Abstract

A planetary entry vehicle is described which is to be launched from a mother vehicle which executes the near miss of a planet. The entry vehicle travels through planetary space and enters the lower atmosphere. Direct measurements of physical quantities are made during the subsonic descent to the surface of the planet which terminates the flight. It can be shown that one pound of payload is sufficient to obtain a large amount of information describing the atmosphere. Measurements are transmitted to a vehicle on a near-miss hyperbolic or elliptic orbit with respect to the planet and stored for subsequent transmission to earth.

Introduction

It has been determined that, by carefully selecting the instruments and evaluating the information which could be derived, a one-pound payload dropping through the atmosphere of Venus with a $W/C_D^A$ of ten could be useful as a micrometeorological probe (1). The feasibility is demonstrated for delivering the probes which are to be launched from interplanetary vehicles or planetary satellites. The requirements are obtained for designing a system which launches an entry vehicle from a mother vehicle on a trajectory which performs a near miss of the planet. The operation is conveniently divided into three parts: 1) the application of the terminal corrective velocity, 2) entry into the planetary atmosphere, and 3) communication from the entry vehicle to the mother vehicle during the experiment.

Application of the Terminal Corrective Velocity

The following ground rules are suggested for developing a satisfactory delivery system.
1. A fixed linear impulse is imparted to the entry vehicle. The entry vehicle is spin stabilized while its engine is active, and is despun following motor burnout and ejection. The electrical system is deactivated after motor burnout.

2. The launch mechanism is designed to minimize the possibility of damage to the mother vehicle.

3. After burnout, the direction of the velocity vector of the entry vehicle has a tolerance sufficiently narrow to allow entry into the planet's atmosphere. The sector of the planet entered permits line-of-sight transmission to mother vehicle.

4. The guidance system carried in the mother vehicle is used for aiming the entry vehicle. The miss distance, distance from the center of the planet at time of launch, and the excess hyperbolic velocity of the mother vehicle are parameters used to orient the launch tube. The control system of the mother vehicle must be adequate to compensate for the reaction imparted to the mother vehicle during launch.

**Launch Tolerances**

A low thrust is used to spin and separate the entry vehicle from the mother vehicle (2). The axis of the firing tube is oriented by a servo in the plane of the path of the mother vehicle and a line through the swivel point of the launch tube and the center of the planet. The tolerance angle in the resultant path of the entry vehicle due to a thrust eccentricity and deviation in specific impulse is taken as ± 1-1/4°. This allows ± 17-3/4° tolerance in the direction of the applied corrective velocity at burnout. The spin required for the above tolerances is 50 rpm, which is obtained from a thrust of 25 lb along a 5° helical angle in the rifling of the launch tube, 3-1/2 inches in length. The ± 1-1/4° tolerance angle requires that the launch be made from 24 effective planetary radii from the planet. An effective planetary radius is the distance of the asymptote of the hyperbola which misses the center of the planet by one planetary radius. In the case of Venus and earth, the factor for obtaining the effective planetary radius is approximately 1.8.

The corrective velocity imparted to the entry vehicle is $s_\infty / 4$, where $s_\infty$ is the magnitude of the excess hyperbolic velocity of the mother ship. The corrective velocity is applied at right angles to the path of the mother vehicle for
maximum effectiveness in turning the course of the entry vehicle into an impact trajectory. A reasonable value of $s_\infty$ is 6 km/sec which results in a corrective velocity of 1.5 km/sec. The corrective velocity applied along an asymptote is sufficiently accurate to impact the planet provided that the miss distance of the asymptote does not exceed seven effective planetary radii.

The weight of the rocket motor required to apply the above correction will be approximately 5.5 lb. The weight of the solid propellant is estimated as 3.3 lb for a specific impulse of 275 sec and a total impulse of 900 lb-sec. The weight of motor inerts is estimated at 1.1 lb; and springs for ejection and spinning, 1.1 lb.

An estimate of the reaction angular impulse due to the rifling in the launch tube is 0.7 in.-lb-sec, and the linear impulse, 0.04 lb-sec. Representing the mother vehicle as a cylindrical tube 10 ft long and 6 ft in diameter, with a mass of 30 slugs and uniformly distributed density, then an estimate of the pitch angular velocity is 0.01°/sec and roll angular velocity is 0.05°/sec. The launch tube is assumed to be attached to the frame of the mother vehicle 3 feet from the center of gravity of the mother vehicle. Assuming that the control system of the mother vehicle can compensate for the anticipated angular impulses due to the launch operation, the residual angular velocities should be considerably less. The damping and compensation for the vibrations due to instruments extended at the end of booms should also be considered.

