COMMUNICATIONS FROM A MARS ENTRY PROBE*

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Summary A low risk experiment to determine the principal properties (density, pressure, temperature) of the Maritian atmosphere has been studied. A. slender conical capsule is ejected from the Mars fly-by or orbiting spacecraft and enters the planet's atmosphere. Theoretical and experimental data support the concept of continuous communication between the sharp cone capsule entering the Martian atmosphere and the spacecraft. The communications system basically consists of a 25 watt transmitter phase modulating a 100 mc carrier on the entry capsule and a wideband receiver on the spacecraft. Transmitting power and spacecraft data storage considerations resulted in a data transmission rate of 147 bits per second in a pulse code modulated format.

Introduction In order to effectively plan a manned mission to Mars, a great deal of information on the properties of the Martian atmosphere and surface must be available. The basic design of a Mars landing vehicle depends on knowledge of the atmospheric properties. The present state of ignorance of the Martian atmosphere precludes the development of a highly sophisticated, costly, unmanned lander as the risk element is too great. Thus, a low cost, low risk, unmanned atmospheric penetration vehicle, which is not expected to survive surface impact, appears to be a logical first step in the exploration of Mars. In addition, sterilization requirements for the conical capsule seem to be much less than that for a blunt body.

As we know little about the Martian atmosphere, the atmospheric penetration vehicle should be able to operate in a wide spectrum of aerodynamic and thermodynamic environments. A configuration with a very high ballistic coefficient can be designed to be virtually independent of atmospheric properties. The sharp conical vehicle meets this objective and has a form that is relatively easy to treat analytically. Since the sharp conical capsule will not be appreciably slowed down by the atmosphere, good altitude versus time correlation can be obtained. In addition, a very compact vehicle can be realized, minimizing space and weight requirements on the spacecraft.

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Figure 1 depicts the events that occur during the spacecraft's encounter with Mars. The entry capsule is in its sterilization container aboard the spacecraft. A. few hours prior to planetary encounter, after the spacecraft/Mars encounter geometry has been determined on Earth, the capsule launch angles from the spacecraft are checked and corrected. Subsequently, all capsule subsystems are monitored by internal and spacecraft telemetry for launch readiness. After capsule checkout, and at a predetermined time before planetary encounter, the capsule is ejected from the spacecraft and starts spinning. When there is sufficient distance between the capsule and spacecraft, the capsule injection motor is ignited to provide a fixed velocity increment for atmospheric entry. The capsule then has a course which leads the spacecraft trajectory in order to assure uninterrupted line-of-sight communications from the capsule to the spacecraft.

When a drag deceleration of 0.01 earth g is sensed by the capsule accelerometer, it is despun and the attitude control system, sensors, telemeter and transmitter are activated. The attitude control system reduces the capsule angle of attack to a negligible value. All capsule data is transmitted in real time and is received and stored in the spacecraft. The stored data on the spacecraft is transmitted to earth upon receipt of a radio command.

Capsule Configuration The inboard profile of the capsule and container conceived for entry into the Martian atmosphere is shown in Figure 2. The basic capsule configuration is a cone flare body with a ballistic coefficient of 500 lb/ft^2 made of ATJ graphite insulated with microballoon loaded phenolic/absestos. The graphite has good electrical characteristics even at high ablating temperatures (3500°F). The heat shield is comprised of two sections which are separated by a quartz insulating ring. The two sections then form two elements of a dipole antenna. The anticipated thermal and aerodynamic stresses to be experienced by the heat shield during re-entry are well below the allowable stresses for graphite, leaving a considerable margin for unexpected effects. The temperature rise on the back of the phenolic/asbestos insulator is expected to be less than 100°F from atmospheric encounter to surface impact.

The interior of the capsule contains the attitude control system, the sensors, the power supply, the sequencer, the telemetry and signal conditioning equipment, and the transmitter. The despin system and injection motor are attached to the rear of the capsule and are jetisoned prior to atmospheric entry for aerodynamic reasons. At entry, the capsule weighs approximately 18 pounds. The rocket motor and despin mechanism weigh about six and a quarter pounds. The launch and ejection system and electronics on the spacecraft will weigh less than twenty-eight pounds. Thus, the total Mars atmospheric entry experiment weight would be about fifty-two pounds.

Measurements and Data Analyses Selection of individual measurements for the Mars entry experiment involved many factors, including usefulness of acquired data, minimization of equipment, reasonable information transmission rates, minimization of data storage requirements on the spacecraft, system sequencing, capsule time in the atmosphere and failure analysis capability. As a result, a small number of relatively uncomplicated measurements were selected to obtain the desired atmospheric data. More sophisticated measurements, such as stagnation pressure and temperature, are of interest, but are difficult to implement and to obtain accurate data. Atmospheric composition measuring instruments are either too large or too slow for the experiment. In addition, feasibility of performing a life detection experiment with the capsule was investigated. All of the life detection experiments must be postponed until the landing survival of a vehicle can be assured. Landing survival requires knowledge of the atmosphere which can be obtained with the sharp body capsule design.

