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A DEVELOPMENT PLAN

FOR

TWO INTERPLANETARY PROBES

(ABLE-4)

14 January 1959

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I. INTRODUCTION

The development plan presented here describes two interplanetary probes. One, a Thor-boosted Able vehicle will carry 80 pounds of payload to the vicinity of Venus. The second, an Atlas-boosted Able vehicle will carry about 330 pounds of payload and injection rocket to Venus and then will attempt to orbit Venus. These interplanetary probes will be utilized to conduct a number of experiments whose purpose is to extend our knowledge of the solar system and of the planet Venus. In addition, these first interplanetary probe flights will establish the basic parameters relating to space communication and guidance for future space missions.

II. BACKGROUND INFORMATION

Early in the calendar year of 1958, the Air Force Ballistic Missile Division (BMD) developed an Advanced Re-entry Test Vehicle (ARTV) for the purpose of testing ballistic missile nose cones at the full 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. The initial flight test was terminated by a failure during first-stage operation.

On 27 March 1958, the Advanced Research Projects Agency (ARPA) directed AFBMD to proceed with a program of three lunar probe vehicles. This program was designated Project Able-1. As much as possible the Project Able-1 vehicles were to utilize existing ARTV designs, spare hardware, and support facilities. Three lunar vehicles have been launched, one each in August, October, and November 1958. The October flight was a partial success and is the source for much of the information concerning space flight. The August flight terminated during the first-stage operation and the November flight terminated because the third stage failed to ignite.

The Able-3 program, presently in progress, to put a large payload in an elliptical orbit about the earth represents, in many respects, a flight test of many of the components to be used in this program.

The Thor-boosted interplanetary probe described here uses basically the same vehicle as that developed for the lunar probe experiments and that being developed for the Able-3 flight. The Atlas-boosted vehicle uses the same second and third stages but the payload is four times larger.

III DESIGN CHARACTERISTICS

A. Objectives

These interplanetary flights have four objectives. First, these flights will be used to gather scientific data, such as a better knowledge of the value of the astronomical unit (semi major axis of the orbit of the earth) measured in terrestrial units, of magnetic field strengths in space, of cosmic ray intensities away from the earth, and of micrometeorite density.

The Venus satellite will be used for the experiments described above and will, in addition, be used to determine the mass of Venus and to gain some knowledge of the Venusian atmosphere. This vehicle can constitute a nearly permanent source of scientific information.

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

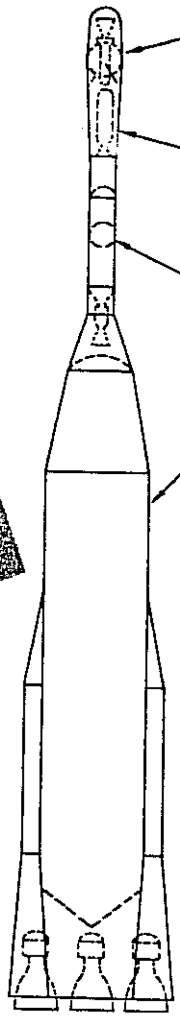
Third, these flights will be valuable for gathering environmental information useful for manned interplanetary space flight. The extended flight times, as well as the fact that the sampling is taken over large parts of the solar system, will increase our knowledge of the problems of interplanetary flight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

Fourth, these flights will establish the fundamental feasibility and effectiveness of a guidance system for space flight and, in particular, demonstrate the value of the proposed midcourse guidance.

B. Technical

1. Vehicles

The Thor-boosted vehicle is essentially the same as the Able-1 vehicle, and is illustrated in Figure 1. The Thor with autopilot steering and fuel-exhaustion shutoff is used as the first stage. The STL Able and Able-1 second stage (Aerojet 10-101 engine) will be used for the second stage for this vehicle. The ABL-248 solid rocket from the Able-1 project will be used for the third stage, giving a payload of 80 pounds.



Fourth Stage

Gross Weight at Burnout	202	lb
Gross Weight at Start*	330	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

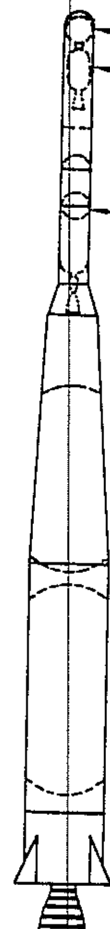
First Stage - Atlas Sustainer

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.



Payload 80 lb

Third Stage - ABL 248-A-3

Gross Weight at Burnout	135	lb
Gross Weight at Start	600	lb
Vacuum I_{sp}	250.4	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 1. Atlas-Boosted and Thor-Boosted Vehicles with Aerojet Second Stage and ABL Third Stage.

The Venus satellite vehicle will use an Atlas booster and sustainer, the STL Able-1 second stage, and the ABL third stage. The payload will be 330 pounds. Figure 1 shows the resulting vehicle. A retrorocket is carried as part (approximately half) of the fourth-stage payload and is used to provide the final thrust to place the net payload into an orbit about Venus. It is possible that the payload may be increased by (1) improved power flight trajectories and (2) by removing some weight from the Atlas booster.

The design, assembly, instrumentation, checkout, and additional similar functions with respect to the second, third, and payload stages will be carried out by STL in the manner established for the Able-1 project.

2. Trajectories

The trajectory is shown in Figure 2. The vehicle will be launched early in June 1959 and will arrive at Venus early in November. Because Venus is not in the plane of the ecliptic, a true minimum energy trajectory cannot be followed. Midcourse velocity corrections are necessary since the expected accuracy at burnout is not itself sufficient to bring the payload close enough to Venus so that when the injection rocket is fired, the vehicle will become a satellite. Variation in launch time will have some effect upon the trajectory flown and a marked effect upon the errors. It is possible to compensate for launch delays by decreasing the payload; however, in estimating the payloads, consideration has already been given to a nominal launch delay.

3. Payload

Using a Thor-boosted vehicle, approximately 80 pounds of payload may be sent to Venus. Use of an Atlas-boosted vehicle permits a payload of 330 pounds. Since the Atlas-boosted vehicle will be put into orbit about Venus, approximately half of the payload must be allocated to a retrorocket.

Table 1 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.

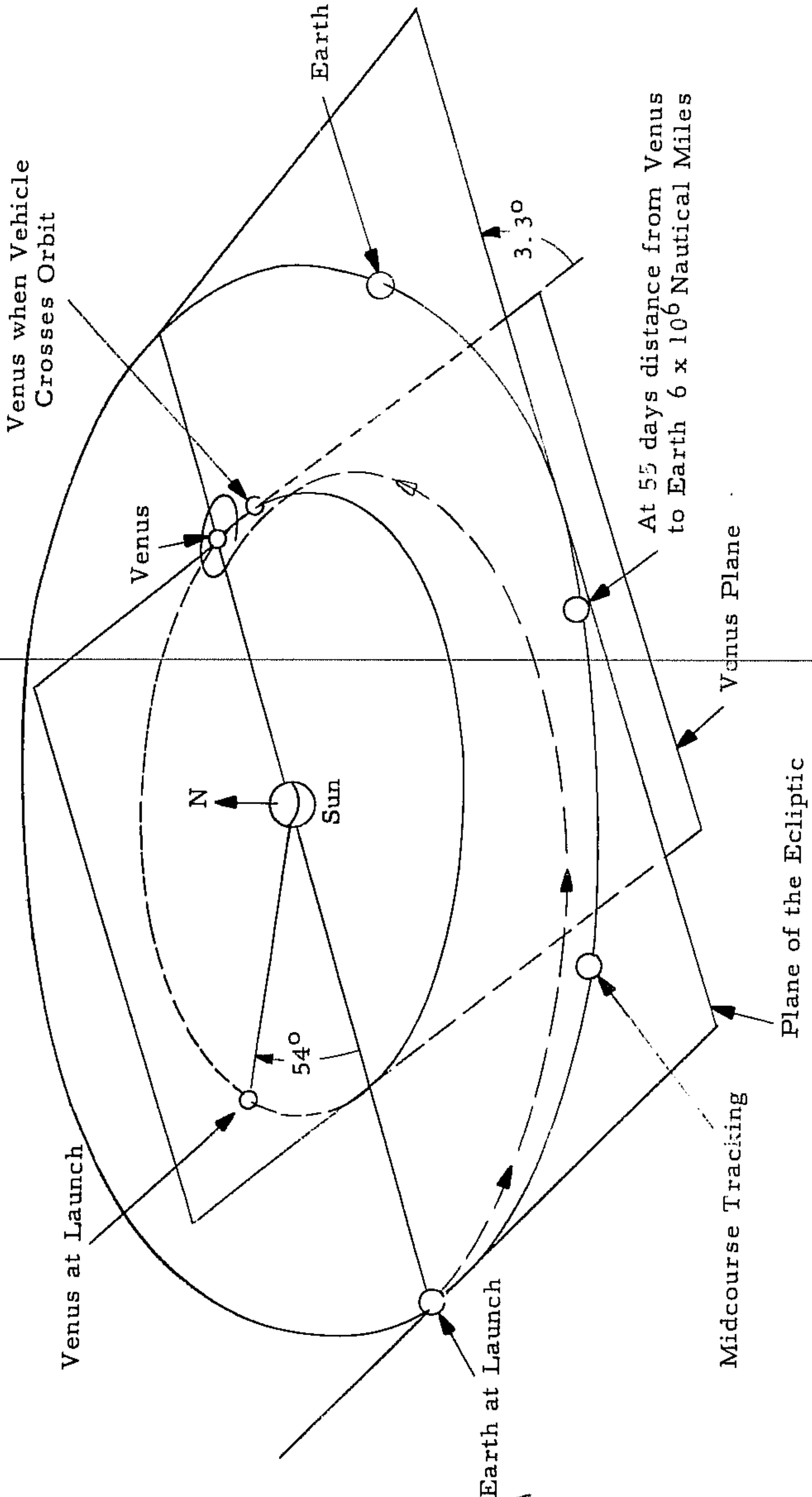


Figure 2. Trajectory of Venus Satellite.

Table 1. Preliminary Estimated Weights of Payload Components.

Item	Thor		Atlas	
Electronics				
Transmitters	9.0		18.0	
Receivers	4.0		8.0	
Power Converter	0.5		1.0	
Antenna System	2.0		4.0	
Telemetry System	2.0		4.0	
Programmer	0.5		1.0	
Wiring, Harness	2.4		3.4	
Sub Total		20.4		39.4
Experiments				
~~Cosmic Ray Telescope~~	~~4.0~~		~~4.0~~	
Search Coil Magnetometer	1.7		1.7	
Geiger Muller and Ion Chamber	2.0		2.0	
Very Low Frequency Monitor	1.0		1.0	
Flux Gate Magnetometer	2.5		2.5	
Micrometeorite	0.7		0.7	
Synchronization and Digital Accumulators	1.5		0.2	
Temperature Indicators (6)	0.2		2.8	
Television			1.5	
Cerenkov Counter			3.0	
Sub Total		13.6		19.4
Power Source (solar cells, batteries and mount)		24.0		48.0
Structure (paddle configuration)		15.6		31.2
Injection Rocket		—		163.0
Verniers (effective weight)		5.0		23.0
Temperature Control System (passive)		1.0		2.0
TOTAL		79.6		326.0

The basic difference between the two vehicles is that the Atlas boosted vehicle will carry redundant components. The transmitters and the receivers will operate at approximately 400 mc for both tracking and telemetry

The payload transponder consists of a coherent receiver, a command decoder, and a 150-watt transmitter The ground system for mid-course and terminal guidance will utilize the 250-foot antenna at 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. They include measurement of the astronomical unit, the radiation level, the mean specific ionization per particle, magnetic fields, micrometeorites and inter-planetary 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 Cerenkov counter and a TV picture of Venus. It may be possible to determine the mass, period of rotation, and oblateness of Venus from measuring the orbit of the vehicle about Venus.

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. A possible configuration is shown in Figure 3. An active temperature control system may be required to meet the temperature requirements of the internal components.

In addition, a vernier velocity correction system will be required both for burnout velocity correction and for midcourse velocity corrections. Two systems are under study. One is a monopropellant system using hydrazine, which would weigh about 15 pounds if it were to supply a correction of 200 ft/sec. The other, in the case of the Atlas, will consist of a number of small solid-propellant rockets of various sizes. Seven of these will be used to correct the velocity errors at third-stage burnout. Seven other rockets will provide mid-course correction after about 30 days of tracking.

Table 2. Basic Instrumentation Package.

