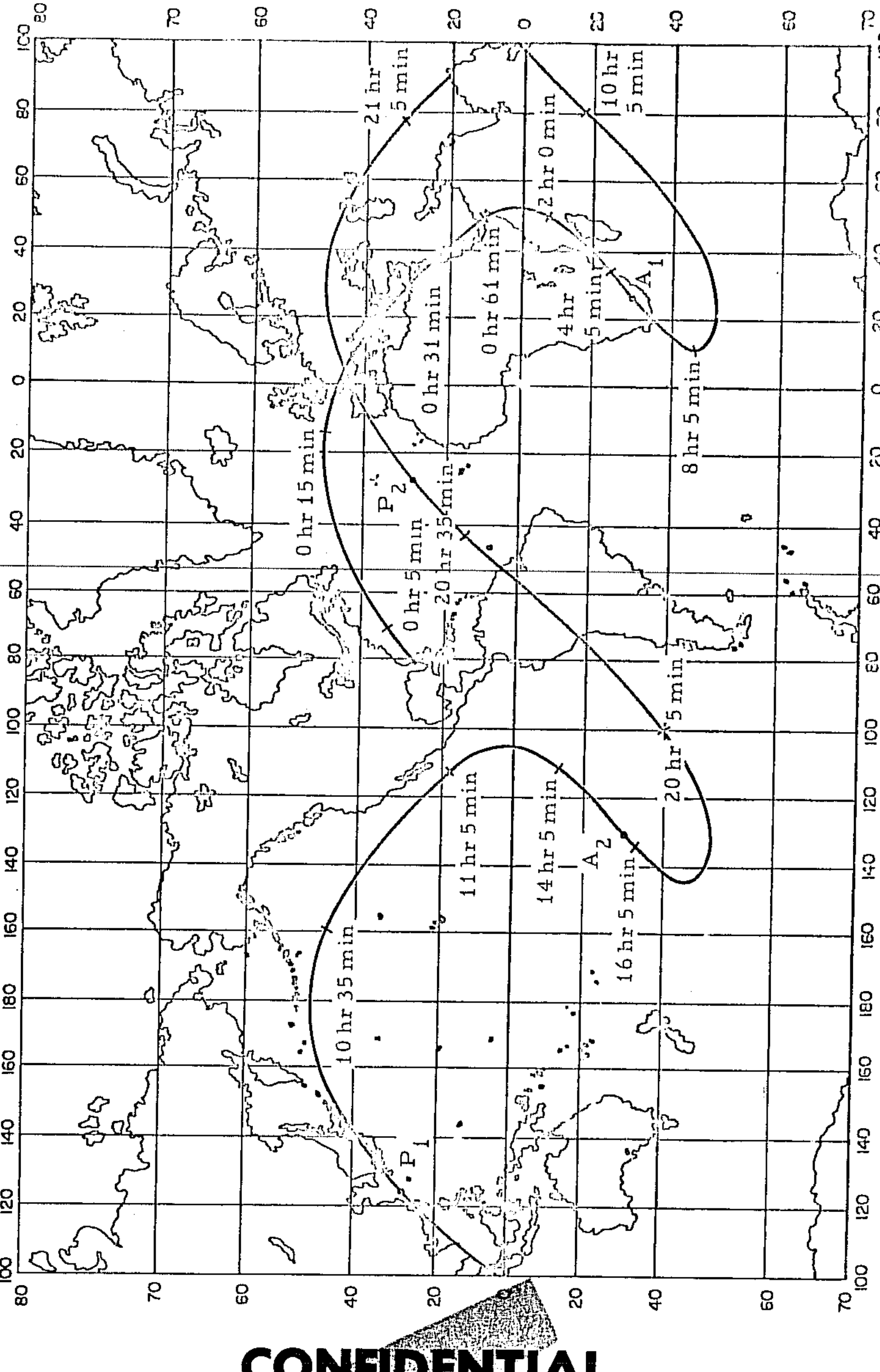


Figure 2. Projection of Nominal Vehicle Trajectory During First 21 Hours of Orbit. (A_1, first apogee; A_2, second; P_1, first perigee, etc.)

performance of this engine has been verified adequately with static firing tests conducted at ABL and the Arnold Engineering Development Center, Tullahoma, Tennessee. The terminal stage (payload) will be mounted on the forward end of the third stage. The structural integrity of the payload has been verified by static and dynamic structural tests conducted at STL. The fiberglass nose fairing used for aerodynamic shielding extends from the second stage forward and covers the entire third stage and payload. This fairing, also used in the Able-1 program, has been proven by static and functional tests conducted at Aerojet General Corporation under the supervision of STL.

Interconnecting structures between each stage of the vehicle were designed by STL. The first and second stages are connected by two semi-monocoque transition structures, both being protected by thin stainless steel heat shields. The interfaces of the transition sections are connected by eight matched fittings, four to carry axial loads and fitted with explosive bolts for first to second stage separation, and four to carry compressive loads only and fitted with shear pins for alignment. The forward transition section is attached to the second stage aft skirt while the aft transition section is attached to the Thor guidance compartment. Adequate exhaust ports for second stage gases are provided in the aft transition region.

The control compartment is located on the forward end of the second stage. This compartment houses the control electronic assemblies, electronic power supplies and instrumentations. It is of semi-monocoque construction with two full length stressed doors. The forward end of this compartment supports the nose fairing and the second to third stage support structure. Exhaust ports for third stage gases are also located in the forward end of the control compartment.

The third stage is supported by four inclined supports which are pivoted on fittings at the forward face of the control compartment and clamped to a ring bonded to the third stage. The clamp is held together by two explosive bolts which, when fired, release the supports, thus freeing the third stage.

The payload is supported by a short cylindrical interstage. The third stage and the payload are attached together with a clamp held in place by two explosive bolts, which when fired release the payload from the third stage.

b. Performance

(1) Powered Flight

The vehicle will be launched vertically from an existing Thor launch stand at AFMTC. During the first 10 seconds of flight, a roll program will be executed which will place the pitch plane of the missile in the flight azimuth. Starting approximately 10 seconds after launch, a pitch program will be initiated to place the missile on the desired trajectory. Main engine cutoff occurs approximately 160 seconds after launch on incipient first stage propellant exhaustion. During the staging process, the first stage vernier engines continue to propel the vehicle in order to maintain a positive head on the second stage propulsion system and assure reliable ignition.

On first main engine cutoff, a signal is transmitted to the second stage to initiate its ignition and separation from the third stage. Since the second stage propellants are hypergolic, ignition occurs on injection into the thrust chamber. Shortly after staging, the aerodynamic nose fairing is jettisoned. The second stage engine will burn for approximately 115 seconds.

After second stage shutdown, 12 small solid propellant spin rockets will be fired to spin the second stage, third stage, and payload assembly to a rate of approximately 150 revolutions per minute. Following spin-up, explosive bolts joining the second and third stages are actuated. After separation, the third stage solid propellant motor is ignited. Burning time of the third stage is approximately 37 seconds. Separation of the payload from the third stage will be accomplished by actuation of explosive bolts and by a spring which will impart a small velocity differential to the payload. The payload separation sequence will be initiated by a timer. A sufficient interval will be provided between third stage burnout and separation to insure that any residual burning of the third stage motor has terminated.

The powered flight plan is similar to that used for Able-1. The sequence of events is shown in Figure 3.

(2) Weights

A preliminary weight breakdown used in the trajectory calculations is shown in Table 3.

Table 3. Able-3 Weight Breakdown.

Stage	Gross Weight (lb)	Expended Weight (lb)	Jettisoned Weight (lb)
Payload	144		
3	517	464	54
2	4,186	3,235	950
1	107,409	98,126	9,283 (includes fairing)
Lift-off weight	112,256		

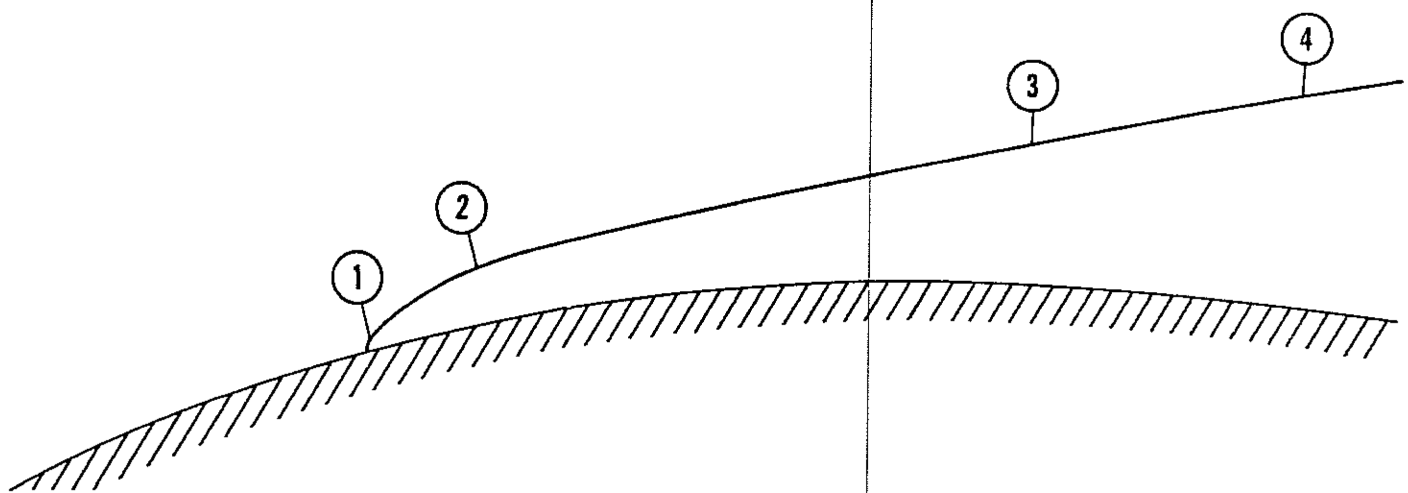
(3) Sequence of Operations

The planned sequence of operations is shown in Table 4. A schematic of this sequence is shown in Figure 4.

5. Guidance and Control

a. Launch Guidance and Control

The first stage is a standard Thor missile airframe, propulsion, and control system without an inertial guidance system. Guidance will be accomplished by use of roll and pitch programmers. The control system has been modified to accommodate the dynamics of the four-stage vehicle by relocation of the rate gyros and redesign of the compensation networks. The engine will cut off on incipient exhaustion of either propellant, i.e., when main engine chamber pressure drops to 80 per cent of nominal.



Point	Flight Time (sec)	Comments	Inertial Speed (ft/sec)	Downrange Distance $R_o \theta$ (nmi)	β (degrees)	Altitude 1000 ft	Weight (lb)
1	10.00	End of Vertical Flight	1,349	0	0	0.632	106,097
2	160.01	Thor Burnout	15,923	86	70.4	300.2	14,147
3	276.9	End of Second Stage	24,088	412	81.8	857.3	1,610
4	316.7	End of Third Stage	33,297	577	87.0	967.3	197.8

Figure 3. Powered Flight Trajectory.

Table 4. Project Able-3 In Flight Sequence

Time (Seconds)	Stage in which Event Occurs	Action		Equipment Initiating Action
X + 0	I	First Motion		
X + 0.1	I	Lift-off Switch Activates		Microswitch
		Programmer Starts		Relay - CEA
		Gyros Uncaged		Relay - CEA
	II	Electrical Umbilical Ejects		Lift-off Signal Operates Releases w/lanyard Backup from Umbilical Mast
		Helium Umbilical Ejects		
		Arm Destruct Initiator - Stage II		Lanyard from Umbilical Mast
X + 2	I	Roll Program Initiated		Douglas Programmer Relays (CEA)
X + 9	I	Roll Program Complete		
X + 10	I	Pitch Program Initiated		
		1st Step Pitch Rate	As Required for Trajectory and detailed in DTO	
X + 28	I	2nd Step Pitch Rate		
X + 70	I	3rd Step Pitch Rate		
X + 98	I	4th Step Pitch Rate		
		Autopilot Gain Change		
X + 140	II	Second Stage Engine Start Circuit Armed R4		Acceleration Switch Set for 6.7 ± 0.1 g 0.8 ± 0.3 sec
	R4	Start 46 sec Time Delay (Tolerance ± 4 sec)		

Table 4. Project Able-3 In-Flight Sequence (Continued)

Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
X + 140	I	Pitch Program Complete	Douglas Programmer Relays (CEA)
		Programmer Output 0°/sec	
X + 148 Approx.	I	Main Engine Cutoff (MECO) Circuitry Armed	
X + 152*	I	MECO Armed	Propellant Float Switch Set to Operate When Either Tank has 3-1/2 sec of Fuel Remaining
		Vernier Tanks Repressurized	
X + 156	I	First Stage MECO	90% Chamber Pressure Switch Initiates Shutdown but Does not Send Signal to Center Engine
	II	Start Staging Sequence	Relay Closure in First Stage Generated by 90% First Stage Chamber Pressure Switch
		Blow Blast Doors in Adapter Section	Relay Activated by the Start Staging Sequence Signal
	R4-	Start 2-sec Time Delay (Tolerance $\pm$ 0.2 sec)	

*This time and all times following are based on the reference trajectory.

Table 4. Project Able-3 In-Flight Sequence (Continued).

Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
X + 158	II	Engine Fire Signal	Relay in Relay Junction Box (RJB)
	R4	Uncage Gyros	
		Start Pitch Program About 0.3°/sec Down	
	R4	Start 100-sec Timer	
X + 158+	II	Blow Separation Bolts	Chamber Pressure Switch Set to Operate at 60 of Second Stage Engine Thrust (TPS w/TVS_2 Back-up)
		Start 10-sec Timer	
X + 158.3	II	Second Stage Separation Complete	
X + 168 approx.	II	Pyrotechnic in Helium R4 Tank is Set Off	10-sec Timer Started at TPS (Tolerance ±1 sec)
		Cutoff Enable	
R4 X + 186		Jettison Nose Fairing	46-sec Timer Started at Arm
		Start 4-min Timer	Lanyard Attached to Nose Fairing
Times vary in accordance with commands	II	Stop Pitch Program and Start Pitch Command ±1°/sec	Radio Command: ** Polarity Determined by Command
		Stop Pitch Command Pitch Rate Zero deg/sec	Radio Command **
		Start Yaw Command ±1°	Radio Command: ** Polarity Determined by Command
		Stop Yaw Command Yaw Rate Zero deg/sec	Radio Command ** Timer

** Commands will not be connected on Able-3. The command link will be tested however.

Table 4. Project Able-3 In-Flight Sequence (Continued).

Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
R4 X + 258	II	Stop Pitch Program	Relay Started at 158 sec
X + 270	II	Cut-off Stage II Engine	Propellant Depletion (TPS)
	R4	Stop Pitch Program (if not stopped by timer)	
		Turn Off AGS Transmitter	
		Start 2-sec Timer	
		Start 3.1-sec Timer	
R4	III - IV	Release Paddles	
X + 272	II	Cage Autopilot Gyros and Ignite Spin Rockets	2-sec Timer (tolerance +0 -0.2 sec)
R4	IV	Paddles Lock and AGS Transmitter is Turned On	
X + 273.1	III	Ignite Stage III Motor R4	3.1-sec Timer (tolerance +0.2 -0 sec)
		Blow Stages II/III Explosive Bolts and Nozzle Shroud	
X + 310	III	Rocket Motor Burns Out	Depletion of Propellant
X + 425	III	Separation Bolt Blows to Separate Stages III and IV	4-min Timer Started at X + 185
	III - IV	Separation Occurs	Separation Spring
	IV	Arm Stage IV Ordnance (2 verniers, 1 injection rocket)	Switch Actuated by Physical Separation

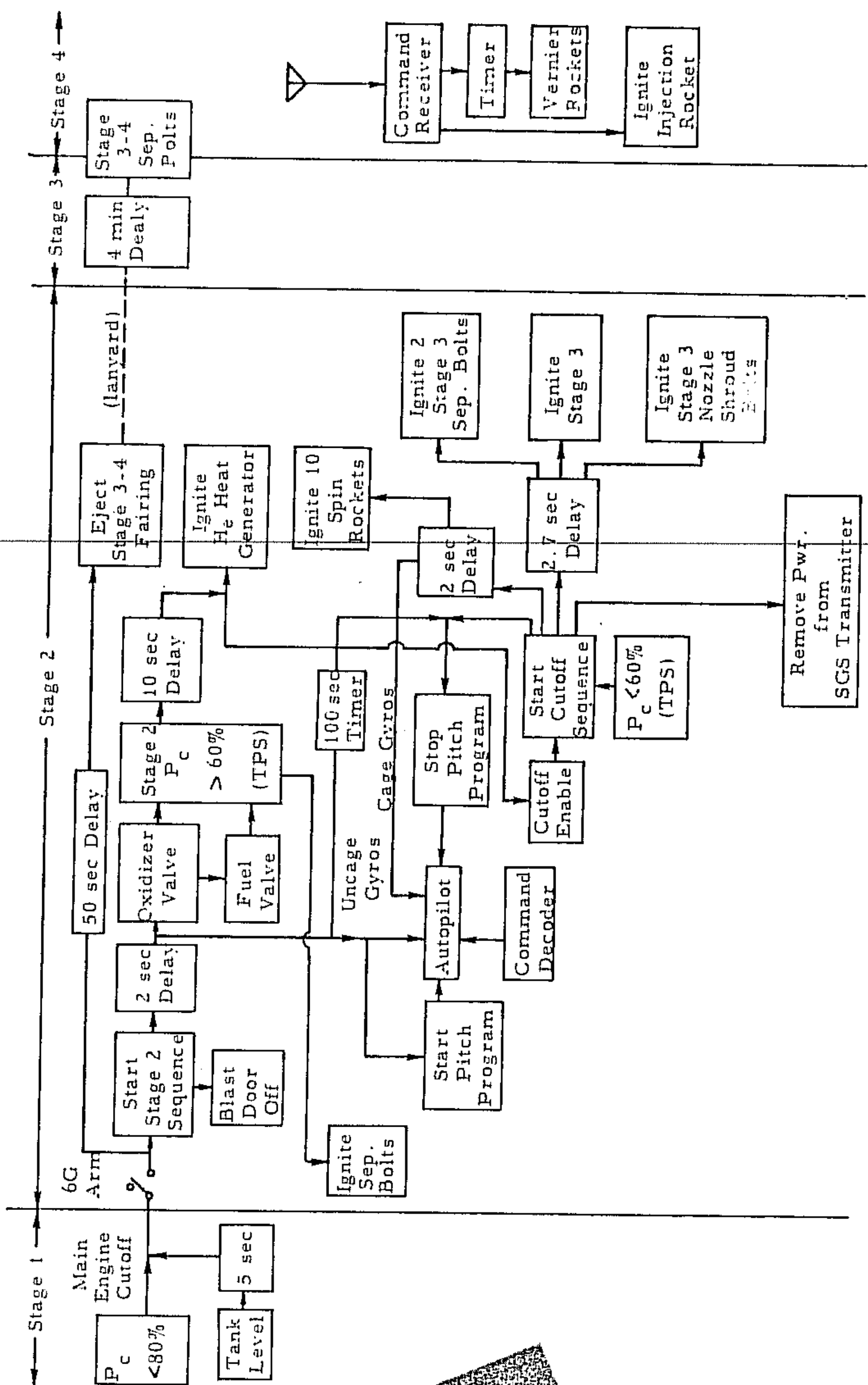


Figure 4. Able-3 Flight Sequence.

b. Second Stage Guidance and Control System

The modified second stage control system used in the Able-1 program required more power and was heavier than could be tolerated for the space missions contemplated. Approval for development of a new system was given in September 1958 and culminated in the scheduled delivery of the last of six lightweight, three-axis, attitude control systems in April 1959.