The atmospheric parameters are measured indirectly by recording the axial deceleration, cone pressure, and surface temperature. Three accelerometers are used having maximum ranges of 1, 5 and 60 earth g. The I g maximum range accelerometer, oriented normal to the vehicle axis, is used during the injection phase and for activating the despin and Attitude control system (at 0.01 g) and the data transmission link to the spacecraft (at 0.05 g). The 5 and 60 g range accelerometers are oriented along the vehicle axis with the lower level deceleration being sensed down through the region of peak loading for a low ballistic coefficient capsule and automatically switched to the higher level accelerometer before the first saturates. Thus, a continuous deceleration history is made available so that velocity and altitude can be derived by integration from the time of impact. The entry angle will be determined from analysis of the closest fly-by measurements made by the spacecraft. Atmospheric density (P) at any point along the trajectory can be determined from knowledge of the vehicle velocity (V), axial deceleration (V_A), angle of attack (α) and drag coefficient (C_D). The first three of these factors are measured to a high degree of accuracy and the last is estimated. A second method for determining density is based on cone pressure (P_C) measurement and is dependent on a reasonable knowledge of pressure coefficient (C_p), gas constant (R), specific heat ratio (γ) and free-stream temperature (T_∞).

The method that will be used to determine the surface pressure (P_O) involves integration of derived velocity and density data, making use 6f the measured impact time (t), Mars gravity (g_M) and entry angle (θ_E). The parameter accuracy resulting from derivations of g_M and θ_E by spacecraft closest fly-by measurements and the accuracy of the derived density and velocity data lead to reasonably low errors in this evaluation of the free-stream pressure. The use of a high ballistic coefficient vehicle improves the evaluation accuracy since deviation from a straight-line trajectory is relatively small. This is particularly true for the nearly vertical descents which are expected to be within ±20° of vertical.

A similar integration technique can be used for calculating the altitude as a function of time by back-integrating capsule velocity (V) from the time of impact.

It is possible to obtain an approximate free-stream temperature profile from the derived density and pressure data using the perfect gas relationship and the gas constant estimated from heating characteristics. The heat flux will be measured by a calorimeter (although it serves mainly to give a total heating) and can be correlated with pa_Vb type trends now known reasonably well for both convective and radiative heating.

An analysis of the errors in parameter values from capsule data has been performed. The error budgets attributed to the measurements and estimates are listed in Table 1. For the postulated atmosphere resulting in the largest error, density, pressure and temperature errors are conservatively determined to be 5%, 5% and 10%, respectively. Thus, the important atmospheric properties can be obtained to a relatively high degree of accuracy utilizing the data from sharp cone capsule.

Data Requirements Data transmission rates and data storage limitations on the spacecraft (for the relay link) serve to limit the quantity of collected information. A.s a result of a series of trade-offs between transmitter power output, measurement sampling rates, data storage unit size, and minimization of system complexity, a data transmission rate of 147 bits per second was selected as shown in Table 2. There will be four measurement channels having a one sample per second rate, six channels sampled at a two per second rate, and one channel with a sampling rate of four per second. In addition, the events generated by the capsule sequence will be monitored and seven bits per second will be required for data frame synchronization. The first three items listed in Table 2 are primarily engineering measurements. However, since these measurements are also very useful during entry, and have low sampling rates, it was decided to switch all the measurements on and off during each active period. Therefore, no channel switching is required except for substituting the high range axial accelerometer for the low range unit when the deceleration value is more than 5 g's.

The trajectory analyses produced information on expected periods in the atmosphere for various atmospheric models and entry angles. These times are:

"Turn On" Deceleration Value	Very Longest Time of Entry	Very Shortest Time of Entry
0. 05 g	43 sec.	7 sec.
0. 01 g	53 sec.	8.5 sec.

Therefore, a maximum entry transmission time of 60 seconds was assumed. Additional transmission periods would occur during pre-separation checkout (3 seconds) and capsule injection (15 seconds). Since there will be inactive periods of considerable length between each of these transmission times, the communication system will be turned off between the active periods to conserve power and data storage space. As the total transmission time is about 80 seconds, the storage unit on the spacecraft will have to store a maximum of 12,000 bits of data. In the event the spacecraft is able to transmit the data obtained during the preseparation checkout, separation and injection periods before the capsule starts atmosphere entry, as is likely, then the size of the data memory can be reduced by one-third.

