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# ASTEROID MISSION ALTERNATIVES

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Six missions are reviewed which cover the three basic asteroid mission concepts: flyby, rendezvous, and sample return, to a variety of objects including Apollos, Amors, main belt members, and Trojans. Mission characteristics and propulsion requirements of each example are provided along with illustrations of flight profiles. A detailed argument is presented for rendezvous encounter as the best alternative for "exploration" level investigation. Assumption of this encounter option leads to the choice of multi-asteroid rendezvous as the best concept option for early mission exploration of asteroids. The propulsion requirements of multi-rendezvous point to the need for the timely development of low-thrust performance capability for NASA's continued solar system exploration program. It is shown that a minimum solar electric propulsion system of 25 kw with array concentrators is needed to perform multiple rendezvous missions of more than two asteroids. This same system is more than adequate for sample return missions as well. A brief discussion of rendezvous maneuvers demonstrates the utility of orbits for objects greater than 10 km in diameter. An encounter strategy is proposed which features adaptability and flexibility; this strategy requires low propulsion expenditure and only basic *a priori* target information. It is concluded that continued mission and systems analyses can bring us to a high state of flight project readiness by the mid-1980's.

## INTRODUCTION

NASA-directed studies of asteroid missions have been performed almost since the agency was formed. Initial results, obtained as early as 1963 by IIT Research Institute (Anon., 1964) dealt principally with flyby missions to the well-known objects, *e.g.*, Ceres, Vesta, Eros, and Icarus. As both analytical capability and propulsion technology evolved, more difficult concepts began to receive consideration, *e.g.*, rendezvous and sample return, with some studies including various forms of low-thrust propulsion. By 1972 mission analysts had generated a substantial base of data on requirements for missions to specific asteroids, *e.g.*, Northrup Services, Inc. (Anon., 1972).

Perhaps the earliest serious consideration of the importance of asteroid missions to solar system exploration by the science community occurred during the 12th Colloquium of the IAU, entitled *Physical Studies of the Minor Planets*, held in Tucson, Arizona in March 1971 (Gehrels, ed., 1971). It became apparent during the course of this meeting, particularly owing to a paper by Anders (1971), that serious planning of an asteroid mission would be premature at that time. This position was strongly supported by two important facts: (1) our state of knowledge about asteroids was based on limited data about several larger or close-approaching objects while the potential for much better information through continued dedicated ground-based observations was very high; and (2) mission concepts to date had concentrated on single well-known targets, which seemed to offer a return of information that was comparatively small, in comparison with the vastly more complex goal of asteroid exploration. Consequently, during the decade of the 1970's, asteroids continued to be studied from the Earth, rather than by missions, and rightly so. Out of this research has emerged an impressive systematic cataloging of asteroid characteristics in the form of the TRIAD data file (Zellner, 1978).

Mission concepts also improved. Specifically, analysis of multi-target concepts, introduced initially by Brooks and Hampshire II (1972) and subsequently analyzed by Bender and Friedlander (1975) and others, have shown that several objects (up to six or seven) can be encountered on a single mission, greatly enhancing its potential science return. The purpose of this paper is to present a brief review of recent progress in asteroid mission analysis, and to present arguments for a preferred concept for early exploration of the asteroids. The asteroid targets discussed include Apollos, Amors, main belt objects, and the Trojans. Mission concepts reviewed include fast and slow flybys, rendezvous, and sample return. Both single and multiple target examples are cited. Propulsion requirements of both ballistic and solar electric low-thrust flight modes are included in the mission examples examined. Mission concepts and associated characteristics are presented first by way of typical example summaries, followed by rationale and supporting arguments for selection of the multi-asteroid rendezvous mission concept for early flight exploration. The paper concludes with a brief discussion of rendezvous strategies capable of global and detailed investigations of individual bodies in the presence of the small but not insignificant asteroid gravity fields.

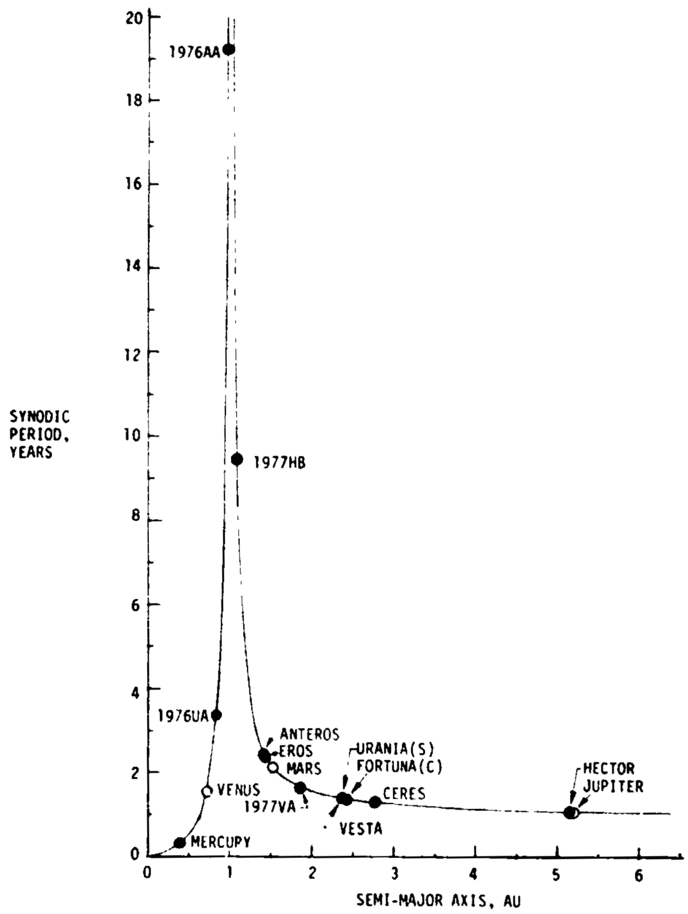



Fig. 1. Asteroid mission opportunity frequency.

## ASTEROID MISSION CONCEPTS

Missions to the asteroids, like all other interplanetary flights from the Earth, are constrained to periodic launch opportunities. Although it is theoretically possible to launch an asteroid mission almost any time, owing to the large number of available targets, missions to specific objects have specific launch opportunities, spaced in time by their synodic period with the Earth. Each object's synodic period is controlled by the semimajor axis (a) of its orbit about the Sun. A plot of synodic period (relative to the Earth) versus semimajor axis is presented in Figure 1. The average opportunity interval (*i.e.*, synodic period) of Mercury, Venus, Mars, and Jupiter are shown in the plot as open circles. Various asteroids from the Apollo 1976UA (a = 0.83 AU) to the Trojan Hector (a = 5.15 AU) are also presented in the plot, as solid dots. It is readily apparent from this presentation that missions to all asteroids beyond Mars can be undertaken with a frequency of less than once every two years; main belt missions have an opportunity frequency averaging once every 16.5 months. Only those objects which have orbits approaching 1 AU (the Earth's orbit) exhibit increasingly longer gaps between direct ballistic opportunities, owing to the low relative motion between themselves and the Earth. These objects are primarily the Apollos and some of the closer Amors. The longest interval shown on Figure 1 is for the Apollo 1976AA (a = 0.97), 19.1 years. It should be noted, however, that whereas objects which are accessible on intervals less than every two years exhibit launch windows of approximately a month, objects such as 1976AA which are accessible only once in a great while remain accessible for many months (perhaps even more than a year) when their opportunities do occur. One final point on mission opportunities to bear in mind is that the flight requirements (*i.e.*, launch energy, flight time, and payload performance) can be highly variable from one opportunity to the next because of the eccentricity and inclination of asteroid orbits. Hence, even though mission opportunities recur on average every 16.5 months or so, favorable opportunities occur with less frequency; specific examples will be cited below.

Six asteroid mission concepts will be discussed as a means of demonstrating the characteristics which are available to the mission planner for the development of flight exploration strategies. The identifying features of each of these missions are summarized in Table 1. As can be seen, this mission set includes near-Earth, main belt, and Trojan asteroid targets. Both ballistic and low-thrust flight modes are represented. Four single missions are included--two rendezvous and two sample return missions; and two multi-target missions will be discussed--one flyby concept and one rendezvous concept. All of these missions could be accomplished with current technologies, although certain hardware elements required for some of the missions have, as yet, not been developed. Each of these six examples is presented individually in the following subsections.