The second stage attitude system weighs 33.85 pounds, compared to 120 pounds for the earlier system, a weight saving of 72 per cent. Components are packaged in a modular design using stacked, etched circuit boards; subassemblies are packaged in such a way that a single portable test set calibrates and tests the entire system when it is being mated to the missile, makes over-all calibrations in the flight-readiness facility, and makes final tests on the gantry immediately prior to missile flight.

The control system is capable of causing the missile to follow a predetermined course in response to commands generated in an internal programmer, or it can also accept discrete guidance commands from the ground.

A gyro reference assembly contains three heat-regulated integrating gyros mounted along mutually orthogonal axes, respectively sensitive to missile attitude deviations in pitch, roll, or yaw. These gyros send signals to the electronic assembly, which converts them to output signals to actuate hydraulic servo valves which gimbal the motor and affect attitude in pitch and yaw channels. Similar electronic channels control roll through selective operation of four pneumatic jets located on the periphery of the airframe. Each jet can exert 15 pounds of thrust. The command converter assembly converts discrete signals from a command receiver into a form suitable for application to gyro torque generators: The following six commands, among others, are available through the receiver: pitch up, pitch down, pitch stop, yaw left, yaw right, and yaw stop. Commands are received in the form of a relay contact closure; contact closures initiate and maintain a constant a-c current in the gyro torque generator of the proper polarity to command the desired missile attitude change. The command converter assembly may be commanded; it also has provision for an attitude program consisting of a constant pitch turning rate for a specific length of time, established with a series rheostat and a transistorized time-delay relay. A static inverter supplies 400-cps power regulated in frequency and output voltage to the other three electronic assemblies.

6. Tracking and Communications

a. General Considerations

(1) Active Tracking

The Able-3 lunar payload package will be actively tracked from the ground during the free-flight trajectory. The tracking is accomplished in the following manner: a CW signal is transmitted from the ground to the vehicle, where it is amplified and retransmitted with a frequency offset. The transponder in the vehicle preserves the phase coherence between the transmitted and received signals. The transponded signal is received on the ground by a large antenna and a highly sensitive receiver. Range rate is measured on the ground by extracting the doppler frequency shift between the transmitted and received signal after correcting for the frequency offset introduced by the vehicle transponder. Accuracies of 1 ft/sec in radial velocity are believed to be achievable. Range is measured by frequency modulating the ground transmitter by low-frequency sinusoids. A modulating frequency of 2 cycles per second should result in a range resolution for this system of better than 100 miles with adequate smoothing. A second modulating frequency of 0.2 cycle per second will be used to resolve the phase ambiguities in the 2-cycle-per-second signal. Further ambiguities will be resolved by measuring the time delay between the start of modulation on the ground and the commencement of modulation in the return signal. Tracking in angle will be performed by nodding the ground antenna alternately in elevation and azimuth. Angular accuracies of about 0.2 degree should be achievable.

(2) Communication Requirements

In addition to furnishing tracking functions, the UHF radio link will be utilized for commands to the fourth stage and telemetry. The command system provides control of vernier and injection rocket firing, telemetry bit rates and channel selection, transmitter control, solar cell charging rates. The telemetry capability includes a pulse code modulated system for scientific experiment data and fourth stage performance, and an analog FM system for TV, VLF, search coil and a subcommutated channel.

On Able-3, two additional UHF transmitters will be incorporated. An FM/PM system composed of 11 standard IRIG subcarriers will be employed to transmit the scientific experiment data and fourth stage performance in analog form.

b. Ground Stations

The world-wide complex of ground stations established for the Able-1 lunar probe program will be supplied additional equipment to be used on the Able-3 program. These stations will provide telemetry reception and tracking data and also will be used to conduct the propagation experiments.

The Able ground station at AFMTC will provide reception of telemetry signals on a continuing basis and will also be used for checkout of the payload telemetry during countdown.

The Manchester ground station (which uses the 250-foot radio telescope at Jodrell Bank) will provide reception of telemetry signals and also precise tracking information on the payload during flight. It will also be equipped to send commands to the payload and will be used to fire the fourth stage injection rocket, if required.

The Millstone Hill, Massachusetts, radar site of the MIT Lincoln Laboratories will utilize the 85-foot parabolic antenna. This station will provide skin-tracking of the second stage by monopulse radar during launch and will attempt to track the third stage. The Millstone station will then switch to telemetry reception after the first few minutes of the launch phase and, when the payload is in orbit, will provide telemetry reception, angular tracking information, and one-way doppler measurement.

The Hawaii station does not participate in the launch phase. After the payload is in orbit, the Hawaii station will provide angular range and range rate tracking information, reception of telemetry signals and transmission of commands. It will be the only ground station instrumented for the propagation and faraday rotation experiments.

The Singapore station does not participate in the launch phase. Once the payload is in orbit the Singapore station will provide reception of telemetry signals. The Singapore antenna beam width is too broad for development of effective angular tracking information.

The BMD/STL Operations Center that was established for Project Able-1 will be expanded and utilized for Project Able-3. This center is located at Space Technology Laboratories, Inc., Building E, Los Angeles. Teletype and leased telephone links between the facility and the various ground stations to be utilized are shown in Figure 5.

Prior to launching, nominal trajectory data will be transmitted to these stations as well as to other cooperating stations. The data will be used to plan nominal steering periods, and will be used for antenna steering until more accurate data from actual tracking is available. After the missile is launched, the trajectory measurements from the various stations will be reduced at the site to a form suitable for transmittal to the Operations Center. The first stations to have this data will be those stations located in and near the launching area. The data normally will be coordinated through the STL operations at AFMTC. Tracking data transmitted to the Operations Center will be used in a continuing operation. Increasingly accurate estimates of the actual trajectory are calculated on the IBM 704 computer of the Computer and Data Reduction Center. After more accurate trajectories are calculated, look-angle data for the various ground stations will be transmitted to these ground stations. The recalculation of trajectory and steering data will proceed on a continuing basis throughout the useful life of the Able-3 payload. The

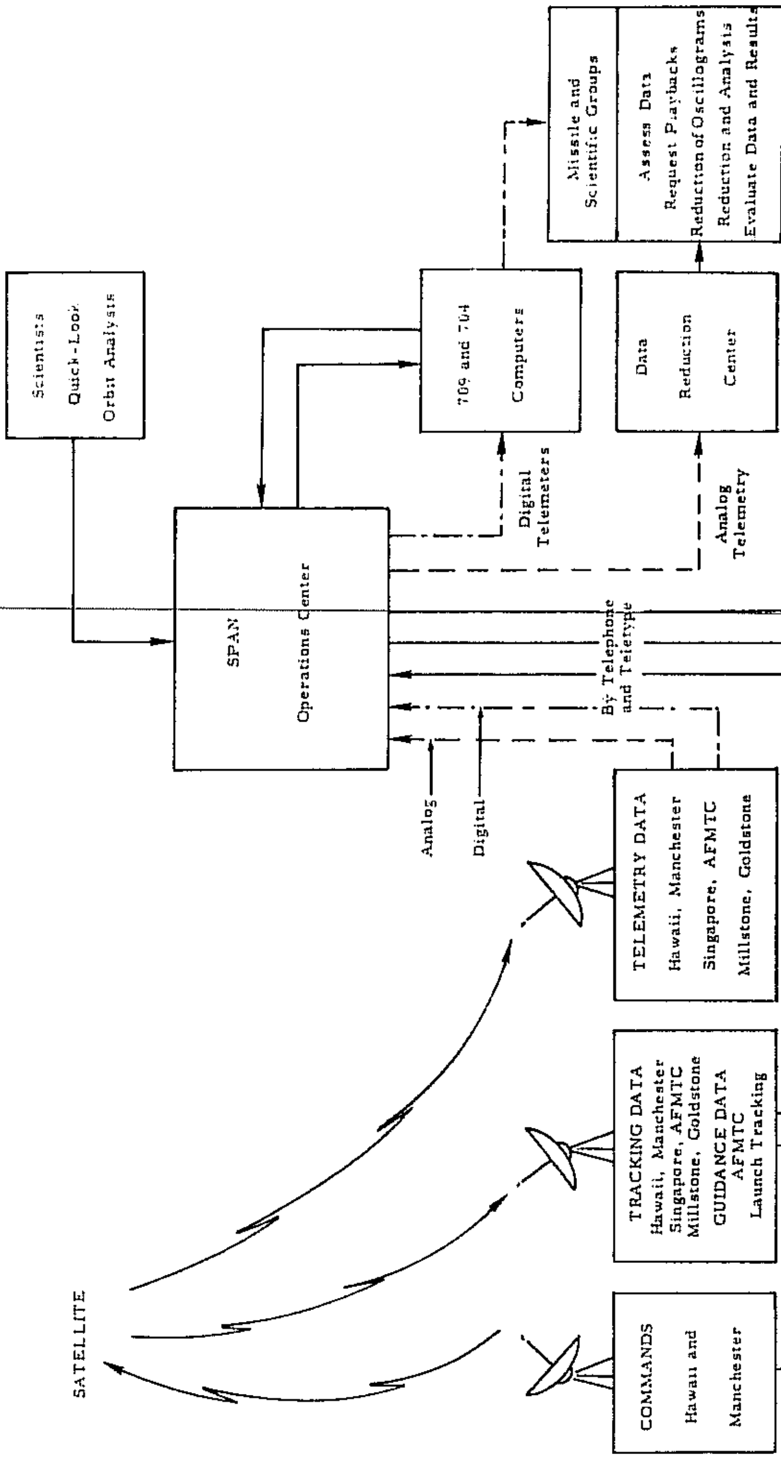


Figure 5. Ground System Network.

trajectory determination task will operate on a full-time basis from launch until the time when a very accurate trajectory has been determined.

Thereafter, trajectory calculations will be made only as required to plan routine tracking operations at the ground stations. After the first few days of flight, scheduling of tracking operations probably will be established as a weekly operation. The emphasis in the initial post-launch period will generally be to quickly establish a sufficiently accurate trajectory to determine the degree of success of the operation and to insure that all tracking stations have accurate look-angle data.

In addition to trajectory data, various sites will also receive information telemetered from the test vehicle. The telemetered data will be automatically punched onto teletype tape and transmitted to the Operations Center for further interpretation.

The expansion of the ground facilities required engineering system changes at Los Angeles, assembling and testing subsystems there, and shipping the equipment to the field.

While the original VHF receiver and analog telemetry system was retained, additional facilities were installed to receive UHF signals employing a new narrow-band digital telemetry system uniquely adapted to reception of information from shore vehicles at great distances. In addition, transmitter facilities were installed at Hawaii and Manchester capable of switching the payload high power transmitters on and off, thus conserving battery power needed for transmission of information.

The added transmitter and UHF receiving facilities greatly complicated the antenna feed structure and diplexing equipment required at Hawaii and Manchester, necessitating the redesign of these facilities.

(1) Telemetry and Tracking Receivers

The tracking and telemetry system will utilize multiple conversion superheterodyne receivers of the phase-lock type provided by Motorola on STL specifications. While these receivers basically met the initial specifications which were considered optimum for the state of the art, it has since become apparent that capabilities beyond those initially prescribed would now be desirable. This desire is occasioned primarily by the fact that the recent change in launch plans may involve much greater distances than initially anticipated. Numerous laboratory experiments and evaluations on the existing receivers have been performed.

It is now evident that with a few modifications to the existing receivers, considerable sensitivity and range extention can be achieved over that initially anticipated. One of the major factors making this possible is the fact that future launches will use a digital telemetry system utilizing a much narrower information bandwidth, thereby making it possible to greatly reduce the noise bandwidth of the receiver.

The preliminary tests indicate that a 15 db improvement over that initially specified can be expected from the modified receiver.

In addition to the UHF receivers described above, station equipment includes UHF receivers previously used on Able-1. These are phase-locked-loop Microlock receivers manufactured by Hallamore Electronics Co. These receivers, like the VHF receivers, are narrow-band type designed for best possible performance at low signal levels.

(2) Transmitter - Multiplexer

The final power amplifier of each of the two ground transmitters for the Able 3-4 program will consist of an RCA UHF air-cooled, power tetrode (Type A 2545), operated in a coaxial cavity. The final power amplifier is capable of 10-kw power output and is approximately 50 per cent efficient at the operating frequency. The driving power is derived from a similar state that is capable of 1-kw power output. Both stages are motor-tuned, and with a frequency doubler and low power driver (for the 1-kw stage) are packaged to be installed and operated in the feed tower of large tracking antennas from a remote position.

Since it is required that a very sensitive receiver be operated with the same antenna, and at a frequency very near the 10-kw transmitter, a multiplexer was designed with very high Q resonant elements that are insensitive to temperature variations. This required the use of spherical cavities made of Invar. The multiplexers are designed to be installed and operated, along with the transmitter, in the feed towers of large tracking antennas.

(3) Ground Station Antennas

The multifrequency feed for the Hawaii installation is composed of four concentric coaxial cavities with a common phase center. The cavities are excited in the TE_{11} mode by probes and possess primary radiation patterns that are almost equal in their E and H plane beamwidths so that the secondary pattern is truly conical in shape.

~~The feed housing contains the necessary switches and~~ hybrids to allow remote operation on either linear or circular polarization for two frequency ranges and circular polarization for the remaining two. The coaxial cavities can be mechanically driven so as to permit rotation of the electromagnetic field vectors to the most favorable orientation for signal transmission or reception.

The multifrequency feed for Manchester is similar in most respects to the Hawaii feed. They do differ in that the mechanical servo drive system at Manchester for orientating the electromagnetic vectors is exterior to the feed housing and the actual layout of the transmission line is dissimilar due to the differences in the attachment of the feed to the associated antenna structure.

c. Airborne Instrumentation

(1) Stage 2

(a) Guidance and Telemetry Antennas

The second stage guidance antenna will employ two driven folded monopoles and associated reflectors spaced on both ends of a diameter of the second stage. The spacing between the driven element and the reflector was determined by experiment so as to yield sufficient antenna gain during all phases of the guided powered flight.

A diplexer will be employed to interconnect the airborne guidance transmitter and receiver to a common antenna.

The second stage telemetry antennas use the same design technique as the guidance antennas but appropriately scaled in frequency. The former are located on a diameter which is orthogonal to that containing the guidance antennas.

A fiberglass fairing completely encases the above referenced antennas and provides a heat shield, and strengthens the antenna as well as streamlining it.

(b) SGS Transponder

The Able-3 and -4 second-stage command transponder is a phase-lock receiver which produces a coherent output of 16/17 of the received frequency. The receiver noise bandwidth is 1 kc which combined with the receiver noise figure of about 60 db allows the receiver to lock on to a signal level of -120 dbm. Acquisition time is less than one second. Seven tones utilizing a double coincidence technique can be used for seven commands, six steering and one shutoff. These will not be utilized on Able-3. Twenty-one tones are possible. The receiver requires less than 3 watts and 28 volts and is 3-1/2 by 7 by 10-1/2 inches. It weighs about six pounds.

(c) Second Stage Telemetry

The second stage telemetry will consist of five FM subcarrier channels and one 18 segment PAM-FM subcarrier channel. The phase modulated r-f carrier will be at 238.5 mc radiated power, 2.0 watts nominal. The antenna system will consist of four folded unipoles, equally spaced around the circumference of the missile and fed in a manner to provide essentially isotropic coverage. The radiated field is right circularly polarized off the aft end of the missile. The system is entirely self-contained, utilizing its own batteries for primary power.

The information transmitted will consist of strategic second stage airframe, propulsion, control measurements and events. In addition, third stage operation and second-third stage separation will be telemetered.

(2) Payload

(a) Power Supply

Electronic components will be powered by storage batteries kept charged by the solar cells. A total of 8000 cells will be carried. Storage batteries of 50-watt-hour capacity are required to provide for intermittent operation of the 378.21-mc telemetry transmitter, which requires more power than the solar cells can provide, and to prevent interruption of experiments when the payload is in the shadow of the earth. A capacity of 15 watts is available from the cells, as shown in Figure 6. Sunlight will be available ~~90 to 95 per cent of the time; eclipses will last at most two hours.~~

The four paddles will furnish 12.2 sq ft of surface area for the solar cells; location of the solar cells on these paddles will permit temperature control of the sphere since its surface can be painted to minimize temperature variation by the constantly changing relationship of the payload to the sun. Each paddle is hinged to permit the four to fold down symmetrically inside the aerodynamic shroud and to extend 22.5 degrees from the sphere's equatorial plane. At second stage burnout, springs will cause the paddles to move to the extended position and latch. When these arms latch into place, a microswitch is mechanically actuated, turning on the payload transmitter.

(b) Payload, Antennas, Diplexers, and Associated Cabling

The Able-3 payload utilizes both VHF and UHF frequencies for the data transmission and guidance functions. The VHF transmission circuit subsequent to the two transmitters consists of a coaxial ring hybrid, a shunt tuning stub to match the antenna system to the hybrid, a diplexer for diverting the whistler signals to the Stanford receiver, and the two monopole antennas.