**Guidance Requirements for Launching**

The mother ship is sent to the planet by means of a heliocentric transfer ellipse which intersects the orbit of the planet. As the planet is approached, the lines tangent to the transfer ellipse are considered to be instantaneous hyperbolic asymptotes. The envelope of the set of two body hyperbolic orbits corresponding to each asymptote represents the approximate solution for the perturbation of the transfer ellipse by the planets gravitational field.

The perturbation of the planetary entry vehicle applied by a rocket motor is obtained by obtaining the correct velocity component at right angles to the path of the mother vehicle.

Let $s_\infty$ = excess hyperbolic velocity of the mother vehicle

$g_o$ = acceleration of gravity at the planet's surface

$R_B$ = miss distance of the asymptote of the mother vehicle's trajectory.
\( R_P \) = radius of the planet
\( F \) = factor for obtaining effective radius of the planet
\( \gamma \) = angle between the line pointing toward the center of the planet and a line tangent to the surface of the planet
\( \gamma' \) = the angle between a line pointing toward the center of the planet and a line tangent to the sphere of radius \( FR_P \)
\( \Delta v_t \) = fixed linear impulse provided by the rocket motor
\( \Delta v_n \) = component of \( \Delta v_t \) at right angles to the path of the mother vehicle
\( \phi \) = angle between the line directed toward the center of the planet and the line tangent to the path of the mother vehicle
\( \theta \) = correction angle in the ecliptic through which the launch tube is swiveled in order to provide the correct component \( \Delta v_n \)

Assume that the launch tube is initially pointed toward the center of the planet. The angle through which the tube is swiveled in order to direct the path of the entry vehicle on a course midway between a normal and grazing entry into the planet's atmosphere is \( \phi - \frac{\pi}{2} + \theta \). The + \( \theta \) correction angle is in the direction of the mother vehicle's motion and provides for an earlier arrival of the entry vehicle. It is therefore advantageous for allowing communication of the entry vehicle with the mother vehicle.

\[
\Delta v_n = \Delta v_t \cos \theta = s_\infty \tan \left[ \phi - \frac{\gamma'}{2} \right]
\]

\[
\theta = \cos^{-1} \left\{ \sqrt{\frac{s_\infty^2 + 2 g_0 R_P \sin \gamma}{s_\infty^2 + 2 g_0 R_P \sin \gamma}} \tan \left[ \phi - \frac{\sin^{-1} (F \sin \gamma)}{2} \right] \right\}
\]

\[
\phi = \sin^{-1} \left\{ \frac{\Delta v_t}{R_B s_\infty} \right\}
\]

\[
\phi = \sin^{-1} \left\{ \frac{\Delta v_t}{R_B s_\infty} \right\}
\]

\[
\phi = \sin^{-1} \left\{ \frac{\Delta v_t}{R_B s_\infty} \right\}
\]
If the corrective velocity is applied at a distance too far from the planet to make use of the measurement of $\gamma$, then the perturbation of heliocentric elliptical orbits rather than straight lines may have to be considered. The above guidance equations are based on the assumption of a two-body problem. The effect of the sun should be considered in order to obtain more general guidance equations. For the case under consideration this degree of refinement might not be necessary due to the modest tolerances required for aiming the launch tube.

An approximate estimate of the travel time of the entry vehicle from launch to the time of subsonic entry into the atmosphere is 8 hours for the 40,000-km miss, 3 hours for the 20,000-km miss and 1 hour for the 10,000-km miss. This suggests that the larger miss trajectories would be advantageous from the standpoint of allowing more time for the fly-by attitude control system to eliminate disturbances caused by the launch operation.

