Versatility of Electrically Propelled Spacecraft for Planetary Missions

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A conceptual solar electric propulsion spacecraft, called Advanced Reconnaissance Electric Planetary Spacecraft (AREPS), having 20 kw of electric power at the target planet can perform repeated coverage of the Mars or the Venus surface with good resolution. An essential concept is that capability for continuous thrust can be exploited to spiral about a target planet, successively occupying consecutive layers of planetary structure for effective synoptic science. At Mars, AREPS is captured in a circular orbit of 10,000 km altitude. It then descends to 820 km through a set of discrete intermediate orbits that allow essentially total photographic coverage at each altitude. Resolutions of 50–60 m with a medium-resolution camera system and 10 m with a high-resolution camera system are obtained. Scientific experiments for studying properties of planetary environment are also described. At Venus, AREPS descends from a capture altitude of 3900 to 1000 km for a complete mapping of the surface of Venus at a resolution of 100 m, using a side-looking radar system. A lander is dispatched to measure atmospheric and surface properties. Compared to a spacecraft that uses only chemical propulsion, AREPS provides an eightfold advantage in scientific payload. In planetary orbit, AREPS has one-third or less of the power that could be obtained from a nuclear-electric spacecraft of the same mass.

Introduction

EACH progressive step in space exploration requires that two essential factors reach their "critical point": the desire for more knowledge in a specific field and the availability of appropriate technologies to acquire this knowledge. It seems that these conditions are presently fulfilled almost ideally for a major advance in planetary exploration. In fact, it appears that planetary orbiters for high-resolution, large-area photoimaging of the surfaces of planets, particularly of Mars and Venus, are well within technological reach for the mid-seventies. Large-area photoreconnaissance of the surfaces of Mars and Venus is extremely desirable not only because of its intrinsic value as a decisive step toward planetary exploration, but particularly in connection with planetary lander projects that will furnish detailed information on the features of the landing sites and of their immediate vicinity. The system to be described here will provide surface images of Mars and Venus comparable to those that the lunar orbiters provided of the moon, with coverage of the total planetary surface once every few weeks throughout the duration of the mission. Photography will be used on Mars orbiters. Side-looking radar must be applied to Venus orbiters because the surface of Venus cannot be imaged by optical systems.

A solar-photovoltaic power source will be used for three tasks: 1) to energize an electric propulsion system that accelerates the spacecraft from near-earth toward the planet and decelerates the spacecraft upon approaching the planet, 2) to provide electric propulsion for orbit control during the entire time of planetary exploration, and 3) to provide 20 kw of power at the planet for high-bit-rate data transmission to Earth. Solar cell arrays are now developed to a high degree of reliability and efficiency, and in these respects they are superior to any other power-producing system applicable to a planetary reconnaissance spacecraft in the mid-seventies. When nuclear-electric systems become available, spacecraft of the type described here can be equipped with them.

Electric thrusters also have reached a high degree of technical perfection. Some have been tested continuously for around 10,000 hr, and lifetimes several times longer can be predicted. Efficiencies are so high that a further increase would not substantially alter the performance of a propulsion system, except at low specific impulse. Systems with precisely defined performance data and mission parameters can be proposed with confidence.[1,2] AREPS is an example of such a system.

The AREPS concept described herein utilizes existing components, so that total development, checkout, and testing time for the system will be 4–5 years.

Summary of Planetary Probe Capabilities

Results of Chemical Probes

During this decade 18 probes[3] have yielded considerable data concerning Mars, Venus, and the moon, and yet relatively little is known about any planets other than Earth. All planetary probes have been flybys, except Venus 4 and the Russian probe, Venera, which were landers. Both the landers and the flybys had extremely limited operating time and transmitting capabilities for returning useful information. Table 1 lists these limitations.

Mariner 4 provided photographs of a small portion of the Martian surface with a resolution of $\sim$1 km, revealing a

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Table 1 Summary of planetary probes

Probe	Target	Launch date	Arrival date	Mass of science package/total mass, kg	Peak power at planet, w	Maximum bit rate at planet, bits/sec	Period of intensive study of target planet
Mariner II, U.S.	Venus	8/26/1962	12/14/1962	22.7/203	220	$8\frac{1}{3}$	10 hr
Mariner IV, U.S.	Mars	11/28/1964	7/14/1965	27.3/262	310	$8\frac{1}{3}$	25 min
Venus 3, U.S.S.R.	Venus	11/16/1965	3/1/1966	[a]/964	300–400[b]	5–15[b]	Successful impact, but no data
Venus 4, U.S.S.R.	Venus	6/12/1967	10/18/1967	[a]/1110	300–400[b]	5–15[b]	94 min
Mariner 5, U.S.	Venus	6/14/1967	10/19/1967	22.7/246	250	$8\frac{1}{3}$	2 hr
Voyager Orbiter	Orbiting Mars	1970's	...	227–318/2,860	1,000	$8–25 \times 10^4$	6 months
Pioneer F, G	Jupiter	ca 1972	...	22.7/182–227	80	500	4–8 hr
Mariner 6	Mars	2/24/1969	7/31/1969	45.5/363.6	350	100	5 hr
Mariner 7	Mars	3/27/1969	8/8/1969				
Mariner 71	Orbiting Mars	...	11/1971	56.8/>546	480	1.62×10^4	>3 months
AREPS[d]	Orbiting Mars	11/8/1977	6/18/1978	272/2,200	19,400	1.2×10^6	1–2 yr
AREPS[d]	Orbiting Venus	12/17/1976	6/19/1977	581/1,990	19,920	1.92 MHz[c]	1–2 yr

[a] Not published.
[b] Estimate.
[c] Bandwidth of side-looking radar.
[d] Proposed, not officially planned.

cratered surface similar to that of the moon. It was also observed that although Mars has an ionosphere, it probably has no radiation belts and a weak magnetic field, if any at all. The observed atmosphere was considerably less dense than expected. However, most of these observations were obtained in one geometry and over a short period of time. For example, the well-known occultation experiment, which implies Martian surface pressures of about 6 mb, represented the atmosphere only at 50° south, 177° east, at a little after local noon, and at a solar zenith angle of ~67°. The surface pressure will probably vary according to time and location. Because of the constraints on encounter time and location, the photographs failed to resolve the question of the existence of Martian "canals."