Because a variety of sensors will be utilized on the capsule, each sensor will incorporate the necessary signal conditioning equipment to ensure signal compatibility with the communications system. This approach eliminates the need for low signal level commutation and provides a clean interface between measurements and communications systems.

Communications Analysis A preliminary analysis was performed on a direct capsule to earth transmission link. Using the PCM/PSK/PM scheme and with negligible capsule antenna gain, the transmitter power required to meet the data requirements of 147 bits per second approximated 3000 watts. A transmitter of this size cannot be accommodated by the capsule under any circumstance.

Attention was then directed to the capsule to spacecraft to earth transmission link. One possible method of retransmission is to repeat the received signal without detection and remodulation. The other possibility is demodulation of the received signal, detection and reconstruction of the data train, and reformatting the data for modulation and transmission by the spacecraft communications system. This method was chosen inasmuch as additional spacecraft and ground station equipment is minimized. The data will be received at a relatively high rate, demodulated and stored, and retransmitted by the spacecraft communications system at a conveniently low data rate. Because of the very short transmission time at capsule entry, signal acquisition by a receiver synchronous detector would be unlikely without significant data loss. Therefore, the receiver predetection bandwidth is made large enough to accommodate crystal frequency drift, doppler shift, and data bandwidth. The detector is a simple, wideband frequency discriminator. Although this method requires considerably more power than a coherent detector, the simplicity and reliabilitylof the system outweighs the power penalty.

The primary considerations in selecting a signal modulation method are measurement sampling rates and desired accuracy. Other important factors are ease of data handling and reduction, available mechanizations, and relative communications efficiency, Since the measurement sampling rates are relatively low and respectable accuracy is desired, the data will be pulse code modulated (PCM) to generate a digital signal. The PCM data from the capsule are compatible with the spacecraft data format. A word length of seven bits per sample was chosen to yield a better than 1% measurement accuracy and is the same as that utilized by the spacecraft communications system. Since there will be no attempt to track the capsule, either from earth or the spacecraft, the carrier can be completely suppressed and no subcarriers are needed. Thus, approximately 3 db is gained in signal power by eliminating subcarriers. Phase modulation was chosen since the mechanization of transmitter and receiver is somewhat simpler than the FM counterpart. A study of carrier frequencies from 100 to 1000 mc was performed to determine the optimum frequency. Although antenna efficiency increases somewhat as the frequency increases and the plasma attenuation is slightly less at higher frequencies, the free space loss and doppler shift increase at a much greater rate at higher frequencies (as compared to 100 mc) and overwhelm the effects of the first two phenomena. The above considerations have led to the selection of a carrier frequency of 100 mc and a modulation choice of PCM-PM.

Since there is no subcarrier for PCM synchronization, the sync pulses must be incorporated in the data train. Frame sync is a unique seven bit data word (Barker code) and bit sync is generated from transitions in the data train. A return-to-zero (RZ) code was chosen to simplify bit sync detection. Although twice the data bandwidth is required to RZ as compared to non-return-to-zero coding, the noise bandwidth required to accommodate receiver oscillator drift and doppler shift is large enough so that less than 0.5 db loss is produced by doubling the data bandwidth. The amount of RF bandwidth available is not a limiting factor in the design.

As the capsule penetrates the Martian atmosphere, it can heat the gases sufficiently to generate an ionization region. Since the sharp conical entry capsule is stabilized, the angle of attack is controlled to be a very small value. Therefore, significant ionization is confined to a thin boundary layer surrounding the capsule heat shield. With a worse case atmosphere, the peak boundary layer electron density (at zero altitude) is sufficient to cause a transmission loss of 2.5 lb. This transmission loss is greatest at zero altitude and diminishes at higher altitudes. In the event of attitude control system failure, the angle of attack will increase considerably. However, it appears that the angle of attack would have to become unreasonably large at low altitudes before there are deleterious effects upon transmission from the capsule. For angles of attack less than 10° at 100,000 feet altitude, there would be no appreciable effect on capsule transmission capability.

As the atmosphere becomes ionized and conductive, the antenna tends to arc or break down. The antenna voltage breakdown problem was analyzed using the work of Scharfman and Morita[(1)]. Breakdown power versus air pressure is a U-shaped function, yielding a critical pressure (or altitude) at which the power required for breakdown is a minimum. This critical pre s sure is given by: $P = \omega / 5.3 \times 10^9$ mm Hg, where ω is the signal angular frequency. For 100 megacycle link, the critical pressure is 0.118 mm Hg. In air, the breakdown power is about 40 watts, and should be greater in the Mars atmosphere because of higher dissociation and ionization potentials. At altitudes above and below the critical altitude, more power can be radiated. Thus, the capsule transmitter design power of 25 watts is well below the value needed for antenna breakdown in the absence of a surrounding plasma. When a plasma is present, the power required for antenna breakdown may decreased by an order of magnitude. However, the capsule will be well beneath the critical breakdown altitudes before plasma formation becomes significant. Therefore, it is expected that antenna breakdown effects are not of importance in the design of the communications link.

