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# PERFORMANCE REQUIREMENTS FOR A NUCLEAR ELECTRIC PROPULSION SYSTEM

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NASA is developing propulsion technologies potentially leading to the implementation of a nuclear electric propulsion system supporting robotic exploration of the solar system. The mission limitations imposed by the use of today's predominantly chemical propulsion systems are many: long trip times, minimal science payload mass, minimal power at the destination for science, and the near-impossibility of encountering multiple targets or even orbiting distant destinations. Performance-based mission analysis, comparing a first generation nuclear electric propulsion system with chemical and solar electric propulsion was performed and will be described in the paper.



## Mission Characteristics and Challenges

### Europa Lander

Candidate Europa lander mission profiles call for transfer to Europa orbit (via Jupiter) followed by lander propulsive descent and landing. An orbiter may remain in orbit as a data relay. Europa has no atmosphere, so aerocapture is not possible. For chemical and SEP missions, the vehicle captures into a highly elliptic Jupiter orbit, followed by Jupiter moon gravity assists to reduce the orbit energy to an elliptic orbit between Ganymede or Callisto and Europa distance from Jupiter. Continued gravity assist in conjunction with propulsion can bring the orbit to near Europa's orbit. The vehicle then approaches Europa and propulsively captures. The NEP can directly capture into Jupiter orbit, followed by propulsive spiral down to Europa altitude and capture into Europa orbit. The time required is less than half that required for the gravity assists.

The science payload may be quite modest for an initial lander, to probe surface conditions and return data. The primary mission objective will likely be to confirm existence of an ocean beneath the surface ice. Much more ambitious missions have also been discussed, including delivery of a large "aquanaut" probe which could melt its way down through Europa's ice to the presumed ice-covered ocean and explore in the ocean to determine whether life could, or does, exist there.

Pluto Orbiter

A Pluto flyby mission has been in and out of U.S. scientific space planning in recent years. Pluto is the only solar system planet never visited by a spacecraft.

There is scientific interest in reaching Pluto by the year 2020. Pluto's orbit is significantly elliptic, and Pluto is now moving farther from the Sun. Its rather tenuous atmosphere is expected to freeze out sometime after 2020. Scientific observations before that occurrence are desired. A Pluto orbiter would be a far more scientifically productive mission than a flyby. The flyby encounter velocity at Pluto is about 50,000 km per hour, so observations at close range will be quite brief. An orbiter could return data for years, and could get close-ups of both Pluto and Charon (Pluto's moon).

Titan Sample Return

Samples from Titan's surface are scientifically important because of the presumed pre-biotic state of organic matter. Titan (Saturn's largest moon) presents a unique mission challenge because of its deep and dense atmosphere. The estimated surface density is about 5 $kg/m^3$, 4 x the nominal density at Earth's surface. The surface is gloomy (solar input 1% of Earth, some haze) and cold, about 80K. Titan's low gravity results in a gradual decrease in atmosphere density with altitude. Consequently, the orbit altitude for a sample return orbiter will be between 1400 and 2000 km. The Titan sample return mission requires landing and ascent vehicles in addition to in-space propulsion. Landing on Titan by aero descent benefits from Titan's dense and deep atmosphere. Sample return would require the ascent system to perform rendezvous in Titan orbit and transfer the surface sample(s) to a return vehicle, which would return the samples to Earth. The return vehicle could use aerocapture or direct entry and landing at Earth.

## Payoff of Advanced Propulsion

Future missions need delta Vs from 20 to 60 km/s [2,3] in order to attain orbits around Neptune and Pluto, to rendezvous with Kuiper Belt objects, to return samples from planets, comets, asteroids and Kuiper belt objects, and to reach orbits suitable for observations of the Sun and other objects of scientific interest. The source of these requirements is the need to obtain "reasonable" trip times, as illustrated in Figure. 1. While modest-energy trajectories to outer planets exist, the time required, even for a one-way trip, becomes too great for practical scientific investigations. Faster trajectories require greater delta V for launch as well as at arrival to cancel the high encounter velocity Figure 2.

To obtain delta Vs in the range of interest, and launch on existing/emerging launch vehicles, we need new technology able to deliver much higher jet velocity (Isp). Candidates are electric propulsion powered by solar or nuclear power, and two 'propellantless' technologies, solar sails and aerobrakes[4,5].

As we seek to enable difficult missions such as a Pluto orbiter and sample return from outer Solar System locations, it is well to remember that when we double mission delta V we roughly square the payload mass fraction for chemical propulsion. For example, a Europa lander mission has a chemical delta V requirement from low Earth orbit roughly 11 km/s. The payload fraction is predicted about 2%. With 10 to 20 metric tons launched to low Earth orbit, we can land 200 to 400 kg, a respectable science payload. A Pluto orbiter requires delta V more than twice this, with a payload fraction less than 0.04%, or 4 to 8 kg. Such a small payload is actually zero. A chemical system can't do the mission.

