ASTEROID RENDEZVOUS MISSIONS

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The earliest flights to concentrate solely on the nature of particular asteroids will probably be of the simple flyby or rendezvous type. That is, data will be obtained by close investigation of these bodies without actually coming in contact with their surface. The technology of flyby missions to bodies at planetary distances is well established; there have been five successful Mariners to date and more flights are scheduled. Unfortunately, ballistic flyby missions suffer from the fact that the spacecraft is near the body of interest for a short period of time (relative velocities at asteroid encounters are 5 to 12 km/s). This problem is acute because spatial resolution of the onboard instruments must be very high, and typical flyby velocities will preclude detailed observations.

These problems lead to consideration of *rendezvous* missions wherein the spacecraft is placed into the same heliocentric orbit as the asteroid and therefore remains close to it over a long period of time. This will allow long-term observations of the body over a range of aspects, distances, and phase angles. Probably most of the surface would be available for observation by this technique.

There are two principal means of achieving rendezvous and these are classified as "ballistic" (or "impulsive" or "high thrust") and "low thrust." In the first method, the spacecraft is given substantial velocity changes at various points in its flightpath and travels ballistically in between them. These velocity changes occur upon leaving Earth, upon arriving at the asteroid, and possibly at one point in between. They are imparted by a conventional rocket engine and occur over a very short time compared with the total flight, hence the term "impulsive." The other method is to continuously thrust the vehicle over most of its flight with a high-specific-impulse, low-acceleration engine. The current concept for doing this uses solar electric propulsion wherein solar power is converted to electricity that drives electron-bombardment mercury-ion thrustors.[1] This requires very large lightweight solar arrays. These engines operate for nearly the whole flight and are directed in such a way as to continuously and optimally change the orbit of the spacecraft until it coincides with the orbit of the asteroid. One concept for a solar electrically powered spacecraft suitable for an asteroid rendezvous is shown in figure 1. The technology of

[1]See p. 491.

