DESIGN AND SCIENCE INSTRUMENTATION OF AN UNMANNED VEHICLE FOR SAMPLE RETURN FROM THE ASTEROID EROS

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Unmanned missions to the asteroids have been proposed and investigated as part of the overall plan of exploration of the solar system. A principal incentive for landing on an asteroid and retrieving a surface sample for return to Earth is the expectation that detailed laboratory analysis of the sample material's chemical composition, crystal structure, surface texture, magnetic characteristics, radioactive state, and age can provide essential clues, not available by other means, to the origin of asteroids and possibly the history and formative processes of the solar system (Alfvén and Arrhenius, 1970*a,b;* Bratenahl;[1] Friedlander and Vickers, 1964; IIT Research Institute, 1964; Öhman, 1963). The results may indicate, for example, to what extent accretion or fragmentation processes have been involved in the formation of asteroids.

Asteroids of the Apollo family, such as Icarus, Geographos, and Eros, periodically approach very close to Earth. Except for the Moon, they are in fact Earth's closest neighbors in space. Eros, in particular, is reasonably accessible to Earth for a landing and sample-return mission with launch opportunities recurring about every other year, at a much smaller propulsion energy than would be required for comparable missions to other planetary bodies, owing to the proximity of its orbit and its almost negligible gravity. A mission to Eros would be desirable also as a precursor to a more complex and costly Mars sample-return mission.

The use of solar electric propulsion as the means of primary propulsion during the outbound and return phases permits the use of a smaller booster than would be required for a ballistic mission with equivalent payload capability and thus can achieve a significant cost saving. The use of electric propulsion also alleviates launch date constraints, provides flexibility in mission profile selection and guidance, and facilitates execution of the final approach and descent phases under remote control from Earth. It permits extended hover phases in close proximity of the asteroid during which television (TV) images can be transmitted to Earth and necessary corrective commands

[1]See p. 561.

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returned to the landing vehicle, with round-trip communication delays of 35 to 40 min. A fully autonomous vehicle that would perform the final approach and landing at Eros without assistance by ground control would be more complex, more costly, and less reliable.

This paper discusses the scientific objectives to be achieved by an Eros landing and sample-return mission, the instrument payload to be carried, and the mission profile and critical mission phases to be executed. The conceptual design of a solar electric-propulsion spacecraft bus, or stage, capable of returning to Earth a capsule with 100 kg of Eros surface sample material will be described. The round-trip mission time is about 1000 days. The results of this investigation indicate the feasibility of this mission based on available electric-propulsion technology plus existing spacecraft design concepts and flight hardware. Such a bus could be developed in time to meet the 1977 launch opportunity. Similar opportunities occur approximately every other year.

MISSION OBJECTIVES AND SCIENTIFIC INSTRUMENTATION

It is believed that detailed examination of matter from an asteroid will provide information valuable to an understanding of the processes of planetary formation and of solar system hetegony (Alfvén and Arrhenius, 1970*b*). No foreseeable development of onsite techniques can hope to achieve the detail or depth of analysis that is possible with full laboratory equipment, as experience with lunar samples from Apollo has readily demonstrated. The central aim of an asteroid sample-return mission, therefore, will be to acquire asteroidal material to permit such an examination by laboratory analysis. It does not follow, however, that ambient or onsite measurements will have no place on an asteroid return-sample payload. Such measurements will contribute significantly to interpretation of laboratory results. Hence, instrumental surveillance of the target asteroid will be an important phase of the mission before, during, and after touchdown and perhaps after takeoff as well.

Surveillance will serve two functions, both of which encompass extensive local measurements. These functions are (1) selection of an optimal landing spot and (2) characterization of the physical context in which the asteroid is found and from which the samples are taken. These functions are naturally interrelated.

Several factors may enter the selection of an exact landing point. One major consideration involves the relative motion of the spacecraft with respect to the terrain at touchdown, which dictates landing near a pole of the asteroid's axis of rotation. Another consideration will be the angle of solar illumination. These are described in a later section. In addition, unpredictable properties of the asteroid may contribute to site selection. Among them would be local topography, which might determine that one area would be more level or more varied in composition than another or that samples would be obtained with more facility there than elsewhere, and local magnetic signature, which might suggest that samples from one spot would be more revealing of the early planetary environment than those taken from another.

The physical significance of laboratory analyses of returned samples could be seriously compromised without reasonably complete specification of the immediate and general present environment from which the samples come. It will be important to know how representative of the body composition and of the site materials the returned samples will be. For example, the cumulative effect of solar-wind impact on some surface materials may be influenced by a strong local magnetic field. Therefore, instruments measuring directly the local field, its gradient, and the resulting solar-wind deflection would be very important as part of the scientific instrument package. Complete field, particle, and optical characterization of the solar wind around the asteroid, of the asteroid as a whole, and of the landing site will therefore be essential for successful completion of the mission objectives.