These results are summarized in Table 3 which presents a detailed link analysis of the proposed capsule-to-spacecraft relay data link. The analysis indicates that the desired bit rate (147 bits/second) can be achieved by utilizing a standard PM receiver and a capsule transmitter power of 25 watts. The net worse case performance margin is +2.75 db.

Communications System Equipment The communication system design philosophy is based on maximum simplicity and reliability. Standard digital modulation and demodulation techniques are utilized. Since the noise bandwidth is a small fraction of the carrier bandwidth, no attempt was made to try to conserve RF bandwidth. In several instances, it was decided to increase the transmitter power requirements to obtain the most reliable equipment. Another design goal was to minimize capsule and associated equipment interfaces with the spacecraft. Capsule volume limitations will require the use of microcircuitry.

In the capsule, all sensors incorporate sufficient signal conditioning equipment to produce a 0 to 5 volt dc output. These high level analog signals are then routed to the telemetry subsystem. Figure 3 is a functional block diagram of the capsule communications system. A high level electronic commutor samples each data channel and furnishes a pulse amplitude modulated (PAM) analog signal to the analog to digital converter. The analog to digital converter digitizes each analog sample to yield seven bits for each data word. These data, which are in returnto-zero (RZ), parallel format, are combined with the sequencer event markers and frame sync words by the buffer register to obtain a serial, RZ, data train. A digital programmer, which includes a digital clockand bit and word counters, sequences the commutator, frame sync generator, event register and buffer register, and supplies clock pulses to the capsule sequencer. The serial PCM data train then enters the transmitter where it phase modulates the 100 mc rf carrier. The PCM/PM signal is amplified to a 25 watt level and then radiated via the foreshortened dipole antenna consisting of the graphite heatshield. Capsule communication equipment will weigh less than three pounds and consume 43 watts of prime electrical power. Checkout of this equipment is accomplished by having the system operate and analyzing the received data for frequency response, distortion, bit errors and overloading. Additional performance information is provided by directly monitoring the transmitter output power via the telemeter. The transmitter provides a 0 to 5 vdc signal for this purpose.

The signal transmitted from the capsule is received by a deployable planar array antenna on the spacecraft. The PCM/PM signal is fed into a wideband receiver which utilizes a simple discriminator to remove the carrier. A post detection filter increases the effective signal-to-noise ratio. The PCM data train simultaneously enters the sync detector and the sampling detector. The sync detector processes the data to obtain clock pulses and bit sync information to gate the sampling detector and the data storage unit. The sampling detector produces clean N-R-Z digital data for entry into the data storage unit. The data storage unit is a 30,000 bit magnetic core memory which is programmed by the spacecraft data processor which treats it as another storage buffer. The data format is compatible with the spacecraft data handling and telecommunications systems. Commands from the spacecraft sequencer deploy the antenna and turn the transformer-rectifier on and off. Prelaunch and preseparation checkout of the auxiliary communication system is accomplished by sequencing the equipment and analyzing the telemetry data for poor performance. Figure 4 is a functional block diagram of the auxiliary communication system.

Conclusions The analysis and predesign of a low risk, low cost experiment to explore the Martian atmosphere has been accomplished. The principal atmospheric parameters can be determined with sufficient accuracy to enable the intelligent design of Mars surface landers and manned exploration vehicles. With the sharp conical entry capsule, continuous communications can be maintained between capsule and fly-by or orbiting spacecraft. A. PCM/PM signal is transmitted from the capsule to the spacecraft where it is received by a simple wideband receiver and stored for retransmission to earth. A data rate of 147 bits per second will require about 12,000 bits of storage for the longest entry trajectory. However, storage unit size and weight increases slowly with growth in capacity. Due to the capsule con-figuration, plasma sheath formation is minimized and very little R. F. signal attenuation is expected. Since the capsule system will weigh so little, two capsules could probably be accommodated by the spacecraft. Thus experiment results and reliability can be increased by having capsule redundancy.

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References

(1) Scharfman and Morita, “Voltage Breakdown of Antennas at High Altitudes,” SRI Technical Report 69, April 1960.