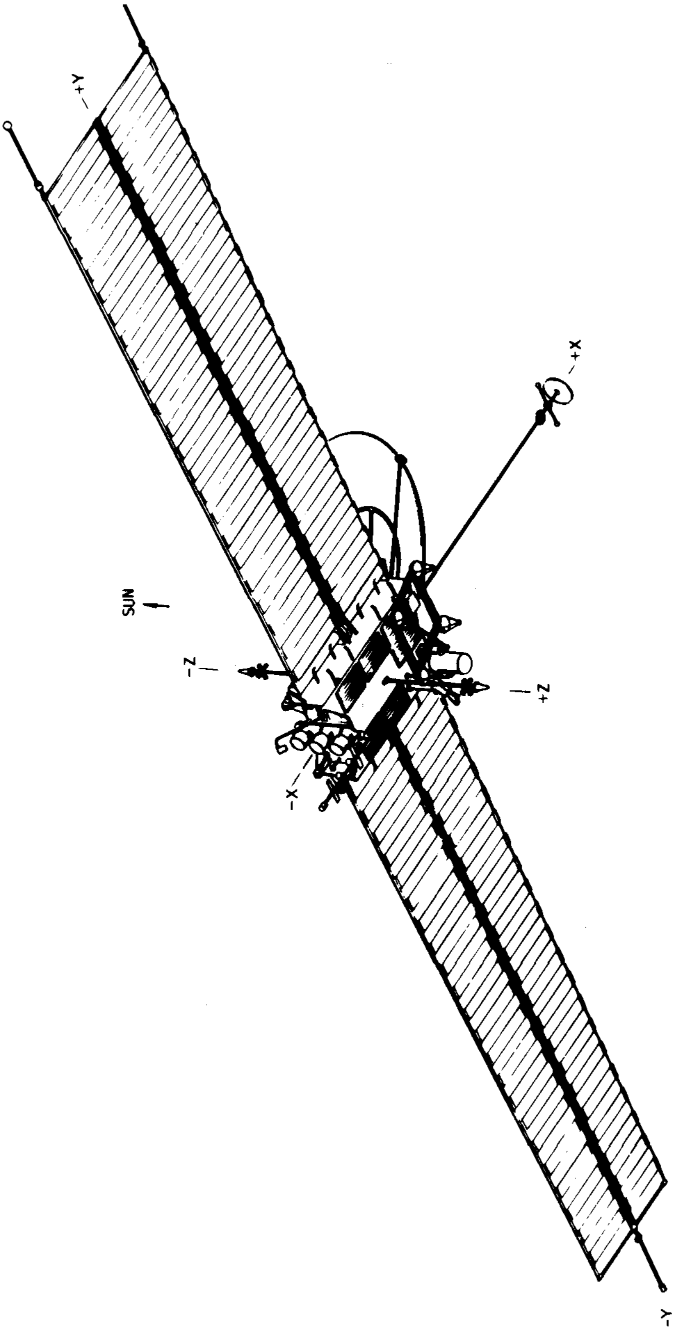

Figure 1.–Typical solar electric spacecraft. Gross power level = 6.4 kW; spacecraft gross mass = 580 kg.

these vehicles is reviewed by Bartz and Horsewood (1969). This review includes an extensive bibliography.

The remainder of this paper deals with the rendezvous by means of a low-thrust, solar electrically propelled vehicle because (1) this approach is generally better from a payload and flight-time basis compared to the high-thrust method and (2) it has a large degree of flexibility and can be applied to a variety of missions. It should be emphasized that here we are concentrating on rendezvous missions only, which are necessary precursors to the more complex landing or sample-return missions. The remainder of the paper deals with our results to date.

ASTEROID SELECTION

In choosing an asteroid for a mission target, one would first assume that size would be the major criterion and pick the largest one because it would present the largest surface for study. Thus a mission to Ceres would be studied at the outset as well as missions to the other asteroids if they are significantly less demanding in terms of energy and/or flight time. Such other possible targets may traverse the main asteroid belt near Ceres but have smaller inclinations, or else they may have smaller orbits passing inside that of Mars. In addition to Ceres, we have considered here missions to three bright asteroids with inclinations successively smaller than Ceres and to one with a significantly smaller orbit than Ceres as illustrated by the orbital elements of table I. Pallas and Juno were not considered because of their high inclinations of 34°.8 and 13°.0, respectively.

To estimate flight times for transfers to be accomplished in less than one revolution about the Sun, we indicate the value of the trip time T_H for a Hohmann transfer from Earth to a circular orbit at the radius of the semimajor axis. This serves as a guide because the Hohmann transfer, which uses a heliocentric central angle of 180°, is an optimal impulsive transfer between two circular orbits. It generally occurs, however, that optimal low-thrust rendezvous trajectories utilize central angles considerably greater than 180° so that flight times can be expected to be 20 to 50 percent greater than T_H.

TABLE I.–*Orbital Properties of Selected Asteroids*

Asteroid no.	Name	i	a, AU	e	T_H, days	Synodic period, yr
1	Ceres	10°6	2.77	0.076	473	1.28
4	Vesta	7.1	2.36	.089	398	1.40
10	Hygiea	3.8	3.15	.100	546	1.22
20	Massalia	.7	2.41	.143	407	1.37
433	Eros	10.8	1.46	.223	249	2.29

T_H = time for Hohmann transfer from r = 1 AU to $r = a$.

Fortunately, for solar electric propulsion, the choice of flight time is not particularly critical for determining mission feasibility.