To summarize, the overall mission purpose of collecting samples of asteroidal material from which comprehensive inferences on solar system formation can be obtained with minimal ambiguity will be served by three interrelated mission objectives:

(1) Examination of the asteroid's geometrical configuration and of its environment, including its interaction with the solar wind, if any
(2) Acquisition of samples of asteroidal material from the surface for return to Earth
(3) Observation and characterization of the site from which samples are taken and documentation of the relationship of the samples to the site and of the site to the asteroid

The scientific objectives described above will be served by groups of instruments that provide the following functions:

(1) Measurement of the ambient solar wind, the distant electromagnetic properties of the body, and the interaction, if any, of the body with the solar wind
(2) Observation of the asteroid's size, configuration, surface features, rotation, and optical properties
(3) Detection of gaseous ionized envelope or plasma sheath
(4) Examination of the surface and subsurface characteristics at the landing site
(5) Observation of surface features at the landing site
(6) Observation of ambient conditions at the site
(7) Acquisition of sample material

The instruments needed to perform the tasks included in the above functional categories are given in the following list. Asterisks denote items that might be assigned a lower priority than the others because their data would be unessential to success of the mission, being partially redundant in relation to what laboratory analysis would discover or being partially deducible from other sources.

(1) Ambient background and interaction measurements:
- (*a*) Plasma probe
- (*b*) dc magnetometer and gradiometer
- (*c*) ac magnetometer
- (*d*) Plasma wave detector
- (*e*) Dust (micrometeoroid) detector
- (*f*) Cosmic-ray telescope*
- (*g*) Gravity gradiometer

(2) Asteroid observation:
- (*a*) Imaging telescope (TV)
- (*b*) IR, UV, and visible spectrophotometers
- (*c*) Photopolarimeter

(3) Gas envelope detection:
- (*a*) Low-energy plasma analyzer
- (*b*) Ion mass spectrometer*

(4) Surface examination:
- (*a*) Surface scraper
- (*b*) Seismic detector, possibly with "thumper"
- (*c*) α-Scattering analyzer*

(5) Surface observation: Imaging telescope (TV), as in function (2)

(6) On-site ambient environment detection: same as function (1)

(7) Sample acquisition:
- (*a*) Loose matter collector
- (*b*) Core borer

The scientific value of the mission would be enhanced if certain instruments were left on the surface together with the communication system required for telemetering their data to Earth. The weight of the devices left behind would be taken up, in part, by the collected samples. Detached instruments would include those for functions (1) and (5) plus the low-energy plasma analyzer (function (3)) and the seismic detector (function (4)). However, extended autonomous operation of a telemetry system and its power source on the asteroid with communication distance to Earth in excess of 2 AU involves technical problems not considered within the scope of this paper.

MISSION PROFILE

Trajectories, performance characteristics, and payload capabilities for one-way and round-trip missions to Eros have been investigated by Friedlander, Mascy, Niehoff, and others (Friedlander and Vickers, 1964; IIT Research Institute, 1964; Mascy and Niehoff[2]) for both ballistic and low-thrust

[2]See p. 513.

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propelled vehicles. Figure 1 shows representative outbound and inbound trajectories of a 1050 day round-trip mission with a 50 day stopover at Eros for the 1977 launch opportunity, based on data obtained by Mascy.[3] The mission uses solar electric propulsion both ways, with thrust characteristics and thrust pointing angles optimized to return a maximum amount of asteroid sample material to Earth. The vehicle is launched by a Titan IIID/Burner II booster and uses 10 kW of initial propulsive power at Earth departure. Low thrust is applied continuously during the outbound phase such that the vehicle arrives at Eros with zero relative velocity v_∞ and can land on the asteroid with almost no additional propulsive effort. Similarly, low thrust is applied continuously during the return trip to reduce the approach velocity on returning to Earth and the required Earth capture maneuver. We assume that the sample-return capsule carried by the interplanetary bus vehicle will be inserted into an eccentric Earth parking orbit for subsequent retrieval by orbital shuttle or by a deorbit maneuver, atmospheric entry, and parachute landing. This mission profile is shown schematically in figure 2 and is used as a basis for defining the vehicle design features and operational characteristics to be discussed below.

We note in figure 1 that the outbound trajectory departing from Earth on February 25, 1977, swings in a wide arc to an aphelion distance of 1.67 AU to achieve the desired velocity matching with the target at the encounter date of July 10, 1978, near perihelion. A gradual plane change necessary to attain the 10°.8 orbital inclination of Eros is included in the outbound propulsion phase.

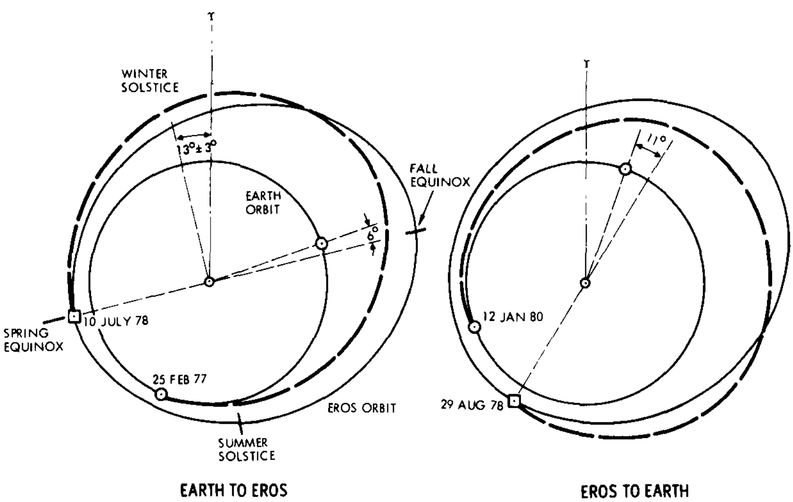

Figure 1.—Eros round-trip trajectory.