TABLE 1

CAPSULE DATA ERROR ANALYSIS

Parameter	Relationship	Measurements	Error Sources	Instrumentation	Estimated or Interpreted	Errors in Result	rss Error $\theta_E = 70^o$
Density	$\rho = \frac{-2\dot{V}_A}{V^2 \frac{C_A A}{m}}$	$\dot{V}_A$ V_E α	$\dot{V}_A$ V_E α C_D V g_M	1% 3 m/sec. 1%	11% 0.2%	11% 0.5% 1.5%	11%
Density	$\rho = \frac{KP_C \gamma^{0.95}}{C_{PC} V^{1.9} (R T_\infty)^{0.05}}$	P_C $\dot{V}$ V_E	C_{Pc} Pc V_E V RT_∞ γ	1% full scale 3 m/sec. 1%	5% 55% 15%	5% 1% P_C max. 0.5% 2.8% 14%	15%
Pressure	$p = \int_0^t g_M \rho V \sin\theta dt$	$\dot{V}$ V_E	$\dot{V}$ V_E $\sin\theta$ g_M	<1% 3 m/sec.	2.2% 4%	0.3% 2.2% 4%	5%
Density	$p = \frac{2P_{ST}}{C_{P_S} V^2}$	$\dot{V}$	$\dot{V}$ $C_{P_S}(X,\gamma)$ P_{ST}	<1% 1% fs	5%	0.5% 5% 1%	5%
Temperature & Heat Flux	$\dot{Q} = \frac{K T}{t}$	T(t)	Calory Losses T	1%	10%	10%	10%
Capsule Altitude	$H(t) = \int_t^{t(IMPACT)} V \sin\theta dt$	$\dot{V}$	$\dot{V}$ V_E $\sin\theta_E$ g_M	1% 3 m/sec.	2.2%	0.2% 2.2% 0.5%	2.3%

TABLE 2

CAPSULE DATA REQUIREMENTS

Number of Channels	Description	Measurement Sampling Rate	Total Sampling Rate
1	Transmitter Output Power	1 SPS	1 SPS
1	Internal Temperature	1 SPS	1 SPS
1	Battery Voltage	1 SPS	1 SPS
2	Attitude Control System Accelerometer Output	2 SPS(each)	4 SPS
4	External Pressure	2 SPS(each)	8 SPS
2	Axial Acceleration	4 SPS(each)	4 SPS(one accelerometer active at a time)
1	External Heat (Calorimeter)	1 SPS	1 SPS
1	Sequencer Events	Pre-emptive	Pre-emptive
1	Frame Sync	7 bits/sec.	7 bits/sec.

TOTAL SAMPLES = 20 SPS + 7 Bits/Sec.

Data transmission rate = 20 SPS X 7 bits/sample + 7 bits/sec. = 147 bits/sec.

Total Number of Bits = 147 bits/sec. X 80 sec. of transmission time = 11,760 bits.

TABLE 3

TELECOMMUNICATION DESIGN CONTROL TABLE

PROJECT: Mars Entry Capsule DATE: 11/29/64
CHANNEL: Capsule to Spacecraft PAGE: 1 of 2
MODE: VHF

No.	PARAMETER	VALUE	TOLERANCE	SOURCE
1	Total Transmitter Power 25 watts	+44.0 dbm	+0.0 -0.5 db	D. Hanson
2	Transmitter Circuit Loss (1)	-3.75 db	+0.5 -0.75db	V. Harrington
3	Transmitting Antenna Gain	+1.75 db	+0.25db	V. Harrington
4	Transmitting Antenna Pointing Loss	-0.0 db	-0.25db	V. Harrington
5	Space Loss	-164.5db		
	@ 100 mc, R=40,000km			
6	Polarization Loss	-1 db	±0.1 db	V. Harrington
7	Receiving Antenna Gain	+8 db	±0.5 db	V. Harrington
8	Receiving Antenna Pointing Loss	-0.0 db	+0.25db	W. Scott
9	Receiving Circuit Loss	-0.6 db	+0.1 -0.3 db	W. Scott
10	Net Circuit Loss	-160.1db	+1.7 db -2.4 db	
11	Total Received Power	-116.1dbm	+1.7 db -2.9	
12	Receiver Noise Spectral Density(N/B)	-168.4 dbm/cps	±1.0 db	L. Turner
	T System=1040°K(2)			
13	Carrier Modulation Loss	Not Applicable		
14	Received Carrier Power	Not Applicable		
15	Carrier APC Noise BW($2B_{LO}$ =)	Not Applicable		
16-18	Carrier Performance-Tracking (one-way)	Not Applicable		
19-21	Carrier Performance-Tracking (two-way)	Not Applicable		
22-24	Carrier Performance	Not Applicable		