Experiment	Quantity Measured	Importance	Weight (lb)	Power (mw)	To Be Supplied By
1. Astronomical Unit	Unit of length	Fundamental unit for location of objects in space	0	0	STL
2. Ion Chamber and Geiger Muller	Radiation level in space and mean specific ionization per particle to 1 per cent		2	500	U. of Minn.
3. Proportional Radiation Counter	Intensity distribution	Radio propagation, solar physics, cosmology theory	4	10	U. of Chicago
4. Scintillation Counter (Cerenkov)	Radiation flux in space		3	250	STL
5. Magnetic Field and Vehicle Aspect using Search Coil Magnetometer	Field strength	Radio propagation, magnetic fields of planets, electric currents in space	1.7	23	STL
6. Flux Gate Magnetometer	Field strength		2.5		STL
7. Meteorites and Interplanetary Dust MM Spectrograph	Intensity and momentum distribution	Nighttime E layer, effect on re-entry problem, cosmology theory	0.7	25	AFCRC
8. Very Low Frequency Monitor	Magnetically guided radiation at very low frequencies	Solar activity, weather on Venus	1	50	Stanford
9. Temperatures	Internal payload temperatures and solar cell paddle temperatures	Effect of space environment on solid propellants and instrumentation	1 oz	1	STL
10. TV (Atlas-boosted vehicle)	Picture of Venus atmosphere	Atmospheric details, possible view of planets	2.8	100-200	NRL/STL

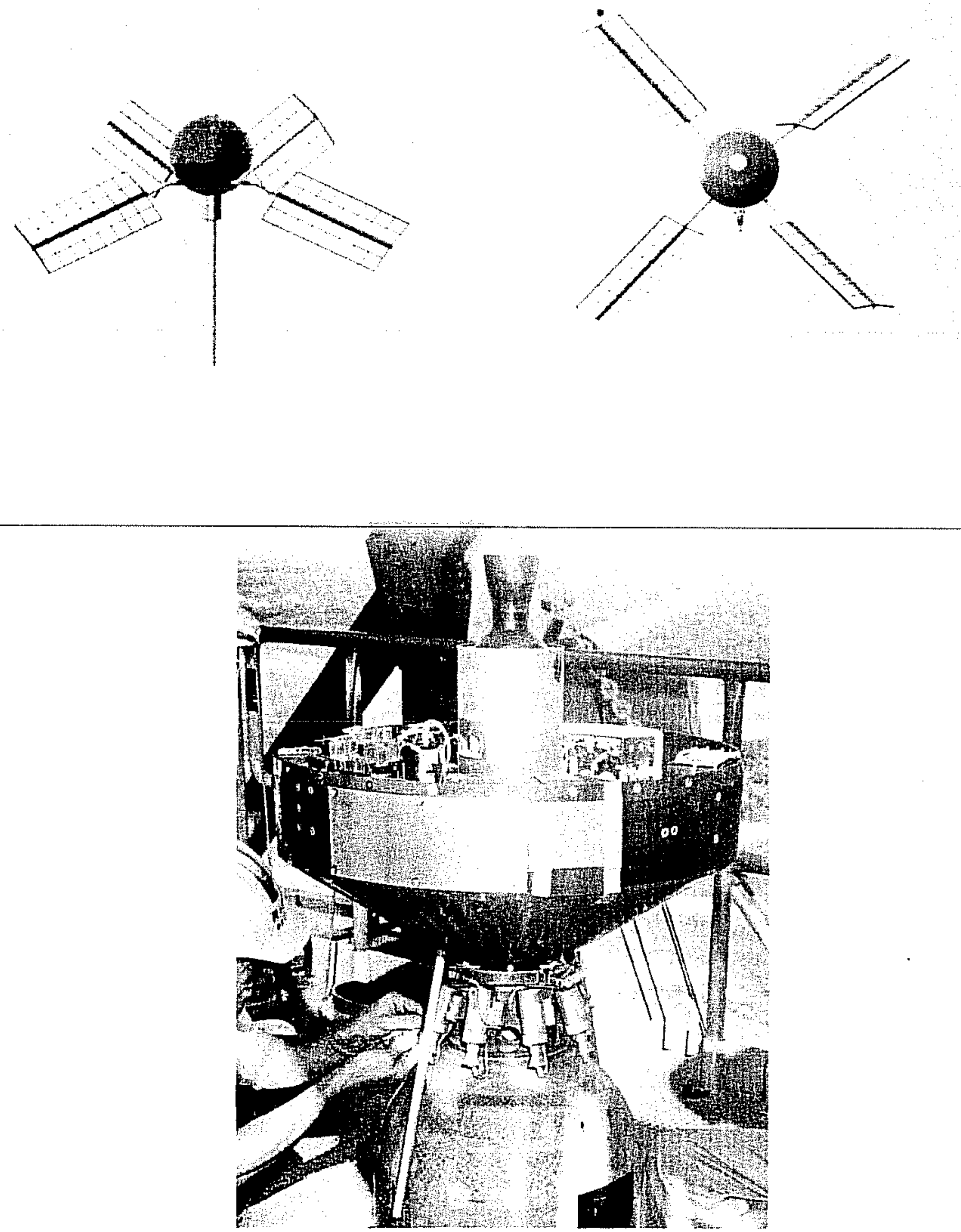


Figure 3. Two Models of Able-4 Payload Showing the Solar Cells and the Able-1 Payload.

4. Guidance

With the Thor-boosted vehicle, the Thor stage is controlled by a gyro-stabilized autopilot. When the Atlas is used, it will be guided by GE Mod III radio guidance. The Able stage will be guided by the STL Advanced Guidance System coupled with the Burroughs Mod I guidance computer. The third stage is spin stabilized and the orientation of the spin axis orients the thrust of the third stage. It is expected that the accuracy of the Atlas-boosted vehicle at burnout will have an equivalent velocity error of 2 to 5 ft/sec after vernier burnout corrections. Burnout flight path angle error will be measured to 0.1 to 0.2 degrees. The velocity error and the in-plane component of the angular error will be largely corrected by midcourse guidance. It is expected that the radio tracking systems can know range to 500 miles and range rate to a foot per second and can determine equivalent velocities to the order of one foot per second after about one month of tracking. For the case of the Thor-boosted vehicle, the accuracy will be the same as that for the Atlas if the limited vernier capability permits adequate burnout and midcourse corrections to be made. The error at the establishment of the Venus satellite depends upon both burnout velocity errors and burnout flight path angle errors. Since both of these errors will be considerably reduced by midcourse guidance, the probability of success for satellite establishment will be improved. Any change in the launch time will have a pronounced effect upon the guidance, and considerable study is planned in this area to optimize all parameters.

C. Vehicle Characteristics

1. Thor-Boosted Vehicle

a. Description

The Thor-boosted vehicle is composed of three propulsion stages plus payload. Figure 1 in Section A shows some of the pertinent weight and propulsion data. Except for payload configurations, the vehicle is very similar to the Able-3 satellite vehicle and consists of a Thor booster for the first stage, the AJ10-101 for the second stage, and the ABL 248 for the third stage.

The sequence of operation is the same as that used for the Able-3, but the altitude and burnout angle have been changed to suit an interplanetary trajectory. The first stage rises vertically for approximately 10 seconds, then enters a gravity turn until approximately main engine cutoff. At 160 seconds, the main engine is shut down and the stage continues to thrust with the vernier engines only. Two seconds after first-stage shutdown the second-stage engine is ignited and the first-stage engine is jettisoned. The vehicle then flies at constant attitude until second-stage burnout at approximately 274 seconds. Approximately two seconds after second-stage burnout, spin rockets located at the center of gravity of the assembly impart a spin of approximately 2.4 rps about the longitudinal axis. (This spinning is necessary to ensure stability of the third stage during its burning and the subsequent coast of the payload). The third-stage motor is then ignited and the second stage is jettisoned. The third stage burns to completion at about 312 seconds and is separated from the payload. The velocity is determined at this point and the proper vernier rockets are fired to impart the desired velocity vector to the payload to place it in the proper trajectory.

b. Performance

The velocity requirements for interplanetary missions vary from escape velocity at the burnout altitude to velocities which are about 10 per cent greater than escape speed depending upon the mission requirements. For example, the minimum-energy requirements at burnout for the Venus mission require a burnout velocity that is some 3 per cent greater than the escape velocity from the earth. A convenient way to show vehicle performance is to plot payload as a function of the theoretical residual velocity at infinity. On such a plot a zero residual velocity would represent exactly the escape velocity at burnout. Figure 4 shows the payload capabilities for the Thor-boosted vehicle with the AJ-10-101 second stage and the ABL 248 third stage.

Figure 5 describes the powered flight of the same Thor-boosted vehicle with a gross payload of about 80 pounds.

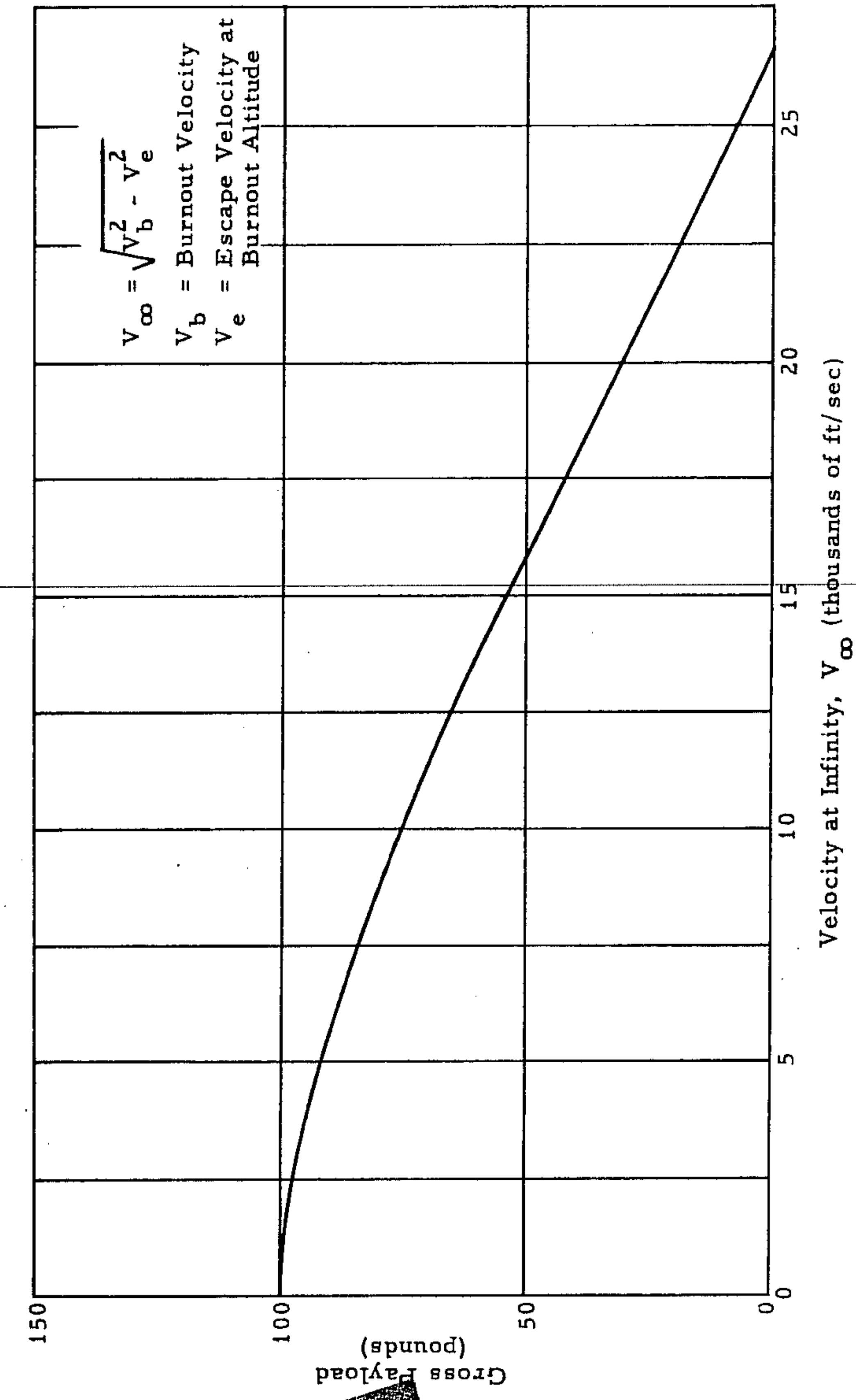
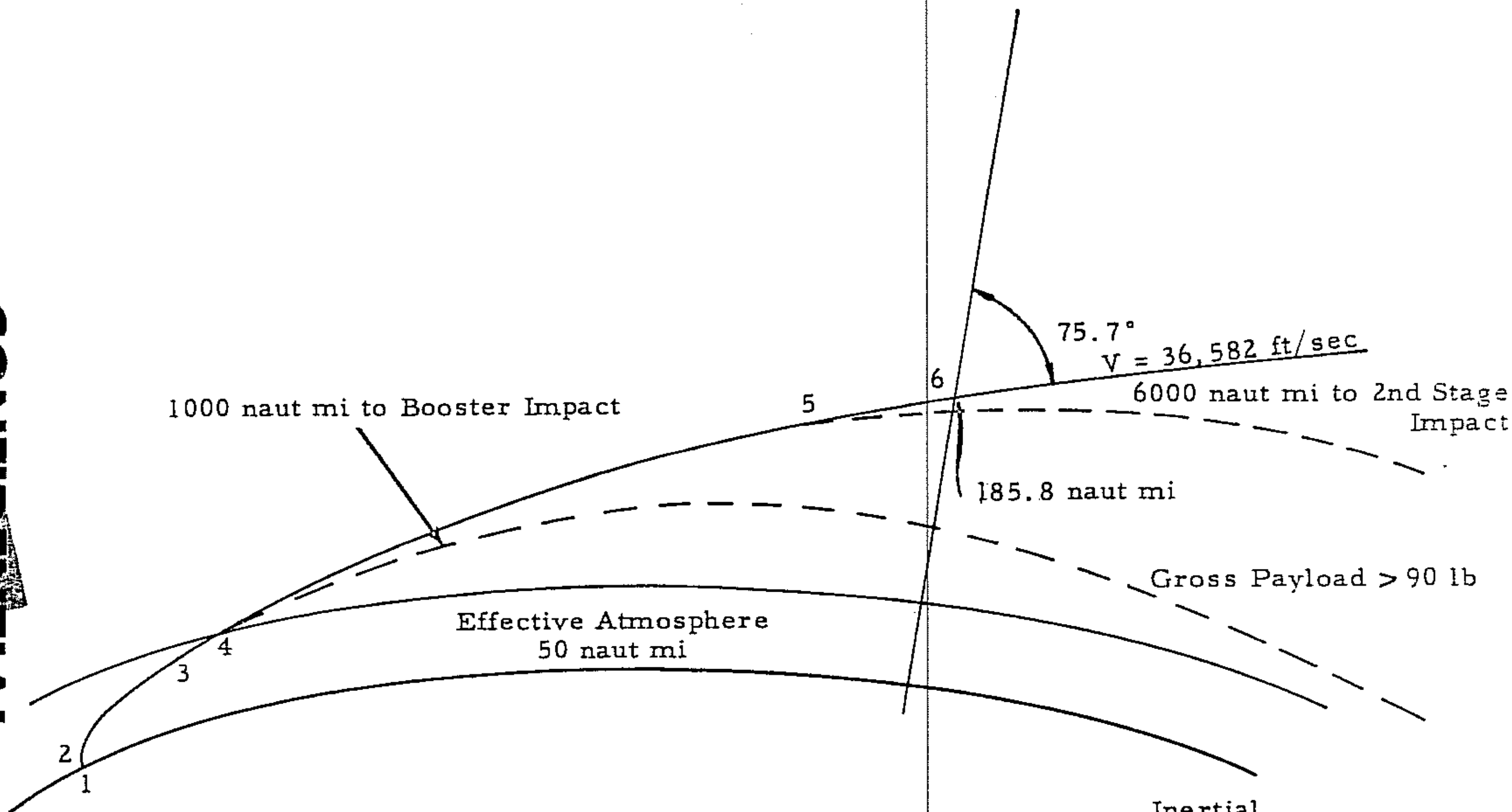


