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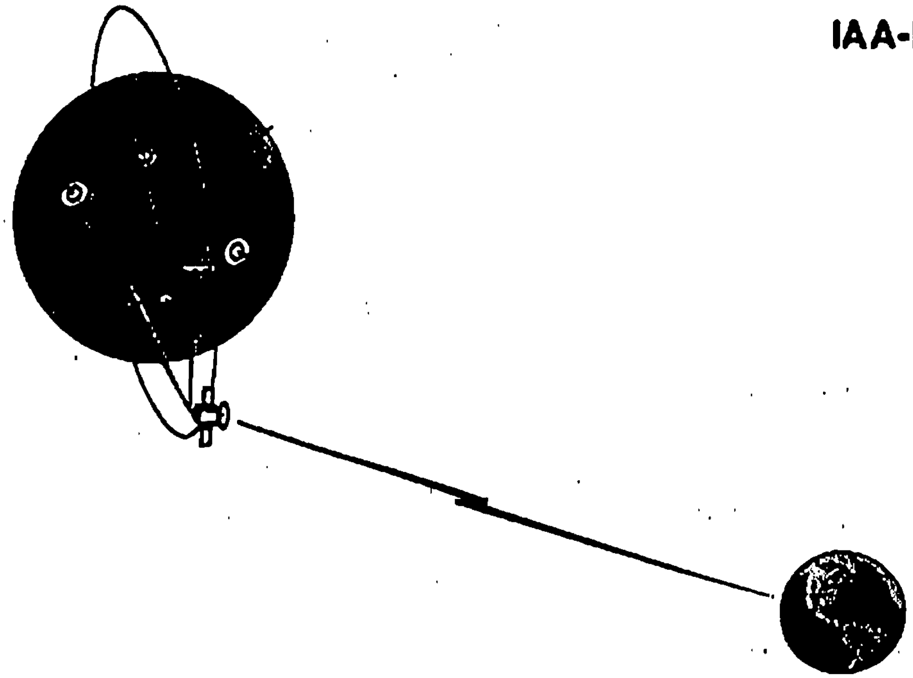

PLANETARY MISSION CAPABILITY OF SMALL LOW POWER SOLAR ELECTRIC PROPULSION SYSTEMS

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ABSTRACT

Planetary mission performance is presented for small, low power Solar Electric Propulsion spacecraft launched on a Delta II (7926) launch vehicle. The planetary missions presented in this paper are those that appear moat attractive for a small, lowcost solar electric propulsion mission. These missions include comet and main belt asteroid rendezvous and high energy outer solar system flyby missions, all requiring a large AV commitment from the propulsion system. The outer molar system missions include solar probe missions using a Jupiter gravity assist and flyby missions to small outer solar system bodies. Included in this paper are examples of heliocentric transfer trajectories and estimates of performance for selected targets for each class of mission.

INTRODUCTION

A vigorous examination of planetary missions using small, lowcost spacecraft is currently underway in the aerospace community. The intent is to develop a program of small inexpensive planetary missions that would complement larger planetary missions and provide mom frequent mission opportunities for the science community. Three lowcost missions will be launched by either a medium class launch vehicle such as a Delta II (7825) or other *small* expendable launch vehicles and will deliver small chemical propulsion spacecraft with a dry mass between 100 and 400 kg. Although many scientifically intimating planetary missions can be performed using conventional chemical propulsion, some proposed planetary missions will require long flight times and possibly multiple planetary gravity assist trajectories to deliver even a small science payload.

Tha current interest in performing these small, lowcost planetary missions has spurred the examination of the use of small, low power 5-10 kW solar electric propulsion (SEP) systems for these planetary missions. Historical impedimenta to the we of solar electric propulsion for planetary missions include development cost end risk due to the uncertainty in the

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advertised performance, reliability and lifetime of these systems. Advances in electric propulsion thruster technology and a proposed flight test of a Xenon ion thruster will do much to reduce the cost and risk of using ion propulsion systems for planetary fissions. The electric propulsion thrusters for these SEP planetary spacecraft will be either segmented ion thrusters proposed by J. Brophy at JPL[8] or 30 cm Xenon thrusters used in past SEP mission studies at JPL and the NASA Lewis Research Center (LeRC). A low power 6-10 kW SEP spacecraft would use a medium Delta II class launch vehicle. For more ambitious missions a larger 10-20 kW SEP system and an intermediate class launch vehicle such as an Atlas IIA are necessary. Only those planetary missions using a 6 kW SEP spacecraft and a Delta II (7925) launch vehicle are considered in this paper.