Other questions that remain unanswered about Mars are the composition of the atmosphere, its thermal structure,[4] the nature of pole-to-pole transport of water vapor, and the composition and structure of the ionosphere. There is also an uncertainty of ~3% in the visual measurements of the oblateness of Mars. Removing this uncertainty may eliminate a discrepancy between the visual measurements and the oblateness values determined by analysis of the orbit of Phobos, the inner moon of Mars. Reconciling the oblateness values would lead to the development of a model of the planet's interior structure, treating questions of solid body structure and composition, and of internal heating by radioactivity. Other observations of Phobos that give information on its age and radioactivity may indicate the history and structure of Mars itself.

The Russian Venus 4, which probably stopped functioning about 26 km above the surface, and the U.S. Mariner 5 probes, produced values of surface and atmospheric conditions which are controversial. Current interpretations[5] of Venus atmospheric surface pressure range from 20 to 60 atm, a composition of approximately 95% CO_2, and a surface temperature of approximately 700°K. Components of the atmosphere other than CO_2 remain indefinite. Venus appears to have little or no magnetic field, and there is no evidence of energetic particle radiation belts.

The acquisition of more meaningful data on Mars and Venus will require observations from many locations at different times. Larger scientific payloads and data-return rates will be needed.

Table 2 Comparison of solar-electric and chemical propulsion

	AREPS	All-chemical
Mass in Mars orbit,[a] kg	2200	850
Science payload, kg	272	85[c]
Available power in addition to science payload, kw	19.4	0.450[d]
Mass in Venus orbit,[b] kg	1990	640
Science payload, kg	573	64[c]
Available power in addition to science payload, kw	19.9	0.8[d]

[a] Capture in 10^4 km altitude circular orbit.
[b] Capture in 4.1×10^3 altitude circular orbit.
[c] Assuming 10% science payload fraction.
[d] Estimated.

Capability of Electric Propulsion Spacecraft

Table 2 shows how AREPS compares with a spacecraft having only chemical thrust.[6] Besides the greatly enhanced scientific payload, AREPS has a large power supply for communication and instrumentation, and a propulsion system that can maneuver the spacecraft with vernier precision into a series of scientifically rewarding planetary orbits. The data-return capability of AREPS is in the order of 10^4 bits/sec. A bit budget for an ideal Martian mission at orbital altitudes of about 500 km with a 2-m focal length, high-resolution optical system is shown in Table 3. Figure 1 shows the number of bits N necessary for any desired percentage coverage P of the Martian surface at selected resolutions R and at one gray level,

$$N = 1.46 \times 10^{14} P/R^2$$

The bit totals in Table 3 differ from those obtained directly from Fig. 1 by 1) a factor of 6 for encoding the 64 gray levels, and 2) an additional factor of either 1.5 or 6 (for the low and the high cases, respectively) that takes into account the range

Table 3 Photoimaging bit budget for 64 gray levels

Item	Surface, %	Area, 10^{12} m²	Resolution, m	10^{12} bits (total) Low	High
Monoscopic	67	100	60	0.24	1.0
Stereo (overlap utilizing I)	10	15	60	0.03	0.13
High-resolution monoscopic	1	1.5	2	3.2	13.3
Monoscopic color	67	100	500	0.011	0.05
				3.5[a]	13.5[a]

[a] Transmission time at 10^6 bits/sec—1.4 months for 3.5×10^{12} bits; 5.3 months for 13.5×10^{12} bits.

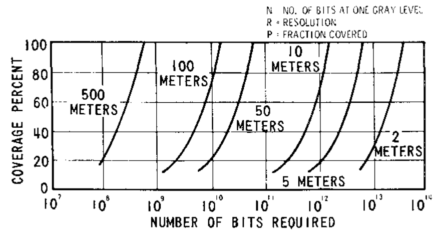

Fig. 1 Bit rate requirements.

of subjective estimates of the resolution requirements for detecting actual planetary features.

Although AREPS uses a large solar array, an alternate power source would be a nuclear reactor. The combination of nuclear power and electric propulsion has been extensively studied and has been shown to have a number of advantages, particularly for the outer planets where the intensity of solar energy is low.

Table 4 gives information on nuclear-electric propulsion missions to Mars and Venus[7] with over-all constaints similar to those of AREPS. A direct comparison is difficult because the nuclear-electric data are based on capture in a very loose elliptical orbit, whereas the AREPS spacecraft is captured in a circular orbit at a specific altitude. The principal advantage of a nuclear-electric spacecraft is the higher power available at the target planet for scientific data return and for extensive maneuvering of the spacecraft with the electric thrusters. The freedom from solar orientation requirements is an additional advantage.

Table 4 Comparison of nuclear-electric and solar-electric spacecraft

	Mars		Venus	
	Solar	Nuclear	Solar	Nuclear
P at 1 a.u., kw	40	65	14.3	65
P at planet, kw	20	65	20.1	65
MIPO,[a] kg	2200	2900	1990	3200
Trip time, days	222	170	191	140
Payload,[a] kg	960	960	1520	1520

[a] MIPO = mass in planetary orbit; payload = MIPO (electric propulsion system and power supply mass).

Planetary Science Objectives

Planetary science objectives for AREPS include both orbital experiments and probes or landers. The orbital experiments emphasize multiband imagery from the uv to the far ir at Mars, and radar mapping at Venus. Probes and landers provide temperature and pressure profiles and possibly other information about the atmosphere, whereas landers can, in addition, include instrumentation to measure deceleration signatures, surface pressures, and temperatures as a function of time, a seismometer, and possibly an inclinometer. At Mars, the lander package would conceivably include a camera.