Figure 4. Gross Payload as a Function of the Velocity at Infinity of the Thor-Able 4.



		Time (sec)	Range (naut mi)	Altitude (naut mi)	Inertial Speed (ft/sec)
1	Takeoff	0	0	0	1,342
2	Begin Gravity Turn	10	0	0.1	1,350
3	Begin Programmed Turn		40	41.7	13,197
4	First-Stage Shutdown	159.9	53	46.6	16,206
5	Second-Stage Shutdown	274.1	383	142.4	25,113
6	Third-Stage Shutdown	311.6	538	185.8	36,582

Figure 5. Powered Flight Trajectory of Thor-Boosted Vehicle.

c. Injection

No injection rocket is provided in the Thor vehicle because of payload limitations. The final correction prior to free flight is made by the vernier rockets described below.

d. Burnout Verniers

With the Thor-boosted vehicle, burnout errors can be corrected by either of two solid-propellant-engine methods: (a) eight 1X550 rockets of 50 lb-sec impulse and total weight of approximately 5 pounds, or (b) six of the following rockets --one 1KS210 (at 250 lb-sec), four 1X 550 (at 50 lb-sec), and one 1X525 (at 25 lb-sec) with a total weight of 6.5 pounds. The second system, while heavier, would permit a smaller velocity increment. In each case, the appropriate number of engines is fired, by command, as determined by ground tracking.

e. Midcourse Verniers

With the Thor-boosted vehicle, payload weight limitations do not permit the use of engines carried for midcourse corrections. However, in the nominal case, not all the rockets for vernier correction will be fired and those remaining can be fired for midcourse velocity correction.

2. Atlas-Boosted Vehicle

a. Description

The Atlas-boosted vehicle is comprised of three propulsion stages plus payload with or without injection rocket. Figure 1 shows some of the pertinent propulsion weight and data. The first stage is the Atlas booster and sustainer, the second and third stages are the same as the Able-4 Thor, namely, the AJ10-101 for the second stage and the ABL 248 for the third stage. In addition there are verniers for velocity corrections and an injection rocket as part of the terminal stage. The first stage rises for 10 seconds, at which time it enters a gravity turn. Booster burnout occurs at 145 seconds and sustainer burnout at 273 seconds. The sequence from this point to the end of third-stage burning is identical to the Able-4 Thor. As in the previous case, the velocity is determined at this time in the flight and a correction is made by vernier firing. In midcourse, approximately a month later, a further correction is made by vernier firing and the vehicle continues to spin and coast to the vicinity of Venus. At the appropriate time in flight, determined by tracking data, the injection rocket is fired to impart a velocity increment of about 3000 ft/sec and this places the vehicle in orbit around Venus. Several rocket engines are under consideration for this use; the Thiokol TE80 may be used but a cluster of five TX8 engines as well as a monopropellant engine are also being considered. The final choice will be made on a basis of payload design criteria, i.e., stability, weight, and size.

b. Performance

Vehicle performance is shown in Figure 6 where payload is a function of the theoretical residual velocity at infinity for an Atlas-C-boosted vehicle. On such a plot, a zero residual velocity would represent exactly escape velocity at burnout.

Figure 7 describes the powered flight using the Atlas booster ABL 248 third stage and a fourth-stage weight of about 330 pounds.

c. Injection Rocket Size and Orientation

To put a vehicle in orbit around Venus requires the use of a terminal thrust period. The magnitude of the thrust is determined by how close the vehicle can come to Venus and upon the satellite orbit desired. As decribed above, this vehicle may use the Thiokol-TE80 or some other appropriate rocket, (see Figure 1 in Section 1 for pertinent weight and propulsion data)

d. Vernier Systems

(1) Monopropellant Systems

A hydrazine unit, similar to the one developed and tested at JPL, appears to be the most promising of the vernier systems that have been considered. The system will use a helical coil tank, pressurized with helium at 400 psi, a hypergolic start, and catalytic decomposition operation, and will have a weight of approximately 15 pounds. The hydrazine will be kept at the exit end of the tank by the centrifugal force of the spin. Squib valves are used for stop and start and the system would require a separate set of valves

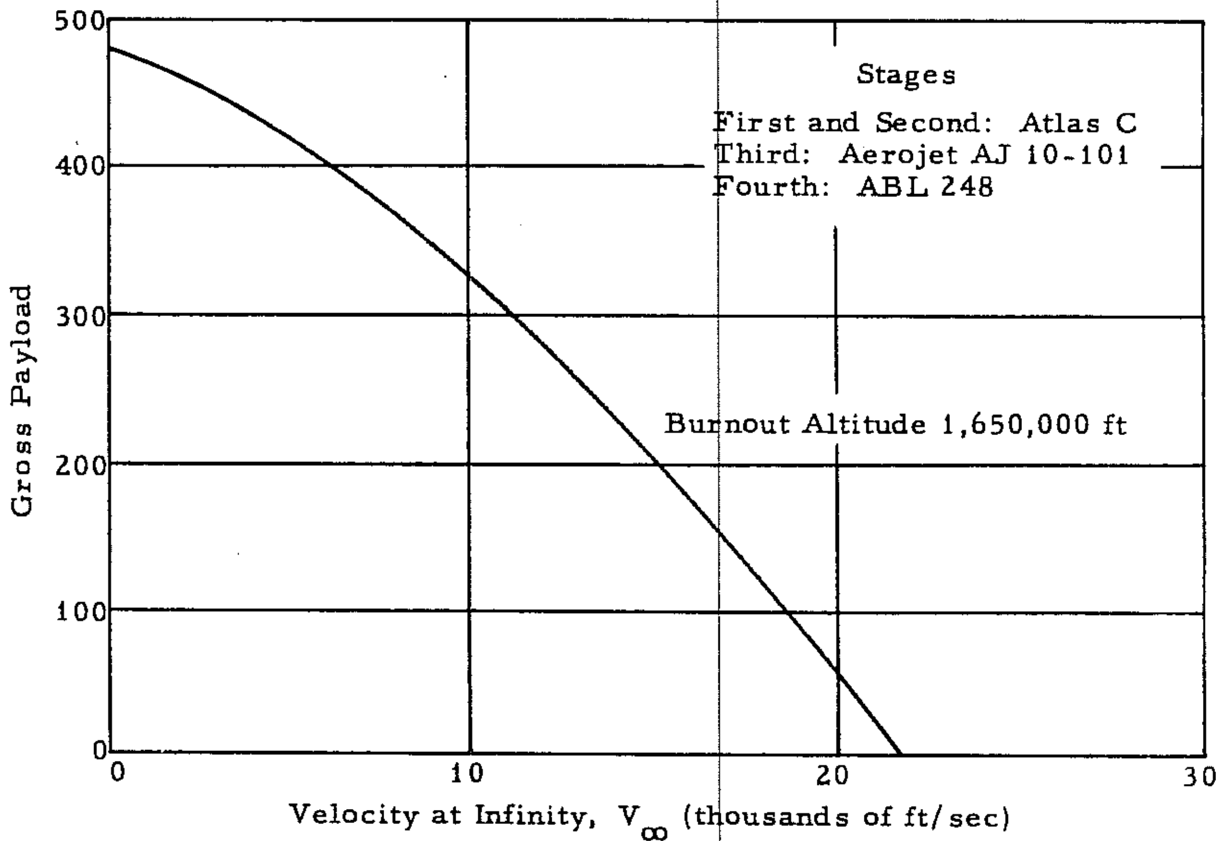
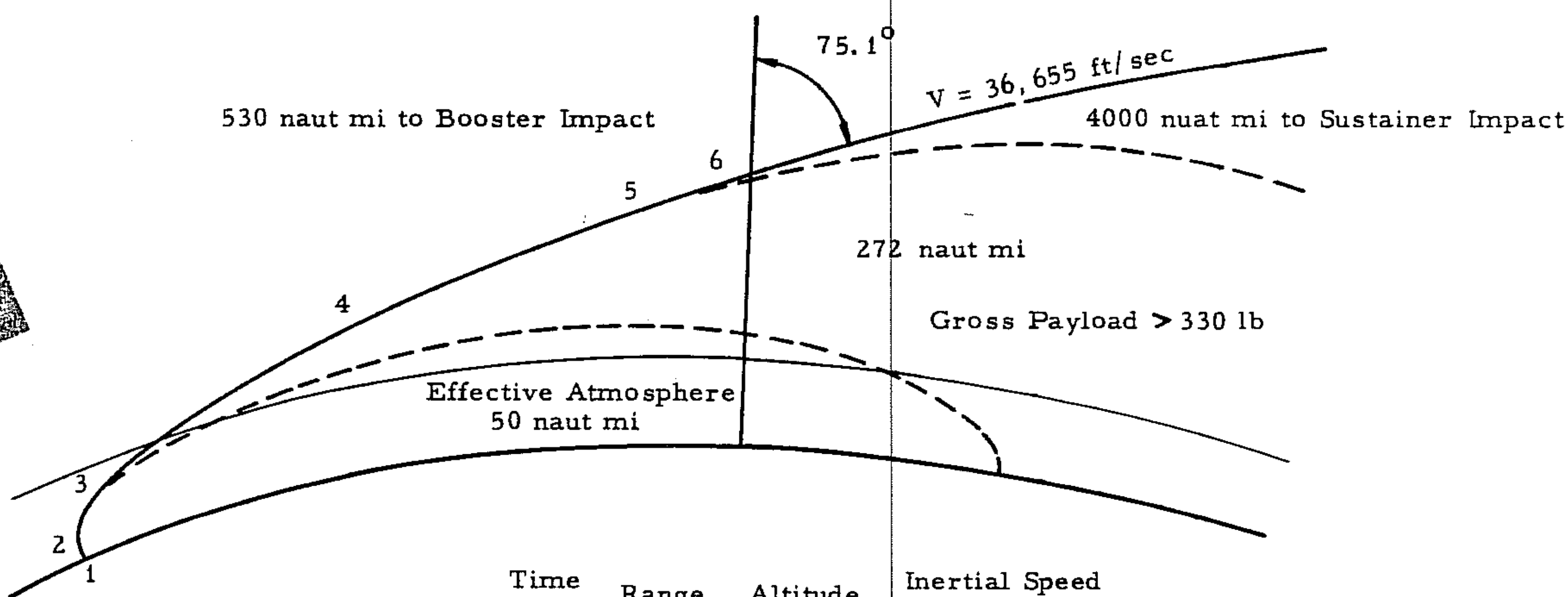


Figure 6. Gross Payload as a Function of the Velocity at Infinity of the Atlas Able 4.