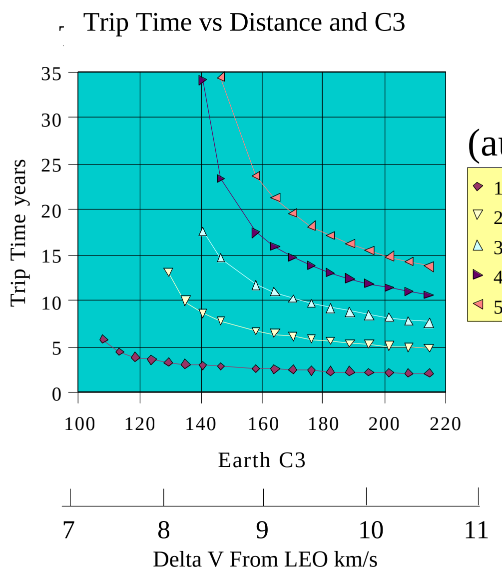



Figure 1. Trip time as a function of earth launch energy

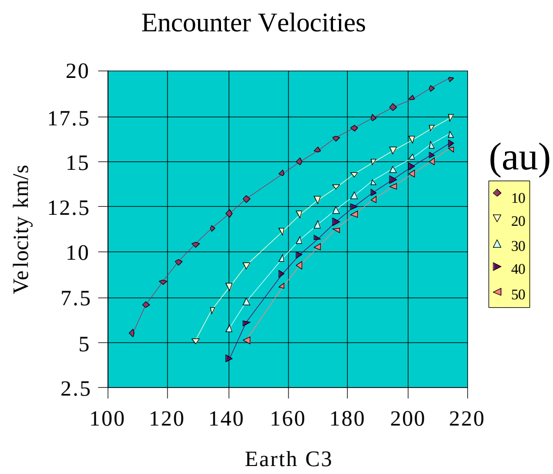



Figure 2. Encounter velocity as a function of earth launch energy

## Payload & Performance Requirements

Europa Lander

A conventional propulsion mission is described by JPL[1] . The mission has a quite small landed science payload, about 15 kg, in the interest of reducing cost. A triple Venus gravity assist is used, with total trip time 8.5 years. Launch C3 is 35 $km^2/sec^2$, using an Atlas IIAS/Star 48. The spacecraft delta V budget is described in Table 1; comments are the authors'. The spacecraft total mass including radiation shielding and contingency is 339 kg. Propellant load for the required delta V, assuming storable propellants, is 638 kg.

**Table 1: Europa Lander Delta V Budget**

| Event | DV, m/s | Comment |
|---|---|---|
| Cruise | 200 | Includes launch correction plus trajectory adjustments for Venus encounters |
| JOI & perijove raise | 910 | |
| Europa/ Callisto Tour | 310 | Using gravity assists to reduce orbit altitude to Europa/Callisto |
| Europa Tour | 350 | Continuing to reduce orbit by propulsion and Europa encounters |
| Europa orbit insert & landing | 2200 | Europa Vesc at surface 2025 m/s |
| Margin for gravity loss | 377 | |

A SEP mission will require all the same delta Vs, except that the cruise phase will be less, probably less than 50 m/s. The SEP cannot provide propulsive maneuvers in Jupiter vicinity. SEP can, however, increase the payload to Jupiter for a given launch condition. At most, a single Venus gravity assist would be needed, and the trip time can be reduced to about 4 years. NEP can provide all the maneuvers except landing. The NEP will execute a low-thrust transfer to Jupiter, enter a high orbit, spiral down to Europa's orbit, enter an orbit about Europa, and spiral down to the mission orbit altitude. The lander will separate and execute the landing maneuver. Since the landing is from orbit, the delta V allocation is 1450 m/s plus 377 m/s for gravity loss. The propellant load would be 283 kg for the same spacecraft. A likely outcome would be to significantly increase the science payload mass and spacecraft bus mass, since NEP can readily deliver 1000 to 2000 kg to Europa orbit.

Pluto Orbiter

A flyby mission can be conventionally launched. If launched in 2004, a Jupiter gravity assist is possible, offering reasonable payload with a Delta IV medium-class launch vehicle. The next Jupiter gravity assist opportunity is in 2006, and the geometry is almost wrong, so the assist is minor. A Delta IV Heavy class is required. If that launcher is used, a direct launch without Jupiter assist is possible by using a Star-48 class solid rocket as a fourth stage. The Delta IV Heavy, however, will not be available until enough demand for its services exists to make the investment pay off. Presently, when that will be is somewhat speculative. A Venus gravity assist flyby mission can also be launched with a smaller (and available) launch vehicle if solar electric propulsion (SEP) is used.

An orbiter mission is only practical with high-performance propulsion at Pluto. For reasonable trip times, as described below, the encounter velocity at Pluto is about 12 km/s. Pluto does not have enough atmosphere for aerocapture, and its gravity well is of low potential. Therefore, the entire encounter velocity must be removed propulsively. SEP does not work so far from the Sun, thus unless nuclear electric propulsion is available, one is left with chemical propulsion (jet velocity about 4 km/s, optimistically). The practical mass ratio (total mass/payload mass) is about 40 to 50. A reasonable mission spacecraft mass is about 500 kg, leading to a trans-Pluto mass requirement about 20,000 kg. Not since the Saturn V has the U.S. possessed such a launch capability. Nor does any other nation presently possess such large launch capability. NEP is the only propulsion option that can reasonably perform a Pluto orbiter mission.

Titan Sample Return

Ascent from Titan requires low acceleration and long burn time (~45 minutes) to climb out of the dense atmosphere without extreme drag losses. Solid propellants are out of the question; a liquid system must be used. While the atmosphere is bad news for ascent, it is good news for aerocapture and aero descent. A typical descent system with ballistic coefficient 100 kg/m$^2$ will have a terminal velocity less than 10 m/s without a parachute (compare with Mars at almost 300 m/s). It can either use a chute and no propulsion with shock-absorbing landing legs, or a small propulsion system with no chute.