Table 1. Asteroid Missions

| Asteroid Object Class | Flight Mode | | Mission Concept[a] | | |
|---|---|---|---|---|---|
| | Ballistic | Low-Thrust | Flyby | Rendezvous | Sample Return |
| Apollo | x | | | S | |
| Amor | x | | | | S |
| Main Belt | x | | M | | |
| Main Belt | | x | | M | |
| Main Belt | | x | | | S |
| Trojan | x | | | S | |

[a]S: single target; M: multiple targets

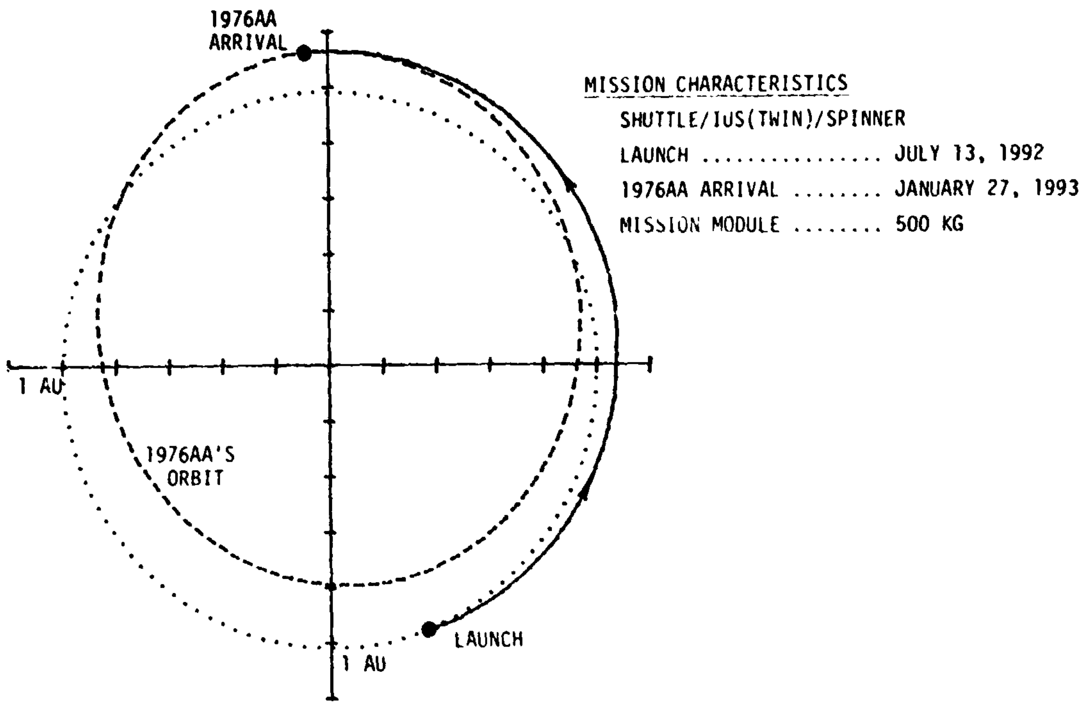



Fig. 2. 1991 1976AA rendezvous mission.

*An Apollo Rendezvous Mission*

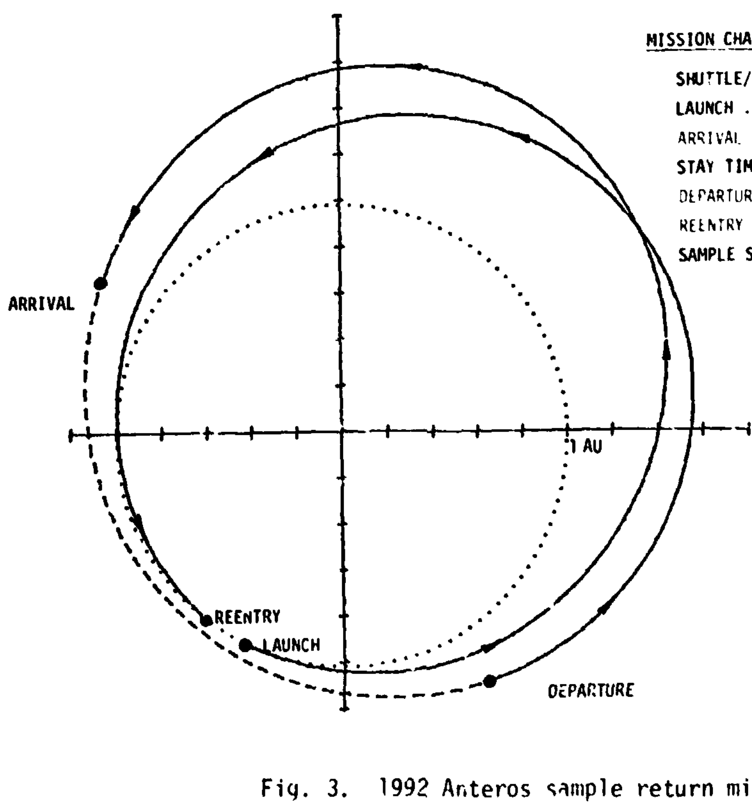
The first example is a ballistic rendezvous mission to the Apollo asteroid 1976AA. 1976AA was discovered two years ago (Shoemaker and Helin, 1978). It is the first asteroid found with a semimajor axis less than the Earth's (0.97 AU). 1976AA's orbit, with a perihelion of 0.74 AU and an aphelion of 1.14 AU, crosses the Earth. Its orbital period is about 347 days. Hence it moves slightly faster about the Sun than the Earth, passing it once every 19 years (note the synodic period of 1976AA in Figure 1). The next window of favorable rendezvous opportunities to 1976AA occurs between 1991-93 (Bender, 1976).

The close proximity of 1976AA to the Earth was immediately recognized as a potentially attractive situation for flight exploration. The asteroid's high orbit inclination of more than 19°, however, was soon realized to be a mitigating factor against low-energy missions. Both unmanned and manned ballistic round-trip missions to 1976AA have been studied (Niehoff, 1977) and both possibilities were found to require multiple Shuttle launches. Yet more practical one-way ballistic rendezvous missions to 1976AA are possible. An example of such a mission is illustrated in Figure 2 with a July 1992 launch date. The Earth's orbit is shown as dots in the figure, with the ecliptic projection of 1976AA's orbit shown as dashes. The heliocentric transfer from Earth to 1976AA is represented by the solid arc marked with arrows. A flight time of [illegible] days is required. Note that the spacecraft would never be more than 0.15 AU away from the Earth during the entire transfer to the asteroid. A rendezvous payload of more than 500 kg could be managed with a Shuttle/IUS(Twin)/Spinner launch system. This would place approximately 100 kg of science instrumentation at rendezvous with 1976AA. By comparison, this mission has approximately the same level of performance difficulty as the Galileo mission, but is accomplished in a much shorter period of time.

On the order of at least 400 Apollo asteroids equal to or larger than 1976AA are thought to exist (Shoemaker and Helin, 1978). If additional objects from this set can be found with lower inclinations than 1976AA, not only will short-time ballistic rendezvous missions be possible, but low-energy short-trip time sample return options can also be considered.

*An Amor Sample Return Mission*

This example is a low-energy ballistic sample return mission to Anteros. This asteroid, discovered in 1973, is a Mars-crosser with a perihelion of 1.06 AU and an aphelion almost in the main belt at 1.80 AU. Its low inclination of 8.7° contributes significantly to low-energy characteristics of this mission, depicted in Figure 3. Again the dotted orbit is that of the Earth and the dashed orbit is that of Anteros. The outbound and return transfers are shown as solid arcs marked with arrows. Launch is in May 1992 with arrival in August 1993. After a stay time of almost six months, departure on the homeward leg is begun in February 1994. The sample is returned to Earth in May 1995, almost three years after launch. A non-propulsive payload design of 780 kg was chosen for the performance analysis of this mission. It includes a 250 kg interplanetary bus (used both outbound and returning), a 150 kg encounter science payload, a 350 kg lander, a sample acquisition and retrieval system, a 29 kg Earth reentry capsule, and a 1 kg sample. If space-storable propulsion can be used for all the large post-launch maneuvers, then the entire mission module (~1700 kg with propulsion) can be launched with a single Shuttle/IUS(Twin) system. This launch capability results from the low post-launch impulse budget required, which is less than 3 km/sec. The total energy demands of this sample return are less than that of the Galileo mission.