TABLE 3

TELECOMMUNICATION DESIGN CONTROL TABLE

PROJECT: Mars Entry Capsule
CHANNEL: Capsule to Spacecraft
MODE: VHF

DATE: 11/29/64

No.	DATA CHANNEL	VALUE	TOLERANCE	SOURCE
25	Modulation Loss	0.0		
26	Received Data Subcarrier Power (3)	-116.1 dbm	+1.7 db -2.9 db	
27	Bit Rate(1/T) Noise Bandwidth=1500 cps(4)	+31.75 db	+0.2 db	L. Turner
28	Required ST/N/B $P_e = 10^{-4}$ (5)	11.5 db	+0 -1.0 db	L. Turner
29	Threshold Subcarrier Power	-125.15dbm	+1.2 -2.2 db	
30	Performance Margin	9.05 db	+3.1 -6.3 db	Net worse case: +2.75 db
31-36	Sync Channel	Not Applicable		

COMMENTS:

(1) Includes 2.5 db plasma loss
(2) Galactic noise 700°K, Mars noise 20°K, line loss noise 40°K, 4 db receiver = 280°K
(3) Identical to carrier received power since no subcarriers are used.
(4) Noise bandwidth: 300 cps, information BW (RZ modulation); 800 cps, worse case crystal drift; 400 cps, worse case doppler shift.

$$\text{Doppler Shift} = \frac{\Delta rf}{C} = \frac{1.2 \text{ km/sec } (100 \text{ mc})}{3 \times 10^{6} \text{ km/sec}} = 400 \text{ cps},$$

For a worse case radial velocity of 1.2 km/sec.

(5) From standard curves for discriminator detection of FCM/FM (in this case, values will be the same for PCM/PM).