[3]Specific data used in this article are based on work by Mascy and Niehoff and are essentially in agreement with data published in their paper in this volume on p. 513.

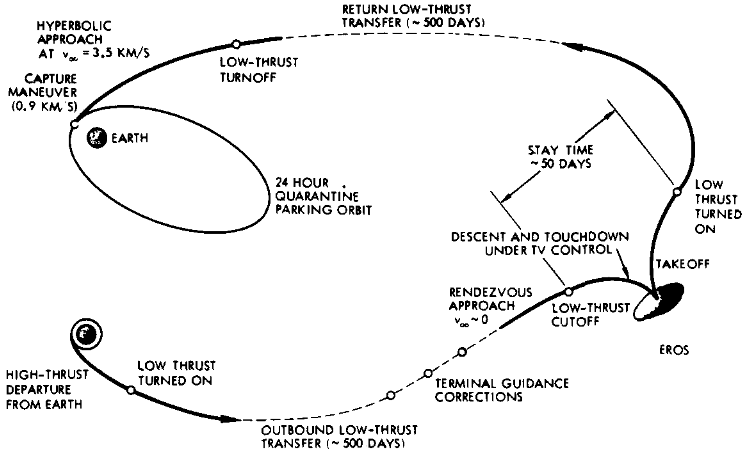

Figure 2.–Eros round-trip mission profile (schematic).

The return trip departing Eros on August 29, 1978, and arriving at Earth on January 12, 1980, has similar characteristics. Mission opportunities with comparable characteristics occur about every 2 yr. 1977, 1979, and 1981 are favorable mission years (Mascy and Niehoff[4]).

A characteristic feature of this class of mission profiles is the large communication range to Earth (2.1 AU) and the fact that Earth and Eros are in almost exact opposition at encounter. These conditions do not change much during the 50 day stopover because Earth and Eros move nearly at the same rate. In the reference trajectory, the arrival at Eros occurs a short time after syzygy. Communication blackout must be avoided during this critical part of the mission. The Earth-Sun separation angle subtended at Eros is initially 3°. This gives a margin of only 1° from the blackout zone, 2° on both sides of the solar disk, which is assumed under conditions of average solar activity. Actually, because during the late 1970's solar activity will be increasing toward a maximum level, a larger margin than 1° would be desirable. The separation angle increases to 5°5 during the 50 day stopover. Therefore, a 20-day delay in arrival will increase the margin by 1°. This can be achieved with only a minor change in payload performance owing to the flexibility of low-thrust missions, as shown by Mascy.[5] A delay in arrival date is also desirable to improve seasonal conditions at the preferred polar site as discussed below.

ADVANTAGES OF USING ELECTRIC PROPULSION

Several major performance advantages accrue in this mission from the use of electric propulsion. The first, and by far the most important one, is the large

[4]See p. 522.
[5]See p. 525.

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reduction of propellant mass due to the high specific impulse at which electric thrustors operate compared to chemical rockets. Typically, the difference in specific impulse is one order of magnitude. To carry a specified amount of payload mass on a trip requiring a major propulsive maneuver away from Earth, as in a rendezvous mission, a spacecraft using electric propulsion is launched with an initial mass two to three times smaller than a corresponding chemically propelled spacecraft. This permits the use of a smaller, less costly booster. Conversely, given the same booster size, the electrically propelled vehicle carries a much larger payload than the chemically propelled one.

Second, out-of-plane maneuvers for a change of orbit inclination or nodal points can be accomplished at a modest extra propellant cost along with the primary in-plane maneuver necessary to achieve the desired aphelion, perihelion, or matching of the target velocity. Changes in mission profile, launch windows, and trip times involve smaller weight penalties than in ballistic missions, essentially as a result of the flexibility in three-dimensional orbit geometry.

Third, the extended thrust phase permits a continuous correction of guidance errors at practically no extra propellant expenditure, simply through deflection of the thrust vector from its nominal orientation. In a rendezvous mission with a target of poorly defined ephemeris such as a small asteroid, low-thrust terminal corrections can be made after the target is acquired by an onboard optical sensor.

Finally, the prolonged low-velocity approach to the target permits extended visual observation and reconnaissance via TV link, the selection of an appropriate landing zone by ground control, and final corrections for obstacle avoidance. The ability to hover for an extended period over the landing site at an altitude where the small electric thrust is sufficient to balance the small local gravity is particularly desirable in view of the very time-consuming round-trip signal transmission process (35 to 40 min). In this context, the ability to transmit video images at high bit rates, using the excess solar electric power not needed for thrust purposes at this time, is a major asset in landing on an entirely unknown target body.