Entry into the Planetary Atmosphere

Of the planets Mars, Earth, and Venus, the most difficult structural and heating problems in the design of a satisfactory entry vehicle are probably offered by Venus. Although the structure and composition of the atmosphere are not definitely known (3), the surface density may be an order of magnitude higher than that of earth, and the surface temperature possibly as high as 300°C, and CO$_2$ appears to be a major constituent of the atmosphere. The speculations on the atmosphere structure and composition are based on interpretations of astronomical data. The conditions actually encountered could be much different from these. However, the above conditions are believed to be the severest that could be encountered by an entry vehicle and are assumed for the purpose of designing the vehicle to survive the entry into the atmosphere of Venus until the surface of the planet is reached.

In order to survive hypervelocity atmospheric entry, a heat shield is designed (4) to withstand maximum heating and loading. The instrumented capsule, designed to make a stable subsonic descent through the lower atmosphere, is housed in the nose of the heat shield and ejected by a spring after the hypervelocity entry is effected. The antenna for the transmitter consists of elastic spring wire arms wrapped inside the heat shield which extend when the capsule is removed from the heat shield. The rate of descent through the lower atmosphere is subsonic and is controlled by elastic vanes which are extended when the capsule is removed from the heat shield.
Entry trajectories were computed using the Allen-Eggers approach. The atmospheric density distribution is taken as \( \rho = 1.0 - 4.86 \times 10^{-5} h \), where \( \rho \) is in lb/ft\(^3\) and \( h \) is altitude above the surface of the planet in feet. A low \( W/C_D A \) (hypersonic) of 8 lb/ft\(^2\) was chosen which provides complete deceleration to subsonic velocities for any entry angle at high altitudes. Entry angles of 90° and 10° were considered, the former providing maximum deceleration and heating rates, and the latter representing a reasonable minimum entry angle and producing maximum total heat transfer to the vehicle. Entry velocity was taken to be 40,000 fps corresponding to the maximum expected hyperbolic excess velocity. Peak deceleration occurs at about 160,000 feet altitude for the 90° entry and has a value of 14,300 fps\(^2\), or about 445 earth g. For the shallow entry peak deceleration is 2,480 fps\(^2\) at about 190,000 feet. Sonic velocities are reached at about 115,000 feet for the 90° entry and 150,000 feet for the 10° entry. The rest of the descent is made at "terminal" velocities and requires 15 to 30 minutes. By contrast, the times of significant heating and loading are about five seconds for the 90° entry and 30 seconds for the 10° entry.

**Heat Transfer.** The principal modes of heat transfer to the vehicle surface are laminar convection and radiation from the gas in the shock layer. The latter was found to be important in the stagnation region and only for the 90° entry. At the stagnation point, laminar heat transfer rates were computed using the equations of Fay and Riddell modified to account for a CO\(_2\) rather than an air atmosphere as in Chapman's equations (NACA TN 4276). The cold-wall assumption was made. Radiation from the shock layer was computed by estimating the ion concentration in the gas (CO\(_2\)), comparing it with air ion concentration at the same temperature and density, and multiplying air emissivity at these conditions by the ratio of the concentrations. Thermodynamic equilibrium was assumed. It was found that for the same total enthalpy and stagnation pressure conditions, the emission from the two gases is about the same. Aft of the stagnation point radiation from the shock layer becomes insignificant. Convective heat transfer rates were related to stagnation point rates using the "local similarity" technique of Lees.

**Heat Shield Thickness.** The basic structure and heat shield are formed of a single material Fiberglas reinforced phenolic. An ablation temperature of 2,500°F and an effective heat of ablation of 800 plus 0.3 \( \Delta h \) Btu/lb were assumed, where \( \Delta h \) is the enthalpy difference across the boundary layer.
Total ablation thicknesses at the stagnation point were 0.056 in. for the 10° entry and 0.027 in. for the 90° entry. The insulation thickness was determined by requiring a maximum temperature in the load-bearing structure of 800°F and was found (conservatively) to be 0.035 in. in the stagnation region for the 10° entry. Maximum aerodynamic loads occur for the 90° entry; about 31 psi at the stagnation point. The failure mode is buckling of the hemispherical shell and requires a wall thickness of 0.010 in. Adding the ablation, insulation, and structural thickness gives a conservative estimate of the total wall thickness requirement 0.10 in. This thickness has been assumed uniform over the hemispherical section. A similar procedure on the flare results in a 0.050 in. wall thickness.