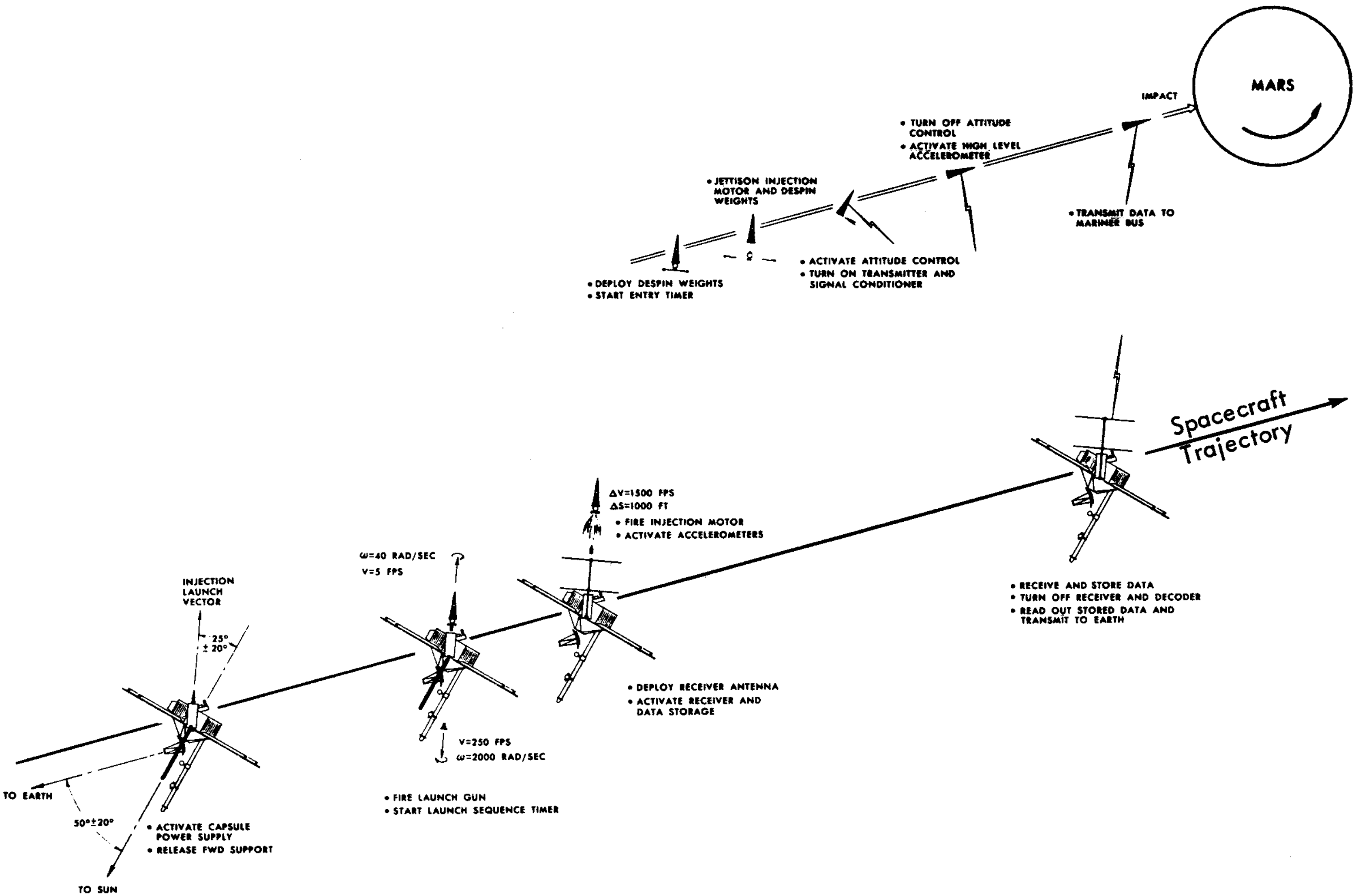

Fig. 1 -Mars Entry Capsule Flight Sequence

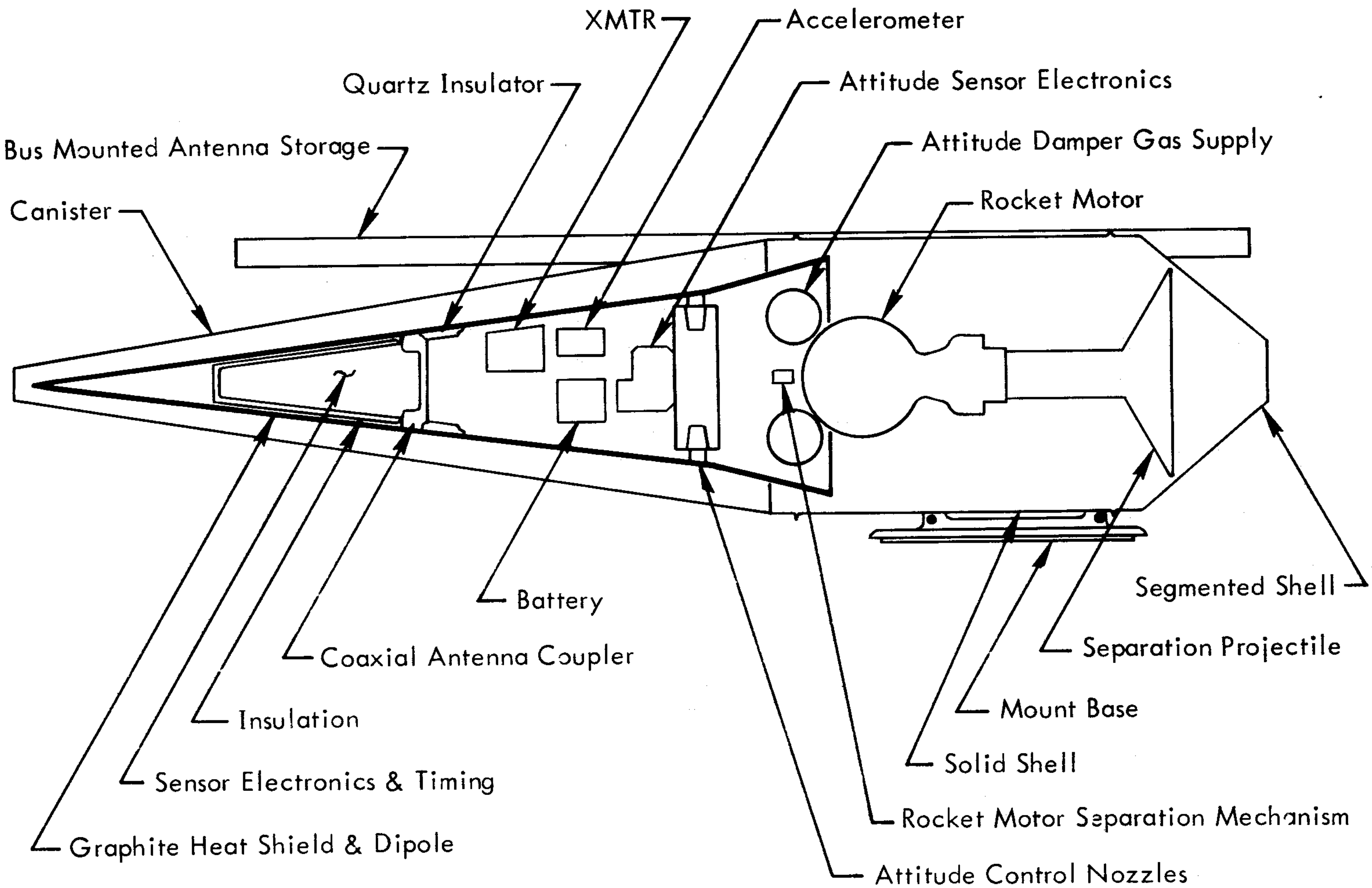

Fig. 2-Mars Atmospheric Capsule