MISSION CHARACTERISTICS

SHUTTLE/IUS(TWIN)

LAUNCH .............. ... MAY 26, 1992

ARRIVAL ... ............. AUGUST 20, 1993

STAY TIME ............... 177 DAYS

DEPARTURE . ....... ... FEBRUARY 14, 1994

REENTRY ................. MAY 14, 1995

SAMPLE SIZE ............ 1 KG

Fig. 3. 1992 Anteros sample return mission.

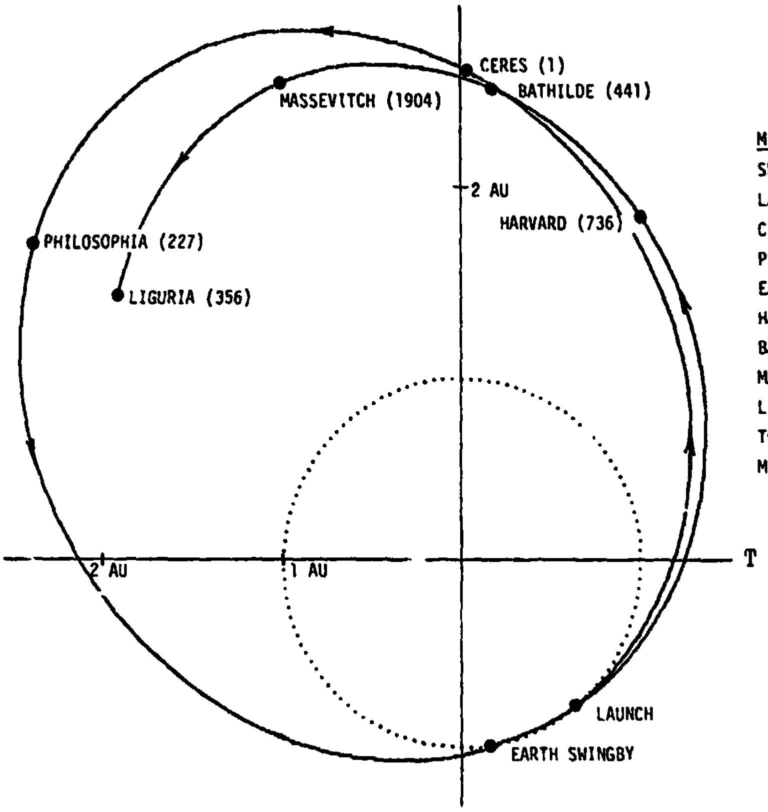
Although the energy required is low, the flight time of three years is comparable to that required for a Mars sample return mission. The stay time alone at Anteros, in this example, is comparable to the flight time of the rendezvous mission to 1976AA presented above. In general, low-energy flight times are proportional to the 3/2's power of the semimajor axis of the object, so that sample return missions require increasingly longer times, the deeper the asteroid belt is penetrated, to reach a desired target. This situation is alleviated somewhat by shorter stay times (Earth is more favorably placed at arrival for immediate departure) of objects in the main belt, and by low-thrust propulsion, but trip times will not decrease for more remote objects.

The rather high eccentricity (e = 0.26) of Anteros, combined with its period of 625 days, results in variable mission energy requirements from one opportunity to the next. With a synodic period of very nearly 2.4 years this behavior is cyclical over a period of five opportunities, or 12 years. In other words, the low-energy sample return examples to Anteros presented above occur only once every 12 years, even though four additional launch opportunities occur during this interval. As it turns out, one of these opportunities occurs with Anteros properly situated in its orbit for a fast one-year sample return mission but the energy requirements are very high. Such mission variability with opportunity is typical of asteroids with eccentric orbits. Additional characteristics of Anteros missions can be found in a recent paper by Niehoff (1977).

### *A Main Belt Multi-Flyby Mission*

Multi-targeted asteroid flyby missions were introduced by Brooks and Hampshire II (1972) as a means of expanding flyby information return for essentially the simple addition of a propulsion system comparable to that of a planetary orbiter. This concept is described generally as a series of several ballistic main belt fly-throughs during which small amounts of propulsion are expended at appropriate points along the trajectory to sequentially acquire targets of opportunity. While there is no way of knowing *a priori* any more than the first target (usually selected to start the search procedure), enough targets present themselves during the course of generating such a mission that some selection is possible. The multi-flyby example selected for discussion here was generated as part of a larger unpublished study at Science Applications, Inc., which specified *a priori* that this particular mission encounter Ceres, at least one M class (metallic) object, and as many other objects as possible. The mission was further constrained to begin during the 1984 launch window for Ceres. The ecliptic trajectory projection of the resulting mission is presented in Figure 4. It consists of two passes through the main belt separated by a reencounter of the Earth, and includes six asteroid flybys. Launch occurs in August 1984 and Ceres (the first target) is encountered in May 1985. No low-energy targets of opportunity were found between Earth and Ceres. An impulse of 390 m/sec is applied shortly after the Ceres flyby enabling an encounter with Philosophia a year later in June 1986. Another impulse of 565 m/sec is applied shortly after Philosophia flyby to reencounter the Earth in July 1987. The Earth's gravity assist along with a 315 m/sec impulse is used to reshape the second main belt fly-through to encounter the M-type object Bathilde in May 1988. One target of opportunity, Harvard, was subsequently found before the Bathilde flyby and two more, Massevitch and Liguria, were found after Bathilde but still on the same orbit revolution. An additional 465 m/sec was needed to add these targets. Liguria, the final flyby, occurs in May 1989, 4.8 years after launch.

The energy requirements of this example are the lowest of the six mission concepts presented. The post-launch impulse requirement is 1735 m/sec plus navigation maneuvers. A 500 kg spacecraft carrying 100 kg of science instruments would require an additional 800 kg of post-launch propulsion to perform this mission. The total injected mass of 1300 kg is easily accommodated by a Shuttle/IUS(Twin) system at the required injection energy (C3) of 54 $km^2/sec^2$. Note the flyby speeds given in Figure 4, which vary from a low of 5.3 km/sec at Philosophia to a high of 12.4 km/sec at Liguria.




**MISSION CHARACTERISTICS**

SHUTTLE/IUS(TWIN)

| | |
|---|---|
| LAUNCH ...................... | AUG 84 |
| CERES FLYBY (10.1 KPS)........ | MAY 85 |
| PHILOSOPHIA FLYBY (5.3 KPS)... | JUN 86 |
| EARTH SWINGBY (8.2 KPS)....... | JUL 87 |
| HARVARD FLYBY (6.7 KPS)....... | FEB 88 |
| BATHILDE FLYBY (8.5 KPS)...... | MAY 88 |
| MASSEVITCH FLYBY (5.4 KPS).... | NOV 88 |
| LIGURIA FLYBY (12.4 KPS)...... | MAY 89 |
| TOTAL TRIP TIME ............. | 4.8 YRS |
| MISSION MODULE .............. | 500 KG |

Fig. 4. 1984 multi-asteroid flyby mission.

This example has two encounters on the first fly-through and four on the second. Three encounters per fly-through are usually experienced in generating multi-flyby mission concepts, so we have here some indication of the variability in number of encounters possible per pass, although the encounters still average three per pass. Multi-flyby missions can theoretically be launched anytime. However, if a single specific main belt first target is desired, as was the case here, launch opportunities will occur only once every 16-17 months with some variability experienced in launch energy, and hence maximum payload, regardless of the subsequent targets.

### *A Main Belt Multi-Rendezvous Mission*

This next example capitalizes on the potential advantage of multiple encounters by attaining rendezvous conditions at each target instead of high-speed flybys. Not only do the spacecraft instruments have more than three orders of magnitude more time to study each object, but lighting conditions are controllable, distances remain constant, and surface probes can be deployed with reasonably small expenditures of energy, if desired. The penalty for this added capability is much higher energy requirements and longer total mission time. The multi-rendezvous mission is, in fact, the most difficult mission to perform of the six examples presented, and requires an advanced low-thrust propulsion system to meet the post-launch maneuver requirements.

