stable transponder on the second element can return the transmitted signal to the first element for comparison. An integration time of 1 sec or less then suffices to determine the phase shift rate (Doppler shift) between transmitted and received LO signals. This information, retransmitted to the second element, then can insure coherence of the two LO signals. The IF signal from the second element can also be transmitted to the first element, corrected for phaseshift in the same way, and correlated in real time.

Thus, coherence times of ~ 1 sec will suffice; the quartz oscillators previously noted will permit rms phase errors of $< 10^{-2}$ rad per GHz. Moreover, orbit determination or on-board ranging can insure knowledge of the separation to a fraction of a kilometer, for the time delay needed for synchronization of the IF signals in processing. This approach offers solutions to the problems of synchronization, phase stability, and signal processing.

System Growth

An orbital VLBI system could be built in stages, to match the available budget and to support a phased experiment program.

First, a single antenna would be orbited. It would be a major radio telescope in its own right, providing experience in space operations of such a structure. It could also serve as one element of an interferometer, the other element being Earth-based, thus meeting the 1970 recommendation of the Space Science Board for use of a satellite for VLBI.[9] This would permit precision orbit determination, for geophysical and geodetic applications.

Next, the second antenna would be orbited, completing the basic system, which would be used in source mapping at extreme resolution. However, it probably would have limited usefulness for astrometry or source position determination. These latter applications require precision determination of the phase of the interference pattern, a quantity difficult to measure directly. But it can be measured, using a second antenna at each interferometer element, to observe a bright source (such as quasar 3C 273) as a phase reference.

Thus, the next stage in system growth would add another antenna at each element, with the element LO generator serving both antennas. Precision astrometry could then be conducted, as well as tests of general relativity involving gravitational bending of electromagnetic waves.

Finally, even higher resolution might be required. This could be achieved by improvements in paraboloidal surface shaping, permitting operation at shorter wavelengths, or by using higher orbits for longer baselines. With $\lambda = 1$ cm in geosynchronous orbit, resolution exceeds 10^{-5} arcsec, still limited by diffraction and not by interplanetary scintillations.[10]

Conclusions

A system such as the one described would offer many advantages. As a dedicated system, it would supplant present-day ad hoc VLBI experiments. As a space-based system, it would permit complete aperture synthesis by two elements, avoiding the administrative difficulties of many antennas in different nations, as required by an Earth-based dedicated system. Real-time signal processing would simplify experiments and permit adaptive use. State-of-the-art technology in structures and electronics would keep costs down, and the system could be built up in phases matched to science budgets and experiment programs.

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Comet Encke Flyby—Asteroid Rendezvous Mission

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Introduction

THE scientific interest in comet and asteroid missions has been growing steadily over the past several years. This is due, in large part, to the activities of the NASA-sponsored science working groups and study panels.[1–3] The discovery this year of comet Kohoutek and the flurry of activity focused on observing this comet has certainly raised the level of interest. A final factor is the advent of advanced propulsion technology, particularly solar electric propulsion (SEP), which will make possible future rendezvous, docking, and even sample-return missions to small-body targets.

The current consensus among scientists and mission planners is that comet Encke will be the primary target of early cometary exploration in the 1980 decade.[4] A two-mission sequence is planned which encompasses a flyby of Encke at its 1980 apparition (perihelion passage) to be followed by a rendezvous in 1984. Several preliminary design studies of the flyby mission are now underway. These studies are expected to provide the necessary tradeoff data from which NASA can make a more definitive selection of mission/spacecraft mode and science payload, should the flight project be approved. The three principal mission modes under consideration are: 1) a short (90 day) ballistic transfer utilizing a modified Helios spacecraft with encounter near Encke's perihelion at a flyby speed of 7–9 km/sec; 2) a short ballistic transfer utilizing a modified Pioneer-Venus spacecraft with encounter 16–26 days before perihelion at a flyby speed of 18–26 km/sec, and possibly retargeted after Encke encounter to flyby the asteroid Geographos; and 3) a long (650 days) low-thrust transfer utilizing a 10–15 kw SEP spacecraft with encounter 20–30 days before perihelion at a slow flyby speed of 4–5 km/sec and possibly pretargeted to flyby an asteroid prior to Encke encounter.