Beginning in a high-altitude capture orbit, the AREPS can controllably occupy many different orbits while magnetospheric, ionospheric, and atmospheric properties are probed. Usually the properties are not spherically symmetric because of spatial and temporal variations in solar wind, temperatures, magnetic fields, radiation, and planetary shadows. In particular, the interaction of the solar wind with the ionospheres and atmospheres of both Venus and Mars is far greater than that with Earth because of the lack of substantial screening magnetic fields at the planets. Hence, this interaction is especially suitable for investigation by spiraling in toward the planet, using electric propulsion.

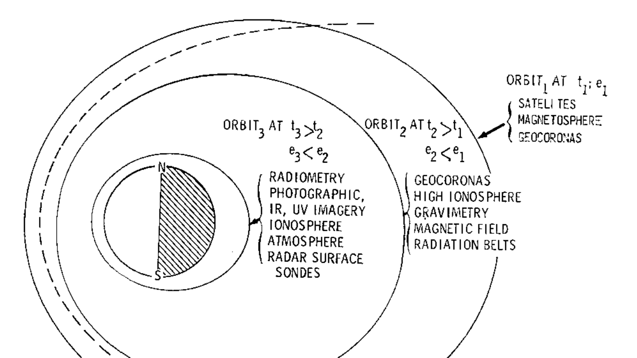

Fig. 2 Representative scientific objectives for planetary exploration.

Although circular orbits have been assumed for orbit computations, and because of photographic objectives, most of the aforementioned phenomena could be effectively studied using elliptical orbits at high altitude that gradually become less eccentric as the altitude is decreased. The great advantage of electric propulsion is that orbital parameters, such as inclination altitude, line of apses, and eccentricity, can be changed continually rather than abruptly, which is the case with chemical propulsion. The resulting orbital evolution occurs over a sufficiently long duration; thus, the layers of planetary structure can be sequentially probed to provide valuable synoptic data. Figure 2 shows three successive orbits and indicates the shift in scientific objectives as the orbit shrinks about the planet. Since most planetary structure above the lithosphere possesses a plane of symmetry in the equatorial plane, a polar orbit is generally preferred. A polar orbit also permits the maximum photographic coverage of the planetary surface. Electric propulsion provides this flexibility in the preselection of orbital parameters and the capability for a constantly evolving orbit as planetary science objectives are met.

Mars

Of the phenomena that could be studied by the photographic system, one of the most important is the progressive "wave of darkening" of certain Martian regions during spring and summer. It affects almost the entire planet. Two theories that have evolved to explain the wave of darkening are: 1) a seasonal response by small living organisms to the increased availability of water vapor, and 2) the transport of dust by winds generated by the temperature gradients, which vary with the seasons. At present, no preference can be given to either theory. AREPS arrives during the early phase of the northern wave of darkening (so called because it emanates from the North Pole) and photographs the rest of the cycle through its peak and gradual recession.

The first photographic map of Mars is made at the initial capture altitude. The mapping is then carried out at discrete orbital altitudes (Table 5) selected for rapid coverage, with photographic swaths which overlap by a predetermined amount on each orbital pass. The altitudes selected depend on the camera field of view α, which was chosen at 22° for illustrative purposes. A smaller α would produce higher resolution and a slightly lower set of altitudes. Figure 3 shows that for a wide range of optical system parameters, the range of allowable orbital altitudes that will produce the overlapping photographic tracks is very narrow. This figure is based on the assumptions of circular orbits, a swath overlap of 10%, pictures of square format, and a ratio of resolution at center to edge of each picture ranging from 2 (at the highest mapping altitude of 10,420 km) to 1.04 (at the lowest altitude of 820 km). The capability to attain precise orbits is an ob-

Table 5 Summary of planetary coverage parameters

	Orbital altitude, 10^3 km	Descent time, days	Resolution Min, m	Resolution Max, m	Orbits for complete coverage	Bit rate, 10^6/sec
Mars	10.42[a]		700	1400	8 (5 days)	0.053
		18.9				
	3.94		260	294	18 (4 days)	0.36
		12.1				
	1.88		125	134	24 (4 days)	1.0
		9.6				
	0.82		55	57	330 (36 days)	1.2
Venus	3.900[a]					
		29.2				
	1.000		100		1450 (110 days)	1.92 MHz[b]

[a] Capture altitude.
[b] Bandwidth of side-looking radar.

vious advantage of electric propulsion. With an all-chemical spacecraft it was calculated that with a periapsis of 1000 km and an apoapsis of 11,930 km, an error of 10 km in the apoapsis would reduce the photographic coverage possible during a 4500-hr period from about 67% to less than 5%.[8]

Orbits whose periods are an exact integer submultiple of a Mars day must be avoided. Otherwise, the spacecraft would cross the planet's equator at the same azimuth or longitude indefinitely; for example, for altitudes of 17×10^3 and 9.5×10^3 km, the planet rotates 360° and 180°, respectively.

From the initial altitude it would be possible to obtain good photographic coverage of Phobos, which orbits at an altitude of about 9400 km in the equatorial plane. Although AREPS is in a near-polar orbit, the orbital altitude can be adjusted during the descent maneuver so that Phobos comes into the field of view. From the capture altitude, the medium resolution photoimaging system would produce a resolution of 60–70 m on Phobos. As the spacecraft descends to the orbital altitude of Phobos, the two satellites would be essentially synchronous (although the orbits would be approximately orthogonal), and the photographic coverage could best be performed by swinging the camera system around to coincide with the apparent direction of Phobos' approach. The encounter could be made arbitrarily close for gravitational field determinations and other measurements.

One of the factors upon which the quality of orbital photography depends is the angle between the plane or the orbit and the terminator. For structure and color on Mars, this angle should range from ~20° to 70°, respectively. No one orbit will be optimum for both. However, the oblateness of Mars and the propulsion capability of AREPS can be used to precess the orbit in phase with the planetary rotation (0.52° per day) or the orbit can be fixed relative to the terminator. The proper conditions can be achieved over a range of altitudes and orbital inclinations. Chemical injection could, in principle, be used to produce specific orbital parameters, but electric propulsion possesses the advantage of fine control.