	Time (sec)	Range	Altitude	Inertial Speed (ft/sec)
1. Takeoff	0	0	0	1,342
2. Begin Gravity Turn	10	0	0.1	1,350
3. Booster Shutdown	140.8	20	36.5	10,811
4. Sustainer Shutdown	290.2	340	129.0	22,430
5. 2nd Stage Shutdown	404.3	768	224	30,696
6. 3rd Stage Shutdown	441.8	946	272	36,655

Figure 7. Powered-Flight Trajectory of the Atlas-Boosted Vehicle.

for restart or two-shot operation. Thrust varies from 17 to 10 pounds as the pressure drops; Isp = 233 seconds; total impulse delivered is 2200 lb-sec for approximately 9.5 pounds of fuel; nozzle expansion ratio is 50:1. This system has the capability to provide a total correction of about 200 ft/sec for the Atlas-boosted Able-4 vehicle.

(2) Solid-Propellant System

An alternative burnout-velocity vernier system for the Able-4 vehicle, as well as the midcourse vernier system, is a "binary-type" solid propellant system. During burnout corrections, one 1KS-420 solid rocket with a total impulse of 498 lb-sec and weighing 5.2 pounds, two 1KS-210 solid rockets, each with a total impulse of 249 lb-sec and weighing 3 pounds each, and four 1XS-50 solid rockets, each with a total impulse of 53 lb-sec and weighing 0.6 pounds each, will be used. These will permit a total impulse of 1208 lb-sec in all possible combinations. The total weight of all of these rockets is 13.6 pounds. A total of seven commands is required to selectively employ the desired amount of impulse, with the smallest increment being 53 lb-sec.

The midcourse corrections for this system will be performed with one 1KS-420 solid rocket, one 1KS-210 solid rocket, four 1XS-50 solid rockets, and one 1XS-25 solid rocket, with a total impulse of 985 lb-sec amd weighs 11.2 pounds. These rockets may be fired in any of their possible combinations using seven commands.

D. Trajectory Considerations

1. General Considerations

The general nature of the trajectories to be followed is determined by the geometry of the solar system which is indicated in Figure 8. For a trajectory to Venus, the vehicle leaves the earth in such a direction that the velocity of the vehicle, once it is free of the earth's graviational field, subtracts from the orbital velocity of the earth, resulting in a vehicle velocity with respect to the sun that is less than the earth's velocity. On a minimum-energy trajectory, the vehicle travels 180° around the sun in a transfer ellipse and reaches a perihelion distance equal to the distance of the target planet from the sun about 150 days after launch.

For two-dimensional minimum-energy trajectories to Venus, the vehicle should be launched when the earth is 53.5° ahead of Venus, which occurs next in June 1959. The proper relative orientation recurs once every 584 days.

The above discussion concerns the two-dimensional trajectories and does not consider the fact that the orbits of the other planets are not exactly in the plane of the earth's orbit (the ecliptic plane). For example, the orbit of Venus is inclined at an angle of 3.4° to the ecliptic, and the point at which Venus crosses the ecliptic in a northerly direction, indicated as the ascending node on Figure 8, is at a heliocentric longitude of 76.2° from the vernal equinox.

In general, the inclination of a planet's orbit prevents a two-dimensional minimum-energy trajectory, because the vehicle flies in the ecliptic. Even if the vehicle has an out-of-the-ecliptic velocity component, it returns to the ecliptic after traveling 180° around the sun. Thus, in general, a flight to another planet requires energy in addition to the minimum energy. Either the trajectory plane must be changed in midcourse or the burnout energy must be increased so that the trajectory angle around the sun is less than 180°. An out-of-the-ecliptic velocity component must then be added to rotate the plane of the trajectory so that it properly intersects the orbit of the target planet.

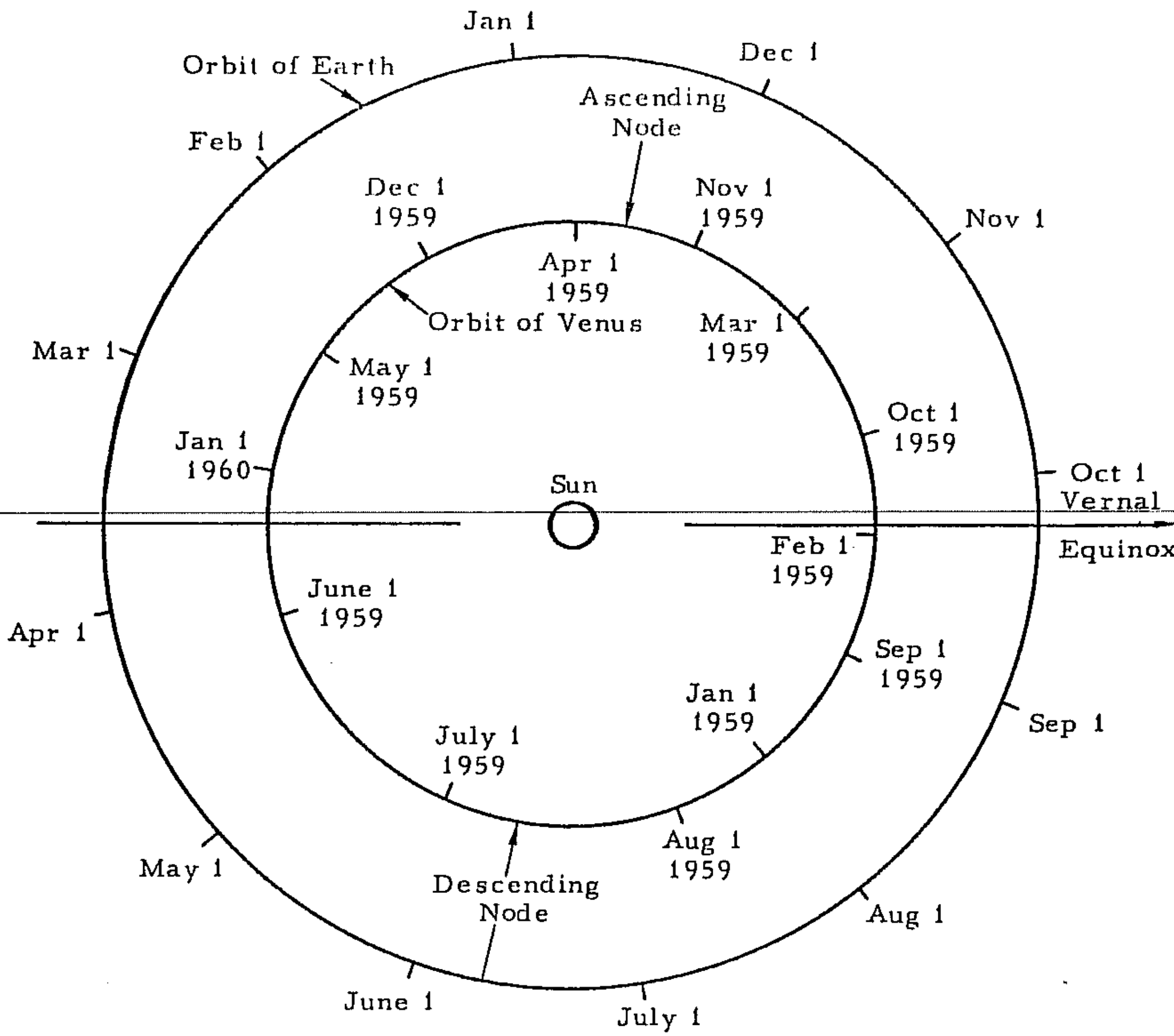


Figure 8. Geometry and Positions of Earth and Venus During 1959.

The next three-dimensional minimum-energy trajectory to Venus also occurs in the first week of June 1959. The planet Venus is at is ascending node (i.e., in the plane of the ecliptic) which is the best point for the vehicle to reach Venus on 2 November 1959.

2. Trajectory to Venus

As explained above, with the nominal trajectory utilized for the flight to Venus, the vehicle leaves the earth in the early part of June 1959, and reaches the vicinity of Venus in the early part of November 1959. The trajectory of the vehicle around the sun, together with corresponding positions of the earth and Venus, is shown in Figure 9. The vehicle will approach Venus on the side toward the sun in order that the retrorocket be properly oriented. The velocity that the vehicle must have with respect to the earth after it is free of the earth should lie in a direction approximately opposite to that of the earth's velocity and should be approximately 8000 to 10,000 ft/sec. For a burnout altitude of 885,000 feet, the burnout velocity should be approximately 37,000 ft/sec. The general nature of the trajectory near the earth is illustrated in Figure 10, which also shows the approximate attitude of the body of the vehicle. The relative position of the earth with respect to the body of the vehicle, which can be seen approximately from Figure 9, is illustrated in Figure 11.

3. Geometry of Satellite Establishment

The trajectory for the Atlas satellite vehicle is so chosen that the vehicle approaches Venus on a parallel path such that the vehicle would miss Venus by a distance, b, if Venus had no gravity. The vehicle is travelling 9000 to 11,000 ft/sec faster than Venus. The gravitational attraction curves the trajectory toward Venus so that the distance of closest approach, D_{ca}, is less than the extrapolated distance, b, as illustrated in Figure 12. The relation between these two distances is illustrated in Figure 13. In order that the distance of closest approach be approximately 8000 miles from the center of Venus, the extrapolated distance is chosen to be approximately 18,000 miles. When the vehicle has reached the distance of closest approach, the velocity has increased to 25,900 ft/sec as illustrated in Figure 12. At this distance the escape velocity from Venus is 23,190 ft/sec, and a velocity change of 2700 ft/sec

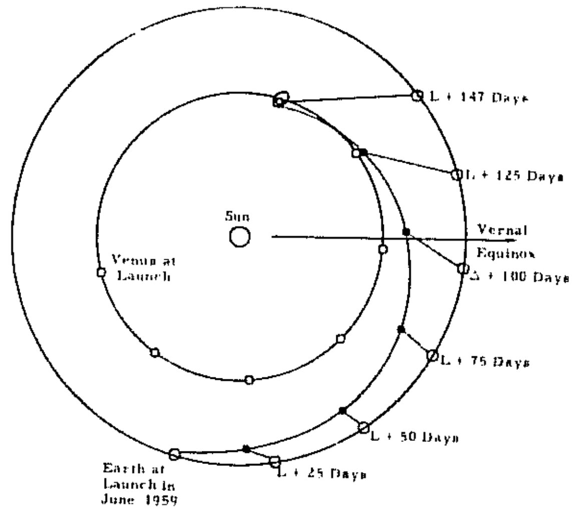


Figure 9. Position of Venus Satellite Vehicle and Earth During Flight to Venus.

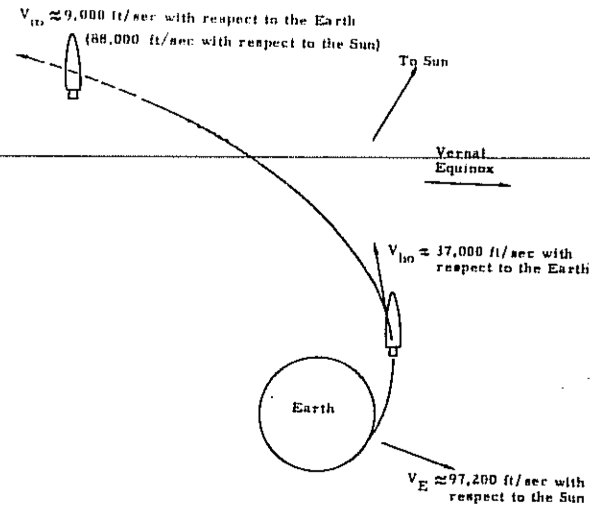


Figure 10. Venus Trajectory near Earth.

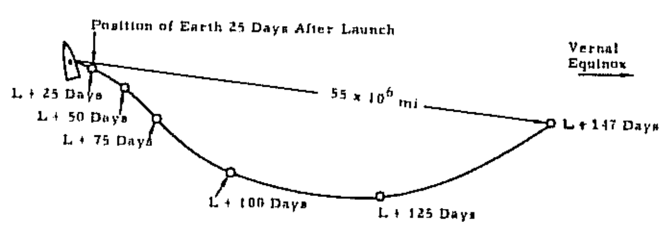


Figure 11. Attitude of Vehicle Going to Venus with Respect to an "Apparently" Moving Earth.