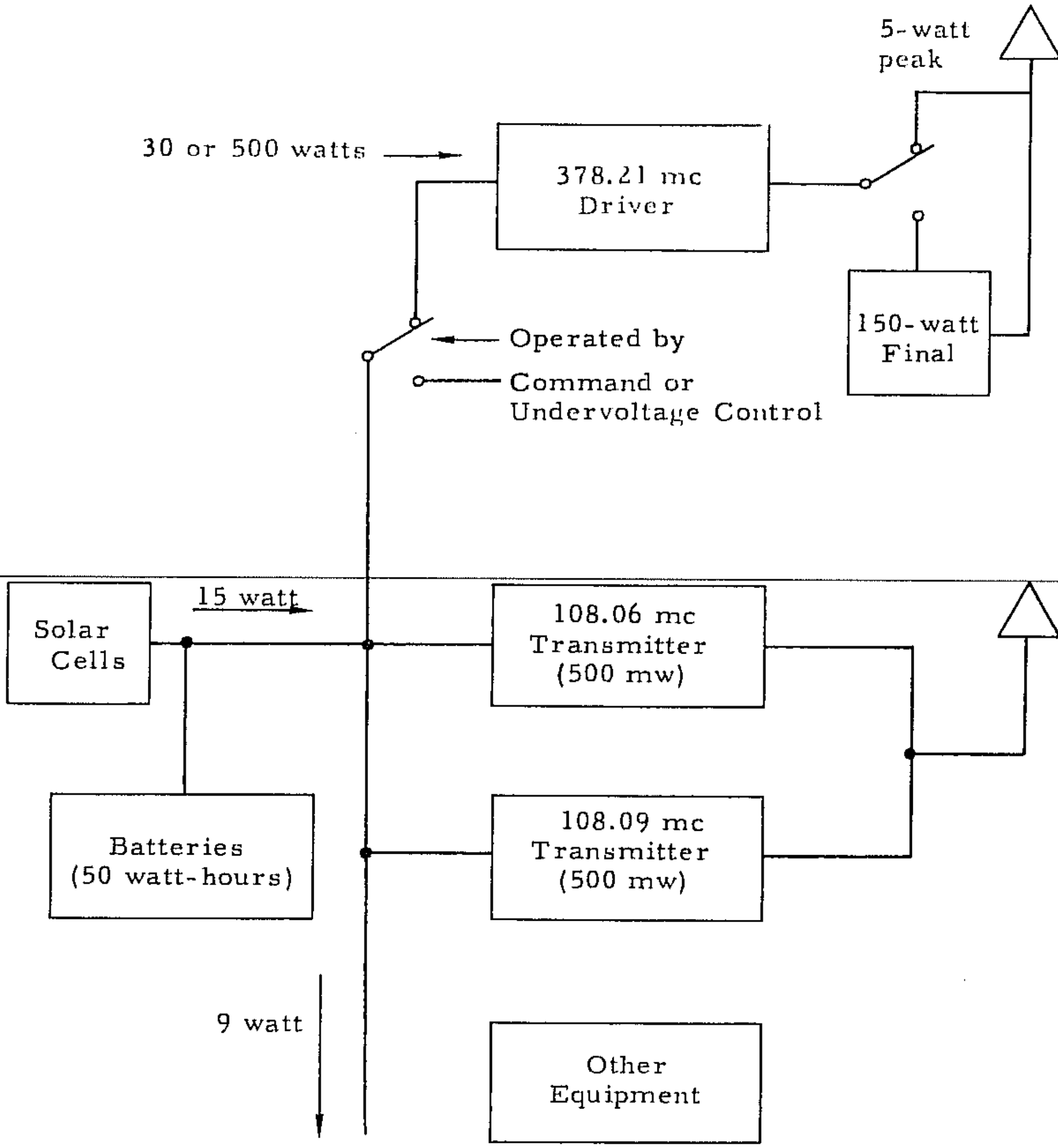


Figure 6. Power Distribution in Payload.

The coaxial ring hybrid connects the outputs of the two VHF transmitters to a common antenna system such that the output powers from the individual transmitters add algebraically as they are separated in frequency. To reduce the hybrid weight, smalldiameter, low loss, coaxial cable was employed.

All the diplexers are purchased parts. The VHF diplexer consists of a resonant circuit tuned to 15 kc for the whistler experiment and is shunted by a bandpass filter for operation at VHF. The elements of the latter filter employ a helical transmission line to provide a compact but rugged over-all structure.

The VHF monopole antennas are conventional in design and operate against the payload shell as the ground plane. The resultant radiation pattern is perturbed by the nonplanar ground plane and the solar cell paddles. Pattern data has been taken with the actual payload and solar cell paddles to insure that the pattern coverage and gain will be adequate for the mission requirements.

The UHF antennas are similar in design to the VHF radiators but approximately scaled in length due to the change in frequency. They are two monopoles operating against the payload shell and are located on the opposite end of the payload from the VHF antennas. Both the antenna systems are symmetrically located with respect to the spin axis of the payload. Lightweight but strong fiberglass construction was employed throughout the antennas.

The UHF diplexer utilized high Q coaxial filter elements in the conventional filter design. The inner conductor diameter is varied to obtain inductive and capacitative elements in the synthesis of the filter. The UHF diplexers in all payloads are identical.

The Able-4 Thor payload radio frequency transmission circuit consists of a diplexer, a quarter wavelength impedance transformer, and a single monopole antenna located on the payload spin axis. Radiation patterns have been taken with a solar cell and payload mock-up.

The real articles will be used in forthcoming radiation pattern studies although pattern and impedance measurements conducted with the mock-up are usually very similar to those made with typical flight articles.

The Able-4 Atlas payload has a redundant UHF antenna system and a VLF antenna configuration for the Stanford whistler experiment.

The UHF antenna uses two monopoles located symmetrically about the spin axis and on both ends of the payload. They are tied to their respective transmitters by quarter wavelength impedance transformers and diplexers.

The VLF antenna is two monopoles. They are resonated at 15 kc by lumped elements in the Stanford receiver. The monopoles are spaced to clear the third stage upon separation and avoid entanglement with the solar cell paddles during their erection.

(c) Command Receiver

The Able 3-4 payload command receiver-transponder is a double conversion phase-lock receiver which produces a coherent output at 16/17 of the received frequency.

The receiver loop noise bandwidth can be commanded to be either 250 cps or 40 cps. The narrow loop noise bandwidth combined with a receiver noise figure of 12 db allows the receiver to lock on to a signal at a level of at least -138 dbm. The receiver incorporates circuitry to enable it to automatically search in frequency over a range of 30 kc. Acquisition time is 20 seconds and 3-1/2 minutes for the two bandwidths.

The receiver requires 1.5 watts at 16 volts and measures 3 by 6-1/2 by 7 inches. The weight of the receiver is 4-1/2 pounds.

(d) Transmitters

The Able-3 payload will have two phase modulated, tracking transmitters (~ 378 mc) which will also be used for transmitting digital telemetry. One will have a power output of 5 watts and provide four times frequency multiplications and power amplification of the IGY signal

(~ 108 mc). The fundamental frequency generator is located in the coherent doppler command receiver. From there the signal will go to a buffer amplifier stage and then the transmitter is initiated with 2 to 8 milliwatts of input power. The transmitter will be about 3 by 6 by 1.6 inches, weigh about a pound. It has an efficiency of about 10 per cent.

The second transmitter (also at ~ 378 mc) will have 150 watts of output power. The 5-watt transmitter will be used as the driver. The transmitter will be 6 by 4 by 3.8 inches and weigh about 2-1/2 pounds. Its efficiency is about 45 per cent. With this transmitter a heat sink is necessary to radiate the 180 watts of heat. The heat sink will be about 4.5 by 8 by 2 inches and weigh about 2.3 pounds.

(e) Digital Telemetry System

Transmitting information over interplanetary ~~distances extending upward of 25 million miles requires the use of a digital~~ telemetry technique since the FM/FM analog technique, utilizing a phase-locked demodulator, requires that background noise be decreased and the transfer function due to the drift of the subcarrier oscillator be reduced. These, however, are incompatible. A digital telemetry system, on the other hand, operating in conjunction with a phase-locked loop demodulator, does not suffer from this basic deficiency, and background noise effects can be decreased without introducing calibration drift errors. Furthermore, a digital telemetry system can be easily controlled to vary its information rate as range increases; it permits a substantial increase in speed of data reduction and it allows for storage of data during periods in which the transmitter is off.

A Digital Telemetry Unit (DTU) will be provided on the Able-3 vehicle to test a telemetry system suitable for interplanetary communications. It accepts both analog and digital inputs from various experiments. The converted information at its output will appear as a binary coded subcarrier (1024 cycles/secong), which then phase modulates either the 5-watt or 150-watt transmitter.

The binary output of the DTU will occur at a synchronous rate and will be composed of repeating sets or frames of words.

For Able-3 and Able-4 Atlas, 11 words per frame will be necessary, while for the Able-4 Thor, a frame will consist of 8 words.

One word of each frame will be used as a frame sync and will be read out as all zeros, while the balance of the words are coded with the digital representation of the input information. Each word will contain 12 bits. The first 2 bits (for information words) are always coded the same (space-mark) and define the start of a word; that is, these 2 bits provide a word sync. The other 10 bits may take on any combination of binary levels to represent a number from 0 to 1023.

The bit rate may be 1, 8, or 64 pps depending upon the ground commands. Thus, at 1 pps a word lasts for 12 seconds and a frame (of n words - as in Able-3) for 132 seconds. For instance, 1 pps is maximum rate at 60 million miles (150 watts), while up to 8 million miles, 64 pps might be used.

A 12-bit combination counter (binary) and shift register referred to as a shifting accumulator (SA) is provided for each word. Pulses from a digital-type experiment are applied directly to a shifting accumulator, while an analog input is applied to an analog-to-digital (A-D) converter whose output is then applied to a shifting accumulator. For Able-3 and Able-4 Atlas, there are 6 digital and 4 analog words, while for Able-4 Thor, there are 5 digital and 2 analog words.

The A-D conversion will be accomplished by comparing the analog quantity to a digital ramp. An electronic commutator running synchronously at the word-rate gates east shifting accumulator for a period of time inversely proportional to the number of words. When a shifting accumulator is gated on, the binary count is read-out at one of the three synchronous bit rates, 1, 8, or 64 pps. The outputs of all shifting accumulators are tied together but only one is gated or read-out at a time, so this common output provides the information of all words. This binary output then encodes a 1024 cycles/second subcarrier by switching its phase by 180 degrees whenever the binary output is a "one." This binary encoded subcarrier then phase-modulates the 5-watt or 150-watt transmitter. Figure 7 shows the schematic of the digital telemetry system.

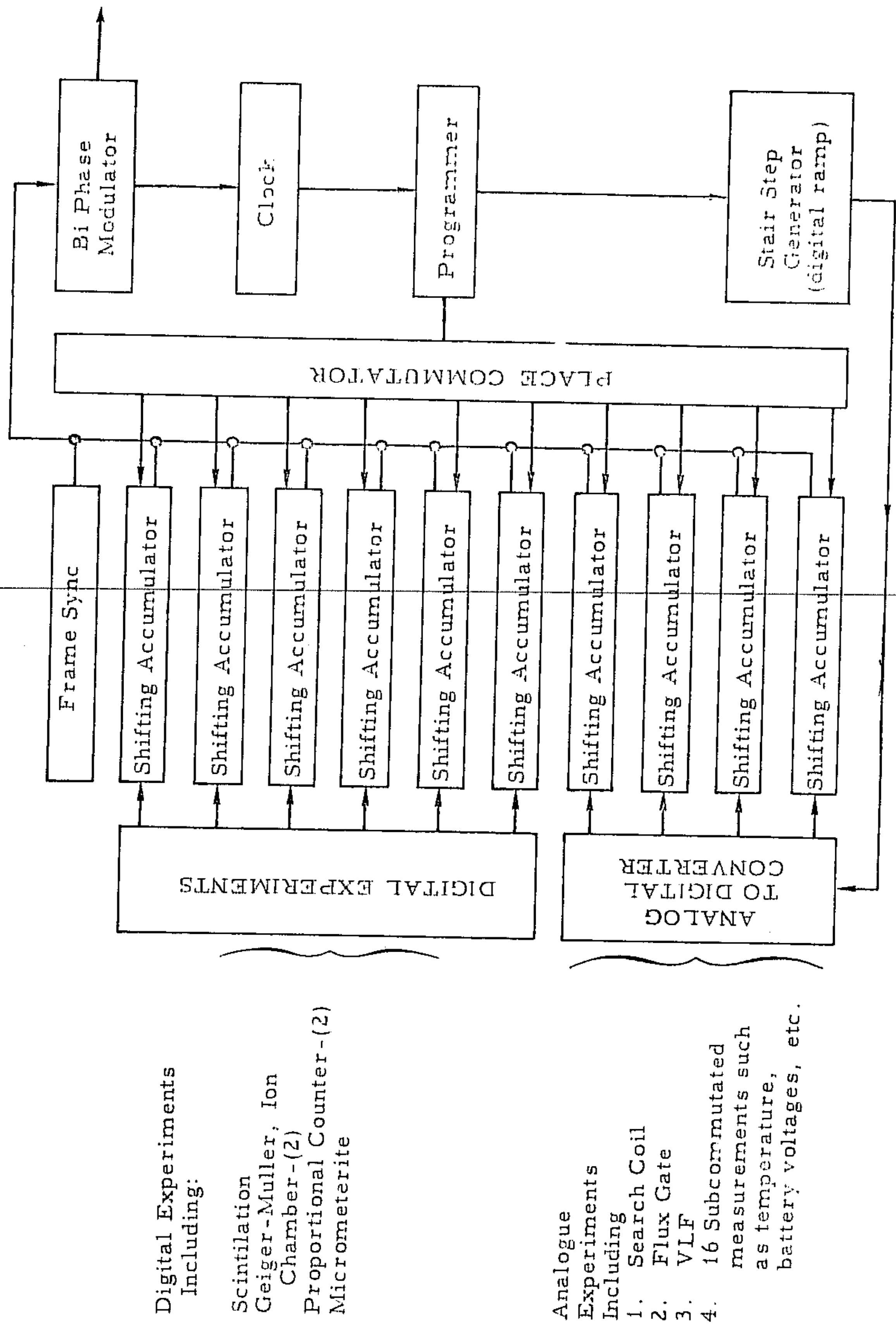


Figure 7. Schematic of Digital Telemetry.

7. Payload Vehicle

a. Experiments

Essentially two classes of experiments will be conducted with the Able-3 payload: (1) environmental experiments in which data of the space environment encountered by the satellite will be gathered and telemetered to ground stations; and (2) propagation experiments in which two of the payload transmitters will be used to transmit signals to Hawaii and possibly Boulder, Colorado. Payload transmitter operation will be controlled and the payload functions selected by use of the doppler and command system. Descriptions of the experiments are as follows:

(1) Environmental Experiments

Ten types of environmental data will be obtained by the Able-3 payload, as follows:

(a) Micrometeorite Flux and Momentum

An apparatus similar to that used on Able-1 (Flights 1 and) will be utilized to count impacts of micrometeorites above about 10^{-4} gr cm/sec momentum. It consists of two metallic plates on opposite sides of the payload and attached microphones designed to detect the vibrations induced by impact. Two momentum levels will be measured. Over-all weight of the equipment is 0.7 pound. The average power requirement is 70 mw.

(b) Magnetometer (Search-Coil)

An STL search-coil magnetometer will be used in conjunction with a flux-gate magnetometer (paragraph (c) below) to enable mapping of the vector magnetic field. Continuous measurements will be made of the magnetic field and its direction. The weight of this equipment is 1.6 pounds; power consumption is 50 mw.

(c) Magnetometer (Flux-Gate)

A flux-gate magnetometer will be used in conjunction with the STL search-coil magnetometer to measure the spin axis component of the magnetic field. Its weight is 2.2 pounds; its power consumption is 8 mw.

(d) Vehicle Position Determination

A fascimile system consisting of both optical and electronic equipment will be contained in the payload to determine the position of the vehicle relative to the earth. This equipment is a modification of the fascimile system used in the Able-1 lunar probe. The television system operates over a 2-cps passband (1-cps information bandwidth), weighs 1.9 pounds complete including modulators, and has an over-all power drain of less than 230 mw. The following is the scanning sequence: a fan beam trigger gates a scaler; the latter initiates a sequence which permits a spot scanner to sample one brightness level per vehicle revolution; each spin cycle advances the spot by a small amount. The over-all effect is that one line is scanned in a number of spin revolutions with some overlap from line to line. The orbital velocity of the vehicle will determine the line sequencing.

Transmitted pictures of the earth will have a resolution of approximately 5 miles. They will be used to determine the position of the vehicle relative to the earth and will provide meteorological information, e.g., cloud cover.

(e) Ion Chamber and Geiger Tube

An ionization chamber developed by the University of Minnesota will be carried to measure the total radiation flux. In conjunction with this chamber, the University of Minnesota is also supplying a Geiger-Muller tube for count rate. The combination of these two instruments will furnish mean specific ionization per particle. Weight of this equipment is 2.1 pounds; power, 140 mw.

(f) Scintillation Counter

An STL scintillation counter will be used to determine the intensity of the very soft component of cosmic radiation as a function of latitude, longitude, and altitude, to detect radiation injected into the geomagnetic field by solar wind, and to correlate changes in intensity with solar activity. The experiment is designed to have a threshold of 150 kev for electrons and 2 mev for protons. Shielding used on this experiment will be of a different material from that used on the University of Minnesota experiment. Equipment weight is 4.8 pounds; power required is 320 mw.

(g) Cosmic Ray Telescope

A triple coincidence proportional counter telescope, designed by the University of Chicago, will be used to obtain a total count of charged particles above two energy thresholds. Except for a possible change in shielding, this is the same experiment carried on Able-1. The apparatus consists of an array of seven counters, six of which are arrayed in a concentric ring about the seventh. Both the singles counting rate of the center counter and the triple coincident rate are recorded, since the outer counters are connected in two groups of three. The apparatus is surrounded by 5 gm/cm^2 of lead. Weight is 5.3 pounds, and the power requirement is 190 mw.

(h) Aspect Indicator

This equipment is a phase comparator which measures the phase relationship between the output of a photoelectric diode "sun scanner" and the search-coil magnetometer; this will provide the "H" direction of the magnetic field encountered. Weight and power requirements are included in paragraph (b).

(i) VLF Propagation

A VLF receiver furnished by Stanford University will be utilized in the Able-3 vehicle to monitor the propagation of 15.5-kc signals from radio station NSS, Annapolis. When correlated with the outputs of ground receivers located at the telemetry stations, this experiment will enable studies of the dispersive properties of the atmosphere at very low frequencies. In addition, the sun, planetary atmospheres, and the interaction of solar plasmas and the geomagnetic and interplanetary fields are all potential sources of very low frequency noise. The analysis of such noise, as observed from the Able-3 satellite, will yield valuable data concerning variations in solar activity and the nature of the plasma streams and magnetic fields in space.

(j) Temperatures

Temperature readings of the payload compartment and of the paddles which contain the solar cells will be telemetered from the satellite. Thermistors for sensing temperature will weigh 0.2 pound and will require 1 mw.

(2) Propagation Experiments

The propagation experiments will utilize one-way transmission from the Able-3 payload to ground receiving stations. Three types of propagation measurements will be made, as follows:

(a) Electron Density

Two coherent transmitters operating at frequencies of 108.06 mc and 3.5 times 108.06 mc (378.21 mc) will be used for electron density measurement. To obtain electron density at the satellite, the doppler ~~frequency shift of the transmitters will be compared at the Hawaii station.~~ As the payload passes through space where there are no electrons, the doppler shift is exactly proportional to frequency. The presence of electrons, however, forms a "dielectric" medium. This "dielectric" will have a larger effect at low frequencies than it will at higher frequencies. Careful comparison of the doppler effect at two widely separated frequencies will thereby provide a measure of ion density.

(b) Faraday Rotation

The faraday rotation caused by a change in the total ions along the propagation path from payload to ground is to be measured in Hawaii. This measurement involves observation of the rotation of the plane of polarization of the arriving 108.06-mc signal. This rotation will be relatively slow and will be recorded automatically.