The planetary science objectives are affected somewhat by the conservatism adopted with regard to the distance of closest approach to the planet and its atmosphere. For some purposes it may be desirable to examine the possibility of orbits lower than 800 km. Factors affecting the distance of closest approach[9] to the Martian surface are: 1) structure and orientation of the spacecraft (effective mass-to-drag ratio); 2) the atmospheric density profile and magnetic drag; 3) the oblateness of Mars; 4) local gravitational perturbations; 5) solar pressure; 6) periapsis, apoapsis, and other orbital parameters; 7) seasonal variations; 8) accuracy and precision of guidance and control; and 9) quarantine requirements. Of the factors listed, the first is analyzed here, and representative information is presented for factors 2, 6, 8, 9.

The spacecraft may orbit Mars near the plane of the terminator; hence, the solar panels would be oriented with their plane parallel to the velocity, resulting in a greatly reduced drag compared to the orientation with an orbit in the plane of the solar meridian. The atmosphere (Fig. 3) derived

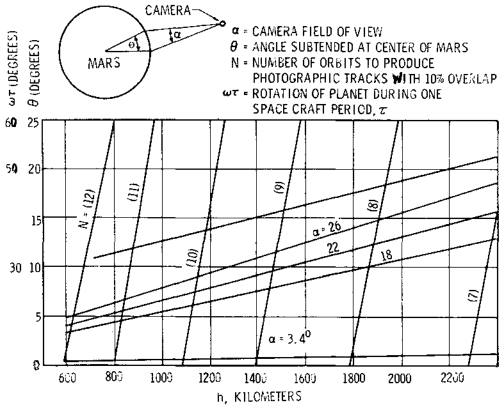

Fig. 3 Spacecraft altitude vs angular planetary coverage (Mars).

by Weidner[10] is employed. Requirements on the accuracy and precision of guidance and control will be met by the variable direction, low-thrust propulsion system that can act continuously if desired. The orbit could be nearly circular. With respect to quarantine requirements, the probability of contaminating Mars must be $<10^{-3}$ up to 1985—implying a 12-year lifetime for an orbiter injected in 1972.[11] The presence of a highly reliable, active spacecraft propulsion system conceivably would permit the spacecraft to descend below the nominal quarantine limit derived for a passive spacecraft.[9] Then, the spacecraft would gather data from the more advantageous altitude and then ascend to a longer lifetime orbital altitude.

Using these considerations to approximate a minimal altitude, the force of drag is

$$D = \rho V^2 C_D A/2 \tag{1}$$

where ρ is atmospheric density, kg/m³; V is velocity, m/sec; A is projected area on plane perpendicular to V, m²; C_D is drag coefficient. Also,

$$a_D = D/M = \rho V^2 C_D A/2M \tag{2}$$

where a_D is acceleration due to drag, m/sec², and M is total mass of the spacecraft in planetary orbit, kg. From Weidner's model (Fig. 4), ρ is approximately 10^{-12} kg/m³. Other studies have shown that a conservative value for C_D is 3. For AREPS, $M = 2200$ kg; $A = 18.6$ m² (assuming 5% of the total solar array of 372 m²); $V^2 = 10^7$ m²/sec² at 500 km altitude. Using these values, $a_D = 1.2 \times 10^{-7}$ m/sec², which is over three orders of magnitude lower than the acceleration produced by the electric propulsion system. This in conjunction with Eq. (2) and Fig. 3 implies that the altitude at

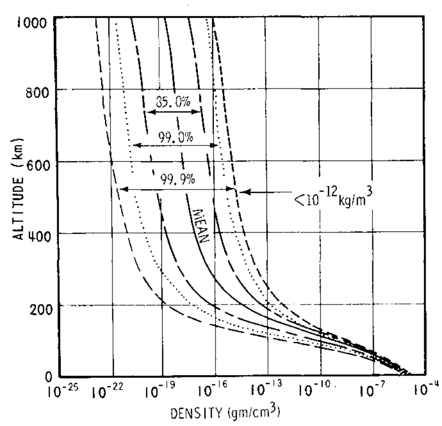

Fig. 4 Preliminary mean profile and confidence envelope for Martian atmospheric density.

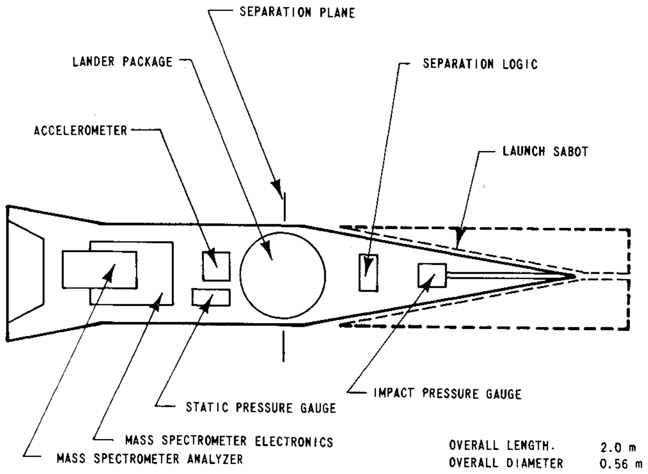

Fig. 5 General arrangement of Venus lander.

which drag deceleration becomes equal to the electric acceleration is well under 200 km. Therefore, the spacecraft could easily descend to 500 km altitude, or probably even lower, and subsequently return to a higher orbit. For a highly eccentric orbit, it would be possible, in principle, to enter the atmosphere at a periapsis altitude at which the drag exceeds the instantaneous acceleration, provided the negative velocity increment were replaced during the higher altitude portion of the orbit.