**Configuration.** The vehicle shape is dictated by the high drag requirements and the desire for a reasonable static stability margin to provide aerodynamic attitude control at entry. The nose was assumed hemispherical with a four-inch radius. This is followed by an eight-inch conical frustum of 10° half-angle, providing a static stability margin of 3 to 4 inches. For the vehicle weight estimated here, this shape will have $W/C_D A = 8$ psf. (The rocket motor case is discarded before entry.) The configuration is shown in Fig. 1.

The instrument package is supported on the forward side of a Fiberglas panel 0.04 in. thick and attached to the outer structure at the hemisphere-flare junction. This panel, with the payload attached, is separated from the heat shield after the acceleration is reduced below a certain value.

**Communications**

The entry vehicle may be released from the fly-by vehicle when it is seven hours away from Venus. The most significant data, however, will be gathered during the last 20 minutes or so when the entry vehicle is dropping through the lower Venusian atmosphere. Therefore, in order to minimize the battery load, the telemetry will be on only during this final descent phase.

A typical operation would proceed as follows: The entry vehicle leaves the fly-by vehicle at zero-time and proceeds towards Venus. The telemetry antenna of the fly-by vehicle has been extended. The entry vehicle is inactive electrically. The phase-locked receiver in the fly-by vehicle is sweeping the expected frequency range. At time plus 6.5 hours the entry vehicle enters the Venusian atmosphere and shortly after the heating phase of entry has ended, the instrumentation package separates from the temperature insulating shell,
Figure 1. Preliminary Capsule Configuration.
the dipole antennas are extended, and the transmitter and electronics system are energized. Transmissions will continue throughout the descent and will last until the batteries expire well after the vehicle has landed on the surface of Venus.

The basic requirement is to transmit the maximum amount of information from the entry vehicle to the fly-by vehicle with the constraint that the size and weight requirements of both vehicles be minimized. It is assumed initially that this system will operate independently of other operations of the fly-by vehicle; therefore, both a receiver and an antenna for the fly-by vehicle are included. A solid-state, phase-locked receiver is used in order to minimize the noise bandwidth and permit reception in the fly-by vehicle with a minimum demand on entry vehicle transmitter power. The data output of this receiver would either be retransmitted directly to earth or stored aboard the fly-by vehicle for later transmission, depending upon the particular communications system employed by the fly-by vehicle.

Transmitter and Batteries

In this study the capsule weight was considered the fixed parameter. However, about half of the weight of the communications system is that of the transmitter and batteries. It was determined that a 450-mw solid-state transmitter and a battery pack sufficient to power both the transmitter and the rest of the system could be designed within the volume and weight limitations of the capsule. Therefore, one parameter in the system was established: 450 mw of transmitter power at 100 mc.

Bandwidth and Modulation Considerations

With the transmitter power specified and knowing the maximum range required, the receiver noise bandwidth may be determined after establishing reasonable values for the receiver noise figure, antenna gains and the signal-to-noise ratio desired at the receiver. This latter value was set at 8 db for a phase-locked receiver.

Bandwidth is determined primarily by the quantity and accuracy of data desired, transmitter and receiver stability, and doppler shift due to relative motion between the capsule and the fly-by vehicle. The maximum doppler shift is expected in the event of a minimum miss fly-by trajectory where the vehicle passes Venus at a range of 10,000 km. It may be shown that this results in a rate of change of doppler frequency of about 8 cps/sec, assuming a transmitter frequency of 100 mc. The loop modulation bandwidth may be considered as
about 15 per cent of the noise bandwidth (5). Since the transmitter power and other system parameters permit a loop noise bandwidth of 60 cps, the loop modulation bandwidth is about 9 cps. Under these conditions carrier phase modulation is permitted wherein the carrier phase is deviated linearly up to \( \pi \) radians in about 0.044 seconds (5).

Modulation must be such that simple circuitry is permitted and bandwidth is minimized. These conditions may be met by commutating 11 channels at a rate of about one channel per second and using pulse position modulation. Two signals will be transmitted for each channel; the first indicating the start of each channel and the second indicating the end of the channel. The value of the telemetered signal is directly proportional to the time between the two signals. Each of the time signals phase modulates the carrier, resulting in a PPM-PM system, (pulse position modulation, phase modulation). This is indicated in Fig. 2 where the upper diagram shows the phase modulation of the carrier and the lower diagram shows the equivalent frequency modulation of the carrier. The pulse rise time \( \Delta \) is about 0.044 seconds. Pulse "a" represents the reference pulse or the start of each channel. Pulse "b" represents the midscale value of the channel data, and pulses "c" and "d" represent the minimum and maximum data extremes, respectively. The distance measured in time between the extreme pulse positions "c" and "d" (2 \( t_o \)) is about 1 second. Since time must be allowed for pulse widths and channel separation, each channel will require 1.35 seconds and a frame of 11 channels is transmitted in about 15 seconds.