SOLAR ELECTRIC PROPULSION CONSIDERATIONS

Three' main benefits of using solar electric propulsion for those missions considered are shorter flight times, more frequent target accessibility, and use of a smaller launch vehicle than that required by a comparable chemical propulsion mission.

For instance small SEP rendezvous missions to comets and mainbelt asteroids are accomplished without using complex gravity assisted trajectories such as those required for a ballistic mission. As an example a mainbelt asteroid rendezvous mission to a target such as Vests can be done in two to 2.6 years with a SEP powered spacecraft as compared with a flight time of four years or more required far a Mars gravity assist ballistic mission. A more diverse selection of targets and more frequent launch opportunities is also available for a SEP spacecraft for many comet and mainbelt asteroid missions.

Possible disadvantages to the use of SEP include concerns with the reliability of the thrusters due to the extended thrust times characteristic of SEP missions and the more intensive navigation and guidance functions required during mission operation. The environmental infraction between the electric propulsion thrusters and the science payload is another issue that must be addressed for these systems.

SEP powered spacecraft may also be used for planetary orbiter missions although the flight time and delivery capability of SEP as compared with conventional them.id propulsion is not as great as for the small body rendezvous missions. Outer planet orbiter missions using SEP further require additional chemical propulsion for the orbit insertion maneuver since these low power SEP systems have insufficient power to function at heliocentric distances greater than around 2.5 AU. For these missions the SEP system is likened to a high energy upper stage augmenting the launch vehicle. The 6EP propulsion system and possibly also the solar array is jettisoned following final thrust shutdown to reduce the burden on the retro propulsion system. These same consideration apply to a lesser extent to terrestrial planet orbiter missions except the solar array would be retained to provide power for spacecraft operations while in orbit. The performance advantage of SEP over conventional chemical propulsion may be only marginal or not exist at all for terrestrial planet orbiter missions however.

Other classes of planetary SEP missions not considered in this paper include a near Earth asteroid rendezvous mission. This mission is performed quite adequately by a small low power SEP spacecraft although the payload provided by a small chemical propulsion spacecraft is probably sufficient for this mission. A near Earth asteroid sample return or a comet sample return is another candidate for a small SEP mission. Another mission, very difficult to do with a chemical propulsion spacecraft but possibly enabled by a SEP spacecraft, is that ofa multiple main belt asteroid rendezvous. This mission also is not addressed in this paper but considerable work has been done previously to illustrate this SEP capability.[3] These last missions more likely require a larger 10- 20 kW SEP and an intermediate class launch vehicle such as an Atlas IIA and would not appear to fall into the low cost mission category. ,

SEP PERFORMANCE

Performance for the planetary missions in this paper was calculated assuming a simplified mode of operation of the SEP system. This model assumes that the thrusters operate at a constant specific impulse and efficiency with the variation of array output power affecting only the thrust level. These effective values of specific impulse and efficiency were selected to give equivalent performance for the Vests asteroid rendezvous mission used as an example in Reference 2. The effective values of specific impulse end efficiency* a SEP system are functions of the solar array power led, thruster throttling characteristics and trajectory profile and consequently vary for each mission, therefore the performance quoted in this paper should be ueed with caution. The delivery capability should be indicative of expected near term SEP performance and should provide a good indication of mission feasibility however. Probably the first SEP mission to be flown may well operate with a higher power solár array power and with engines that have lower effective values of specific impulse and efficiency lower those wed in this papal,

The SEP system assumed for the missions in this paper is modeled with a constant specific impulse of 3000 seconds and an efficiency of conversion of electric to jet power of 60%, The propulsion system mass, which includes the 5 kW solar array, weighs 275 kg *with an additional* allowance of 16% of the expended propellant included for SEP propellant tanks. This tankage allowance appears typical for storage of the Xenon propellant. The solar array output power of 5 kW used in the trajectory emulation is the effective power measured at 1 AU. To account for *environmental* and other degradation factors *the beginning of* life *or* BOL solar array power is 10 to 20 percent higher, A fixed power demand of 260 watts is allocated for spacecraft housekeeping during the mission.