Venus

At Venus, the prime scientific objective of the AREPS is a 100-m-resolution radar map of the entire surface. (The radar system will be described later.) In addition, the release of a lander (which also serves as a probe during descent) for detailed studies of the atmosphere and the surface will enhance scientific return from this mission. The importance of this scientific objective has been emphasized.[12] Tables 6 and 7 list representative weights of equipment, and components of a lander that is compatible with the payload availability of the spacecraft.[13] Figure 5 shows how such a lander may be arranged. The dimensions are approximate, since detailed studies will be required before such a system can be defined accurately.

During descent, several small atmospheric probes will be used to measure temperature, pressures, density, and sonic velocity. These probes will be programed to sample the atmosphere at critical points in the planetary atmospheric structure, e.g., at the poles and at the antisolar and subsolar points. In later missions, lightweight, limited-resolution vidicon cameras to delineate the position of cloud layers and cloud structure should be considered for inclusion in these small probes. Such a system should also be equipped to study the polarimetric properties of the clouds to obtain a better definition of particulates. Deceleration measurements are used in conjunction with the known aerodynamic characteristics of the probe to establish an approximate density profile. Measurement accuracy is, of course, limited by uncertainties in probe characteristics in the Venus atmosphere.

High-priority instruments for the lander package are listed in Table 6. These instruments can perform their measurements in periods ranging from seconds to minutes. The value of a lander would increase only slowly with its lifetime on the surface, since all diurnal variations must be relatively slow. (A Cytherian solar "day" is 118 days.)

The most suitable communications link would be a solid-state VHF transmitter operating at several tens of watts, with a moderately directional downward-pointing antenna in the spacecraft. The probe also requires a simple solid-state memory to store data collected during the anticipated blackout period, an encoder and control logic unit to control operation of the instruments carried and to assemble their data into a format suitable for transmission, and a battery power supply. Current progress in the development of sterilizable batteries indicates that high-capacity silver-zinc batteries may be available. In this case, a 4-kg power supply will furnish 100 to 200 w-hr and a transmitter will radiate as much as 100 w; this will be ample power to operate all instruments. Because of the short period of transmission (descent and landing), transmitter output may, in fact, be limited by antenna performance and available output transistors rather than the available power.

Table 6 Mass summary of Venus lander, kg

Instruments used after impact		
Thermometer, pressure gage, anemometer, photometer	1	
Gas chromatograph	2.3	
Radiation counter	1	
Penetrometer	1	
		5.3
Instruments used during descent		
Accelerometers (2 ranges)	1	
Pressure spectrometer	1	
Mass spectrometer	4	
Temperature instrumentation	0.2	
Density instrumentation	1.2	
		7.4
Communications		
Transmitter (50–100 w VHF)	3.2	
Encoder and control logic	1	
Solid-state memory	1	
Power supply (100–200 w-hr)	5	
Antenna	1	
		11.2
Wiring and mounting provisions		6
Sterile container/launch tube, checkout and launch provisions, receiver and antenna		32
Launch rocket		16
Structure and environmental protection		50
Contingency		32
Total lander mass, kg		160

Table 7 Summary of Venus science package weights, kg

Side-looking radar		270
Lander/probe		160
Atmospheric probes 2 (nonsurviving)		94 (47 each)
Atmospheric structure	1.2	
Mass spectrometer	4	
Ancillary equipment (data handling and communication, power, thermal control structure)	42	
Entry science		70
Microwave spectrometer[14] (atmospheric 0.3–3 cm wavelength)	16	
Microwave imager[14] (surface thermal mapping—3 cm wavelength)	10	
Infrared interferometer[14] (1.5–5 wavelength)	16	
Magnetometer	2	
Plasma and particles experiment	8	
uv spectroscopy	11	
Photo-imaging experiment[14]	7	
		594 kg

Table 8 Mission parameters

	Mars	Venus
Launch date	Nov. 8, 1977	Dec. 9, 1976
Arrival date	June 18, 1978	June 19, 1977
Launch mass, kg	4420	5355
Mass in planetary orbit, kg	2200	1990
Travel angle, deg	143	234
C_3, km²/sec²	14.8	4.02
Capture stage fraction	0.40	0.60
ISP, sec	2786	3162
Thrust at planet, N	0.866	0.815
Power at 1 a.u., kw	40	14.3
Power at planet, kw	19.4	19.92

Mission Sequence

After launch, the spacecraft will begin the heliocentric trajectory. In the case of the Mars 1977 project (Table 8), the retropropulsion system places the spacecraft in a near-polar, circular orbit at 10,420 km (Table 5). Due to planetary oblateness, the orbital inclination is chosen at 95° to provide a natural retrograde precession of the orbit and to match the angular change of the terminator (0.52° per day). The descent time from initial capture altitude to 820 km will be 42 days, and the time required for photography at the altitudes shown in Table 5 will be 49 days. The descent and mapping operations will be completed 122 days before conjunction. In practice, communications between the spacecraft and Earth may become poor at 10° (30 days) in communication angle before conjunction, because of interference from the sun. Additional time may be required for orbital trim, particularly for rotating the orbital plane to remain near the terminator for optimum photography.

For the Venus mission (Table 8), the sequence is similar to that at Mars, except that the capture into polar orbit is performed at a much lower altitude to reduce the descent time required at the more massive planet. The oblateness of Venus is not known, so precessional effects cannot be estimated. Assuming a Venusian rotation rate of approximately 243 days and neglecting precession, the side-looking radar system maps the entire surface from 1000 km altitude in 122 days. Then the combination lander and probe can be released. Its retrorocket firing leads to entry into the atmosphere. After partial descent, the vehicle ejects a heat shield and continues toward the surface as it telemeters environmental data to the spacecraft. The advantage of carrying out preliminary mapping of the surface is that careful selection of the landing site will improve the lander's chance for survival.