RESULTS

Launch Date Selection

Ranges of possible launch dates to be used as initial estimates of detailed searches may be found using a simple procedure that is described below. We first replace the orbits of Earth and the target asteroid by circular coplanar orbits, and we choose the time of flight and central angle to be traversed. The procedure is illustrated in figure 2, which shows launch date possibilities for Vesta from 1975 to 1977. We plot the mean anomaly of Vesta M_2 versus the mean anomaly of Earth M_1 as it actually occurs for the launch years we wish to consider. That is

$$M_2 = \frac{n_2}{n_1} M_1 + \text{const} \tag{1}$$

Using the time of flight t_f and the central angle $\Delta\theta$ to be traversed on the transfer orbit, we plot the mean anomaly on the target orbit M_2 at the time of departure versus the mean anomaly on Earth orbit M_1 at the time of departure. The relationship to be plotted is obtained by expressing the longitude of the arrival point in terms of the motion of the spacecraft along the transfer orbit and the motion of the target. Thus we obtain

$$M_1 + \lambda_{p_1} + \Delta\theta = M_2 + \lambda_{p_2} + n_2 t_f \tag{2}$$

or

$$M_2 = M_1 + \lambda_{p_1} - \lambda_{p_2} + \Delta\theta - n_2 t_f \tag{3}$$

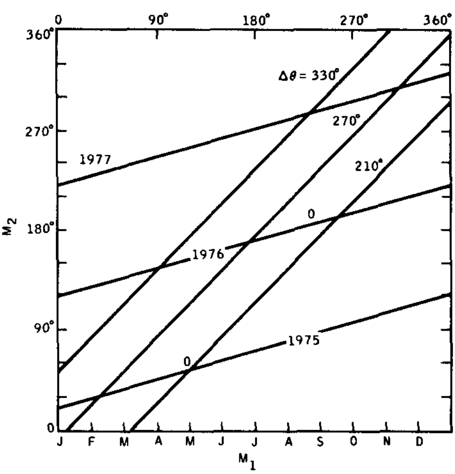

Figure 2.—Vesta launch possibilities (1975-77), 500 day flights.

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where λ_p is the longitude of perihelion and n is the mean orbital rate. Intersections of lines for equations (1) and (3) indicate values of M_1 for which launches meeting the requirements chosen for t_f and $\Delta\theta$ selected are possible. The corresponding calendar date can be read from the abscissa.

We may choose a different value for $\Delta\theta$ and plot another curve of M_2 versus M_1 thus obtaining a range of launch dates for the chosen range of central angles. Having found a reasonable range of launch dates, we make a search over that range by means of a computer program that determines payloads and accurate trajectory characteristics for the particular type of mission under consideration. The optimum launch dates found for the solar electric-propulsion missions are indicated by an O.

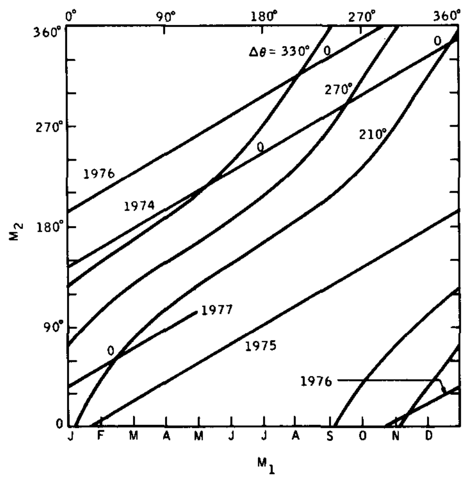

Figure 3.–Eros launch possibilities (1974-77), 360 day flights.

Equations (2) and (3) can be modified to be more representative of the true situation by including the effect of the eccentricities of the two orbits. True anomaly intervals are used to obtain the longitude at arrival and thus M_2 versus M_1. The relation is now a curve that oscillates about the straight line and must be computed for each case of $\Delta\theta$ and t_f desired. Such a graph is shown for Eros for 1974 to 1977 in figure 3.

Mission Data

For each asteroid chosen, the low-thrust program CHEBYTOP (Han, Johnson, and Itzen, 1969) was used in a launch-day scan mode to locate the departure data that resulted in greatest payload for a given flight time. The launch vehicle chosen was the Titan IIID/Centaur, except for Eros in which the Atlas/Centaur was used. The remaining spacecraft and launch parameters required as input for Ceres, Vesta, Massalia, and Hygiea rendezvous are as follows:

(1) Hyperbolic departure velocity at Earth, ν_{hl} = 5 km/s
(2) Total mass injected (at ν_{hl} = 5 km/s), 3450 kg
(3) Thrustor specific impulse, I_{sp} = 3500 s (exhaust velocity c = 34.3 km/s)
(4) Tankage factor, 6 percent (propellant and tank mass = 1.06 × mass of propellant needed)
(5) Engine efficiency, 0.667
(6) Propulsion system specific mass, 30 kg/kW

The power level at 1 AU P_0 is calculated by the program, but the payload results quoted have been scaled to correspond to a power level of 16 kW.