LANDING SITE SELECTION

Eros is an elongated body with dimensions estimated to be 35 by 16 by 7 km; it rotates around the axis of its shortest dimension at a rate of one revolution per 5.27 hr. By landing at or near one of the poles as illustrated in figure 3, appreciable relative terrain motions in circumferential and radial direction are avoided. Other major advantages in selecting a polar landing site are these:

(1) The local gravity is greater than at the tips of the elongated body and centrifugal effects are minimized, hence the tendency to bounce off after landing is reduced (Staley, 1970).

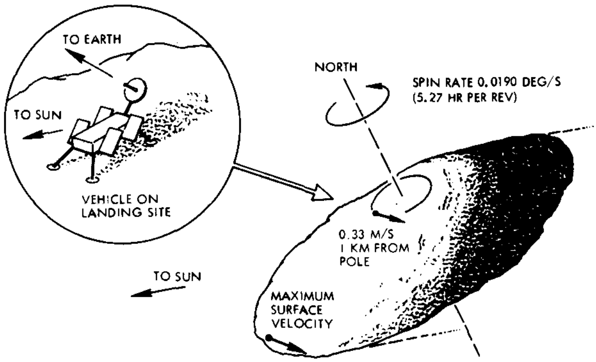

Figure 3.–Landing site selection on rotating small asteroid; assumed dimensions: 35 by 16 by 7 km. The body has the following estimated surface motion characteristics: maximum horizontal velocity = 5.8 m/s, maximum radial velocity = 1.6 m/s, maximum radial excursion = 9.5 km. The effective accelerations (cm/s^2) at the tip of the cigar-shaped body are the following (Staley, 1970): gravity, 0.264; centripetal, 0.192; downward, 0.072.

(2) For the same reason, sample material to be picked up at the landing site includes meteoroids that cannot collect at points of lower effective gravity.

(3) Day/night cycles are avoided by landing in a Sun-illuminated polar area. This simplifies power generation and minimizes thermal design problems.

(4) Uninterrupted communication with Earth is facilitated, as will be discussed below.

(5) The polar area provides greater visual contrast of surface features and smaller contrast fluctuation during the daily revolution than other areas, particularly at the time of arrival. This assists landing point selection and obstacle avoidance.

Present best estimates of the polar axis orientation of Eros are given by Vesely[6] as $13° \pm 3°$ in longitude and $28° \pm 1°$ in latitude relative to the ecliptic. (Earlier estimates gave the same mean values but included uncertainties of about $\pm 25°$). This information enables us to select in advance the polar region best suited for landing, given the reference trajectory shown previously in figure 1. The trajectory plot indicates the position of equinoxes and solstices of Eros derived from the estimated mean polar axis orientation. We note that the nominal arrival date of July 10, 1978, practically coincides with the estimated time of the asteroid's vernal equinox. This timing is quite unfavorable for a polar landing because of the uncertain illumination of the landing site, marginal availability of solar electric power, and marginal Earth

[6]See p. 133.

visibility for downlink telemetry. However, with time elapsing after equinox landing, conditions in the north polar region become favorable as the terminator recedes and the shadow-free area around the pole spreads out. Therefore, the vicinity of the north pole is clearly the preferred landing area.

A 20 day delay in arrival, previously suggested for improvement of the Earth-Sun separation angle, would allow the subsolar point to move about 18° north of the equator. This assures adequate illumination of the polar region, with 18° elevation of the Sun, and adequate downlink capability with 21° elevation of Earth above the horizon. At a 2 to 3 km distance from the pole these conditions are still approximately valid because of the very pronounced flattening of the figure of Eros. This is illustrated in figure 4 by contours of the shadow-free area around the pole at three dates past equinox, with the subpolar point 10°, 20°, and 30° north of the equator.

The uncertainty in the actual pole orientation of Eros requires a conservative landing date selection with a postequinox time margin of at least 2 to 3 weeks. There is even the possibility of postponing the landing maneuver for several weeks if necessary after arrival, and using the time interval for exploration of ambient conditions and remote observation of surface features. Electric propulsion provides easy and inexpensive maneuverability during this exploration phase. A particularly valuable task would be to monitor physical properties of the asteroid from a stationary position and to detect periodic variations with the changing relative orientation of the rotating asteroid body. This can be accomplished most effectively by hovering at a station close to the asteroid's equatorial plane. Comprehensive imaging of a large portion of the asteroid's body can thus be obtained. Certain quantities, e.g., magnetic field strength, solar-wind direction, or gravity gradient, might display repetitive

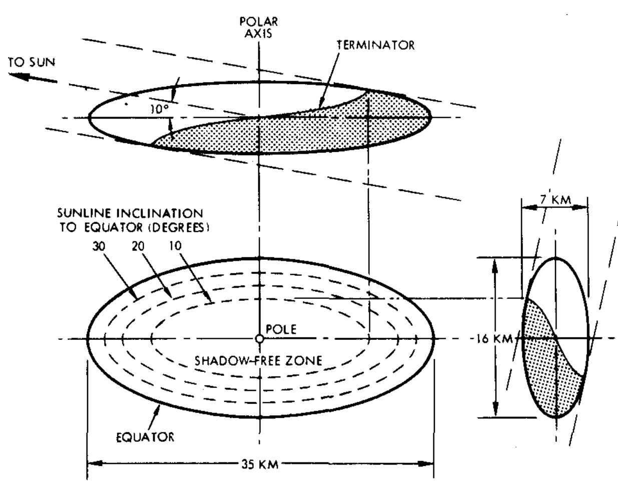

Figure 4.–Lighting geometry on ellipsoidal asteroid with assumed dimensions of Eros.