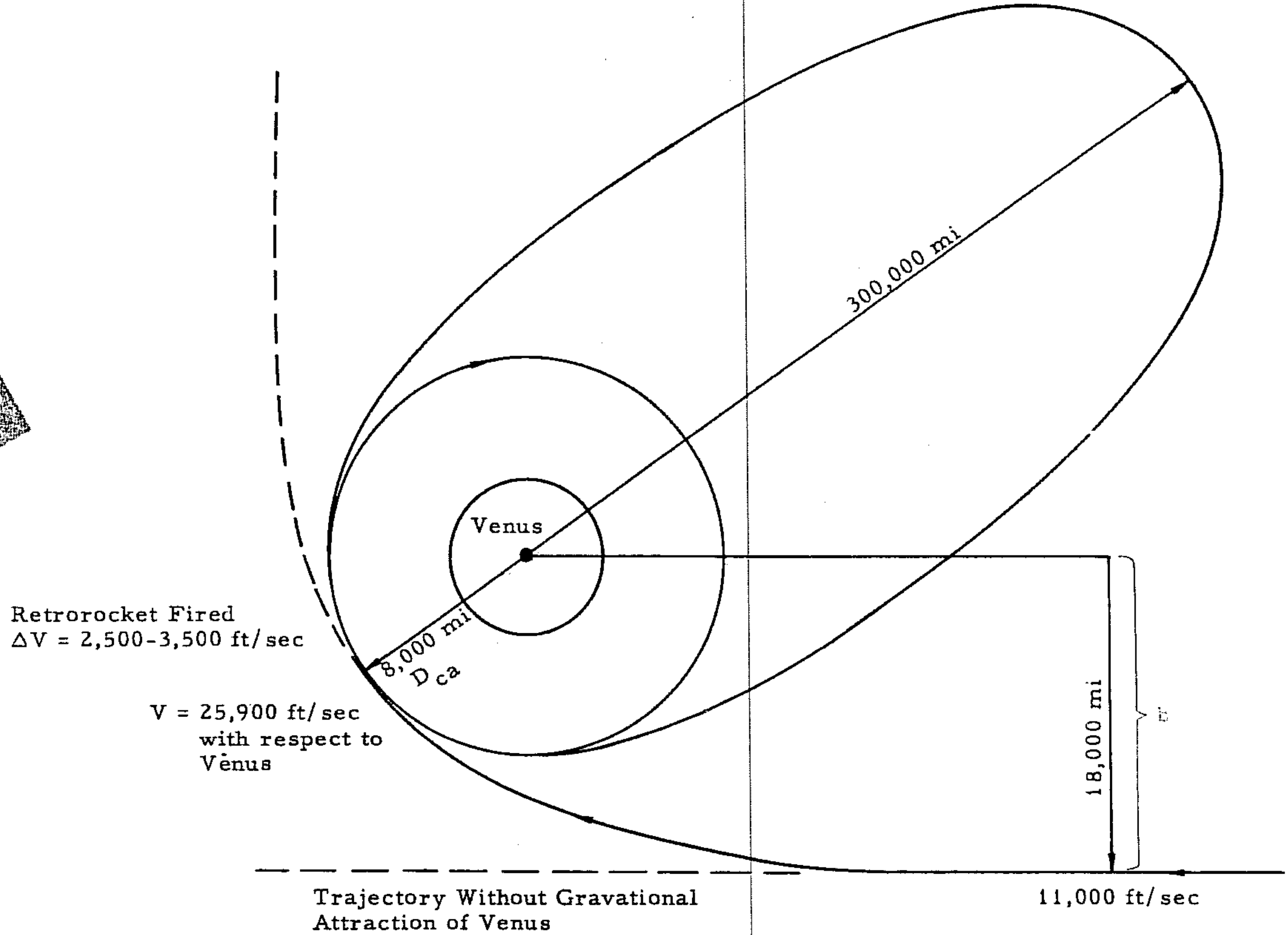


Figure 12. Sketch of Venus Satellite Capture Conditions.

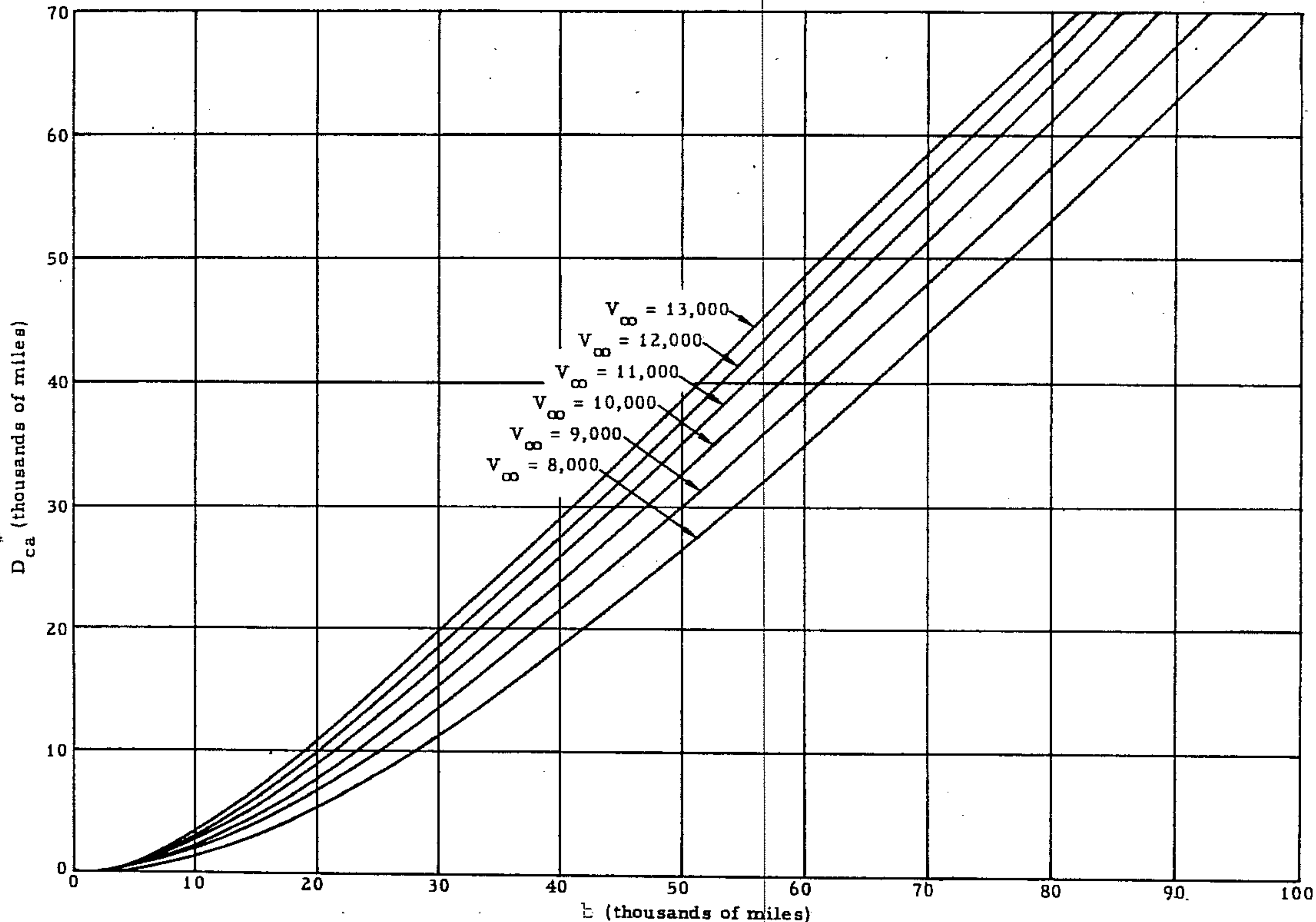


Figure 13. Distance of Closest Approach, b (Miss Distance Including the Gravity of Venus), as a Function of Extrapolated Miss Distance, b (Miss Distance Without the Effect of the Gravity of Venus).

is required for the satellite to be just barely captured. With this velocity, the greatest distance in an eliptic orbit which the satellite can get from Venus will be 300,000 miles.

4. Allowable Variation in Launch Time

The nominal trajectory is based upon launching during the first week in June 1959. The extent to which variations in launch time are allowable is determined by the additional velocity required to comply with other launch times and the extent to which these additional velocities can be achieved with the propulsion system. The effect of a 5° variation in the burnout velocity vector angle is a 7.5° variation in the inertial orientation of the residual velocity after the vehicle is free of the earth's gravitational field. This amount of control of the orientation of the residual velocity enables the same trajectory around the sun to be followed with a total launch time variation of 30 minutes. This is a preliminary estimate, and an analysis of the extent to which the November arrival at Venus can be relaxed may indicate a larger allowable variation in launch time.

E. Guidance Requirement and Accuracy

1. Launch Guidance

a. Launch Guidance of the Thor Booster

When the Thor is used as the first stage, the launch guidance is essentially the same as for Able-1. The angular orientation of the first stage is controlled by a gyro-stabilized autopilot which follows a pitch programmer. The termination of the thrust in the first stage is determined by the chamber pressure change occurring upon fuel exhaustion.

b. Launch Guidance of the Atlas

When the Atlas booster and sustainer are used instead of the Thor, the Atlas will be guided by the GE Mod III radio guidance. The basic guidance procedure would be similar to an ICBM guidance except for modification of the guidance equations to increase the accuracy with which the attitude of the Atlas sustainer body is controlled at shutdown.

c. Guidance of Terminal Stages

The Able stage will be guided using the STL Advanced Guidance System (AGS) coupled with the Burroughs Mod I Guidance Computer at AFMTC. This system will steer and shut off the Able stage with IRBM accuracy and control the initial attitude of the spin-stabilized third stage. The AGS will consist of two legs of an interferometer on 1000- to 2000-foot baselines which will determine vector velocity and position (resolving ambiguities by means of subcarriers) plus an airborne transmitter and receiver in the Able stage. Since attitude changes will be effected by turning at constant rate, only discrete commands will be sent to the Able stage. After stage 2-3 separation, the AGS will track the payload using another command receiver-transmitter configuration in the payload. After a short coast period (to reduce an ionospheric error), the AGS 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

Tracking data obtained in the first 30 to 60 days will be used to make more accurate estimates of the trajectory parameters than are available shortly after third-stage burnout. These trajectory parameters will be used to calculate the amount of trajectory correction required. The amount of necessary trajectory correction will determine the amount and time of the midcourse vernier correction. At this time, suitably coded commands are transmitted from earth to the vehicle to initiate firing. Seven command channels will be able to initiate 128 propulsive combinations.

The radio tracking system can furnish range, doppler velocity (range rate), and angular data. The use of all three forms of data is somewhat redundant and one form might be eliminated. Equipment limitations indicate the desirability of not relying strongly on angular data. An analysis of the effects of burnout errors on trajectory parameters shows that primary reliance can be placed on range and range-rate measurements, with angular data being used only for correcting the effects of earth rotation on doppler velocity. For example, the range will be known to about 500 miles and range rate to 1 ft/sec. Therefore, after 30 days the equivalent velocities will then be known to the order of 1 ft/sec.

3. Error Coefficients for Venus-Satellite Establishment

The trajectory for the Venus-satellite vehicle is close to a minimum-energy trajectory and has been calculated for 6 June. One of the primary errors is that caused by burnout velocity error and causes a miss of 10,000 nautical miles in b per ft/sec. The midcourse tracking described in the previous section should correct the trajectory so that the resulting error is equivalent to a burnout velocity error of 0.25 ft/sec.

The second error, flight path angle, causes a residual error of 50,000 miles/degree. Since angular errors at the burnout point become both angular and velocity errors in the sun's field, vernier and midcourse velocity corrections substantially reduces the effect of the angular error.

The third error, launch azimuth, results in an error of 100,000 miles/0.5 degree. However, since the trajectory to be flown is approximately in the Venus orbit plane, this error can be almost entirely removed by vernier and midcourse velocity corrections.

Section 4.3 has indicated that sufficient performance is available to allow a variation of the launch date from a nominal of approximately one week for the Thor-boosted vehicle. The effect of the guidance accuracy, however, may limit the launch date variation to a smaller interval. Preliminary estimates show that, if the probability of success of the mission is not to be seriously compromised, the launch date variation must be limited to ± 2 weeks. This estimate is not final and is based upon the restriction that arrival at Venus during the first week in November be maintained. Relaxation of the arrival-time constant will increase the performance required but may decrease the accuracy requirements. A more detailed study is needed to deter-mine the appropriate compromise between performance required and the guidance accuracy.

4. Tracking Accuracy Required for Astronomical-Unit Determination

The method planned for determination of the astronomical unit (AU) involves making accurate radio measurements from the earth of the missile position with respect to the earth. The trajectory relative to the earth is influenced primarily by burnout conditions and by the gravitational mass of the earth. When the missile is some distance from the earth (millions of miles), the influence of the uncertainty in the mass of the sun becomes appreciable.