Conventional launch to Saturn and Titan will require multiple Venus gravity assists to make the mission feasible. The solar electric propulsion (SEP) vehicle uses a single Venus gravity assist. The NEP vehicle flies a direct low-thrust trajectory to the Saturn vicinity and propulsively spirals down to the desired Titan orbit. Aerocapture is used at Titan for either conventional launch or solar electric propulsion transfer to Titan. For conventional and SEP launch, the lander makes a direct entry at Titan from the transfer trajectory, with entry speed 6 to 8 km/s. The NEP delivers the lander to a low approach velocity, and the entry speed is about 2 km/s. The parking orbit for the return system (which is the NEP vehicle in that case), as noted, is 1600 km altitude. The sample-carrier ascent vehicle returns to Titan orbit, where the return system performs rendezvous and the sample is transferred to the return system. For the conventional and SEP cases, chemical propulsion trans-Earth injection is required. The NEP vehicle performs this maneuver on electric propulsion, which gives it a large overall mass advantage. In all cases, the sample is contained in an Earth entry vehicle, which performs direct entry and landing at Earth. For the conventional and SEP cases, a spacecraft bus is required for the return trip, to accommodate midcourse correction, communications, attitude control and spin-up requirements. These functions are performed by the NEP vehicle in that case. Comparative mission profiles are presented in Table 2.

For this paper we used the Yelle nominal atmosphere [6] to estimate lander and ascent performance, and the high-density atmosphere to select parking orbit altitude at 1600 km. If a multi-year mission in Titan orbit were desired, a slightly higher parking orbit might be chosen. Terminal velocity for a typical lander, without a parachute, is about 10 m/s. We found ascent trajectories need about 30 minutes' burn time at rather low thrust, with a coast period. Thrust is low, and a single-stage expander cycle pump-fed LOX-methane propulsion system should do the job. We estimate 1000 kg start mass for the ascent system. Characteristics are summarized in Table 3.

***Table 2: Titan Sample Return Mission Profiles***

| Profile⇒ Parameter ⇓ | Chemical& Aero-braking | SEP/ Aero/ Chemical | NEP |
|---|---|---|---|
| Trans-Titan Injection | Launch vehicle + 1 or more Venus swingby | Launch vehicle C3 ~ 15 + Venus swingby | Launch vehicle to C3 0, NEP direct |
| Trip time, years | 5 | 6.6 | 6 |
| Titan entry velocity | 7 | 7 | <2 |
| Landing | Direct entry | Direct entry | From orbit |
| Orbiter Capture | Aerocapture | Aerocapture | NEP |
| Ascent | Adv. Chem. | Adv. Chem. | Adv. Chem |
| Rendezvous | Titan orbit | Titan orbit | Titan orbit |
| Trans-Earth inj | Adv. Chem. | Adv. Chem. | NEP |
| Transfer time | 6 | 6 | 6 |
| Earth landing | Direct entry | Direct entry | Direct entry |

| Table 3. Ascent Vehicle Characteristics | |
|---|---|
| Propulsion | |
| SL thrust, N | 1800 |
| Vac thrust, N | 2300 |
| SL Isp | 270 |
| Vac Isp | 343 |
| Pc, atm | 60 |
| Masses in kg | |
| Avionics | 7.00 |
| Power | 10.00 |
| Tanks | 54.43 |
| Body Structure | 10.00 |
| Nose Cone | 5.00 |
| Engine | 9.18 |
| Press | 5.00 |
| Feed | 20.00 |
| Ident Dry | 120.61 |
| Growth Allow | 0.30 |
| Growth Allow | 36.18 |
| Residuals | 15.00 |
| Inert | 171.79 |
| Propellant | 750.00 |
| Payload | 78.21 |
| Total | 1000.00 |
| Pump Data | |
| Total Flow | 0.680919 kg/s |
| LOX flow | 0.495214 |
| Methane flow | 0.185705 |
| Speed | 50000 rpm |
| Methane rotor dia | 0.085 m |
| Methane power | 6.5 kW |
| LOX rotor diam | 0.075 m |
| LOX power | 4.75 kW |

For the chemical and SEP missions, the lander and return stage are delivered to Titan orbit by aerocapture. Optionally, the lander may use a direct entry and landing. Estimated lander mass including aerobrake is 2700 kg. The return stage delivers a cruise bus and an Earth entry vehicle (EEV) to a trans-Earth ballistic trajectory. The EEV is adapted from a LaRC EEV concept for Mars Sample Return, illustrated in Figure 3. Together with the bus the estimated mass is about 300 kg. The return propulsion system also uses LOX-methane propulsion; the return delta V is a little over 4 km/s and the initial mass in Titan orbit is about 1800 kg. With the aerocapture system, the total is coincidentally also about 2700 kg. These payloads exceed the Delta IV H capability for direct injection to a Saturn transfer C3 about 110 $km^2/sec^2$. In the case of SEP, scaling from trajectory analyses for lesser payloads indicates required payload about 5600 kg to C3 between 5 and 6 $km^2/sec^2$, well within the Delta IVH capability of over 7000 kg.

## NEP Performance Analysis

Performance Factors

Electric propulsion performance analysis involves several important factors. Unlike ballistic trajectories, an infinite number of trajectories is possible between two points at fixed times. Therefore, optimization is required to select the best trajectory. The departure dates and the specific impulse must also be optimized. (If a fixed trip time is assigned, the departure date fixes the arrival date.) The result is an optimized trajectory and specific impulse for particular values of trip time, mass-to-power ratio (alpha), and power level. The optimum is defined as the maximum value of net payload arriving at the destination. The power level and alpha fix the mass of the propulsion system; the remainder is payload, which was variable in the present study.

The mass-to-power ratio, alpha, is expressed either as mass per unit jet power or mass per unit electrical power. The difference is a multiplier, the efficiency of the power processing system and the thrusters for converting electrical energy into collimated kinetic energy (which may be thought of as momentum flux, i.e. thrust) of the jet. Since the unit is mass/power, less is better and alpha-jet is a larger number than alpha-electric. The optimization and trend charts in this paper are all in terms of alpha jet, except as noted. A typical efficiency is 65%, so an alpha electric of 65 kg/kWe would become an alpha jet of 100 kg/kWj.