The signal-to-noise improvement in a PPM system is a function of both the maximum deviation in time, \( t_o \), and the rise time, \( \Delta \), of the pulses. It may be shown (6) that the improvement in signal-to-noise ratio which results under the conditions specified is about 18 db. Since the signal-to-noise ratio in the loop is 8 db, the receiver output signal-to-noise ratio will be effectively 26 db.

Telecommunications System Parameters

Significant parameters of the telecommunications system are summarized as follows:

- **Frequency**: 100 mc
- **Distance**: 21,900 nautical miles
- **Propagation loss**: -164.9 db
- **Receiver equivalent noise bandwidth**: 60 cps
Equivalent Frequency Modulation of Carrier

Phase Modulation of Carrier

Fig. 2. Modulation Waveforms.
Receiver noise figure 4.5 db
Sky temperature $850^\circ K$
Effective receiver temperature $1370^\circ K$
Receiver noise $-179.4 \text{ dbw}$
Receiver loop signal-to-noise ratio 8 db
Required receiver signal $-171.4 \text{ dbw}$
Receiving antenna gain +3 db (circular polarization)
Polarization loss -3 db
Miscellaneous losses 3 db
Transmitting antenna gain (linear polarization) 0 db
Transmitter power 450 mw

Telemetry Concept

A simplified block diagram of the entry vehicle telemetry system is given in Fig. 3. Eleven channels are commutated by a two-pole stepping switch. Two of the channels, such as channels 1 and 2, are for voltage reference (0 and +4 volts) so that the system is calibrated during each frame, (eleven channels constitute a frame). These two channels also serve as synchronizing channels since their value are known. The lower pole of the commutator provides power to the transducer being sampled and to the chopper amplifier if required. Therefore, power is dissipated only in the specific channel of interest at any given time. In addition to conserving power, this prevents heating of sensitive transducer elements which could degrade accuracy.

Although all transducer outputs will be voltage levels, some will be in the millivolt region, while others will be on the order of volts. A chopper amplifier is provided to amplify the low-level signals to a maximum value of 5 volts.

The integrator generates a linear voltage ramp which is compared with the voltage of the channel being sampled by
Figure 3. Telemetry Block Diagram.
the commutator. The comparison time begins at the start of the ramp and a pulse phase modulates the transmitter. When the comparator indicates that the ramp voltage from the integrator has equalled the sampled channel voltage, another pulse is generated which again phase modulates the transmitter. The time between these two transmitted pulses is proportional to the amplitude of the signal being sampled by the commutator.

The integrator also provides an appropriately timed pulse for stepping the commutator between measurements.

Transmitter

The transmitter must provide a stable carrier frequency with good efficiency and must be rugged enough to withstand the entry deceleration of up to 450 g. These requirements may be satisfied by a solid-state unit incorporating a crystal oscillator.

A schematic of the transmitter is given in Fig. 4. The crystal oscillator is phase-modulated. The output of the crystal oscillator section is a low-level signal at a frequency of about 6.67 mc. This is amplified in the buffer amplifier and multiplied by 5 and then by 3 in two varactor multiplier sections. The 100-mc signal is then amplified to a level of about 450 mw in two power amplifiers and is radiated. Although a transmitter could be built which would deliver this power without the use of varactors, it appears at this time that varactors result in a better over-all efficiency. The over-all efficiency of this unit is about 30 percent so that 1.35 watt of power is required to deliver 450 mw of power.

Commutator

A mechanical commutator will provide better than a two-to-one advantage in weight, power, and size over an equivalent solid-state unit for this application. The commutator operating time will be less than one hour, and the switching rates will be very low (about one per second). The unit would be hermetically sealed to provide adequate lubrication of the moving parts, and since there are no rapidly moving parts or hard wear requirements, high reliability can be expected. Therefore a small, two pole, eleven channel, mechanical stepping switch commutator appears most desirable. Switching will be solenoid-actuated, thus dissipating power only during the actual switching period.
Figure 4. Solid-State Transmitter.
Integrator and Comparator

The integrator and comparator are conventional circuits and will not be discussed in detail (7). In generating the one-second voltage ramp in the integrator, only nominal linearity is required since the ramp circuit will be well calibrated on the ground. At the end of each ramp period, the integrator section discharges a capacitor through the solenoid of the stepping switch commutator, thus switching channels.