The optimum hyperbolic escape velocity depends on the size of the target orbit, and for Eros it is very close to 2 km/s. With a Titan IIID/Centaur launch vehicle, payloads for Eros missions would be more than 2000 kg, which is considered to be more than necessary for preliminary missions. For this reason, the smaller Atlas/Centaur launch vehicle, which results in a total injected mass of 1000 kg at ν_{hl} = 2 km/s, was used. Otherwise the values listed pertain also to Eros missions.

Mission data results are presented in table II. Here flight time, payload, and encounter geometry are shown for at least two launch dates for each asteroid in the period from 1974 to 1977. It is interesting to note that inclination does

TABLE II.–*Solar Electric Asteroid Rendezvous Missions (Launch Vehicle Titan IIID/Centaur)*

Asteroid	Estimated window	Launch date	Flight time, days	Payload, kg	Encounter conditions: Solar distance, AU	Encounter conditions: Earth-Sun-asteroid angle	Encounter conditions: Earth asteroid distance, AU
Ceres	7/75-9/75	7/20/75	600	588	2.58	11°	1.61
	10/76-12/76	11/21/76	600	643	2.92	6	1.91
Vesta	2/75-5/75	4/22/75	560	795	2.55	30	1.76
	6/76-9/76	8/11/76	560	476	2.18	68	2.03
Hygiea	7/75-9/75	9/4/75	650	204	2.94	80	2.94
	10/76-12/76	11/6/76	650	450	2.89	50	2.37
Massalia	2/74-4/74	2/6/74	500	358	2.70	40	2.04
	5/75-7/75	5/15/75	500	553	2.10	69	1.98
Eros	4/74-2/75	6/18/74	360	[a]580	1.47	36	.88
		12/7/74	360	[a]534	1.78	129	2.52
	8/76-2/77	2/14/77	360	[b]485	1.55	140	2.40

[a]Atlas/Centaur launch vehicle (fully optimized trajectory data) ($P_0 \leqslant$ 10 kW).
[b]Atlas/Centaur launch vehicle (P_0 = 10 kW).

not have a significant effect on payload over the range selected. Rather the payloads fluctuate a great deal because of the varying relationships between relative orbit nodes and perihelion from one launch opportunity to the next. It is clear that missions to any of these five asteroids (and of course many others) are possible with solar electrically propelled spacecraft launched by a Titan IIID/Centaur or an Atlas/Centaur and that a substantial scientific payload can be delivered.

Eros Trajectory

To determine requirements on the design of an asteroid rendezvous spacecraft, some details of a flight to Eros have been generated. The selected launch date[2] is in December 1974 and the launch vehicle postulated is the Atlas/Centaur. The spacecraft is equipped with the solar panels developing 10 kW at 1 AU and, except for a 10 day coast period in the middle of the flight, the engines thrust continuously for the 360 day trip to Eros.

This combination of launch vehicle, power, and flight time allow a net spacecraft mass at the target in excess of 500 kg. This includes 50 to 100 kg of science instruments. A breakdown of the spacecraft mass at launch is given in table III.

Figure 4 shows the relative positions of Earth, the spacecraft, and Eros projected into the ecliptic for various dates along the trajectory. After May 1975, the spacecraft and Eros are in the same position to the scale of this drawing, but a good deal of their separation is normal to the ecliptic and hence does not show in this projection.

The most vital part of the flight from the standpoint of scientific return begins when the spacecraft approaches the asteroid. In this phase there are a number of problem areas, two of which are discussed below.