The arrival date at Venus will be 218 days before the conjunction; after descent to 1000 km, approximately 185 days remain. As with Mars, communication may be impossible when the communication angle becomes ~10°. Before conjunction, the time available for communications with the spacecraft would be reduced by about 30 days in this case. Following the conjunction, it would be reasonable to expect the spacecraft to continue its exploratory functions, including repeated SLR mapping of the surface for a year or more. The amount of electric propulsion fuel would have to be sized for the attitude control and orbit trimming requirements for the desired length of the mission.

Spacecraft Description

The suggested launch vehicle for both the Mars and the Venus missions is the Titan III D/Centaur, which is designated Titan IIIx(1205)/Centaur in the OSSA Launch Vehicle Estimating Factors Document.[6] It offers very attractive payload capability and low cost. The Saturn IB/Centaur has an equivalent payload capability.

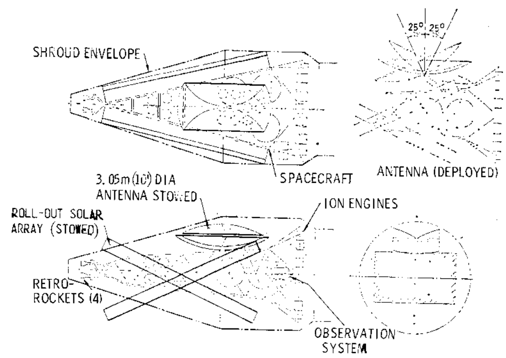

Fig. 6 Spacecraft configuration with parabolic dish antenna.

The spacecraft structure is composed of lightweight, truss-like hollow tubular aluminum and composite members. Figure 6 shows such major features of the Mars spacecraft as the folded parabolic antenna, rollout solar array, 16 rear-mounted main electric thrusters and 8 side thrusters for orbital precession control at Mars.

Figure 7 is a photograph of a spacecraft model with a phased array antenna. Most of these subsystems and those that follow have been treated elsewhere in some detail.[2] Four chemically fueled retroengines, developing 46-kg thrust per engine, are used to capture the spacecraft into planetary orbit at Mars or Venus. For both missions, the rolled-out solar arrays can rotate ±25° about the spacecraft pitch axis, placing less demands on the attitude control system during heliocentric transfer orbit and on the communications antenna pointing control system.

The mercury-propellant electric ion engines are rated at 2.5 kw each. The thrusters are mounted on actuated plates, which can be moved upon demand of the guidance system to permit attitude control of the spacecraft and to align the resultant thrust vector through the spacecraft center of mass.

Table 9 shows subsystem mass breakdowns. Spacecraft structural mass is 6% of total spacecraft mass at Earth escape in both missions. Other load-carrying structures such as tanks and thrust vector alignment mechanisms are tabulated as separate items. The subtotal may be considered as the payload delivered into planetary orbit, especially since the solar arrays and thruster subsystems are used thereafter to generate power and "tune" the spacecraft's orbits. The 40 kw-rated solar array (at 1 a.u.) will yield 19.4 kw at Mars. Thus, only 8 ion engines operate when approaching Mars, instead of the 16 engines that operate at Earth escape (in effect this provides 8 standby engines). The Venus probe has 14.3 kw available at Earth that increases to 19.92 kw near Venus. In this case, 6 engines are used initially, and 8 are used at Venus (with 1 additional standby aboard).

Figures 8 and 9 depict the communication angles and distances. The communication angle is positive when the angle

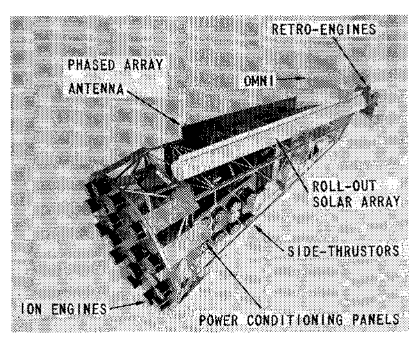

Fig. 7 Photograph of spacecraft model (Mars mission).

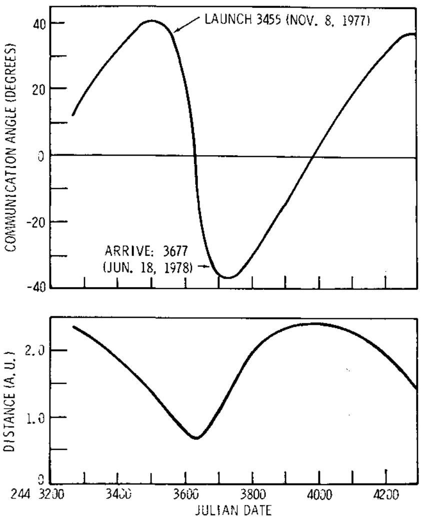

Fig. 8 Communication parameters, 1977–1978 Mars opportunity.

from planet-sun line to Earth is clockwise, as viewed from the north pole of planet. The launch times chosen in both cases enable the spacecraft to be captured at the planet with ample time for photoreconnaissance and scientific data acquisition. The heliocentric transfer trajectories were calculated using an elaborate three-dimensional, two-body, variable low-thrust trajectory optimization program.[15] Once capture was effected, circularization of the orbit and descent after circularization was accomplished by a simple minimization procedure[16] where the thrust was assumed constant.

Communication Subsystem

The communications subsystem performs three vital functions: 1) active participation in tracking of the spacecraft by the Deep Space Instrumentation Facility (DSIF) ground stations, 2) command reception, and 3) transmission of spacecraft engineering and science data. Tracking information (range, velocity, and angular position) gathered by ground stations of the DSIF is used for spacecraft navigation during the interplanetary leg of the mission. The onboard DSIF transponder receives, coherently offsets in frequency, and retransmits the carrier and range-modulation signal to the ground stations. The range measurement is performed at each ground station by measuring the time delay between the transmitted and received pseudo-random binary sequence of range-modulation signal. Velocity is determined by measuring the Doppler frequency shift of the transponded carrier. Angular position is measured at each ground station by a simultaneous multilobe receiving antenna pattern. Commands to the spacecraft and low-data-rate telemetry of spacecraft engineering performance data are provided by the DSIF transponder and an omnidirectional antenna for the entire duration of the mission. The DSIF transponder and omniantenna weigh $\sim$22 kg and require 115 w d.c.