(c) Amplitude and Phase Fluctuation

A third propagation experiment which may be made will use signals from the 108.06-mc transmitter. Two receivers spaced on a 475-meter baseline will be used to measure amplitude and phase fluctuations induced by the ionosphere. Ground equipment which would be used for this experiment is already in existence at the National Bureau of Standards Laboratory in Boulder, Colorado.

b. Payload Structure

The payload structure consists of a welded truss assembly supporting a fiberglass honeycomb payload platform. Four hinge brackets are bolted to the truss at the four quadrants to support the solar cell paddles.

The resin bonded fiberglass nose fairing covering the third stage and payload consists of a cylindrical section with a hemispherical head. The fairing is split longitudinally and its halves are connected by two explosive bolts (at the head end) and two explosive actuators (in the plane containing fairing center of gravity) to insure jettisoning.

The concept of physical design of the Able-3 and Able-4 payload assemblies was severely complicated by their comparatively large size and weights. The Able-1 lunar probe consisted of three basic elements, namely, the propulsion assembly comprising about one-half of the total weight, the structure-electronic assembly that was one integral assembly, and a very light forward cover.

To overcome some of the difficulties experienced in the utilized construction as in Able-1, convenient interfaces were created and the ground rules were established that the payloads would be constructed of "plug-in" subassemblies. These subassemblies followed closely the sequence of assembly on the missile, and are in the following order:

(1) Lower truss structure assembly,

(2) Propulsion units,

(3) Electronic structure assembly,

(4) Forward cover assembly,

(5) Solar cell assemblies.

The physical requirements imposed on the electronic structure assembly are many. Its primary function is to serve as a "plug-in" electronic chassis. All electronic components, such as transmitters, receivers, batteries, experiment, interconnection wiring, etc., are mounted to this platform and, except for a few interface connections, is completely operable for assembly and checkout purposes.

A second function is to act as a main payload structural stiffener. The design "L" shape is such that it is extremely stiff in any plane of stress and when fastened into the lower truss structure, substantially reinforces the whole assembly.

The material is an electrical nonconductor to eliminate the circulating current problem. It must also be a resiliant material to absorb the vibration transmitted from the propulsion stages.

The material selected is 0.250 thick nylon honeycomb of 0.010 thick fabric, faced on both sides with 0.010 thick fiberglass bonded with a high temperature curing epoxy resin. It was given a subsequent heat treatment to slightly improve its tensile properties.

Holes are cut where required for clearance or for component mounting. The clearance holes are reinforced either with fiberglass and patted with epoxy on a metallic grommet.

For component mounting, special ferrules were required, in two styles. One is a threaded fastener with a nylon insert that would act as a captive nut. The other is similar except without threads for screw clearance. These ferrules are double-sided; two mating halves are inserted from opposite sides of the honeycomb surface and are squeezed together to form a solid compression pad. Epoxy patting compound is then injected in the circular cavities in the honeycomb to rigidly hold it in place.

Strength tests have been conducted on this type of fastener. The weakest configuration is in the cantilevered load method q mounting. Failures occurred at about 100 lb-in. q torque.

c. Temperature Control

Temperature control of the payload is achieved by arranging that the thermal balance between input solar radiation and infra-red emission from the payload occurs at a temperature within the desired operating range. This is done by selecting a surface coating that will give a value of the ratio of solar absorbence to long wavelength emittance (α/ϵ) in the range of about 1.0 to 1.3.

To dissipate heat from the 5-watt transmitter, which dissipates about 40 watts altogether and which operates for long periods of time (of the order of 1 to 2 hours continuously in a four hour period), it is necessary to provide as low a resistance path as possible to the shell so that the heat can be radiated away by a large area of the shell. This is accomplished by providing a good conduction path and taking care to eliminate radiation gaps at the mechanical connections.

The silicon solar cells will be placed on both sides of four paddles which are extended out laterally from the payload during the spin-up and staging operation. The paddles will be oriented in such a way that the power output of the solar cell paddles is relatively independent of sun angle. The cells are quite temperature sensitive, however, and their conversion efficiency drops off about 0.6 per cent per degree centigrade above a nominal value of 25°C. Because the absorptivity-to-emissivity ratio of a silicon cell is about 3, the cells will run excessively hot unless their surfaces are coated to alter the long wavelength emissivity. Thus, glass plates, 0.003 inch thick, cemented individually to each cell, are being used to increase the long wavelength emissivity. In order to protect the cement from ultra-violet radiation and to reduce the cell temperature even more by eliminating a part of the solar spectrum to which the solar cells are functionally insensitive, an ultra-violet reflective coating is applied to the glass before it is cemented to the cell. For the glass plates to reduce the temperature successfully, it is necessary to insure that the glass is bonded to the cell over the whole surface, so that there is no gap to cause a greenhouse effect, which would raise the cell temperature.

The shadowing of one paddle on another and of the paddles and payload on each other is taken into account in achieving proper temperature control in the payload and to determine temperatures on the solar cell paddles.

The large thermal capacity of the payload will prevent large drops in the payload internal temperature for eclipses expected during the first 1000 days of flight. Temperature drops of 15 to 20°F will be the most severe during this time. After about 1000 days, however, long eclipses (about 2-1/2 hours duration) will occur; the payload temperature drop by the end of such an eclipse will be of the order of 50°F.

Temperatures of the solar cells, on the other hand, drop rapidly during an eclipse. While there is, of course, no operating requirement for the solar cells during an eclipse, they must be able to survive the low temperatures. For a one-half hour eclipse, the lowest temperature on the solar cell paddle should be about -50°F and about -175 to -200°F for a 2-1/2 hour eclipse.

d. <u>Stability Requirements</u>

The condition for long-term stability of a spinning payload is that the roll moment of inertia must be greater than the pitch moment of inertia. The radially deployed solar cell paddles and the annular instrument platforms to be used on the Able-3 and Able-4 payloads will have the effect of insuring that this condition will be satisfied. The expected ratio of roll-to-pitch moment of inertia is greater than 1.2 for all three payloads.

The minimum spin rate required to meet the third stage velocity vector specifications is 120 rpm. The maximum allowable spin rate is determined by the allowable centrifugal acceleration on components and structure and is in excess of 180 rpm. Spin is achieved after second-stage cutoff and before separation.

Each flight payload will be dynamically balanced. The allowable angle between principal axis and axis of symmetry is determined by trajectory error considerations. This allowable angle will be approximately 1 degree for the Able-3 and Able-4 payloads.

The Able-3 and Able-4 Atlas payloads will have annular dampers to remove precessional motion of the payload, which would interfere with TV scanner operation.

The solar cell paddles will be deployed after second-stage burnout and before spin-up. This procedure removes the possibility of trajectory errors caused by asymmetrical paddle motion while the vehicle is spinning.

External torques on the spinning payload, such as radiation pressure, magnetic fields, and gravitational gradients, cause an undesirable precessional motion and a decrease in the spin rate. However, calculations of the upper bounds of these effects show that they are negligible for the Able-3 and Able-4 missions.

Aerodynamic forces acting on the Able-3 payload at perigee cause the spin rate to decrease. If the perigee is significantly lower than the expected 150 miles, this effect may reduce the spin to zero in a period of weeks. If the injection rocket is used to raise the perigee above 300 miles, this aerodynamic de-spin effect becomes negligible.

Figures 8 to 11 show the prototype of the Able-3 payload with each component identified.

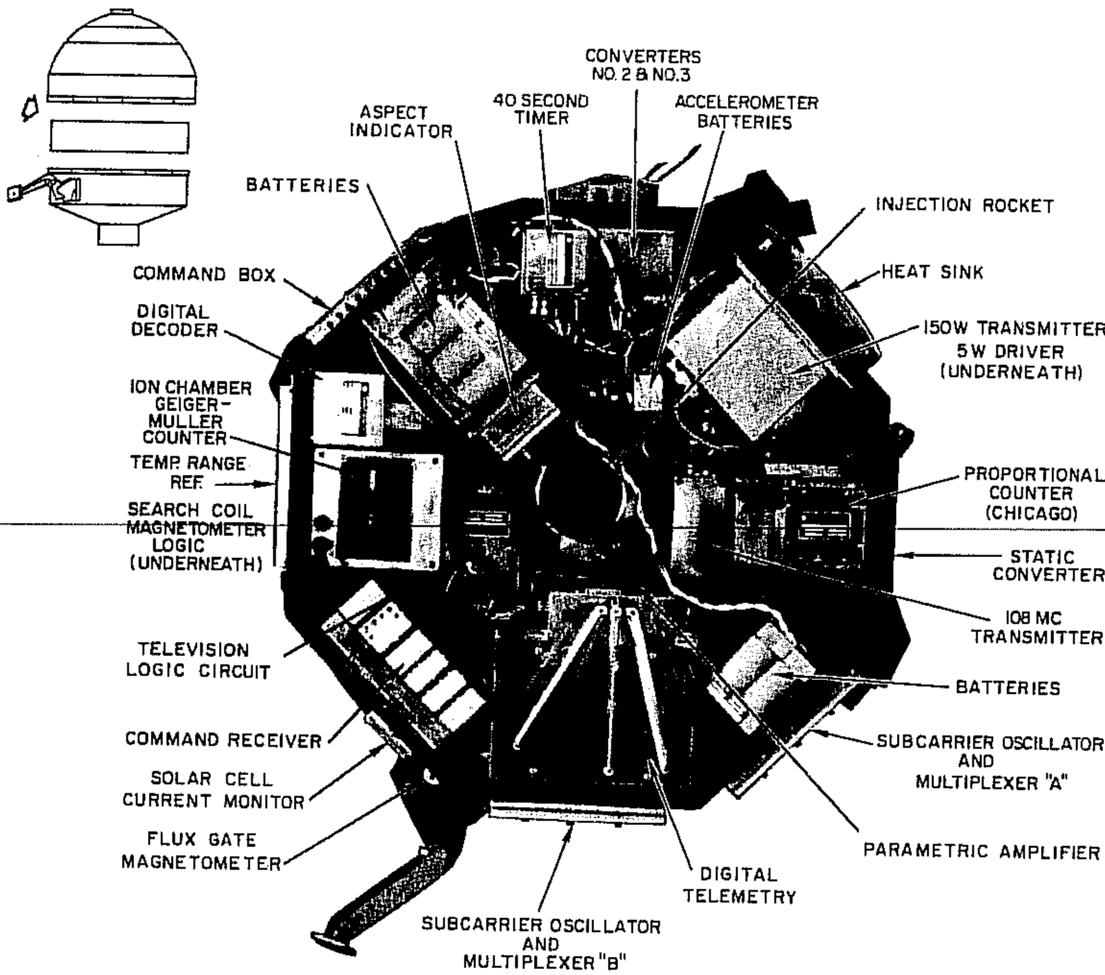


Figure 8. Able-3 Payload (Top View).

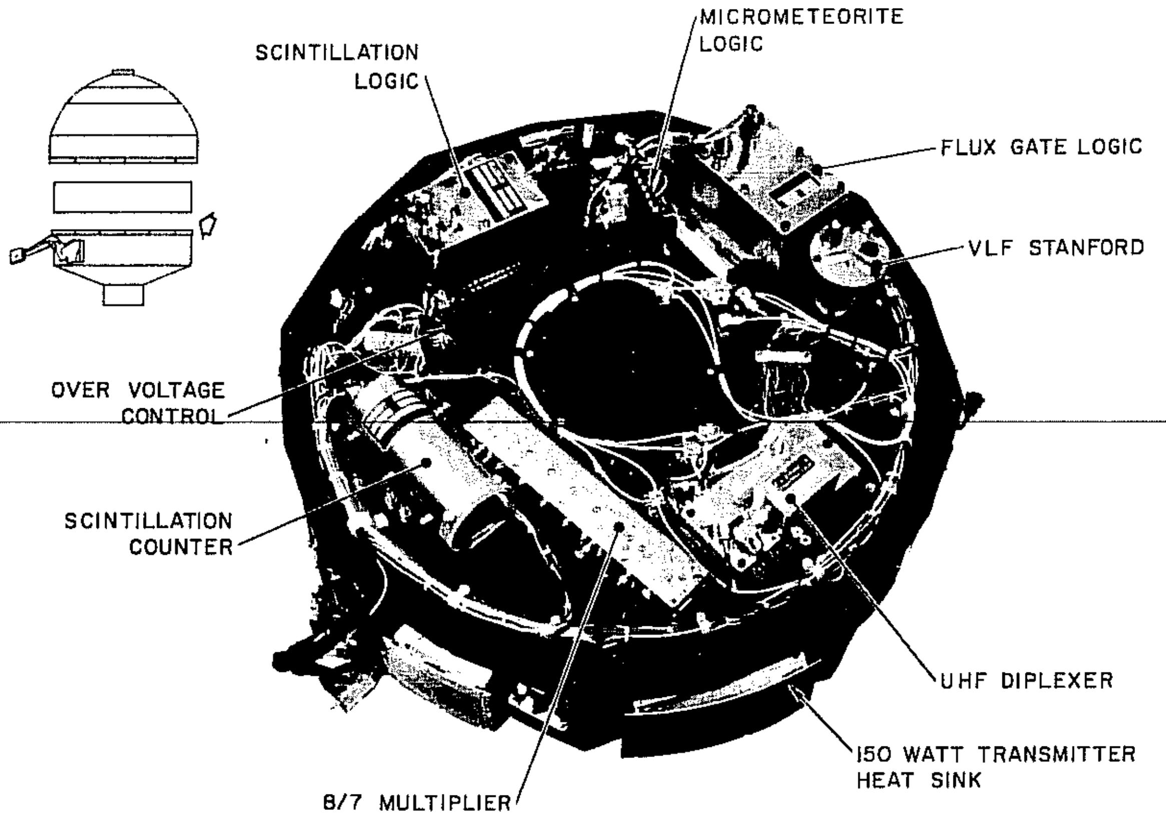


Figure 9. Able-3 Payload (Bottom View).

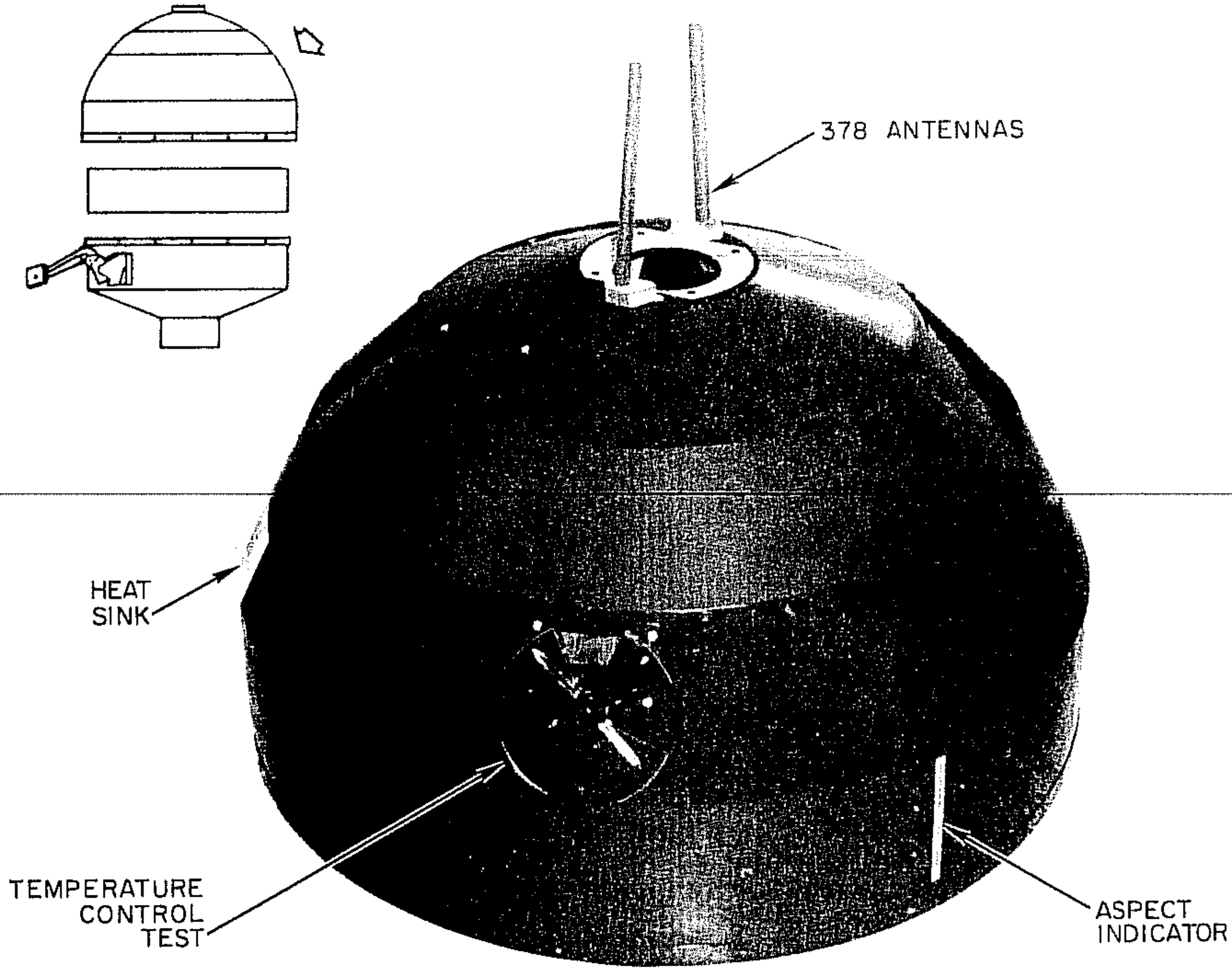


Figure 10. Able-3 Payload Cover.