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Attractive features of the SEP mission mode include the probable enhancement of science value due to the slow flyby speed, and the operational flight test of the SEP system as a precursor to the follow-on rendezvous mission to Encke. The main drawback is the inherently higher risk of mission failure associated with the new SEP technology. Furthermore, programatical constraints could force a delayed project start putting the necessary Jan.–March 1979 launch period in jeopardy. If this should prove to be the case, are there viable alternatives for a SEP spacecraft launched on a 1980 Encke flyby mission?

The purpose of this Note is to describe one such alternative. Basically, it is a multitarget mission mode which utilizes the SEP capability to rendezvous with an asteroid after the encounter with Encke. We could define this mode as a "no-risk" Encke flyby mission relative to SEP technology. Launched in mid-1980, the Earth-Encke transfer is all-ballistic, and SEP operation begins after comet encounter and is relied upon only to accomplish the secondary target objectives. The following discussion is based on an exploratory analysis and is therefore limited in scope to a description of trajectory profile and spacecraft mass characteristics.

Results

Encke has an orbital period of 3.3 yr, is inclined 11.9° to the ecliptic plane, and passes through a perihelion distance of 0.34 AU on Dec. 6, 1980. The launch period for short ballistic transfers lies in Aug. of 1980 near the ascending nodal longitude of Encke. As the encounter date varies between Nov. 6 and Dec. 6, launch energy C_3 increases from 42 to 97 km^2/sec^2, while flyby speed decreases from 27 to 7 km/sec. Such transfer arcs have a 1 AU aphelion distance, a perihelion distance between 0.73 and 0.34 AU, and are inclined about 12° to the ecliptic. It is this type of orbit which must be reshaped to intercept a second target of opportunity after Encke encounter.

Geometrical considerations lead one to the conclusion that the asteroid target should be chosen either from the Amor group (Mars crossing orbits) or the Apollo group (Earth crossing orbits). This choice tends to ensure flight times which are not excessively long and minimizes the propulsion energy needed to achieve rendezvous conditions. Upon examining the time-position characteristics of particular bodies in these groups, two asteroids were selected for investigation: Eros (433) and Geographos (1620). The orbit of Eros is inclined 10.8° and has perihelion and aphelion distances of 1.13 and 1.78 AU. Similar data for Geographos are 13.3°, 0.83 AU, and 1.66 AU. Both bodies appear to have small characteristic dimensions (2–35 km), are elongated in shape like a football or a cigar, and have rotational periods on the order of several hours.

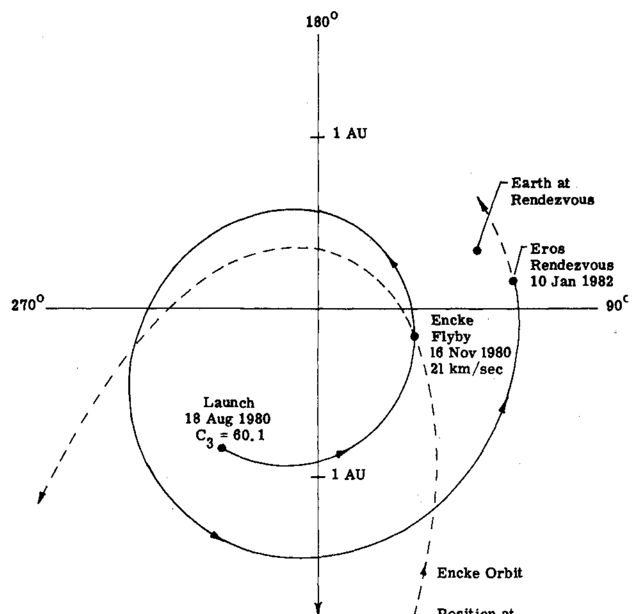

Fig. 1 Trajectory profile for 1980 Encke flyby—Eros rendezvous mission.