The example chosen for discussion is a five-target mission that was generated by Bender (1977). Its ecliptic flight profile is depicted in Figure 5. As before, the dotted orbit is that of the Earth, and the solid arcs marked with arrows are the heliocentric transfers between targets. The dashed arcs indicate the periods of rendezvous (stay time) with each target. This time is typically set at 90 days per target, but is slightly longer at the first target, Vesta (112 days) for performance reasons. Two differences are immediately apparent compared to the multi-flyby profile (Figure 4). First, once the flight path reaches the asteroid belt it stays there. Second, the arcs, and hence flight times, between targets are longer. This is a necessary result of reducing the encounter speed at each target to zero for rendezvous, and directly increases total trip time. The five-target example shown in Figure 5 has a 1987 launch and requires almost nine years to complete if the stay time at Klytaemnestra is added to its May 1996 arrival date. The energy-efficient spiral character of the flight path was possible in this example because the four asteroids encountered after Vesta were targets of opportunity. If specific targets are desired, less efficient flight profiles are likely to occur, which could result in fewer targets being accessible within performance capabilities.

**MISSION CHARACTERISTICS**

| | |
|---|---|
| SHUTTLE/IUS(TWIN)/ION DRIVE (60 KW) | |
| LAUNCH | OCTOBER 3, 1987 |
| VESTA ARRIVAL | MAY 14, 1989 |
| ASIA ARRIVAL | APRIL 26, 1991 |
| CAMPANIA ARRIVAL | FEBRUARY 15, 1993 |
| PSYCHE ARRIVAL | OCTOBER 7, 1994 |
| KLYTAEMNESTRA ARRIVAL | MAY 31, 1996 |
| STAY TIMES | 90 TO 112 DAYS |
| TRIP TIME | 8.7 YEARS |
| MISSION MODULE | 500 KG |
| SURFACE PROBES | 75 KG PER ASTEROID |

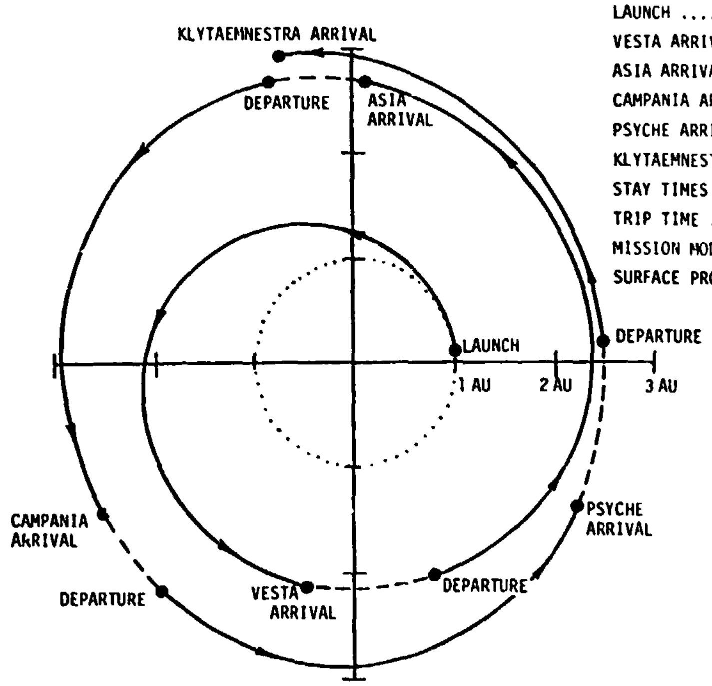


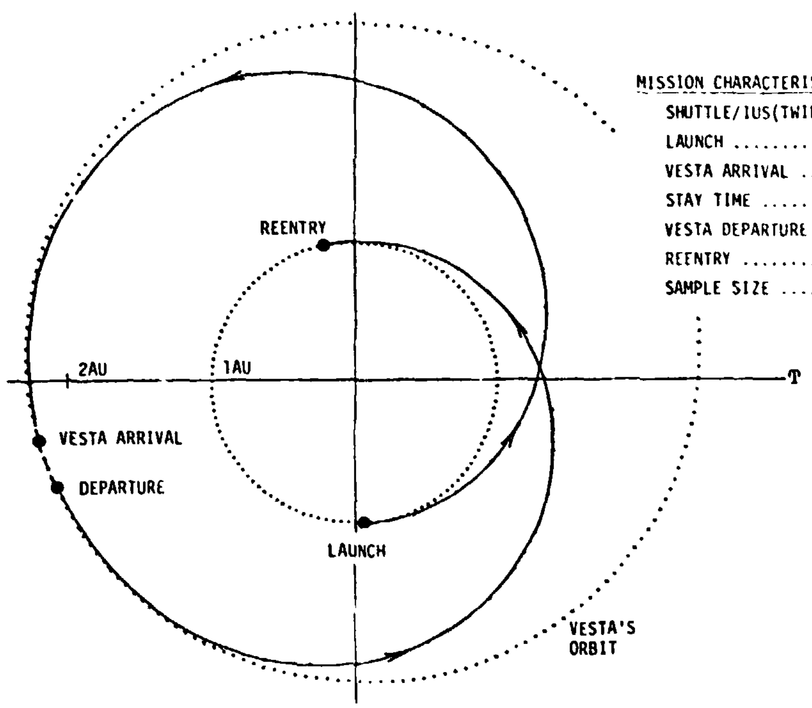

Fig. 5. 1987 mult. asteroid rendezvous mission.

A 500 kg, non-propulsive mission module carrying 100 kg of science was assumed for determining the performance requirements of this example. A further allowance of 75 kg per target was added to permit the deployment of a penetrator at each of the five asteroids, as well as 100 kg of mercury propellant for low-thrust station-keeping/orbital maneuvers. With these payload assumptions the mission requires a 60 kw Ion Drive low-thrust propulsion system similar to that recently designed by the Jet Propulsion Laboratory (Anon., 1977) for a Halley Rendezvous mission. The payload and this low-thrust system can be launched with the Shuttle/IUS(Twin).

Launch opportunity characteristics are similar to those for the multi-flyby mission discussed above. The average time between rendezvous encounters, in this example, is 1.5 years, which is typical for main belt objects. Hence, each inter-asteroid transfer and encounter is similar to an inner planet mission in time and operations. The benefit of the multi-rendezvous mission is, therefore, not so much in savings in time as it is in savings in hardware costs, since only one system is employed to explore many targets. Additional information on the tradeoff between number of targets, propulsion requirements, and flight time is given in the next section, which presents a rationale for why the multi-rendezvous mission concept should be the baseline approach to flight exploration of the asteroids.

*A Main Belt Sample Return Mission*

This example, a sample return mission to a main belt asteroid, is presented for several reasons. First, main belt sample return is a very probable element of any comprehensive asteroid exploration strategy. Second, sample return from main belt asteroids is considerably more difficult than from well-placed Apollos or Amors (such as Anteros discussed earlier), an important point relevant to planning exploration strategies.




| MISSION CHARACTERISTICS | |
|---|---|
| SHUTTLE/IUS(TWIN)/SEP(21 KW) | |
| LAUNCH | JUNE 25, 1990 |
| VESTA ARRIVAL | MAY 25, 1992 |
| STAY TIME | 30 DAYS |
| VESTA DEPARTURE | JUNE 24, 1992 |
| REENTRY | DECEMBER 27, 1993 |
| SAMPLE SIZE | 1 KG |

Fig. 6. 1990 Vesta sample return mission.

The specific example chosen for illustration of requirements and characteristics is a 1990 Vesta sample return. The heliocentric flight profile is presented in Figure 6. Launch takes place in June 1990. A low-thrust interplanetary transfer delivers the sample return mission module to Vesta almost two years later in May 1992. A short stay time of 30 days is assumed for sample acquisition on the presumption that the target has already been explored by a precursor rendezvous mission. The same low-thrust system begins a spiral departure of Vesta in June 1992. The 1.5-year return trajectory reencounters the Earth in December 1993 where the sample capsule is released on a direct reentry flight path for surface recovery. The total mission time for this example is 3.5 years, which is typical for missions of this kind.