The net spacecraft mass** including the science payload is typically about 200 kg for these low cost planetary missions. To accommodate launch periods of 20 days or more and to allow for additional launch vehicle and SEP performance contingencies, a net spacecraft mass of around 300 kg is adopted as a criterion for a viable SEP mission.

Each of the following sections present performance for a particular set of SEP planetary missions. These missions are presented with the most attractive SEP missions considered

" Net spacecraft mass is defined as the dry spacecraft mass at the end of the mission less the propulsion system mass.

first. Exhaustive comparisons of SEP and ballistic missions are not made in this paper since the entire mission scenario, including launch vehicle costs and mission operation, must be considered in any realistic evaluation and not just the trajectory and delivery capability as presented in this paper.

MAINBELT ASTEROID RENDEZVOUS

A comprehensive examination of the delivery options available for ballistic mainbelt asteroid rendezvous missions has been made at JPL by Chen-wan Yen.[5] This examination showed a moderate launch energy requirement but a very high post launch AV requirement of 4 to 6 km/s for direct ballistic rendezvous missions to asteroids in the inner mainbelt. To reduce the high post launch AV requirement, it is necessary to perform one or more gravity assists of Mars during the transit to the asteroid. These gravity assiststed trajectories have the effect of reducing both the launch energy and the post launch AV that the spacecraft propulsion muet handle but at the expense of an increase in flight time.[6]

The result of wing a SEP system for a direct asteroid rendezvous mission is to decrease the launch energy and increase the post launch energy over that of the direct ballistic rendezvous mission. The high specific impulse characteristic of solar electric propulsion enables the SEP system to contribute much more efficiently to the total required mission energy than a chemical propulsion system, Solar array power and minimum throttling capability of the SEP thrusters define the maximum heliocentric distance where the propulsion system is effective. The 5 kW SEP system described in this paper limits the available asteroid targets to the inner mainbelt with rendezvous distances less than around 2.6 AU&em the sun.

An illustration of the heliocentric trajectory for a 2.1 year rendezvous mission to the asteroid Vesta is shown in Figure 1 to the right. The trajectory shown in this figure does not contain any intermediate mast arcs and the SEP spacecraft thrusts continuously from launch to rendezvous. Many of these asteroid rendezvous trajectories do contain short coast arcs however. The flight time for this Vests mission is typical for asteroid rendezvous missions to the inner mainbelt with net delivery masses around 300 kg. It is possible to reach other large asteroids such as Ceres farther out in the main belt by increasing the solar array power however the net spacecraft mass in this case will be considerably less than the desired 300 kg,'