Chopper Amplifier

Some of the transducers will be used to develop signal voltages of only a few millivolts. The comparator, however, requires voltages on the order of 5 volts or less for satisfactory operation. Therefore, an amplifier is required for the low-level channels, and a chopper is used to avoid the inherent drifts of d-c coupled amplifiers. Again, these circuits are conventional and will not be detailed here.

Transmitter Antenna

The capsule antenna will consist of a center-fed half-wave dipole. Each arm will be about 30 inches long and will be made of tapered spring steel. Prior to entry, the antennas are fitted inside the heat shield. The antennas extend themselves after the heating phase terminates and the instrumentation package separates from the heat shield.

In order that the antennas remain extended without bending excessively when the vehicle is falling through the Venusian atmosphere, it may be shown that the following relation must be satisfied.

\[
\frac{M_v}{A_v} = \frac{M_a}{A_a}
\]

where

- \(M_v\) = vehicle mass
- \(A_c\) = projected area of the vehicle
- \(M_a\) = antenna mass
- \(A_a\) = projected area of the antenna

Since \(A_v\) may be varied almost independently of \(M_v\) over a nominal minimum value, this relation can be satisfied by the vehicle for this project.
Consideration will be given to adding a parasite antenna in addition to the driven antenna in order to obtain some gain, resulting in a simple Yagi array. Since the bottom of the entry vehicle is always oriented toward the surface of Venus, transmitter power may be conserved by radiating only into the upper hemisphere, thereby achieving about 3 db gain. A suitable cardioid pattern is obtained by placing the parasitic element cut to less than a half wavelength approximately one foot in front of the driven element.

Receiver on the Fly-by Vehicle

The receiver will be solid-state, incorporating a phase-locked loop discriminator and will have a noise figure of 4.5 db. Transistors such as the 2N1141 (Texas Instruments) are currently available with a noise figure of 4.5 db at 100 mc, and it appears that even lower noise figures will be available in the near future.

A block diagram of the receiver is given in Fig. 5. The R-F signal at 100 mc is mixed with a 90-mc signal generated by a crystal oscillator whose stability is at least as good as that of the transmitter. The difference frequency is amplified by the I-F amplifier stages, is limited, and enters a phase-locked demodulator. An optimized lag network is used in the low-pass filter. The crystal stabilized VCO operates at 10 mc with a frequency range more than adequate to track out doppler, transmitter, and receiver frequency drifts. By using this relatively high VCO frequency, a second mixer stage is eliminated.

In order to pick up the transmitter signal when it first begins to radiate, and in order that the receiver may recapture the signal when it loses lock, a search sweep circuit is provided. This circuit generates a ramp voltage which sweeps the VCO frequency until lock is established. The output of the VCO is phase-shifted 90° and compared with the I-F signal in the correlation circuit. Prior to lock-in, the output of the correlation circuit is zero, the sweep circuit is energized and passes directly into the VCO. When the transmitter signal is detected, the output signal of the correlation circuit rises, cuts off the sweep circuit, and the VCO frequency is again controlled by the output of the low-pass filter.

The demodulated signal out of the low-pass filter is similar to that shown in Fig. 2. The signal is processed in a decision circuit which demands that both the derivative and the one complete cycle characteristic of the signal be within required limits before it recognizes the signal as the true one. The output of this circuit is a pulse with a width equal to the time spacing between the two pulses which modulated the transmitter. This output signal may then be converted in the signal
Figure 5. Receiver for Fly-By Vehicle.
processor to a voltage level, binary code, subcarrier frequency, or whatever form is required by the fly-by vehicle for either direct transmission to earth or storage.

Receiver Antenna

It is assumed that the entry vehicle carries a linearly polarized antenna. Since the aspect of the entry vehicle with respect to the fly-by vehicle may vary over a wide range, it is evident that a circularly polarized antenna should be used on the fly-by vehicle. The fly-by vehicle will be attitude stabilized and the relative direction of Venus will be known. This suggests that a directive antenna might be used in order to obtain a few decibels of gain. The antenna, however, must be simple and lightweight.