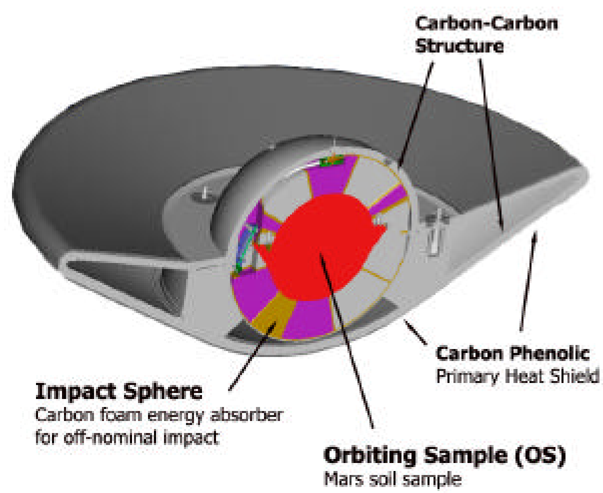



**Figure 3.** Earth Entry Vehicle Concept (Courtesy NASA Langley Research Center)

## Method of Analysis

Chebytop The present study used the Chebytop code, which performs an approximation to calculus of variations to optimize the path. An additional level of optimization is needed to select the best specific impulse and the best departure and encounter dates. This optimization was automated by one of the authors, Jonathan Jones. The launch condition was selected as (a) launch to 2500 km circular orbit (an orbit with extremely long lifetime) with 10,000 kg initial mass at that orbit altitude, or (b) C3 = 0, Earth escape, with 7500 kg. Most of the analysis was done for the first of these conditions. All of the NEP trajectories were direct from the launch condition to the target planet; gravity assists were not used. SEP trajectories came from earlier studies.

Isp optimization is required because low-thrust systems always exhibit an optimum. One definition of a low-thrust system is that the mass of equipment required to produce (even the low value of) thrust is comparable to or greater than the propellant mass. If one selects too low a specific impulse the propellant mass becomes too large; if one selects too high specific impulse, the propulsion hardware mass becomes too large (at fixed thrust, power is approximately proportional to specific impulse). The optimum value is usually a few thousand seconds. Figure 4 illustrates typical Isp optimizations for a range of alphas. As Isp is optimized, the departure date and the power are also optimized to give minimum propulsive energy and maximum payload mass delivered for that propulsive energy. The result is a best system and trajectory for a particular alpha and trip time. The figure also shows optimum power trends with Isp. The overall optimum power occurs at the Isp value for maximum payload.

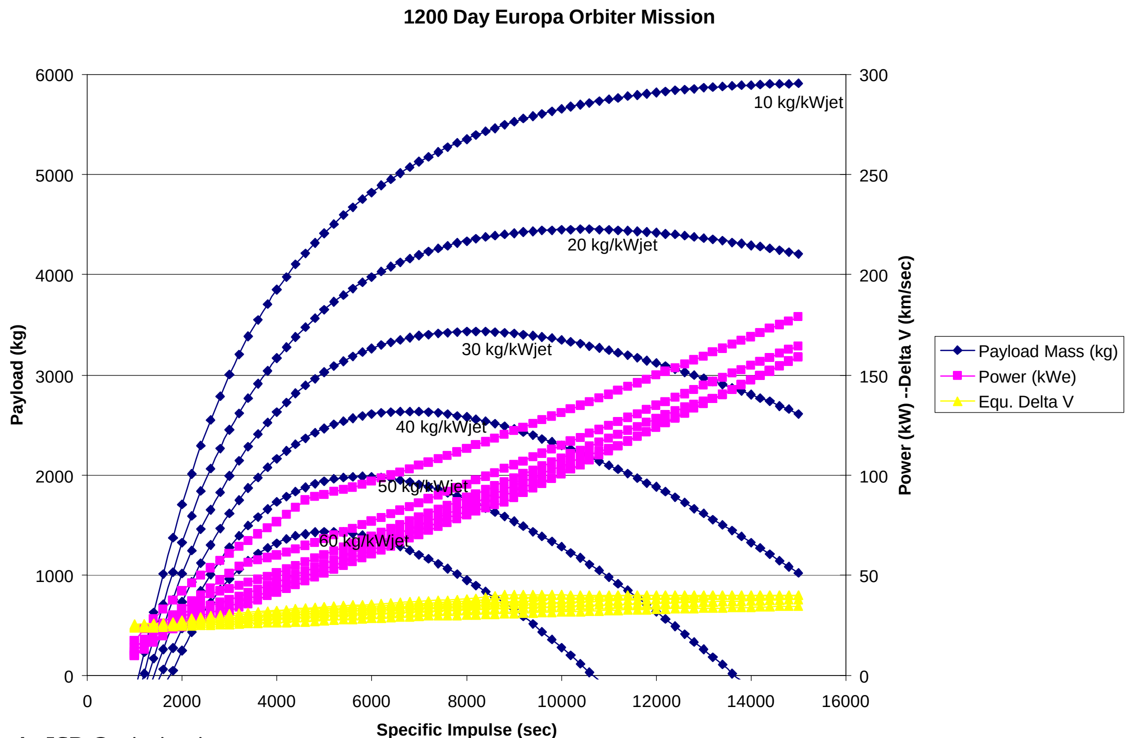



**Figure 4**. ISP Optimization

The occurrence of a power optimum is a result of trajectory characteristics. One might imagine that the optimum power at any particular Isp would be the minimum power that just permits the mission to be completed with continuous thrusting. This is not true because as power and thrust are increased, allowing insertion of coast time, the ideal delta V drops so rapidly that savings in propellant more than offset the mass increase due to power increase. As power continues to increase, a point of diminishing return is reached and an optimum occurs when the propellant savings no longer keep pace with mass increases due to increasing power. This typically happens when the coast time is roughly half the transit time, although each case is unique and a formal optimization is necessary.