To begin with, the position of the asteroid is not sufficiently well known for spacecraft navigation purposes. We expect an a priori error on the order of a few thousand kilometers. Because this is much larger than the intended closest approach distance to Eros (which is on the order of the dimensions of Eros

TABLE III.–*Spacecraft Mass Budget*

Component	Mass, kg
Power and propulsion system (10 kW)	300
Mercury propellant	150
Net spacecraft (including scientific payload)	500
Gross spacecraft mass at launch	950

[2]There are no firm NASA plans to launch an Eros mission at this date; it is used for example purposes only.

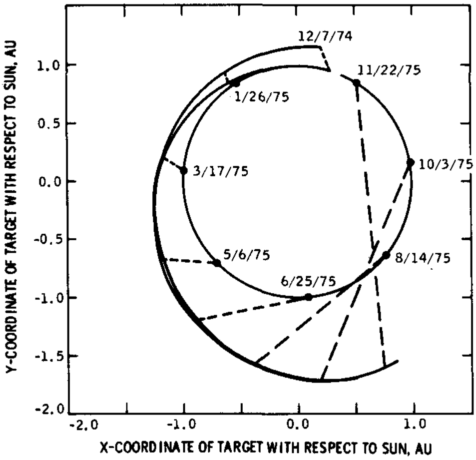

Figure 4.–Eros rendezvous ecliptic plane projection. Thrust is fixed at 90°, P_0 = 10 kW, I_{sp} = 3500, and t_f = 360 days. Launch date is December 7, 1974. The launch vehicle is the Atlas/Centaur.

itself), the error must be corrected. Before a flight, the ephemeris of the target would be improved to the largest extent possible by ground-based observations.[3] Further improvement can be made by correcting the ephemeris of the target by using optical observations taken from the spacecraft while in flight. This procedure was tested on Mariners 6 and 7 and it is scheduled to be operationally demonstrated on the Mariner-Mars 1971 orbiter arriving late this year. It is a vitally necessary component of the Outer Planet Grand Tour Flight.

Using typical optical system performance, it should be possible to begin resolving Eros when the spacecraft is 200 000 km away although it may be visible as a bright object well before this. The position of the asteroid against the star background, when combined with Earth-based radio-tracking data from the spacecraft, provides sufficient information to determine the spacecraft's position relative to the asteroid. The information can then be used to reprogram the thrust history for the remainder of the flight, thereby correcting any errors.

Another problem that arises is due to the fact that the thrust beam must be directed nearly toward the asteroid near the end of the flight. This can be seen from figure 5, which shows the asteroid-spacecraft-thrust beam angle as a function of date throughout the mission. Note that the angle goes to zero as the spacecraft approaches the asteroid. (This phenomenon is characteristic of any rendezvous trajectory because the spacecraft must slow down with respect to the target on the approach.) Thus observations from the spacecraft during

[3]See p. 639.

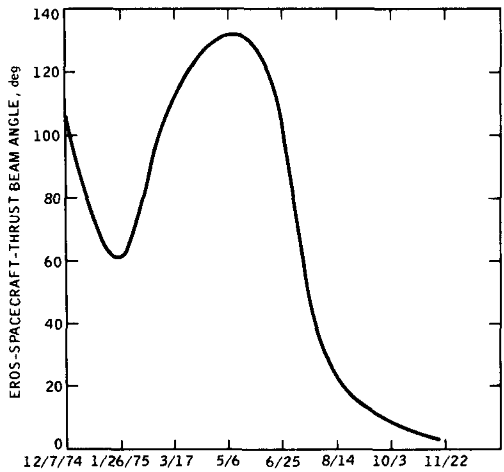

Figure 5.–Eros rendezvous approach conditions.

this phase would have to be made through the exhaust. Two possible solutions to this problem are (1) to momentarily cease thrusting on the approach and reorient the spacecraft to allow observations or (2) to initially rendezvous with a point some distance from the asteroid, take the appropriate observations, then slowly close in on it. These and other methods are currently under study.

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