The estimation procedure must involve enough measurements to permit the burnout conditions (or an equivalent set of trajectory parameters) to be calculated sufficiently accurately so that the AU uncertainty can be recognized despite the effect of burnout errors. Although an optimum procedure has not been devised, preliminary calculations indicate that the measurement accuracies described in paragraph F.1 makes possible a determination with reasonable probability of the astronomical unit to an accuracy of 10^{-4}. Estimates of the accuracy with which the astronomical unit is now known vary from 10^{-4} to 3×10^{-5}, the most accurate determination being that of Eugene Rabe. The value of the measurement is thus essentially a confirmation by a basically different method rather than an improved measurement. A significant improvement in the accuracy of the midcourse tracking system, or a more optimum processing of the measurements than the simplified method assumed, would make possible a better determination of the astronomical unit.

F. Tracking and Communications

1. Active Tracking

The Able-4 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 1ft/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 cy/sec should result in a range resolution for this system of better than 100 miles with adequate smoothing. A second modulating frequency of 0.2 cy/sec will be used to resolve the phase ambiguities in the 2 cy/sec 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 degrees should be achievable.

2. Communication Requirements

The same radio link which performs the tracking function described in the preceding paragraph furnishes two-way communications between the ground and the vehicle. The transponder is used during third-stage burning as well as for active tracking during free flight. In addition, the payload transmitter serves to transmit telemetry data. During third-stage burning and in free flight, 32 different commands will be provided. Two of these commands will be used to turn on and turn off the transponder transmitter from the ground. One command will be available to change transponder characteristics after vernier staging or vernier firing. Another command will be capable of staging the vernier assembly if required. During midcourse and terminal flight, provision must be made for a vehicle address, since two payload packages may be in orbit about the sun simultaneously. Seven commands will be available for firing up to 7 vernier rockets in 128 combinations for midcourse correction, one command for the firing of the injection rocket, and one or more commands for the firing of the injection rocket after a time delay. The time delay to firing of the injection rocket will be used in case the best time for injection rocket firing does not occur when the vehicle is above the horizon of the terminal guidance antenna. A transpond command will also be employed during the free-flight and terminal-tracking phases. The payload transmitter is turned on and off by command.

3. Frequency Considerations

The choice of frequency for the Able-4 tracking and communication radio link requires a compromise between several, sometimes divergent, considerations. These considerations fall under the broad headings of receiver noise, antenna efficiency, and the availability and efficiency of equipment and components.

In general, more development effort should be applied to the reduction of ground equipment receiver noise than to improvements in any individual payload package. Significant increases in payload weights will usually be utilized for more complex instrumentation and control systems or for increasing transmitter duty cycles or information bandwidths. Complexity in ground equipment is justified because the ground equipment may be used for a number of successive missions. Also, it is available for maintenance and is not subject to the vibration environment of the missile powered-flight phase.

Two low-noise receivers are available, molecular amplifiers or masers, and parametric amplifiers. On the basis of internally generated noise alone, development of both types of amplifiers for use in ground receivers is indicated. However, external radio noise from cosmic sources, as well as the atmosphere above the ground antenna, is of such a magnitude that reduction of internally generated noise below that available from parametric amplifiers may not be required. Although more information on effective antenna temperatures at frequencies above 100 mc is needed, present best estimates indicate there is a broad minimum in antenna temperature in the neighborhood of 400 mc, particularly for low antenna elevation angles. At frequencies below 400 mc, galactic noice is the dominant factor. Above 400 mc, the effective antenna temperature is determined by the temperature of the atmosphere and the temperature of the ground seen through the antenna sidelobes. At the higher frequencies, the effective noise temperature is determined largely by the extent to which the antenna sidelobes are suppressed. Therefore, efforts on the ground receiver will be directed toward development of a parametric amplifier for use at 400 mc.

Payload weight limitations require a high electrical efficiency for the transponder package. This in turn implies transistorization wherever possible. For the Able-1 transponder, the IGY frequency at 108 mc was chosen primarily because at that time this was very nearly the highest frequency at which the entire package could be transistorized in the short time available. The general availability of ground equipment led to the precise choice of frequency. However, at the higher transmitter powers required for the Able-4 mission, transistors are not available for the final transmitter stages even at 100 mc. At an output frequency at about 400 mc, it appears feasible to transistorize all stages of the transponder package except the last two stages of the transmitter and perhaps the input stage of the receiver. Noise figures of conventional receivers increase with frequency and the efficiencies of amplifiers decrease somewhat with frequency. Careful consideration of these factors and the availability of tubes at the powers required has indicated that the most desirable frequency at this time is in the neighborhood of 400 mc. The ground-to-air frequency will be 401.748 mc and the transponded frequency will be 378.116 mc.

4. Antennas

One of the possible objectives of the Able-4 mission is to obtain data on the oblateness of Venus and the period of its axial rotation. To perform this type of experiment, measurements of velocity and position must be performed at ranges of approximately 55 million miles, and an injection rocket must be fired; the initial orbit of the payload package about Venus must be determined; and subsequent observations of the perturbations on this orbit must be made. Since at the time of capture Venus is moving away from the earth, the amount of information regarding Venus that can be obtained is limited primarily by the range to which the payload package can be tracked from the earth. This experiment alone requires the use of the largest available ground antenna for the tracking system. As previously noted, the largest antenna available is the 250-foot parabolic reflector at Jodrell Bank which is operated by the University of Manchester. With the limitation on payload weight available for the the transponder and power supply, this is the only antenna which will perform the tracking mission for this probe.

Although use of a directional antenna on the vehicle would result in additional signal-to-noise ratio and hence a greater communication range or more useful communication bandwidth, an attempt to employ a directional antenna at this time would result in an undue reduction in over-all reliability of the vehicle.

For a spin-stabilized vehicle, a high-gain airborne antenna design, which would give a gain of about 10 db at injection of the vehicle into an orbit about Venus and also coverage over the entire trajectory, should have the following characteristics. The power should be equally split between a main lobe of half-power width of 5° and a secondary lobe whose peak likes 15° from that of the main lobe and whose half-power width is about 30°. There should be no null in the directivity pattern in the 30° solid angle between the directions of the main and secondary beams. A pattern with these characteristics may be obtained by first constructing an antenna which has the directivity desired of the main lobe and then distorting the pattern to give the additional secondary lobe. Since this is not an attractive design for the present payload package, the radiation pattern of the Able-4 vehicle will be essentially that of a dipole with a broadside gain of about 1-1/2 db and a gain of not less than -5 db at angles greater than 30° from the spin axis.

5. Power Requirements

The space and weight limitations in the vehicle require a minimum of power and only part-time operation. For a distance of 55 million miles, a transmitter output of 150 watts on the vehicle will be sufficient for tracking and communications purposes. The time of operation will be seriously limited in order to conserve power, i.e., about 5 minutes every 10 hours.

The power transmitted from the ground will be 10,000 watts. With the antennas specified above, the ratio of received signal level to the receiver lock-on sensitivity is approximately 10 db when the injection rocket is fired. The 150 watts transmitted from the vehicle should result in an operating margin of about 10 db at this range. The above calculations allow approximately 6 db for polarization losses and miscellaneous losses.

6. Commands

Commands will be transmitted to both the AGS transponder in the second stage and to the payload transponder. Commands to the AGS transponder are generated by simultaneously modulating the ground transmitter by any two ~~of six possible tones. The frequencies of these tones will be between 2000 and~~ 5000 cy/sec with the frequencies so chosen that second-order intermodulation products will not result in false commands. Figure 14 is a block diagram of the airborne command receiver system.

During midcourse and terminal flight, commands to the payload transponder are transmitted by phase modulation by various pulse sequences of a subcarrier, whose frequency is 400 or 500 cy/sec. The two possible subcarrier frequencies will provide a vehicle address when two vehicles are in flight at the same time.

7. Ground System

The ground system for midcourse and terminal guidance and tracking will be located at Jodrell Bank, England. The ground system consists of a transmitter, which feeds the 250-foot antenna through a diplexer, and a receiver. The transmitter has an output power of 10,000 watts and employs phase modulation for transmission of commands transmission of commands to the vehicle.

The receiver will be an improved version of the Microlock-type receiver with a low-noise input and an effective noise bandwidth of between

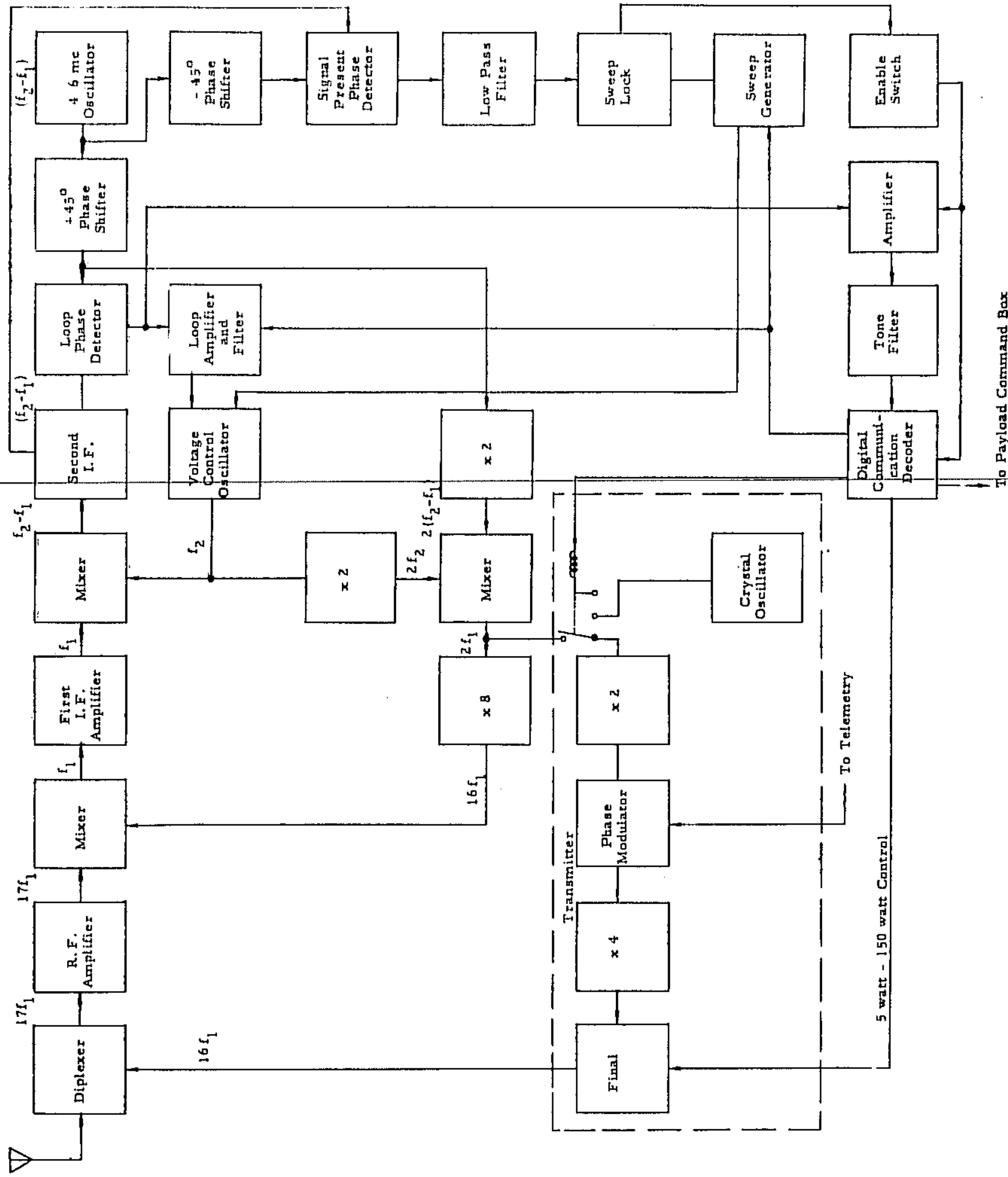


Figure 14. Schematic of Airborne Receiver.

10 and 40 cy/sec. The required sensitivity of -155 dbm for the ground receiver will require development of a low-noise parametric preamplifier and additional narrow banding of presently available Microlock receivers. Since this station is not used for guidance during powered flight, only the 0.2, 2, and the 400 or 500 cy/sec subcarriers are used for modulation of this transmitter. The command subcarrier will be phase modulated $\pm 90^{\circ}$ by the command pulse sequences. The individual pulse width will be about one second. The modulating subcarrier will be phase modulated onto the main carrier with a phase deviation of about ± 1 radian. The frequency of the ground transmitter should be adjustable to permit transmission not only at the nominal frequency f_t but also at f_t + 12 kc/sec and f_t + 24 kc/sec. This provision allows a reduction in the vehicle receiver search range and acquisition time, since the transmitting frequency will be so chosen that the received signal at the vehicle is close to the nominal receiver frequency.