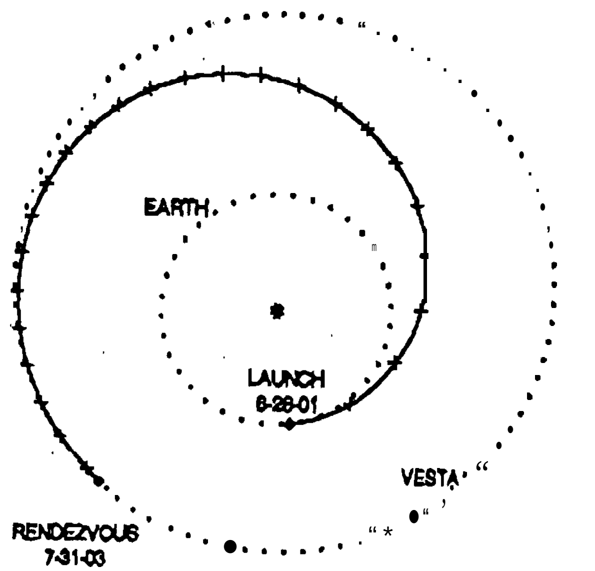

Figure 1. 2.1 Year 2001 Vesta Rendezvous

A survey of performance for large Inner mainbelt asteroids is presented in Figure 2. The ordinate in this figure is launch date and the abscissa is the flight time required for delivery ofa net spacecraft mass of 300 kg. Except asteroid 43 Ariadne, all the targets shown in Figure 2 have radii greater than 60 km. This figure illustrates both the attractive target availability and frequency of launch opportunity for selected targets available to even this small low power SEP spacecraft. Generally there exist launch opportunities to any of the targets shown in Figure 2 every 15 to 16 months, however only about every third launch

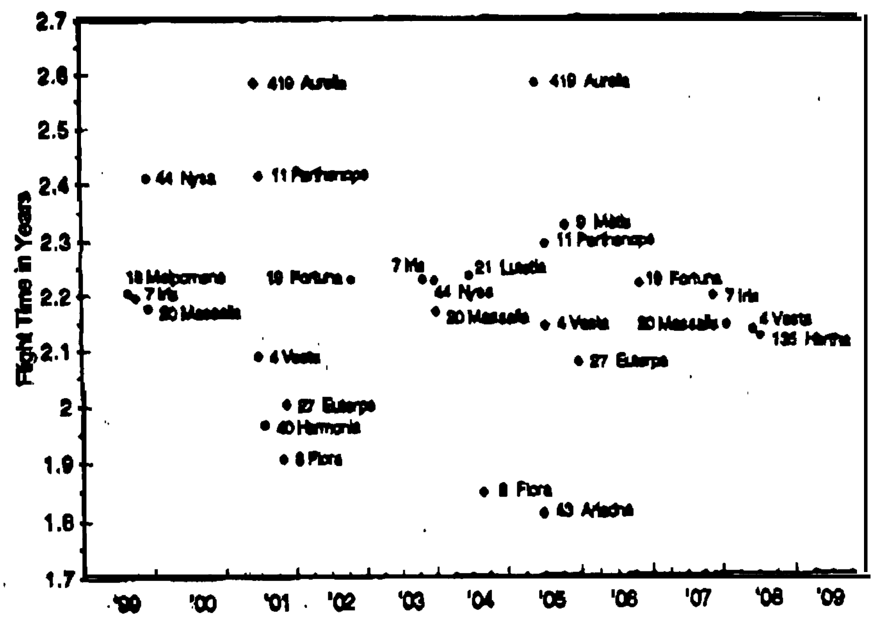

Figure 2. Mainbelt Asteroid Rendezvous Missions

opportunity has acceptable performance, As an example attractive launch opportunities exist to the asteroid Vests in 2001,2004 and 2008. Asteroids with large eccentricity will show more variation in performance between launch opportunities than will asteroids such as Vesta with low eccentricity.

COMET RENDEZVOUS

Rendezvous missions to comets and mainbelt asteroids have total mission energies that are" comparable, however the division of mission energy between the launch and poet launch mission phases is different for these two types of missions because of the much higher orbital eccentricity of short period comets. The best performance is realized for comets having orbits with low inclination and with a perihelion distance around 1 AU. Since much of the mission energy for ballistic missions is obtained from the launch phase of the *mission the* post launch AV requirements are often quite modest. The use of Earth and Venus gravity assisted trajectories for these ballistic comet rendezvous missions effectively reduces the energy requirements on the launch vehicle and enables the use of smaller launch vehicles for this mission. To keep the post launch AV as low as possible, these ballistic missions frequently rendezvous with the comet around aphelion at 4 to 5 AU or greater.