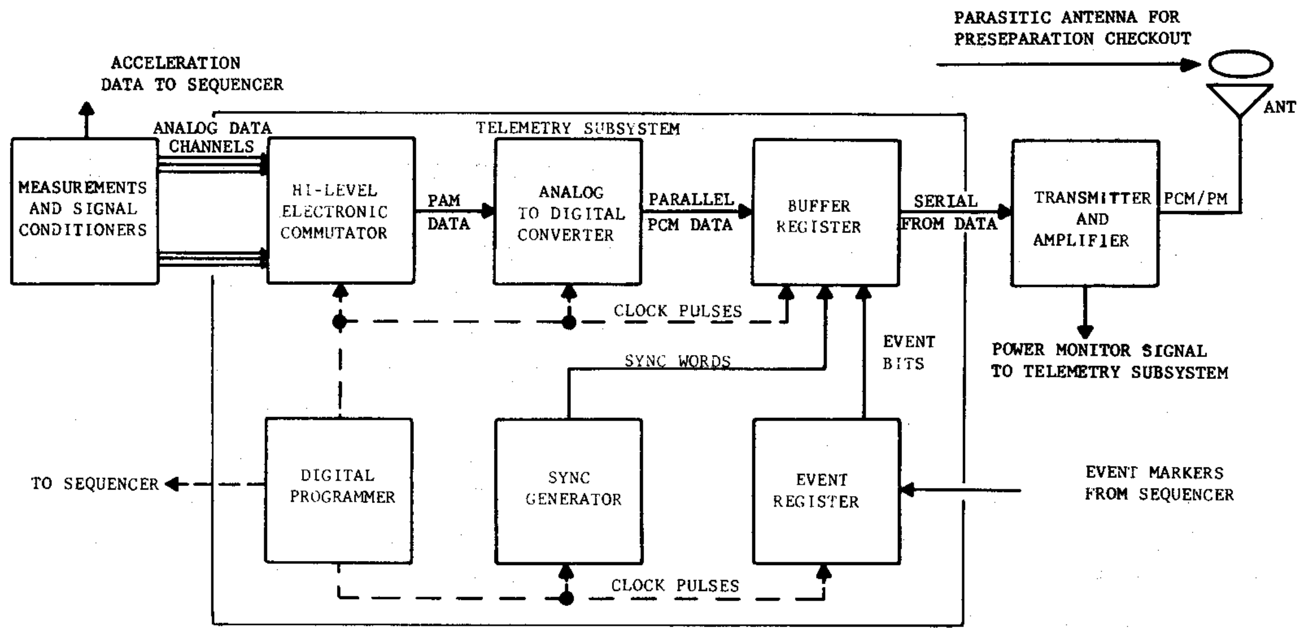

Fig. 3-Capsule Communications System

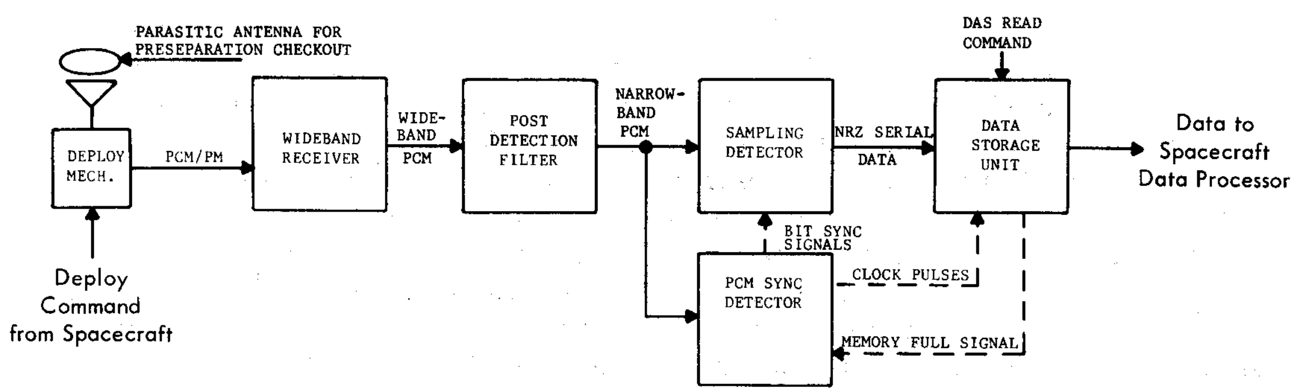

Fig. 4-Spacecraft Auxillary Communication System