Payload assumptions for a performance analysis of this mission are similar to those assumed for the Anteros ballistic sample return discussed above. An interplanetary mission module of 400 kg is needed, together with the low-thrust propulsion system. Encounter operations, including initial orbit capture, descent, sample acquisition, ascent, and rendezvous with the waiting interplanetary low-thrust system and mission module are handled by a 495 kg lander/ascent/rendezvous (LAR) module. The final hardware system needed is the sample reentry capsule budgeted at 30 kg including a 1 kg sample. The 30-day stay time is divided into four segments: (1) a three-day approach phase terminated with impulsive capture of the entire system using LAR propulsion into a low circular orbit; (2) one week of orbital reconnaissance for site selection; (3) one week for descent, acquisition, and delivery of the sample by the LAR to the waiting interplanetary spacecraft; and (4) low-thrust spiral escape from Vesta in the remaining 13 days.

A preliminary assessment of interplanetary flight options clearly showed that low-thrust propulsion is needed for main belt asteroid sample returns such as the Vesta example discussed. To perform this mission ballistically, even with optimistic energy and post-launch propulsion assumptions, would require four Shuttle launches. These launches would be used to assemble 11 IUS stages in orbit needed to inject the required payload (including post-launch propulsion) on a ballistic transfer trajectory to Vesta. By comparison, the low-thrust mission can be performed with a single Shuttle/IUS(Twin) launch. A 25 kw solar electric low-thrust propulsion module easily performs the interplanetary transfers. It should be noted that this system is considerably less advanced and less costly than the 60 kw Ion Drive system used in the previous multi-asteroid rendezvous mission example. It follows that single main belt asteroid sample return missions are more easily performed than main belt multi-rendezvous missions, from a propulsion point of view.

*A Trojan Asteroid Rendezvous Mission*

The final example to be discussed is a rendezvous mission to a Trojan asteroid, captured at one of the stable libration points of Jupiter. Launch opportunities to the Trojans occur at 13 month intervals (see Hektor, Figure 1). Little, if any, mission analysis has been performed on the Trojan asteroids. Therefore, the example presented here was selected to be representative of minimum requirements for Trojan rendezvous, to determine if ballistic flight performance is adequate for this class of asteroid missions.

The selected target is the Trojan asteroid Odysseus, which has the rather small orbit inclination of only 3.2°. Propulsion requirements were further minimized, in the case examined, by selecting an optimum launch opportunity, *i.e.*, November 1988. The ballistic flight profile is shown in Figure 7, using the same orbit profile formats as in the preceding examples. Rendezvous occurs in September 1991, almost three years after launch.

It is apparent from the energy requirements for this flight profile that this would be a difficult ballistic mission to perform, considerably more difficult than the Galileo mission, for example. Hence, the performance analysis was based on the full capability of a Shuttle/IUS(Twin)/Spinner launch vehicle in order to determine maximum payload capability. Using a two-stage, high-energy, space-storable retropropulsion system, the

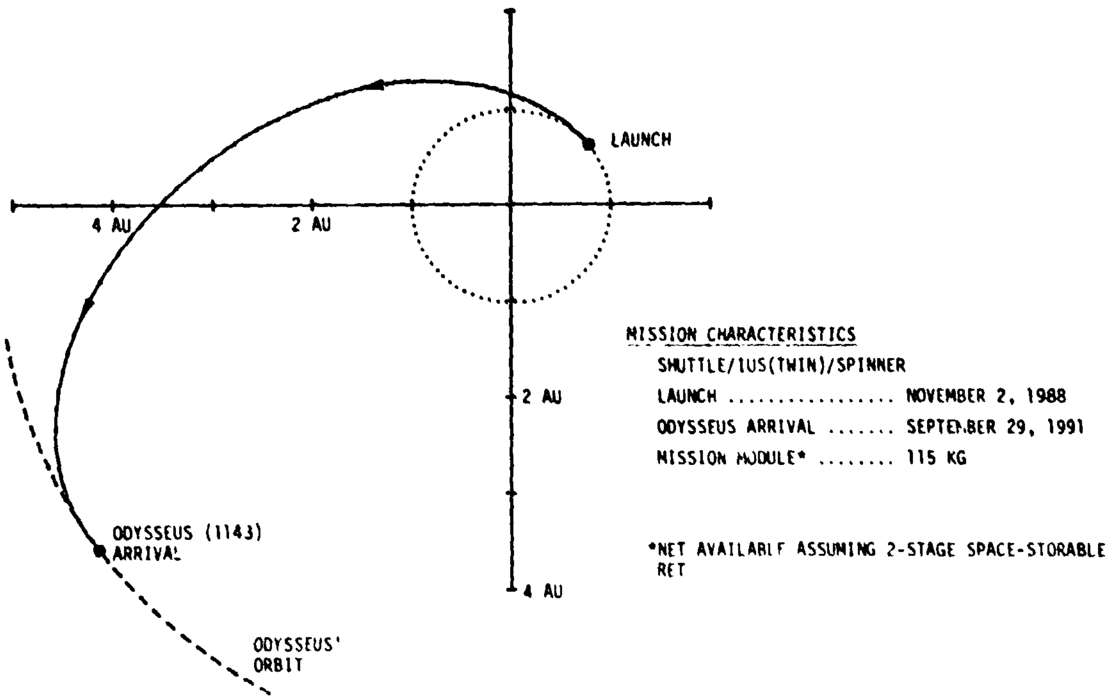



Fig. 7. 1988 Odysseus rendezvous mission.

maximum delivered rendezvous payload was found to be only 115 kg. Without exploring this example any further, two conclusions are apparent. First, even the most accessible Trojan asteroids will require low-thrust propulsion (or gravity-assisted trajectories) for rendezvous. Second, the likelihood of multi-rendezvous Trojan asteroid missions is doubtful in light of these energy requirements and the substantial differences in orbit inclinations of the larger known bodies.

## MULTI-ASTEROID RENDEZVOUS: PREFERRED EXPLORATION MODE

The Committee on Planetary and Lunar Exploration (COMPLEX) (Anon., 1976) has defined three levels of planetary investigation which are, in increasing order of comprehension and sophistication: (1) reconnaissance, (2) exploration, and (3) intensive study. The detailed ground-based program of asteroid observations, currently in progress, is often cited as the reconnaissance phase of asteroid exploration. If this premise is correct, then initial flight projects should address "exploration" level questions of asteroid investigation. With this perspective in mind, an important question to be answered prior to the planning of asteroid exploration strategies is: "What is an appropriate mission concept to undertake 'exploration' level investigation of the asteroids?"

Three mission concepts embrace the six asteroid mission examples just discussed: flyby, rendezvous, and sample return. The flyby concept can be further divided into two subconcepts, fast flyby and slow flyby. (Only the fast flyby concept has been discussed above, *i.e.*, the ballistic multi-flyby main belt mission example.) There are, therefore, four basic mission concepts for exploration level study of the asteroid. In planetary exploration strategies, the sample return concept is considered a part of intensive study. The preferred approach is to develop a broad base of "exploration" level knowledge with less costly one-way multi-targeted asteroid missions, then proceed with sample return missions to a few specific representative asteroids, in order to effectively pursue "intensive study" level objectives of asteroid exploration. It should be noted, however, in the specific case of low-energy Apollo and Amor objects, it may be possible to combine "exploration" and "intensive study" with multiple object sample returns. The practicality of such a hybrid approach will depend on additional discoveries of such low-energy objects as well as further engineering studies of mission requirements.

The assignment of sample return to the "intensive study" level still leaves three basic mission concepts to choose from for "exploration" level investigations, *i.e.*, fast flyby, slow flyby, and rendezvous. Assuming that all of these concepts are capable of carrying a comparable comprehensive science payload (on the order of 100 kg mass), then the effectiveness of each can be judged in terms of those payloads' encounter performance. Encounter performance will be assessed here by considering the capability (spatial resolution and time) of a visual imaging experiment, the premise being that if imaging encounter capabilities are unsatisfactory, so also will be most, if not all, of the other remote sensing instruments. In other words, if an asteroid remote sensing payload cannot produce acceptable imaging science because of encounter conditions (primarily velocity), it will not produce good science with its other instrumentation either.