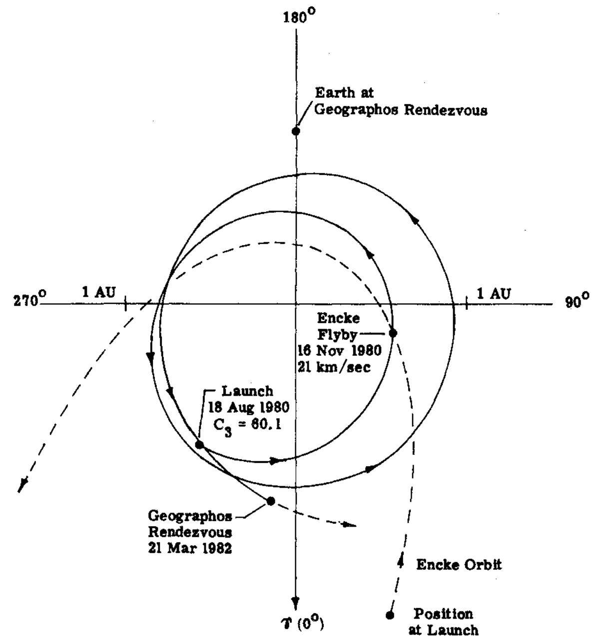

Fig. 2 Trajectory profile for 1980 Encke flyby—Geographos rendezvous mission.

Figure 1 illustrates a typical trajectory profile (ecliptic plane projection) for the 1980 Encke flyby–Eros rendezvous mission. Launched on Aug. 18 with escape energy $C_3 = 60$ km^2/sec^2, the spacecraft encounters Encke 20 days prior to its perihelion passage at a heliocentric distance of 0.6 AU and a relative flyby velocity of 21 km/sec. Earth is at 54° longitude at this time and is therefore in a good position for communications and correlation of ground-based and spacecraft science measurements. The SEP thrust program initiated after Encke flyby shapes the subsequent trajectory once around the sun for rendezvous with Eros on Jan. 10, 1982. Note again that Earth is in a very favorable position at the time of rendezvous operations. Typical thrust on-time is 350–400 days over the 420-day second-leg transfer to Eros. The interspersed coast periods, particularly near rendezvous, will aid the attainment of high navigation accuracy. Also, it should be noted that the optimum thrust direction angle relative to the sun remains close to 90° during the entire flight; this implies a desired simplification in thrust vector control mechanization relative to solar array pointing requirements.

Figure 2 shows a typical trajectory profile for the Encke flyby-Geographos rendezvous mission. The Earth-Encke ballistic transfer is the same as before. In this case the asteroid's time-position characteristic is such that the SEP spacecraft must "bide its time," setting up the rendezvous conditions over nearly 2 revolutions around the sun. The Encke-Geographos flight time is 490 days with rendezvous occurring on Mar. 21, 1982 after Geographos has passed through perihelion and is at 1.2 AU from the sun. Unfortunately, the Earth is in near-conjunction during this time which means that ground-based communications and observation geometry is relatively unfavorable compared to the Eros mission. Longer coast periods can be specified for the Geographos mission with a typical thrust on-time of 250–300 days. Thrust pointing relative to the solar direction varies over a wider range, 55°–120°.

As it turns out the payload delivery capability of SEP is nearly the same for either asteroid target. Performance results are summarized in Fig. 3 where the Titan IIIE/Centaur/BII launch vehicle and a 11 kw SEP powerplant are assumed. The initial mass corresponds to the maximum launch vehicle performance as determined by the ballistic C_3 requirements, less a 10% penalty to account for a launch window of about 2 weeks. With the thrust subsystem operating at a specific impulse of 3000 sec and a total efficiency of 64%, the mercury propellant requirement varies from 370 kg to 465 kg over the 10-day Encke encounter window shown. The propulsion system is comprised of the solar arrays, power conditioners, thrusters and thrust vector control mechanisms, and is estimated to weigh 330 kg for the 11 kw system. Delivered "payload" or net spacecraft mass is given by the lower curve in Fig. 3; this would be comprised of the science experiments ($\sim$60 kg) and the nonSEP functional support subsystems such as communications, data handling, thermal control, integrating structure, etc. One possible vehicle configuration would have the SEP Module attached to a 3-axis stabilized spacecraft based on Mariner and/or Viking technology. A net mass requirement of about 600 kg can be expected. This effectively constrains the Encke encounter date to be no later than 17 days before perihelion with a resultant minimum flyby speed of about 19 km/sec. It is important to note that the Titan/Centaur launch vehicle may be utilized without the BII kick stage since the C_3 requirement is sufficiently low in this region of constraint.