Telemetry of Mars Mapping Data

As previously mentioned, two transmitters are proposed: the low-power DSIF system with one function of providing telemetry for performance measurements and a high-power system for experimental measurements. The maximum bit rate for the latter is a primary consideration. To calculate it, we assume: 1) coherent operation with matched-filter detection and PCM/PM modulation will be used; 2) the bit error probability, $P_e{}^b = 1 \times 10^{-3}$; 3) the ratio of received signal energy per bit-to-noise power per unit bandwidth, E(joules)/(N/B) (W/Hz) = 7 db; 4) receiver noise temperature, T = 40°K; 5) receiving system noise spectral density, ϕ_K = −182.6 db for 40°K; 6) receiving antenna gain, GR = 61.0 db (210-ft dish); 7) transmitter antenna gain, G_T = 35 db (10-ft dish); 8) carrier modulation loss, L_M = 3 db; 9) design safety margin, L_T = 6 db; 10) carrier frequency, f = 2300 MHz; 11) rf transmitter power 10,000 w; P_T = 67 db (mw).

The space-loss equation expressed in db is

$$L_S = 32.45 + 20 \log f + 20 \log R \tag{3}$$

where f is in MHz (2300 MHz), and R is the communications range in km. R values for Mars vary from 5.6×10^7 to 4.0×10^8 km. Here a nominal value of $R = 2.2 \times 10^8$ km is assumed and, hence, L_S = 266.5 db. The normalized received signal-to-noise ratio for the telemetry link expressed in db is

$$S/(N/B) = P_T + G_T + G_R - \phi_K - L_S - L_M - L_T \tag{4}$$

For the Mars photo mission this gives

$$S/(N/B) = 67 + 35 + 61 + 182.6 - 266.5 - 3 - 6 = 70.1 \text{ db}$$

The bit rate b (bits/sec), expressed in db is

$$b = S/(N/B) - E/(N/B) = 70.1 - 7.0 = 63.1 \text{ db}$$

This yields 2.04×10^6 bits/sec with a safety factor of 4(6 db).

Telemetry of Venus Mapping Data

Received signal bandwidths for the synthetic-aperture radar used in mapping Venus vary from 0.384 to 3.86 MHz. The S/N of the unprocessed radar signal is quite low and thus may prohibit using the efficient pulse-code-modulation technique for transmitting the data to Earth. Therefore, pending further study, it is proposed to use low-index frequency modulation to limit the bandwidth of the transmitted spectrum to twice the signal bandwidth. To prevent degradation of the weak data signal, the noise contribution of the spacecraft-to-Earth link should be very small.

Table 9 Spacecraft subsystem masses, kg

	Mars	Venus
Science instruments	136	164
Lander	...	160
Camera system	136	...
Side-looking radar system	...	270
Solar array (18.2 kg/kw)	727	360
Communications	190	190
Electric thrusters (5 kg/engine)	120	45
Thrust vector control mechanisms	36	11
Power conditioning (4.1 kg/kw)	163	82
Electric propellants (for use in capture orbit)	150	76
Tankage for preceding	5	6
G&C	90	90
Batteries	18	18
Cabling	91	90
Thermal control	22	25
Structure of spacecraft	252	320
Tankage for electric propellants (heliocentric transfer orbit)	21	11
Retro-engines, tanks, plumbing	43	72
Subtotal: mass in Mars orbit (Payload)	2200	1990
Electric Propellants (heliocentric transfer orbit)	680	368
Pyrotechnic	12	12
Retro-propellant (for planetary capture)	1440	2878
Contingency (2%)	88	108
Total: mass at Earth launch, kg	4420	5356

Assuming the same transmit-receiver link parameters as for the Mars mission (the space loss for Venus is essentially the same as for Mars), the signal received by the DSIF station has a 70.1 db S/N for a unity noise bandwidth. For the synthetic aperture signal bandwidth of 1.92×10^6 Hz,

$$S/N_{\text{received, db}} = 73.1 - 10 \log(1.92 \times 10^6) = 73.1 - 62.9 = 10.2 \text{ db}$$

Any further increase of signal bandwidth would require a corresponding increase of transmitter power, since 10 db is a nominal value required for input S/N.

Transmitter-Parabolic Antenna Spacecraft Integration

The folding, 3-m dish in Fig. 6 is made of a 6.3-mm-thick honeycomb aluminum with 0.13-mm-thick face sheet and 3.75-cm-square by 3.2-mm-thick torque tubular support structure. The estimated weight of the dish structure including the antenna feed is 46 kg. The mechanical deployment system utilizes a lever arm and a pair of electric motors, each of which is coupled to a worm gear and pinion. The estimated weight of the deployment mechanism is 45 kg. The transmitter utilizes two electrostatically focused klystron amplifiers and power supplies and weights 37 kg. This yields a total weight for the paraboloid antenna system of 128 kg.

Alternative Communication System Concepts

The solid-state phased array, operating at S band, was proposed and discussed by Wood.[2] Its chief advantages are the potentially high reliability because of the large number of individual transmitting elements, and the capability for electronic steering of the rf beam. The latter capability permits the communication system to track the Earth throughout the Martian orbit without the use of a mechanical tracking system such as a gimbal system, which, depending on the communication angle (Fig. 8), must swing through as much as $\pm 45°$. Unfortunately, it is not possible at this time to assess the capability of phased arrays for deep-space communication, since efficiencies and operating temperature limits, which contribute to to both heat rejection problems and reliability considerations, have not been adequately studied. If the state-of-the-art continues to advance rapidly, such systems might be a very desirable substitute for the parabolic dish system. The spacecraft with a phased-array is shown conceptually in Fig. 7.