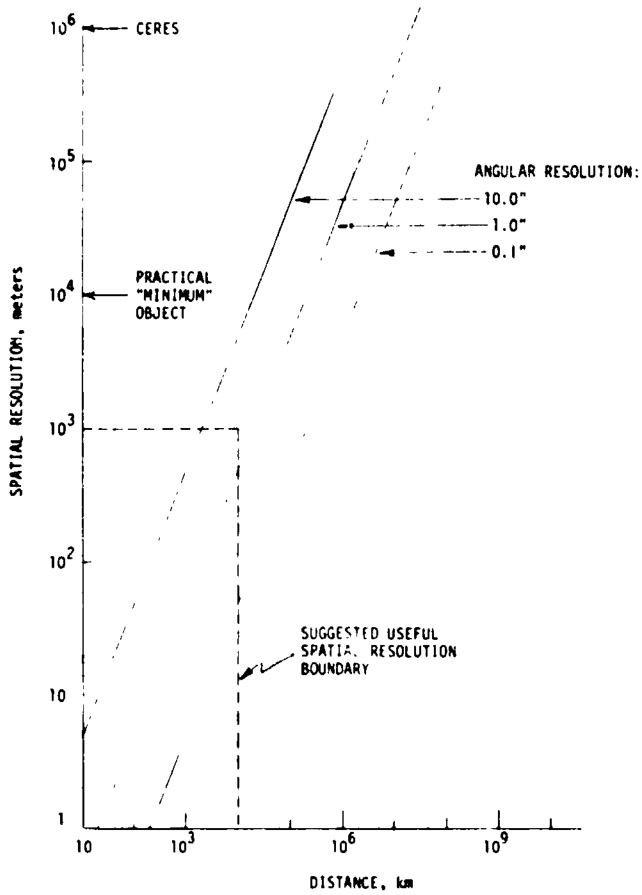



Fig. 8. Spatial resolution.

Spatial resolution is presented in Figure 8 as a function of distance for several effective angular resolutions. The purpose of presenting this graph is not to illustrate the relation between resolution and distance, which is straightforward and well understood, but to use the plot to establish a boundary of productive imaging data for asteroids. The diameter of Ceres, $\sim 10^6$ m, is noted near the top of the abscissa. Except for a few others, most notably Vesta, the remaining asteroids are less than $2 \times 10^5$ m diameter with main belt objects as small as $10^4$ m being candidate mission targets based on ground-based observational data currently being collected. From this perspective, it seems reasonable to assume that encounter imaging information would always be useful at resolutions greater than $10^3$ m, *i.e.*, resolvable picture elements an order of magnitude smaller than the smaller targets of interest. This still leaves the lower limit of resolution at 1 km, certainly quite crude even by comparison with lower resolution planetary imaging capabilities. The limiting effective resolution of spacecraft imaging instruments, based on Viking orbiter and Voyager design, is about 2 arcsec. Hence, for resolutions of better than $10^3$ m, the instrument must be within $10^4$ km of the target as illustrated by the dashed lines in Figure 8. This then is a suggested boundary on encounter distance. Perhaps more subjective, but just as important, is a secondary boundary for distinguishing surface features on the encountered objects. A suggested limit of $10^2$ m as the coarsest useful resolution for these investigations yields an upper distance boundary of $10^3$ km. To summarize, assuming that the current planetary spacecraft imaging instruments represent an advanced state-of-the-art, asteroid encounter distances must be within $10^4$ km before acceptable data on size and shape can be safely assumed (presuming acceptable phase angle conditions) and within $10^3$ km before useful data on surface features can be assumed. These values will now be used to evaluate the effectiveness of the three candidate "exploration" level mission concepts defined above.

Asteroid encounter tradeoffs between flyby velocity and time within specified resolution boundaries are presented in Figure 9. Flyby velocity is shown along the abscissa. Time within resolution boundaries is given on the left ordinate in minutes, with solid curves for the two resolution boundaries defined above, *i.e.*, $10^3$ km and $10^4$ km, plotted in the graph. For the purpose of computations an encounter of Ceres has been assumed with a closest approach of 100 km. The effect of assuming a smaller target will be mentioned in a moment. Consider first the fast flyby encounters. In the main belt multi-asteroid flyby example discussed earlier (see Figure 4), a minimum flyby speed of 5.3 km/sec was noted for the second target, Philosophia. This value is not much above the theoretical minimum flyby speed of 4.5 km/sec for a main belt asteroid at a mean distance of 2.4 AU, encountered by a ballistic coplanar transfer from the Earth. The average flyby speed of the six encounters is 8 km/sec. The resulting times spent within the resolution boundaries for speeds of 5 and 8 km/sec, found in Figure 9, are as follows:

| Flyby Speed | Resolution Boundaries | |
|---|---|---|
| | 1000 km | 10,000 km |
| Minimum, 5 km/sec | 9 min | 70 min |
| Average, 8 km/sec | 6 min | 45 min |

These times are, of course, maximum possible values. Image smear near closest approach, as well as unfavorable solar lighting geometry would reduce these times. For smaller objects the times within the boundaries (measured from the surface) would also be less.

Typical asteroid rotation periods are 5-10 hr. Hence, to obtain full longitudinal coverage of asteroids with remote sensing spacecraft will require *at least* 5 hr (300 min) within acceptable resolution boundaries. A conservative value of 10 hr (600 min) seems more reasonable for planning purposes (assuming the specific mission targets do not have

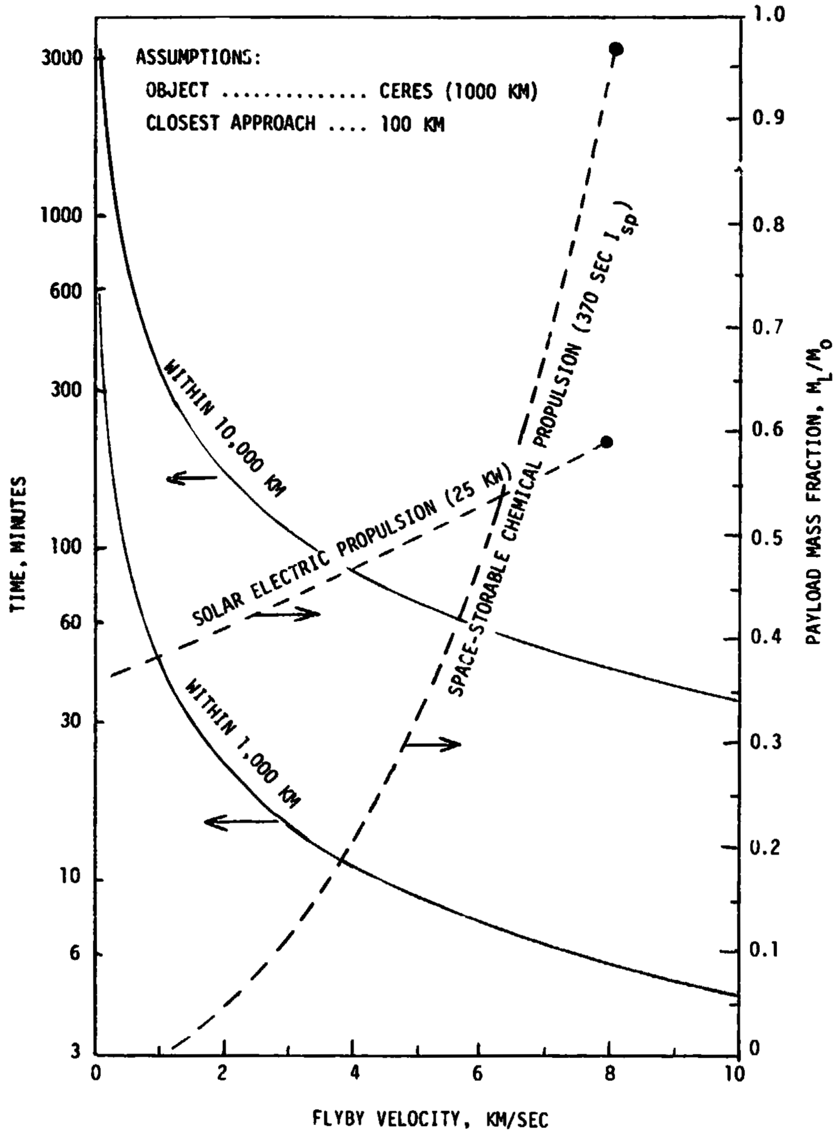



Fig. 9. Asteroid encounter tradeoffs.

known rotation rates before encounter), especially when the extent of possible rotation variations, lighting conditions, object size, and exposure/smear factors are all taken into account. Returning to Figure 9, it is found that the flyby speeds required to provide 60C min within the boundaries of 10,000 km and 1,000 km are 0.58 km/sec and 0.08 km/sec, respectively. These velocities are between one and two orders of magnitude slower than the average speed of the multi-flyby mission example presented in Figure 4. Fast flybys severely restrict the amount of useful data obtained. Hence, it can be concluded that only the slow flyby and rendezvous mission concepts are suitable candidates for "exploration" level asteroid investigation, where slow flyby refers to encounter velocities of less than 1 km/sec.