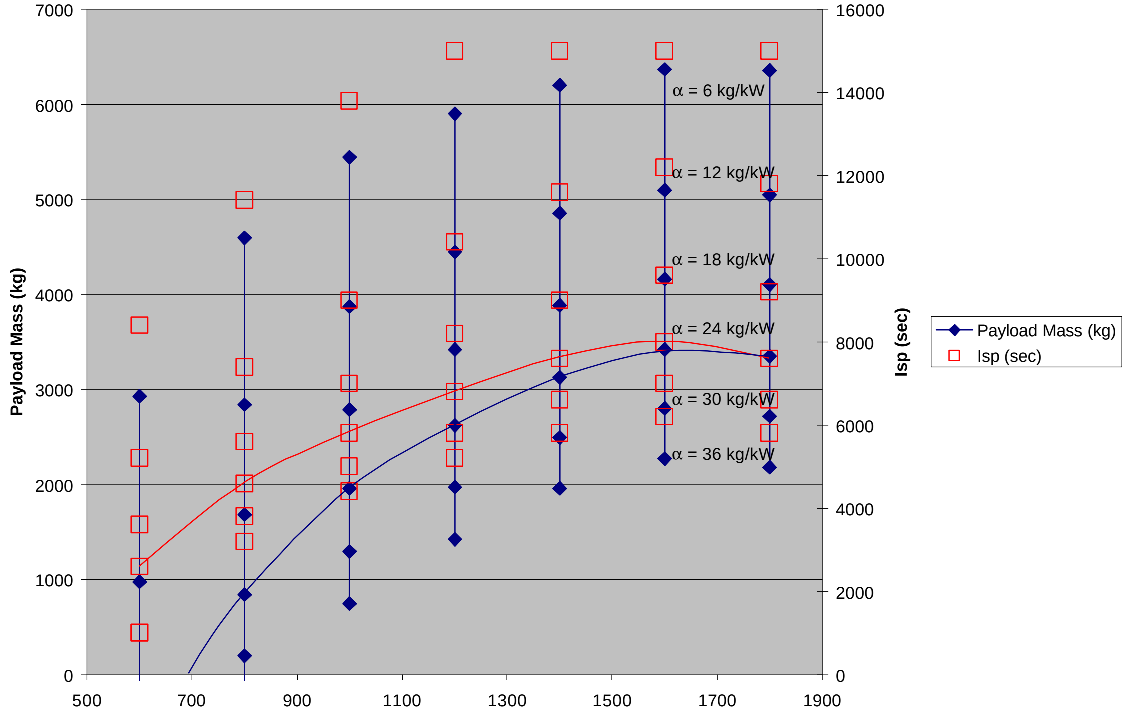



**Figure 5.** Europa Trip Times

Trip time, power and payload: In order to understand the relationship of payload to trip time, alpha and payload, one must perform the described optimization over ranges of these variables. If there are multiple destinations, all this must be done for each destination, requiring on the order of a hundred trajectories optimized for departure date, Isp and power for each destination. Figure 5 shows a typical family of optimized cases for transfer to Jupiter's moon Europa. This figure shows the expected trend that more payload is obtained if longer trip time is allowed. The reason this is expected is that electric propulsion systems require operating times on the order of the desired trip times to deliver enough delta V to perform the mission. If we permit more time, a more efficient trajectory, requiring less delta V, can be found and a higher Isp can be used. (Since, at constant power, thrust is inversely proportional to Isp, longer operating time permits a higher Isp to deliver the required delta V in the time available.)

Multiple Destinations: We wish to select a single system that can serve as many destinations as practicable. The idea is to examine the trends for all destinations to select a compromise system that performs reasonably well for many destinations rather than selecting a system that is optimum for one destination. While most of this paper considers only three missions, NEP trajectory analysis was conducted for all outer planet destinations to address this issue.

## Results of Analysis

Alpha: Alpha is a power and propulsion performance parameter. It represents what the technology can do, and the lower the number the better. Therefore, alpha is an input to the mission analysis. Our analysis covered a range of alpha since the achievable value is the subject of considerable ongoing study and is not known. One of the results of analysis such as reported here is the desired maximum (worst) value of alpha that gives reasonable mission performance. On the basis of the results achieved, we recommended 50 kg/kWj as a reasonable target. This is about 30 kg/kWe for representative efficiency. An example of the results leading to this conclusion is shown in Figure 6, for Pluto orbiter missions. Alpha 50 provides payload approaching 1000 kg for a trip time of 10 years.