8. Transponder

The transponder carried in the payload package consists of a coherent receiver, a command decoder, and a 150-watt transmitter. The receiver uses very narrow banding by phase-lock techniques and automatic search circuitry to provide acquisition in frequency. Although the command receiver is somewhat similar to the Able-1 command receiver, a number of significant changes have been made. For low-noise operation at the received frequency of 401.8 mc, a tube rf preamplifier may be used if a suitable tube can be found. Since the Able-4 mission requires many more commands than did the Able-1 vehicle, more tone channels are provided and also a digital decoder will be used after staging of the vernier assembly. This receiver will perform two functions during the Able-4 flight. During third-stage tracking, it will operate as a transponder for the advanced guidance system. The command channels will be used for firing the vernier rockets. During this phase of operation, the received signal levels will be high, and rapid acquisition by the airborne receiver is required. For operation as a doppler command receiver for midcourse velocity correction and for tracking and command functions at maximum ranges on the order of 55 million miles, the receiver sensitivity is an acquisition and command channel sensitivity of -140 dbm.

To obtain this sensitivity, the equivalent noise bandwidth will be reduced to 40 cy/sec or less.

Figure 1[illegible] is a block diagram of the airborne system transmitting and receiving equipment.

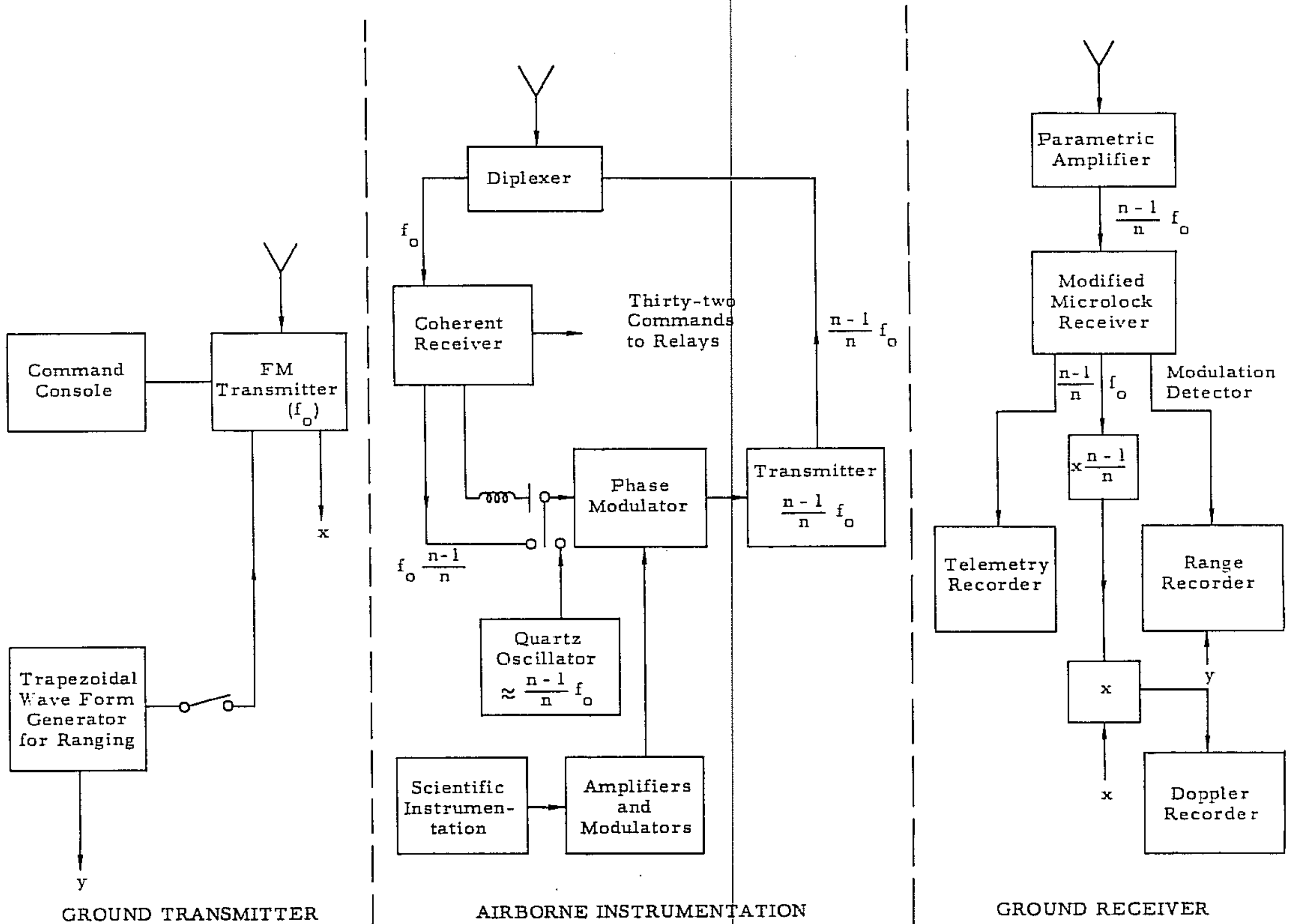


Figure 15. Schematic of Transmitting and Receiving Equipment.

G. Payload Characteristics

1. Scientific Experimentation and Associated Instrumentation

Data gathered will fall naturally into two classes: that characteristic of the general region of the solar system at distances of 66×10^6 to 150×10^6 miles from the sun, and that peculiar to the near vicinity of Venus. Information of the first type will be obtained, of course, from both types of vehicles: the solar system probe and the Venus satellite

There are several measurements of very great theoretical and practical interest which can be made in these vehicles. One of the first is an independent determination of the size of the solar system. By measuring and artificial orbit in terrestrial units, feet or feet per second for instance, one can determine an equivalence between astronomical units(major semi-axis of the earth's orbit taken as unity) and terrestrial units.

Table 2 in Section 3 summarizes these experiments. Estimated weights and powers are included.

a. Measurements of the Astronomical Unit of Length

An accurate knowledge of the astronomical unit of length (AU) is necessary for almost any type of space experiment. The equivalence between the AU and the centimeter is established through the measurement of the solar parallax, π_o. This measurement has been carried out in five ways in the past but the results of these methods are inconsistent with each other. Therefore, a measurement by a different means would prove to be quite valuable.

The AU can be calculated if the position and velocity of a space vehicle are known at two different times, preferably widely separated. In the probes, this information will be available from the radio guidance tracking equipment (Section D). Therefore, it should be possible to carry out this experiment without adding any additional equipment to the vehicle.

b. Magnetic Field Measurements

The purpose of this experiment is to measure the magnetic field in space and in the vicinity of Venus.

Magnetic field theories concerning the nature of solar flare material rejected from the sun are in conflict. Measurements made from a space probe in the vicinity of such flare material would cast light upon the mechanism responsible for their behavior. At sporadic intervals interplanetary fields are expected to rise to as high as 5 milligauss. During quiet times, such fields should be down several orders of magnitude. Determinations of the interplanetary field would be made during the flight and correlated timewise with solar observatory data.

The magnetic fields of the planets are unknown. On the Venus probes this experiment may allow a determination of any magnetic fields in the environs of Venus and might give sufficient data to calculate the magnetic moment of the planet. There will be two magnetometers: one will be the Able-1 search coil, measuring amplitude and phase of component of magnetic intensity ~~normal to spin axis; the other will be flux gate along spin axis.~~

c. Meteorites and Interplanetary Dust

The purpose of this experiment is to determine the momentum, and flux density of meteoric material. The complete assembly will weigh less than 1.0 pound including the auxiliary electronics exclusive of telemetry.

d. Cosmic Ray Experiments

The University of Chicago cosmic-ray telescope, carried in Able-1, will be used for determining cosmic-ray flux and the existence and location of disorded magnetic field.

A Cerenkov counter, consisting of photomultiplier, high-voltage power supply, amplifier, and scalers for determining high-energy proton and electron fluxes in space, will be carried with the Atlas-boosted vehicle.

e. Geiger-Muller Experiment

The University of Minnesota is preparing an experiment consisting of a Geiger-Muller tube and an ionization chamber of the Neher type. This experiment is intended to yield a measurement of the primary cosmic-ray flux accurate to one per cent and, in addition, to measure the mean specific ionization per incident particle. The equipment weight is two to three pounds.

f. VLF Monitor

The purpose of the very low frequency experiment is to monitor the naturally occuring noise in the vlf band of the radio spectrum. Sources of such noise are the sun, planetary atmospheres, and the interaction of plasma and magnetic fields. The results of this experiment may therefore yield valuable data concerning the variations in solar activity, the existence of possible weather effects on the planet Venus, and the reaction of interplanetary plasma streams with the magnetic fields in space.

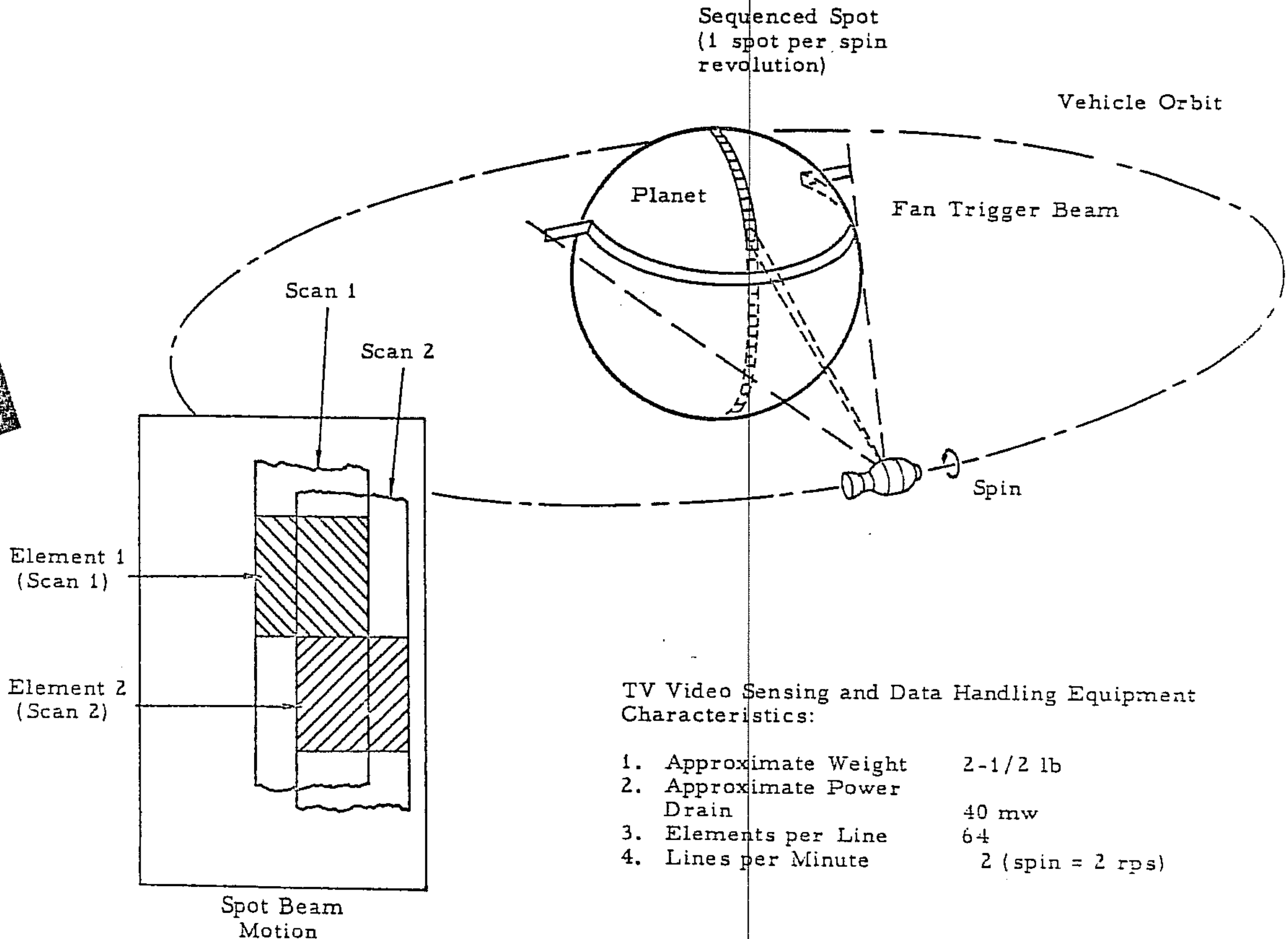
g. Temperature Measurements

Provision for a maximum of eight temperature measurements, using thermistors, will be made, four inside the payload and four on the solar cell paddles. The internal measurements will establish the temperature of various parts of the package and determine the over-all temperature gradients. The four measurements on the solar cell paddles will be used to determine the temperature at which the solar cells are operating and consequently their efficiency. The total number of temperature measurements will be determined, in part, by the results of Able-3 measurements.

h. TV Pictures of Venus and the Earth

A television system to be carried on the Atlas-boosted vehicle for operation over a 2-cps passband (1-cps information bandwidth) will weigh 2.8 pounds complete up to the modulators and have an over-all power drain of less than 40 milliwatts. The scan consists of a fan beam trigger which gates a scaler, the latter sequencing a spot scanner to sample one brightness level per vehicle revolution, each spin cycle advancing the spot by a small amount. The over-all effect is to scan one line in a number of spin revolutions with some overlap from line to line. The orbital velocity of the vehicle will be used for line sequencing. This system is illustrated in Figure 16.