The key tradeoff which dictates the choice between slow flyby and rendezvous mission concepts for initial asteroid flight exploration is payload performance versus encounter time. The time difference between a 10 hr slow flyby encounter and a 30 day rendezvous is a factor of 72. The question is: "What is the associated propulsion penalty for achieving rendezvous instead of the easier slow flyby?" The answer to this question is also given in Figure 9. Payload mass fraction is shown along the right-hand abscissa as a function of flyby speed, where flyby speed in this case is the speed *after* a propulsive maneuver is performed, assuming that the initial speed (*i.e.*, unbraked encounter speed) is 8 km/sec--the average flyby speed of the multi-flyby mission example discussed earlier. Two dashed curves are shown relating braked flyby speed to payload mass fraction: (1) a space-storable chemical propulsion curve assuming impulsive braking, and (2) a solar electric propulsion curve assuming gradual low-thrust braking. It is apparent that solar electric propulsion is favored for reducing flyby speeds by more than a few kilometers per second, owing to its superior specific impulse. Of even more significance is the fact that the additional performance penalty to bring the encounter speed to zero, *i.e.*, rendezvous, is almost negligible. Hence, the tradeoff between payload performance and encounter time strongly favors rendezvous over slow flyby, assuming the use of low-thrust propulsion. It is concluded that multi-asteroid rendezvous is the preferred mission concept for "exploration" level asteroid investigation.

A preliminary performance summary in terms of required injected mass versus number of rendezvous targets, prepared by Friedlander (1978), is presented in Figure 10. A 40 kw Ion Drive interplanetary low-thrust propulsion system and a 500 kg mission module (~100 kg science payload) is assumed. Injected mass is given along the ordinate and number of targets is shown along the abscissa. The lower curve assumes just the mission module payload which is carried from target-to-target. The upper curve assumes an additional 100 kg small lander system (*e.g.*, penetrator), is dropped at each target. The dashed line across the plot

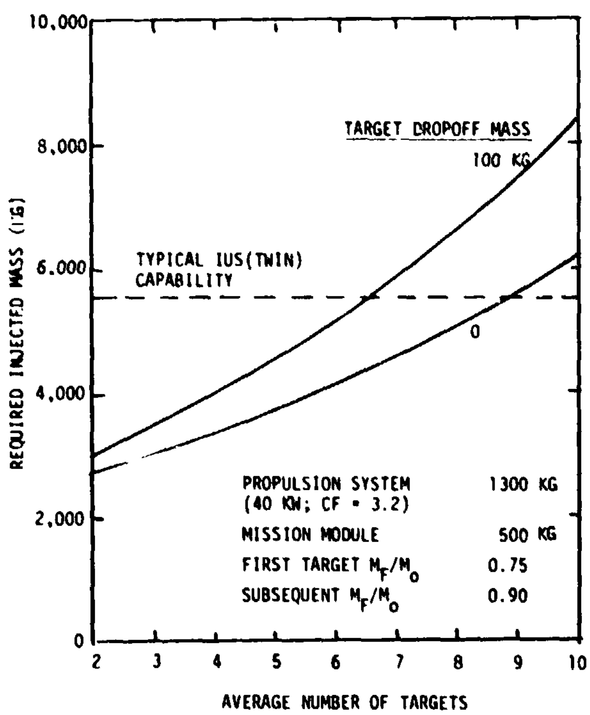



Fig. 10. Capability of advanced solar electric propulsion for multiple asteroid rendezvous missions.

shows the maximum injected mass which can be launched by a Shuttle/IUS(Twin) vehicle at the typical *vis viva* injection energy requirement. Up to eight targets can be encountered without landers, and up to six targets reached with small landers, without exceeding single launch Shuttle capability. Because of certain simplifying assumptions, these preliminary results represent upper limits on number of targets. Also, it should be noted that typically 1.5 years of flight time is required to reach each target. Hence, an eight-target mission would have a trip time from launch to the final target of 12 years. The impact of such long endeavors needs to be assessed both in terms of reliability and planning of science investigator participation.

The results shown in Figure 10 are derived for a 40 kw Ion Drive low-thrust system with solar array concentrators, a design considerably advanced over early low-thrust development plans. Using a less advanced 25 kw design, still with concentrators, would decrease the number of targets to three or four. Further decreasing the design to a 25 kw system without concentration, representative of current SEP technology, would decrease the number of targets to only two. Hence, the potential exploration capability of the multi-asteroid rendezvous concept very much depends upon the level of low-thrust performance available at the time such missions are to be implemented.

## RENDEZVOUS OPERATION CONSIDERATIONS

A few remarks on maneuver strategies during rendezvous should be made since the effects of asteroid gravity fields are different from those encountered in planetary experience. The gravity of asteroids larger than 10 km in diameter is greater than that usually assumed for comet nuclei. Hence, station-keeping rendezvous strategies typically assumed for comet rendezvous missions can be very costly at asteroids larger than 10 km, particularly when close approaches (<100 km) are desired. The preferred alternative is to orbit these objects, just as is done on planetary rendezvous missions.

Orbital periods (ordinate) are presented as a function of orbit altitude (abscissa) in Figure 11 for four size (diameter) asteroids: 1, 10, 100 and 1000 km, assuming a mean density of 3 g/cm$^3$. Two-body equations of motion dictate that all spherical bodies of equal density have the same orbital periods at zero altitude, *i.e.*, 1.9 hr. However, as altitude increases the associated orbital periods about smaller bodies increases more rapidly than for larger bodies. This characteristic is evident in Figure 11. Hence, at 10 km altitude the orbital period about the largest asteroid Ceres (~1000 km diameter) is still only 1.9 hr, whereas for a 1 km object it is 180 hr (from Figure 11). At 100 km altitude the Ceres orbiter would still have a short period of only 2.5 hr, whereas a 1 km asteroid orbiter would have an extremely long period of 5400 hr (225 days). In fact, a 1 km asteroid at a mean solar distance of 2.75 AU would have a sphere-of-influence of only 100 km.

Given the orbital characteristics just described, what should the encounter strategy be for a first remote sensing payload? Many possibilities exist. One attractive scenario, which is sequentially phased in three steps from broad global reconnaissance to very detailed study, goes as follows:

Step 1: Slowly approach the asteroid from a rest position beginning at the order of 50,000 km. During this time (~3 days) continue processing low-resolution imaging to determine object size, shape, rotation rate, and polar axis.

Step 2: Establish a polar observation orbit with a period at least several times longer than the asteroid rotation period for global medium resolution coverage.

Step 3: When global coverage is complete transfer to a low altitude circular orbit which is nearly resonant with the rotating object so that sites of specific interest can be studied repetitively in detail.