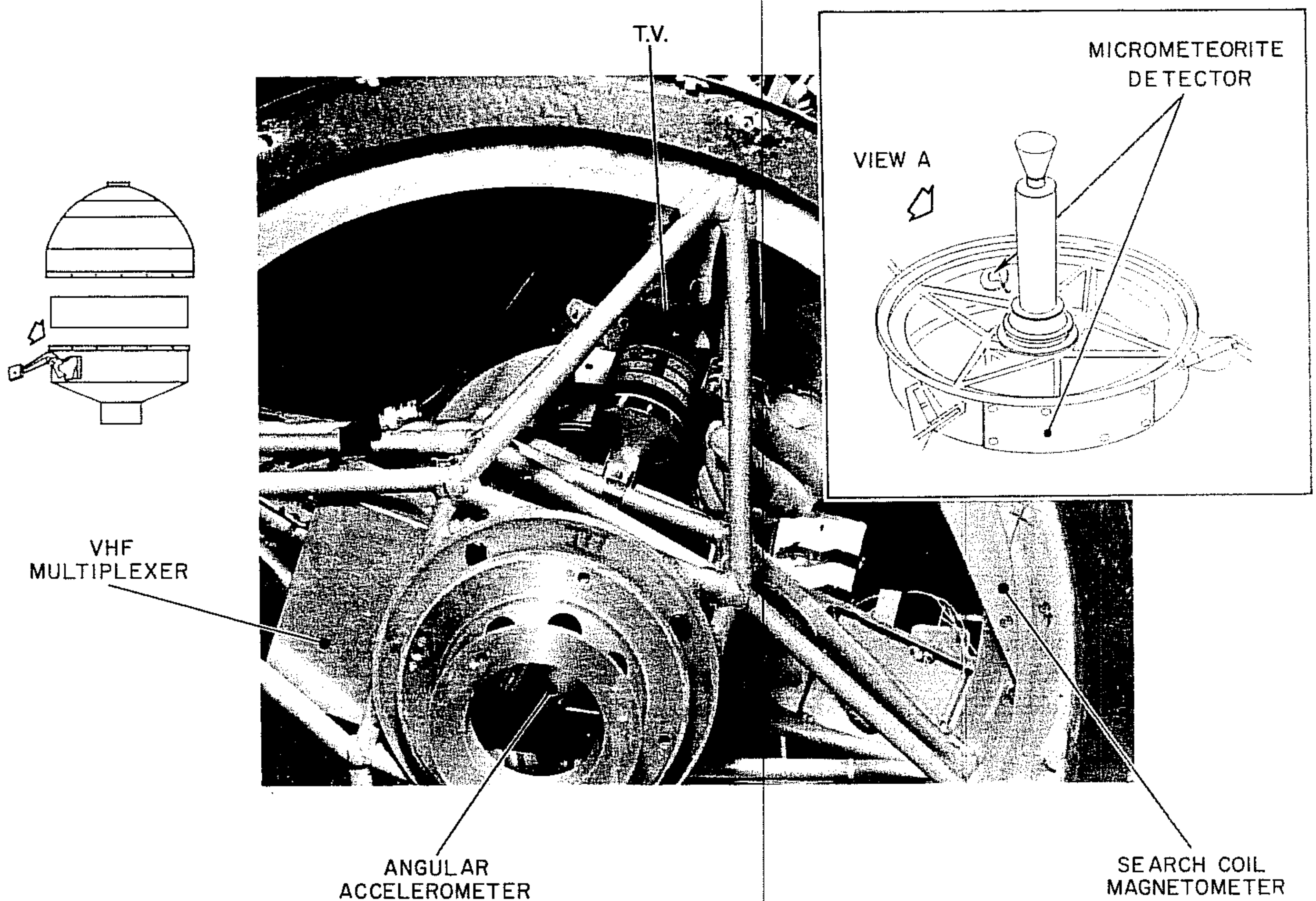


Figure 11. Able-3 Payload and Solar Cell Paddle Structure (View A).

B. Able-4 Thor (Deep Space Probe)

1. Objectives

The mission of the Thor-boosted Able-4 program is to launch a vehicle several million miles into space to the vicinity of the orbit of Venus, where it will become a permanent solar system body in an elliptic orbit with perihelion near the orbit of Venus and aphelion near the orbit of the earth. The launch will gain the following objectives.

First, this flight will be used to gather scientific data, such as a more accurate knowledge of the value of the astronomical unit (semi-major axis of the orbit of the earth), of magnetic field strengths in space, of cosmic ray intensities away from the earth, and of micrometeorite density.

Secondly, this flight will establish the basic communication parameters involved in extremely long-range transmission in space.

Third, this flight will be valuable for gathering environmental information useful for manned interplanetary space flight. The extended flight times, as well as the fact that the sampling is taken over large parts of the solar system, will increase our knowledge of the problem of interplanetary flight enormously.

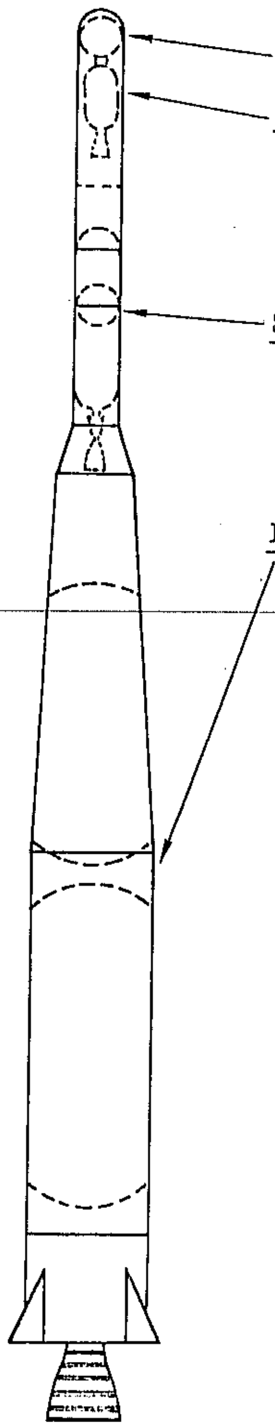
Fourth, this flight will establish the fundamental feasibility and effectiveness of a guidance system for space flight.

Fifth, this flight will evaluate the operation of the second, third, and fourth stages of the Able-4 test vehicle.

2. Technical Summary

The first three stages of the Able-4 Thor vehicle are essentially the same as the Able-3 vehicle (see Figure 12). The payload, however, will be lighter, will not carry an injection rocket, and will contain fewer experiments.

A weight summary of the Able-4 Thor vehicle is given in Table 5, and the present weight breakdown for the payload is given in Table 6.



Payload	80	lb
Third Stage - ABL 248-A-3		
Gross Weight at Burnout	135	lb
Gross Weight at Start	600	lb
Vacuum I_{sp}	250.5	sec
Thrust at Altitude	3,150	lb
Second Stage - AGC 10-101		
Gross Weight at Burnout	1,534	lb
Gross Weight at Start	4,728	lb
Vacuum I_{sp}	271	sec
Thrust at Altitude	7,670	lb
First Stage Thor		
Gross Weight at Burnout	13,978	lb
Gross Weight at Liftoff	111,765	lb
Vacuum I_{sp}	289.7	sec
Sea Level I_{sp}	248.7	sec
Sea Level Thrust	153,248	lb

Figure 12. Able-4 Thor Vehicle.

Table 5. Able-4 Thor Weight Summary.

Stage	Gross Weight (lb)	Expended Weight (lb)	Jettisoned Weight (lb)
Payload	82.5		
3	521.5	465	56.5
2	4,306	3,248	1,058 (including nose fairing)
1	106,754	97,423	9,331
Lift-off weight	111,664		

Table 6. Able-4 Thor Payload Weight.

Item	Weight (lb)
Receiver and control	4.7
Transmitter system	7.7
Telemetry system	9.4
Converters (including heat sink)	5.2
Digital command unit	1.5
40-second timer	0.3
Batteries	16.0
Wire, fasteners, miscellaneous	3.2
Structure	17.3
Solar cells	6.6
Temperature control	1.0
Experiments	9.6
TOTAL	82.5

3. Trajectory Considerations

The free-flight trajectory followed by the payload will be similar to a minimum-energy trajectory to Venus, except that the payload will not come close to Venus because the launch date is not the requisite date. The payload will follow an elliptical orbit around the sun with aphelion near the earth's orbit and perihelion near the orbit of Venus. Such an orbit will be suitable for determining the astronomical unit from radio tracking of the payload and, in addition, will allow scientific data to be gathered over a significant region of the solar system.

In order for the trajectory to reach the orbit of Venus, the payload velocity with respect to the earth, once the payload is free of the earth's gravity, must be approximately 8000 ft/sec. Thus, the third stage burnout velocity must be approximately 900 ft/sec above escape velocity, and therefore a burnout velocity of 36,700 ft/sec is required at a burnout distance from the center of the earth of approximately 22 million feet (200 nautical miles).

The Able-4 Thor can achieve the velocity with an 82-pound payload. Assuming that the vehicle will be launched during a specified seven-day period, a single launch azimuth appears to be sufficient. The launch azimuth of 77.73° (true) probably will be used. On each of the seven possible days for launch, the launch period will be restricted to 30 minutes, to minimize the payload temperature changes resulting from its attitude with respect to the sun. Also, to minimize temperature variations, the guidance constants for the second stage will be changed during each launch period and from day to day.

4. Vehicle Characteristics

a. Description of Vehicle

The Thor-boosted vehicle is comprised of three propulsion stages plus a terminal stage and a nose fairing. Except for the terminal stage configuration, the vehicle is the same as the Able-3 vehicle.

b. Sequencing and Timing

Table 7 shows the complete in-flight sequence of events and indicates the device which initiates each action. A schematic of this sequence is shown in Figure 13.

5. Guidance and Control

a. Thor Stage

The Thor stage will utilize a programmed autopilot for control.

b. Second Stage

Launch control guidance of Stage 2 of both Able-4 vehicles will be accomplished from Cape Canaveral by an experimental guidance system. This system consists of an STL measuring subsystem, the Burroughs J-1 computer, an STL command subsystem, and STL Stage 2 and payload transponder. This system will be operated open-loop in the Able-3 operation to verify the system performance. In addition to Stage 2 guidance, this system will be utilized for payload tracking and in the case of the Able-4 Atlas vehicle for final velocity correction of the payload after Stage 3 burnout.

Guidance geometry for the Able-3 and Able-4 flights is given in Table 8. Guidance requirements for these flights in terms of the required accuracy of the measuring subsystem are given in Table 9. Figure 14 is a block diagram of the system.

c. Measuring Subsystem

The measuring subsystem is a rate and position interferometer consisting of the following units:

(1) One 1000-watt transmitter and transmitting antenna.

(2) Three receivers and receiving antennas forming the two base lines of the interferometer.

(3) One digital data extraction equipment.

This equipment is mounted in five trailers.

Table 7. Project Able-4 Thor In-Flight Sequence

<table>
<tr><th>Time (Seconds)</th><th>Stage in which Event Occurs</th><th colspan="2">Action</th><th>Equipment Initiating Action</th></tr>
<tr><td>X + 0</td><td>I</td><td colspan="2">First Motion</td><td></td></tr>
<tr><td rowspan="6">X + 0.1</td><td rowspan="3">I</td><td colspan="2">Lift-off Switch Activates</td><td>Microswitch</td></tr>
<tr><td colspan="2">Programmer Starts</td><td>Relay - CEA</td></tr>
<tr><td colspan="2">Gyros Uncaged</td><td>Relay - CEA</td></tr>
<tr><td rowspan="3">II</td><td colspan="2">Electrical Umbilical Ejects</td><td rowspan="2">Lift-off Signal Operates Releases w/lanyard Backup from Umbilical Mast</td></tr>
<tr><td colspan="2">Helium Umbilical Ejects</td></tr>
<tr><td colspan="2">Arm Destruct Initiator - Stage II</td><td>Lanyard from Umbilical Mast</td></tr>
<tr><td>X + 2</td><td>I</td><td colspan="2">Roll Program Initiated</td><td rowspan="8">Douglas Programmer Relays (CEA)</td></tr>
<tr><td>X + 9</td><td>I</td><td colspan="2">Roll Program Complete</td></tr>
<tr><td rowspan="2">X + 10</td><td rowspan="2">I</td><td colspan="2">Pitch Program Initiated</td></tr>
<tr><td>1st Step Pitch Rate</td><td rowspan="4">As Required for Trajectory and detailed in DTO</td></tr>
<tr><td>X + 28</td><td>I</td><td>2nd Step Pitch Rate</td></tr>
<tr><td>X + 70</td><td>I</td><td>3rd Step Pitch Rate</td></tr>
<tr><td rowspan="2">X + 98</td><td rowspan="2">I</td><td>4th Step Pitch Rate</td></tr>
<tr><td colspan="2">Autopilot Gain Change</td></tr>
<tr><td rowspan="2">X + 140</td><td>II</td><td colspan="2">Second Stage Engine Start Circuit Armed R1-</td><td rowspan="2">Acceleration Switch Set for 6.7 ± 0.1 g's + 0.8 ± 0.3 sec</td></tr>
<tr><td>R1-</td><td colspan="2">Start 46 sec Time Delay Relay</td></tr>
</table>

Table 7. Project Able-4 Thor In-Flight Sequence (Continued)

Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
X + 140	I	Pitch Program Complete	Douglas Programmer Relays (CEA)
		Programmer Output 0°/sec	
X + 149	I	Main Engine Cutoff (MECO) Circuitry Armed	
X + 158	I	MECO Armed	Propellant Float Switch Set to Operate When Either Tank has 3-1/2 sec. of Fuel Remaining
		Vernier Tanks Re-pressurized	
X + 162*	I	First Stage MECO	80% Chamber Pressure Switch Initiates Shutdown but Does not Send Signal to Center Engine
	II	Start Staging Sequence	Relay Closure in First Stage Generated by 90% First Stage Chamber Pressure Switch
		Blow Blast Doors in Adapter Section	Relay Activated by the Start Staging Sequence Signal
		Start 2-sec Time Delay (Tolerence ± 0.2 sec)	
X + 164	II	Engine Fire Signal	Relay in Relay Junction Box (RJB)
		Uncage Gyros	
		Start Pitch Program About 0.1°/sec Down	

*This time and all times following are based on the reference trajectory.

Table 7. Project Able-4 Thor In-Flight Sequence (Continued)

Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
X + 164 +	II	Blow Separation Bolts	Chamber Pressure Switch Set to Operate at 60% of 2nd Stage Engine Thrust (TPS w/TVS_2 Backup)
		Start 10-sec Timer	
X + 164.3	II	2nd Stage Separation Complete	
X + 174 Approx.	II	Pyrotechnic in Helium Tanks is set off	10-sec Timer Started at TPS (Tolerence ± 1 sec)
		Cutoff Enable	
X + 185		Jettison Nose Fairing	46 sec Timer Started at Arm (Tolerence ± 4 sec)
Times vary in accordance with commands	II	Stop Pitch Program and* Start Pitch Command ± 1°/sec	Radio Command: Polarity Determined by Command
		Stop Pitch Command Pitch Rate Zero deg/sec	Radio Command
		Start Yaw Command ± 1°	Radio Command: Polarity Determined by Command
		Stop Yaw Command Yaw Rate Zero deg/sec	Radio Command

*These commands may be applied in arbitrary order. Steering commands should cease one second prior to command cutoff (i. e., stop pitch and stop yaw commands must be sent by this time).

Table 7. Project Able-4 Thor In-Flight Sequence (Continued)

	Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
	X + 275	II	Cutoff Stage II Engine	Radio Command with Propellant Depletion (TPS) Backup
			Stop Pitch Program (if not stopped by command)	
			Turnoff SGS Transmitter	
			Start 2-sec Timer	
			Start 3.1-sec Timer	
R1		III - IV	Release Paddles	
R1	X + 277	IV	Turn on SGS Transmitter	Paddles in Place Due to 0 g's
R1		II	Cage Autopilot Gyros and Ignite Spin Rockets	2-sec Timer (Tolerance $^{+0}_{-0.2}$ sec)
	X + 278.1	III	Ignite Stage III Motor	2.1-sec Timer (Tolerence $^{+0.2}_{-0.0}$) sec
			Blow Stages II/III Explosive Bolts and Nozzle Shroud	
	X + 315	III	Rocket Motor Burns Out	Depletion of Propellant
	X + 425	III	Separation Bolt Blows to Separate Stages III and IV	Radio Command
		III & IV	Separation Occurs	Separation Spring

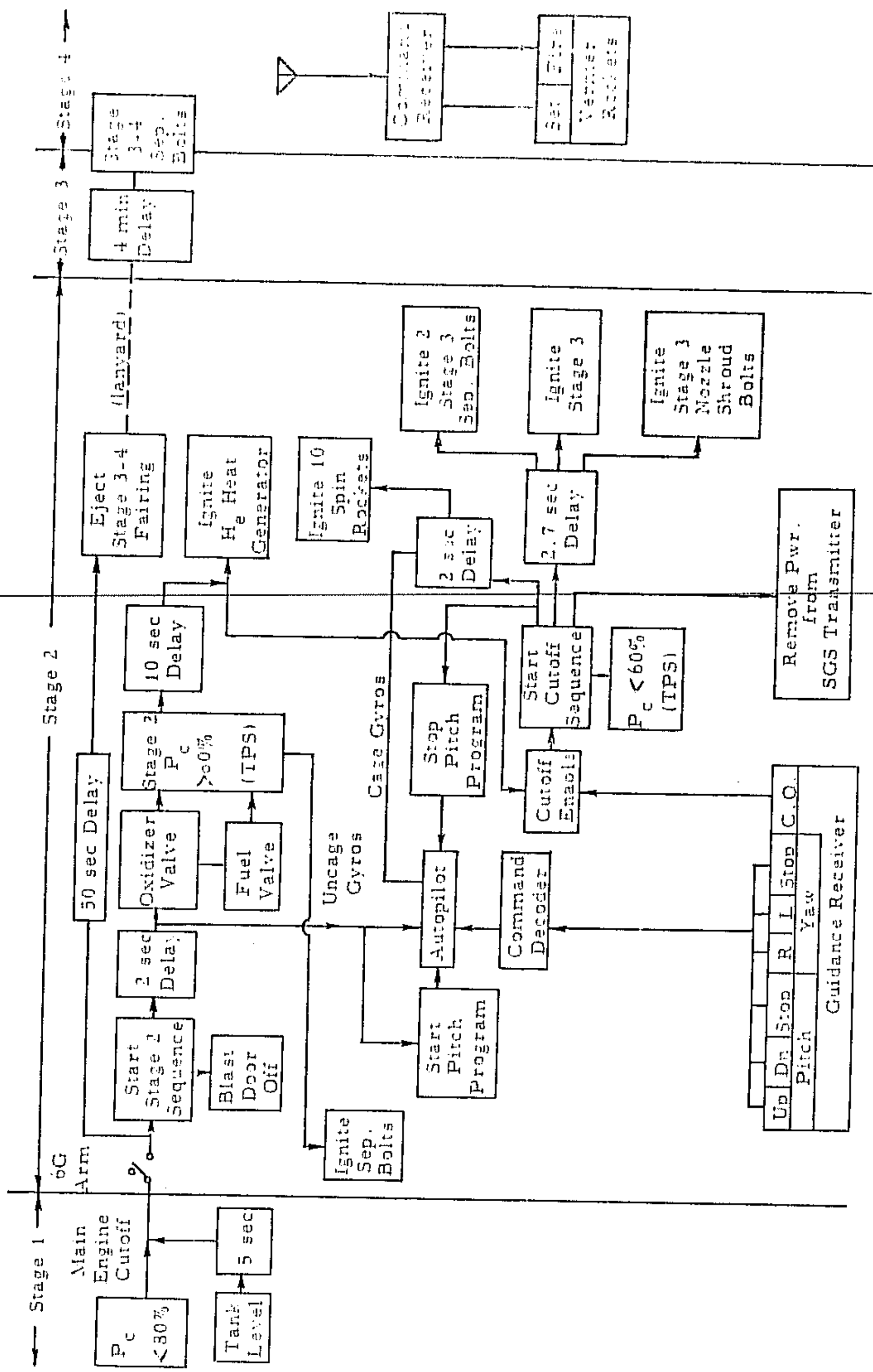


Figure 13. Able-4 Thor Flight Sequence.

Table 8. Guidance Geometry from Cape Canaveral.