The same constraint apply to SEP comet rendezvous missions as to mainbelt asteroid rendezvous missions with respect to thrusting at extended heliocentric distances. If thrusting is constrained to distances of 2.5 AU or less from the sun, then rendezvous with the comet must occur within one year of comet perihelion. To obtain' sufficient performance for a SEP comet rendezvous mission, thrusting near comet perihelion is necessary and rendezvous with the comet occurs after comet perihelion. Pre-perihelion rendezvous with the comet generally extracts a considerable performance penalty. Although them may be a pair of launch opportunities a year apart, generally comet launch opportunities occur approximately every five to six years corresponding to the orbital period of the comet. Because of the limited performance from these small, low power SEP systems, adequate spacecraft performance is only possible with low inclination comets with perihelia less than around 1.6 AU.

An example of a SEP comet rendezvous trajectory to the comet Kopff is shown in Figure 3. This trajectory mode is rolled an indirect, post perihelion rendezvous since the spacecraft makes more than one full revolution about the sun before rendezvous with the comet. This trajectory mode offers much flexibility for comet rendezvous missions since the phasing between the Earth and the comet is handled by varying the aphelion distance during the initial part of the trajectory following launch thus allowing acceptable performance for each comet launch Opportunity. The long coast arc appearing in the trajectory in Figure 3 is quite common on the comet rendezvous trajectories.

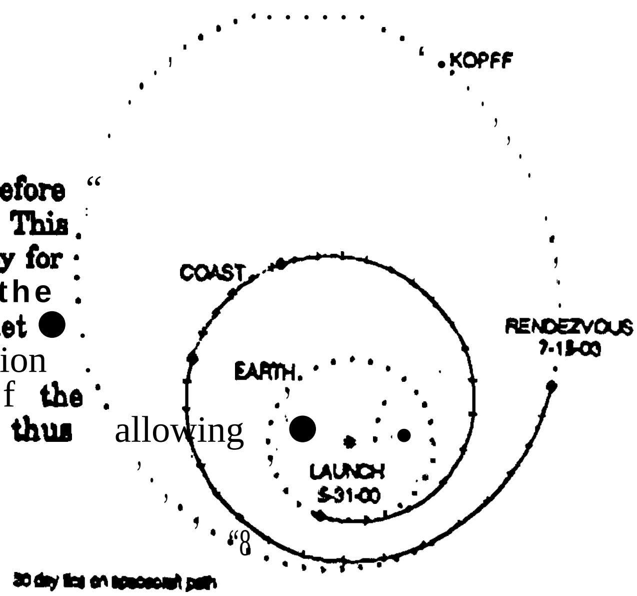

Figure 3. 2000 Comet Kopff Rendezvous

Net spacecraft mass for the selected short period comet missions is shown in Figure 6 on the next page. The rendezvous position on the cornet orbit is optimal or occurs at a thrust cutoff distance of 2.5 AU where the available array power has decreased to a point insufficient for further thruster operation, The comet targets shown in Figure 6 include several CRAF comets selected for their scientific interest plus others with low perihelion and orbital inclination. There are *comet* launch opportunities with adequate performance of 300 kg or greater in nearly all the years covered in Figure 6.

Another potential comet mission is that of a comet nucleus sample return minion. However none of the comet missions shown in Figure 6 have sufficient performance for a sample return mission. A remet sample return mission has been examined in the peat for higher power SEP systems that use an intermediate or large launch vehicle such as an Atlas IIA or Titan IV/Centaur[8]. These sample return missions would only we the SEP system to rendezvous with the comet. The return phase of the mission would be accomplished with a separate chemical propulsion system with a direct atmospheric entry at the Earth, The *mass* requirements at comet rendezvous for a sample return mission dictate a minimum of at least a 10 kW SEP system and an Atlas IIA launch vehicle.