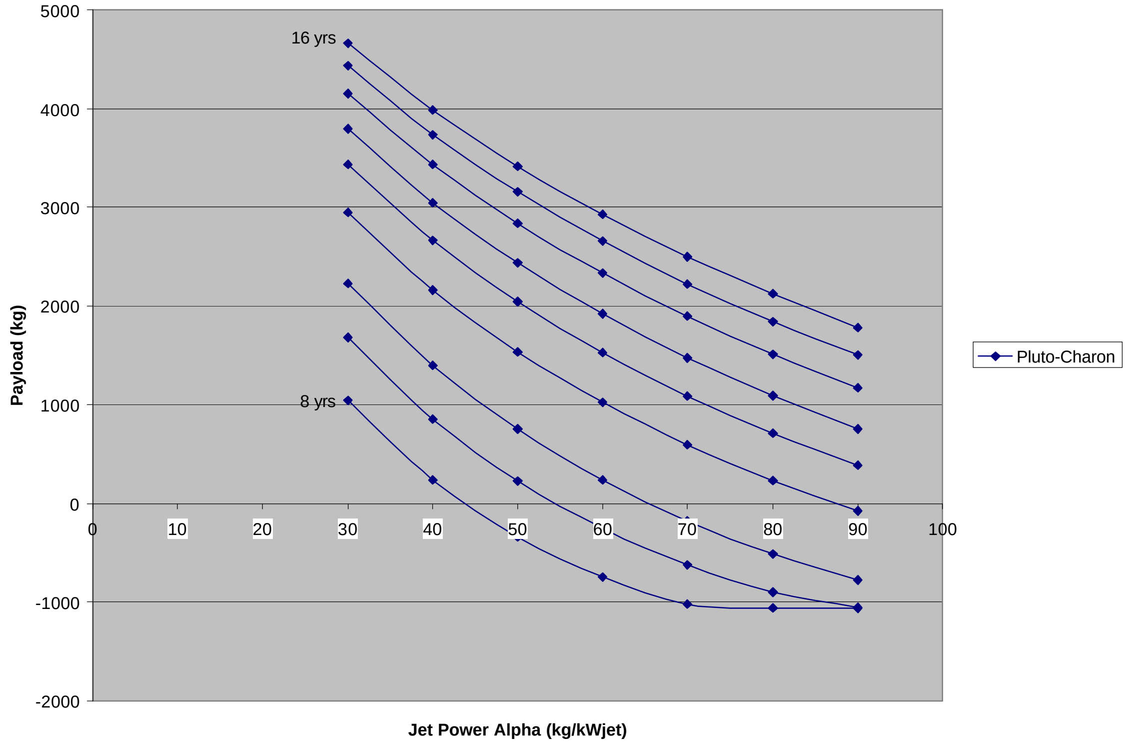



**Figure 6.** Pluto Orbiter Missions

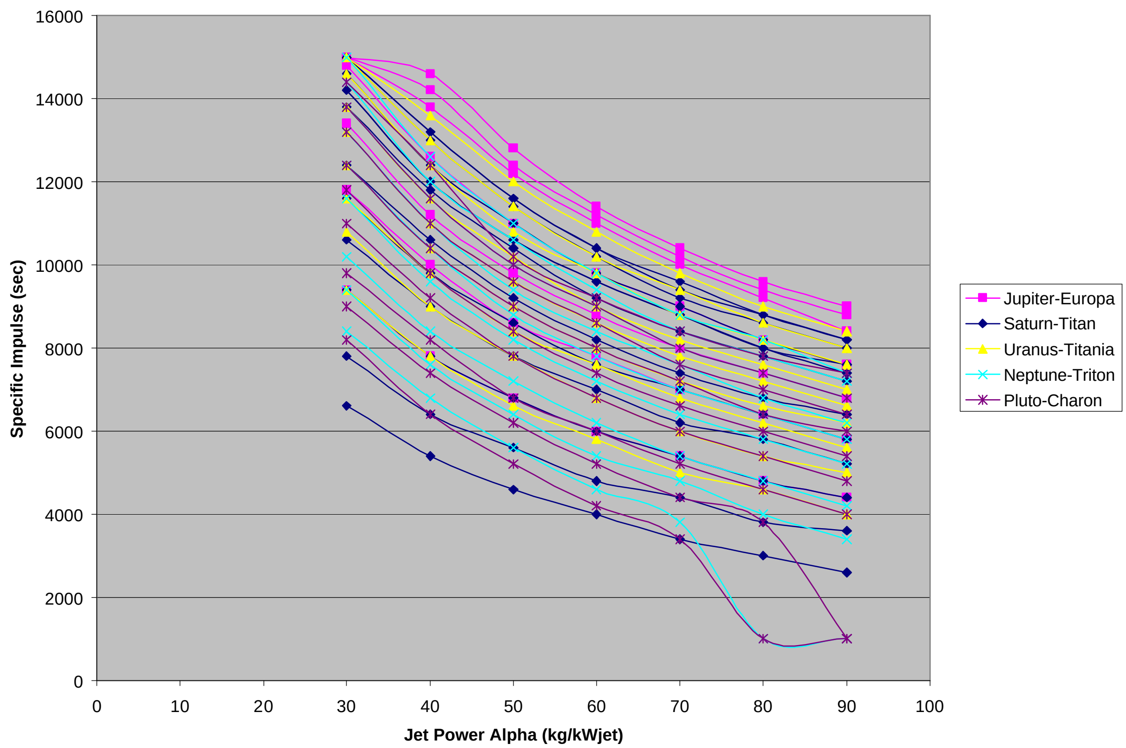



**Figure 7.** ISP Jet Power Alpha Range

Isp: Figure 7 shows optimal Isp trending for a range of jet power alpha for all outer planet destinations over a range of trip times. The plot has several curves for each destination; these are different trip times in the range of interest, 8 years to 12 years. At a jet power alpha of 50 kg/kWj, Isp in the range 6000 to 10,000