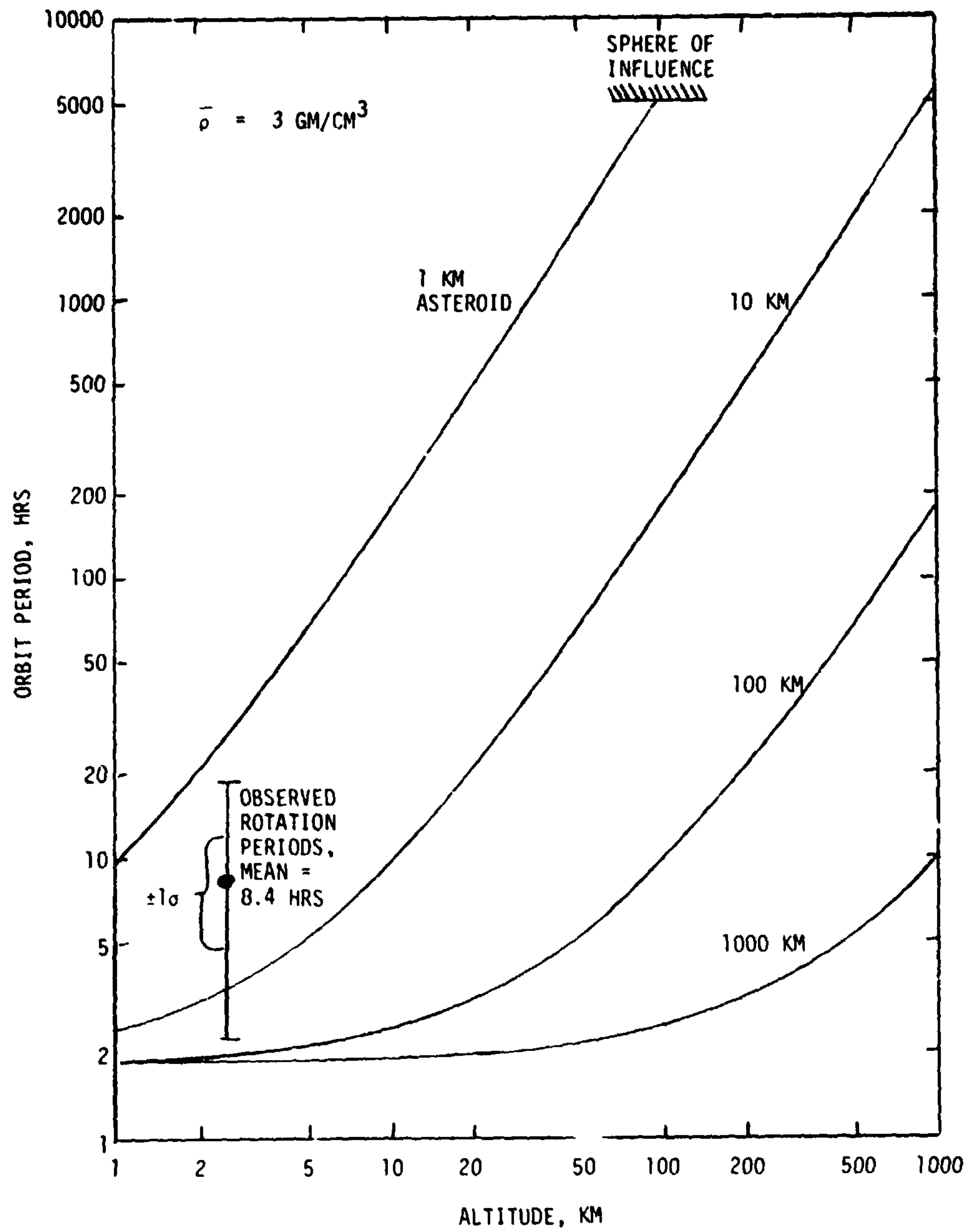



Fig. 11. Orbital characteristics around asteroids.

Three examples of this rendezvous maneuver strategy are summarized in Table 2 for three large asteroids: Fortuna, Urania, and Vesta. Global coverage orbit altitudes near 800 km were chosen to provide 100 m imaging resolution. Minimum coverage times for these objects varied from 1-4 weeks. Resonant low altitude orbits of less than 70 km provide resolutions of better than 8 m with the same camera system. Note also the small amount of impulsive $\Delta V$ which would be required to establish these orbits, the most being 141 m/sec for Vesta. Hence, these orbits can be established in a short period of time with the low-thrust inter-asteroid propulsion system, or even with a small auxiliary chemical propulsion system, if preferred. In summary, an adaptive orbital sensing strategy is suggested for asteroid rendezvous payloads which offers considerable investigation flexibility at minimum propulsion cost.

Table 2. Asteroid Rendezvous Profile Example

| Parameters | Asteroids | | |
|---|---|---|---|
| | Fortuna | Urania | Vesta |
| Class Object | C | S | U |
| Diameter (km) | 215 | 91 | 538 |
| Rotation Period (hr) | 7.5 | 13.6 | 10.6 |
| Initial Orbit Altitude for Global Mapping (km)[a] | 800 | 800 | 829 |
| Initial Orbit Period (hr) | 48.6 | 176.8 | 15.5 |
| Minimum Global Mapping Time (days)[a] | 6.1 | 7.4 | 29.7 |
| Final Orbit Altitude for Detailed Studies (km)[a] | 61 | 36 | 66 |
| Final Orbit Period (hr) | 3.7 | 4.6 | 2.6 |
| Final Orbit Rate: Rotation Rate Resonance | 2:1 | 3:1 | 4:1 |
| Equivalent Total $\Delta V$ for Orbit Capture (m/sec) | 52 | 21 | 141 |

[a]These data presented for 250 mm focal length camera with a 49 mrad field-of-view and 120 $\mu$rad per line pair resolution; mapping resolutions are $\leq$100 m, detailed studies resolutions are $\leq$8 m.

## ACKNOWLEDGMENT

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## DISCUSSION

CHAPMAN: Does the 500 kg mission module for the Anteros sample return mission include some science instruments?

NIEHOFF: Yes, there is a separate lander vehicle that actually does the acquisition and which has some surface science. There is also an allowance of 50 kg for remote sensing instruments.

VEVERKA: What is the typical $\Delta V$ for a multi-asteroid flyby mission?

NIEHOFF: In the 1994 mission shown it is 1700 m/sec. This is the $\Delta V$ after Earth escape, making this mission easier than Galileo.

VEVERKA: I am confused between the two Ion Drives. Is the conventional one the same as the one currently being discussed for a possible Comet Encke or Tempel 2 mission?

NIEHOFF: Yes. Twenty-five kilowatts is enough to do Encke or Tempel 2 rendezvous.

WETHERILL: Is there anything essentially difficult about going to the 60 kw instead of 25 kw? Is it just a matter of making it bigger or does the cost go up enormously?

MORRISON: There are some engineering changes with the larger system. One tries for higher efficiencies and that costs extra. For example, the solar concentrator for the 25 kw system is simpler, a factor of two instead of a factor of four.

FANALE: I would like to make two comments. One is that the example you gave of imaging is very useful. However, if you did the same exercise for other instruments which have wider fields-of-view, it is even more devastating. My second comment is that it is not just the resolution that we are concerned with. We mentioned briefly global coverage that you get only if you watch the asteroid rotate. You also want variations in the Sun-spacecraft-object geometry for photopolarimetry, radiometry and fields and particles experiments. I agree the resolution is a basic thing but there are other important things as well, all of which argue for the advantages of rendezvous.

ARNOLD: Usually on missions like this, the gamma-ray spectrometer is the critical item as far as time is concerned and I am delighted to have other people also wanting to be around for a long time. For the gamma-ray spectrometer, and for the x-ray system as well, one really needs to be pretty close. The angular field is typically 30°, which means in your description of the asteroid encounter, the time spent at 100 km from the object is very interesting. Being 800 km away is not. Many things I came here to argue for from one point of view are surfacing as useful from other points of view and it sounds as if a stay time on the order of 60 days is realistic.

NIEHOFF: The reason for that higher orbit was to get a quick global map. It would probably take much longer and be a much bigger burden on the imaging system if the initial map were done at closer range.

ARNOLD: I think if you tried to picture yourself in that room planning the sequence of steps, having that quick map would be very, very valuable before you started trying to think what you were going to do next.

NIEHOFF: One thing I didn't mention was that we are in the process of doing an analysis of gravity mapping with Doppler tracking and the initial results look encouraging. No separate instrument may be necessary to get the mass distribution.

ARNOLD: There is a distinction, which is important to me, between orbiting and station-keeping. I would surely think of orbiting Ceres or any really big object. I would want to get into a polar orbit and look at the whole thing. At smaller objects you do not orbit, but you would try to go to say six or eight close points and hold a position. Where does the transition between these operational modes occur?

FANALE: I indicated that station-keeping is easier than orbit-keeping except for the big ones which have a surface escape velocity of more than 100 m/sec.

ARNOLD: What diameter is that roughly, do you recall?

NIEHOFF: Orbits are possible around asteroids which are surprising small, maybe less than 10 km. As an additional point, for a very small object (less than 10 km), the surface weight of a lander would be measured in grams, not in kilograms; this is an important operational problem for sampling.