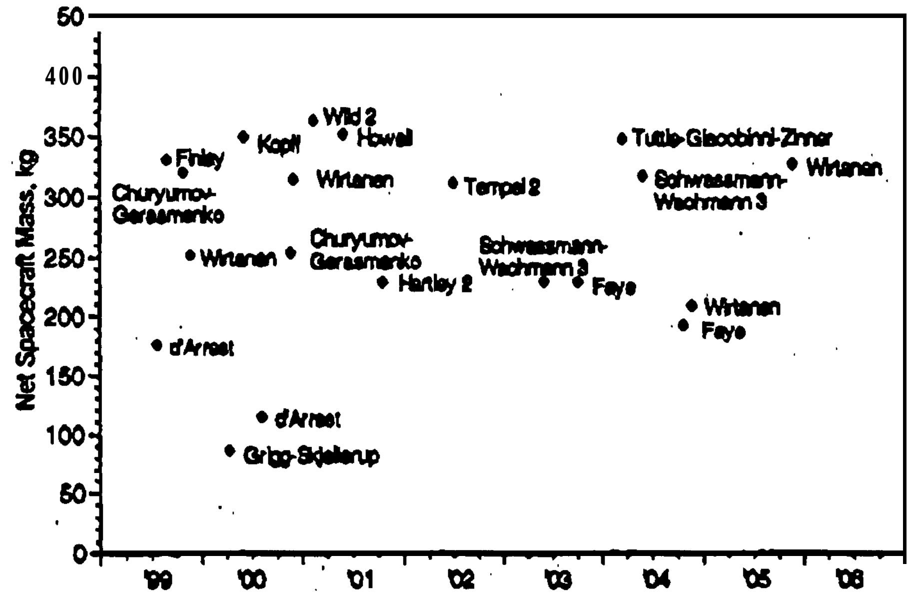

Fig. 4 Comet Rendezvous Missions

OUTER SOLAR SYSTEM FLYBY MISSIONS

The small, low power SEP systems discussed in this paper cannot deliver significant payloads on direct flyby trajectories much beyond 4 AU in a reasonable amount of time, To transfer to objects at the distance of Jupiter and beyond with adequate performance, a gravity assisted trajectory using either Venue or the Earth is necessary. These trajectories allow the spacecraft to achieve the additional heliocentric energy necessary for this mission. Even *with a* gravity assist of the Earth there is a little performance advantage in using a SEP system in place of a conventional chemical system in performing an outer planet orbiter mission. The reason for this is that a low energy arrival at the target is necessary to accommodate the chemical orbit insertion maneuver. At least for Jupiter and Saturn orbiter missions the delivery capability of a SEP spacecraft may be only marginally better than that of an equivalent ballistic mission. Use of a SEP system for fast, high energy flyby missions to objects at Jupiter's distance or beyond appears more attractive. The ability of a SEP system to thrust both before and after Earth swingby results in the addition of a significant amount of energy to the trajectory.

Them are many interesting outer solar system flyby missions that may prove attractive for a small low-power SEP spacecraft, At the distance of Jupiter there are the Trojan asteroids and more distant, a large cometary body Schwassmann-Wachmann L Located farther out at around the distance of Saturn them is the large asteroid 2080 Chiron. More distant is the interesting asteroid 5145 Pholus and finally there are objects at the distance of Neptune and beyond that include thenewly discovered asteroid 1992 QB1. Another outer solar system target is the planet Pluto and its satellite Charon. A close observation of even one of these objects should be very scientifically rewarding.

One high energy mission that does not fit into the above categories is that of a solar probe mission. This mission would use a gravity assist at Jupiter to place the spacecraft into an orbit inclined 90 degrees to the solar equator that passes at a distance of four solar radii from the sun. The hyperbolic excess speed required at Jupiter for a solar probe mission is around 13 lends and the swingby distance at Jupiter is at nine to 10 Jupiter radii.

An example of the trajectory for a high energy Earth gravity assist outer solar system flyby mission is shown in Figure 6 for a 6.5 year flyby trajectory to the asteroid Chiron. Only that portion of the trajectory is shown that is less than around 2.5 AU where thrusting occurs. The remainder of the trajectory to Chiron is ballistic and along essentially a straight line to Chiron.