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signal patterns containing harmonics of the asteroid's rotational frequency. These would indicate the presence and approximate locations of magnetic and gravitational inhomogeneities.

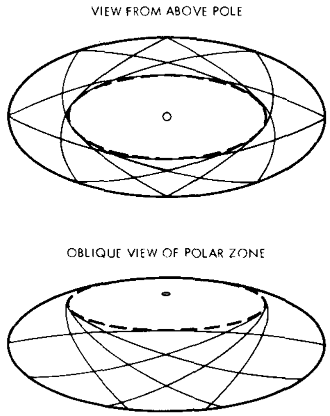

Figure 5.—Delineation of polar region by successive terminators.

Visual identification of the pole prior to selecting the site is a reasonably simple task using the polar illumination characteristics explained above. The spacecraft is targeted to arrive at a position from where it can observe the north polar region. An extended hover period lasting from one to several revolutions of the asteroid, with low thrust used for position control, is sufficient to obtain a sequence of TV images that permit reconstruction of the shadow-free envelope. The envelope is formed by successive terminator lines as illustrated in figure 5. The pole can be readily determined as the centroid of the shadow-free zone. Irregularities of the terrain can also be detected conveniently by this technique. Landmarks at the pole identified by contrasting features serve as a guide for the subsequent descent and landing phase.

FLIGHT SEQUENCE

Major events of the flight sequence were previously illustrated in figure 2. The vehicle is launched from the Eastern Test Range on a near-easterly azimuth and injected into an escape trajectory at a hyperbolic excess velocity of 3 km/s. After booster separation, the solar array paddles are deployed and the vehicle assumes cruise attitude with the Sun and a selected reference star providing three-axis orientation. The electric thrustors operate continuously during the 500 day transfer phase to Eros to provide the necessary velocity increment for a zero-velocity rendezvous. The thrust is oriented in forward direction approximately at right angles to the sunline with an out-of-plane component added to attain Eros' orbit inclination of 10°.8.

Guidance corrections are carried out by thrust angle adjustments on command from the Deep Space Instrumentation Facility (DSIF) ground station, which supports the mission by intermittent tracking and orbit determination. Typical guidance accuracies of 1000 to 3000 km relative to Earth coordinates are achievable; however, the uncertainty of the asteroid ephemeris, estimated to be at least 5000 km, necessitates the use of onboard terminal guidance sensors. An optical technique will be used that determines the orientation of the target line of sight relative to selected reference stars. Repeated fixes taken during the final 50 to 100 days before encounter permit updating the relative trajectory and performing terminal guidance corrections. A terminal guidance accuracy of 100 to 200 km can thus be achieved.

The transfer trajectory is aimed at a point located about 50 nominal asteroid radii (400 km) above the north polar region. The terminal approach and descent phase includes several extended hover intervals to permit observation by scientific instruments and reconnaissance by ground control via TV link. The descent sequence is illustrated in figure 6. As previously discussed, the initial approach might also include a hover period of several days for observation near the equator (not shown in the diagram).

The first hover phase at 400 km altitude is used to determine the pole position of the rotating asteroid from a sequence of TV frames. The vehicle then descends to a second hover position about 25 km above the pole, from which a higher resolution image of the pole region can be obtained. A preferred landing area can be selected at this time.

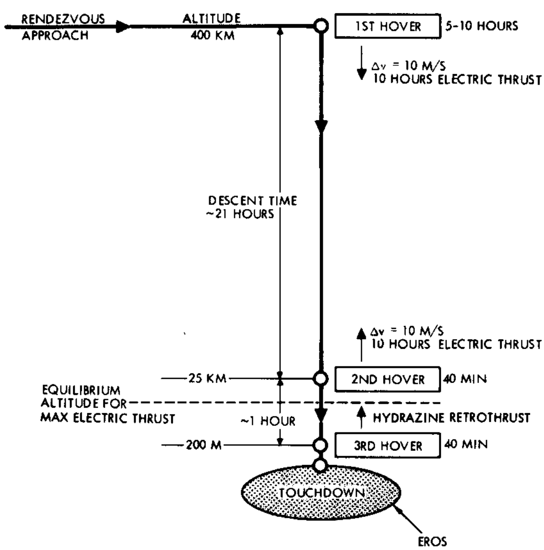

Figure 6.–Terminal approach and descent phases. Total descent time is 28 to 33 hr. A hover period of several days near the equator could also be included during the approach (as discussed in text).

Electric thrust is sufficient to hold the vehicle in stationary position against the small local gravity during hover phases 1 and 2. A set of small hydrazine rockets provides acceleration and deceleration on descent from hover position 2 to position 3.