		Able-3	Able-4 Atlas	Able-4 Thor
Stage 2 Burnout	Sec after L.O.	276.1 sec	382.2 sec	278.0 sec
	Elevation	15.55°	10.9°	17.0°
	Range	2.711×10^6 ft	4.965×10^6 ft	2.799×10^6 ft
	Height	0.884×10^6 ft	1.476×10^6 ft	0.982×10^6 ft
Stage 3 Burnout	Sec after L.O.	314.7 sec	419.1 sec	317.0 sec
	Elevation	11.03°	8.96°	14.46°
	Range	3.172×10^6 ft	6.118×10^6 ft	3.830×10^6 ft
	Height	1.015×10^6 ft	1.772×10^6 ft	1.268×10^6 ft

Table 9. Accuracy Requirements for Position Measuring Subsystem, Able-3 and Able-4.

Stage 2	Payload
$\delta A \leq 2.0$ mils	2.0 mils
$\delta E \leq 0.5$ mils	0.5 mils
$\delta R \leq 2$ naut mi	not applicable
$\delta \dot{A} \leq 10$ μrad/sec	10 μrad/sec
$\delta \dot{E} \leq 10$ μrad/sec	10 μrad/sec
$\delta \dot{R} \leq 1^*$ ft/sec	1^* ft/sec

* Neglecting ionospheric errors which in daytime may be 2 to 3 ft/sec.

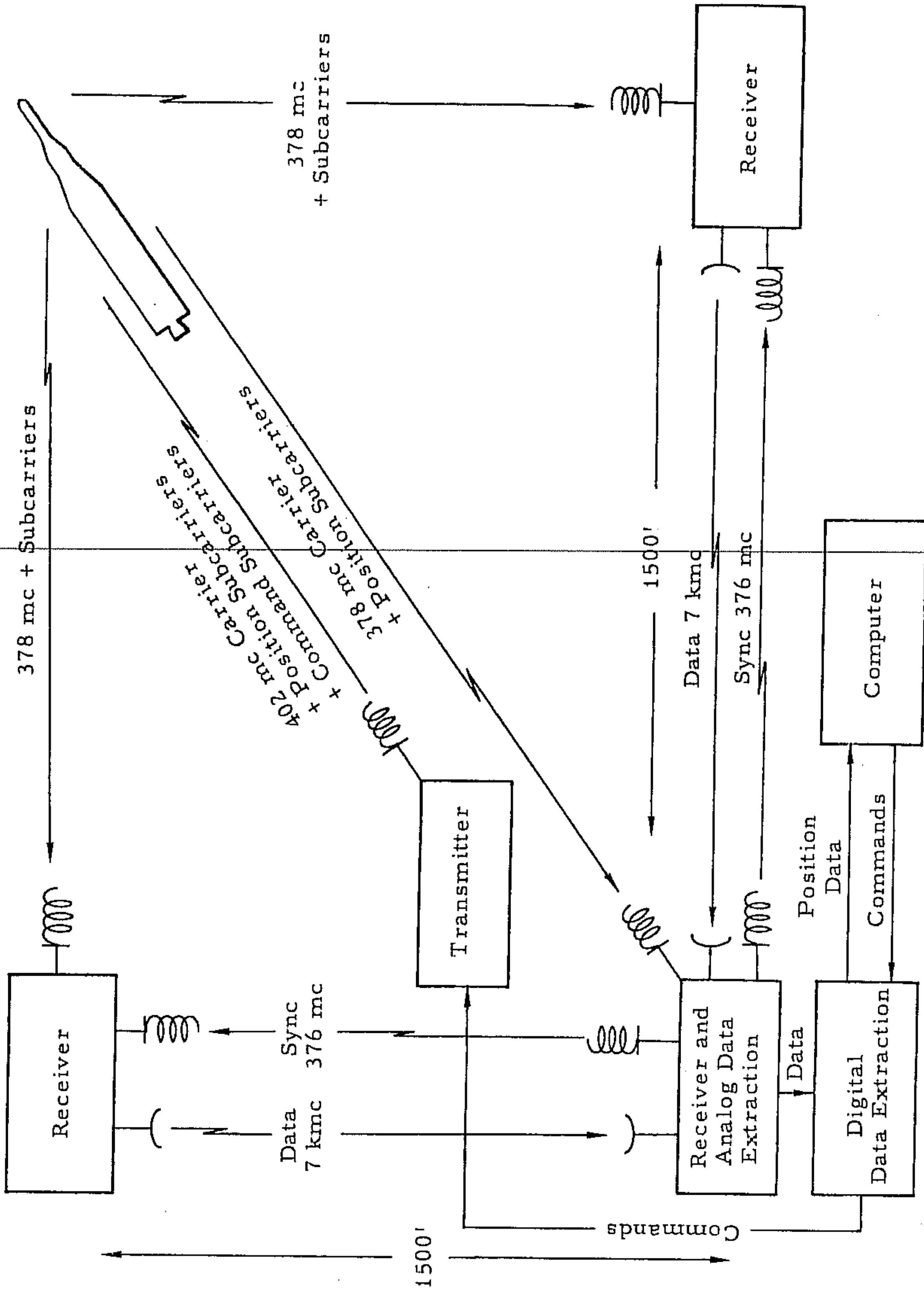


Figure 14. Space Guidance System.

The central receiver of the interferometer provides the synchronizing signals for the two outlying receiver stations by means of a 376-mc synchronizing link. Microwave links transmit the received signal information from the outlying receivers to the analog data processing equipment of the central receiver station. The analog signals are processed by the digital data extraction equipment to yield position and velocity information. Position information (range and two-direction cosines) is derived from phase data. Velocity (range rate and angular rates) is derived from the carrier doppler frequency data. Subcarriers at 19.53-kc and 137.3 cps frequency modulated on the 402-mc ground transmitter frequency are employed for determination of range and resolution of range and angle ambiguity. These subcarriers are demodulated by the missile-borne transponder and are remodulated on the 378-mc transponder carrier frequency. To provide for additional angle ambiguity resolution, a single sideband subcarrier, 4-mc above the 378-mc transponder carrier, is generated in the missile-borne equipment. During tracking of the payload transponder, the subcarriers are not employed on either the ground or airborne transmissions.

d. Computer and Command Subsystem

A command system for automatically controlling the Able-4 missiles will be tested during the Able-3 launching, but it will not actually control the missile. The system has as inputs digital outputs of the digital data extraction equipment specifying range and velocities in an r, p, and q nonorthogonal coordinate system. The system automatically generates command tone subcarriers (2 tones per command) for control of the second stage and automatically generates digital commands for controlling vernier firing at the end of the Able-4 Atlas third stage. The command system consists of digital buffering equipment, a digital guidance computer (Burroughs Mod I (J-1), a command tone generator, and a digital command generator.

The digital computer is used to assemble, edit, and transform inputs to a suitable coordinate system. It then determines errors in position and velocity and generates output signals to control the command generators so that they in turn will generate the correct tones and digital commands for missile control.

The output of the tone command units are used to control the phase modulation on the transmitter so that the commands are sent to the missile with the regular transmitter.

6. Tracking and Communications

General Considerations

Tracking to burnout will be carried out as for Able-3, and, in general, free flight tracking will be the same except for the limitations imposed by the enormous distances involved and the consequent requirement for increased power. The increased power requirement makes it necessary that the duty cycle of the airborne transmitter be reduced. The shorter duty cycle, coupled with the rotation of the earth, sets particular requirements on the ground system. Moreover, due to the great distances involved, only the 250-ft antenna at Manchester, England, will be capable of receiving from the vehicle a month or so after launch.

In addition, because of payload weight limitations, the airborne portion of the tracking and communications equipment will be reduced by eliminating the analog telemetry (which is not effective at interplanetary distances), reducing the weight of solar cells by about 50 per cent, the weight of batteries by about 15 per cent and the cabling by about 50 per cent. In addition, there will be no injection rocket, no vernier rockets, and no angular accelerometer and associated power supply.

7. Payload Vehicle

a. Structure

The payload configuration is similar to the Able-3 payload structure except that four machined arms are used instead of the welded truss structure. These arms terminate at the outer edge of the payload platform at the forward end for the solar paddle hinges and intersect to form a cruciform at the aft end, where they are connected to the third to fourth interstage structure.

b. Experiments

The following experiments will be carried in the Able-4 Thor payload and are identical with those carried in Able-3.

(1) Proportional counter,

(2) Geiger-Muller and ion chamber,

(3) Search-coil magnetometer and aspect indicator,

(4) Micrometeorite detectors,

(5) Temperature measurements.

c. Payload Temperature Control

A vehicle traveling from the earth to the orbit of Venus experiences an increase of solar energy by a factor of about 1.92. This would normally cause an increase of 80 to 100°F in payload temperature over the course of the flight. An active temperature control scheme would be most desirable, but because of the critical weight problem in the Able-4 Thor payload a passive scheme was selected. This scheme takes advantage of knowledge of sun look-angle during the flight, or, more specifically, the sun look-angle as a function of distance from the sun. The payload shell is then coated so that it has a relatively high α/ϵ, as seen from the angle of the sun, when it is farthest from the sun (at the earth) and has a relatively low α/ϵ, relative to the new sun position, when it is at the point of the trajectory closest to the sun. By properly determining the coating pattern it is possible to keep the temperature range within desired limits.

The large amount of power dissipated by the 150-watt transmitter during its short operating time requires the presence of a heat sink to store the dissipated energy until it can be radiated away, at a much slower rate, from the payload shell. The transmitter-heat sink assembly and converter-heat sink assembly will be connected to the payload shell to provide as low a thermal resistance path as possible.

As the payload travels closer to the sun, the temperature of the solar cells increases, thereby reducing the efficiency of the cells. The increased solar input more than compensates for the reduction in efficiency, however, so that there is a significant net gain as the sun is approached.

d. Stability Requirements

The stability requirements for the Able-4 Thor are similar to those discussed in the Able-3 section on stability requirements, except for those paragraphs specifically related to the Able-3 and Able-4 Atlas.

C. Able-4 Atlas (Lunar Satellite)

1. Objectives

The Able-4 Atlas mission has two primary objectives. First, the flight will be used to gather scientific data such as magnetic field strengths, cosmic ray intensities, and micrometeorite densities in the near vicinity of the moon; also, once in lunar orbit a precise determination of the mass of the moon can be made. Second, the flight will establish the fundamental effectiveness of a guidance system for space flight, and, in particular, demonstrate the general value of the proposed midcourse guidance in providing a nearly circular, close-in lunar satellite.

2. Technical Summary

Both the first and fourth stages of the Able-4 Atlas vehicle (see Figure 15) differ from the other Able vehicles. The first stage* will consist of a Series E Atlas ICBM modified to accept the Able second stage at its nose cone interface. The large thrust-to-weight ratio of the Atlas permits a heavier and hence more sophisticated payload for this mission than for any space vehicle yet launched. A weight summary of the total vehicle is given in Table 10.

The 364-pound fourth stage includes a hydrazine monopropellant engine which by incorporating fore and aft nozzles will be used for both terminal orbit injection and for midcourse velocity corrections. Payload weights are given in Table 11.

The external requirements on the trajectory together with the desire for a tight, safe satellite orbit about the moon dictate that a low velocity orbit be flown on September 6, 7, or 8. The nominal satellite orbit is so oriented that the vehicle will be in orbit over a month before the first

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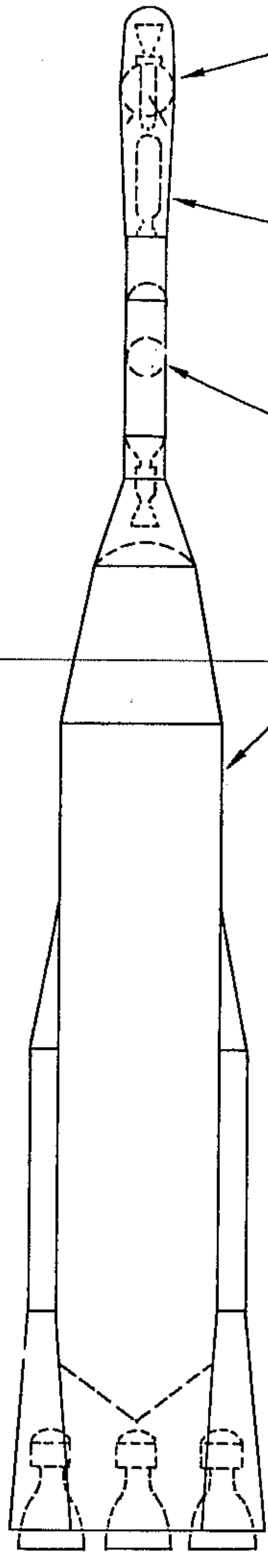
* Because the Atlas jettisons its main engine and continues on its vernier engines, it is sometimes referred to as a 1-1/2 stage vehicle. Here, however, it will be considered a single stage.



Fourth Stage

Gross Weight at Burnout	202	lb
Gross Weight at Start*	364	lb
Vacuum I_{sp}*	225	sec
Thrust at Altitude*	4,200	lb

Third Stage - ABL 248

Gross Weight at Burnout	380	lb
Gross Weight at Start	845	lb
Vacuum I_{sp}	250.4	sec
Thrust at Altitude	3,150	lb

Second Stage - AJ 10-101

Gross Weight at Burnout	1,779	lb
Gross Weight at Start	4,973	lb
Vacuum I_{sp}	271	sec
Thrust at Altitude	7,670	lb

First Stage - Atlas Sustainer

Gross Weight at Burnout	14,095	lb
Gross Weight at Booster Jett.	45,907	lb
Vacuum I_{sp}	310	sec
Vacuum Thrust	79,640	lb

Atlas Booster

Gross Weight at Burnout	55,800	lb
Gross Weight at Liftoff	267,720	lb
Vacuum I_{sp}	293	sec
Sea Level I_{sp}	243	sec
Sea Level Thrust	357,400	lb

*Assuming a Thiokol TE-80.

Figure 15. Able-4 Atlas Vehicle.

Table 10. Able-4 Atlas Weight Summary.

Stage	Gross Weight (lb)	Expended Weight (lb)	Jettisoned Weight (lb)
Payload	364		
3	525	465	60
2	4,218	3,259	959
1	265,279	248,061	8,414
			8,804 (including booster stage and nose fairing)
Lift-off weight	270,386		

Table 11. Able-4 Atlas Payload Weight.

Item	Weight (lb)
Structure	43.2
Hydrazine system structure	43.0
Hydrazine, propellant	138.0
Nitrogen, pressurization	8.0
Receiver and control	9.4
Transmitter system	15.4
Digital decoder	4.1
Digital telemetry	12.4
Command assembly, including timer	2.0
Converters (including heat sink)	9.7
Wire fasteners, miscellaneous	6.4
Batteries	34.0
Solar cells	14.0
Temperature control	5.0
Experiments	19.4
TOTAL	364.0

eclipse occurs. The maximum duration of eclipses when they do occur on this orbit is less than 70 minutes. After a 2.6-day flight from the earth, the satellite orbit begins beneath and climbs up behind the moon and then down across the visible face at about a 42-degree angle with respect to the equatorial plane. The nominal orbit is roughly circular with a radius equal to four moon radii (approximately 4000 nautical miles).

During the Atlas portion of powered flight, the orbit will be guided by GE Mod III radio guidance. The Able stage will be guided by the STL Space Guidance System coupled with the Burroughs Mod I guidance computer. Third stage is spin stabilized and the orientation of the spin axis determines the thrust direction of the third stage. The accuracy of the determination of vector velocity after third stage burnout will be approximately 10 ft/sec in magnitude and 0.1 degree in direction, and within the first hour the trajectory knowledge has improved to better than 1 ft/sec. The effects of dispersions introduced by the third stage can, in principle, be completely corrected to the accuracy of the measurements by vernier corrections during the first day. Therefore, the probability of success is almost one, excluding equipment malfunction.

Tracking and communications will be the same as in Able-3 except that the 150-watt transmitter will not be used.

By using an Atlas-boosted vehicle, a payload of about 364 pounds will be put into orbit about the moon. Approximately half of the payload must be allocated to a retrorocket.

Table 11 shows an approximate payload weight breakdown.

The component weights as shown include the weights for all necessary mounting hardware, special cables, connectors, and temperature control equipment particular to a piece of equipment. The weight of converters for voltages not supplied from the standard converter is also charged to component weight. All required analog-digital converters are also an integral part of the experiment weight.

The payload transponder consists of a coherent receiver, and a command decoder. The ground system for midcourse and terminal guidance will utilize the 250-foot antenna Jodrell Bank, England. The transmitter has an output power of 10,000 watts and employs phase modulation for transmission of commands to the vehicle.

The experiments included on these flights are listed in Table 2, Section III, A. They include the radiation meteorites and interplanetary dust, cosmic rays, vehicle temperature measurements, as well as a study of magnetically guided radiation at very low frequencies, and for the Atlas-boosted vehicle a scintillation counter and a TV picture of the moon. It may be possible to determine the mass, period of rotation, and oblateness of the moon from measuring the orbit of the vehicle about the moon.

The power supply will consist of solar cells mounted upon arms extended out from the vehicle and set at angles so that at least two will always be facing the sun. These cells will charge storage batteries to meet the peak power load of 500 watts of input power required for five minutes every ten hours. The payload will be roughly a sphere. An active temperature control system may be required to meet the temperature requirements of the internal components.

3. Trajectory Considerations

a. Selection of Trajectory

The transit orbit from earth to moon was selected to achieve the tightest circular satellite orbit around the moon consistent with a high probability of success. In addition to establishing a tight, safe satellite orbit, the transit orbit had to fit within certain other stringent conditions.

(1) The velocity of approach to the moon must be sufficiently small for the injection rocket velocity increment. This condition essentially determines the burnout velocity.

(2) Range safety requirements set 75 degrees as the minimum launch azimuth. This condition puts a severe limitation on available launch days.

(3) Maximum eclipse time while in satellite orbit should be minimized. Length of time to first eclipse should be maximized. These requirements determine the position of injection relative to the moon.

(4) For payload considerations, the minimum burnout flight path angle must be greater than 68 degrees.