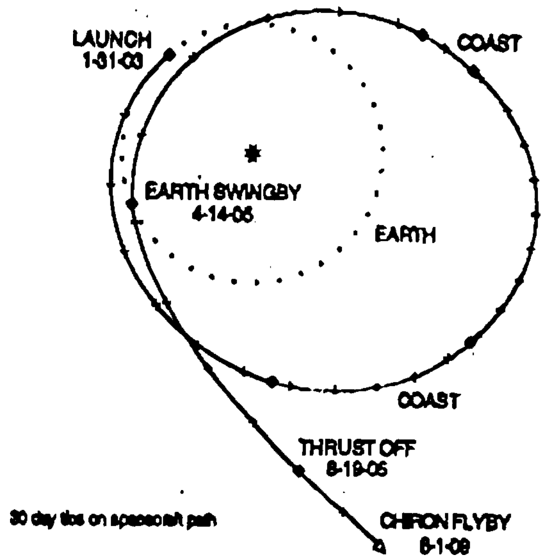

Figure 5. 6.5 Year 2003 Chiron Flyby

There are several coast arcs in the initial transfer phase of this trajectory and there is considerable thrusting following the Earth swingby. Note that this trajectory passes a short distance inside the orbit of the Earth. This is characteristic of all post perihelion Earth swingby trajectories end is more pronounced for higher energy missions such as those to Pluto. The consequence of these thrust arcs inside 1 AU is to greatly increase the demands on the SEP thrust system as compared with the other missions previously presented in this paper. This demand on the thrust system could imply that additional thrusters would need to be added to satisfy the thruster lifetime constraints. Consequently the propulsion system masses used in this paper may be too small for this class of missions end the accompanying performance estimates may be too optimistic.

A performance summary of the various representative outer solar system flyby missions discussed above is shown in Figure 6 on the next page. The performance for each target is presented as a function of flight time. Only a single point is shown for the solar probe mission since that mission is constrained by the flyby conditions at Jupiter. Performance is included for two launch opportunities for both Chiron end Pholus since both objects have orbits with considerable eccentricity or inclination and performance varies considerably with launch opportunity. Although adequate performance is achievable for the Chiron flyby mission, *the* net delivered spacecraft mass is less than adequate for the mission to Pholus. Only one launch opportunity is shown for the other objects in this figure since these objects have either such low eccentricity that the performance does not vary significantly from year to year (Schwassmann-Wachmann 1) or are so far out in the solar system that the position of the object does not change much *over the* 10-year period covered by the mission capability map shown in Figure 6. Not all possible missions are shown in this figure; flyby missions to the Trojan asteroids, for instance, are not included. The performance for a flyby mission to these bodies should about the same as that for the Schwassmann-Wachmann 1 comet mission however,