At hover position 3, located 200 m above the terrain, a final check of the landing site is made to take corrective action for obstacle avoidance, if necessary. This 40 min hover phase requires use of the hydrazine rockets in an intermittent thrust mode. The final descent to touchdown ends in a free fall from about 50 m altitude, resulting in an impact velocity of about 0.7 m/s based on an assumed gravity acceleration of 0.5 cm/s^2.

Autonomous control of the descent phase velocity profile from an altitude of about 10 km is achieved by means of a radar altimeter and a three-beam Doppler radar system that operates in a manner similar to the Apollo lunar module landing radar. The threshold velocity detectable by this system is 1.5 m/s. The descent is in nearly vertical direction, which simplifies the onboard computation of velocity and attitude corrections. In addition to the radar system, the vehicle uses a vertical attitude gyro as a redundant attitude control reference.

The solar array is designed for retraction by the rollup storage mechanism. After completion of the final electric thrust operation, about 2 hr prior to landing, the solar array is retracted for protection against dynamic loads at impact but with a sufficient portion of array paddles protruding to provide about 200 W to operate housekeeping systems and the landing radar.

The liftoff and return flight sequence will not be discussed in detail. In principle, this sequence is a reversal of the outbound transfer but with the approach guidance to Earth made simpler by the absence of target ephemeris uncertainty and by the availability of a DSIF station on the ground.

VEHICLE CONFIGURATION AND DESIGN CHARACTERISTICS

Conceptual configurations of the solar electric spacecraft during cruise and after landing are shown in figure 7. The vehicle consists of a center structure that houses the electric-propulsion module, engineering subsystems, scientific instruments, sample collection tools, and the sample-return capsule. Attached to the center body are two pairs of lightweight solar array paddles that are deployed from storage drums in window-shade fashion by means of extendable tubular booms. The flexible landing gear consists of four legs having footpads lined with crushable material for absorbing impact energy as in the Surveyor spacecraft. Spring-released anchoring devices, not shown in the sketch, are used to secure the vehicle's position after touchdown under the extremely small surface gravity of Eros. This configuration is derived from an earlier solar electric spacecraft design study (TRW, Inc., 1970).

The electric-propulsion module, mounted opposite the payload bay, is shown in greater detail in figure 8. It consists of electric power conditioning units (PCU's), an array of electron-bombardment mercury ion thrustors

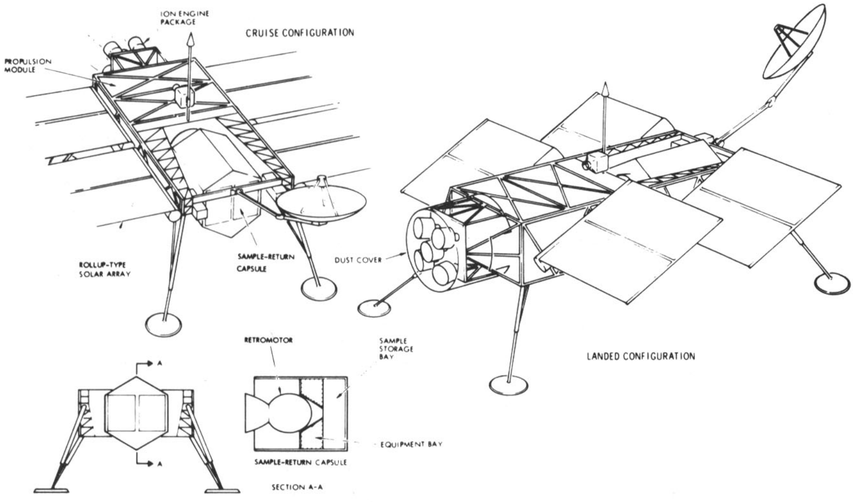

Figure 7.–Solar electric bus vehicle in cruise and landed configurations.

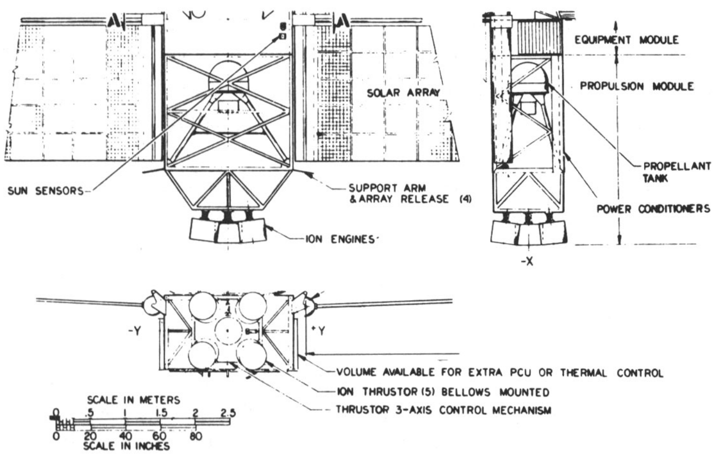

Figure 8.–Electric-propulsion module.

mounted on a flexible support fixture for thrust vector control, a propellant tank, and a feed system. The high density of the mercury propellant permits storage of a 750 kg propellant load in a 0.5 m diameter compact spherical tank located near the vehicle's center of mass. This minimizes the change of mass distribution and its effect on attitude control during cruise and landing. Five ion thrustors are provided, only two of which are normally in use during the outbound and inbound transfer phases. The three remaining thrustors serve as standby units to assure a high system reliability in a mission that requires a total thrust time of 1000 days. The delicate thrustor system uses a shroud for protection against dust stirred up during touchdown and surface operations. Dust covers must also be provided for the apertures of optical and other sensors.