Increased payload performance would be possible if the Shuttle/Centaur vehicle is available for a 1980 launch. This is indicated by the performance map for the Encke-Eros mission shown in Fig. 4. Net mass curves are linear functions of SEP power rating assuming a constant value of propulsion system specific mass (α is taken as 30 kg/kw). Optimum SEP power is 10–11 kw for the Titan/Centaur launch vehicle as shown by its maximum performance constraint boundary. In the case of Shuttle, optimum power is 16–18 kw which yields a net mass increase of 60%–70%. Hence, Shuttle availability would provide a good measure of payload design margin and would allow later Encke encounters up to about 12 days before perihelion.

The pertinent conclusions from this analysis are: 1) an attractive multitarget mission alternative exists for Encke 1980 exploration; 2) SEP technology would be employed, at virtually no risk to cometary objectives, to rendezvous with an asteroid after Encke encounter; 3) of the two asteroid targets studied, Eros offers the better mission profile; 4) this mission could be the maiden SEP voyage replacing the proposed SEP slow flyby if its earlier launch date should prove to be programatically impossible; and 5) in any event, many future opportunities should exist for comet flyby-asteroid rendezvous missions (e.g. Halley 1986) which are uniquely suited to SEP capabilities. Other multitarget asteroid flyby concepts have been proposed elsewhere—rendezvous is much preferred for bodies of such small dimension. Finally, it appears that the proposed mission concept warrants further detailed analysis to verify its design and cost feasibility.

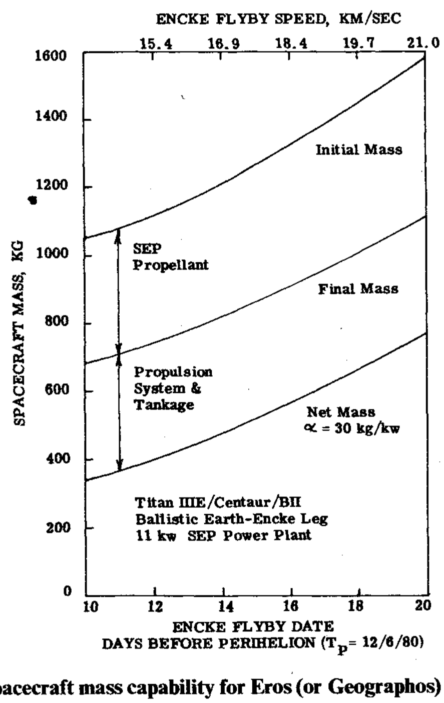

Fig. 3 Spacecraft mass capability for Eros (or Geographos) rendezvous via solar electric propulsion.

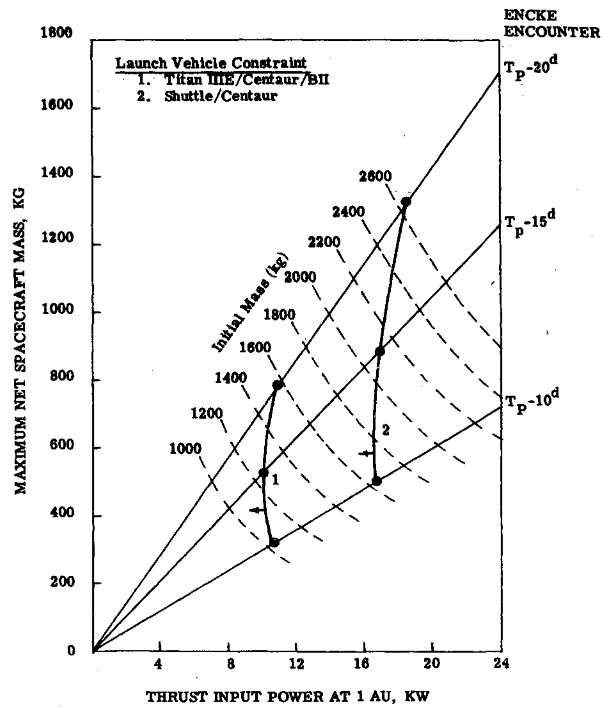

Fig. 4 SEP performance map for 1980 Encke flyby—Eros rendezvous mission. (Continuous thrust, $\alpha = 30$ kg/kw.)

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