There is evidence of surface shading in Venus as seen from earth. Visual and photographic observations made at the Pic du Midi Observatory and at the Flammarian Observatory show large markings.



TV Video Sensing and Data Handling Equipment Characteristics:

1.	Approximate Weight	2-1/2 lb
2.	Approximate Power Drain	40 mw
3.	Elements per Line	64
4.	Lines per Minute	2 (spin = 2 rps)

Figure 16. TV System for Venus Satellite Vehicle

The markings are difficult to observe because of their low contrast, and at the Pic du Midi Observatory have been studied by a process of integrating a number of separate plates to increase the contrast of the markings. A study of such markings at close range will contribute significantly to the knowledge of the planet's atmosphere, and possibly establish definitely its rate of rotation. The television system will have a selected spectral sensitivity, the long wavelength region of the spectrum being preferable for Venusian observation because of atmospheric scattering.

i. Operational Characteristics

The experimental equipment has been selected so as to be compatible with available telemetering equipment. In addition to experimental equipment, digital accumulators will be required for the proportional radiation counter, the Geiger-Muller, ion chamber, the micrometeorite experiment, the temperature indicators, and the Cerenkov counter, so that these data accumulated during periods when information is not telemetered to the ground. These accumulators with converters will weigh about 1.5 pounds and use about 150 mw of continuous power.

The search coil magnetometer, the flux gate magnetometer, and the vlf monitor experiments cannot have their data digitalized and therefore they will be turned on and the data sent only upon command. The length of time that they are on will depend upon the power required. Thus, while the vehicle is within 8 million miles of the earth and use of the 5-watt transmitter is still possible, more data will be available (because of the very small drain on the power supply.) At greater distances, when the 150-watt transmission is used, the " on' times will be quite short - - 5 minutes every ten hours.

2. Telemetry

The telemetry data will be transmitted to ground simultaneously with transmission of the tracking data. The telemetry will utilize a digital system. Twelve digits per word will be transmitted, two for synchronizing and ten for carrying digital information from the experiments. Because the data will be sent intermittently, a total count must be stored in digitalized form for such experiments as cosmic ray counters and the micrometeorite experiment.

The transmitter can radiate at 150 or 5 watts. Three choices of pulse rates are possible: 1 pulse per second; 8 pulses per second; and 64 pulses per second. This will permit 64 bits per second at 150 watts to be sent at a distance up to 8 million miles or will permit 1 bit per second from launch to 60 million miles These are equivalent to bandwidths of 64 cy/sec and 1 cy/sec, respectively. The transmitter will operate at 150 watts upon command approximately 5 minutes out of every 10 hours or at 5 watts for 3 hours every 13 hours. The telemetry will weigh 2 pounds, the transmitter for both telemetry and tracking will weigh 9 pounds, and the receiver, 4. The programmer will weigh 0.5 pound.

3. Power Supply

The electrical power requirements for the payload of the Able-4 space probe vehicles are largely dictated by the transmission distances involved. Consideration of this distance along with the antennas and receivers available for the ground tracking stations, lead to the requirement for 150 watts rf transmitted power. Assuming an over-all efficiency of 30 per cent for the airborne transmitter and voltage converter, the total power available for a communication link to the earth must be 500 watts.

Continuous transmission of such quantities of power is completely impractical at present, therefore the airborne transmitter must operate intermittently. In view of the quantity of information to be transmitted and the acquisition problems attendant to very narrow band receivers, it is desirable that the "on" time of the transmitter be no less than 5 minutes. This is compatible with practical weight of storage batteries which can be carried. The "off" time is determined by the time required to recharge the storage batteries from the solar cells.

To supply a 500-watt peak load requires a combination of silicon solar cells and nickel-cadmium cell batteries with a watt-hour per pound ratio of 8.

The solar cell chosen is the Hoffman 120C which has a 1.8 square centimeters of active area. With an over-all conversion efficiency of 8 per cent, each cell will deliver approximately 14 mw at 25°C. A continuous load of 15 watts thus requires 1070 active cells; i.e., each cell must be fully exposed to the sun's rays at normal incidence.

The intermittent load of 500 watts for 5 minutes is approximately equivalent to 42 watt hours of energy. Multiplying by 1.4 (the charge-to-discharge "inefficiency factor" of nickel-cadmium cells), this becomes approximately 60 watt hours of energy necessary to recharge the batteries. Over a 10-hour period this is 6.0 watts required from the cells or 430 fully active cells at 25°C.

The configuration chosen is that of four paddles extending from the central payload. Each paddle will be covered by solar cells on both sides. The paddles will be canted at the appropriate angle to maximize the average number of cells normal to the sun's radiation. Such a configuration results in a utilization factor of 1/6. Therefore, the number of cells computed above must be multiplied by a factor of 6 resulting in a total of 9000 cells.

The output of the solar cells must be derated at the rate of 0.6 per cent degree centigrade from 25°C. The absorptivity-to-emissivity ratio of uncoated silicon solar cells is approximately 3. This results in calculated equilibrium temperatures in space of 100 to 250°C. Coating the surface of the cells will result in considerable improvement in the equilibrium temperatures reached. This problem is undergoing intense investigation and some form of coating will be used, possibly vacuum sputtered silicon monoxide.

The configuration of the paddles will be a light aluminum spar into which are fastened modular "pallets" of solar cells. The base of the pallet will be aluminum honeycomb to which solar cells are cemented on both sides. All the cells on one side will be connected in series such that their output, under full sun radiation, will be 30 to 40 milliamperes of current at the appropriate voltage to charge the batteries. A blocking diode will be incorporated in each series string to eliminate electrical loading of the illuminated cells by the dark cells. The outputs of all pallets will be connected in parallel and fed to the battery pack located in the central payload.

The nominal output voltage of the central battery pack will be 16.8 volts (fourteen nickel-cadmium cells in series) This is considered an optimum compromise when taking into account the working voltage of available components, such as relays and transistors, the rate of current drain required and the number of solar cells required in series. Only one central voltage will be availabe from this system.

4. Temperature Control

In the Able-4 mission, the payload will experience a factor of two increase in solar radiation above that at the earth. This will cause a mean payload temperature rise of the order of 90 to 100°F. Because of the temperature limitations imposed by the payload electronics, the fourth stage rocket, and the vernier motor, temperatures must be maintained within a range of approximately -25 to +105°F. It is evident, then, that some form of temperature control system is required to meet these temperature specifications. Temperature control, in general, is affected by controlling the solar absorptivity and long wavelength emissivity. The trajectory is such that the sun look angles will cover almost a full 360° so that the payload will be made approximately spherical so as to always present the same area for intercepting solar energy.

If reduced performance can be tolerated in the vicinity of the earth, it might be possible to reduce the lower operating temperature in the vicinity of the earth to such a low value that the temperature increase on traveling to Venus would still be within the desirable operating range. For example, if the equilibrium temperature of the position of the earth were -25°F or less, then the maximum temperature would be below the upper limit of 105°F when the vehicle reaches Venus. To the extent that such reduced temperature is acceptable in the vicinity of the earth and during midcourse, such a scheme is feasible.

It is expected that the payload transmitter will dissipate about 350 watts with a duty cycle of about 1 per cent. Since the on-time is of the order of 5 minutes, it may be necessary to isolate this transmitter from the remaining payload components and probably provide a means to smooth out the temperature fluctuations by an energy sink such as the phase change of a solid to a liquid. Again, depending on the duty cycle, it may be necessary to provide extra area for emitting the dissipated power such as by extended surfaces on the payload in the vicinity of the transmitter or by mounting the transmitter out of the main payload package in an out-board package.

The silicon solar-cell power supply will be removed from the payload surface partly to avoid conflict with the active temperature control surfaces and partly because there is likely to be insufficient area on the payload surface to meet the power requirements. The solar cells will then be placed on both sides of four paddles which are extended out laterally from the payload sometime during the spin-up or 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, 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 solar cell is about three, the cells will run excessively hot unless their surfaces are coated to alter the long wavelength emissivity. Silicon monoxide and glass are two possible coatings which will control the temperature of the solar cells and will keep it at about room temperature near earth. Near Venus the efficiency will fall somewhat even with the coating. The shadowing of one paddle on another and of the paddles and payload on each other must be taken into account in achieving proper temperature control in the payload and to determine temperatures on the solar cell paddles.

5. Stability Requirements

Assuming a long slender injection rocket in the fourth stage (say the TE-80) of the Atlas-boosted vehicle, the remaining fourth stage weight must be placed at a large distance from the axis of symmetry to achieve a larger moment of inertia in roll than in pitch. The use of solar cell paddles (arms projected radially outward from the payload) greatly increases the roll moment of inertia. Nevertheless, the payload diameter may run to 4 feet or more in order to achieve long-term spin stability. A careful appraisal of the fourth-stage weights must be made in order to minimize the payload diameter and paddle length, yet retain long-term spin stability.

Because the Thor-boosted vehicle has no injection rocket in the fourth stage, long-term spin stability can probably be achieved within a 40-inch payload diameter if solar cell paddles are used.

The spin rate required to meet the third-stage velocity vector specifications is a minimum of 2 revolutions per second. Spin is achieved after second-stage cutoff and before separation.

Deploying the solar cell paddles, which are contained within the nose fairing during first-stage thrusting, will require careful consideration. If the vehicle is spinning during paddle deployment, then a serious attitude error may be generated by nonsymmetrical paddle motion. To avoid these difficulties, paddle deployment during second-stage thrusting, or between second-stage cutoff and spin-up, is possible.

The short interval between second-stage cutoff and spin-up can probably be provided for solar cell paddle deployment.

Finally, external torques on the spinning payload, such as radiation pressure, magnetic fields, gravitational gradients, etc., might cause an undesirable precessional motion. However, calculations for the upper bound of such effects indicate that, for the Venus mission, the attitude perturbation from these effects is certainly less than a fraction of a degree, a negligible amount.

H. 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 Thor-boosted proposed test vehicle will be essentially identical to the equipment used for the Project Able-1 Lunar Probe Program and to the equipment anticipated for the Able-3 Satellite Program.

The launch complex facilities and ground support equipment required for the first, second, and third stages of the Atlas-boosted test vehicles are not in existence at this time. 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 modification 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.

Detailed program support requirements are spelled out in Section V of this development plan.

IV. PROGRAM SCHEDULE

A. Program Schedule

The major milestones in the proposed program schedule are shown in Table 3. The schedule is tentatively based on the tests being conducted early in June 1959. Launch dates will be determined after detailed planning and coordination with AFMTC, the booster contractors and the respective STL program offices, but the sequence of operations will be essentially as shown.

Table 3. Major Milestones, Able-4 Program.

	First Flight (Thor)	Second Flight (Atlas)
Receipt of Stage 2 at LAX	28 January 1959	2 February 1959
Payload Design Freeze	30 January 1959	30 January 1959
Payload Prototype Operating	16 March 1959	15 April 1959
AGS Flight Unit Delivery	6 April 1959	20 April 1959
Payload Complete	17 April 1959	30 April 1959
Covers on Complete LAX	17 April 1959	1 May 1959
Stage 2 at AFMTC	27 April 1959	11 May 1959
Payload to AFMTC	7 May 1959	15 May 1959
MOS	22 May 1959	28 May 1959
Launch Date	~3 June 1959	~6 June 1959

V. PROGRAM SUPPORT REQUIREMENTS

The following items of support in the form of government-furnished equipment, services, and facilities will be required:

1. A Douglas Aircraft Company Thor first-stage booster and a Convair Atlas first-stage booster plus airlift, engineering services, and field support services.
2. Two Aerojet General Corporation AJ 10-101 second-stage propulsion assemblies (including hydraulic system) plus ground support equipment and checkout equipment.
3. Aerojet General Corporation assistance to STL in Los Angeles for propulsion system assembly and checkout plus field support services at AFMTC for Stage 2 and 3 assembly, checkout, and launch.
4. Two Stage 3 solid rocket motors for flight use plus additional units for static tests and backup.
5. Arnold Engineering Development Center facilities and services for static firing tests.
6. Air Force Missile Test Center launch facilities, range support, hangar space, and office space.
7. Ground communication links.
8. Military airlift.
9. Military travel.
10. Miscellaneous Air Force stock items.