The rollout solar array with each paddle measuring 2 by 14 m when fully extended provides 12 kW of initial power at 1 AU, 10 kW of which is used to operate the electric-propulsion system. The remaining power is used for housekeeping and telemetry and includes a 10 percent margin against contingencies such as solar array performance degradation due to solar flares. Prior to landing, the paddles are retracted for protection against the landing impact and flying dust. Subsequently, small paddle segments are extended to generate power of about 400 W for surface operations, housekeeping, and high-data-rate telemetry of TV pictures. After takeoff from Eros, the array is again fully extended for the return cruise.

As shown in the design illustration, the four solar paddles can be rotated around their deployment booms to improve array illumination primarily during the approach, descent, and hover phases, and after landing. During the transfer phase, small changes in solar array orientation relative to the center body are useful but not required, permitting an additional degree of freedom for optimum thrust vector pointing. During the landed phase, array reorientation may be required to accommodate changes in Sun elevation. By splitting the array into four narrow paddles instead of two, field-of-view obscuration of the optical sensors and the high-gain antenna due to paddle reorientation can be avoided.

These deployment and orientation sequences are compatible with design specifications of the rollout array shown in the design illustrated. A full-scale 2.5 kW engineering model of this array design has been developed for NASA by General Electric and has successfully completed extensive electrical, thermal, and mechanical tests in the laboratory.

A set of four differentially throttlable hydrazine thrustors, each with a maximum thrust level of 4 N (1 lb), are centrally mounted on the underside of the center body. They provide thrust required during the descent and ascent phase and support the vehicle during the extended final hover phase prior to touchdown. With an assumed small surface gravity of 0.5 cm/s^2, the total hydrazine propellant consumption for a 40 min hover period is 16 kg for a vehicle of 1400 kg gross mass. The total maneuver sequence performed by the hydrazine thrustors requires about 40 kg of propellant, equivalent to 60 m/s of Δv expenditure.

A variety of optical sensors are required to control vehicle operations on approach to the asteroid, during the final descent phase, during landed operations, and during ascent. The use of a TV system is required both for the vehicle control operations and the scientific observations that follow:

(1) Approach guidance
(2) Reconnaissance and landing site selection

(3) Obstacle avoidance by command control from Earth
(4) Remote manipulation on the surface under TV monitoring
(5) Surface panoramic view and detail feature observation

These operations will require wide-angle and narrow-angle, high-resolution TV image systems. Three or more separate TV systems mounted on two-axis gimbal platforms with different and overlapping fields of view are envisioned. For landing site inspection and selection an image system with at least 0.0014° resolution will be required comparable to the planetary image system of the TOPS spacecraft.

The attitude reference sensors include fixed, fine and coarse, Sun sensors, and a one-axis gimbaled star sensor that uses an electronic image scanning principle similar to the Mariner star tracker. This device permits use of alternate reference stars that come into view at different times in the mission and can be tracked without interference by the long solar array paddles and other deployed appendages (Meissinger and Benson, 1970). An additional optical sensor is required for locking the high-resolution TV camera on the asteroid during the approach and hover phases when successive TV frames of the polar area must be taken for determining the location of the pole and selecting a landing site.

The communication system operates on S-band and uses design principles developed for and successfully used on Pioneer spacecraft. The 2.4 m diameter parabolic high-gain antenna dish is mounted on a hinged deployment arm that permits Earth pointing in all directions relative to the vehicle body without obstruction by other deployed appendages, using a two-axis rotation joint. The same deployed configuration is used during cruise and landed operations. In addition to the high-gain antenna, a pair of omniantennas is provided, one on each side of the center body, to maintain an uplink command capability in all vehicle attitudes.

A 100 W solid-state transmitter composed of four parallel 25 W channels provides incremental power options desirable for flexible use of the telemetry system. Low telemetry power and bit rates are used at times when power is needed primarily for propulsion purposes. High bit rates are available for telemetry of TV images during critical mission phases. Table I lists representative bit rates and communication intervals per TV frame at 2.1 AU communication range for 25 and 100 W of transmitter power with ground coverage by 25.9 and 64.0 m DSIF antennas. The effect of TV image data compression by a 2:1 ratio is also shown. The unprocessed TV image is assumed to contain 2.5×10^6 bits. Bit rates of 4096 bps are available for telemetry to the 25.9 m ground station and 65 536 bps to the 64.0 m ground station using 100 W of transmitter power. This corresponds to telemetry intervals of 610 and 38 s per TV frame without data compression and to 305 and 19 s with data compression, respectively. The lower time intervals are quite small compared to the 17.5 min one-way transmission time delay from Eros. A principal advantage of the electric spacecraft is the ability to meet high data

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TABLE I.—*Eros-to-Earth Communication Data Rates*

Downlink option		Data rate, bps	Single TV frame transmission time, s	
Transmitter power, W	DSIF antenna, m		Unprocessed data	With 2:1 data compression
25	25.9	1 024	2440	1220
100..................	25.9	4 096	610	305
25.....................	64.0	16 384	152	76
100..................	64.0	65 536	38	19

Average communication range during stopover: 2.1 AU; S-band telemetry; 2.4 m spacecraft antenna; TV frame contains 2.5×10^6 bits (unprocessed).

rate requirements without demanding coverage by the 64.0 m DSIF station, owing to the large unused power capacity available for telemetry during critical mission events.