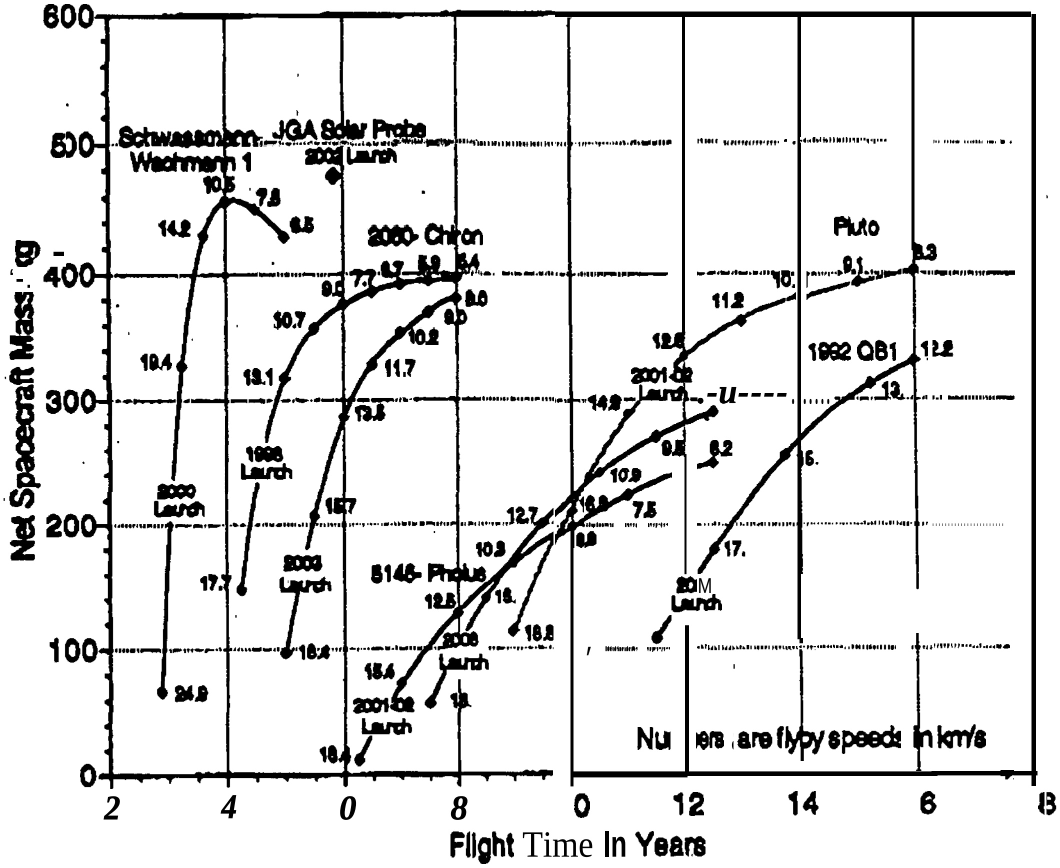

Figure 6. Outer Solar System Flyby Missions

SUMMAW

This paper has presented the most attractive missions for a first generation small, low-power SEP spacecraft. The main intent in this paper is to show the flexibility and viability in target selection available for a lowcost planetary program using a small, low-power SEP spacecraft.

The spacecraft performance presented in this paper is based upon expected near-term performance of ion bombardment thrusters however the net mass performance estimates should be used with discretion since the calculated performance is only as accurate as the models of thruster and array parameters that are used in the trajectory simulations. More reliable performance estimates for theme planetary missions will depend upon better knowledge and modeling of both thrusters and solar arrays.

REFERENCES

1. **Curran, F.M., Brophy, J.R., and** Bennett, **G.L.,** "The NASA **Electric Propulsion** Program", **AIAA** Paper **93-1935, AIAA/SAE/ASME/ASEE** 29th Joint Propulsion **Conference** and Exhibit, **Monterey,** CA June **28-30,** 1993

2. **Brophy,** J. IL, **"A Segmented** Ion Engine **Design for** Solar **Electric** *propulsion Systems".* **IAF** Paper 92"0608, **43 Congress** of the International **Astronautical** Federation, Washington, DC, Aug **28-Sep** 6,1992.

3. Yen, **Chen-wan** L., **"Mission Options for the First** SEPS Application", Paper **81-186, AAS/AIAA Astrodynamics** Specialist **Conference,** Lake Tahoe, NV, **Aug** 9-6, 1981

4, **Kakuda,** R, "Small Body **Rendezvous Missions Using** Solar **Electric** Ion **Propulsion:** . Low **Cost Mission** Approach and **Technology Requirements", IAA** Paper IAADLO71O, **IAA** International **Conference** on **Low-Cost** Planetary **Missions,** Laurel, MD., April **12-** 1/5, 1994

5. Yen, C. L., "Main-Belt **Asteroid** Exploration, **Mission** *options* **for the** 1990s", The Journal **of the Astronautical Sciences, Vol** 37, No 3, **July-September** 1989, pp. **333-361.**

6. **Veverka, J.,** et al, "Master, A **Discovery-Class Mission for** the Detailed Study of Large **Mainbelt Asteroids", IAA** Paper **IAA-L-0311, IAA International Conference** on **Low-Cost** Planetary **Missions, Laurel,** MD, **April** 12-16,1994

7. **Sauer,** O.G., "Planetary Mission **Performance for Small** Solar Electric **Propulsion** Spacecraft", Paper **AAS 93-561, AAS/AIAA Astrodynamics Specialist** Conference, Vie@-ia, C.C., Canada, **Aug** 16"19,1999

8. **Sauer, C. G.,** "Currant **Trajectory** Options for a Comet Nucleus Sample Return Mission", AAS Paper 01-474, **AAS/AIAA Astrodynamics** Specialist Conference, **Durango,** CO, Aug **19-22,** 1991.