The requirements for circular polarization can be met as indicated in Fig. 6 by using a turnstile antenna, while gain may be achieved by adding a parasitic turnstile, forming Yagi arrays with about 3 db gain over a wide angle. Higher gain is possible if more elements are added. An antenna beamwidth of about 90° is required to assure coverage under the worst trajectory conditions, (assuming a minimum close approach of 10,000 km).

Environmental Considerations

From the time the entry vehicle leaves the fly-by vehicle until it enters the atmosphere of the planet, it will be electrically inert. During this period of approximately six hours, the temperature of the vehicle must be controlled so that the batteries and electronic equipment are within their operating ranges when the vehicle enters the final (descent) phase. The absorptivity of sunlight energy will be optimized by suitable surface coloring of the vehicle to control the temperature. The vehicle should rotate in order to equalize the temperature. If a greater degree of temperature control is required, rotating color vanes controlled by temperature-sensitive bimetallic springs may be used in a manner similar to the Able lunar satellite system.

During the brief deceleration phase, both temperature and deceleration will be extreme. The instrumentation must be sufficiently well insulated from the anticipated 2500°F temperature outside of the heat shield so that a serious internal temperature rise does not occur.

The heating phase may last from 5 sec for a normal entry to 30 sec for a low angle entry. The deceleration will rise linearly for a normal entry to a peak value of about 450 g
Figure 6. Turnstile-Yagi Array for Fly-By Vehicle.

- Driven Elements
  \[ \frac{\lambda}{2} \]

- Parasitic Elements
  \[ \left( \frac{\lambda}{2} \text{ each} \right) \]

- \( \alpha, \lambda \)
in a period of about 1.5 sec, and then decrease in a similar manner. All components must be packaged to withstand these stresses.

At the end of the entry phase, when aerodynamic heating has subsided, the instrumentation package must be separated from the heat shield. At this time the antennas are extended, the electronics system is activated, and telemetry transmissions begin. The rate of descent and the stability of the vehicle may be controlled by using a suitable drag device. The vehicle should continue to rotate slowly during the descent so that any radiation pattern nulls will not be continually directed toward the receiving antenna on the fly-by vehicle.

The atmospheric temperature of Venus may be as high as 300°C. Therefore, the instrumentation package itself must be insulated to withstand an exposure in this environment for at least 20 minutes. Temperature sensitive elements such as the solar cells must be protected. The antenna should be transformer-coupled to the transmitter so that conducted heat does not destroy the final power stage.

The receiver will be required to operate for about one hour. During this period, it requires a suitable transistor operational environment.

Although the antenna is required only during the short transmission period of the entry vehicle, it will probably be most convenient to extend it at the same time that the main earth communications antenna is extended so that no disturbances are introduced as the fly-by vehicle nears the planet.

The fly-by vehicle should be controlled so that it passes within the limits of 40,000 km and 10,000 km of the surface of the planet. At ranges much greater than 40,000 km, the transmitted signals are attenuated to an undesirable extent. At ranges considerably less than 10,000 km, the fly-by vehicle may disappear below the horizon of the entry vehicle and thus prematurely terminate communications.

Summary of Fly-By Vehicle Component Power and Weight

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<th>Item</th>
<th>Average Power (milliwatts)</th>
<th>Weight (ounces)</th>
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<tr>
<td>Receiver</td>
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<td>16</td>
</tr>
<tr>
<td>Antenna (2 turnstile, Yagi)</td>
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<td>8</td>
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### Summary of Entry Vehicle Component Power and Weight

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<th>Item</th>
<th>Average Power (milliwatts)</th>
<th>Weight (ounces)</th>
</tr>
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<tbody>
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<td>Transmitter (300 mw at 100 mc)</td>
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<td>Integrator and comparator</td>
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<tr>
<td><strong>Totals</strong></td>
<td><strong>1500</strong></td>
<td><strong>16.7</strong></td>
</tr>
</tbody>
</table>

### Conclusion

A significant amount of data may be gathered from a very small entry vehicle launched from a fly-by vehicle. All techniques required for the system are well within the current state of the art.

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References