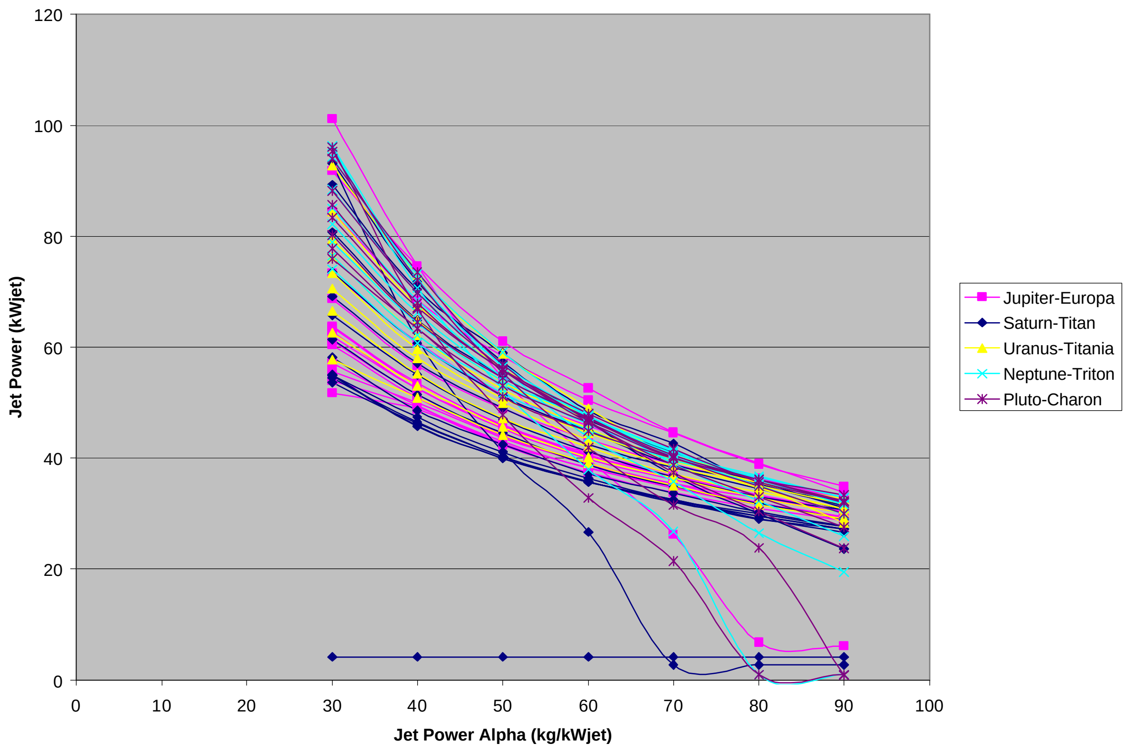



**Figure 8.** Multiple Destination Plot

seconds covers all the destinations. The NSTAR thruster delivered Isp 3100 seconds. Current concepts for next-generation thrusters are in the range 4000 to 5000 seconds; this has been shown to be near the optimal value for typical solar electric propulsion missions. The results here for NEP are expected, in that NEP missions exhibit ideal delta Vs approaching twice those for typical SEP missions. The reason is that NEP missions perform significant delta Vs at the destination; for outer planet missions SEP cannot do this, and whatever delta V is delivered at the destination is delivered another way, such as by aerocapture. Technology advancements for electric thrusters for NEP will need to focus on Isp in the range indicated, 6000 to 9000 or 10,000 seconds.

Power: Figure 8 is a multiple-destination plot much like the previous Isp plot. Convergence of optimal power is even more striking than the convergence of Isp. At jet power alpha 50 kg/kWe, a power range of 40 to 60 kWj, which is 67 to 100 kWe at a representative efficiency of 60%, captures all the destinations and all the trip times. Accordingly, a NEP powerplant rating of 100 kWe is a good nominal figure to use for concept definition studies. As these studies progress, and provide better indication of the alpha and efficiency to be expected, these recommendations should be revisited.

## Comparisons

Figure 9 captures the essence of results of this study. Payloads deliverable to various destinations by NEP are shown as a function of trip time, and conventional propulsion points are plotted on the graph. For the three missions discussed in detail in this paper: (1) All-chemical propulsion, SEP and NEP can all perform the Europa lander. For small-payload missions, little advantage is gained for SEP or NEP except that in the case of NEP the mission profile is much simpler. (2) Only NEP can do the Pluto orbiter mission, given launch mass constraints of current and planned launch vehicles. (3) Chemical propulsion cannot do Titan sample return; SEP with aerocapture can perform this mission with two Delta IV heavy launches, and NEP can perform it with one such launch. The NEP Mission profile is much simpler, and therefore one would expect a higher probability of mission success.

Figure 10 shows the trade between mission time and required alpha for Titan sample return. These curves are approximate. They are based on trajectory characteristics estimated by Carl Sauer of JPL, from converged optimized trajectories for a slightly different Titan round trip case. The achievable mission duration for alphas in the expected range, about 30 kg/kWe, is about 15 years for a 100 kWe NEP. A more powerful system could perform the mission in about half of the time if a much more aggressive alpha of about 12