In addition, the transit orbit has been sufficiently biased to guard against 3σ dispersions in the following burnout quantities:

(1) Burnout velocity (after midcourse corrections) = ±1 ft/sec

(2) Burnout flight path angle (after midcourse corrections) = ±0.15 deg

(3) Burnout azimuth angle (after midcourse corrections) = ±0.15 deg

(4) Tip-off = ±10 deg

(5) Uncertainty in injection position (from ΔV_o error) = ±200 nmi

The position of injection has been biased 1000 nautical miles away from the moon to ensure a safe orbit in spite of items (1) through (5),

b. Specific Trajectory

The nominal lunar satellite trajectory (shown in the xy plane in Figure 16 and in the xz plane in Figure 17) resulting from the above considerations has the following burnout parameters:

(1) v_o = 35,490 ft/sec

(2) β_o = 70 deg

(3) α_o = 78 deg (launch)

(4) h_o = 200 naut mi

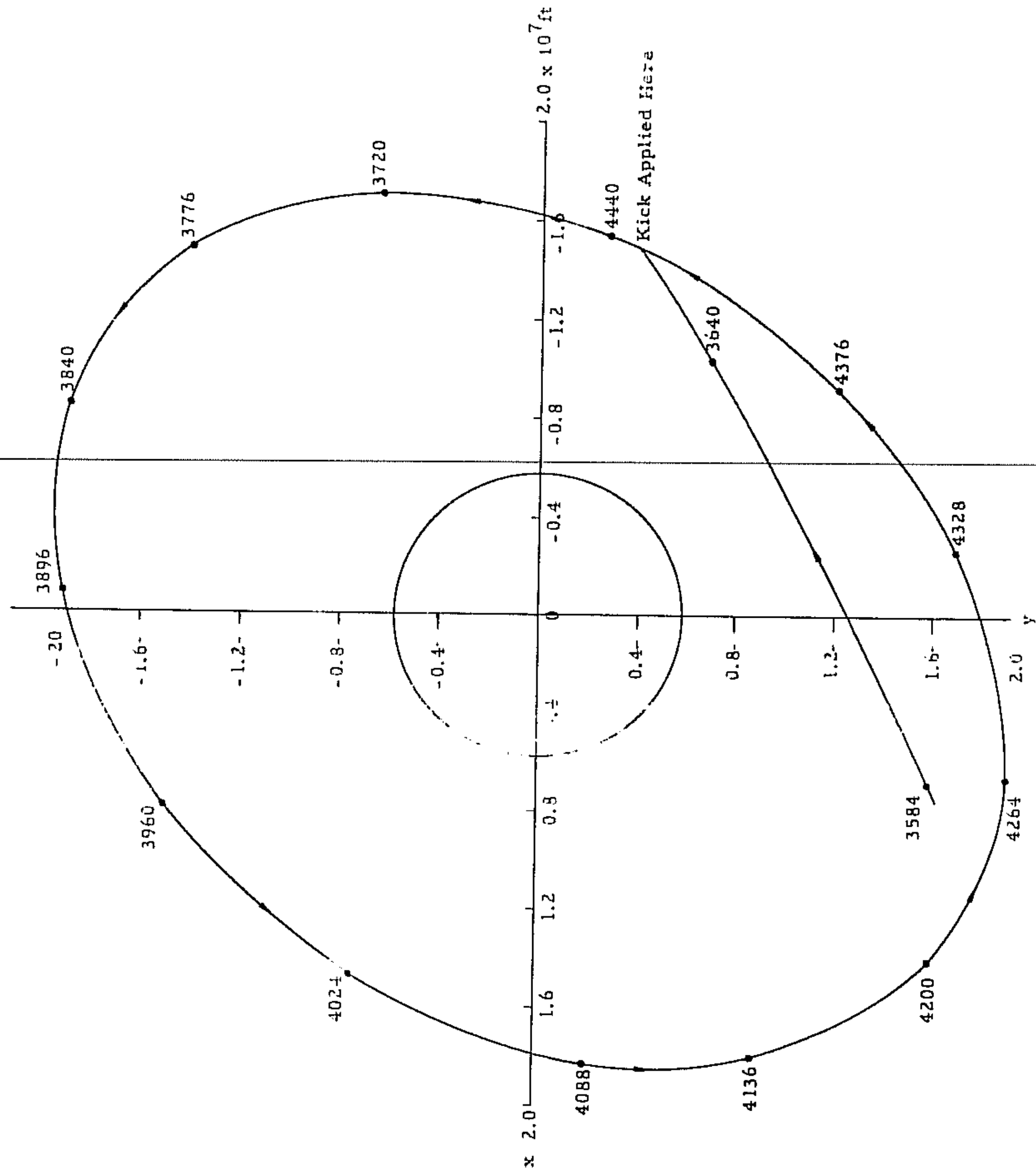


Figure 1[illegible]. Lunar Satellite Orbit in xy Plane.

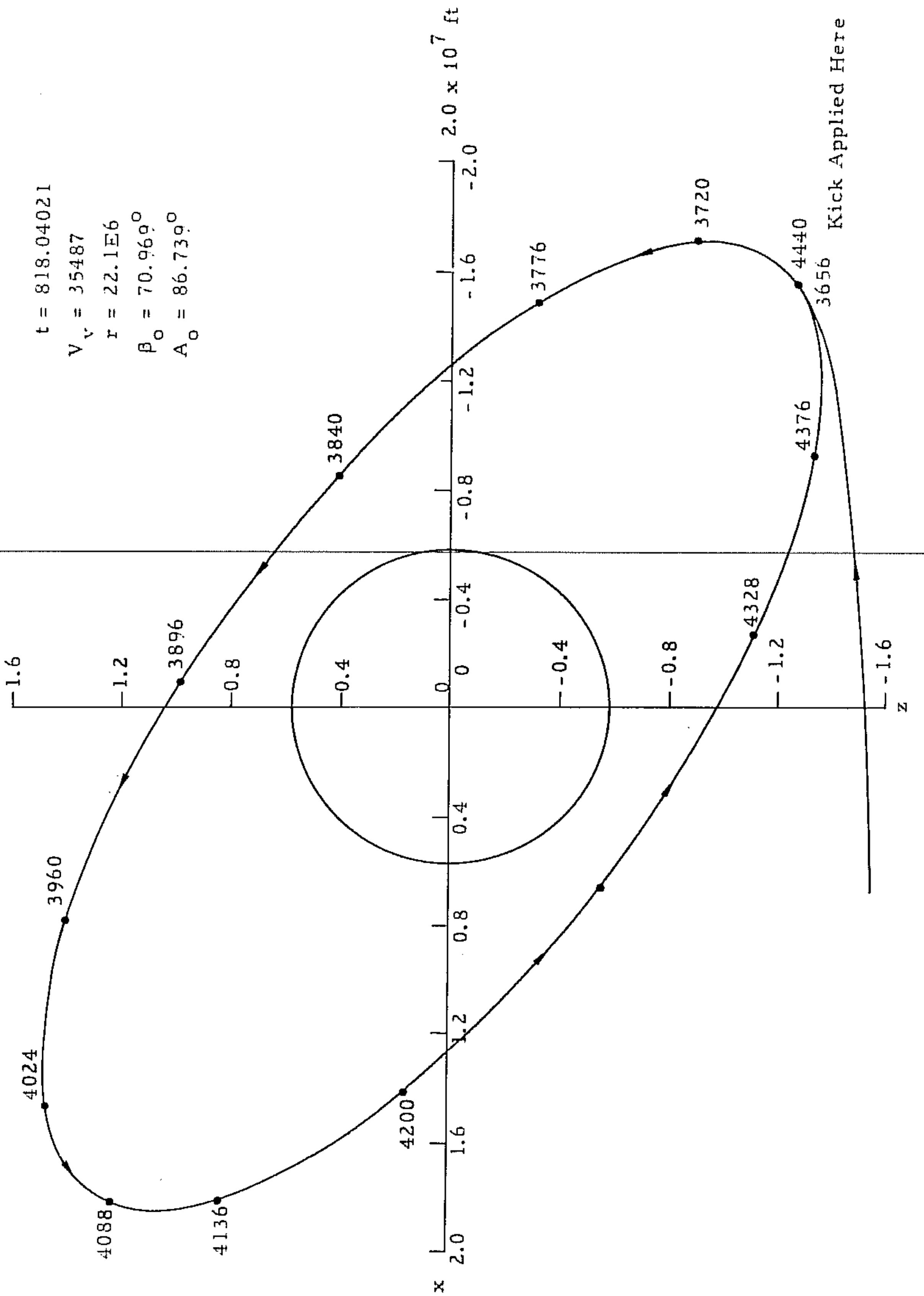


Figure 17. Lunar Satellite in xz Plane.

(5) Miss coefficients (impact parameter sensitivity)

$\Delta b/\Delta \alpha_o$ = 2300 nmi/deg

$\Delta b/\Delta \beta_o$ = 4800 nmi/deg

$\Delta b/\Delta v_o$ = 84 nmi/ft/sec

(6) Time to injection = 720 minutes = 2.6 days

(7) Hold time = 30 minutes

(8) Launch dates = September 6, 7, 8

c. Satellite Characteristics

The nominal transit trajectory results in a satellite orbit with the following characteristics:

(1) Retardation velocity = 3500 ft/sec

(2) Orbital elements

(a) eccentricity = 0.0270

(b) apogee distance = 2.35×10^7 ft = 3900 naut mi

(c) perigee distance = 2.23×10^7 ft = 3700 naut mi

(d) period = 12 hours

(e) equatorial position of ascending node = -127 deg

(f) angle from ascending node to perigee = -76 deg

(g) inclination with respect to equatorial plane = 42 deg

(3) Time to first eclipse 40 days; maximum duration of eclipse = 70 minutes.

The vehicle is injected below and slightly behind the moon. The 3σ dispersions in burnout variables can alter the eclipse situation from nominal to zero days to first eclipse, and maximum duration equals 120 minutes.

d. Launch Time Available

The range safety requirements of $\alpha_o = 75°$ (launch) limits the feasible launch dates severely. To maintain a 30-minute launch period in any given day, the launch dates must be restricted to September 6, 7, and 8. In general, for dates before and after this period, either launch azimuths smaller than 75 degrees must be used or the trajectory must be lofted so that β_o is less than 68 degrees. Figure 18 shows the required variation of α_o and β_o over the 30-minute hold period for September 6. It is assumed that V_o has been held constant. As can be seen from Figure 18, the launch azimuth decreases with time so that the maximum length of hold time is determined by range safety requirements. Orbital sensitivities and eclipses, on the other hand, are essentially unchanged over the hold period on any given day. The length of time to first eclipse does decrease as the launch day moves from September 6 to 8. From all considerations, including safety of the mission, the earlier launch day is preferable.

4. Vehicle Characteristics

a. Description of Vehicle

The Atlas-boosted vehicle is comprised of three propulsion stages plus a terminal stage and a nose fairing. The first stage is a Series C Atlas booster and sustainer. This vehicle has been satisfactory in the Atlas flight test program. The second and third stages are respectively the Aerojet General Corporation AJ 10-101A liquid propellant rocket and the Allegheny Ballistic Laboratory 248 solid propellant rocket. Both of these stages are used in the Able-1, Able-3, and Able-4 Thor vehicles. In addition, a hydrazine rocket propulsion system designed by STL for velocity corrections and final injection is part of the payload.

The first to second interstage structure is similar to the corresponding Able-3 interstage with the exception that the aft transition region has been redesigned to accommodate the Atlas. The forward transition region and stage separation devices are the same as for the Able-3. The second to third interstage is the same as in the Able-3.

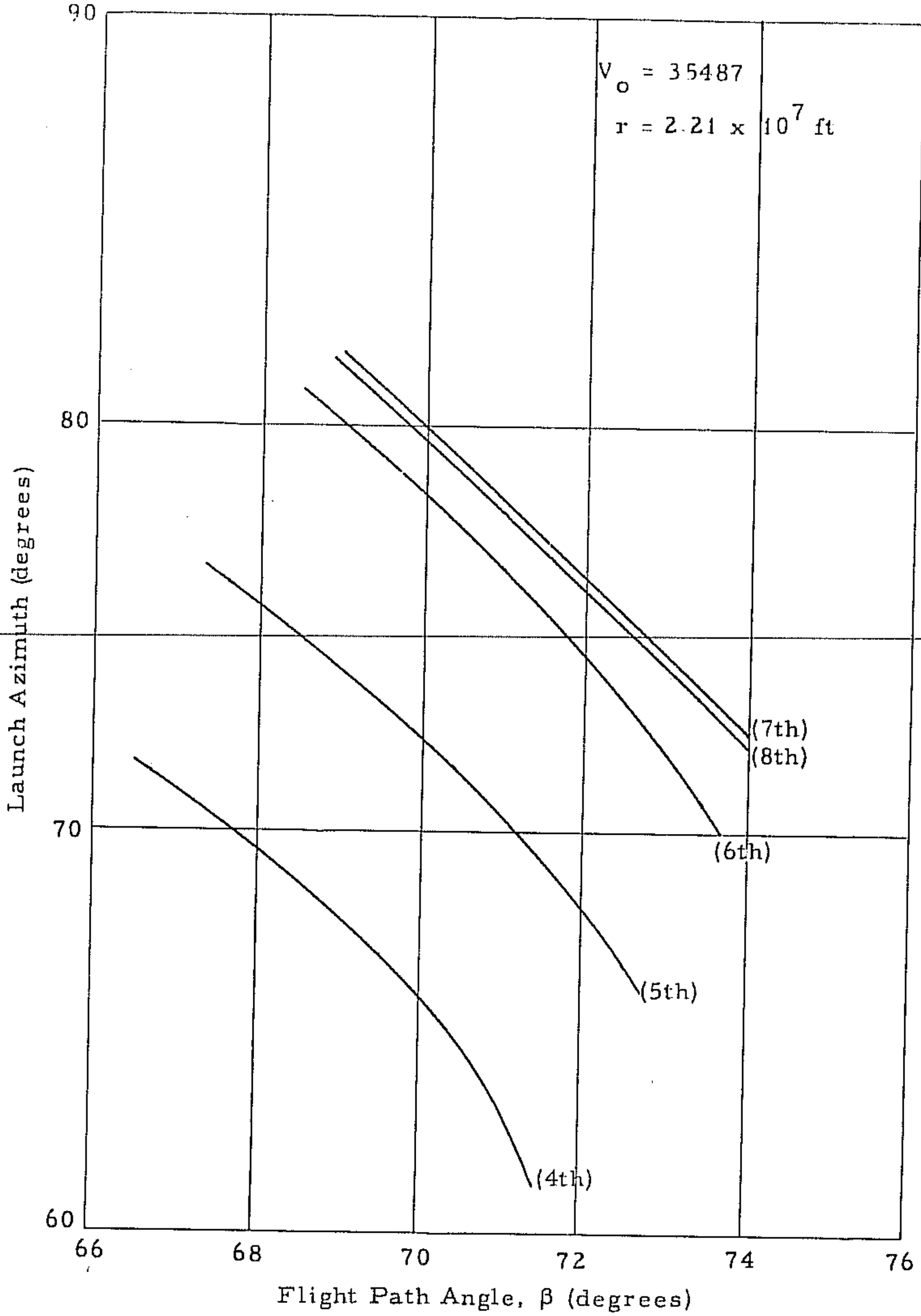


Figure 18. Launch Azimuth as a Function of Flight Path Angle for the 4, 5, 6, 7, and 8 of September 1959.

The payload is supported by a fiberglass honeycomb conical structure. Separation is similar to the Able-3. The payload structure is of built-up sheet metal construction. It consists of an inner cylinder to carry axial and bending loads and rings and webs to which the payload platform and solar paddle hinges are attached.

The nose fairing for this vehicle has been modified to enclose the larger payload. Construction is similar to the Able-3 except for the larger diameter. Separation by means of actuators is the same as the Able-3.

b. Vernier System

The hydrazine monopropellant vernier system will be used to provide a total of 3000 pound-seconds impulse for vernier velocity adjustment. The thrust chamber will be mounted on the spin axis of the payload such that the thrust acts in the same direction as third stage engine thrust. The thrust chamber and nozzle will extend from the nearly spherical payload by approximately eight inches. The equipment consists of:

(1) One thrust chamber assembly with nozzle,

(2) One tank assembly,

(3) One pressurization tank, with regulator,

(4) Four squib-operated start tanks mounted on the thrust chamber assembly,

(5) Four normally-closed, squib-operated start valves,

(6) Four normally-open, squib-operated start valves,

(7) Interconnecting structure and fuel lines.

For each of the four operations of the vernier engine, a separate start and stop signal is required. The desired duration of firing for each operation is determined from tracking data and from the impulse time calibration for the flight engine. The calibration will be determined from vacuum firing data resulting from tests at AEDC.

The amplitude and duration of controlling signals will be selected to provide the maximum, practical reliability, and to increase the probability that the ignition squib wires will not remain shorted and continue to load the battery. In normal operation, each squib should draw current for a maximum of 10 milliseconds. Figure 19 is a sketch of the vernier and injection system.

c. Injection Rocket

The hydrazine monopropellant system will have been developed to meet the injection rocket requirement. The chamber will be mounted on the spin axis of the payload opposing the thrust direction of the vernier engine and will protrude from the nearly spherical payload by approximately eight inches. The equipment consists of:

(1) One thrust chamber assembly with nozzle,

(2) One fuel tank assembly,

(3) One pressurization tank, with regulator,

(4) One squib-operated start tank mounted on the thrust-chamber assembly,

(5) One normally-closed, squib-operated start valve,

(6) Support structure, fuel lines, and servicing fittings.

Items (2) and (3) may, in fact, be common with the vernier system tanks.

One operation only of the injection rocket will be made and the injection-rocket fuel tank will be loaded with the weight of fuel necessary to provide a velocity increment of approximately 3000 ft/sec. The hydrazine injection rocket will have a maximum fuel capacity of 150 pounds.

4. Sequencing and Timing

Table 12 shows the complete in-flight sequence of events and indicates the device which initiates each action.

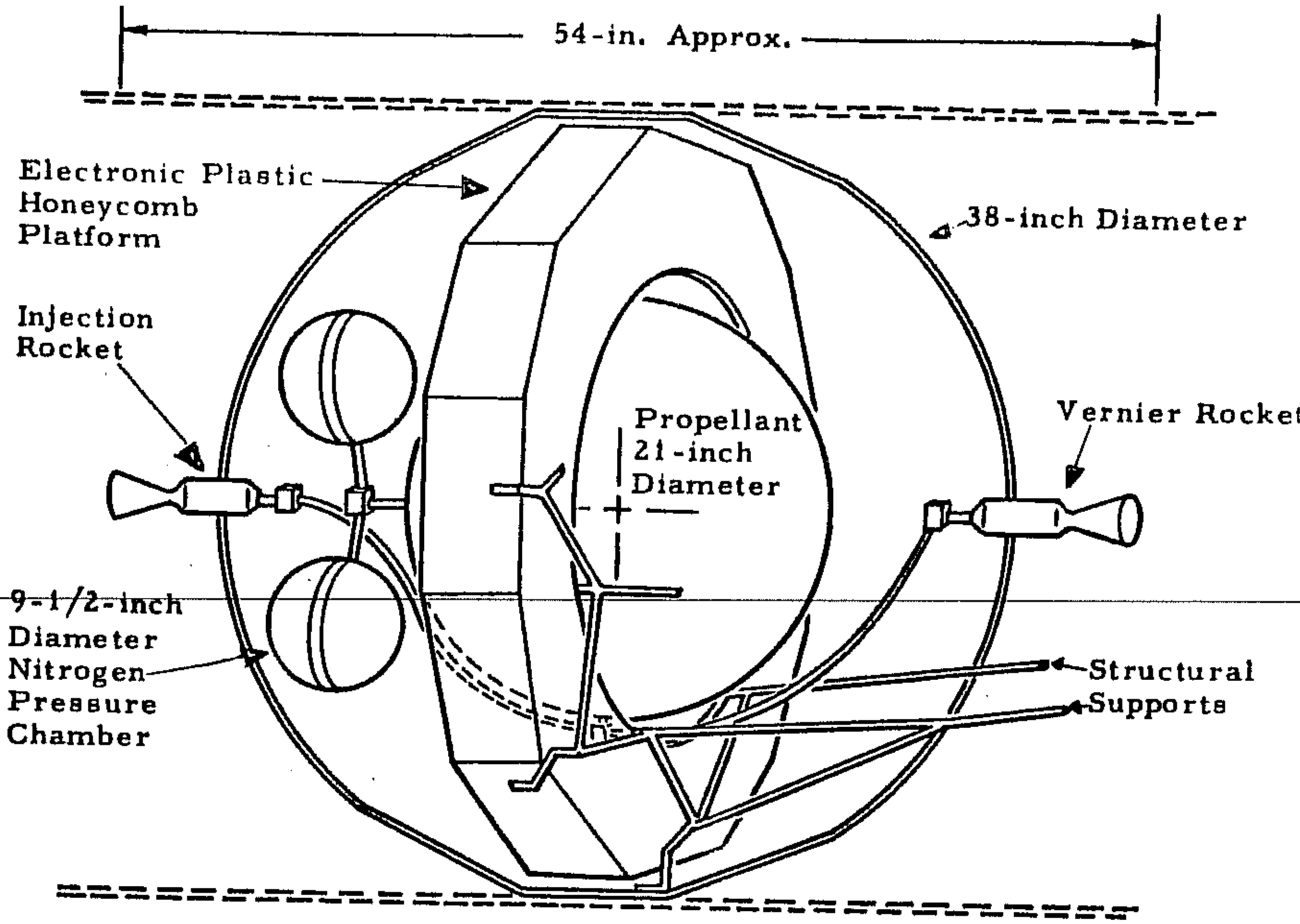


Figure 19. Hydrazine Monopropellant Vernier and Injection Rocket System.