The following list is a summary of system characteristics based on performance data from Mascy and Niehoff:[7]

(1) Launch vehicle: Titan IIID/Burner II
(2) Launch date: February 25, 1977
(3) Arrival at Eros: July 10, 1978
(4) Return to Earth: January 12, 1980
(5) Round-trip time: 1050 days
(6) Stay time: 50 days
(7) Propulsion power at 1 AU: 10 kW
(8) Mercury ion thrustors: five (two plus three spares)
(9) Specific impulse I_{sp}: 3000 s
(10) Peak power to thrustor: 4.6 kW
(11) Maximum thrust force per thrustor: 0.214 N (48 mlb)

Mass estimates of the electric bus vehicle and the sample-return capsule, also based on the performance data of Mascy and Niehoff, are given in table II. By holding the return sample mass to 100 kg, a 10 kW bus vehicle launched by Titan IIID/Burner II has adequate performance margin. A much larger sample mass of 200 to 300 kg can be returned by using a higher powered bus vehicle (15 kW), which would require the more costly Titan IIID/Centaur booster.

We conclude this discussion with a chart (fig. 9) showing the current status of critical subsystems that are required to develop the electric bus vehicle for this mission. In all categories except the solar array, a first generation subsystem with adequate performance to achieve the mission has been flight proven and would be ready for application to the system. Technology

[7]See p. 524.

TABLE II.–*Mass Estimates*

System	Mass, kg
Vehicle:	
Earth departure	1760
Eros arrival	1415
Earth approach	1000
Sample-return capsule in Earth orbit	380
Sample material	100
System breakdown:	
Bus vehicle plus capsule at Earth departure:	
Solar electric propulsion	300
Low-thrust propellant	520
Hydrazine propellant	40
Structure and subsystems	420
Science instruments	200
Return capsule (includes retropropellant, no sample)	280
Total	1760
Capsule (including sample):	
Structure and mechanisms	60
Subsystems and sample storage	120
Retropropellant	100
Sample material	100
Total:	
Before retro	380
After retro	280

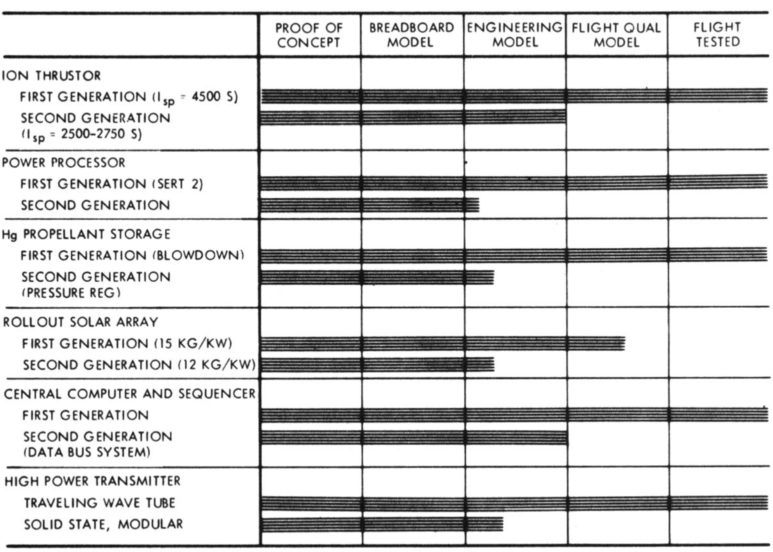

Figure 9.–Technology status of critical subsystems as of January 1971.

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improvements obtainable from second generation subsystems will add performance gains that are, however, not critical to mission accomplishment. As seen in the chart, the improved subsystems are well along in their development toward flight application.

CONCLUSION

Preliminary analysis and conceptual design study of a solar electric bus vehicle for an Eros sample-return mission show that no major obstacle exists today in terms of technical feasibility, design approach, and operational concepts to early adoption of a program aimed at exploring nearby asteroids such as Eros and returning soil samples. Solar electric propulsion provides basic advantages in payload capacity, mission flexibility, and operational convenience needed to make such a mission more cost effective, more reliable, and more exciting from a scientific exploration standpoint. However, more detailed study of vehicle design, mission implementation, mission timing, performance tradeoffs, and cost factors are required to further substantiate these predictions. It appears that even with an early start of such a program it would not be realistic to expect to meet a launch date prior to the 1977 opportunity. Subsequent launch opportunities for Eros sample-return missions occur about every 2 yr. These opportunities as well as missions to other nearby asteroids require further study.

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