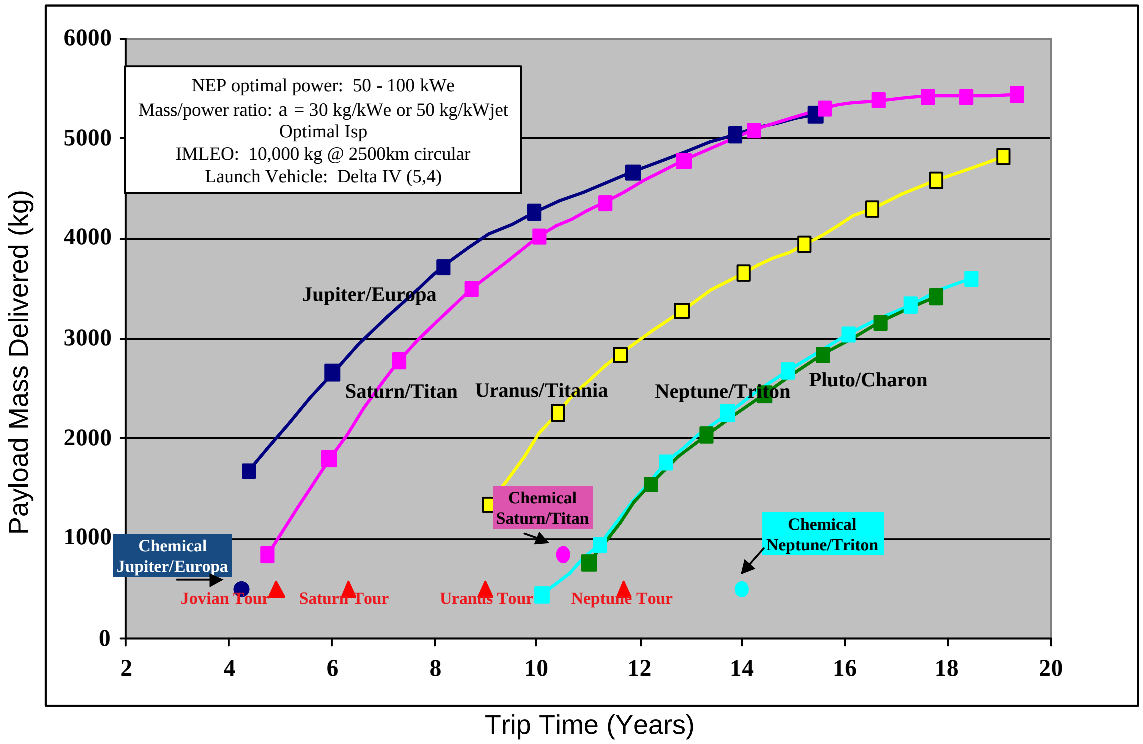
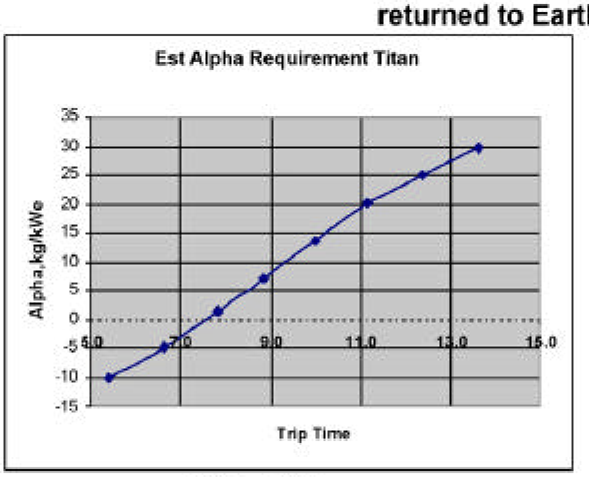



**Figure 9.** Comparison of Trip Times

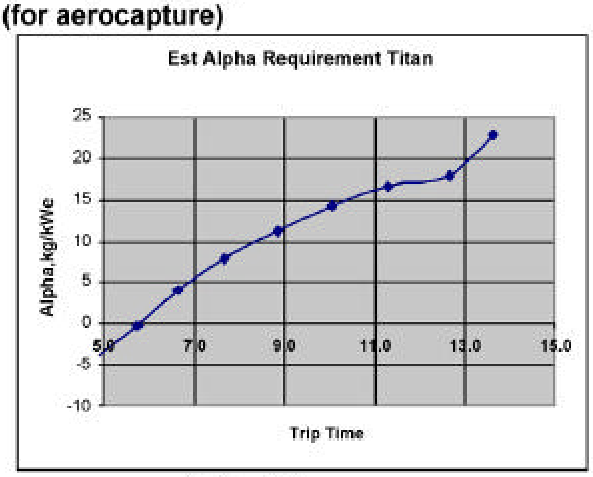
kg/kWe. Note that this curve uses alpha-electric where most of the curves presented use alpha-jet. NEP enjoys a huge advantage for heavier payloads. The payload-trip time characteristic offers large payoff for a few years added trip time. This characteristic either does not exist or is much weaker for other propulsion systems.

These characteristics of NEP missions derive from the underlying characteristics of NEP systems. NEP systems are reactor-powered. Reactor-shield systems have a minimum practical size dictated by neutronics considerations. The minimum-size reactor is capable (from the standpoints of heat transfer and nuclear fuel burnup) of producing a few hundred kilowatts thermal power. At typical dynamic conversion efficiency, this means the minimum system can generate roughly 100 kWe. If it generates less, the power conversion machinery and the thrust-producing system become smaller and lighter, but the reactor/shield does not.

**2000 kg payload to destination, 100 kg payload returned to Earth (for aerocapture)**




$P_0/M_0 = 0.01$
Power = 100 kWe
10,000 kg IMLEO
Isp = 6000-8000 sec




$P_0/M_0 = 0.02$
Power = 200 kWe
10,000 kg IMLEO
Isp = 12,000 - 15,000 sec

Combine this with other economies-of-scale effects, and one finds that the alpha for NEP improves rapidly up to power levels about 100 kWe; above that point it continues to increase but less dramatically.

A 100-kWe NEP with alpha 35 kg/kWe has mass 3500 kg. A general trend for electric propulsion systems is that well-balanced and optimized systems, at electric propulsion start, are about 1/3 power and propulsion system, 1/3 propellant, and 1/3 payload. Thus one expects an idealized NEP system, at 10,000 kg start mass and alpha 35, to be about 100 kWe, with about 3400 kg propellant and 3300 kg payload. By reducing payload we can increase propellant and decrease Isp (still delivering the required delta V in the same run time), thereby saving some trip time. But we expect this tradeoff to be costly in payload for the relative amount of trip time saved, as is the case. Our general finding is that the NEP performance advantage increases as mission difficulty increases, until we get to missions that are not practical with any less-powerful propulsion system. It is especially true that a mission requiring significant propulsive delta V far from the Sun is, unless that Delta V can be provided by aerocapture, not practical without NEP.

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