Laser communication systems capable of extremely high data rates at low power requirements, with masses competitive with those of rf systems, have been demonstrated in recent years. Pointing and tracking problems have been solved in the laboratory and in limited range field tests. The testing of laser communication systems over great distances in space should be given the highest priority in the development of space technology. Statements to the effect that laser communications provide far more data-return capability than is needed for practical space missions are frequently encountered; however, the reverse is actually the case. Data-return rates of recent successful planetary probes have been low, not because of insurmountable problems in realizing scientific systems capable of more rapid data acquisition, but because of the severe limitations placed on them by available communications capabilities. A fact relevant to the Venus scientific program discussed in this paper is that side-looking radar, like cameras, can acquire data at mb/sec rates.

Table 10 Side-looking radar operating parameters

Operating frequency	10 GHz
Peak transmitted power	99 kw
Average transmitted power	220 w
Pulse repetition frequency	4250 Hz
Pulse length	0.5 sec
Signal bandwidth	1.92 MHz
Minimum antenna pointing angle	42°
Beamwidth	0.74°
Antenna dimensions	3m × 2m
Signal-to-noise ratio	12 db
Resolution at 1000 km altitude	100 m
Transmitted power to Earth (communication system)	10 kw

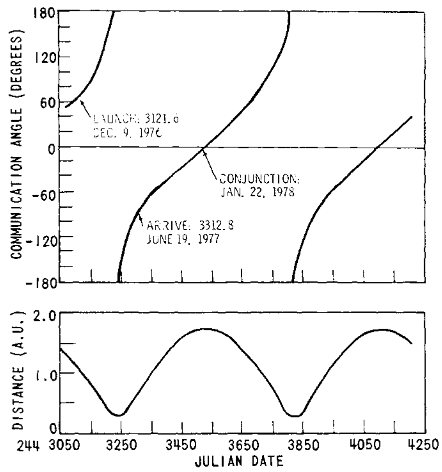

Fig. 9 Communications parameters, 1977-1978 Venus opportunity.

Photoimaging System

A medium-resolution (50 to 100 m) mapping of the entire Martian surface would provide a reference for all future unmanned and manned explorations. In addition, large amounts of high resolution (≤ 10 m) coverage, along with stereoscopic overlapping and multicolor photography are necessary to obtain accurate information about the Martian surface. Candidate photoimaging systems are photographic film, dielectric tape, and vidicon cameras. Each possess unique advantages and disadvantages. However, their resolution capabilities are somewhat similar (provided that sufficient image-motion compensation is available for the less sensitive devices) and, as will be shown, resolution is one of the principal determinants of the information rate.

The surface features resolvable in the television photos are related as follows:

$$R_I = 2.82\, Hd/NF \cos^2 n \tag{5}$$

where R_I is intrinsic resolution (m), H is orbital altitude (m), d is TV vidicon target height (mm), N is number of active scan lines for vidicon image readout, F is focal length of camera lens (mm), and n is angle between optical axis and normal-to-Mars surface. Thus, the minimum dimension of surface features discernible in the TV photos decreases (increasing resolution) with decreasing altitude and increasing focal length. The focal length of TV cameras flown on the Ranger and Mariner Mars missions have varied from 25 mm to 1 m to provide a field of view and surface feature resolution commensurate with the available transmitter power. The ruggedized 25.4-mm-diam vidicon to be used on the Mariner Mars 1969 mission[17] has 704 active scan lines over a target height of 9.6 mm, achieving a scan line density of 73 lines/mm. A 50.8-mm-diam return-beam vidicon currently under development at RCA promises to achieve a scan density of 200 lines/mm, an increase in resolution by a factor of 3. (The achievable scan density is also influenced by exposure duration, illumination, and frame rate). The resolution achieved with the latter is plotted in Fig. 1 vs altitude for the lens focal lengths most suitable for the AREPS mission. A phenomenon that is not very obvious is the fact that the 50.8-mm return-beam vidicon produces 1.5×10^8 bits/picture when the 25-mm by 25-mm target raster is encoded with a 6 bits/picture element (0.005 mm by 0.005 mm).

Side-Looking Radar System

The side-looking radar is now a proven instrument for all-weather reconnaissance and ground mapping. It is potentially a very valuable sensor to place on board a Venus orbiter, because microwaves can penetrate the cloud cover of the planet.

There are three types of side-looking, ground-mapping radars: real aperture, unfocused synthetic aperture, and focused synthetic aperture. Real aperture radar systems have comparatively limited resolution capabilities and were not seriously considered for this mission. The unfocused system is suitable, provided that resolution requirements are kept to ~100 m at 1000 km altitude. It coherently adds successive returns from a fixed range in the signal processor. The physical aperture assumes the positions of elements of a "synthetic" array as a function of time, as the spacecraft moves along its orbital path. The element spacing depends upon spacecraft velocity and the pulse period of the radar. The maximum array length is determined by the tolerable difference in phase shifts between the signals received at the center and the ends of the array. The focused radar system can provide somewhat better resolution because the equipment is more sophisticated.

The SLR operating parameters chosen for the AREPS mission to Venus are listed in Table 10. The beamwidth which results from these parameters is sufficient at 1000 km to produce mapping swaths, which overlap ~10%, on the surface as the planet rotates beneath the orbital trace at the currently estimated rate of once every 243 days. Oblateness of Venus, if present, would cause a precession of the orbit at inclinations other than 90°, thereby changing the mapping parameters. Such a condition could be anticipated during the descent, and a final orbit could be selected which provides the proper mapping relationships.

Conclusions

The electric propulsion concept offers attractive mission possibilities at Mars and Venus, not only because of the payload delivered into planetary orbit, but also because of the extreme versatility of the spacecraft once in orbit and because of the power available to transmit information to Earth at rates that are several orders of magnitude higher than those normally associated with chemically propelled spacecraft. Although Mars and Venus were chosen to illustrate the capability of an electrically propelled spacecraft for planetary exploration, the advantages also apply to proposed missions to other planets, such as Jupiter.

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