Table 12. Able-4 Atlas In-Flight Sequence.

Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
X + 0	I	First Motion	
X + 0.1	I	Lift-off Switch Activities	Microswitch
		Programmer Starts	Relay (2 in Motion Switch)
		Gyros Uncaged R1	Relay (8-inch Umbilical)
	R1	Close Servo Loops	Relay (42-inch Umbilical)
	II	Electrical Umbilical Ejects	Lift-off Signal Operates, Releases w/Lanyard Back-up from Umbilical Mast
		Helium Umbilical Ejects	
		Arm Destruct Initiator - Stage II	Lanyard from Umbilical Mast
X + 3	I	Roll Program Initiated	Convair Programmer
X + 14	I	Roll Program Complete	
X + 15	I	Pitch Program Initiated (Program described in detail in DTO)	
X + 138	II	Able Stage Engine Start Circuit Armed R1	Acceleration Switch Set for 6.7 ±0.1 g + 0.8 sec ±0.3 sec)
	R1	Start 37-sec Time Delay Relay Tolerance ±3 sec	
X + 144*	I	Booster Engine Cutoff R1	GE Guidance
		Booster Jettison Sequence Starts	

* This time and all times following are based on the reference trajectory.

Table 12 . Able-4 Atlas In-Flight Sequence (Continued).

Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
X + 147	I	Back-up Staging Signal Initiates Above Events	Convair Programmer
R1 X + 144	I	Sustainer Phase Pitch Program Starts Pitch Down 0.12°/sec	Convair Programmer
X + 165	I	Vernier Tank Repres-surization	Convair Programmer
R1 X + 175	II	Jettison Nose Fairing	37-sec Timer Started at Arm
		Start 5-min Timer	Lanyard Attached to Nose Fairing
X + 270	I	First Stage SECO (Sustainer Engine Cutoff) R1	Fuel Pump Outlet Pressure Switch (80%Pc) Initiates Shut-down but does not send Signal to Center Engine
		Pitch Program Stops	
	II	Start Staging Sequence	Relay Closure in First Stage Generated by 80% SECO/backed up by GE Guidance Radio Command (loss of thrust + 2 sec)
		Blow Blast Doors in Adapter Section	Relay Activated by the Start Staging Sequence Signal
		Uncage Gyros	
	R1	Start 2-sec Timer Delay Tolerance ±0.2 sec	
X + 272	II	Engine Fire Signal Start Pitch Program About 0.1°/sec Down	Relay in Relay Junction Box (RJB)

Table 12. Able-4 Atlas In-Flight Sequence (Continued).

Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
R1 X + 272.3	II	Blow Separation Bolts	Chamber Pressure Switch Set to Operate at 60 of Second Stage Engine Thrust (TPS w/TVS$_2$ Back-up)
		Start 10-sec Timer	
X + 272.3+	II	Able Stage Separation Complete	
X + 282 approx.	II	Pyrotechnic in Helium Tanks is Set Off	10-sec Timer Started at TPS
		Cutoff Enable	
Times vary in accord-ance with commands	II	Stop Pitch Program and Start Pitch Command* ±1°/sec	Radio Command: Polarity Determined by Command
		Stop Pitch Command Pitch Rate Zero deg/sec	Radio Command
		Start Yaw Command ±1°	Radio Command: Polarity Determined by Command
		Stop Yaw Command Yaw Rate Zero deg/sec	Radio Command
X + 386	II	Cutoff Stage II Engine R1	Radio Command with Propellant Depletion (TPS) Back-up
		Stop Pitch Program (if not stopped by command)	
		Turn Off SGS Transmitter	
		Start 2-sec Timer	
		Start 3.1-sec Timer	
R1	III - IV	Release Paddles	

* These commands may be applied in arbitrary order. Steering commands should cease 1 second prior to command cutoff (i. e., stop pitch and stop yaw commands must be sent by this time).

Table 12. Able-4 Atlas In-Flight Sequence (Continued).

Time (Seconds)	Stage in which Event Occurs	Action	Equipment Initiating Action
X + 388	II	Cage Autopilot Gyros and Ignite Spin Rockets	2-sec Timer $^{+0}_{-0.2}$ sec
R1	IV	Turn On SGS Transmitter	Paddles Lock Due to Zero g
X + 389.1	III	Ignite Stage III Motor R1	3.1-sec Timer $^{-0}_{+0.2}$ sec
		Blow Stages II/III Explosive Bolts and Nozzle Shroud	
X + 425	III	Rocket Motor Burns Out	Depletion of Propellant
R1 X + 476	III	Separation Bolt Blows to Separate Stages III and IV	5-min Timer Started at X + 175
	III - IV	Separation Occurs	Separation Staging
	IV	Arm Stage IV Igniter Circuits	Switch Actuated by Physical Separation
X + 605	IV	Vernier Engine Set for Proper Velocity Increment	Radio Command
X + 665	IV	Vernier Engine Fired	Radio Command

The Able-4 Atlas flight sequence is the same as that shown in Figure 13 for the Able-4 Thor, except that there is a 40-second delay from the arm to eject Stage III-IV and a 5-minute delay in Stage III.

5. Guidance and Control

a. Atlas GE Mod III Guidance

The Atlas portion of the Able-4 powered flight will be guided by a GE Mod III radio guidance system at Cape Canaveral. Guidance equations used are identical to those used for ICBM guidance. Several virtual targets are selected during the half-hour launch interval in order to reduce the amount of maneuvering required of the upper stages.

b. STL Space Guidance System

(1) Stage 2 and Stage 3

The Able stage will be guided using the STL Space Guidance System coupled with the Burroughs Mod I guidance computer at AFMTC (see Able-4 Thor discussion, Section III, B,5). This system will steer and shut-off the Able stage and control the initial attitude of the spin stabilized third stage. The SGS (for this application) consists of two interferometer legs on approximately 1500-foot baseline which determine vector velocity and position (resolving ambiguities by means of subcarriers) plus an airborne transmitter and receiver in the airborne stage. Because attitude changes will be effected by turning a constant rate, only discrete commands are sent to the Able stage. After Stage 2-3 separation, the SGS will track the payload using another command receiver and transmitter in the payload. After a short coast period (to reduce ionospheric error), the SGS will make a final accurate determination of vector position and velocity, and the guidance computer will compute the desired vernier correction which will then be automatically transmitted to the payload.

(2) Midcourse Guidance

Midcourse guidance is accomplished by using tracking measurements from the ground stations together with the SGS determination of injection conditions. To determine the precise trajectory, vernier corrections are commanded from various ground stations at appropriate times. A maximum of five vernier corrections can be made prior to lunar injection.

The present plan is to correct the energy in the first few hours using accurate range rate data obtained at Manchester. Over the first day the effect of a velocity change along the direction of the vehicle spin axis changes markedly in direction. Therefore, one additional vernier correction can eliminate the majority of the residual errors. Tracking data will also be used to determine the optimum time for firing the vernier injection system as an injection rocket. Subsequent tracking will establish the character of the lunar orbit and will permit modification of the orbit by subsequent vernier firings.

c. Accuracy

The accuracy requirements for establishing a stable lunar orbit are quite modest (50 ft/sec velocity errors and 1.2 deg angular errors can be tolerated). Since the accuracy which will be attained using SGS and midcourse guidance will be at least an order of magnitude better than above, capabilities then exists for establishing a close-in lunar satellite of a fair amount of precision. Therefore, the probability of lunar capture is exceedingly high.

6. Tracking and Communications

General Considerations

To track and communicate between the moon and the earth, at the maximum distance less than 260,000 miles, places substantially smaller requirements upon the ground and airborne system than does interplanetary tracking and communications. Hence, the Able-3 airborne and ground system will readily meet the requirements for this lunar mission. The 150-watt airborne transmitter will not be required.

7. Payload Vehicle

a. Structure

This structure is built up of sheet metal and consists primarily of a 0.032-inch magnesium alloy inner cylinder to carry axial and bending loads, rings and webs forward to which are attached the payload platform and the solar paddle hinge brackets, and a ring aft for clamping to the Stage 3-4 interstage structure. Separation is accomplished by firing two explosive bolts securing the clamp at this interface.

As in the Able-3 and Able-4 Thor, the payload is covered by spun, thin, aluminum alloy spheres made in two or more parts and screwed on at the equator of the payload.

b. Experiments

The experiments in this payload are the same as those in Able-3 (see Section A, 7) except for the photomultiplier scintillation counter. The objectives of the photomultiplier scintillation counter are similar to those discussed in Section III, G, 1 for the Able-3, except that the threshold is to be lowered to 100 kev for electrons and 1 mev for protons by use of a more sensitive detecting crystal.

c. Payload Temperature Control

Because of the hydrazine motor, the allowable temperature range of the Atlas payload is smaller than for the two Thor payloads. Because of the degradation of surface radiation properties by ultra-violet radiation, micrometeorites, sublimation, and other stresses of the space environment, it is highly desirable to provide an active closed loop temperature control scheme which will correct for gradual changes in the radiation properties. Inasmuch as only a small change in properties could be tolerated (because of the narrow allowable temperature range) such a control scheme was incorporated in the Atlas payload.

The control scheme consists of about 50 individual control units on the payload shell. Each unit consists of a very light, fourbladed fan, which at one end of its 45 degree of travel exposes a high α/ϵ material and at the other end exposes a low α/ϵ material. The fan is driven by a bimetallic spring, arranged so that it is well coupled to the interior payload temperature. The fan adjusts itself to yield a payload temperature within the desired range. To increase the effectiveness of the control, the remainder of the payload surface, outside of the control unit areas, is made to have as little influence on the temperature of the payload as possible. This is done by providing a surface (vacuum-deposited aluminum) with as low an α and ϵ as possible. The control surfaces are also subject to change by the space environment, but this simply serves to reduce the effectiveness of the control. By providing a large margin of control, in the form of a large range of α/ϵ, the control system should be able to cope with an appreciable amount of change in the properties of its own control elements.

(1) Temperature Measurements

Temperatures are being measured, by means of thermistors, at a number of key locations in the payload and on the solar cell paddles.

(2) Solar Cell Temperature Control

(Same as for Able-3)

(3) Eclipses

Because of the low emissivity surface covering most of the payload, the thermal time constant of the payload is quite long with the consequence that the internal temperature will drop only slightly (about 10° to 15°F) even for the long eclipses expected to occur.

The solar cells experience temperature drops during eclipses comparable in magnitude to those of the Able-3 mission.

d. Stability Requirements

The stability requirements for the Able-4 Atlas payload are similar to those discussed in the Able-3 section on stability requirements. However, unlike the requirements stated for the other Able payloads, there is a possibility of unstable motion in this payload because of sloshing of the liquid propellant in the Able-4 Atlas vernier tank. However, to date experiments with a real model have indicated no instabilities. Further studies will be made on the motion of the liquid propellant.

D. AFMTC Launch Complex Facilities and Ground Support Equipment

Launch complex facilities and ground support equipment required for the first, second, and third stages of the proposed test vehicles will be essentially identical to the equipment used for the Project Able-1 Lunar Probe Program. Ground stations at AFMTC and overseas will provide guidance, tracking, and telemetry coverage for the fourth stage.

The launch complex facilities and ground support equipment required for the first, second, and third stages of the Atlas-boosted will require modification of existing facilities. The blockhouse area and the pad and tower area of the Atlas launch complex to be used during the launching of the test vehicle will, of necessity, require changes to support the launch. A summary of the required modifications to the Atlas launch complex is detailed below.

1. A platform to hold the second stage propellant servicing trailers.
2. Drain pans to hold the servicing trailers.
3. Propellant drain lines.
4. Electrical cables.
5. Safety showers.
6. Storage space.
7. Weather protection and flood lights.
8. Umbilical mast extension.
9. Communication loops between blockhouse and all Able operational areas.
10. Blockhouse space for Able launch consoles.
11. Parking space for missile checkout trailers and propulsion instrumentation vans near the blockhouse and base of tower.
12. Space for helium pressurization consoles.

The exact nature of equipment and stand modifications will be furnished to Convair early in the program to insure compliance with the STL requirements well in advance of the proposed launch date.

Ground stations at AFMTC, Manchester, Hawaii, Millstone, and Singapore will provide guidance, tracking, and telemetry coverage for the fourth stage.

E. Data Reduction and Storage

1. Analog Data

Power available in the Able-3 payload will allow continuous operation of the analog telemetry in that vehicle, but except for the television signal and some special measurements of payload characteristics, there will be no analog telemetry of experiments in either of the Able-4 vehicles. Because of the distances involved the majority of that telemetry will be transmitted in digital form.

Recording of analog telemetry will be in two forms: first on magnetic tape, along with the digital telemetry, as a complete flight record, and, second, on stylus recorder strip charts to facilitate visual inspection and analysis of the data. The signal representing the image from the television receiver, however, will not be displayed as a pen recording; the picture will be obtained directly from the magnetic tapes when they are returned to STL.

For a short time immediately following launch and at intervals throughout the flight, information from the pen recordings will be teletyped to Los Angeles for quick-look purposes to assess certain vehicle characteristics, to check the condition of the experiments, and to allow for the recognition of any special environmental situations at the payload.

The volume of data taken from the Able-3 payload will be considerable, necessitating a library facility where magnetic tapes and stylus recordings can be stored. These records will form the basis for exhaustive analysis and evaluation of all results.

Since the analog results will be more complete than the digital, it will be subjected to more extensive analysis. The magnetic field and radiation zones will be mapped and results compared with various proposed models. Then the records will be carefully examined for transient phenomena to ascertain the effects of solar and cosmic activity on the values of magnetic field strength and radiation rate. In addition, results of the various experiments will be correlated since the understanding of one phenomena may very likely be enhanced by results in related phenomena.

2. Digital Data Reduction

The STL Operations Center established for Project Able-1 will constitute the center of activity for data reduction for Projects Able-3 and Able-4. The center will perform the following digital data reduction:

(a) Immediate Quick-Look Data Reduction (output approximately one hour after transmittal of data to the center)

Subcommutated data (to be used for monitoring payload temperatures and battery voltages) will be transmitted to the Operations Center via teletype immediately after reception. The paper tape obtained will be entered into the 704 or 709 computer and will be processed to yield uncalibrated listings of these data versus time. At the same time as the subcommutated data are being generated, an uncalibrated listing versus time will be prepared for all of the experiments contained in the payload.

(b) Quick-Look Data Reduction (one day after receipt of paper tapes by the center)

The program developed for this task will be used to provide listings of calibrated experimental values from the payload along with appropriate trajectory information.

(c) Complete Data Reduction

Analytical procedures to be utilized to a great extent will be a function of the quantity and quality of the data generated. As a result, a detailed formulation of exact procedures in this area will not be made until the data have been received.

E. PROGRAM SCHEDULE

The major milestone schedules for the Able 3-4 program are shown in Figures 20, 21, and 22. The dates are tentative and may be changed after detailed planning and coordination with AFMTC, booster contractors, and respective STL program offices; however, the sequence of operation will remain essentially as shown.

	Completed	1959 June 6	13	20	27	July 4	11	18	25	August 1	8	15	22	29	September 5	12	19	26
Payload Prototype Components on hand		X																
Prototype Payload Operating					X													
Payload Flight Components on hand			X															
Flight Payload Operating						X												
Payload "A" Ship to AFMTC									X									
Payload "B" Ship to AFMTC									X									
Ship 2nd Stage to AFMTC	X																	
Ship 2nd Stage SGS to AFMTC							X											
Covers Off 2nd Stage at AFMTC								X										
Covers On 2nd Stage at AFMTC									X									
MOS									X									
Mock Countdown										X								
Launch											X							

Figure 20. Able-3 Major Milestone Schedule.

	1959																
	August				September				October					November			
	8	15	22	29	5	12	19	26	3	10	17	24	31	7	14	21	28
Payload Prototype Components on Hand		X															
Prototype Payload Operating			X														
Payload Flight Components on Hand			X														
Flight Payload Operating				X													
Payload "A" Ship to AFMTC											X						
Payload "B" Ship to AFMTC											X						
2nd Stage SGS on Hand						X											
Covers Off 2nd Stage at LAX							X										
Covers On 2nd Stage at LAX								X									
Ship 2nd Stage to AFMTC										X							
Covers Off 2nd Stage at AFMTC											X						
Covers On 2nd Stage at AFMTC											X						
MOS											X						
Mock Countdown													X				
Launch													X				

Figure 21. Able-4 Thor Major Milestone Schedule.

		1959																
	Completed	July				August					September				October			
		4	11	18	25	1	8	15	22	29	5	12	19	26	3	10	17	24
Payload Prototype Components on hand		X																
Prototype Payload Operating					X													
Payload Flight Components on hand			X															
Flight Payload Operating					X													
Payload "A" Ship to AFMTC							X											
Payload "B" Ship to AFMTC									X									
Ship 2nd Stage SGS to AFMTC			X															
Ship 2nd Stage to AFMTC	X																	
Covers Off 2nd Stage at AFMTC					X													
Covers On 2nd Stage at AFMTC						X												
MOS						X												
Mock Countdown								X										
FRF									X									
Launch											X							

Figure 22. Able-4 Atlas Major Milestone Schedule.